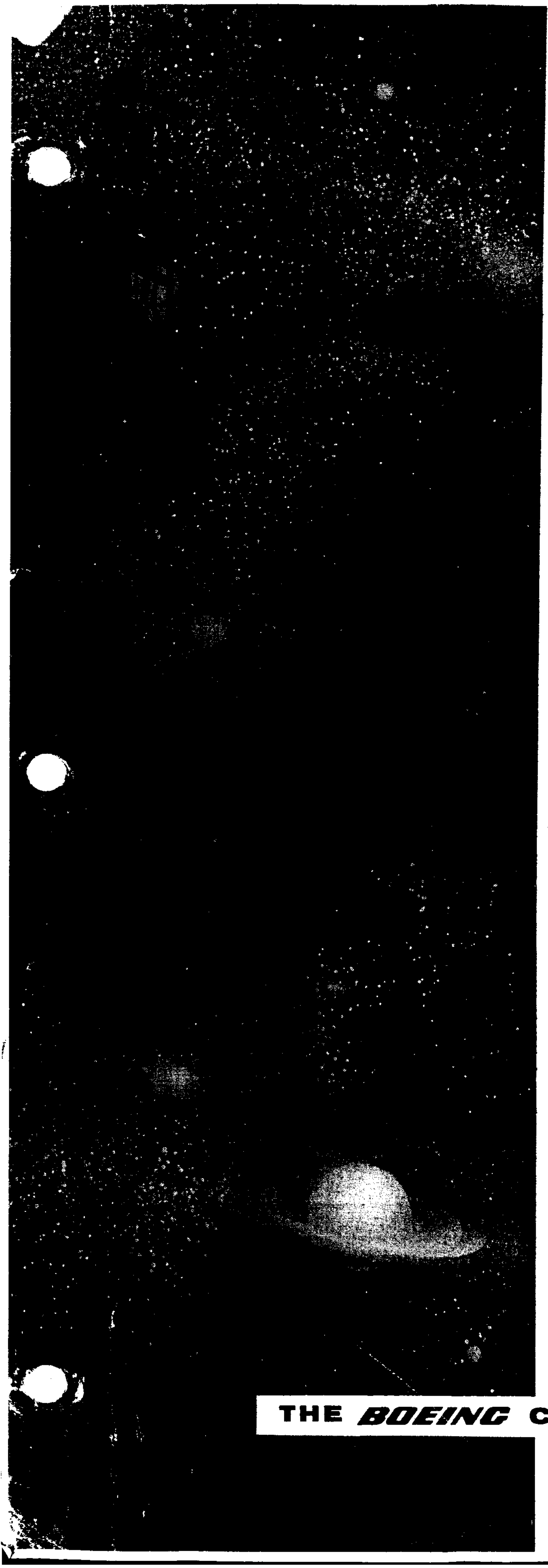

D5-15772-2

2108

SYSTEMS DESCRIPTION - J-2S IMPROVEMENT STUDY

APRIL 30, 1969

THE BOEING COMPANY · AEROSPACE GROUP
SOUTHEAST DIVISION

DOCUMENT NO. D5-15772-2

TITLE SYSTEMS DESCRIPTION - J-2S IMPROVEMENT STUDY

MODEL NO. CONTRACT NO. NAS8-5608, SCHEDULE II, PART VII, TASK 3.0

APRIL 30, 1969

APPROVED BY: J. E. MARTIN
STUDY MANAGER

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THE BOEING COMPANY SPACE DIVISION LAUNCH SYSTEMS BRANCH

REVISIONS

REV. SYM	DESCRIPTION	DATE	APPROVED

ABSTRACT

The Systems Description Document, J-2S Improvement Study, provides a condensed compilation of stage/vehicle systems and support operations changes, schedules, and costs for incorporation of J-2S engines in the upper stages of the Saturn V launch vehicle. Detailed back-up data for this condensed document are shown in supplemental documents. These documents report the detailed results of the mission/performance analysis, vehicle control and control systems dynamic characteristics, structural analysis, stage systems impact, launch operations impact, and resources analysis studies.

The Phase I trade study activity consisted of determining the design vehicle/payload configuration, mission flight profiles and J-2S/stage operational sequence which would be studied in detail in the second phase of the study. In addition, trades were conducted to establish preferred J-2S engine calibration thrust range and desirability of using mixture ratio shift schedules in the S-II and S-IVB stages. From these trades, four mission/vehicle configurations were selected for detailed analysis in the Phase II activity.

The Phase II activity consisted of studies to develop detailed design information for making an assessment of total Saturn V vehicle, support operations and KSC launch operations impact to arrive at a master schedule and program costs required to design, develop, and produce the J-2S/Saturn V vehicle. Resources required to incorporate J-2S engines for the LOR mission were identified separately from resources required to adapt the J-2S/Saturn V for synchronous orbit or low-Earth orbit missions.

The study concludes that incorporation of the J-2S engine in the Saturn V vehicle is desirable for the following reasons: (1) systems simplification, (2) improved reliability, payload, and operational flexibility, and (3) cost savings.

LIST OF KEY WORDS

D5-15772
Systems Description Document
J-2S Improvement Study
J-2S/Saturn V
J-2S/S-II
J-2S/S-IVB
J-2S Engine
Simplified J-2 Engine
Simplified J-2 Applications
Resources J-2S/Saturn V

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ABBREVIATIONS

A	Flight Azimuth
ACE	Automatic Checkout Equipment
ACS	Automatic Checkout System
ADC	Analog-to-Digital Converter
AFC	Automatic Frequency Control
AG	AND Gate
AGC	Automatic Gain Control
AHI	Aerodynamic Heating Indicator
AM	Amplitude Modulation
AMP	Amplifier
AN	Ascending Node
APS	Auxiliary Propulsion System
ASC	ST-124M Accelerometer Signal Conditioner
ASI	Augmented Spark Igniter
ASP	Accelerometer Signal Processor
ATP	Authority to Proceed
ATP	Acceptance Test Procedure
AUX	Auxiliary
AVP	Address Verification Pulse
A_z	(A) Flight Azimuth
Acc/Reg	Accumulator Register
A/S	Add/Subtract

ABBREVIATIONS (Continued)

AZTEC	Aerospace Saturn Test and Evaluation Console
B/W	Bits/Word
CAT	Control Attenuation Timer
CCP	Customer Connect Panel
CCS	Command and Communication System
CCSL	COD Counter Sequence Latch
CDDT	Countdown Demonstration Test
CDF	Confined Detonating Fuse
CDR	Critical Design Review
CDSV	Change Data Sector - Voted
CEI	Contract End Item
CG	Center of Gravity
CIU	Computer Interface Unit
CL	Center Line
CN	Criticality Number
COD	Crossover Detector
COM	Common
COV	Calculus of Variations
CP	Center of Pressure
CPP	Command Program Patch
CRP	Command Received Pulse
CT	Control Transformer
CTCA	Component Test Control Authority

ABBREVIATIONS (Continued)

CSTN	Single Step Operation - Not
CSTV	Single Step Operation - Voted
CVD	Command Voltage Demodulator
CVS	Continuous Vent System
CX	Control Transmitter
CX :CT	Control Transmitter: Control Transformer
DATAV	Nominal Data Input from External Equipment
DB	Disagreement Bit
DCR	Design Certification
DCS	Digital Command System
DCW	Data Command Word
DDAS	Digital Data Acquisition System
DIN	Special Data Input
DN	Descending Node
DOMS	Data Output Multiplexer Serializer Latch
DSIF	Deep Space Instrumentation Facility
DVT	Design Verification Tests
EAMV	Error in Even Memories - Voted
EAS	Engine Actuation System
EBM V	Error in Odd Memories - Voted
EBW	Exploding Bridge Wire
ECCS	Engine Compartment Conditioning System
ECP	Engineering Change Proposal

ABBREVIATIONS (Continued)

ECS	Environmental Control System
EDS	Emergency Detection System
EF	Emitter Follower
EMR	Engine Mixture Ratio
ESE	Electrical Support Equipment
EVT	Equipment Verification Test
FAV	Fixed Angle Variable
FCC	Flight Control Computer
FCI	Flight Critical Item
FCM	Flight Combustion Monitor
FCS	Flight Control System
FIR	Far Infrared
FIOR	Flight Information and Operations Report
FM	Frequency Modulation
F/M	Thrust Acceleration (Force/Mass)
FPSD	Fuel Pump Seal Drain
FS	Factor of Safety
FTC	Florida Test Center (MDAC)
g	Gravity
GBS	Gas Bearing Supply
GCC	Ground Control Computer

ABBREVIATIONS (Continued)

GG	Gas Generator
GM	Universal Gravitational Constant
GRR	Guidance Reference Release
GRRA	Guidance Reference Release Alert
GSCU	Ground Support Cooling Unit
GSE	Ground Support Equipment
GU	Spin Reference
H	Spinning Mass (Angular Momentum)
HALTV	Halt - Voted
H&CO	Handling and Checkout
HOPCIV	Generate HOP Constant - Voted
i	Inclination of Orbit (Angle)
I&C	Instrumentation and Communication
IA	Imput Axis
ICD	Internal Control Discrete
ID	Identification
IECO	Inboard Engine Cutoff
IF	Intermediate Frequency
IGM	Iterative Guidance Mode
IIR	Intermediate Infrared
INTC	Interface Output Latch (Interrupts LVDC)

ABBREVIATIONS (Continued)

INTV	Interrupt - Voted
INT	Inverter
IP&C	Instrumentation Program and Component
IP&CL	Instrumentation Program and Components List
IR	Infrared
IRIG	Interrange Instrumentation Group
LCC	Launch Control Center
LEO	Low Earth Orbit
LES	Launch Escape System
LET	Launch Escape Tower
LM	Lunar Module
LO	Lift-off
LOR	Lunar Orbital Rendezvous
LOX	Liquid Oxygen
LSD	Least Significant Digit
LTE	Laboratory Test Equipment
LUT	Launch Umbilical Tower
LVDA	Launch Vehicle Data Adapter
LVDC	Launch Vehicle Digital Computer
MAX Q	Maximum Dynamic Pressure and Bending Moment
MAP	Message Acceptance Pulse

ABBREVIATIONS (Continued)

MCC	Mission Control Center
MCR	Magnetic Core Register
MFV	Main Fuel Valve
MGSE	Mechanical Ground Support Equipment
Mod	Modification
Mono	Monostable (Multivibrator)
MR	Mixture Ratio
MRD	Measurement Requirements Drawing
MS	Margin of Safety
MSD	Most Significant Digit
MSFN	Manned Space Flight Network
mV	Millivolt
MVS	Majority Voting System
MVB	Multivibrator
MW	Megawatt
N	"Not" Condition, a Binary "0"
NIR	Near Infrared
NPSH	Net Positive Suction Head
NRZ	Non-Return-to-Zero
N_C	Compression Running Load
N_T	Tension Running Load

ABBREVIATIONS (Continued)

OA	Output Doppler
OECO	Outboard Engine Cutoff
OM/D	Orbital Mode/Data
Oper	Operational
OPR	Orbital Processing Routine
P	Probability
PACPS	ST-124M Platform AC Power Supply
PAM	Pulse Amplitude Modulation
PCM	Pulse Code Modulation
PD	Propellant Dispersion
PDD	Program Description Document
PDECO	Propellant Depletion Engine Cutoff
PDP	Program Definition Phase
PDR	Preliminary Design Review
PEA	ST-124M Platform Electronic Assembly
PIRN	Preliminary Interface Revision Notice
PIO	Process Input Output
PM	Phase Modulation
PRF	Pulse Repetition Frequency
PRN	Pseudo-Random Noise
PSK	Phase Shift Keyed
PSR	Parallel Storage Register

ABBREVIATIONS (Continued)

PTL	Prepare to Launch
PU	Propellant Utilization (System)
Q	Aerodynamic Pressure
Q/D	Quick Disconnect
R	Stage Reliability
R	Reset
r(or)R	Vehicle Position
r_o	Initial Velocity
RACS	Remote Auto. Calib. System
RAM	Reliability Analysis Model
RASM	Remote Analog Submultiplexer
RCS	Reaction Control System
RDSM	Remote Digital Submultiplexer
REM	Reliability Engineering Model
RET	Return
RF	Radio Frequency
RFA	Redundant Force Analysis
rms	Root Mean Square
RPM	Revolutions
RSO	Range Safety Officer
RTC	Reasonable Test Constant
RUNV	Start Signal - Voted

ABBREVIATIONS (Continued)

S	Set
S&A's	Safe and Arm Devices
SA	Spin Axis
SC	(S/C) Spacecraft
SCCS	Saturn Command and Control System
SCFH	Standard Cubic Feet per Hour
SCFM	Standard Cubic Foot per Minute
SCN	Specification Change Notice
SCO	Subcarrier Oscillator
SE&I	Systems Engineering and Integration
Servo	Servomechanism
SDH	Simplex Driver - High
SDI	Simplex Driver - Intermediate
SDL	Simplex Driver - Low
SG	Signal Generator
SLA	Spacecraft Launch Vehicle Adapter
SNAP	Stage Networks Acceptance Program
SPTS	Solid Propellant Turbine Starter
SR	Shift Register
SRA	Spin Reference
SS	Single Sideband
STC	Sacramento Test Center

ABBREVIATIONS (Continued)

STDV	Start Tank Discharge Valve
STE	Special Test Equipment
STP	Standard Temperature and Pressure
SUBS	Subsequent
SYNC	Synchronous
TBD	To be Determined
TCC	Test Control Center
TCD	Test Control Drawings
TCS	Thermal Control System
TD	Time Delay
TDH	TMR Driver - High Current
TDL	TMR Driver - Low Current
TDM	TMR Driver - Medium Current
TEMP	Temperature
TG	Torque Generator
TLC	Simultaneous Memory Errors
TLCV	Two Simultaneous Memory Errors - Voted
TM	Telemetry
TMR	Triple Modular Redundnace
TP&D	Test Planning and Evaluation
TRS	Transfer Register Serial
TRSV	Transfer Register Serial - Voted

ABBREVIATIONS (Continued)

TVC	Thrust Vector Control
TWT	Traveling - Wave Tube
UHF	Ultra-High Frequency
ULT	Ultimate
USB	Unified S-Band
VAB	Vehicle Assembly Building
VCO	Voltage-Controlled Oscillator
VHF	Very-High Frequency
VSWR	Voltage Standing Wave Ratio
WACS	Workshop Attitude Control System
W/M	Water/Methanol
X	Roll Axis
Y	Pitch Axis
Z	Yaw Axis

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FOREWORD

This document, the Systems Description Document of the J-2S Improvement Study, is one of eight in a series documenting the results of a thirteen-month study program, performed under National Aeronautics and Space Administration Contract NAS8-5608, Schedule II, Part VII, Task 3.0. The study effort was supervised and administered by the Marshall Space Flight Center. The purpose of the study was to determine the improvements in stage and/or vehicle operating simplicity, reliability, payload capability, mission flexibility and operational costs resulting from the incorporation of J-2S engines in Saturn V. Stage data were supplied by: The Space Division of North American Rockwell (S-II Stage); McDonnell Douglas Corporation (S-IVB Stage); and the Federal Division of IBM Corporation (I. U. and astrionic systems). The Rocketdyne Division of North American Rockwell Corporation supplied J-2S engine data. The Boeing Atlantic Test Center, Southeast Division provided launch operations impact data. The Boeing Huntsville Operation, Southeast Division provided S-IC stage data, and was the System Engineering and Integration (SE&I) contractor for the study.

The study compares the impact of incorporating the J-2S engine in the Saturn V vehicle to a standard baseline Saturn V vehicle (SA-511) with J-2 engines and in addition describes other system and vehicle changes required for missions other than LOR. This document provides a condensed compilation of system changes, schedules, and costs from the S-II, S-IVB and IU stage contractors; J-2S description from the engine contractor; launch operations impact and schedules from the KSC contractor; and integrated total systems assessment, master schedule, and costs from the SE&I contractor.

Program documentation includes this Systems Description Document, D5-15772-2, and other documents as listed below:

D5-15772-1	Summary
D5-15772-2	Systems Description
D5-15772-3	Mission/Performance Analysis
D5-15772-4	Upper Stage Guidance
D5-15772-5	Vehicle Control and Control System Dynamics Characteristics
D5-15772-6	Structural Analysis
D5-15772-7	Resource Analysis
D5-15772-8	Retrofit Analysis

Primary study effort was expended in defining requirements for in-line incorporation of J-2S engines in SA-518 and subsequent Saturn V vehicles. These results are reported in documents D5-15772-1 through D5-15772-3 and D5-15772-5 through D5-15772-7. Results of a study to define Iterative Guidance Mode Scheme

FOREWORD (Continued)

modifications for rendezvous missions with the S-II and S-IVB stages incorporating J-2S restart and throttling capabilities are reported in D5-15772-4. Requirements for retrofit of assembled and stored Saturn V SA-511 with J-2S engines are reported in D5-15772-8.

Associate stage contractors and KSC launch operations contractor investigations detailing the effect of the J-2S/Saturn V design configurations and vehicle environments on their system, operation and resource requirements are reported in the following documents:

S-II STAGE IMPACT:

SD-69-82-1	J-2S Improvement Study, Phase I: In-line Implementation

S-IVB STAGE IMPACT:

DAC-56749	J-2S Implementation on the Saturn V/S-IVB Stage (LOR and Synchronous Orbit Missions)

IU STAGE IMPACT:

69-K44-0001	Final Report (Assessment of Astrionic System and Instrument Unit Impact for J-2S Implementation on Saturn V Vehicles)

LAUNCH OPERATIONS IMPACT:

D5-16793	Study of J-2S Engine Impact on Saturn V Launch Operations

SECTION 1
INTRODUCTION

1.0 GENERAL

The Saturn V is currently designed to use J-2 engines in accomplishing its assigned Apollo/LOR mission. If simplified, more flexible engines (J-2S) replaced the current J-2 engines, improvements would be realized in payload capability, mission flexibility and system cost effectiveness.

The general objective of this study was to assess total system impact and to determine resource requirements when J-2S engines replace J-2 engines in the Saturn V vehicle. The J-2S engine has idle mode and three-start operating capabilities and when compared to the J-2, provides greater thrust and higher specific impulse. These features offer improvements in stage and vehicle simplicity, reliability, operational cost, payload capability and mission flexibility.

The thirteen month study program was divided into two phases (see Study Logic Diagram, Figure 1.0-1). The eight week Phase I activity was directed toward conducting vehicle trades for determining the basic vehicle configurations which would be studied in detail in Phase II. Trade studies were made of missions/trajectory profiles, payload size and shape, engine thrust, etc. Four mission/vehicle configurations were selected from these trades as basic vehicles for detailed design analysis in Phase II. The LOR mission was used for measuring the advantages or disadvantages associated with incorporating J-2S engines into the Saturn V. Other candidate missions were selected for measuring J-2S engine associated changes related to mission peculiar requirements.

Phase II activity was directed toward defining and documenting in detail the four selected vehicle/mission configurations, their performance capability, flight environment, necessary vehicle/stage modifications, GSE and facilities modifications and KSC launch operations impact and comparing them to the Baseline SA-511 Saturn V (J-2 engines). Schedules and costs for development and implementation of the design vehicles into the space program were defined. This document provides a consensed compilation of total study results.

Data as reported in this document were prepared by: The Space Division, North American Rockwell Corporation (S-II Stage); McDonnell Douglas Corporation (S-IVB Stage); The Federal Division, IBM Corporation (I.U. and Astrionic Systems); The Rocketdyne Division, North American Rockwell Corporation (J-2S Engine Data); The Boeing Atlantic Test Center, Southeast Division (Launch Operation); and, The Boeing Company-Huntsville Operation (S-IC Stage and System Engineering and Integration).

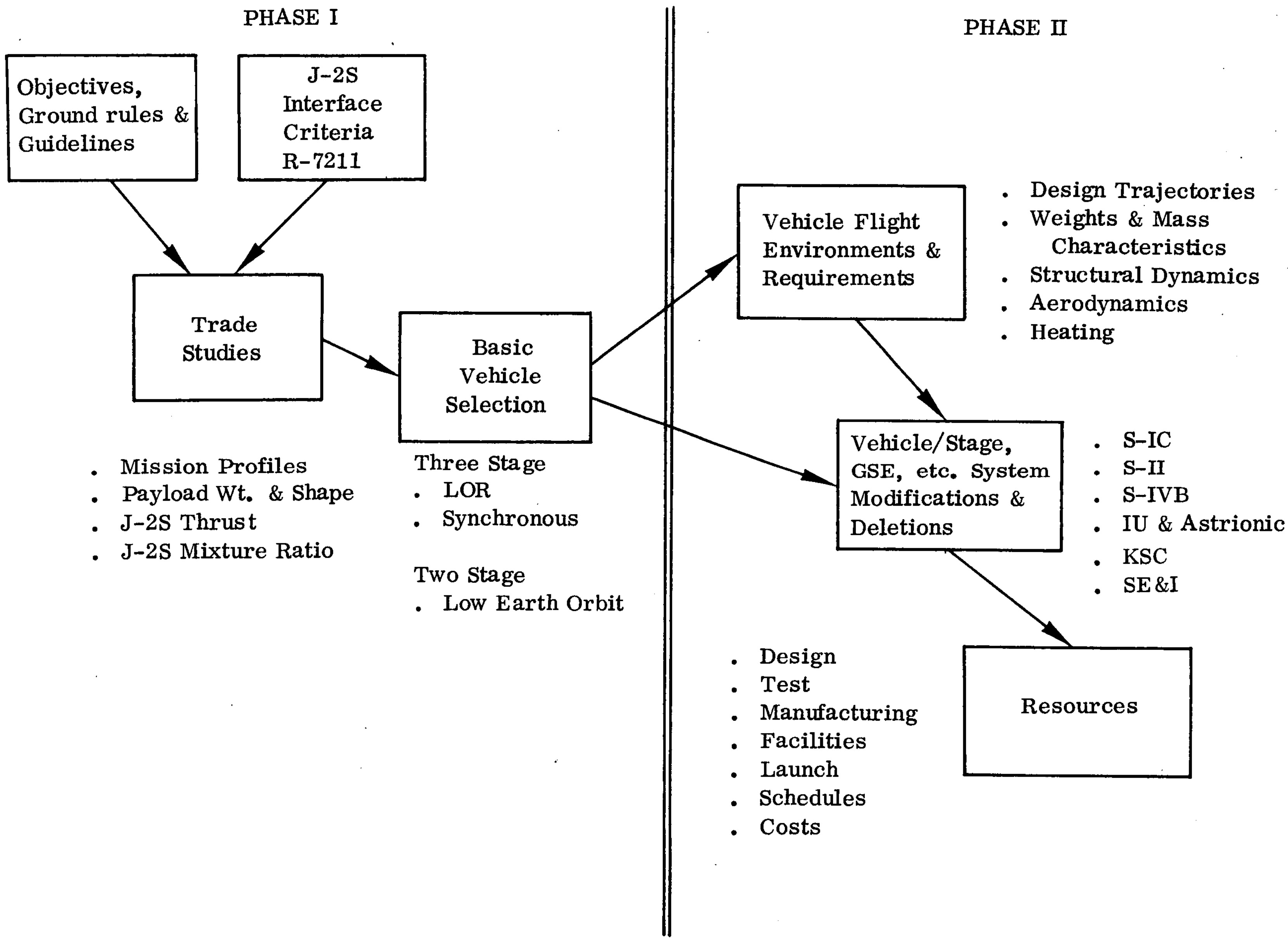


FIGURE 1.0-1 STUDY LOGIC DIAGRAM

SECTION 2

SUMMARY

2.0 GENERAL

Incorporation of J-2S engines in Saturn V increases systems simplicity and reliability, and results in cost savings. Improvements in engine thrust, specific impulse, and operational features results in greater payload capability and mission flexibility.

2.1 OBJECTIVES

The objective of this study is to determine and report the impact of using J-2S engines in the S-II and S-IVB stages of the Saturn V launch vehicle. The improvements in stage/vehicle simplicity, reliability, payload capability, mission flexibility and operational costs were to be identified. The stages and program definitions are to be in sufficient detail to enable writing of contract end items detail specifications during a Phase C Program Definition Phase.

2.2 J-2S ENGINE DESCRIPTION

The J-2S engine is a complete, integrated package of minimum cylindrical dimensions designed to operate at any precalibrated nominal vacuum thrust level between 230,000 and 265,000 pounds with a minimum instantaneous specific impulse of 427 seconds. The engine uses liquid oxygen oxidizer and liquid hydrogen fuel as propellants and is calibrated at a mixture ratio (O/F) of 5.5. Engine mixture ratio can, however, be varied over the range of 4.5 to 5.5 for the purpose of stage propellant management. The engine is capable of being used for single or multi-engine installations on an interchangeable basis. The J-2S engine general arrangement is shown in Figure 2.2-1.

The engine has a single thrust chamber with an expansion ratio of 40:1 and is capable of being gimbaled to provide thrust vector control. The engine can be operated in an idle mode (0-5000 pounds nominal vacuum thrust) for a maximum of 1000 seconds and a mainstage mode for a maximum of 500 seconds from stage command signals. The engine can be shutdown from mainstage to engine cutoff (zero thrust), from mainstage to idle mode operation, and from idle mode to engine cutoff by appropriate stage command signals. It also can be shut down safely as a result of stage liquid oxygen depletion. The engine is capable of restarting twice to mainstage while at orbital conditions without inflight servicing for periods up to 20 hours (two 10-hour orbital coasts).

For flight use, the engine requires no temperature preconditioning from ground equipment or from the vehicle prior to initial start or restart. Idle mode operation conditions the hardware prior to mainstage during engine restart. The engine requires that the user monitor only one engine parameter as a prelaunch "redline" (helium tank pressure) for flight operational use, providing minimum propellant turbopump inlet requirements are met. The engine has a service life of 3750 seconds of engine mainstage mode operation and 3500 seconds of engine idle mode operation. The service life of the engine includes 30 engine starts to mainstage and ten starts to the idle mode thrust level only.

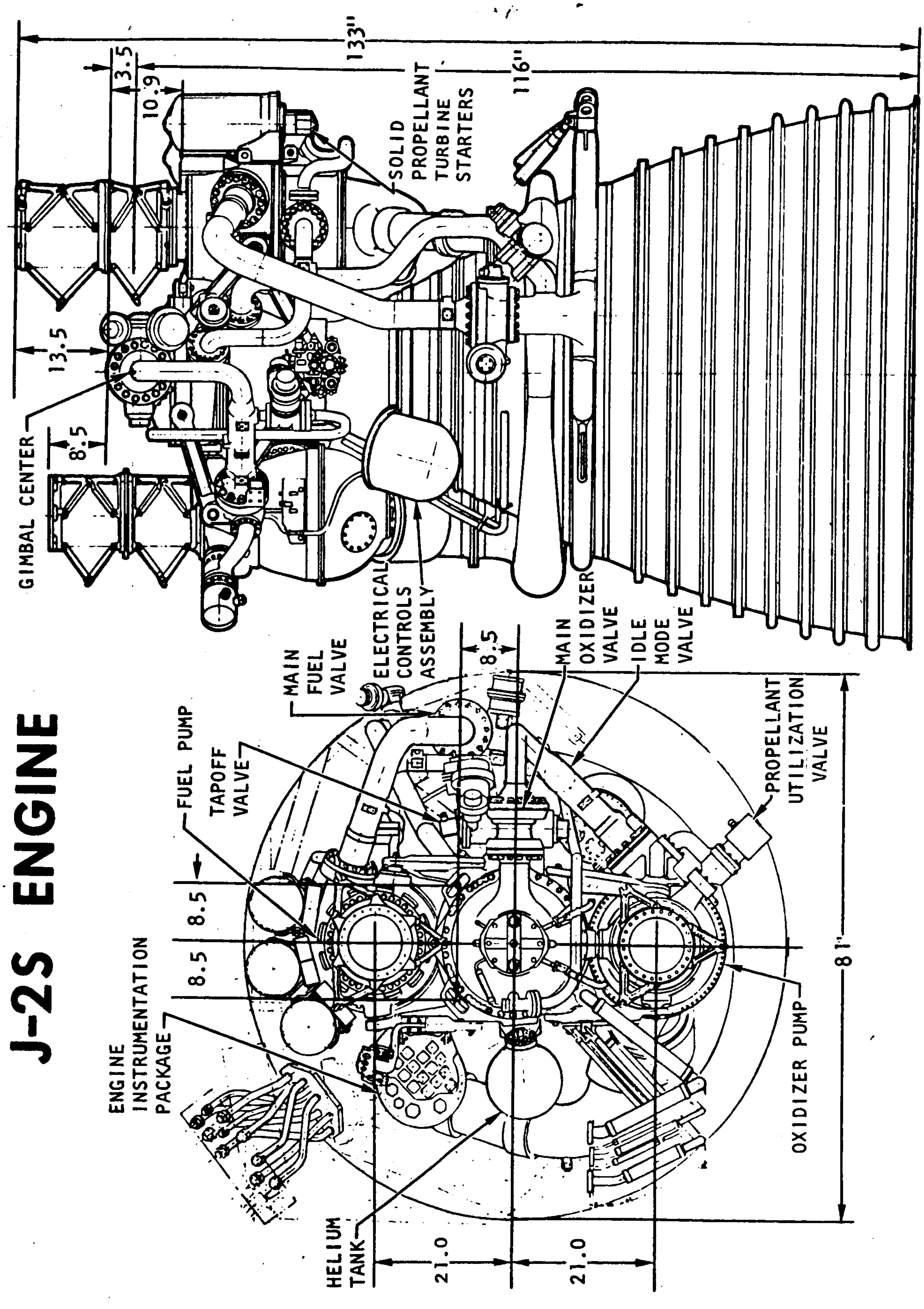


Figure 2.2-1 General Arrangement

2.3 TRADE STUDIES AND CONFIGURATION SELECTION

The purpose of Phase I trade studies was to determine the J-2S/stage system requirements and vehicle/payload configurations that would have application in the time period predicted for J-2S/Saturn V. Four design vehicle configurations were selected for detailed study in Phase II. These vehicle configurations utilize the J-2S engine features without major structural or systems changes to Saturn V stages or support operations.

2.3.1 J-2S Thrust Trades

Thrust trades were conducted to determine the preferred engine thrust calibration level. As a result, J-2S engines calibrated to 265,000 pounds thrust at a mixture ratio of 5.5 were selected for both the S-II and S-IVB stages. The S-II is thrust sensitive for a LOR mission, while the S-IVB is not. However, the S-IVB is thrust sensitive when used for other missions. The increase in the S-IVB thrust structure weight of only 75 pounds to accommodate the J-2S 265,000 pound thrust engines, and the $56,500 development and zero recurring costs were considered acceptable.

2.3.2 J-2S Mixture Ratio Trades

Mixture ratio operating levels in both the S-II and S-IVB stages were optimized for the LOR mission. Operation with mixture ratio schedules and operation with constant mixture ratios were compared. Performance was determined with, and without, maximum utilization of impulse propellant. The S-II stage performance was optimized when operating with a mixture ratio schedule of 5.0 to 5.5 to 4.5 with the shift times selected to maximize impulse propellant. The S-IVB stage performance was optimum when operating at a constant mixture ratio selected to maximize impulse propellant.

2.3.3 Missions, Flight Profiles, and Payload Envelopes Trades

Candidate vehicle/payload configurations and mission flight profiles were examined to establish engine/stage operational sequence requirements and vehicle load criteria. The predicted J-2S/Saturn V missions were LOR, Synchronous, two-stage low-Earth coplanar, and two-stage low-Earth polar. Candidate flight profiles and payload envelopes for each mission were examined to obtain a reasonable compromise between J-2S engine/stage systems modifications and mission peculiar requirements. The four design vehicle configurations selected from these trade data for Phase II study are:

a. J-2S/Saturn V (Lunar Mission) - The lunar orbital rendezvous (LOR) flight profile was selected. The standard Apollo payload was selected for the payload envelope. This shape maintains the Saturn V vehicle geometry

2.3.3 (Continued)

and aerodynamic characteristics. The increased payload weight was assumed to be uniformly distributed within the current Apollo spacecraft. Structural loads were expected to increase due to the increased payload weight, but should be within baseline SA-511 stage capability. Subsequent analysis showed this to be true (para. 2.4.5).

This design vehicle eliminates mission peculiar requirements and gives only program requirements when J-2 engines are replaced with J-2S engines. The other design vehicles (b, c, and d) identify program requirements which are identified as "mission peculiar".

b. J-2S/Saturn V (Synchronous Orbit Mission) - The design flight profile selected was a three-stage boost to 100 N M parking orbit, coast for five revolutions in the parking orbit, S-IVB mainstage burn out of parking orbit into the Hohmann transfer ellipse, coast to synchronous orbit altitude, and S-IVB third mainstage burn for orbit circularization and plane change. This profile is a "worst case" condition for stage systems modifications and payload capability. However, it is a "best case" condition for mission flexibility. The payload envelope used was the Apollo shape. Since payload weight to synchronous orbit will be less than LOR, a reasonable payload density will be maintained. Design loads were expected to be less than LOR due to the decrease in payload weight and the use of identical payload envelopes.

c. J-2S/Saturn V (Low-Earth Orbit Mission) - The flight profile selected was direct S-IC/S-II boost (no parking orbit) to Hohmann transfer orbit perigee at 100 N M, coast to 300 N M apogee, and S-II start in idle mode for orbit circularization. This profile demonstrates the usefulness of the J-2S idle mode feature for mission impulse maneuvers. Use of J-2S idle mode for orbit circularization mininizes S-II stage modifications since a repressurization system, required for S-II mainstage restart, is not needed. The payload shape selected was the S-IVB/Apollo shape. This shape is expected to satisfy postulated manned mission volume requirements. Since this payload envelope has the same geometry as the three-stage Baseline SA-511, it has minimal effect on stage structural loads and thus should result in none or minimum structural modifications. The payload weight was assumed to be uniformly distributed in the payload volume. Subsequent analysis showed that no structural modifications were required (Para. 2.4.5).

d. J-2S/Saturn V (Polar Orbit Mission) - The flight profile selected was two-stage direct injection to a 100 N M altitude orbit. Yaw steering of the S-IC and S-II stages was used to avoid spent stage impacts on major land masses. The payload shape selected was identical to the payload shape selected for the low-Earth orbit mission. Subsequent analysis showed that minor structural modification might be required (Para. 2.4.5).

2.4 VEHICLE ENVIRONMENTS

2.4.1 Vehicle Description

Two three-stage configurations and two, two-stage configurations were selected for detail study. Both versions have the Baseline Saturn V SA-511 external configuration. The Apollo payload shape was used for the three-stage vehicles and the "S-IVB Workshop"/Apollo payload shape was used for the two-stage vehicles. Each configuration is an assembly of SA-511 type stages modified only for installation of the J-2S engine or to withstand increased structural loads.

Analyses were conducted to define the flight environments of these four design vehicles and to determine the vehicle systems requirements for the LOR, LEO, and two-stage Polar orbit missions. All J-2S engine associated changes are referenced to the Baseline SA-511 Saturn V vehicle.

2.4.2 Design Trajectories

Flight simulations were made for the four design mission/flight profiles selected in the Phase I trade study. The design trajectories are nominal trajectories in that they do not include 3 σ dispersions. They identify payload capability and vehicle design characteristics for structural, control, and heating analyses.

A comparison of payload capability and vehicle design characteristics of the Baseline SA-511 vehicle and the J-2S/Saturn V design vehicles are made in Table 2.4-I. The LOR payload is increased approximately 9 percent when using J-2S engines. The load criteria parameters for the synchronous vehicles are bounded by the LOR and LEO vehicle parameters. Since the structural loads of these vehicles do not exceed capability, it was considered unnecessary to perform a synchronous vehicle loads analysis.

2.4.3 Aerodynamics

All basic aerodynamics data were provided by MSFC. The external shape of all J-2S design vehicles is identical to the Baseline SA-511 vehicle. The applicable Saturn V aerodynamics were used.

2.4.4 Acoustics

The acoustic environments for the design vehicle were provided by MSFC. Although these environments are somewhat greater than Baseline SA-511 acoustics, they are within specification limits and no modifications are required.

TABLE 2.4-I
VEHICLE DESIGN CHARACTERISTICS

	J-2/SATURN BASELINE SA-511 LOR	J-2S/SATURN LOR	J-2S/SATURN V SYNCHRONOUS	J-2S/ SATURN V LEO	J-2S/ SATURN V POLAR
PAYLOAD (LBS)	100,078	110,066	65,991	247,224	215,818
LOAD CRITERIA					
MAX Q (LBS/FT^2)	699	709	723	746	698
G'S AT MAX Q	2.08	2.09	2.1	2.13	2.07
MAX G	4.15	4.14	4.23	4.41	4.49
HEIGHT (FT)	363.0	363.0	363.0	363.0	363.0
WEIGHT ABOVE S-II (LB) (INCLUDES IU)	364,330	374,950	328,220	251,407	220,000
AEROHEATING (AHI) ($FT-LBS/FT^2-RAD$)	46.43×10^6	+5.74%	+9.17%	+14.82%	-7.0%
PROPELLANT LOADING (LB)					
S-IC	4,577,113	4,577,113	4,577,113	4,577,113	4,577,113
S-II	970,441	970,441	970,441	970,441	970,441
S-IVB	227,991	229,097	219,330		
THRUST (LBS.)					
S-IC	7,610,064	7,610,064	7,610,064	7,610,064	7,610,064
S-II	1,149,635	1,318,850	1,318,850	1,318,850	1,318,850
S-IVB	231,012	237,500	265,000		

2.4.5 Structural Loads and Dynamics

The ultimate compressive design loads (N_C) along with structural capabilities are shown in Figure 2.4-1. These loads are envelopes of design loads developed for the vehicle design conditions (Rebound, q α max, g max, etc.). Structural loads are less than capability for all J-2S/Saturn V vehicles except in the S-IC fuel tank for the two-stage polar orbit mission. Structural modification would increase S-IC weight approximately 74 pounds. Revised mill tapes would be needed to increase structure cross-section. Another approach would be to use load alleviation techniques to reduce ultimate compressive loads. These could consist of trajectory shaping, restricting launch availability and restricting design criteria.

Structural dynamics (lateral vibrations) properties were computed at the critical times during boost using the stage mass properties supplied by stage contractors. These data were used in the structural loads and vehicle control analyses.

The structural effects of one engine-out and all-engine out malfunctions during maximum qα product flight region was determined although not imposed as a design requirement or constraint. Structural failure results from vehicle tumbling for an all-engine-out malfunction and for a one-engine-out malfunction on either engines one or two. Structural failure was defined as the condition when the factor of safety was reduced to one. For all failure cases, the time-to-failure was calculated to give an indication of mission abort requirements. The minimum time-to-failure for the uncontrollable vehicle was 0.25 seconds. Vehicle control was maintained for a one-engine-out malfunction on either engines 3 or 4. However, structural tension capability was exceeded in the S-II aft skirt. The minimum time-to-failure for this condition was 0.4 seconds.

2.4.6 Heating Environment

Thermal design criteria were provided by MSFC. A comparison of relative aerodynamic heating of geometrically similar vehicles with dissimilar trajectories can be made by comparing aerodynamic heating indicator (AHI) values. Vehicles with the same AHIs are assumed to have comparable aerodynamic heating environment. A comparison of J-2S design vehicles and Saturn V vehicle AHIs shows that Saturn V thermal environment are applicable for the J-2S design vehicles.

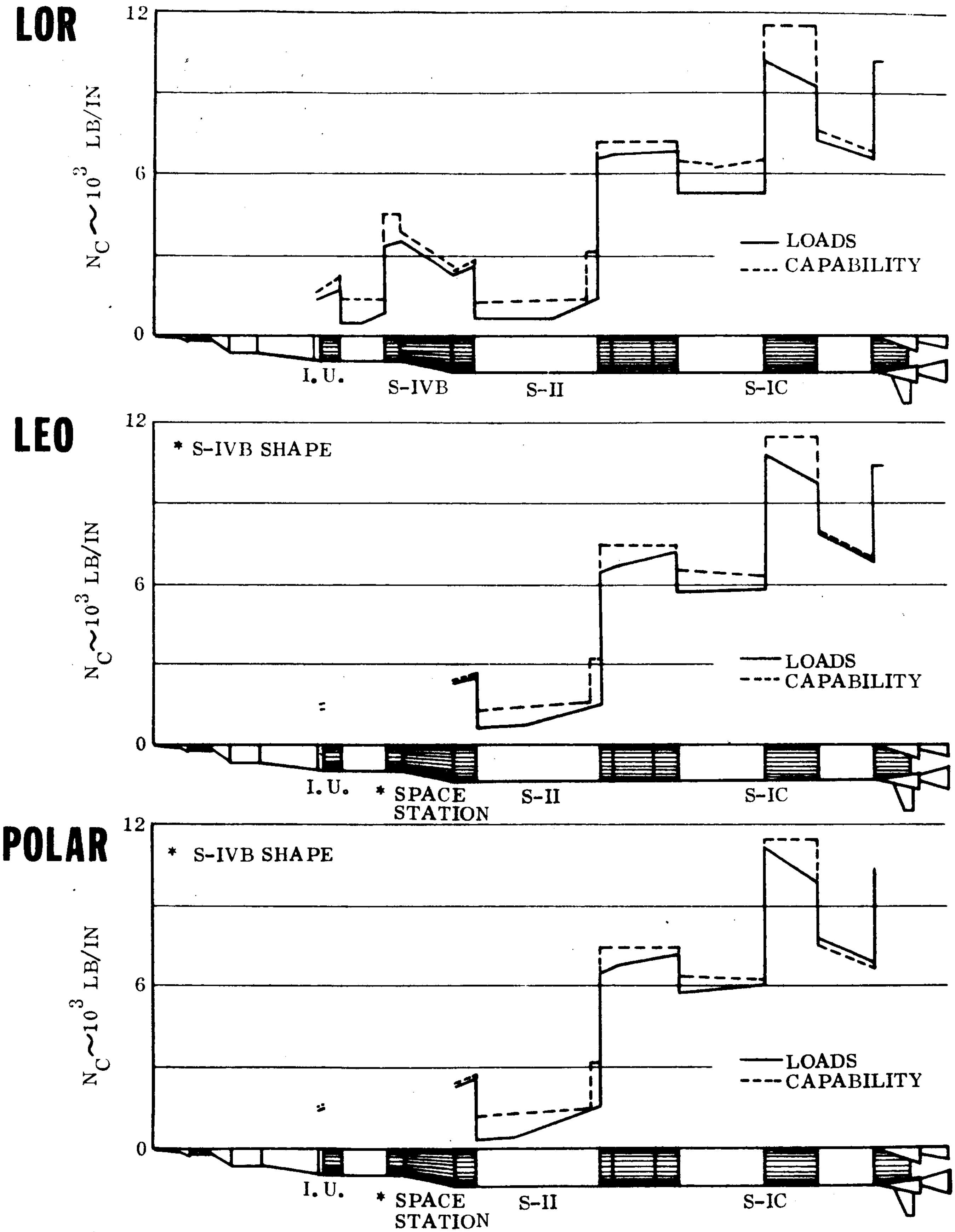


FIGURE 2.4-1 J-2S/SATURN V ULTIMATE COMPRESSIVE DESIGN LOADS (N_C)

2.4.7 Vehicle Weights and Mass Characteristics

Control of all the J-2S/Saturn V design vehicle weights was maintained. These data consist of integrated vehicle mass properties, drop weights, fuel residual summaries, and stage dry weights. The final stage weight summary for the four J-2S/Saturn V vehicles is shown in Table 2.4-II. The Baseline SA-511 weights are shown for comparison. These stage weights reflect system and structural modifications resulting from stage design changes described in Paragraph 2.6, Vehicle/ Stage Descriptions.

2.4.8 Vehicle Control

Rigid body control requirements and control system dynamic characteristics analyses were conducted for the LOR, LEO and two-stage polar orbit design vehicles.

Rigid body control system gains and thrust deflection requirements, including effects of scatter parameters during first stage flight were determined. The maximum required F-1 thrust deflection angle of 2.8 degrees occurred for the LEO vehicle. This is within the Saturn V maximum capability of 5.15 degrees.

Control system dynamic characteristics were determined by generating uncompensated control frequency response Nyquist and Bode plots. This data was provided for the most unstable boost phase times. The plots indicate the controllability of the design vehicles and the complexity of the control system compensator networks required to stabilize the vehicle. The J-2S design vehicles uncompensated frequency response characteristics were compared to uncompensated Saturn V data and show that the compensation required is within the Saturn V design.

TABLE 2.4-II
FINAL STAGE WEIGHT SUMMARY

DESCRIPTION	WEIGHT - LBS SA-511 BASELINE	LOR MISSION	SYNCH MISSION	LEO MISSION	POLAR MISSION
S-IC					
STRUCTURE	140,835	140,835	140,835	140,835	149,909
PROPULSION SYST. & ACCESS.	139,525	139,525	139,525	139,525	139,525
EQUIPMENT & INSTRUMENTATION	8,618	8,618	8,618	8,618	8,618
GROWTH	431	431	431	431	431
TOTAL DRY WEIGHT	289,409	289,409	289,409	289,409	289,483
RESIDUAL WEIGHT	67,585	67,585	67,585	67,585	67,585
S-IC/S-II INTERSTAGE					
SHORT INTERSTAGE	1,548	1,548	1,548	1,548	1,548
LONG INTERSTAGE	9,435	8,336	8,336	8,336	8,336
S-II					
STRUCTURE	47,064	49,013	49,013	49,203	49,013
PROPULSION SYST. & ACCESS.	26,439	26,472	26,472	27,943	26,472
EQUIPMENT & INSTRUMENTATION	6,765	6,645	6,645	7,085	6,645
GROWTH	418	418	418	418	418
TOTAL DRY WEIGHT	80,686	82,548	82,548	84,649	82,548
RESIDUAL WEIGHT	14,134	12,735	12,735	10,250	12,735
S-II/S-IVB INTERSTAGE	7,021	7,021	7,021	NOT APPLICABLE TWO-STAGE VEHICLES	
S-IVB					
STRUCTURE	13,241	13,235	13,544		
PROPULSION SYST. & ACCESS.	7,263	7,539	6,845		
EQUIPMENT & INSTRUMENTATION	4,580	4,176	4,958		
TOTAL DRY WEIGHT	25,084	24,950	25,347		
RESIDUAL WEIGHT	2,272	2,795	2,640		
I.U.					
STRUCTURE	621	705	710	705	705
EQUIPMENT & INSTRUMENTATION	3,266	3,297	3,623	3,297	3,297
SERVICE ITEMS	296	299	342	299	299
TOTAL DRY WEIGHT	4,183	4,301	4,675	4,301	4,301

2.5 PERFORMANCE/MISSION ANALYSIS

This section presents mission payload capability of the four J-2S/Saturn V design vehicles described in Paragraph 2.4.1. Comparisons are made with the Saturn V using J-2 engines (Baseline SA-511).

2.5.1 Lunar Missions

The initial J-2S LOR design trajectory, reported in Paragraph 2.4.2 was based upon initial best estimates of stage changes required to incorporate the J-2S engine. After determination of final stage modifications, a final J-2S LOR payload was calculated. This payload which reflects the stage dry and residual weight changes is 109,000 pounds as compared to 100,078 pounds for the Baseline SA-511 (J-2 engines), a nine percent gain. Most of the nine percent payload gain is due to increased J-2S thrust and higher specific impulse.

Direct injection lunar payload is 115,320 pounds. This flight profile does not use an Earth parking orbit and results in a reduced launch window but may be acceptable for unmanned missions. J-2S direct injection payload is six percent greater than J-2S LOR payload. A comparison of J-2 LOR, J-2S LOR, and J-2S direct injection payload is shown in Table 2.5-I.

2.5.2 High Energy Missions

The flight profile for high energy missions is the same as the LOR flight profile except for the 90° launch azimuth. Payload capabilities as a function of C_3 (twice the specific energy) are shown in Figure 2.5-1 for a LOR type vehicle. Shown in the figure are approximate energy levels for representative missions.

2.5.3 Synchronous Orbit Missions

The design synchronous orbit mission used five revolutions in parking orbit and an equatorial (0°) orbit inclination. These restraints imposed "worst case" conditions on vehicle payload capability. When these constraints are relieved by varying the inclination and decreasing the number of parking orbit revolutions, the payload increases are as shown in Figure 2.5-2. For a 28.5 degree orbit inclination, the payload is increased to 75,800 pounds, a payload gain of 15 percent over the 66,000 pound design synchronous orbit payload. Reducing the revolutions in parking orbit to one, increased the payload to 70,500 pounds.

The J-2/Saturn V payload is 59,500 pounds to an equatorial orbit with a five revolution parking orbit. The J-2 performance assumed the availability of S-IVB three-start capability without penalty. The J-2S/Saturn V payload of 66,000 pounds is an eleven percent increase.

TABLE 2.5-I LUNAR MISSION PAYLOAD CAPABILITY

J-2 LOR BASELINE	J-2S LOR	J-2S DIRECT
100,078 LBS.	109,000 LBS	115,320 LBS

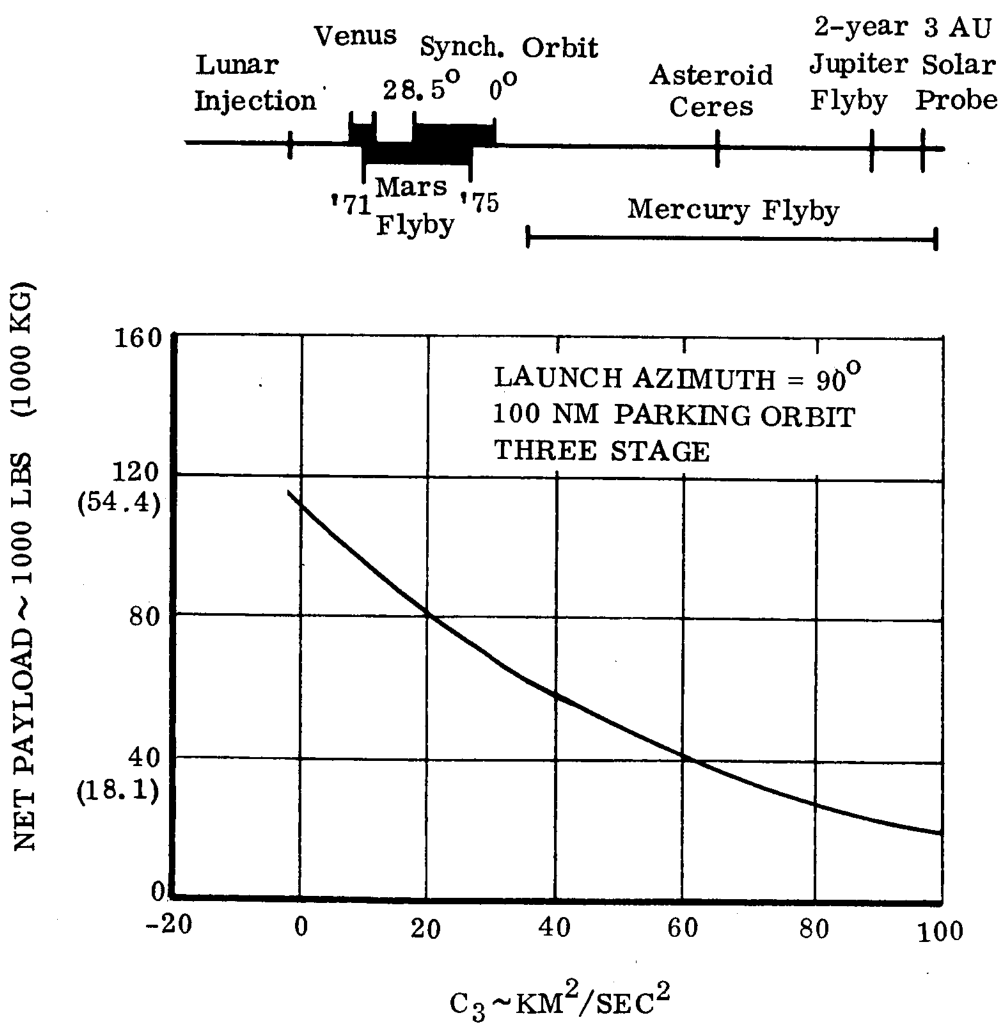
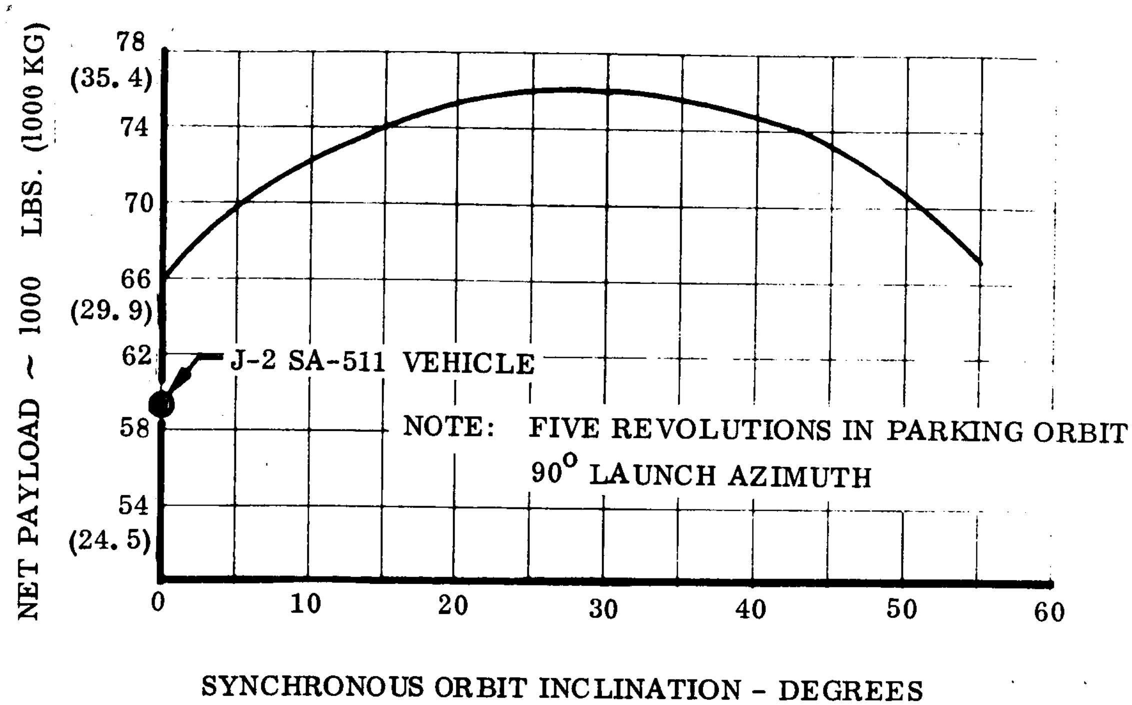


FIGURE 2.5-1 J-2S/SATURN V HIGH ENERGY PERFORMANCE CAPABILITY THREE-STAGE VEHICLE



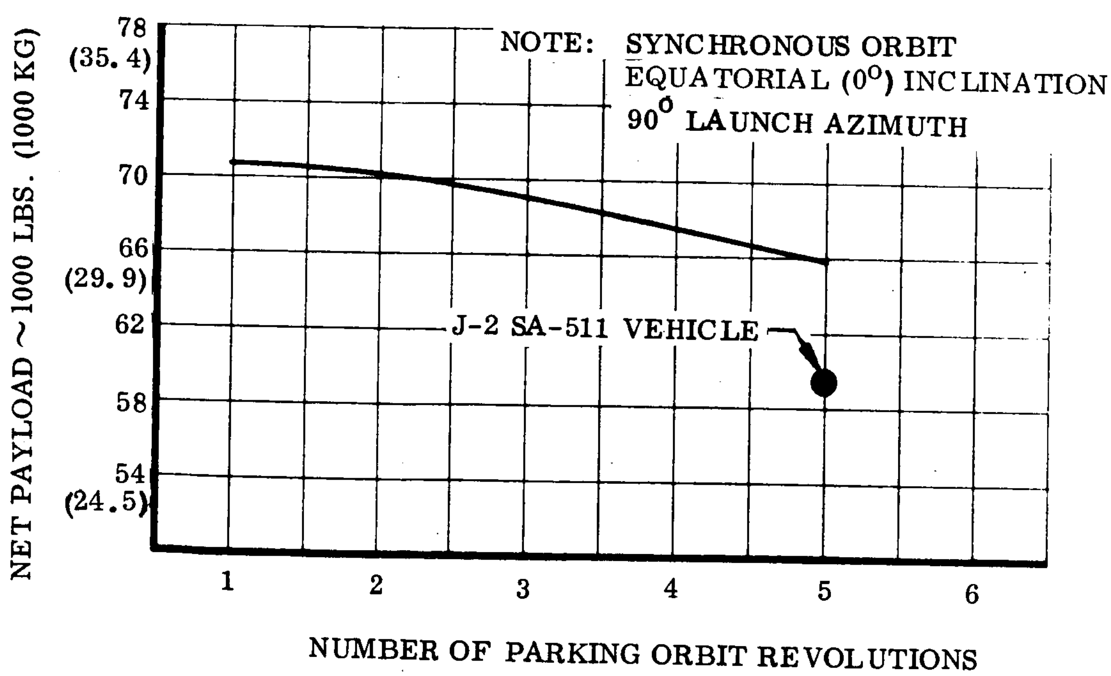


FIGURE 2.5-2 J-2S/SATURN V SYNCHRONOUS ORBIT PERFORMANCE

2.5.4 Low Earth Orbit Missions - Two Stage

2.5.4.1 Coplanar (LEO)

Two-stage (S-IC/S-II) J-2S and J-2 direct injection payload capabilities to orbit altitudes up to 500 NM are compared in Figure 2.5-3. The data was generated using 265K thrust J-2S engines. No attempt was made to optimize the thrust calibration level as a function of orbit altitude. Lower thrust calibration levels would increase payload performance at the higher orbital altitudes.

J-2S Hohmann transfer payload capability is also shown in Figure 2.5-3. At 300 NM altitude, payload capability with Hohmann transfer flight profile is approximately 50 percent greater than with direct injection. S-II stage modification is not required to support the J-2S/S-II idle mode operation used for orbit circularization at Hohmann ellipse apogee. To perform the same orbit circularization with a J-2/S-II stage, major S-II modification would be required for a repressurization system to support the J-2 mainstage operation. A S-II or payload RCS system would be required by both J-2 and J-2S vehicles for attitude control during coast.

2.5.4.2 Polar

J-2S Hohmann transfer and direct injection payload capabilities to orbit altitudes up to 300 NM are compared in Figure 2.5-3. A 140 degree launch azimuth was used. The yaw maneuver was initiated at 100 seconds with a yaw rate of one degree per second during S-IC flight. During S-II flight, a constant thrust yaw angle of 44.9 degrees was used. The J-2S payload trends for polar orbit missions are similar to those for coplanar flight. At 300 NM altitude, the J-2S payload capability is increased approximately 50 percent when using the Hohmann transfer flight profile.

The J-2 direct injection payload capability of 201,000 pounds to 100 NM altitude is also shown in Figure 2.5-3. This payload is some 7 percent less than J-2S direct injection payload.

The yaw maneuver required during boost to avoid overflight of major land masses results in significant payload degradation. For a 100 NM orbit, the payload degradation from a LEO coplanar mission is approximately 73,000 pounds.

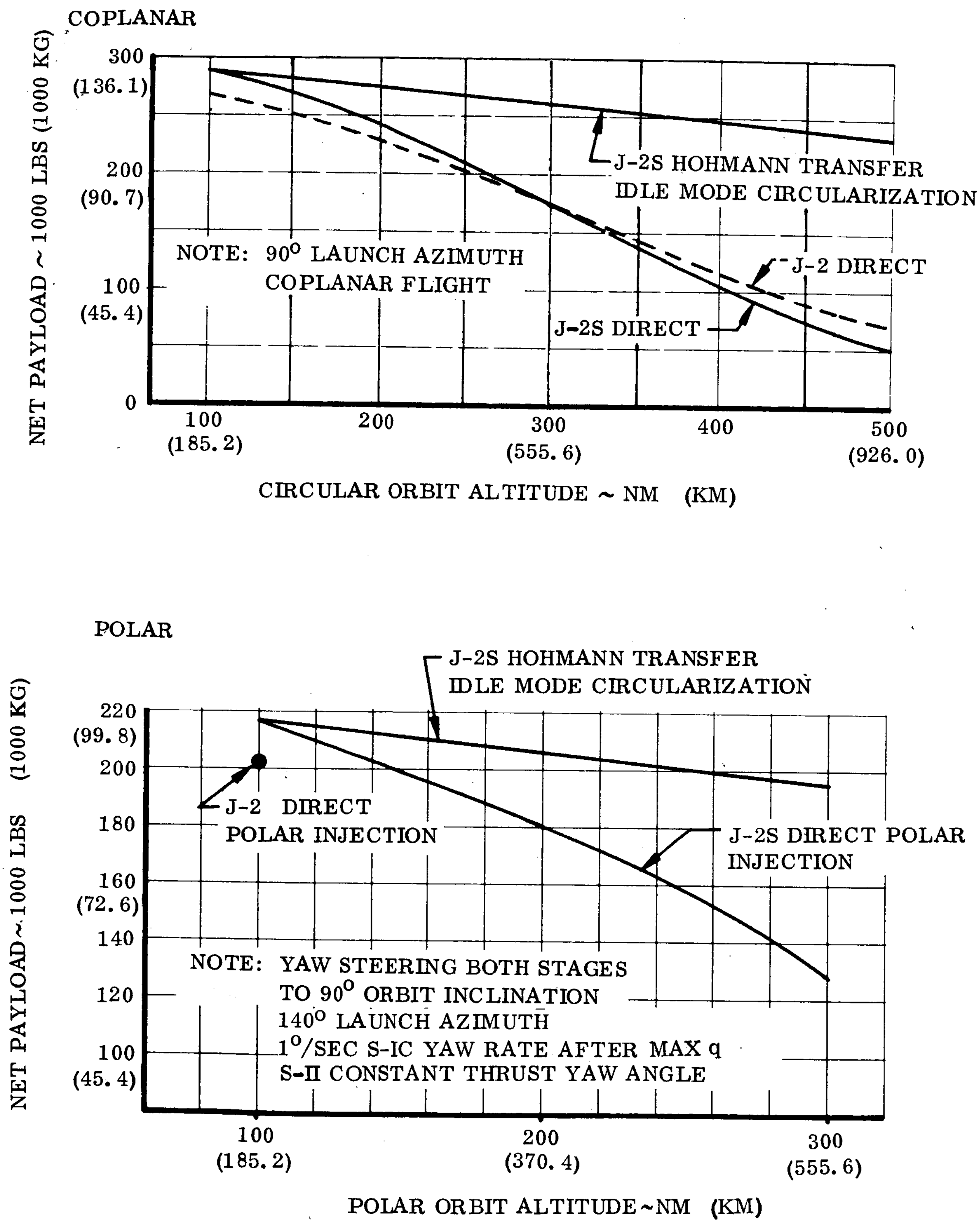


FIGURE 2.5-3 J-2S/SATURN V LOW EARTH ORBIT PERFORMANCE - TWO-STAGE VEHICLE (S-IC/S-II)

2.6 STAGE DESCRIPTIONS

This section summarizes system and operational changes required when J-2S engines are incorporated in Saturn V.

2.6.1 S-IC Stage Description

The S-IC stage is the first stage of the Saturn V launch vehicle. The S-IC is a cylindrical booster 138 feet long and 33 feet in diameter. It is made up of a forward skirt, two cylindrical propellant tanks separated by an intertank structural assembly, the thrust structure, and four aerodynamic fairings and fins. The S-IC is powered by five F-1 engines burning a mixture of RP-1 and LOX. One engine, on the vehicle longitudinal centerline, is fixed; the four remaining engines mounted in a square pattern about the center engine, are gimballed for control. Each F-1 engine provides a sea level thrust of 1.522 million pounds. The thrust structure transmits the engine thrust loads to the propellant tanks.

The S-IC stage interfaces structurally and electrically with the S-II stage. It also interfaces structurally, electrically, and pneumatically with two umbilical service arms, three tail service masts, and certain electronic systems by antennae.

The S-IC stage configuration is identical to the SA-511 S-IC stage. S-IC structural loads are less than structural capability for all design missions except the two stage polar orbit mission. For this mission, structural loads exceed capability in the S-IC fuel tank. Structural modifications to the fuel tank longitudinal stringers and also to the tank sidewall are needed to increase the capability to match the loads. The S-IC weight is increased approximately 74 pounds. However, load allevitation techniques can be used to reduce the design load to the structural capability. These include trajectory shaping, restricting launch availability, and restricting design criteria.

2.6.2 J-2S/S-II Stage Systems Description

This section of the report summarizes the major changes required to the S-II stage/GSE systems and operations required for incorporation of the J-2S engine for the missions studied. The changes for the LOR mission are treated as basic and changes for the other missions are described as additive requirements. The three outstanding changes are redesign of the thrust structure to accommodate the increased thrust of the J-2S engines (all missions), elimination of the stage recirculation systems (all missions), and addition of a reaction control system (RCS) to the stage (LEO mission only).

2.6.2.1 Stage Structural Systems

System	Change
Thrust Structure (Increased engine thrust) (All missions.)	General redesign required to maintain structural integrity and stiffness properties. The approach is to maintain component design and material type but, to increase thickness.
Base Heat Shield (Increased clearance requirements and thermal environment) (All missions.)	Redesign rigid panel for 1/2-inch greater hot side core thickness and addition of wire mesh screen. Relocate support brackets and modify tubes and fittings. Modify flexible curtains to match engine changes.
Tank Skins and Bulkheads (Deletion of recirculation system) (All missions.)	Delete holes and bosses required for recirculation system.
Aft Interstage (Ullage motor deletion) (All missions.)	Delete ullage motor attachment fittings and reinforcements.
Aft Skirt (Addition of RCS) (LEO mission only)	Add holes and reinforcement for RCS monitoring.

2.6.2.2 Stage Propulsion/Mechanical Systems

System	Change
Recirculation System (Characteristics of the J-2S engine permit starting without propellant preconditioning to subcooled conditions.) (All missions.)	The entire system on both LOX and LH_2 feed systems have been deleted.
Engine Servicing System (Purges and drains on the J-2S are revised and/or relocated from the J-2 requirements.) (All missions.)	General redesign to satisfy requirements specified by the engine contractor.
Engine Actuation System (Hydraulic pump operating speed is increased due to an increase in speed by its driver, the LOX pump.) (All missions.)	Component revision within the hydraulic pump to meet flow and speed requirements.
(For vehicle attitude control during idle mode operation (LOX pump is not operating), a hydraulic power source is required.) (LEO mission.)	A new flight auxiliary power system based upon an MDC system is added to the vehicle.
Propellant Management System (The J-2S engine has the capability of performing a safe shutdown by use of the electrical drop-out of the mainstage OK pressure switches.) (All missions)	The stage electrical cutoff circuitry is modified to utilize the low level sensors to arm the mainstage OK cutoff circuit.

System	Change
(Following mainstage operation, propellants are required for an idle mode operation to perform an orbital circularization maneuver.) LEO mission.)	The low level cutoff sensors are raised in the tanks to a level corresponding with the idle mode propellant requirement.
Pressurization System (Heat exchanger output of the J-2S engine is higher than J-2.) (All missions.)	Pressurization manfolds and lines require increased capability, and LOX tank pressurant is bled from four engines rather than five.
(The vehicle requires a small positive acceleration during coast for propellant settling and heat input to the LH_2 tank raises the ullage pressure.) (LEO mission.)	A continuous vent system with axially directed nozzles provides the low acceleration and tank ullage pressure control.
Engine System (The J-2S starting system uses a solid propellant turbine spinner.) (All missions.)	Start tank system deleted and SPTS added to the system.
Ullage Motors (Engine start characteristics of the J-2S do not require the acceleration provided by the ullage motors.) (All missions.)	The ullage motor system and supporting components and systems are deleted in their entirety.
Valve Actuation System (Deletion of the recirculation system reduces the system requirements to one prevalve actuation.) (All missions.)	The pressure requirements of the pneumatic system are reduced to the point where several components may be deleted.

2.6.2.3 Stage Electrical System

System	Change	
Electrical power system	Component deletions:	
(Recirculation system is not required for J-2S engines)	Inverter assembly - recirculation pumps	5
	Recirculation power transfer switch	1
(All missions.)	Recirculation batteries	2
	Inverter filters	5
	Recirculation motor power distributor	1
	Engine ignition power distributor	1
	Total	15
	Component additions:	
(45-minute coast period and Idle Mode Operation)	New main battery	1
	New Instrumentation battery	1
(LEO mission.)	New TVC batteries	2
	New TVC container	1
	New TVC power transfer switch	1
	New TVC motor start switch	4
	Redesign 207 container	1
	Redesign 206A35 container	1
	Total	12
Electrical Control System	Component deletions:	
(Deletion of recirculation and ullage motor systems)	Nonlatching relays	5
	Magnetic latching relays	9
	EBW firing units	2
(All missions)	Pulse sensors	2
	TOTAL	18
(Deletion of LOX tank low level engine cutoff)	Nonlatching relay	10
	TOTAL	10
(All missions.)		

System	Change	
	Component additions:	
(New controls for J-2S engine system)	Nonlatching relays	13
	Magnetic latching relays	4
(All missions)	Total	17
(Controls for the RCS and TVC)	Nonlatching relays	31
(LEO mission)	Magnetic latching relays	11
	Total	42
Instrumentation System	Component deletions:	
(Deletion of recirculation system and ullage motors)	Voltage attenuator	1
	Current monitor	1
(All missions)	Dc/dc transducers	4
	Low level flexible surface transducers	2
	Temperature bridge modules	2
	Total	10
(RCS and TVC system measurement)	Component additions:	
(LEO mission)	Asymmetrical bridges	19
	Dc/dc pressure transducers	5
	Transducer potentiometer	1
	Low level temperature probe	1
	Dc amplifier	1
	Temperature bridge submodule	1
	Low level flexible surface transducers	13
	Low level differential amplifier assemblies	2
	Low level switch assemblies	4
	Decoding matrix assembly	1
	Current monitors	2
	Voltage attenuators	2
	Total	52

System	Change	
(New J-2S engine system measurements and changes in stage thermal environment)	Component additions:	
(All missions)	Low level temperature probes	6
	Low level temperature bridges	10
	Low level flexible surface transducers	9
	Low level dc amplifiers	9
	Low level H-SH temperature submodules	9
	New temperature bridge modules	15
	Total	58
Ordnance system	Component deletions:	
(Deletion of the ullage motor ignition system)	EBW Detonators	2
	CDF Manifolds	2
	CDF Assemblies	9
	Pyrogen Initiators	8
	Total	21

2.6.2.4 Reaction Control System

For the LEO mission, the S-II stage performs an orbital circularization function after a 45-minute coast. A reaction control system (RCS) consisting basically of two S-IVB (Saturn V) type thruster modules mounted opposite each other on the aft skirt is added to provide vehicle attitude control during the coast period. Each S-IVB module contains three hypergolic bipropellant (N_2O_4 - MMH), 150-pound-thrust rocket engines for pitch, roll, and yaw control. Each module also contains a positive expulsion propellant feed system, a 3,000 psig helium pressurization system, valving, lines and electrical control system.

The stage/GSE modification required are shown below.

System	Change
Aft skirt	Add holes and reinforcement for RCS module monitoring. Add insulation to offset aerodynamic and RCS motor impingement heating.
Engine compartment purge system	Provide tap-off for thermal conditioning of RCS modules.

System	Change
Electrical system	Add RCS control and measurement circuitry.
Aft umbilical	Add a single disconnect and engine compartment manifold system to pressurize RCS helium tanks.
GSE servicing system	A mobile servicing console is required for manual servicing, purging, and filling the RCS propellant tanks.

2.6.2.5 Stage Weights

The installation of the J-2S engine will increase the S-II stage dry weight by 1,862 pounds. The major components affected are shown below:

Component	Weight Change (lb)	
	Increase	Decrease
Engine and accessories	1218	
Thrust structure	2000	
Fairing		81
Base heat shield	30	
Purge and leak detection	30	
Fuel recirculation		910
LOX recirculation		305
Electrical system		120
Total	3278	1416
Net increase:	1862	

The S-IC/S-II interstage will have a weight reduction of 1099 pounds resulting from the following:

System	Weight Reduction (lb)
Deletion of ullage motor system	834
Deletion of recirculation system components	155
Decrease in measurement system	70
Decrease in structural elements	40
Total decrease	1099

In addition to the above net stage weight increase of 1862 pounds for J-2S engine installation for an LOR mission, the additional equipment required for the LEO mission will further increase stage weight 2101 pounds, distributed as shown below:

Component	Weight Increase (lb)
Add slosh baffle to LOX tank	150
Additional fairings for RCS and vent systems	40
LH_2 balanced vent system	25
Stage thrust vector control system for idle mode	500
Reaction control system	946
Additions to environmental control system	210
Additions to telemetry and measurement system	200
Additions to electrical system	30
Total increase	2101

Total stage weight increase for a LEO mission is, therefore, 3963 pounds.

2.6.2.6 Ground Support Equipment

The ground support equipment changes required as a result of the stage changes are as follows:

System	Change Description
Engine servicing system (Required by changed engine requirements) (All missions)	Partial deletions or deactivations in: Umbilical Carrier Plate - Aft LH_2 Heat Exchanger Pneumatic Checkout Console Pneumatic Servicing Console Fluid Distribution System Static Firing Skirt Electrical GSE
Recirculation system (Deletion of the stage recirculation system) (All missions)	Partial deletions or deactivations in: Umbilical Carrier Plate - Aft Pneumatic Checkout Console Pneumatic Servicing Console Electrical GSE
Valve actuation system (Deletion of stage recirculation system) (All missions)	Partial deletions or deactivations in: Umbilical Carrier Plates Pneumatic Checkout Console Pneumatic Servicing Console Electrical GSE

System	Change Description
Ullage motors (deletion) (All missions)	Delete handling/storage equipment and monitoring of ignition system.
Heat shield (Modified dimensions for J-2S engine) (All missions)	Modify center engine platform set.
Reaction control system (LEO mission)	Add disconnects and equipment for RCS servicing and control in: Umbilical Carrier Plate Pneumatic Servicing and Control Consoles Electrical GSE
Thrust vector control system (For idle mode operation of engine gimbaling) (LEO mission)	Add disconnects and equipment for the Dc motor/pump in umbilical carrier plate, pneumatic checkout servicing control consoles, electrical GSE
Balanced vent system (For LH_2 tank vent during coast) (LEO mission)	Add electrical ground power and control of vent system.
Engine shutdown, coast, and restart controls (LEO mission)	Add electrical power, control, and monitoring for idle mode shutdown and idle mode restart

2.6.2.7 Integrated System Tests

Design verification testing requirements have been developed for installation of the J-2S engine into the S-II stage. Testing will be accomplished on the S-II Battleship stand at the Santa Susana Field Laboratory, at MSFC Facilities, at NR/SD Downey facilities, and on the first two production vehicles during acceptance testing at MTF.

The major elements of the program are as follows:

Battleship Test Stand	MSFC	NR/SD Downey	MTF
Boattail Environment Tests Subsystem Development Tests Short Duration Firings Standard Duration Firings	Thrust structure (aft skirt tests) Thermal (plume heating) tests	NR Downey facility Heat shield tests	Full mission duration tests on the first two production vehicles to demonstrate total system integration design verification.

The total test program time span is 13 months from go-ahead - including a seven-month period for Battleship modification and J-2S engine installation.

2.6.2.8 Systems Integration Documentation

Documentation requiring changes for employment of J-2S engines on the S-II stage are as follows:

Contract End Item Specifications, Parts I and II

Automatic Checkout
- Stage Networks Acceptance Program (SNAP)
- Pressurization System
- Electrical Power System
- Flight Control System
- Measurement System
- GSE Integrated Checkout
- Simulated Flight Test
- Interface Dictionary and Command Reference List
- Common Sub-procedures: Recirculation System and Engine System

Acceptance Test Requirement (Non-static Firing)
- Seal Beach/MTF Acceptance Test Specification
- Manual Functional and Leak Check Acceptance Specification
- Vendor Acceptance Test Specification Review (LEO Mission)

Static Firing Test and Acceptance Requirements
- MTF Static Firing Requirements Document
- MTF Static Firing Acceptance Test Specification

Interface Control Documents (ICD's)

KSC Test Specifications and Criteria

Schematic Drawings
- Electrical Checkout GSE
- Mechanical Checkout GSE
- Static Firing Electrical GSE
- Stage Mechanical Schematics

Flight Sequence Program

2.6.2.9 Launch Operations Requirements

The impact on launch operation of incorporation of J-2S engines in the S-II stage is being assessed in a separate NASA/KSC study. The information presented below is a preliminary description of this impact as determined by NR/SD, Seal Beach Operations.

Requirement	Impact
Facility/Equipment	Partial deletion of engine servicing systems Deletion of ullage motor handling/inspection equipment Deletion of recirculation system support systems Addition of equipment for solid propellant turbine starter handling/storage Addition of equipment for RCS checkout and handling (LEO mission)
Low Bay Operations	Simplification of leakage and functional tests Additional RCS tests (LEO mission)
High Bay Operations	Reduction of propulsion system testing Launch vehicle/spacecraft integrated tests simplified slightly Addition of RCS electrical checkout (LEO mission)
Launch Pad Operation (before CDDT/Countdown)	Addition of RCS tests (LEO mission)
CDDT/Countdown	Slight decrease in complexity Significant increase in probability of successful countdown because of redline reduction Add RCS tasks (LEO mission)
Launch Mission Rules	Of a total of 78 redline measurements, 34 will be deleted Add 10 redlines for RCS (LEO mission)

2.6.2.10 Reliability Assessment

In the 1966 preliminary analysis J-2S study, it was determined that approximately 30 percent of the S-II stage single point failure mode that could cause mission loss or abort would be eliminated by the system simplifications, resulting from J-2S engine incorporation. In the present study, this analysis was revised and updated, and the benefits of such simplification on the stage quantitative reliability values were then determined. The results are shown below for the LOR and LEO missions as compared to the baseline S-II-11 configuration. In the case where the boost period reliability value of 0.95 (meaning 95 of every 100 attempted missions will be successful) is retained for the LOR and LEO missions, a program reliability benefit is indirectly implied in the lower apportioned reliability values required of the remaining stage systems. In the case where the boost period reliability value is increased, a program reliability benefit is directly indicated.

Mission	Boost Period Reliability 0.95 All Cases			Boost Period Reliability Increased		
	Boost	Orbital	Total	Boost	Orbital	Total
Baseline S-II-11	.95	-	.95	.95	-	.95
J-2S/S-II LOR	.95	-	.95	.954	-	.954
J-2S/S-II LEO	.95	.94	.893	.954	.945	.902

2.6.3 S-IVB Stage Description

The S-IVB stage is built in two configurations: (1) the S-IVB/IB which is the second stage of the uprated Saturn I and, (2) the S-IVB/V which is the third stage of Saturn V. Application of the simplified J-2 engine (J-2S) on the S-IVB/V was investigated to determine its impact on the stage subsystems, performance, payload potential, and mission flexibility. The stage modification requirements identified were used to generate detailed implementation plans, tentative schedules and data for estimating budgetary requirements.

To fully understand the effects of the J-2S engine on the S-IVB stage, an analysis of its requirements on each of the subsystems was made by the affected technologies. These analyses were based on requirements of currently scheduled Saturn/Apollo missions. The Lunar Orbital Rendezvous (LOR), the equatorial synchronous orbit mission, the polar orbit mission, and the Low Earth Orbit Mission (LEO) were reviewed. Early in the study, the polar orbit mission was determined to be impractical and further analyses was discontinued. Since the LEO mission does not employ the S-IVB stage, it was excluded from the MDAC-WD analyses.

Lunar Orbital Rendezvous Mission

The prime mission investigated for implementation of the J-2S engine on the S-IVB/V stage was the LOR. This mission requires propulsion of the first two stages (S-IC and S-II) and a portion of the propellant from the third stage (S-IVB) to achieve a 100 nautical mile circular parking orbit. The S-IVB and spacecraft then coast for up to 3 revolutions in this orbit (4-1/2 hours) for vehicle checkout and precise orbit determination prior to S-IVB stage engine restart. As the lunar launch window is entered, the third stage engine reignites propelling the spacecraft into the lunar transfer trajectory. The S-IVB stage is then utilized to maintain proper attitude control for spacecraft transposition and docking.

Synchronous Orbit Mission

This mission will also utilize the first two stages (S-IC and S-II) plus a portion of the propellant from the third stage (S-IVB) to achieve a 100 nautical mile circular parking orbit. After up to a 7-1/2 hour coast period (5 revolutions) the S-IVB stage will be reignited for insertion into an elliptical transfer orbit whose apogee altitude is equal to the earth synchronous orbit altitude of 19,323 nautical miles. Transfer coast time to apogee is approximately 5.25 hours. Upon reaching apogee, the S-IVB is again restarted to circularize the orbit.

A parking orbit coast period of 7.5 hours or 5 revolutions was selected for these analyses since it represents the most stringent attitude control system, hydrogen boiloff, and power requirements that would be imposed on the S-IVB stage in providing complete longitudinal coverage of the earth. Additional requirements peculiar to this mission are that the trans-synchronous coast phase must occur in total sunlight, and the vehicle must be oriented broadside to the sun vector and rolled continuously about its longitudinal axis at a rate of 1 roll per hour. Both these requirements are based on S-IVB stage systems thermal criteria.

2.6.3 (Continued)

J-2S Application

Implementation of the J-2S engine on the S-IVB stage will improve stage reliability, mission flexibility, and performance. It can provide either a two or three burn capability at 431 seconds specific impulse. Increased reliability is primarily due to system simplification. The chilldown system (both LOX and LH_2), solid ullage motors, LOX low level sensors, and APS ullage engines and considerable instrumentation and wiring are not required for the J-2S/S-IVB configuration and have been deleted.

The J-2S/S-IVB does not require a recirculation chilldown system to condition its hardware for ignition during a LOR mission. Instead it utilizes an idle mode operation prior to mainstage ignition to perform the chilldown function thus eliminating the requirement for this system. This simpler operation greatly facilitates extension of the two-burn to a three-burn capability. The only additional modification required to implement a three-burn capability is the addition of a third solid propellant turbine starter (SPTS) and additional cold helium spheres to accommodate the second repressurization. If this capability were implemented using the current J-2 engine, additional batteries and stage plumbing would be required.

Incorporation of a synchronous mission capability results in some stage modifications; however, these modifications are not a result of incorporation of the J-2S but are caused by the mission requirements. These changes consist of adding baffles and/or deflectors in the LOX and LH_2 tank, adding power amplifiers to increase the range of the T/M transmitters, adding active and passive thermal control systems to accommodate the extended mission coast temperature environments and adding instrumentation.

All stage modifications have been identified in considerable detail. Drawings, part numbers, and procedures have been identified with respect to deletions, revisions and additions. These lists are predicated on the stage requirements resulting from incorporating the J-2S engine as described in the J-2S interface document. However, if future developments reveal J-2S operations or interface requirements would be materially different than those considered at present, some modifications to the changes described in this document may be necessary.

Test Requirements

To assure that the engine will operate with the stage in the manner predicted, a ground test program is required. This ground test program will be accomplished primarily at the Arnold Engineering Development Center (AEDC) and on the S-IVB Battleship vehicle at MSFC. The test requirements at AEDC have been defined and are directed primarily toward demonstrating engine operations with respect to its interface with the S-IVB stage. The test at MSFC will verify the modifications to stage equipment resulting from incorporation of this engine. A series of acceptance firing tests on the J-2S equipped S-IVB also are planned to verify engines/stage interfaces and operations.

2.6.3 (Continued)

Ground Support Equipment

Modifications to the ground support equipment are required because of the new stage operations. However, these modifications consist primarily of deleting procedures, capping off systems no longer needed, and changing name plate labels. The net result of incorporation of the J-2S engine into the S-IVB is a simpler stage with attendant program benefits. The basic cost of fabrication, checkout, launch, and post flight evaluation is materially reduced. This is accompanied by an increase in payload from the improved engine performance coupled with a stage weight reduction of 161 lbs for the LOR configuration. The reliability of the stage increases because of simplification and results in a 22 percent decrease in stage criticality.

2.6.4 Instrument Unit (IU) Stage Description

The current IU is a 260-inch diameter assembly 36 inches high that fits atop the S-IVB stage. Fully configured, the IU weighs 4,500 lbs, has a useful lifetime of approximately eight hours and a reliability over its 6.8 hour design life of 0.988. The IU is designed to be applicable to the wide range of missions for the Saturn family of vehicles. These missions can be placed in the five general classifications listed below:

Ascent and insertion into earth orbits having varying accuracy requirements.

Transfer orbits in the vicinity of the earth, e.g., rendezvous.

Injection into escape trajectories following extended earth orbiting.

Direct ascent and injection into escape trajectories.

Extended missions beyond injection.

Each of these missions places a different set of requirements on the system. Such factors as requirements for accuracy, in-flight turn-on and turn-off, in-flight system checkout, varying launch windows, operating time, and differences in equipment reliability, influence the choice of an equipment configuration for a particular mission.

The IU configuration is designed to meet the following requirements for any of the five categories of missions:

Navigation - maintain an up-to-date knowledge of the vehicle position, velocity and orientation.

Guidance - provide vehicle steering commands to obtain desired trajectories with minimum fuel consumption.

Control - establish and maintain desired vehicle attitude.

Sequencing - provide primary and contingency event queuing and execution and maintain the onboard mission clock.

Command - accept and respond to ground instructions.

Onboard checkout - assist in preflight test and verification of flight worthy status and provide capability for in-flight tests and status monitoring.

2.6.4 (Continued)

Spacecraft control - provide for accepting attitude error signals during flight from the spacecraft.

Telemetry - make numerous flight control and engineering measurements and provide for their transmission to the ground.

Tracking - provide RF signals for use in ground tracking.

Structure - provide the structural member connecting the thrust stages with the payload.

Emergency detection - detect abnormal or emergency situations and inform the crew.

Power - provide and distribute power to all equipment.

Environment - provide thermal conditioning to electronic components.

The IU system is considerably more than a guidance, navigation and control system. The control, assembly and testing of a system of this complexity is a significant portion of the total cost of the IU.

2.6.4.1 Summary of IU Design Changes

The large majority of changes, which in effect convert the J-2/LOR IU to the J-2S/LOR IU, fall within the category of mission-to-mission changes which, in the current program, require a steady stage "maintenance-of-design" work force in the engineering design and release cycle. The basic universality of the IU design, the electrical support equipment, and the digital computer analytical programs, makes accommodation to new missions very straightforward. An exception is the Synchronous mission where lifetime extension requires modifications that were not designed in, but again are easily implemented in exercising the growth potential of the IU.

a. J-2S/LOR IU Changes

Minor changes to the Flight Control System include provision for a reset capability into one of the two switch points in the S-IVB pitch and yaw attitude error and attitude rate filters. The changes are confined to the Flight Control Computer and the Control Distributor.

2.6.4.1 (Continued

Flight programs and electrical support programs require minor parametric value changes and sequence reprogramming because of the idle mode and minor variations in event sequences.

b. J-2S/Synchronous IU Changes

It is assumed that all changes made to the J-2S/LOR IU cover all engine-imposed changes. Additional changes which are mission imposed, follow.

Additional Switch Selector commands will be required to initiate the desired sequencing and inhibiting of the power amplifiers and antennas. Minor rework will be required in the distributors for the sequencing and inhibiting logic.

Five additional CCS directional antennas are employed. To maintain satisfactory circuit margins, two modified Power Amplifiers are used. A power divider and two coax switches are added to permit antenna selection.

A four uprated IU battery configuration was chosen for the J-2S/Synchronous mission which extends the IU power system lifetime from 6.8 to 15 hours. The batteries are a new design and result in a non-recurring cost impact.

Extensive rework of the flight program results from the mission. Yaw biasing of the three gimbal ST-124 platform and retargeting equation implementation are unique to the application.

The Environment Control System is modified by the addition of an additional two cubic feet GN_2 storage sphere and mounting appurtenances and a redesigned TCS orifice regulator assembly.

Extensive logic changes are required to the Electrical Support Equipment automated checkout program with no change to the basic program. The ESE hardware changes include network modifications for accommodation of the new batteries and antennas.

c. J-2S/LEO IU Changes

Moderate redesign of the Flight Control Computer (FCC) is required to eliminate the S-IVB circuitry and to add the S-II idle mode pitch and yaw attitude rate filters. Blank boards will replace populated boards to minimize the redesign impact. Major modification is required of the FCC wiring harness.

2.6.4.1 (Continued)

IU OCM/DDAS Telemetry limits to the S-II stage and transmission of data via the S-II PCM telemetry system will be required. The design impact is minor.

The LEO mission will impose careful optimization of the guidance filters. Implementation is unique for restart stage and estimation of the terminal targeting.

Additional gain switching will be required in the S-II pitch and yaw attitude error shaping networks for the S-II idle mode and one gain switch will be required in the S-II roll shaping network for S-II idle mode.

The IU Environmental Control System liquid coolant lines to the S-IVB stage will be capped off.

d. J-2S/Polar IU Changes

Because the S-II RCS is not used, the elimination of all S-IVB circuitry, including APS circuitry from the FCC, results in extensive rework. The design impact is minimized by use of dummy cards rather than total repackaging of the FCC.

Both the LEO and Polar missions impose excessive tension loads which required minor structural modification.

2.7 KSC LAUNCH OPERATIONS IMPACT

The study determined the total impact at KSC when J-2S engines are used in Saturn V vehicles. Changes to operations and equipment were identified, and implementation schedules developed.

2.7.1 Propulsion Related Processing Operations

The impact on processing operations primarily concerns modifications to existing operations rather than the addition or deletion of a large number of operations. There were 133 S-II and S-IVB propulsion related operations identified for the SA-503 baseline vehicle (those affected by an engine change). For a vehicle configured for a LOR or Synchronous mission, 70 percent of these operations would be reduced by an average of 7 percent and only 2 operations would be deleted. For a vehicle configured for a LEO mission, 45 percent of the operations would be reduced by an average of 7 percent, and 8 operations would be added due to S-II Reaction Control System (RCS) and Solid Propellant Turbine Starter (SPTS). Forty percent of the operations can be eliminated because an S-IVB stage is not used for this mission.

2.7.2 Facility and Ground Support Equipment

The primary effect of J-2S engines on Facilities and Ground Support Equipment located at KSC was in the Pneumatic Systems on the Mobile Launcher (ML) and the control and Monitor panels in the Launch Control Center (LCC) for both LOR and LEO Missions. The Pneumatic System changes consisted of the deletion of one console, modification of three consoles, and modification of ML plumbing and umbilical. The S-IVB Pneumatic System changes consisted of the deletion of one console, modification of two consoles, and modification of ML plumbing and umbilical. S-II RCS servicing equipment would have to be added to Platform 1 and the 133-foot level of the Mobile Service Structure (MSS) for the LEO Mission only. This new equipment consists of two control assemblies, two valve isolation boxes, and an nitrogen purge system.

2.7.3 Interlock System

Changing to J-2S engines impose modified requirements on the Interlock System. These result in the deletion of 54 interlocks for a LOR or Synchronous mission. Approximately 34 interlocks on the S-II would be deleted and four top level interlocks must be added for RCS, Propellant Settling System and the Auxiliary Hydraulic System for a LEO mission. All S-IVB related interlocks would be deactivated for a LEO mission.

2.7.4 Redline Parameters

The LOR mission effect on the Redline Parameters consisted of deleting 20 of the original 75 redlines and adding five new redlines. The LEO mission effect consisted of deleting 15 of the original 42 S-II redlines and adding 11 new redlines.

2.7.5 Operations Improvement

It is anticipated that KSC will realize improvements in GSE reliability, processing flow times and launch complex flexibility as a result of using J-2S engines in the S-II and S-IVB stages. The overall reliability of the pneumatic system would improve due to the deletion of consoles and the reduction of complexity of other consoles. Servicing time would be reduced by the elimination of potential leakage points. The processing flow time was estimated to be reduced by approximately 10 percent due to the procedural changes which result from processing the J-2S equipped stages. Increased launch complex flexibility at KSC results because the design changes outlined allow for launching different vehicles of various configurations and mission objectives such as a three stage LOR mission and a two-stage LEO mission.

2.8 SYSTEMS ENGINEERING AND INTEGRATION (SE&I)

SE&I is presently performed by The Boeing Company for the Saturn V vehicle and is divided into four activity categories.

Activity Category	Tasks
System Integration	Program Control, Configuration Management, Test Program Integration and Launch Readiness Assessment, and Logistics
Systems Engineering	Interface Engineering, System Definition, Prelaunch System Analysis, Design Certification Review, Reliability Analysis Mission Rules, and System Safety
Technology	Flight Evaluation, Propulsion System Analysis, Structural System Analysis, and Instrumentation System Analysis
Launch Vehicle and Mechanical Ground Support Equipment (LVGSE)	LVGSE Logistics and System Development Facility (SDF)

The tasks, such as Program Control, Configuration Management, Test Program Integration, System Definition, Design Certification Reviews, etc., investigate the overall launch vehicle and its systems including the engines, but are independent of type of engine used, in that the same efforts are required for J-2 or J-2S equipped vehicle. These efforts are normal SE&I efforts which are performed under normal manpower and schedule requirements.

The tasks, Interface Engineering, Flight Evaluation, and Propulsion System Analysis require additional SE&I development and recurring efforts. The LOR Mission requires efforts to develop the necessary guidance system modification and computer trajectory simulation modifications to both digital and hybrid simulators. The 3-stage synchronous mission with a 3-burn S-IVB and 2-stage LEO mission imposes changes in analysis which are unique and partly independent of the use of the J-2 or J-2S engines. Additional analysis and development is required to assess J-2S synchronous orbit mission capabilities and to rework the assumed developed synchronous orbit guidance to incorporate J-2S capabilities. Additional work is also required in guidance and simulation modifications to implement the 2-stage configuration capabilities.

2.9 RESOURCES

2.9.1 Stage Resources

2.9.1.1 S-IC Stage

The S-IC stage is not impacted by the J-2S engine and/or mission requirements except for that of the two-stage polar orbit which causes an overload condition in the S-IC fuel tank. This overload condition may be solved by (1) load alleviation or (2) fuel tank skin/stringer beef-up. If the second option is assumed a minimal development cost of approximately $20,000 is required for new manufacturing mill tapes.

The S-IC stage resources plans (design, test, manufacturing, facility, schedule and cost) are the same as those of current stage production program. The S-IC schedule cycle is 42 months from ATP to on dock KSC, which coincide with the lead time boundaries of the integrated J-2S program schedule. The S-II stage also requires 42 months prior to on dock KSC.

2.9.1.2 S-II Stage

Design Plan

The engineering design activities associated with an inline incorporation of J-2S engines into an advanced S-II stage for either an LOR, Synchronous Orbit, two-stage Polar Orbit or LEO mission is broken into two phases; a program definition phase (PDP), and a development operational phase. The PDP will involve six months of engineering effort, whereas the development operational phase will continue through vehicle delivery and launch. It is to be noted that all of the identified changes for this stage upgrading are well within the technical state-of-the-art and available facility capabilities.

Due to the nature of the task of installing J-2S engines in place of J-2 engines, the concept or approach for specification and drawing preparation shall be that of making revisions to existing S-II-15 released documentation (documents may be released with new identifying numbers to aid configuration control).

Whereas the design plan as presented satisfies the four study missions from a schedule viewpoint, the LEO mission will require a higher technical manpower level during the design and analysis periods to fulfill the additional requirements. These additional requirements include the addition of a reaction control system, auxiliary power source for engine actuation (gimbal) during idle mode operation, engine idle mode operation capability, and an LH_2 tank vent system to provide a low vehicle acceleration force during the coast portion of the flight.

2.9.1.2 (Continued)

Test Plan

The planned tests represent the minimum development and/or verification tests considered necessary to provide assurance that the advanced S-II stage, as modified to incorporate J-2S engines and added mission capability as directed by NASA, has retained a man-rated status and will therefore require no development flight tests.

The Test Plan includes:

a. Integrated system tests

1. Structural Testing. Test verification that design changes to the S-II structure will accommodate the increased thrust and base heat flux of J-2S engines will be conducted using selected portions of full scale structure and special heat shield panels.

2. Battleship Tests. Development testing will be conducted in the Battleship test stand (COCA 1) at the Santa Susana Field Laboratory to verify stage/engine interface compatibility and J-2S cluster operation.

3. All Systems Verification Tests. All Systems tests will be conducted at MTF utilizing the first production stages.

b. Design Verification Tests

Design verification tests (DVT) will be conducted on selected components to verify attainment of basic operating parameter requirements. The primary objectives of DVT are as follows:

1. Verification of proper operation with parameter limits
2. Verification that design approach was satisfactory
3. Resolution of design difficulties
4. Verification of parameter stability
5. Verification of hardware selection by comparison
6. Preliminary verification of parameter stability under stress conditions

c. Qualification Test Program

The primary purpose of a qualification test is to assure equipment performance during exposure to the environmental stresses predicted in service. To accomplish this purpose, a selected number of specimens representing production hardware are subjected to environments selected and arranged to best represent service conditions.

2.9.1.2 (Continued)

These tests are designed to assure the reliability of the equipment design with respect to flight operation.

The qualification test program to be conducted on a J-2S/S-II stage will be a continuation of the same program employed for the current J-2/S-II stage as delineated in the Saturn S-II Reliability Plan. It is considered that this approach best meets the requirement to achieve or retain a man-rated S-II stage at minimum cost.

Incorporation of the J-2S into the S-II stage requires certain modifications peculiar to specified missions. These modifications are evaluated relative to the following factors to determine necessity for qualification or requalification.

1. New or modified component designs in criticality categories I, II, and III
2. New or changed environments
3. Increased or significantly changed duty cycles
4. New or revised operating modes
5. New or revised significant failure modes
6. Dynamics Program 63-2 requirements
7. Prior test history
8. Common applications
9. Contractual requirements

J-2S Manufacturing Schedule Plan

All Manufacturing sequences of the Manufacturing Plan remain basically the same for the current J-2 engine configurated stages and LOR mission J-2S engine configurated stages. However, changes will be necessary in internal schedules on a subsystems and subassembly level.

Facilities Plan

The Contractor anticipates no additional requirements for expansion of Government-funded facilities to accomplish effort proposed in support of the J-2S engine incorporation into the S-II stage.

Schedule Plan

The schedule for vehicles 518 through 537 with J-2S engines present no problems. Thirty-eight months leadtime has been allowed from raw material release to delivery of the first stage at KSC, which is consistent with the conclusions previously reached in the low production rate study. Although the schedule is reflected on an ATP schedule, some variations in the schedule would be encountered depending upon the "capability position" at the time of go-ahead. In general, the overall 48-month span

2.9.1.2 (Continued)

from go-ahead to delivery of S-II-18 provides adequate time for start-up from minimum capability levels.

Cost Plan

The S-II stage cost to incorporate the J-2S engine is divided into non-recurring cost (development) and recurring (operational). The costs are for a block of 20 vehicles, 10 year production program, at two per year production rate. The baseline stage is J-2/S-II-LOR. The total delta non-recurring cost for a LOR or synchronous mission is $19.48 millions. The delta recurring cost is a saving of $44.68 million ($2.23 million per stage). The development cost ($19.48 million) is amortized after the 8th stage.

The total delta non-recurring cost of an LEO mission w/o the RCS (J-2S/S-II - LOR as a baseline) is $25.05 million. The recurring delta cost is a saving of $31.69 million of $1.58 million per stage. The development cost ($25.05 million) is amortized after the 12th stage.

The cost of the RCS system is $5.66 million non-recurring and $28.12 million recurring.

2.9.1.3 S-IVB Stage

The net result of incorporation of the J-2S engine into the S-IVB is a simpler stage with attendant program benefits. The basic cost of fabrication, checkout, launch and post flight evaluation is materially reduced. This is accompanied by an increase in payload from the improved engine performance coupled with a stage weight reduction of 161 pounds for the LOR configuration. The reliability of the stage increases because of simplification and results in a 22 percent decrease in stage criticality.

Design Plan

The two existing S-IVB Battleship Vehicles are to be updated for use in the J-2S Engine Test Program. MDAC engineering shall provide technical support for the modification and testing phases. Engineering shall also provide personnel to witness J-2S testing at the North American Rockwell Corporation, Santa Susanna Facility.

Sustaining engineering shall be performed for the duration of the Program, which is based on a fabrication and launch rate of two vehicles a year.

Basic S-IVB/V drawings shall be configurated for all additions, modifications, and deletions that are required to incorporate the J-2S engine for a LOR mission.

2.9.1.3 (Continued)

The J-2S/S-IVB/V stage drawing shall be configurated to incorporate modifications (scars) that will permit late incorporation of a synchronous orbit mission kit. Scars are defined as changes to the stage that do not affect the stage performance, but permit late incorporation of mission peculiar kits.

Engineering shall support the following design reviews:

a. Preliminary Design Review

At this review, Engineering shall present to the customer, concept layouts and written material that depict the proposed design approach for new and revised vehicle and GSE systems.

b. Critical Design Review

At this review, Engineering shall present detailed drawings and supporting analyses of all design changes resulting from the J-2S implementation.

Test Plan

The incorporation of the newly developed J-2S engine into the S-IVB stage and the mainline Saturn/Apollo Program will require careful and deliberate integration in order to maintain the man-rated, high confidence level of the present Saturn/ Apollo Program.

Based on the J-2S/S-IVB Development Test Plan in DAC-56749 and the J-2S/S-IVB Rocketdyne Interface Criteria Document, R-7211, development tests required for concept verification, operational limit establishment, and acceptance testing requirements will be defined.

These tests must start at the engine development (Battleship) test level, complemented by S-IVB stage component tests and special instrumentation on mainline Apollo flight, progress through the stage acceptance firings at STC, comprehensive launch site system checkout demonstrations and finally verified by inflight system analysis through adequate placement of instrumentation.

a. Battleship Test

MDAC-WD recommends a joint NASA/Contractor cooperative test approach in the Battleship tests presently planned for Tullahoma (altitude facility) and MSFC (sea level facility). This approach involves simultaneous development of stage modifications while in the process of developing the engine itself. Concurrently, through MDAC-WD test support, the Contractor plans to develop operational limits

2.9.1.3 (Continued)

and test procedures which can be employed during later component development and stage acceptance firing tests.

b. Component Tests

As a result of J-2S engine implementation into the basic S-IVB/V Stages, MDAC-WD presently envisions that the following qualification tests must be performed.

1. LH_2 Tank Pressurization Module
2. Hydraulic Pump and Thermal Isolator
3. Auxiliary Hydraulic Pump
4. Component Heater Blanket for Synchronous Mission

c. Special Instrumentation for Mainline Apollo Flights

Where Mainline Saturn Apollo Program missions permit, various aspects of the J-2S engine performance might be verified by appropriate instrumentation. This shall be predominantly true in the stage-to-engine interface conditions of the space environment.

d. Acceptance Firing Tests

MDAC-WD recommends a minimum of three Saturn S-IVB vehicles to undergo a acceptance firing with the new J-2S engine installed. This shall include three full duration and three partial duration J-2S engine ignitions.

Manufacturing Plan

Manufacturing operations will occur in the general areas of structural, propulsion, mechanical systems including piping and ducting, electrical components and wiring, the APS, and GSE. As presently conceived, all of these modifications will be accomplished with conventional manufacturing methods, processes, and tools. Modification of the stages to accommodate the J-2S engine will be accomplished by the Manufacturing organization of MDAC. The tasks will be supported by Quality Assurance, Procurement, Manufacturing, Engineering, Planning and Tooling, Facilities Engineering, and Materials and Methods Research and Engineering (MM-RE) organizations.

Facilities Plan

Based on the current visibility (design, manufacturing, and testing) of the Propulsion Improvement Study, and on the ground rules and assumptions, it appears that implementation of the J-2S engine on the S-IVB stages can be accomplished with the existing facilities currently used on the Saturn Program.

2.9.1.3 (Continued)

Schedule

The lead time for design, development, procurement, tooling, and manufacturing is similar to the schedule now being used on the S-IVB program. The largest percentage of detail manufacturing will be performed at the Santa Monica Facility. Primary welding, major, and final assembly, and checkout will be performed at the MDAC Huntington Beach Facility.

A fourteen month lead time for procurement will be required prior to the initial major assembly effort. Required tooling will be phased and scheduled to meet all fabrication, assembly, and checkout requirements as needed.

A three-burn synchronous mission kit will be developed and available for installation on any J-2S equipped LOR stage which NASA may designate. This installation would be performed preferably in-line at the Huntington Beach Facility but could also be handled at STC.

The J-2S stage will be delivered to KSC three and one-half years after ATP.

Costs

a. Development Costs

The total development (non-recurring) cost for implementing the J-2S on the S-IVB stage for the LOR mission is $3,474,721.

The total development cost of extending a J-2S/S-IVB to synchronous mission capability is $1,886,413.

b. Recurring Costs

The delta recurring costs between the LOR J-2/S-IVB and J-2S/S-IVB result in a net saving of $458,319 per stage on a two per year production rate (10 year program). Therefore, J-2S/S-IVB LOR development costs are amortized by the eighth stage.

The delta recurring costs between a J-2S/S-IVB LOR stage and J-2S/S-IVB synchronous mission stage are $354,548 per stage at a production rate of 2 per year. These recurring costs are based on the production of 20 kits to be installed on J-2S/S-IVB LOR stages in-line and in-position. For a single synchronous mission kit the recurring cost per stage increases to $657,788 due to the absence of a learning curve. Based on production of 5 kits, the cost is $463,443 per kit.

2.9.1.4 IU Stage

Design Plan

The Flight Control Computer (FCC) of the control system and the Control Distributor of the electrical system are the only end-item components which require hardware modification as a result of the J-2S engine. A minimum-modification approach will be taken to design the required changes to the FCC and Control Distributor. This approach will consider the peculiar requirements of the LOR, Synchronous, LEO and Polar missions and provide one configuration level of these components for use with any J-2S/Saturn V vehicle configuration independent of the mission application.

With exception of the above modification, the present Saturn V IU design meets the requirements of the LOR, LEO and Polar mission applications of the three-stage or the two-stage Saturn V vehicle with the J-2S engine improvement.

Test Plan

There are no test requirements peculiar to J-2S engine improvement modifications on the IU. The minor hardware modifications to the FCC and Control Distributor do not warrant requalification testing. Subsequent to incorporation of the J-2S improvement changes into these components, they shall be considered qualified by similarity.

There are no known test requirements peculiar to J-2S/Saturn V IU's for the LOR, LEO or Polar mission applications. Only those test requirements peculiar to the incorporation of the IU life extension modifications to satisfy requirements of the Synchronous mission are discussed here. Further, this discussion is based on the content of IBM-ECP-1558.

Manufacturing Plan

The IU manufacturing effort is described in terms of three distinct phases: Fabrication, Assembly and Preparation for Shipment controlled by manufacturing routings which outline, step-by-step, the procedure to accomplish all the discrete operations required, including the essential inspections.

The manufacturing routings are machine prepared and afford the flexibility of being responsive to changes in manufacturing instructions as brought about by engineering releases of new or changed requirements.

There are no new tooling or fixture requirements for the manufacture of IU assemblies for the three-stage or two-stage Saturn V vehicle with the J-2S engine improvement. Configuration variations between IU's for the LOR, Synchronous, LEO or Polar vehicle configurations can be handled by the issuance of separate sets of manufacturing

2.9.1.4 (Continued)

instructions (routings) which are unique to a particular IU. In effect, there would be no essential differences from the manner of manufacturing Saturn V and Uprated Saturn I on the current program. Further, the nominal times given for each of the manufacturing phases, including systems checkout, would be the same.

Facilities Plan

Existing facilities are designed to satisfy the broad mission requirements of the current program. These facilities are sufficient to support the production rate of two IU's per year, independent of configuration for the LOR, Synchronous, LEO or Polar vehicle/mission application. No facilities modifications are required due to J-2S engine improvement or mission application.

Schedule Plan

The master phasing schedule shows that the minor modification required for J-2S engine control requirements can be accomplished within the normal IU mission cycle which is representative of such cycles on the current program.

An IU mission cycle is defined as that period of time between the first issue of an Instrumentation Program and Components (IP&C) List for a given IU and the launch of a vehicle with that IU. This cycle is 24 months, established by scheduling availability of the IP&C List 18 months prior to an IU delivery and an IU delivery which is scheduled against an arbitrary launch date to occur six months later.

The schedule effects of Synchronous mission dependent changes were selected for illustration because Synchronous mission requirements represent the most severe lead times for schedule considerations and the greatest impact on program costs. This study did not reveal any requirements peculiar to the LOR, LEO or Polar mission/vehicle application which would require time outside of the time span for an IU mission cycle for a Synchronous mission.

Cost

a. Non-Recurring Cost

Only the Synchronous mission has extensive development costs involving the flight control computer, structure modification, network and cabling changes, flight program environmental control system, batteries and battery installation, checkout programming changes and hardware requalification.

The delta non-recurring cost for J-2S/IU-LOR (J-2/IU-LOR as baseline) is $0.03 million. The delta non-recurring cost, from the J-2S/IU-LOR, for J-2S/IU-synchronous is $1.12 million and for J-2S/IU-LEO is $.05 million.

2.9.1.4 (Continued)

b. Recurring Cost

Based on a two per year production rate for ten years, there are no delta recurring costs for the J-2S/LOR and LEO IU's. There are numerous changes in the flight program, network cabling and in the entire release cycle of design and manufacturing. The normal vehicle-to-vehicle "maintenance of capability" accommodates these changes in a routine manner.

The recurring cost of J-2S/IU-synchronous mission is $0.32 million. This cost is in the assembly and testing line items, which results from mounting the additional Gas Bearing Supply (GBS) cold plate, additional antennas and additional plumbing.

2.9.2 KSC Launch Operations

Changes in terms of additions and deletions of LC-39 support functions and equipment for the J-2S vehicles were identified. Implementation schedules were developed to assess the total impact to KSC.

Schedule

The schedules for accomplishing the modification to KSC facilities assume only the modification and activation of one VAB High Bay, one Firing Room and associated equipment, one Mobile Launcher, one MSS, and one Pad. The anticipated date for the establishment of interfaces was used for GSE design start. Modification completion date was established as the date when the first modified stage will be on dock at KSC. These ground rules result in a total of 24 months to accomplish the changes. Of this total time span, 17 months have been allocated for design, 6 months for procurement, and 3 months for modification and activation.

The total operations would be reduced from the SA-503 baseline by 3,552 man-hours (approximately 3.70 men crew size) for the LOR mission and 2,238 man-hours (approximately 2.33 men crew size) for the LEO mission. The changes for the Synchronous mission would be essentially the same as for the LOR mission.

2.9.3 Systems Engineering and Integration (SE&I)

The SE&I tasks, such as Program Control, Configuration Management, Test Program Integration, System Definition, Design Certification Reviews, etc., are independent of type of engine used. These tasks are normal SE&I efforts which are performed under normal manpower and schedule requirement, resulting in zero delta development or recurring cost to the J-2S/Saturn V program.

The Interface Engineering, Flight Evaluation and Propulsion System Analysis tasks are impacted by the incorporation of J-2S engines in the Saturn V vehicle or from mission profile variations.

Schedule

Compared to the baseline schedule (recurring twelve months cycle for the J-2/Saturn V - LOR), twelve months of development for the J-2S/Saturn V - Synchronous Mission and six months for either the J-2S/Saturn V - LOR or the J-2S/Two-Stage - LEO mission is required. This applies only to the first flight. Subsequent flights assume a normal and recurring twelve month cycle.

Costs Impact

The SE&I non-recurring (development) cost is minimal. Compared to the baseline vehicle (J-2/Saturn V - LOR), the non-recurring delta cost for J-2S/Saturn V - LOR is $0.55 million. An additional cost to the J-2S/Saturn V LOR of $0.14 million and $0.11 million is required for the synchronous mission and for the J-2S/two-stage LEO mission, respectively. These development costs are mainly for developing and assessing mission capabilities except for the Interface Engineering task which is stage development cost.

The total SE&I recurring cost for the three-stage/Saturn V-Synchronous mission is 110 percent that of LOR mission which results in a delta cost of $1.06 million per year. This cost is due to greater mission complexity and is independent of engine type. The total cost for the two-stage/Saturn V - LEO mission is 70 percent that of a three-stage/Saturn V - LOR. This reduced cost is due to the fact that the S-IVB stage is not included, resulting in a cost saving of $3.18 million per year. This cost is independent of the type of engine.

2.9.4 Vehicle Resources

2.9.4.1 Schedule

The J-2S incorporation imposes no unique schedule problem. The master schedule is shown in Figure 2.9.4-1. It reflects the total phasing of the program through delivery of SA-518 on dock at KSC. The study shows that the first J-2S equipped flight vehicle will be on dock at KSC, 3 1/2 years after Authority to Proceed (ATP); subsequent vehicles will follow at a rate of one every six months.

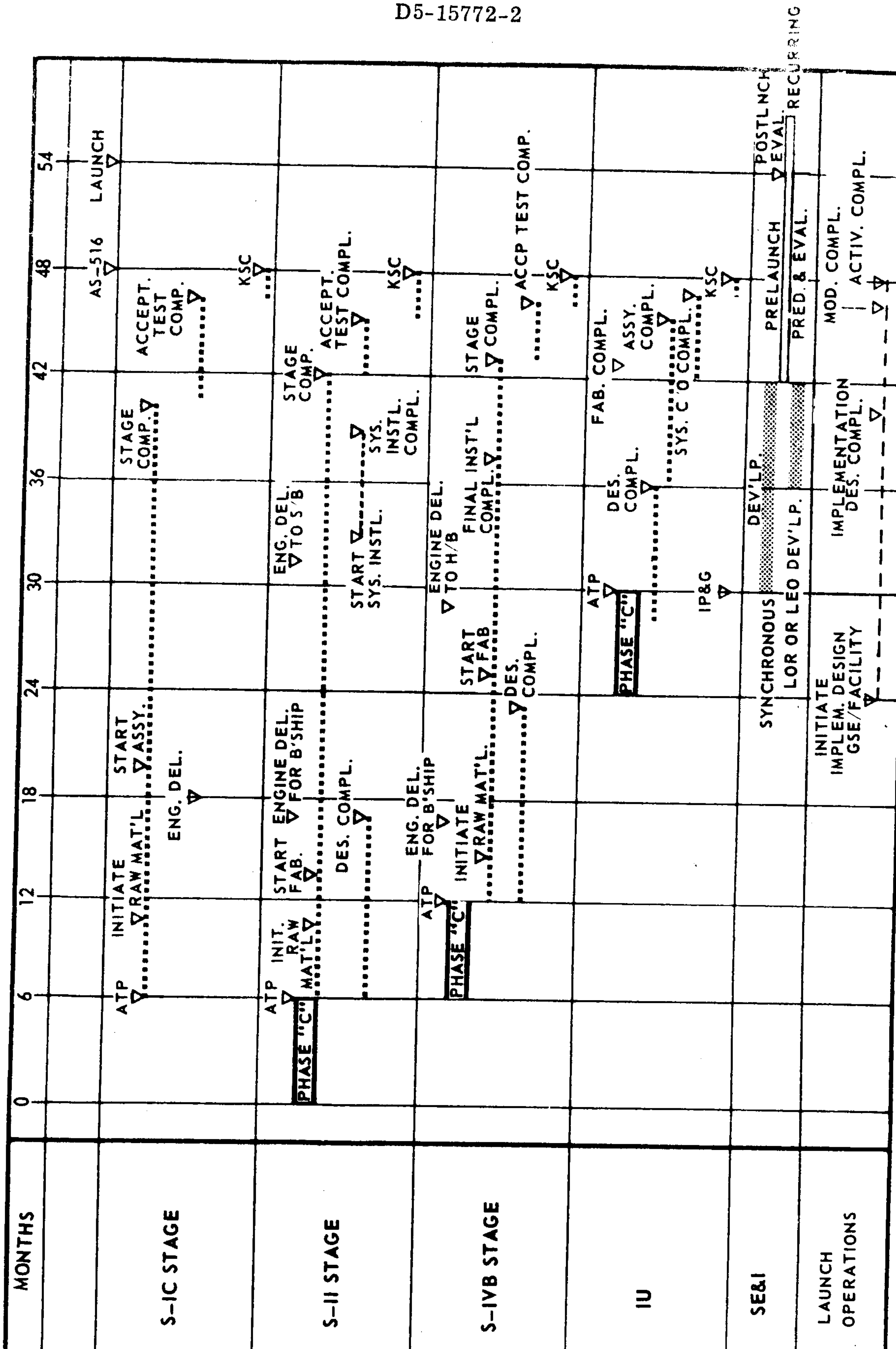


FIGURE 2.9.4-1 INTEGRATED VEHICLE MASTER DEVELOPMENT AND DELIVERY SCHEDULE

2.9.4.1 (Continued)

The 3 1/2 year production period for a J-2S, SA-518 vehicle is identical to the flow time required for a J-2 equipped SA-518 vehicle. Pacing items in the schedule are S-IC and S-II stage material requirements.

2.9.4.2 Cost

Development requirements for the J-2S equipped Saturn V vehicle (assuming LOR mission) result in a non-recurring cost of $23.54 million. These funds will be expended over a four year period. Peak year development fund requirements are less than $8.00 million.

Recurring costs for a ten year period of production at a rate of two vehicles per year, results in an average unit cost saving of $2.7 million compared to a J-2 equipped Saturn V for the LOR mission. Development cost is amortized by the cost saving per unit produced after only nine vehicles.

Tables 2.9.4-I through 2.9.4-III summarized the integrated vehicle costs of the J-2S equipped vehicles for the LOR, Synchronous, and LEO mission, respectively. The costs shown are combined non-recurring and recurring delta costs for a block of 20 vehicles produced at two per year.

Figure 2.9.4-2 shows the program comparative cost of J-2 and J-2S equipped three-stage vehicles. Baseline #1, the J-2 equipped Saturn V-LOR, is drawn horizontally through the zero cost point. Baseline #2, the J-2S equipped Saturn V-LOR, is drawn through the $30.30 million point which is a comparison of the two base line vehicles. The $30.30 million cost saving is the sum of $23.54 million development expenditure plus a stage saving of $53.84 million for 20 units or $2.7 million per unit. This saving has resulted exclusively from the incorporation of the J-2S in the S-II and S-IVB stages. To the right of the figure, the delta costs attributed solely to the mission pecularity are projected. These costs are independent of engine type. The LOR mission peculiar delta cost is zero (baseline). The cost to incorporate the mission peculiar changes for a synchronous mission (with either J-2 or J-2S) is $27.12 million; however, in case of J-2S engine. this mission cost when combined with stage saving of $30.30 million (see Table 2.9.4-I) result in a savings of $3.18 million.

Similarly, Figure 2.9.4-3 shows the program comparative cost of J-2 and J-2S equipped two-stage vehicles. The comparison between the two baselines is a cost saving of $24.61 million which is the sum of $20.06 million development expenditure plus a saving of $44.67 million for 20 units or $2.24 million per unit. The cost to incorporate the mission peculiar changes for a LEO mission (with either J-2 or J-2S) is $18.61 million; however, in case of J-2S engine, this mission cost when combined with stage saving of $24.61 million results in a saving of $6.00 million.

TABLE 2.9.4-I DELTA COST SUMMARY - J-2S/SATURN V - LOR MISSION (DOLLARS IN MILLIONS)
(20 VEHICLES, 10 YEAR PROGRAM, 2 PER YEAR, NO. 518 THRU 537)

	DEVELOPMENT	OPERATIONAL	TOTAL
Launch Vehicle			
S-IC Stage	0	0	0
S-II Stage	18.552	(41.600)	(23.048)
S-IVB Stage	3.145	(9.088)	(5.943)
Instrument Unit	0.034	0	0.034
Launch Vehicle Total	21.731	(50.688)	(28.957)
Ground Support Equipment			
S-IC Stage	0	0	0
S-II Stage	0.926	(3.075)	(2.149)
S-IVB Stage	0.330	(0.078)	0.252
Instrument Unit	0	0	0
Launch Vehicle	0	0	0
GSE Total	1.256	(3.153)	(1.897)
Facilities			
S-IC Stage	0	0	0
S-II Stage	0	0	0
S-IVB Stage	0	0	0
Instrument Unit	0	0	0
Launch Vehicle - KSC	0	0	0
Launch Vehicle - Other	0	0	0
Facilities Total	0	0	0
SE&I	0.554	0	0.554
Other Support	0	0	0
Total (20 Operational Vehicles)	23.541	(53.841)	(30.300)
Average Unit Cost	1.177	(2.692)	(1.500)

( ) = Savings

TABLE 2.9.4-II DELTA COST SUMMARY - J-2S/SATURN V - SYNCHRONOUS MISSION (DOLLARS IN MILLIONS) (20 VEHICLES, 10 YEAR PROGRAM, 2 PER YEAR, NO. 518 THRU 537)

	DEVELOPMENT*	OPERATIONAL*	TOTAL*
Launch Vehicle			
S-IC Stage	0	0	0
S-II Stage	0	0	0
S-IVB Stage	1.853	7.091	8.944
Instrument Unit	1.095	6.310	7.405
Launch Vehicle Total	2.948	13.401	16,349
Ground Support Equipment			
S-IC Stage	0	0	0
S-II Stage	0	0	0
S-IVB Stage	0.033	0	0.033
Instrument Unit	0	0	0
Launch Vehicle	0	0	0
GSE Total	0.033	0	0.033
Facilities			
S-IC Stage	0	0	0
S-II Stage	0	0	0
S-IVB Stage	0	0	0
Instrument Unit	0	0	0
Launch Vehicle - KSC	0	0	0
Launch Vehicle - Other	0	0	0
Facilities Total	0	0	0
SE&I	0.142	10.600	10.742
Other Support	0	0	0
Total (20 Operational Vehicles)	3.123	24.001	27.124
Average Unit Cost	0.156	1.200	1.356

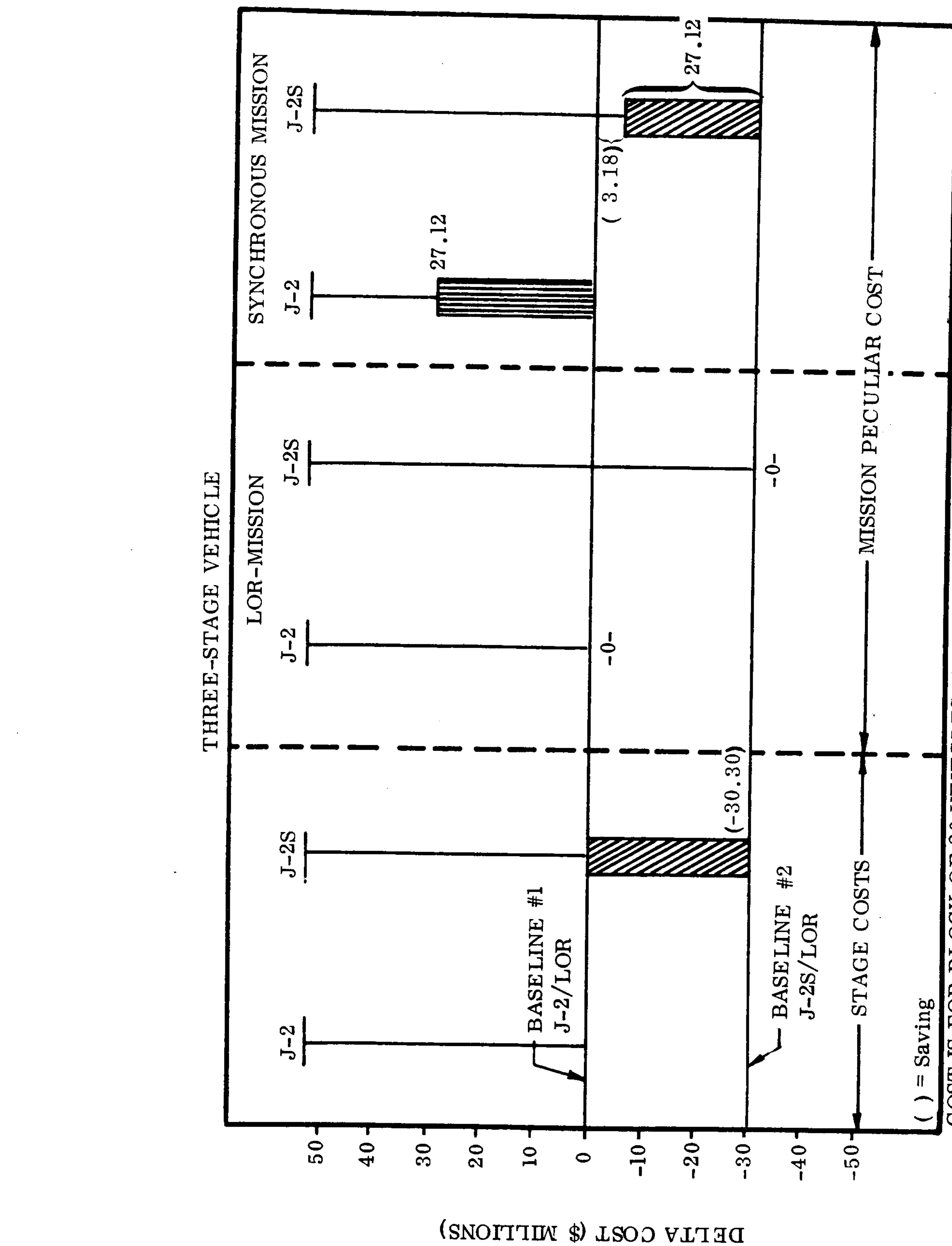
* MISSION COST

TABLE 2.9.4-III DELTA COST SUMMARY - J-2S SATURN V LEO MISSION W/O RCS (DOLLARS IN MILLIONS) (20 VEHICLES, 10 YEAR PROGRAM, 2 PER YEAR, NO. 518 THRU 537)

	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE*	MISSION	STAGE*	MISSION	
Launch Vehicle					
S-IC Stage	0	0	0	0	
S-II Stage	18.552	4.901	(41.600)	12.992	(5.219)
S-IVB Stage	0	0	0	0	0
Instrument Unit	.034	.013	0	0	0.047
Launch Vehicle Total	18.586	4.914	(41.600)	12.992	(5.172)
Ground Support Equipment					
S-IC Stage	0	0	0	0	0
S-II Stage	.926	.596	(3.011)	.064	(1.489)
S-IVB Stage	0	0	0	0	0
Instrument Unit	C	0	0	0	0
Launch Vehicle	0	0	0	0	0
GSE Total	.926	.596	(3.011)	.064	(1.489)
Facilities					
S-IC Stage	0	0	0	0	0
S-II Stage	0	0	0	0	0
S-IVB Stage	0	0	0	0	0
Instrument Unit	0	0	0	0	0
Launch Vehicle - KSC	0	0	0	0	0
Launch Vehicle - Other	0	0	0	0	0
Facilities Total	0	0	0	0	0
SE&I	.554	.110	0.110	0	0.664
Other Support	0	0	0	0	0
Total (20 Operational Vehicles)	20.066	5.620	(44.675)	12.992	(5.997)
Average Unit Cost	1.003	0.281	(2.235)	.650	(0.300)

* STAGE COST CONTRIBUTED BY INCORPORATION OF J-2S

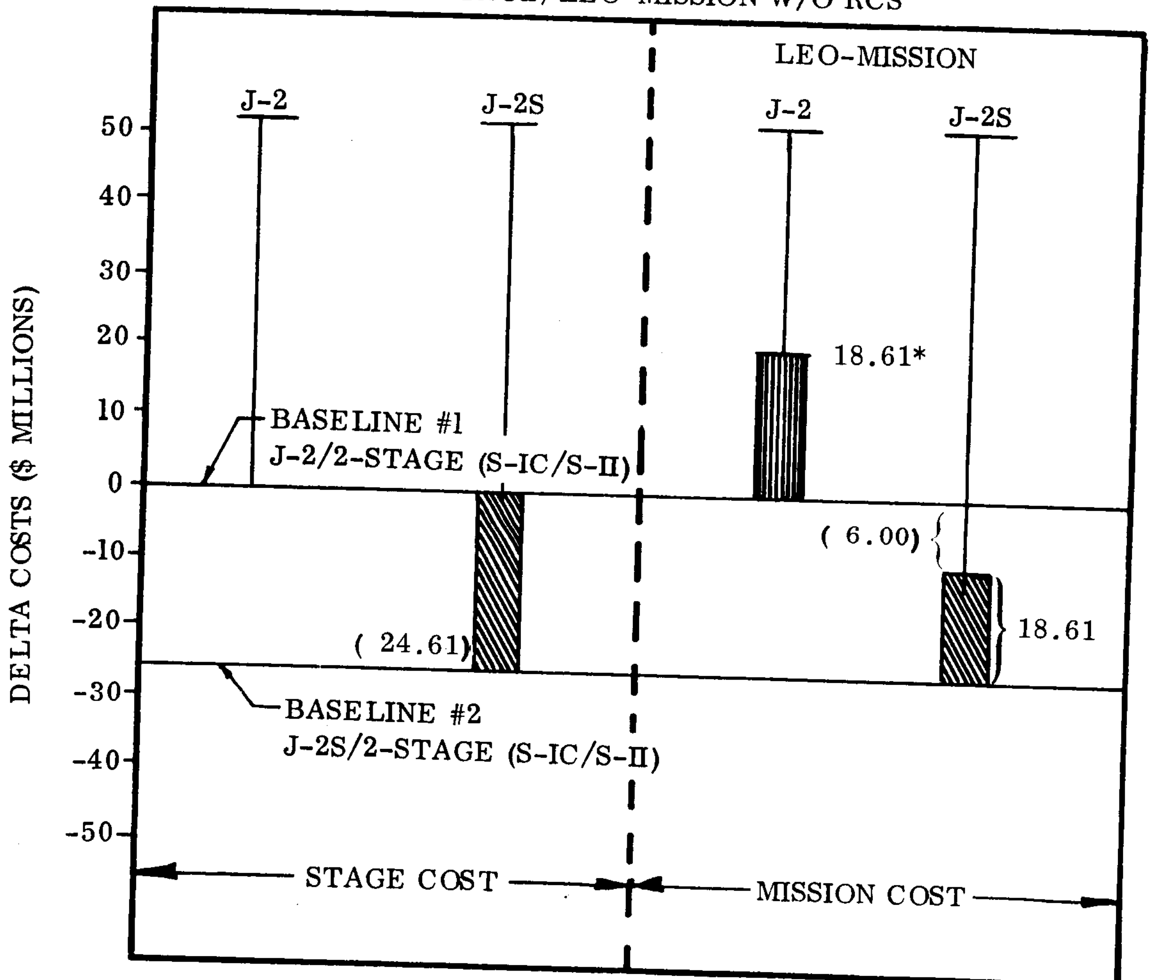
( ) = Savings



COST IS FOR BLOCK OF 20 VEHICLES, TEN YEAR PROGRAM TWO/YEAR PRODUCTION RATE

FIGURE 2.9.4-2 COMPARATIVE COST SUMMARY J-2S/J-2 EQUIPPED VEHICLES

TWO-STAGE/LEO-MISSION W/O RCS



COST IS FOR:
BLOCK OF 20 VEHICLES
10 YEAR PROGRAM
TWO/YEAR RATE

( ) = SAVING *PLUS REPRESSURIZATION SYSTEM COSTS

FIGURE 2.9.4-3 COMPARATIVE COST SUMMARY J-2S/J-2 EQUIPPED VEHICLE

2.9.4.2 (Continued)

The above discussed delta costs are summarized as follows:

Vehicle/Mission Configuration	Development Cost* Plus Mission Cost (20 Vehicles)	Recurring Cost* Per Unit	Cost Amortization
J-2/Saturn V - LOR	Baseline		
J-2S/Saturn V - LOR	23.54	(2.69)	after 9th vh.
J-2S/Saturn V - Synchronous	50.66	(2.69)	after 19th vh.
J-2S/two-stage - LEO	38.68	(2.23)	after 17th vh.

(-) = saving

*($ millions)

2.10 POTENTIAL PROBLEM AREAS AND POSSIBLE SOLUTIONS

No potential problems associated with incorporation of the J-2S engine in the Saturn V vehicle were identified. However, several areas of concern were identified.

a. Due to thermal dynamic conditions encountered during the time from propellant tanks pressurization through the boost periods, engine thrust buildup within the specified start envelope might not be achieved. Additional engine start analysis and engine tests simulating the thermal dynamic conditions should be conducted. If a start problem is discovered, a feed duct conditioning system, e.g., a modified recirculation system, may have to be incorporated.

b. The manpower and critical skills retention associated in a low production rate of two per year is a problem area which was expressed by the stage contractors. This problem is considered to be major but not critical and its effects were included in the stage costs and schedules which were presented for this study.

2.11 CONCLUSIONS AND RECOMMENDATIONS

2.11.1 Conclusions

The study demonstrates that J-2S engines incorporation in the Saturn V Vehicle is both feasible and desirable. It yields cost savings compared with continuing the J-2 application, as well as improvement in vehicle reliability, operational simplicity, increased payload capacity and missions flexibility. The J-2S/Saturn V vehicles are logical improvements to the flight proven Saturn V launch system. Increased versatility extends the application of this system to expanded lunar missions or other missions.

Technical

a. Flight environments were within Saturn V limits or required only minor vehicle modifications.

b. All required stage and/or support operations changes or modifications are minor and within the current state-of-the-art technology. Structural loads were minimized by using the Saturn V external geometry (Apollo spacecraft shape) on all design vehicles.

c. Simplification of stage and support operations results from the use of J-2S engines. Simplified engine operations characteristics reduce the number of required stage systems, prelaunch redlines and failure modes, and simplifies prelaunch operations.

2.11.1 (Continued)

d. The J-2S idle mode increases mission flexibility. Idle mode operation for slosh control and engine start preconditioning eliminates the need for ullage rockets and propellant conditioning systems. Idle mode operation allows the Hohmann transfer flight profile to be used without an auxiliary injection stage. S-II idle mode operation provides the low impulse maneuver for orbit circularization. This mode of operation eliminates any major S-II modification for a repressurization system.

e. J-2S when compared to the J-2 engine, provides increased payload capability as a result of the increased thrust and higher specific impulse.

Resources

a. The J-2S program is self-supporting, resulting in a net cost saving of $30.3 million, for the combined development and a ten year production program (a block of 20 vehicles, produced at a rate of two per year). The development cost of $23.5 million will be amortized by the recurring saving after the production of nine vehicles. The saving is the comparison in cost between a J-2 equipped vehicle for a LOR mission and a J-2S equipped vehicle for the same mission.

b. J-2S engine incorporation incurs no schedule problems. The 3 1/2 years production period for a J-2S equipped SA-518 vehicle is identical to the flow time required for a J-2 equipped vehicle. Pacing items in the schedule are S-IC and S-II stage material requirement.

c. No major or pacing problems are anticipated in manpower requirements, skill retention or skill utilization; however, the S-IVB and S-II contractors are concerned about the skill-mix and management problems associated with a low production rate of two per year. To retain the skills required and yet hold manpower costs at low levels, a combination of cross training, team effort and multi-skill use of individuals would be required.

d. The existing Saturn V government owned facilities and the contractors facilities are adequate to support J-2S engine incorporation requirements.

e. No major vehicle development tests such as dynamic test or wind tunnel test, are required. Development, verification and qualification tests of selected stage components are required to verify operating parameter requirements. Battleship tests are required for the S-II and S-IVB stages to verify stage/engine interface compatibility, characteristics of the J-2S engine and subsystem operating parameters.

2.11.1 (Continued)

The S-II stage contractors recommend acceptance test fire of all the stages. The S-IVB stage contractor recommends at least three S-IVB stages to undergo acceptance test firing.

f. No development flight test is required. The stage test plans will provide assurance that the advanced vehicle, as modified to incorporate the J-2S engines and added mission capabilities, has retained a man rated status.

g. The increase in production rate reduces the stage average unit cost. The average unit cost of a production rate of one per year cannot be estimated accurately without further in depth analysis of manpower and skill retention requirements.

h. The delta costs of the J-2S equipped vehicle for synchronous mission are $27.12 million. These costs are purely mission peculiar costs, independent of the type of engine used. The cost saving of $30.30 million contributed by J-2S engine will pay for mission cost ($27.12 million) and results in a net cost saving of $3.21 million. The above costs are for the ten year program (a block of 20 vehicles produced at a rate of two/year).

i. The delta costs of the two-stage/J-2S equipped vehicle for a low Earth Orbit mission are $18.60 million. The costs are purely mission peculiar costs, independent of the type of engine used. The cost saving of $24.37 million contributed by J-2S engine will pay for the mission cost ($18.60 million) and results in a net saving of $5.77 million. The above costs are for the ten year program (a block of 20 vehicles produced at a rate of two per year).

2.11.2 Recommendations

a. Based on the above conclusions, it is recommended that the J-2S engines be incorporated in the S-II and S-IVB stages of all follow-on Saturn V's (i.e., SA-516 and on).

b. Additional studies should be conducted to further define the merits of a two-stage J-2S/Saturn V for low-Earth orbit missions. Trades should be made to determine the advantages of orbit insertion using J-2S/S-II idle mode operation as compared to using an injection stage. The abort problems should be considered.

A trade study of the Reaction Control System (RCS) for stage/payload attitude control during coast should be conducted to determine whether the RCS system should be on the payload or on the S-II stage.

The studies should include both technical merits and costs for comparisons.

SECTION 3

STUDY OBJECTIVES AND GROUND RULES

3.0 STUDY OBJECTIVES

The primary objective of the J-2S Improvement Study was to assess and report the total systems impact of using J-2S engines in the Saturn V launch vehicle. This primary effort was concerned with the application of the J-2S engine improvements as described in the J-2S Interface Criteria Document, Rocketdyne Report Number R-7211. The principal operational design features of the J-2S included:

a. Idle mode operation;
b. Three-start capability with solid propellant turbine starters; and,
c. Increased thrust and specific impulse.

Vehicle trade data to be developed during Phase I was for the purpose of selecting design vehicles for Phase II.

Phase II was to define in detail the performance capabilities, design environments, vehicle/stage simplifications, support operations, and system engineering and integration analysis; to provide total system impact; and to determine resource requirements. The system changes related to J-2S engines incorporation for the LOR mission were to be distinguished from those related to other candidate missions peculiar requirements.

The study was to define the stages and program in sufficient detail to enable writing of contract end items detail specifications during a follow-on six months Phase C Program Definition Phase.

3.1 GROUND RULES, GUIDELINES AND ASSUMPTIONS

Applicable data from previous and current studies were used wherever possible. The following guidelines and assumptions controlled this study.

3.1.1 Vehicle/Payload Definition

a. The baseline vehicle was SA-511 with J-2 engines in the S-II and S-IVB stages as defined by MSFC. The baseline vehicle definition included payload capabilities and vehicle environment to provide a base for measuring vehicle improvements.

3.1.1 (Continued)

b. Modification to the selected design vehicles was limited to those changes required for compatibility with J-2S engine, and to accommodate, if necessary, additional loads and environments due to increased payload weight and requirements of mission profiles. The S-IC stage was the same as the baseline SA-511 first stage.

c. Apollo/Saturn design criteria were used except where otherwise specified or approved by MSFC.

d. Payload Configurations

1. The Apollo payload shape was used on three-stage vehicles. Payload density was varied with the payload weight.

2. Payload shape for the two-stage vehicle was a S-IVB dry workshop shape plus an Apollo command-service module (same as three-stage shape).

3. Maximum payload envelopes for the two-stage and three-stage vehicles were determined by mission analysis and payload effects studies.

4. Homogeneous mass distribution was assumed for all payloads other than Apollo.

3.1.2 Mission Profiles

The following profiles were used to generate vehicle performance data and to select the mission/vehicle configurations for detail design analysis.

a. Injection into a 72-hour lunar orbit rendezvous (LOR) trajectory, with three stages, after utilization of a 100 NM parking orbit.

b. Direct injection lunar mission (no parking orbit) with three stages.

c. Synchronous orbits of 0^{o} and 28.5^{o} inclination with three stages.

d. Direct ascent with two stages to a 100 NM circular orbit.

e. Direct injection performance capabilities of the two-stage vehicle with orbit altitudes of 100 to 300 NM and launch azimuths of 44 to 110 degrees (measured from North to South over East).

f. Hohmann transfer to circular orbit with two and three-stage vehicles from 100 NM perigee to higher orbit altitudes.

3.1.2 Mission Profiles

g. Direct ascent with three stages to a 100 NM polar orbit using yaw steering to avoid overflight of major land masses. Nominal launch azimuth will be 140 degrees measured from North to East.

h. High energy missions with three-stages direct ascent into 100 NM circular parking orbit (suborbital burn of the third stage), reignition and burn of third stage to a specified energy level. Range of energy levels; C_3:

$$(-20 \leq C_3 \leq +100 \text{ KM}^2/\text{Sec}^2)$$

Nominal launch azimuth will be 72 degrees measured from North to East.

3.1.3 Technical

a. Trajectory and propellant distribution procedures were compatible with methods in use at MSFC. Detailed assumptions concerned with ascent trajectories were discussed and approved by MSFC.

b. A flight performance reserve of 3/4 percent of the total vehicle characteristic velocity was provided for in the last stage. These reserves were considered to be part of the usable mainstage propellants.

c. Atmospheric model and geopotential function were provided by MSFC.

d. Nominal wind assumptions, as furnished by MSFC, were consistent with Apollo wind restrictions. These conditions were:

   1. 99.9 percent probability on pad.

   2. 99.9 percent probability during lift-off (for the first twenty seconds).

   3. 95.0 percent probability during powered ascent with 99 percent wind shear.

   4. Gust conditions as specified by MSFC.

e. Modes of rigid body control were at a minimum control frequency of 0.15 cps and a damping 75 percent of critical. The gains were chosen to produce a desirable compromise between lateral drift, gimbal angle requirements and maximum dynamic pressure.

3.1.4 Program Resources

a. Non-interference with the Saturn/Apollo hardware was assumed.

b. J-2S engine incorporation effectivity was assumed to be vehicle SA-518.

c. A six-month Program Definition Phase would precede J-2S engine incorporation effectivity.

d. Funds were assumed to be available as required.

e. Cost estimates were in 1968 dollars without inflationary factors applied.

f. Schedules were based on assumed delivery of two Saturn V vehicles per year from SA-515, prior to introduction of J-2S engines.

g. The program outlined to qualify the vehicle for operational flights included all facility modifications, hardware, and test operations for all necessary ground testing.

   1. The R&D program would not include a test flight.

   2. The time between the delivery of the last standard Saturn V and the introduction of the first Saturn V with J-2S engines would be six months, unless facility modifications require greater time.

h. The operational program used a rate of two deliveries per year. Costs were calculated for the first ten years of operation with J-2S incorporated (total of twenty operational vehicles). The influence of annual production rate on costs were shown for 1, 3 and 4 J-2S/Saturn Vs per year.

i. Launch vehicle and facility modifications were compatible. Necessary launch facility modifications were identified by KSC.

j. Cost and schedules were based on a one-shift, 5-day week for manufacturing and engineering.

k. Man-rating was required.

l. All necessary propulsion data was supplied by MSFC. Information affecting S-IC, S-II and S-IVB stages was coordinated directly between the stage contractors. Change in ground rules, emphasis, objectives or work statements were issued by MSFC.

m. In documentation, dimensionless parameters were used to the greatest extent possible. The "International System of Units (SI)" were used in addition to the "English Gravitational System" in final presentations and reports where appropriate.

SECTION 4

J-2S ENGINE

4.0 J-2S ENGINE DESCRIPTION

The J-2S engine is designed to operate at any precalibrated nominal vacuum thrust level between 230,000 and 265,000 pounds, using liquid oxygen oxidizer and liquid hydrogen fuel as propellants at a mixture ratio, MR, of 5.5.

The engine is integrated into a complete package of minimum cylindrical dimensions and is capable of being used for single or multi-engine installations on an interchangeable basis.

The following are features of the engine:

a. The engine shall have a single thrust chamber having an expansion ratio of 40:1 and shall be capable of being gimbaled to provide thrust vector control. The gimbal bearing shall be mounted on the thrust chamber injector dome and shall have provisions for attachment to the vehicle stage.

b. The engine start shall be accomplished by the use of a solid propellant turbine starter (SPTS) that is fired to spin the propellant turbopumps and to flush propellant inlet ducts of poor quality propellants.

c. The engine shall be capable of operating in an idle mode (0-5000 pounds nominal vacuum thrust) and a mainstage mode upon stage command signal.

d. The engine shall be capable of being shut down from mainstage to engine cutoff (zero thrust), from mainstage to idle mode operation, and from idle mode to engine cutoff by appropriate stage command signals. The engine shall be capable of shutting down safely as a result of stage liquid oxygen depletion.

e. The engine shall be capable of restarting twice to mainstage while at orbital conditions without inflight servicing for periods up to 20 hours (two 10-hour orbital coasts).

f. The engine will require no preconditioning or servicing of any type from ground equipment or from the stage vehicle within a mininum of one hour prior to vehicle launch, other than the electrical requirements listed in the Electrical Power (External Requirements) section of this document.

g. The engine will require that the user monitor only one engine parameter as a prelaunch "red-line," viz., the He control tank pressure, for flight operational use, provided the environment and other external input requirements are as prescribed in the Engine Start Requirements section of this document.

h. The engine shall be capable of operating in the low thrust idle mode for a maximum of 1000 seconds.

i. The engine will provide a mixture ratio control system for stage propellant management.

j. The engine will have provisions for supplying stage propellant tank pressurization gases during mainstage operation.

k. The engine will provide an accessory drive mechanical output.

l. The engine will have provisions for dumping residual stage propellants, separately, and without ignition.

m. The engine shall have a service life of 3750 seconds of engine total mainstage mode operation and 3500 seconds of engine total idle mode operation. The service life of the engine includes 30 engine starts to mainstage and ten starts to the idle mode thrust level only.

ENVELOPE

The engine is within an envelope approximately 81 inches in diameter and a maximum of 133 inches in length. The envelope is shown in Fig. 2.2-1.

WEIGHT

Single Start Engine

The dry weight of a single start engine including accessories is 3800 pound maximum. A further weight breakdown is shown in Table 4.0-I.

Multiple Start Engine

The dry weight of a multiple start (two restart) engine including accessori and restart accessories is 4050 pounds maximum. A further weight breakdown is shown in Table 4.0-II.

TABLE 4.0-I

SINGLE START ENGINE WEIGHT

Rocket Engine	Weight, pounds
Dry	3235
Burnout*	3288
Wet	3347
Accessories	
Dry	565
Burnout*	625
Wet	630
Total Weight, dry	3800

*Dry weight plus the instantaneous weight of fluids remaining at the termination of rated duration

TABLE 4.0-II

MULTIPLE START ENGINE WEIGHT

Rocket Engine	Weight, pounds
Dry	3235
Burnout*	3288
Wet	3347
Accessories	
Dry	565
Burnout*	625
Wet	630
Restart Accessories	
Dry	250
Burnout*	221
Wet	250
Total Weight, dry	4050

*Dry weight plus the instantaneous weight of fluids remaining at the termination of rated duration

ENGINE PERFORMANCE

MAINSTAGE PERFORMANCE

The engine is capable of being calibrated at any thrust level between 230K and 265K at a mixture ratio of 5.50 ±2 percent as shown in Table 4.0-III. With mixture ratio excursion from the calibration point, the engine will operate within the thrust vs mixture ratio envelope shown in Fig. 4.0-1. The 3σ bands shown in Fig. 4.0-1 include all thrust variations due to engine calibration and variations in engine inlet conditions within the limits established in the External Requirements section of this document. Thrust vs altitude and specific impulse vs mixture ratio are shown in Fig. 4.0-2 and 4.0-3. The oxidizer and fuel flowrate vs mixture ratio are shown in Fig. 4.0-4.

TABLE 4.0-III

MAINSTAGE PERFORMANCE
(AT STANDARD VACUUM STATIC CONDITIONS)

Nominal Thrust, pounds	Minimum Instantaneous Specific Impulse, seconds	Engine Mixture Ratio, o/f	Engine Duct Inlet Pressure psia Total		Rated Duration, seconds
			Fuel	Oxidizer	
230,000 ±3% 265,000 ±3%	427	5.50 ±2%	30*	39**	500

*Fuel density = 4.40 lb/ft^3
**Oxidizer density = 70.79 lb/ft^3

PROPELLANT UTILIZATION

The propellant utilization valve is capable of varying the engine mixture ratio in accordance with Fig. 4.0-1. Propellant utilization capability is provided by bypassing oxidizer from the oxidizer pump discharge.

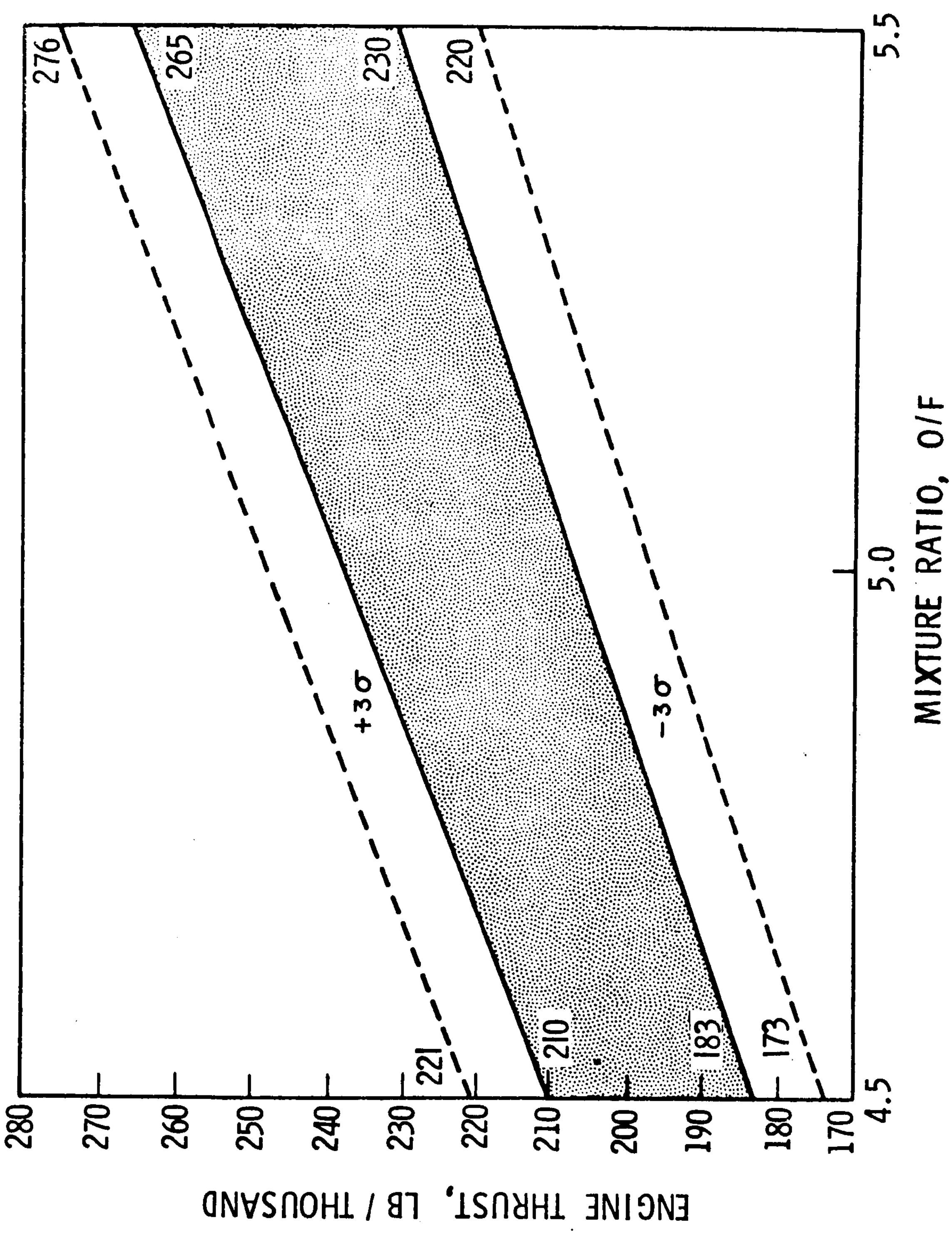


FIGURE 4.0-1 MAINSTAGE CALIBRATION THRUST RANGE

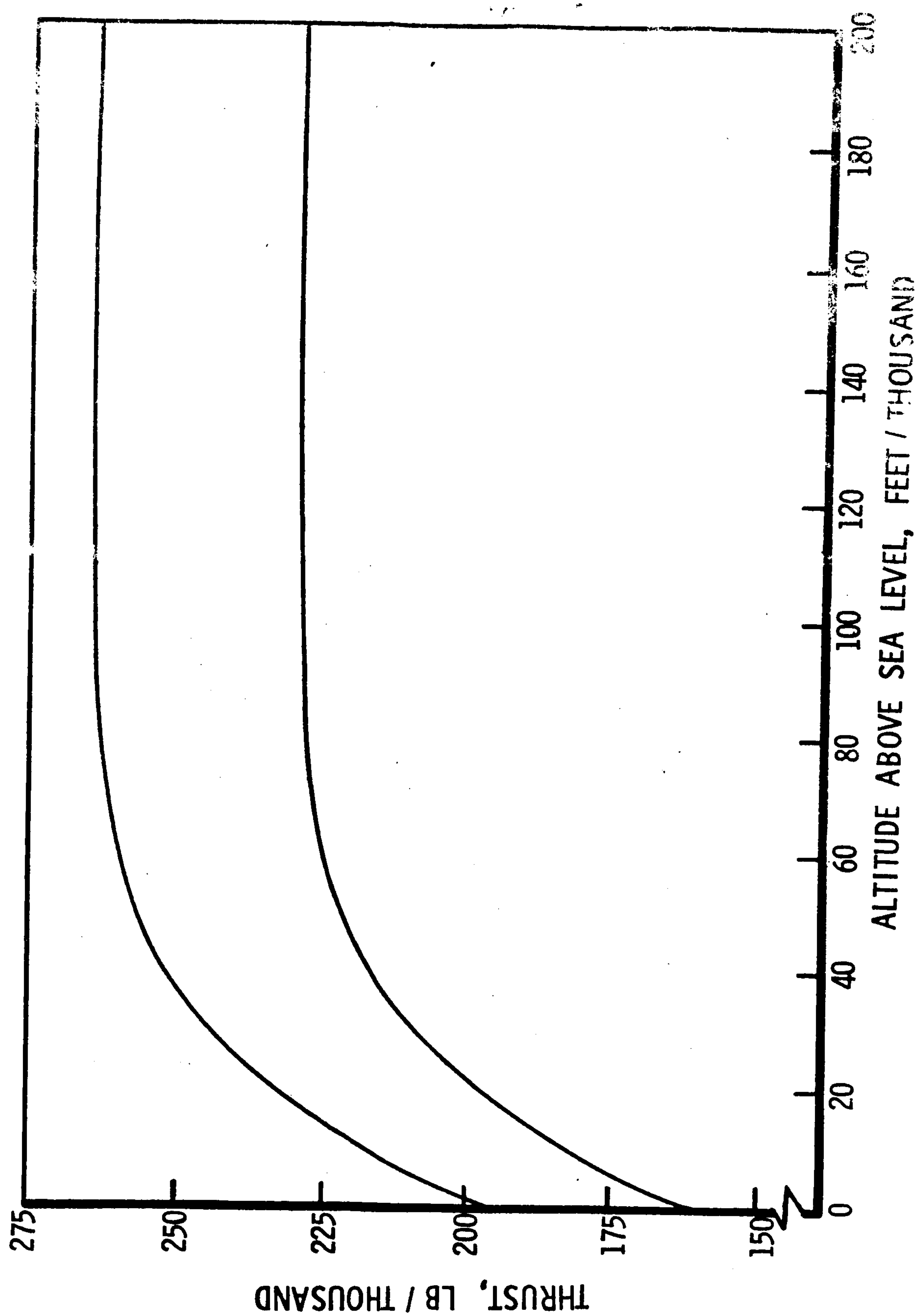


FIGURE 4.0-2 THRUST VS ALTITUDE

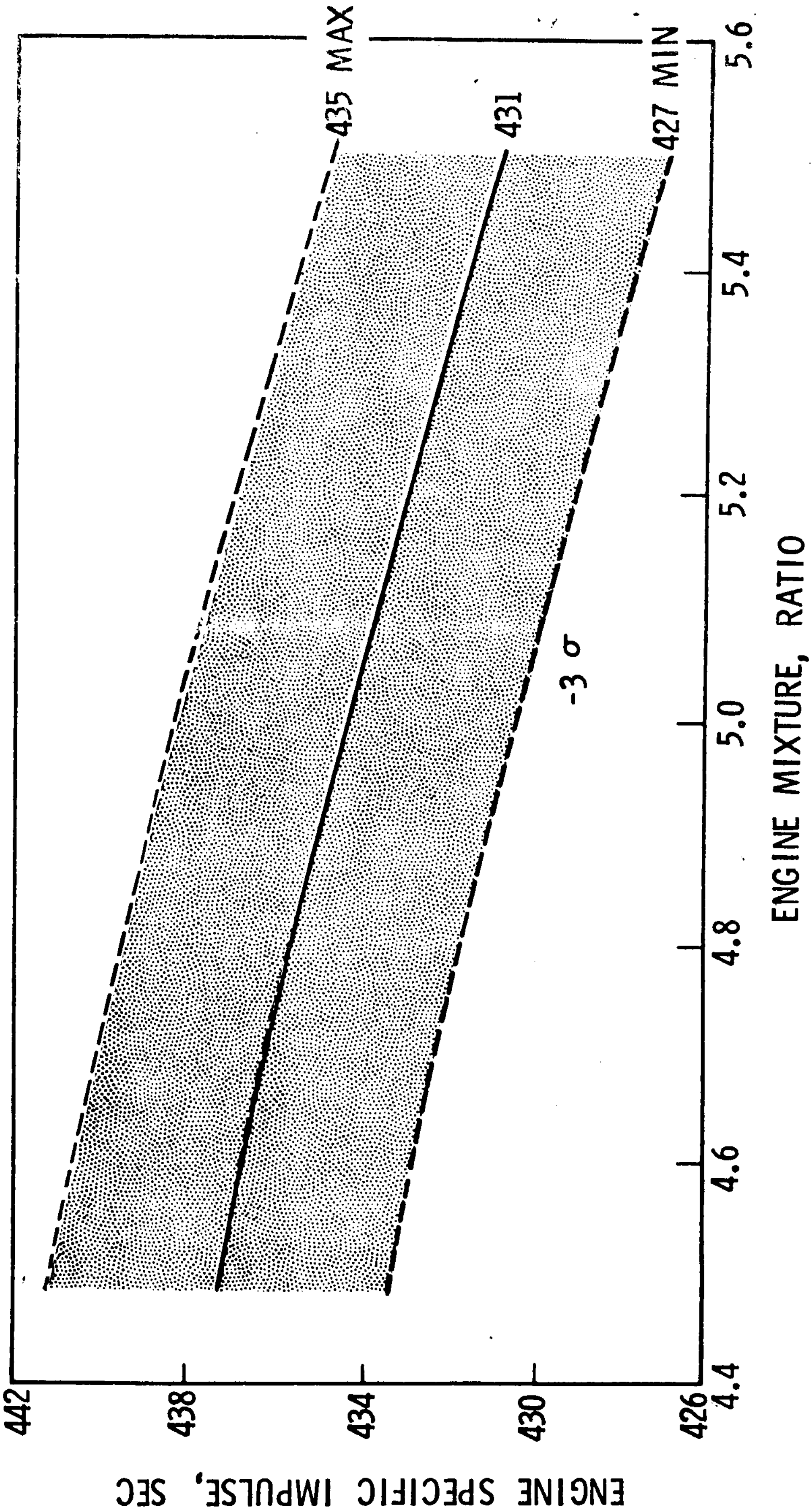


FIGURE 4.0-3 SPECIFIC IMPULSE VS MIXTURE RATIO

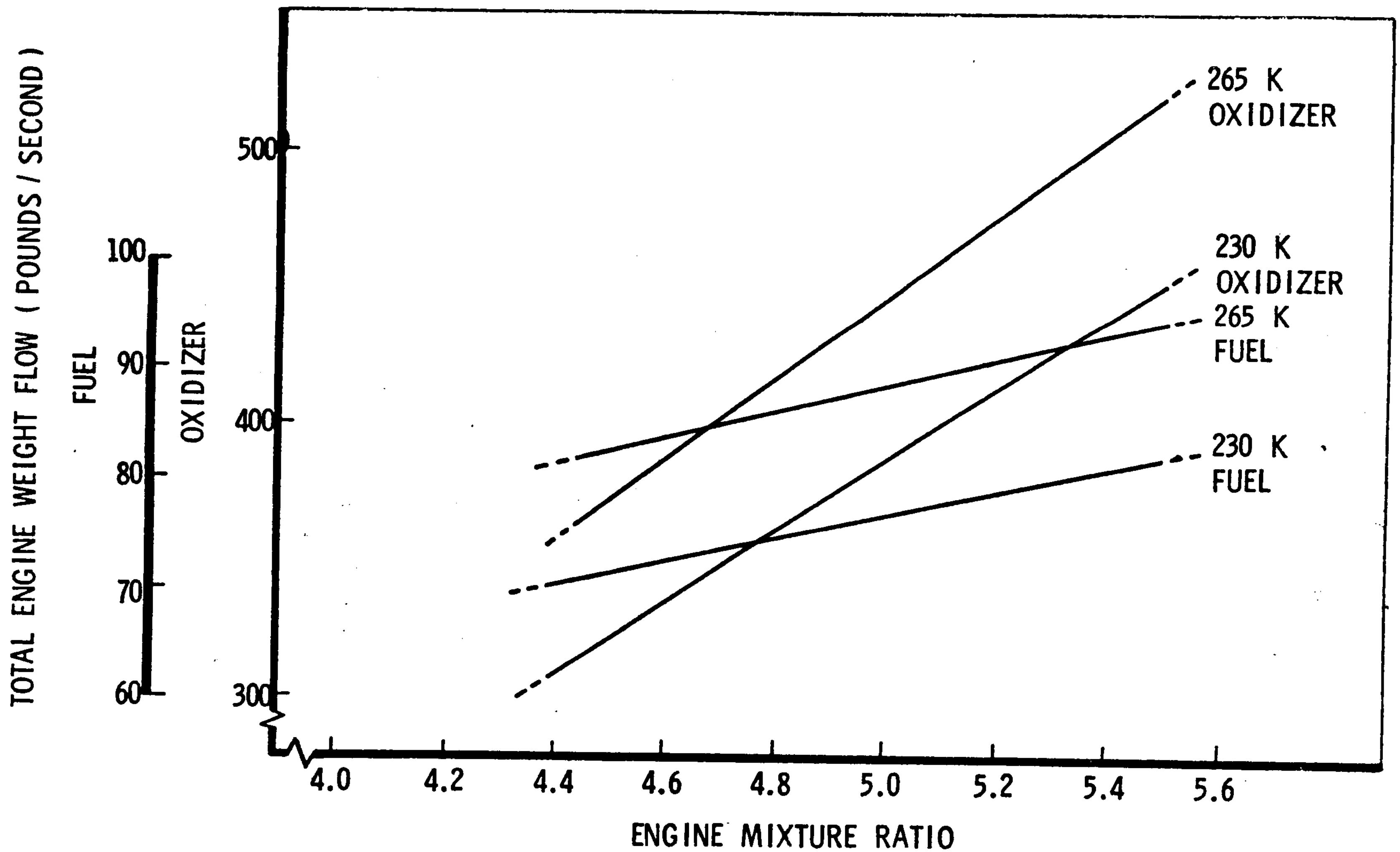


FIGURE 4.0-4 ENGINE MIXTURE RATIO VS TOTAL ENGINE WEIGHT FLOW

IDLE MODE PERFORMANCE

Idle mode steady-state performance is shown in Table 4.0-IV. Idle mode specific impulse vs mixture ratio is shown in Fig. 4.0-5. Idle mode performance as a function of inlet conditions is shown in Fig. 4.0-6 through 4.0-9.

TABLE 4.0-IV

IDLE MODE PERFORMANCE (STEADY STATE, LIQUID)

Engine Duct Inlet Pressures (Total)		Flowrates						
LOX, psia	Fuel, psia	LOX lb/sec	Fuel lb/sec	Mixture Ratio	Minimum Specific Impulse, seconds	Nominal Thrust, pounds (Vacuum)	Chamber Pressure, psia* (Vacuum)	Rated Duration (Total), seconds
39**	30***	12.5 ±2.5	5.0 ±1.0	2.5 ±0.5	280	5000 ±1000	26 ± 5	1000

*Injector end
**Oxidizer density = 70.79 lb/ft^3
***Fuel density = 4.40 lb/ft^3

THRUST TRANSIENTS

Thrust Increase

The rate of thrust increase will not exceed 40,000 pounds per any 10 milliseconds. The thrust increase, corrected to altitude conditions, for all engines will be within the thrust increase envelope of Fig. 4.0-10, provided that the external input requirements to the engine are satisfied. Propellants consumed during the start sequence for engine first start and from mainstart signal after long idle-mode operation are 280±30 lbm fuel and 1010±110 lbm oxidizer.

Idle Mode Thrust Increase. The maximum idle mode duration required to achieve the steady-state condition of Table 4.0-IV shall be 100 seconds.

Inflight Mainstage Thrust Variations. After mainstage signal plus 20 seconds the difference between values of thrust reduced to site conditions averaged over 3 seconds at 5-second intervals shall not exceed 0.5 percent

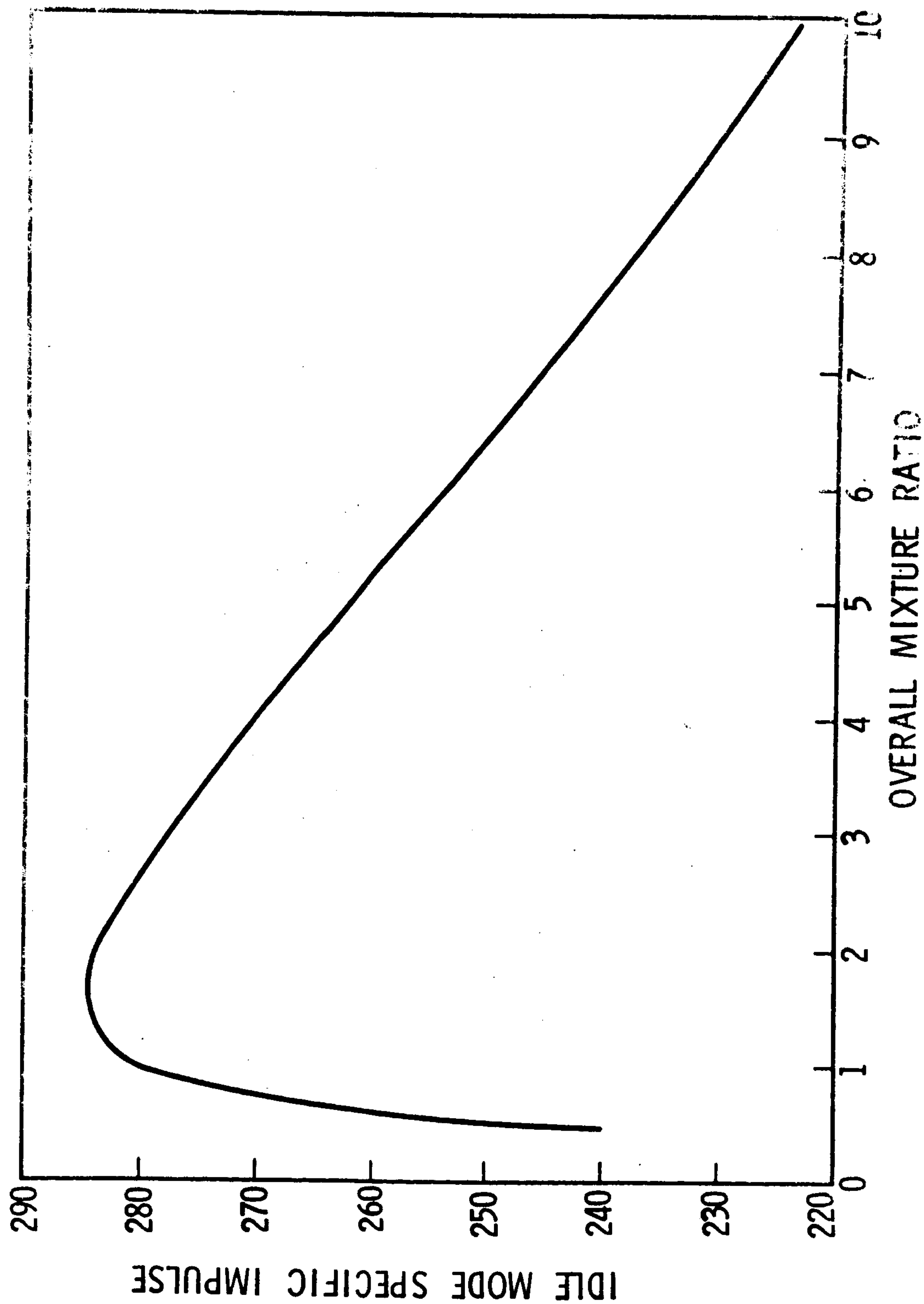


FIGURE 4.0-5 IDLE MODE SPECIFICATION MINIMUM SPECIFIC IMPULSE VS MIXTURE RATIO

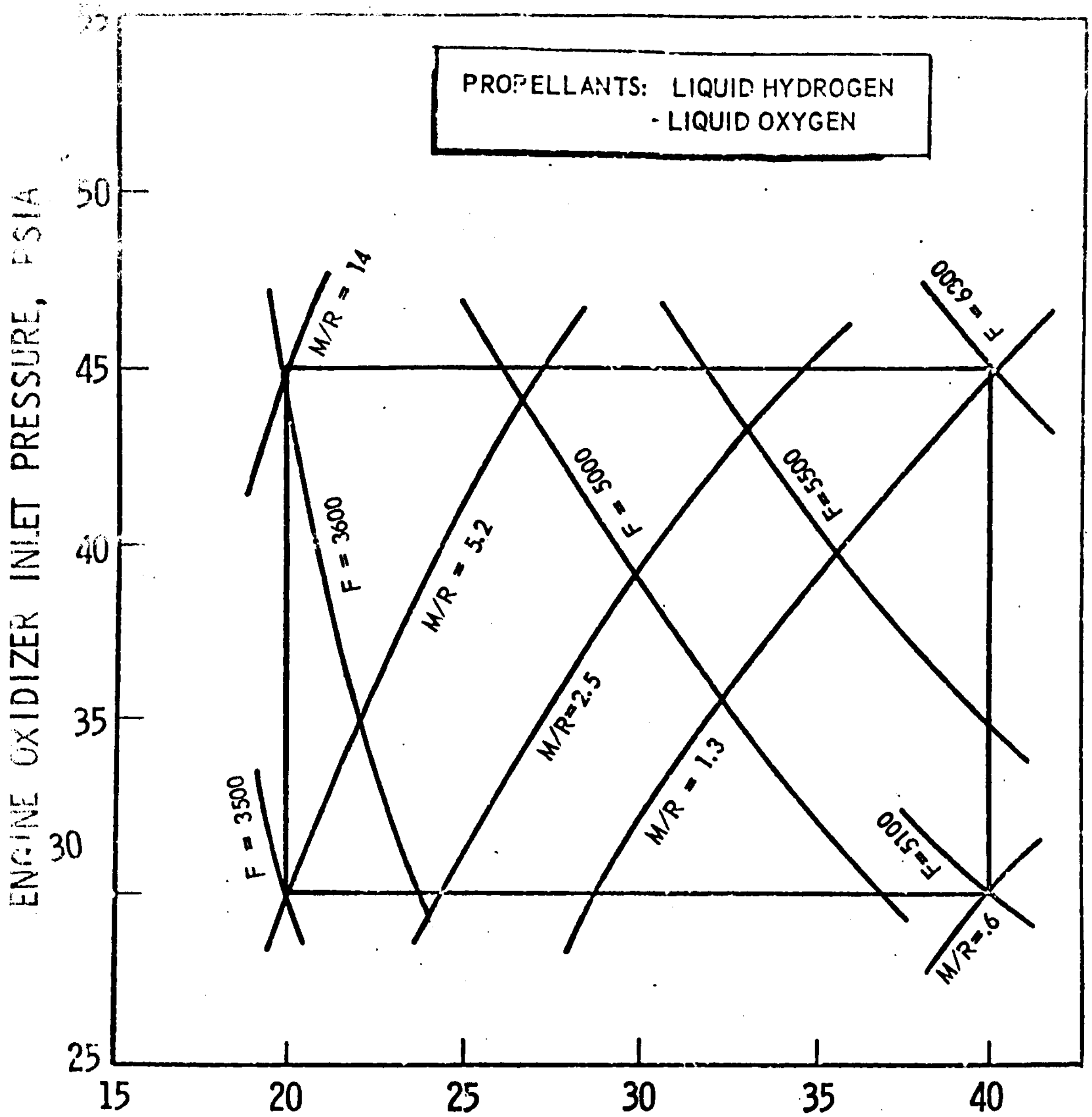


FIGURE 4.0-6 ENGINE IDLE MODE PERFORMANCE

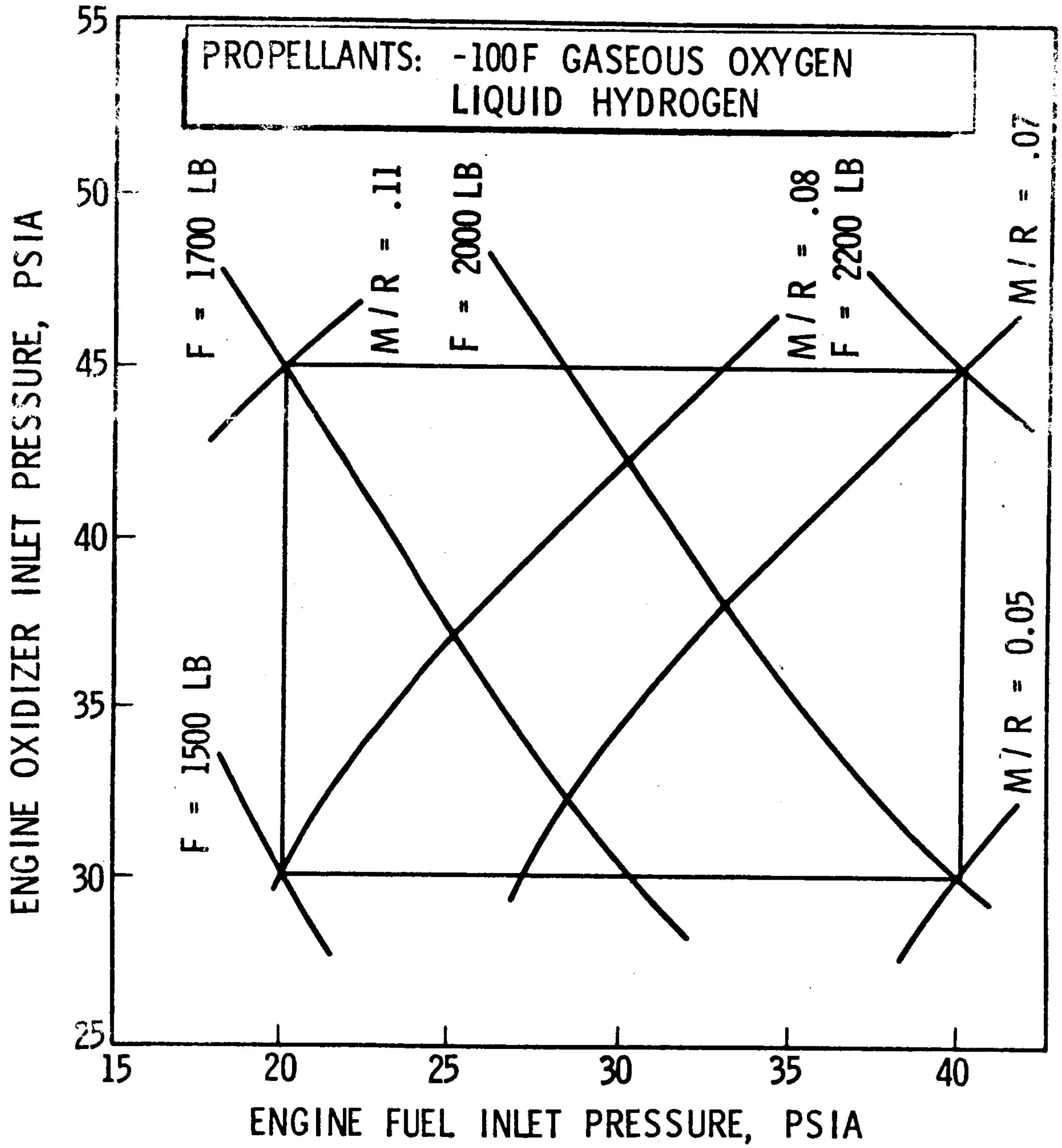


FIGURE 4.0-7 IDLE MODE PERFORMANCE

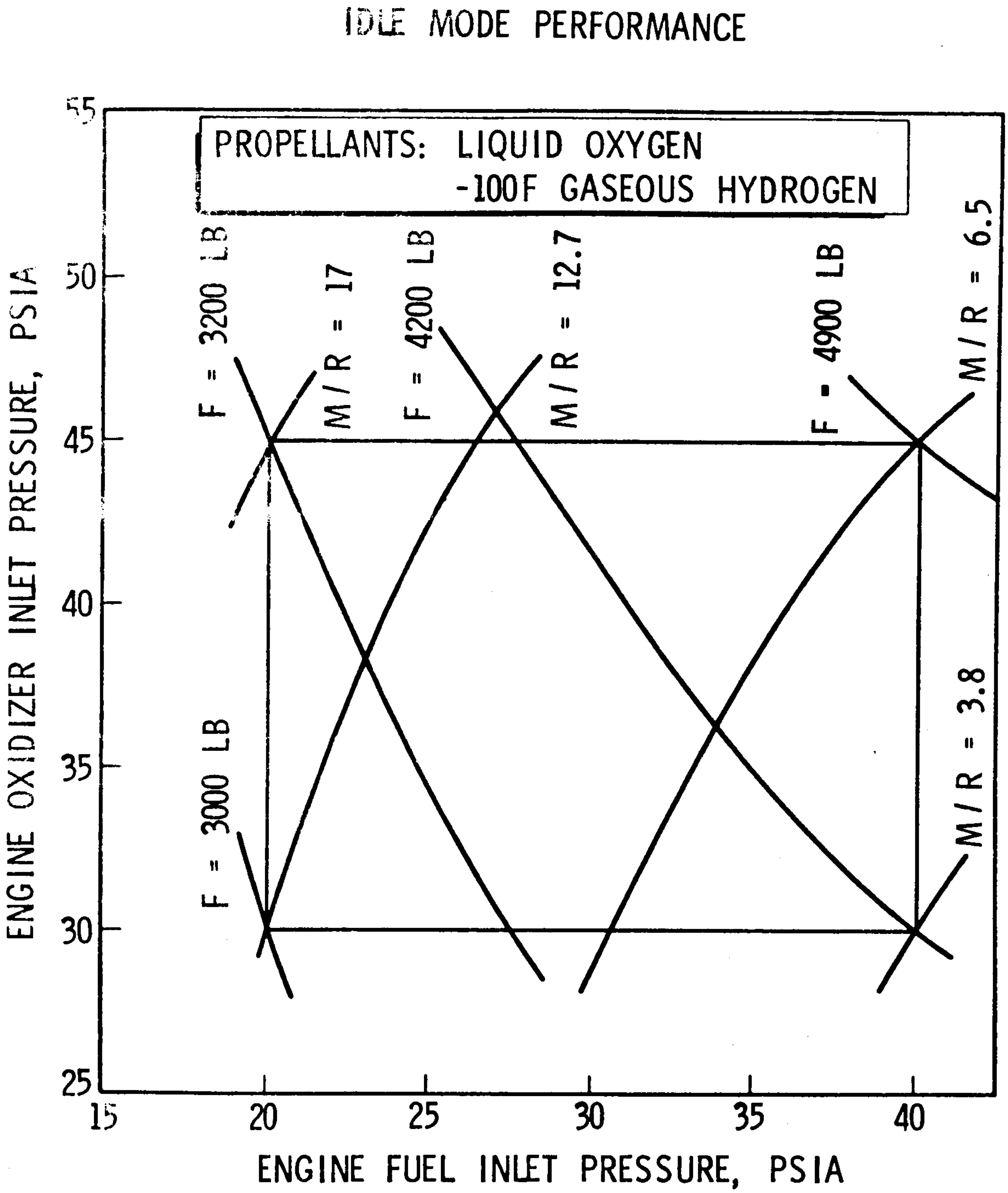


FIGURE 4.0-8 IDLE MODE PERFORMANCE

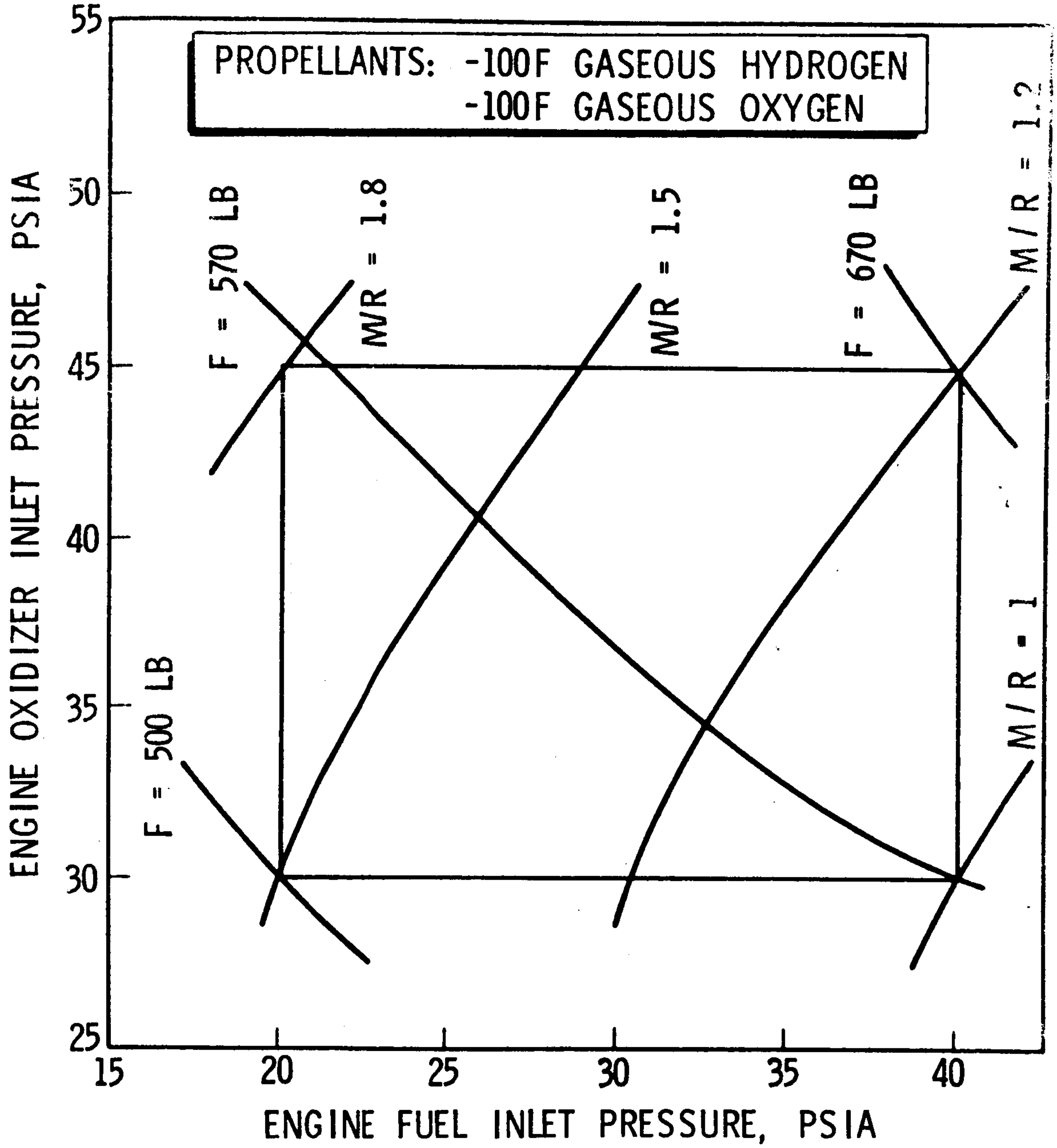


FIGURE 4.0-9 IDLE MODE PERFORMANCE

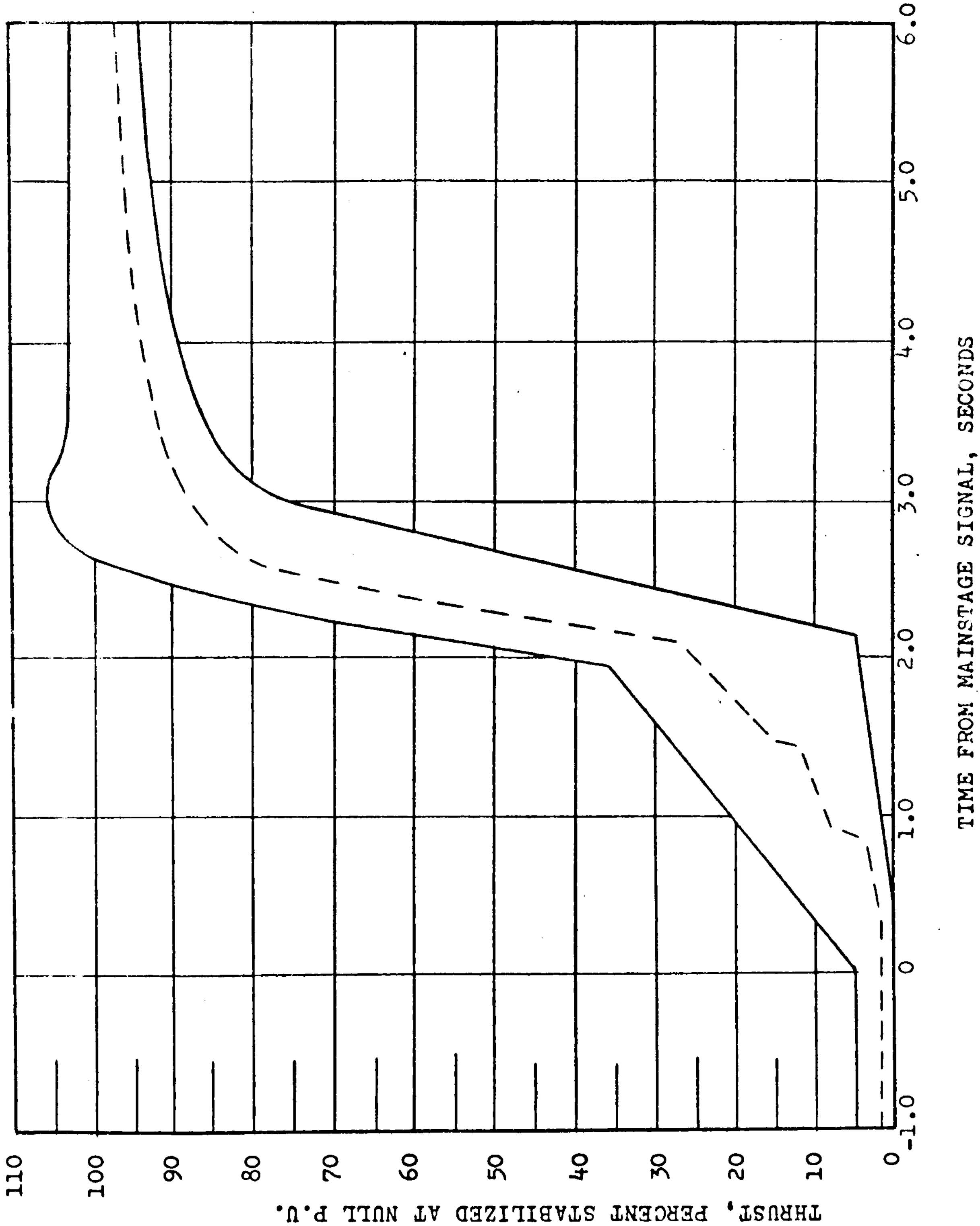


FIGURE 4.0-10 ESTIMATED J-2S THRUST BUILD-UP ENVELOPE

of the rated engine thrust for any 2 consecutive intervals. The sum of the absolute values of the differences between 15 consecutive intervals shall not exceed 3 percent of the rated thrust and the algebraic sum of the differences shall not exceed 1.5 percent.

Thrust Decrease

The thrust decrease requirements of the J-2S engine shall be as follows.

From Mainstage Mode to Shutdown. The thrust decay time from engine receipt of the cutoff signal to a level of 8000 pounds thrust shall not exceed 600 milliseconds and the thrust decay time to zero thrust shall not exceed 2.5 seconds. Engine cutoff impulse is defined as the thrust time integral between engine receipt of the cutoff signal and the time when the thrust reaches zero pounds. The engine cutoff impulse will not exceed 67,000 lb-sec when corrected to the closed propellant utilization valve position under all operating conditions. The test-to-test variation of corrected cutoff impulse for a given engine will not exceed ±6500 lb-sec under all operating conditions. Propellants consumed during the cutoff sequence are 40 ± 8 lbm fuel and 91 ± 18 lbm oxidizer.

From Mainstage Mode to Idle Mode. The thrust decay time from engine receipt of the cutoff signal to an idle mode thrust less than 8000 pounds, shall not exceed 4 seconds and the idle mode thrust level shall be less than 8000 pounds thereafter. Mainstage cutoff impulse is defined as the thrust time integral between engine receipt of the cutoff signal and 4 seconds later. The engine cutoff impulse shall not exceed 110,000 lb-sec when corrected to the null propellant utilization valve position. The test-to-test variation of corrected cutoff impulse for a given engine shall not exceed ±11000 lb-sec under all operating conditions. Propellants consumed during the cutoff sequence are 83 ± 10 lbm fuel and 138 ± 23 lbm oxidizer.

From Idle Mode. The thrust decay time from engine receipt of the cutoff signal to zero thrust shall not exceed 2.0 seconds. Idle mode cutoff impulse is defined as the thrust-time integral between engine receipt of the cutoff signal and the time when the thrust reaches zero pounds. The engine cutoff impulse from idle mode shall not exceed 7000 lb-sec. The test-to-test and engine-to-engine variation of cutoff impulse shall not exceed ±1000 lb-sec under all operating conditions. Propellants consumed during the cutoff sequence are 20 ± 4 lbm fuel and 7.5 ±1.5 lbm oxidizer.

START/RESTART SEQUENCING

Typical timing sequences for three types of initial start and restart operations are shown in Fig. 4.0-11.

VEHICLE TANK PRESSURIZATION SYSTEMS

The engine shall provide a heat exchanger capable of heating either liquid oxygen, tapped from the engine system, or vehicle supplied helium, for use in pressurizing the vehicle LOX propellant tank during engine operation. The engine shall also provide a source of gaseous hydrogen for use in pressurizing the hydrogen tank during engine operation.

Oxygen Pressurant

The gaseous oxygen discharge design point shall be 190 F and shall be based upon nominal 230K engine operating conditions (5.5 mixture ratio) using liquid oxygen as the pressurant at a flowrate of 2 pounds per second. Outlet pressure at the customer connect point shall be a minimum of 400 psia under all engine operating conditions while the liquid oxygen flowrate is varied from 1.0 to 3.0 pounds per second. Refer to Fig. 4.0-12 and 4.0-13 for heat exchanger performance.

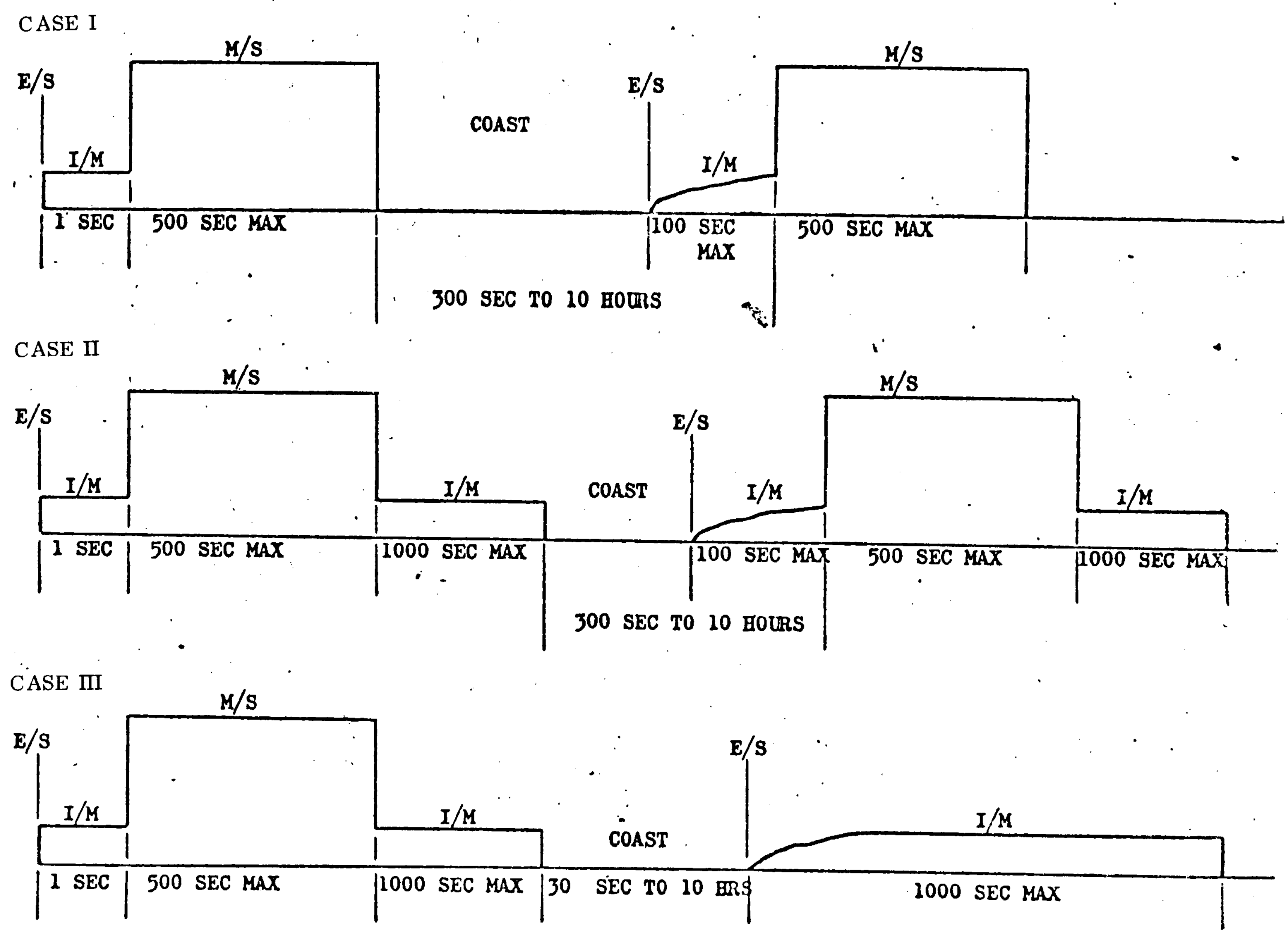


FIGURE 4.0-11 START - RESTART REQUIREMENTS

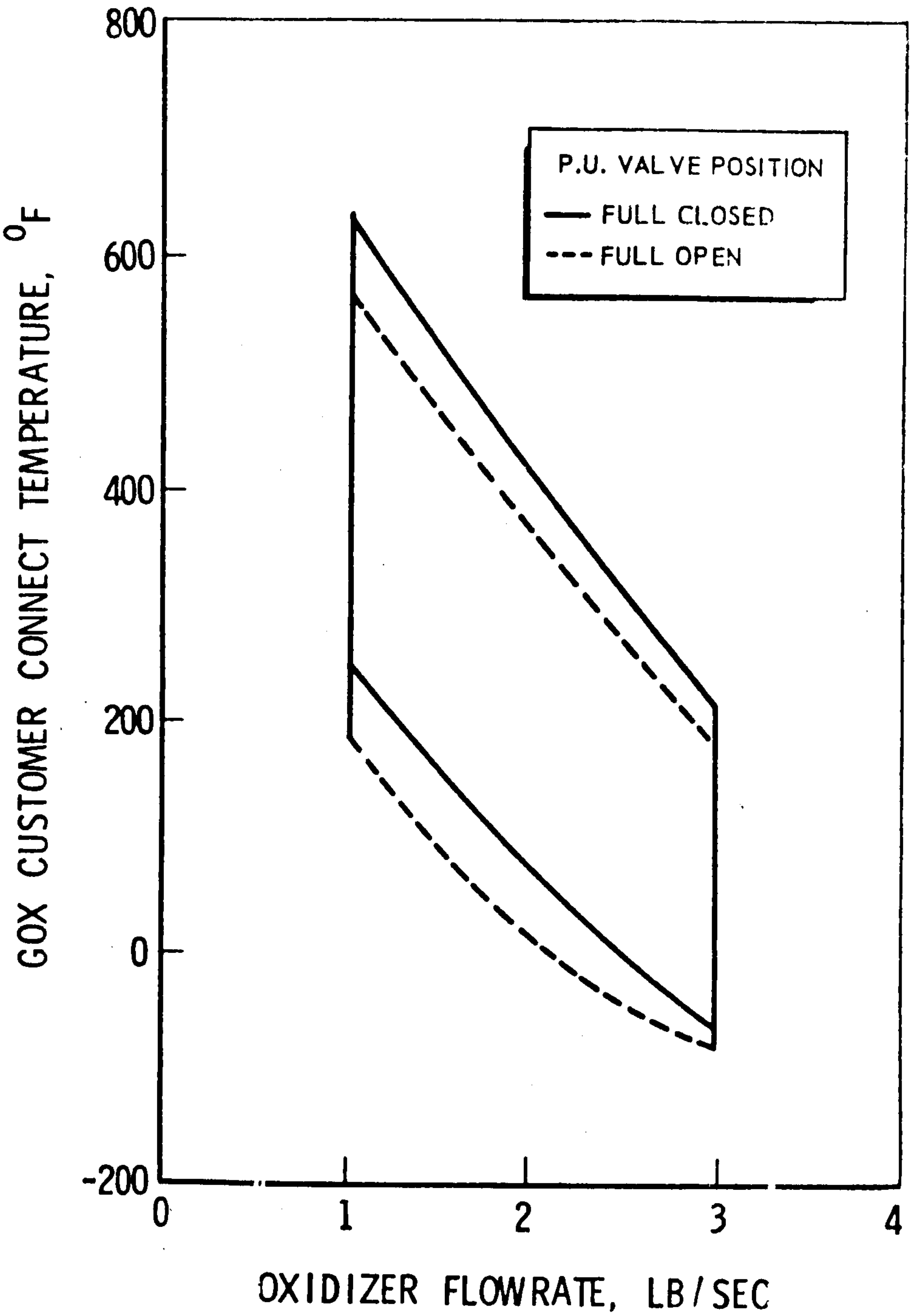


FIGURE 4.0-12 HEAT EXCHANGER GOX CUSTOMER CONNECT TEMPERATURE VS OXIDIZER FLOWRATE

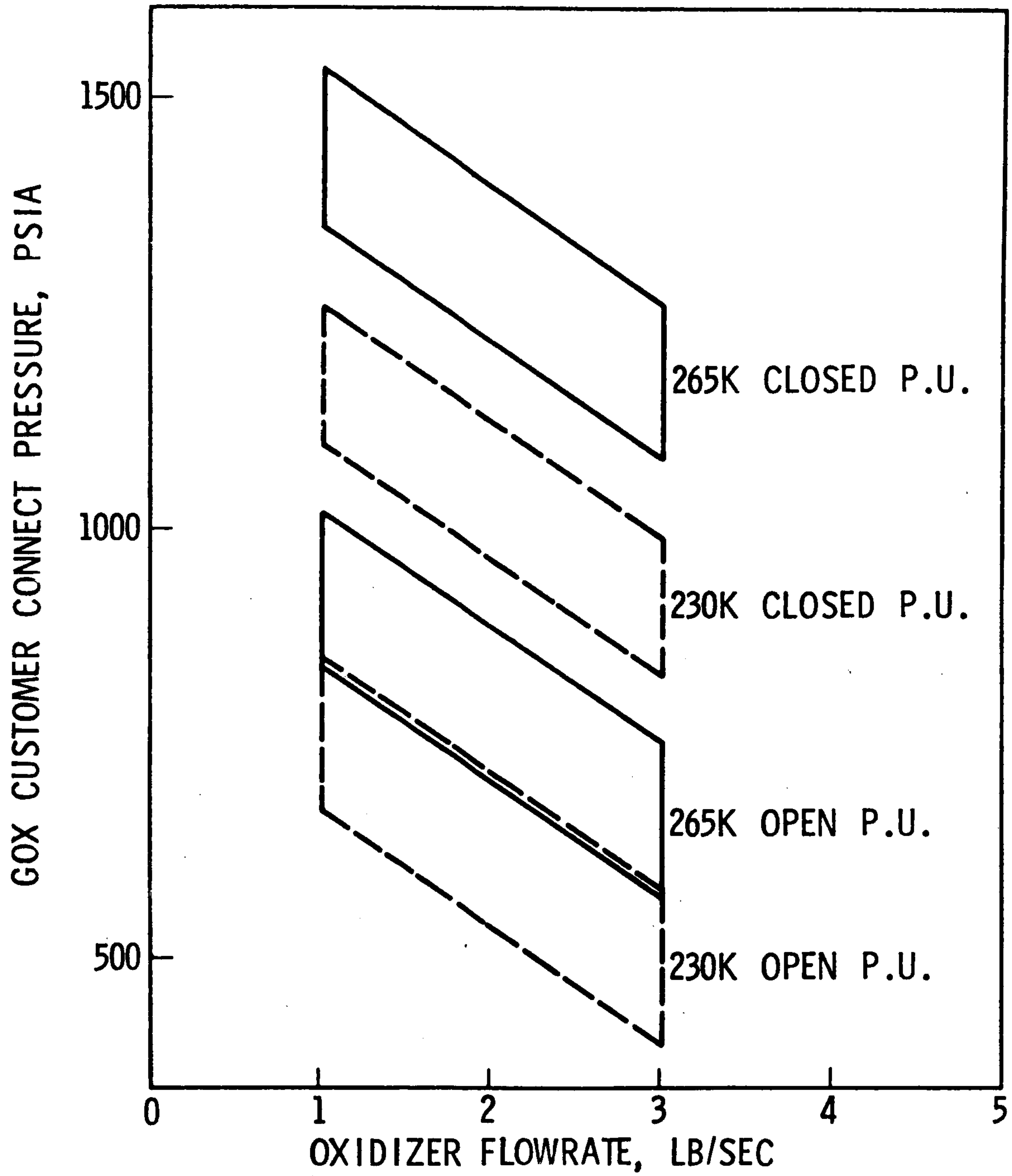


FIGURE 4.0-13 HEAT EXCHANGER GOX CUSTOMER CONNECT PRESSURE VS OXIDIZER FLOWRATE

Helium Pressurant

With helium supplied at an inlet pressure of 350 psia and at an inlet temperature of 40 R, the outlet pressure at the customer connect point shall be a minimum of 200 psia under all engine operating conditions while the helium flowrate is varied from 0.05 to 0.34 pounds per second. Refer to Figures 4.0-14 and 4.0-15 for heat exchanger performance with helium.

The gaseous oxygen and helium Mach number in any portion of the system shall not exceed 0.3 for any specified condition of flowrate and temperature.

Hydrogen Tapoff

The engine shall provide hydrogen pressurizing gas from a thrust chamber tap at a flowrate up to 3.0 lb/sec and at a temperature between -320F and -190F. The gas pressure vs pressurant flowrate at the engine interface are shown in Fig. 4.0-16.

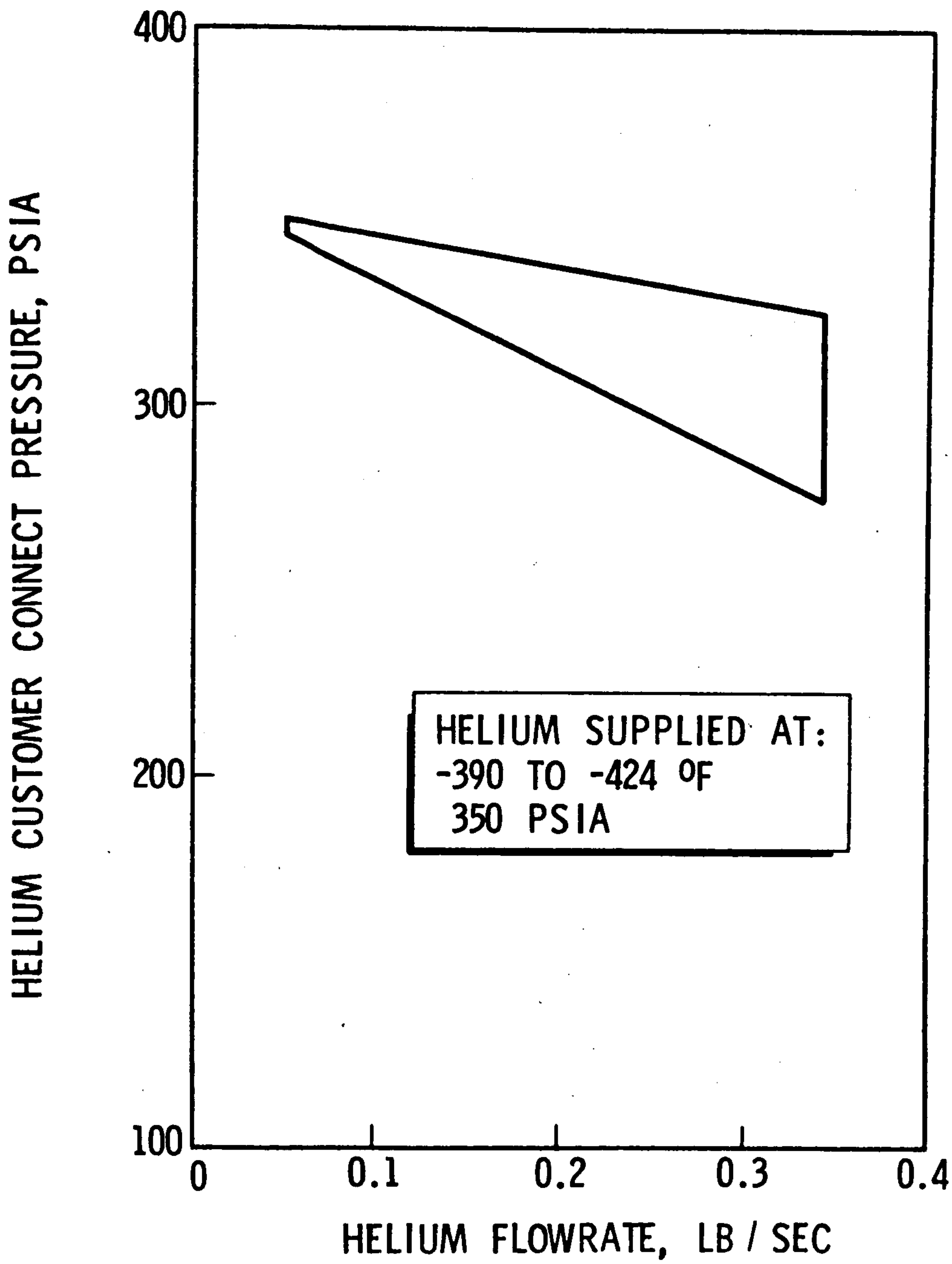


FIGURE 4.0-14 HEAT EXCHANGER HELIUM CUSTOMER CONNECT PRESSURE VS HELIUM FLOWRATE AT ALL ENGINE OPERATING CONNECTIONS

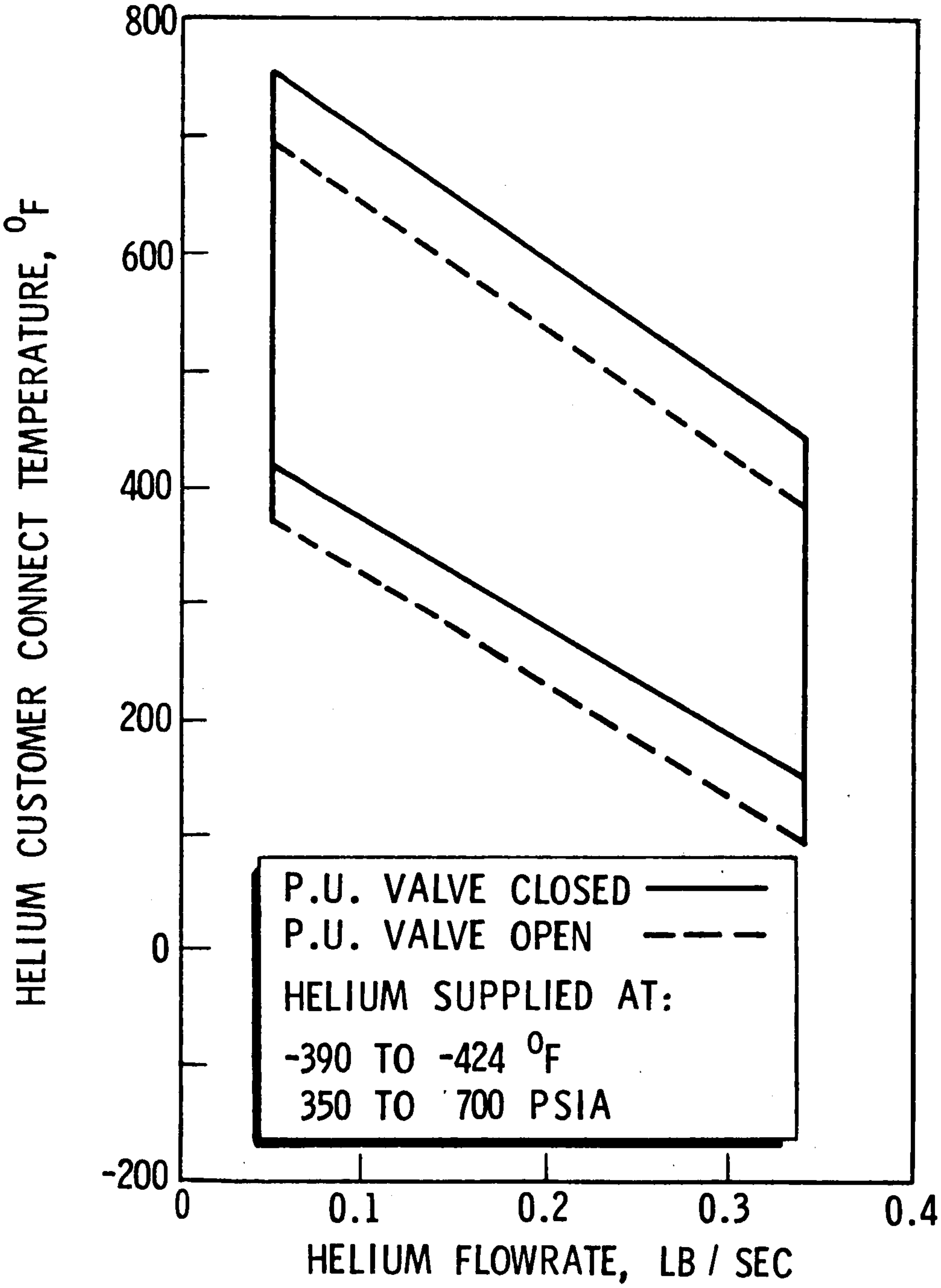


FIGURE 4.0-15 HEAT EXCHANGER HELIUM CUSTOMER CONNECT TEMPERATURE VS HELIUM FLOWRATE

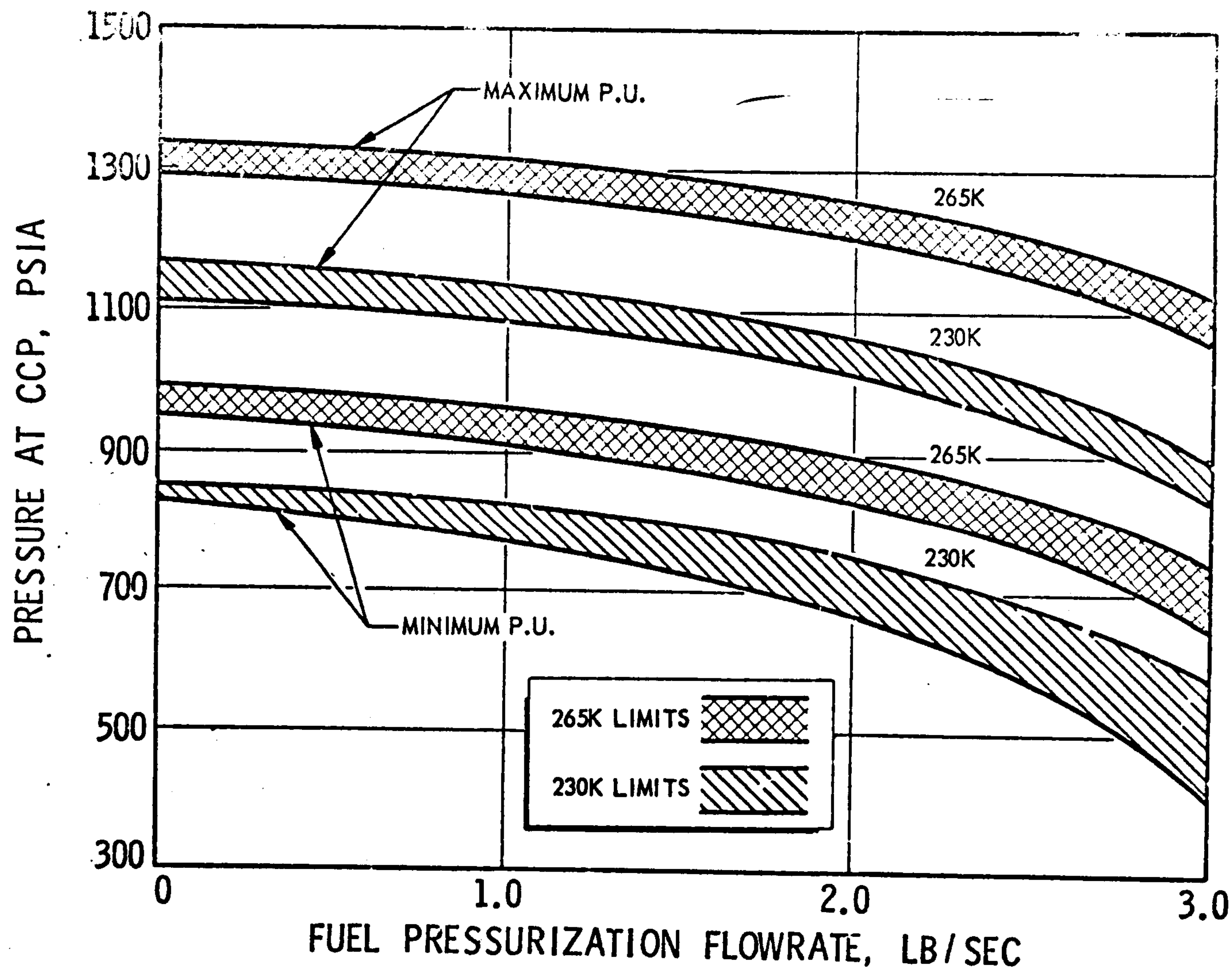


FIGURE 4.0-16 FUEL PRESSURIZATION PERFORMANCE CUSTOMER PANEL PRESSURE VS FLOW

EXTERNAL REQUIREMENTS

PROPELLANTS AND PRESSURANTS

Control of solid impurities in the propellants and fluids must be maintained in accordance with the applicable military specifications.

Oxidizer

The oxidizer supplied to the engine must be oxygen in accordance with MIL-P-25508D, except that total hydrocarbon content must not exceed 75 parts per million (ppm), acetylene content must not exceed 1.5 ppm, the purity must not be less than 99.2 percent, and the particulate content of the oxygen must not be limited by the total weight. Oxygen supplied to the engine must have been processed with ground loading equipment containing nominal 72-micron or finer filters. These filters must have a maximum orifice size of 175 microns.

Fuel

The fuel supplied to the engine must be hydrogen in accordance with MIL-P-27201, and must be supplied from ground loading equipment, equipped with nominal 72-micron or finer filters. These filters shall have a maximum orifice size of 175 microns.

Helium

The helium supplied to the engine must be Grade A in accordance with the Bureau of Mines. The helium must have a water content less than that defined by a dewpoint of -80 F at standard atmospheric pressure. Helium supplied to the engine must have been processed with equipment containing nominal 10-micron or finer filters. These filters shall have a maximum orifice size of 50 microns.

The helium supply required for pneumatic actuation of all engine controls is stored in a spherical tank under 2800 to 3450 psia pressure at ambient temperature for the single start engine and 3000 to 3450 psia pressure at ambient temperature for the multiple start engine. The size of the helium tank is 700 cu in. for the single start engine and 4000 cu in. for the multiple start engine. Helium usage is as shown in Table 4.0-V.

TABLE 4.0-V

HELIUM CONSUMPTION

The table presents the helium consumption requirements during the various phases of operation of the J-2S engine. The usage requirements have been corrected to standard conditions in the helium tank and standard temperature in control system.

		Helium Consumption
a.	Idle Mode Start	10.4 scf
b.	Idle Mode Operation	2.78 scfm
c.	Mainstage Start	1.29 scf
d.	Mainstage Operation	2.89 scfm
e.	Mainstage Cutoff	1.42 scf
f.	Engine Cutoff	2.75 scf
g.	Idle Mode Cutoff	1.33 scf

Gaseous Nitrogen

Gaseous nitrogen in accordance with MIL-P-27401B may be used for purging. The nitrogen total hydrocarbon content shall not exceed 10 ppm. Nitrogen supplied to the engine must have been processed with equipment containing nominal 10-micron or finer filters. These filters shall have a maximum orifice size of 50 microns.

ELECTRICAL POWER

Separate power supply circuits (from stage buss) must be provided for control power, ignition power, instrumentation power, and ground (prior to launch)-supplied power. The external electrical power requirements to be supplied to the engine connect point to operate the engine must be as follows.

Direct Current for Control Power and Ignition Power

Separate power supply circuits (from stage buss) must be provided for ignition power and control power. The 30 vdc (maximum) required for engine start or for initial voltage application must be applied to the engine for not more than 60 seconds. The d-c, peak-ripple voltage must not exceed 2.1 volts for the engine system when measured by using a peak-reading, vacuum-tube voltmeter in series with a 4.0-microfarad capacitor.

The higher of the two values measured when the voltmeter is successively connected for each of the two polarities must be considered the ripple voltage. The maximum voltage transient limit is a 50-volt positive pulse with a time width of 10 microseconds and a repetition rate of 20 pulses per second.

Control Power

A maximum of 20 watts shall be provided for control power prior to engine start, 200 watts during both pre- and post-mainstage idle mode, 375 watts throughout mainstage, 100 watts for 1 sec following C/O, and 20 watts thereafter (refer to Fig. 4.0-17).

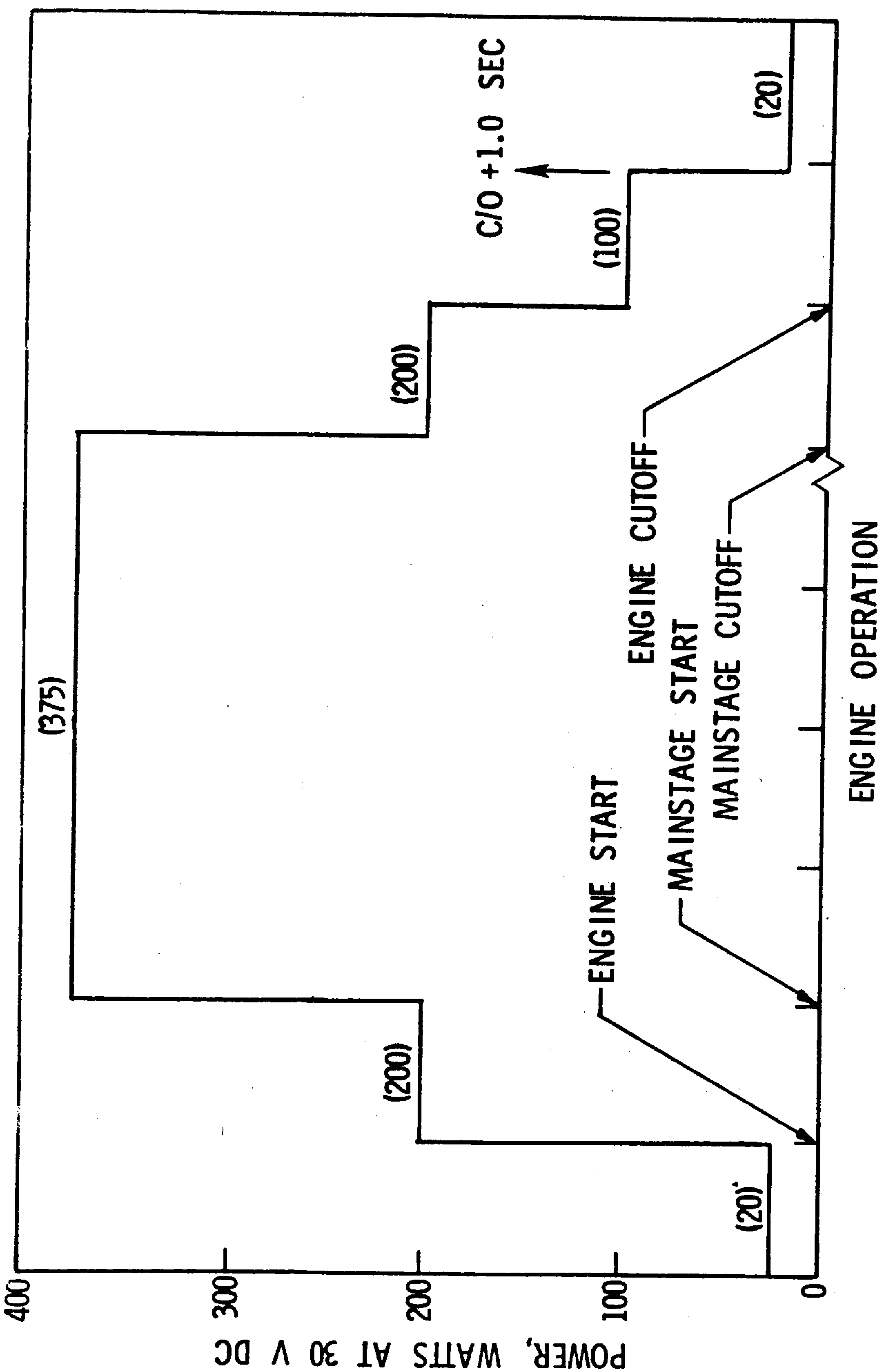


FIGURE 4.0-17 J-2S ENGINE CONTROL POWER REQUIREMENTS

Ignition Power

SII/S-IVB First Start Ignition Power. A maximum of 475 watts at 24 to 31 vdc shall be provided for 1.0 second after engine start, then 325 watts until 4.0 seconds after mainstage start signal, refer to Fig. 4.0-18.

Extended Idle Mode Ignition Power. A maximum of 475 watts at 24 to 31 vdc shall be provided for 1.0 second after engine start, 310 watts for 22 seconds of idle mode and 25 watts for the duration of idle mode. Refer to Fig. 4.0-19.

Alternating Current

A maximum of 40 watts continuous at 108 to 121 vac, single phase, 400 cps, for fixed-phase winding; 15 watts control-phase winding of the propellant utilization actuator motor shall be provided.

Engine Instrumentation

Instrumentation power is as shown in Fig. 4.0-20 and will be as follows.

Direct Current. Separate d-c circuits are provided for instrumentation power and all electrical simulation circuits except pump speeds and flows. The d-c voltage limits for all, except valve position potentiometers and temperature transducers, is 24 to 32 vdc. The peak ripple voltage on this supply will not exceed 0.1 volt when measuring with a peak-reading, vacuum-tube voltmeter in series with a 4.0-microfarad capacitor. The higher of the two values measured, when the voltmeter is successively connected for each polarity, shall be considered the ripple voltage. The maximum voltage transient limit is 50 volts (positive pulse), with a width of 10 microseconds, and a repetition rate of 1 pulse per second.

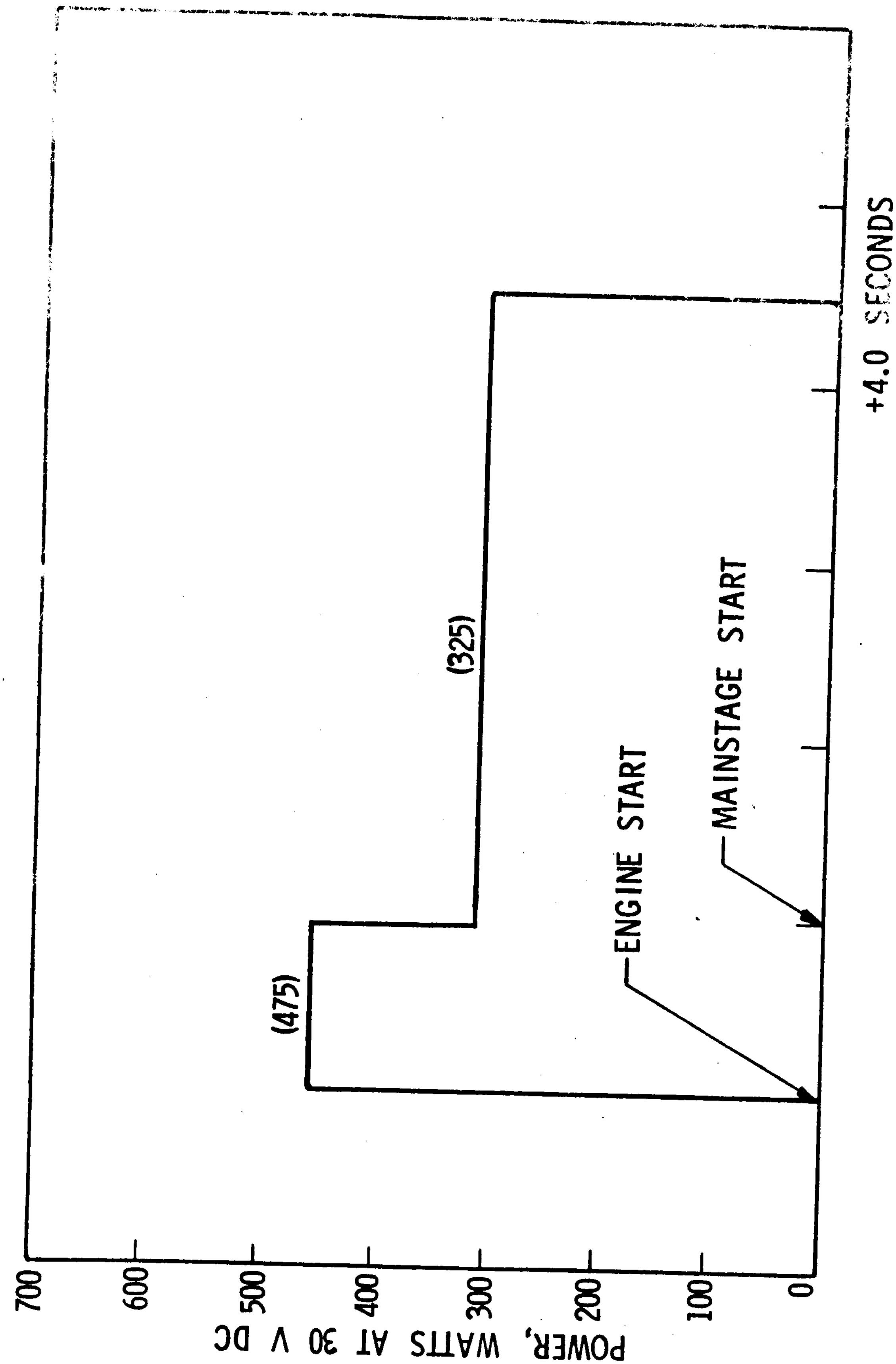


FIGURE 4.0-18 J-2S IGNITION POWER REQUIREMENT

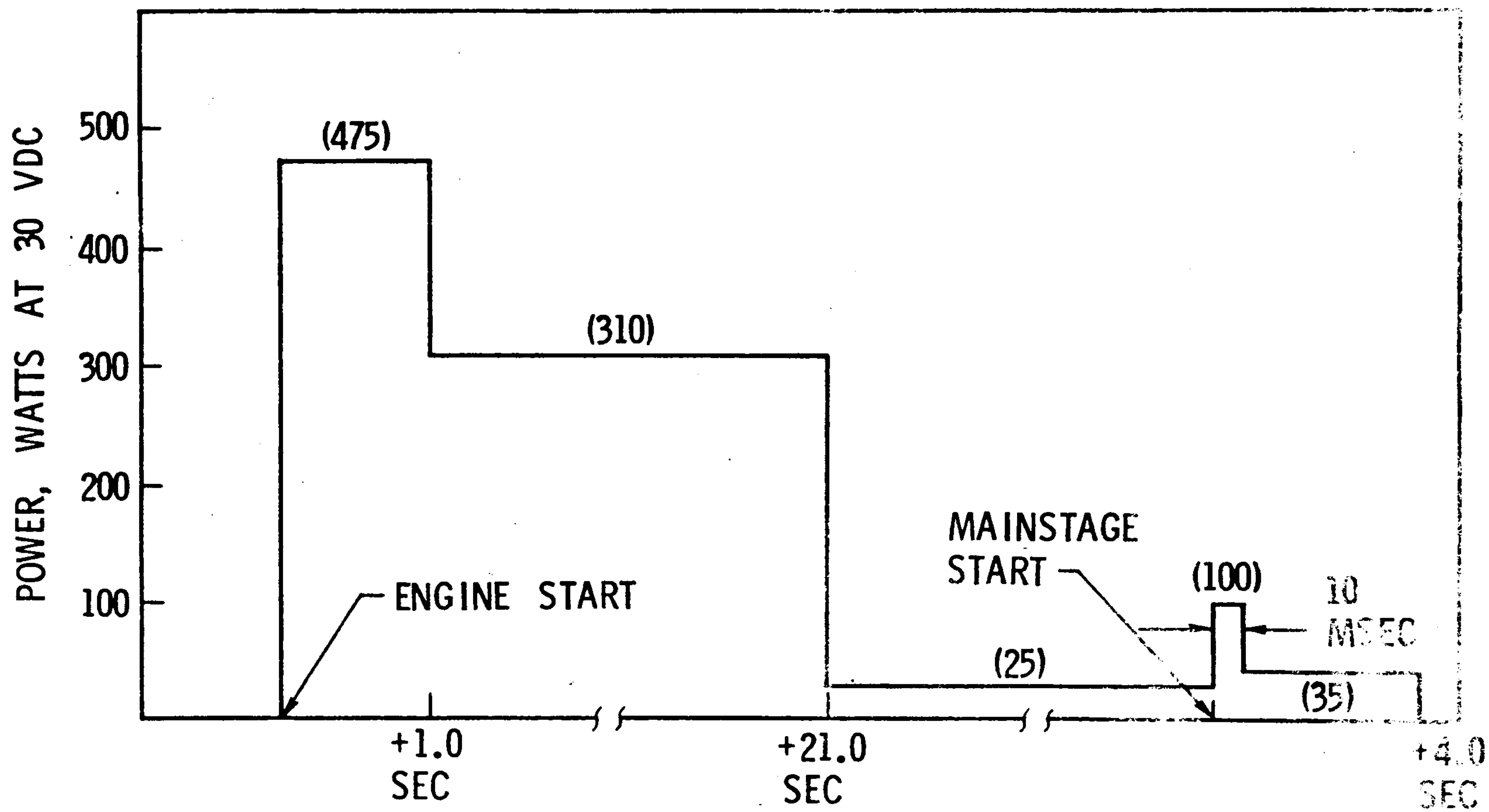


FIGURE 4.0-19 EXTENDED IDLE MODE START

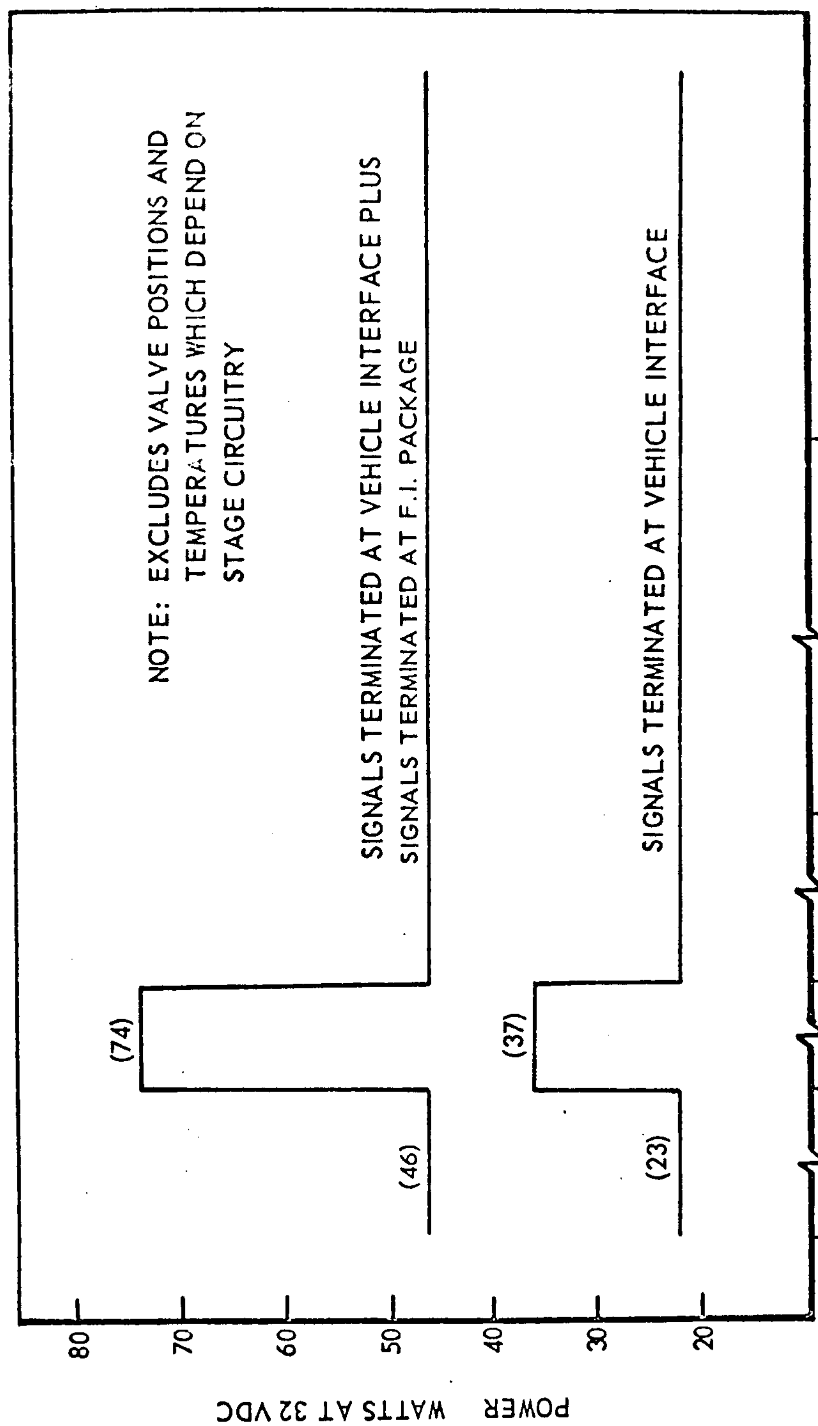


FIGURE 4.0-20 INSTRUMENTATION POWER REQUIREMENTS PRESSURE AND FLOW TRANSDUCERS

The d-c voltage limit for the valve position potentiometers is 5.000 ±0.010 vdc. The peak-ripple voltage on this supply will not exceed 0.025 volt when measuring with a peak-reading vacuum-tube voltmeter in series with a 4.0-microfarad capacitor. The higher of the two values measured, when the voltmeter is successively connected for each polarity, will be considered the ripple voltage.

The voltage level required for the temperature transducers is determined by the stage signal conditioning circuits, as limits to that value which does not cause the current through the resistive element to exceed 1 milliampere.

The engine instrumentation power is as follows:

Parameter	Voltage	Power
Pressure Transducers and Flowmeter Pickups	24 to 32 vdc	Eighteen watts (maximum) for flight. See Fig. 4.0-22 for other requirements. Two-watt addition for each simulation during electrical simulation
Valve Position Switches	24 to 32 vdc	To be determined by stage circuitry
Valve Position Potentiometers	5 vdc	0.5 watt (maximum)
Temperature Transducers	5 vdc	To be determined by stage circuitry
Heaters	TBD	To be determined
Pump and Flowmeter Speed Simulation	10 vac	0.2 watts per simulation

Alternating Current. A-c voltages required for obtaining simulation on the pump speed transducers and main propellant flowmeters are as follows:

1. 0.2 watt maximum for each coil at 10 vac, single phase, 208.2 ±0.2 cps for the flowmeter pickup simulation
2. 0.2 watt maximum at 10 vac, single phase, 6200 ±5 cps for fuel pump speed transducer simulation
3. 0.2 watt maximum at 10 vac, single phase, 2290 ±2 cps for the oxidizer pump speed transducer simulation

PURGE REQUIREMENTS

The engine feed systems, upstream of the main valves, must contain a dry inert atmosphere prior to the introduction of any propellant in order to preclude any ice or combustible mixture formation. Prior to chilling the engine hardware by introducing liquid propellant into the feed system upstream of the main propellant valves, all moisture must be removed from downstream of the main valves by purging as shown in Fig. 4.0-21 for static testing or Fig. 4.0-22 for flight.

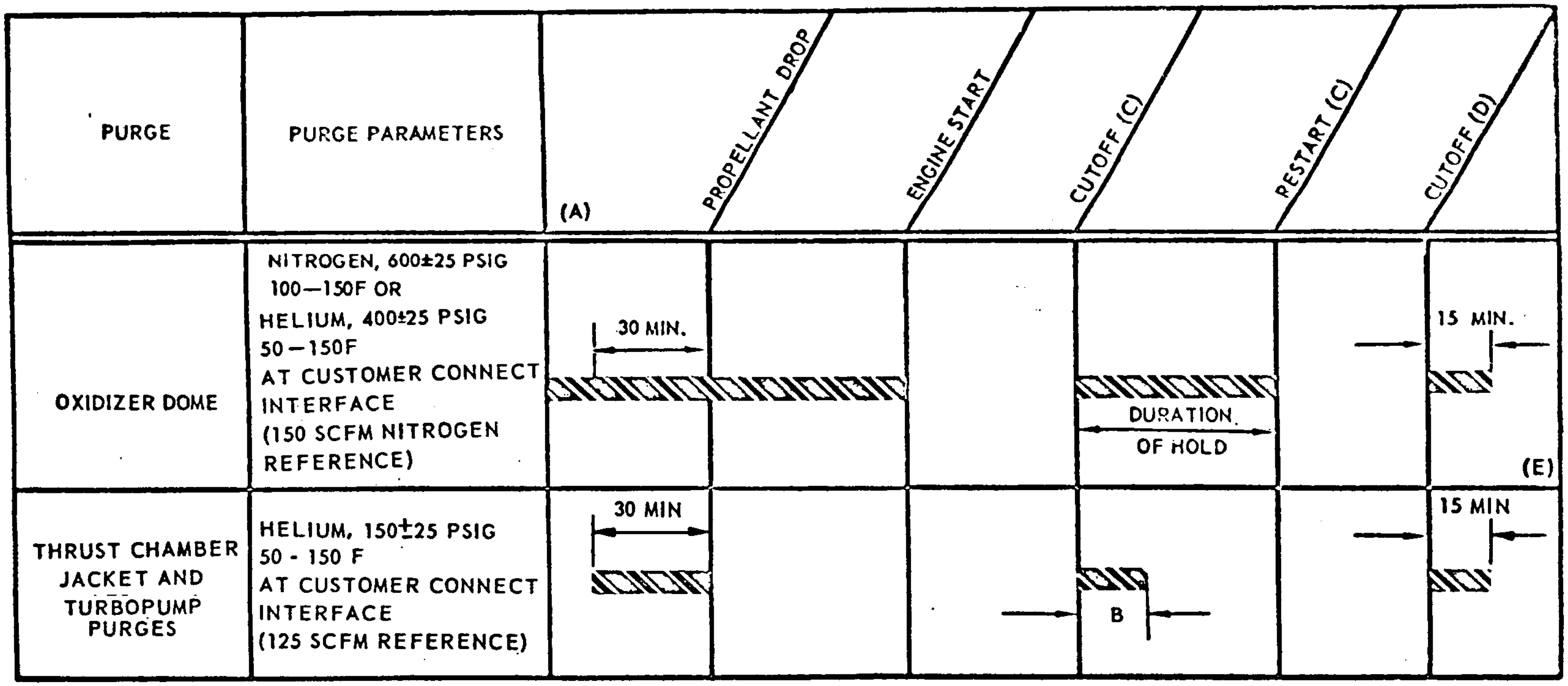
In a test situation where the normal operational sequence has been interrupted and the engine firing is delayed and where the engine is still to be fired before its hardware is expected to return to ambient temperature, a suitable quantity of dry inert gas must be introduced through the oxidizer dome to prevent ice or frost formation on the main injector hardware prior to subsequent tests. Visual inspection of the main injector hardware must be made prior to engine start to verify an ice-free condition.

Special Flight Requirements

In an abort situation (a preliftoff situation where the normal operational sequence has been interrupted after the drying purges have been completed), the engine hardware must be satisfactorily protected from atmospheric moisture by either protective closures, a dry boattail environment, or dry gas flow through the thrust chamber.

ENGINE NPSH (REQUIRED)

Engine operation at maximum propellant utilization conditions will be stable with a pump head loss due to cavitation not to exceed 4 percent at the NPSH values shown in Fig. 4.0-23 and Fig. 4.0-24.



NOTES:

A. PRIOR TO INITIATING PRE-TEST PURGE PROCEDURE FOR A SINGLE TEST ON THE FIRST TEST OF A SERIES, A MOISTURE CHECK SHALL BE MADE OF THE INJECTOR, TAPOFF PORTS, AND FILM COOLANT TUBES. IF NO MOISTURE IS PRESENT, NORMAL PRE-TEST PURGES SHALL BE USED. IF MOISTURE IS FOUND, NORMAL OXIDIZER DOME AND THRUST CHAMBER JACKET PURGES SHALL BE INITIATED FOR 30 MINUTES AFTER WHICH A MOISTURE INSPECTION SHALL BE MADE. THIS PROCEDURE SHALL BE REPEATED UNTIL THE SYSTEM IS DRY.

B. DURATION OF HOLD OR 15 MINUTES MINIMUM.
C. REPEAT FOR SECOND RESTART.
D. CUTOFF PROCEDURE FOR SINGLE FIRING.
E. NORMAL POST-TEST PURGE SHALL BE USED FOLLOWING AN ABORTED TEST.

FIGURE 4.0-21 STATIC TEST PURGE PROCEDURE

PURGE	PURGE PARAMETER	START VEHICLE PROPELLANT LOADING	LIFTOFF	ENGINE START	ENGINE CUTOFF	
NITROGEN BOATTAIL PURGE						
OXIDIZER DOME	NITROGEN, 600 ± 25 PSIG AT 100° TO 150° F OR HELIUM 400±25 PSIG 50° TO 150° F AT CUSTOMER CONNECT PANEL (150 SCFM NITROGEN REFERENCE)	30 MIN				
THRUST CHAMBER JACKET AND TURBOPUMP PURGES	HELIUM, 150±25 PSIG (a) 50° TO 150° F AT CUSTOMER CONNECT PANEL (125 SCFM REFERENCE)	30 MIN				

(a) THIS THRUST CHAMBER JACKET PURGE MUST BE ACTIVATED DURING ANY TIME INTERVAL SUBSEQUENT TO PROPELLANT LOADING THAT THE VEHICLE NITROGEN BOATTAIL PURGE IS NOT IN OPERATION.

FIGURE 4.0-22 ENGINE PURGE SEQUENCE FOR FLIGHT

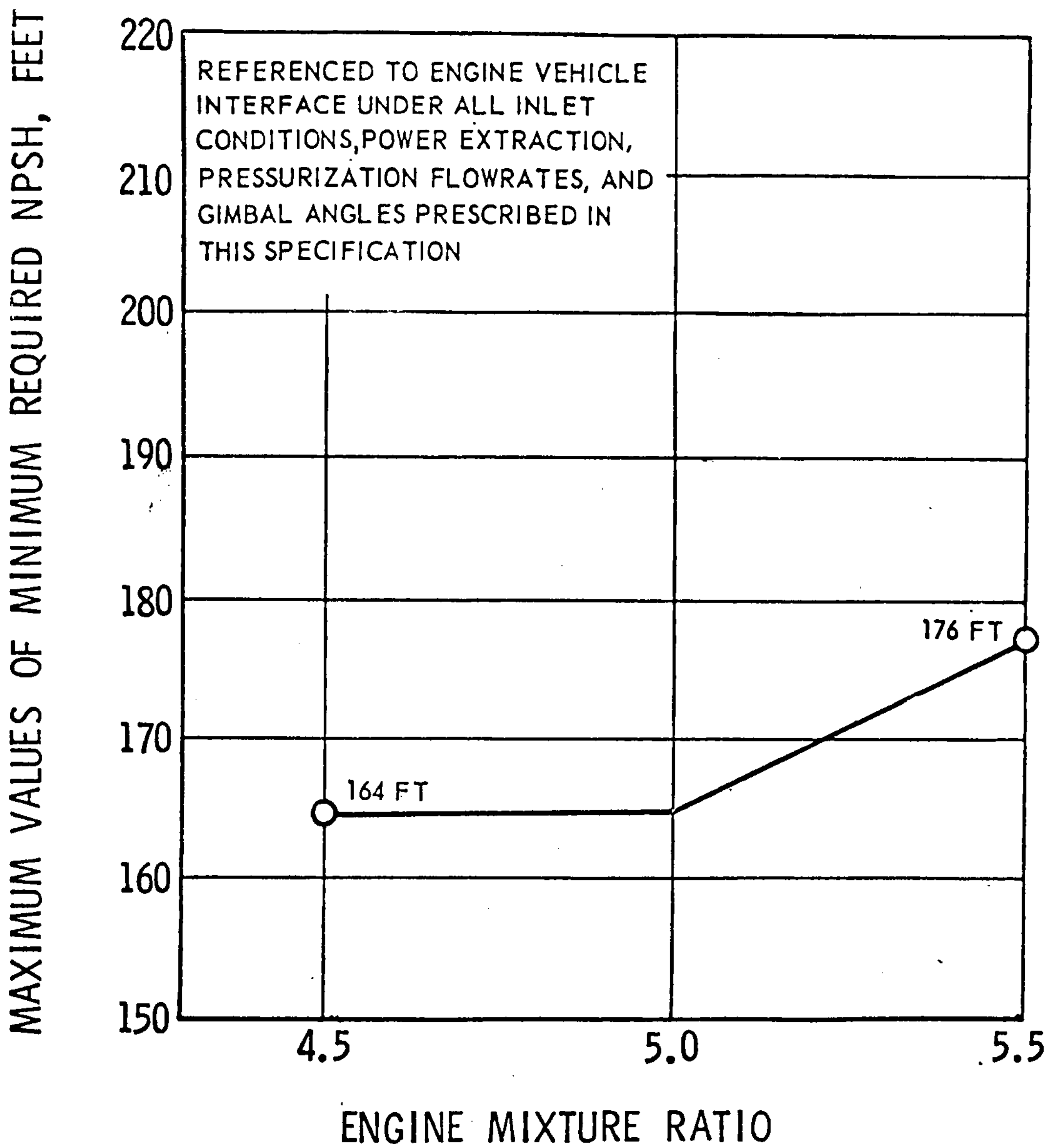


FIGURE 4.0-23

HYDROGEN ENGINE INLET NPSH REQUIREMENTS

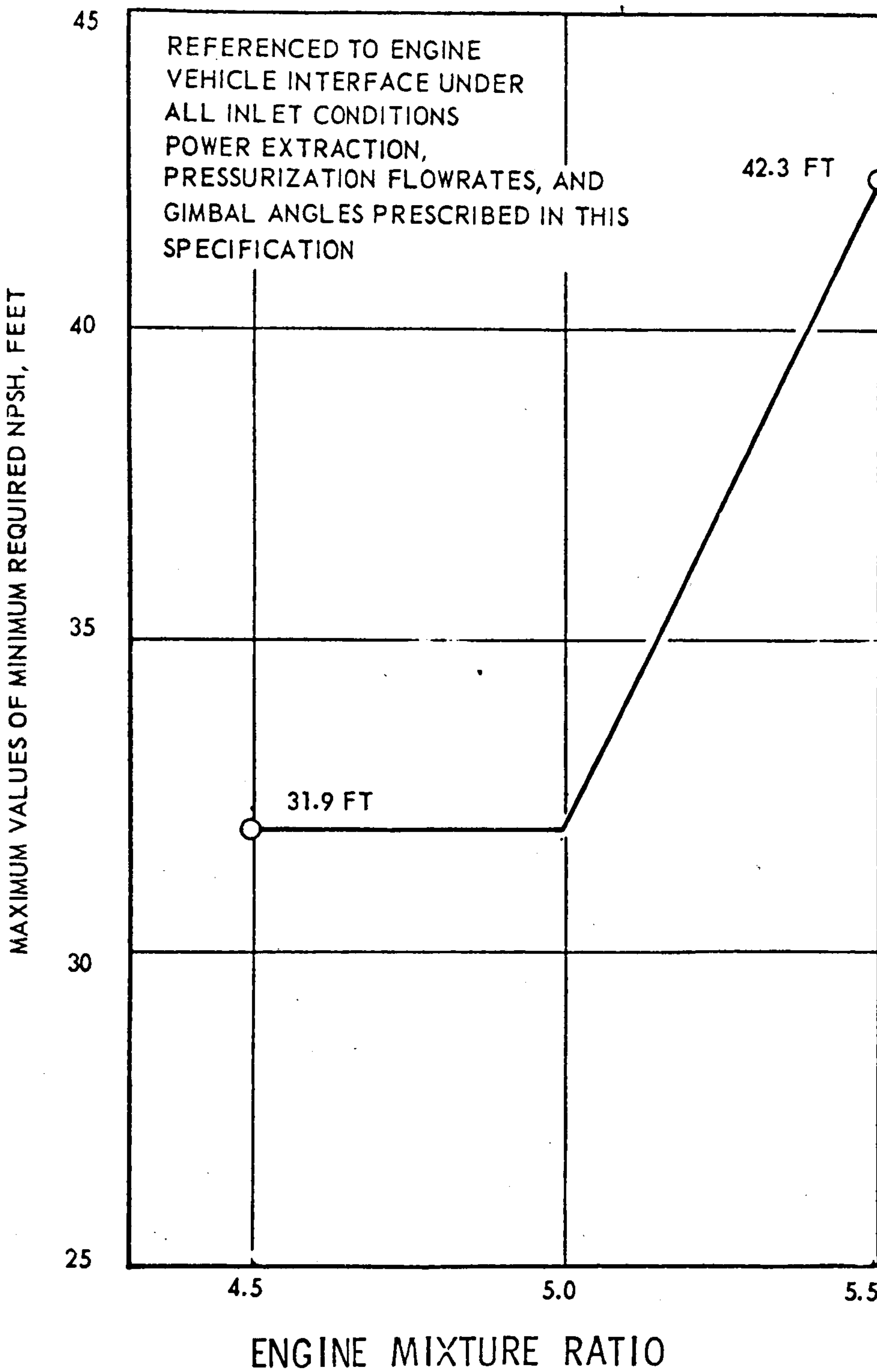


FIGURE 4.0-24

OXIDIZER ENGINE INLET NPSH REQUIREMENTS

ENGINE START REQUIREMENTS

The engine start requirements for static firing and for flight are shown in Fig. 4.0-25 and 4.0-26.

For first start, the oxidizer and fuel shall be supplied to the engine for a minimum of 1 hour and a maximum of 18 hours prior to engine start.

For engine restart to idle-mode restart, the oxidizer and fuel inlet temperatures must be colder than -100 F as shown in Fig. 4.0-26. Mainstage start shall be preceded by sufficient idle-mode duration to chill the propellant feed systems properly. The required duration will be 30 seconds minimum and 100 seconds maximum. Idle mode operation is described in Fig. 4.0-27.

PROPELLANT UTILIZATION SYSTEM

The propellant utilization system consists of a servocontrolled valve to bypass liquid oxygen from the oxidizer pump discharge. The valve is positioned by electrical inputs from a source external to the engine. The propellant utilization valve is capable of varying the engine mixture ratio from 4.5 to 5.5 with the characteristics shown in Fig. 4.0-28.

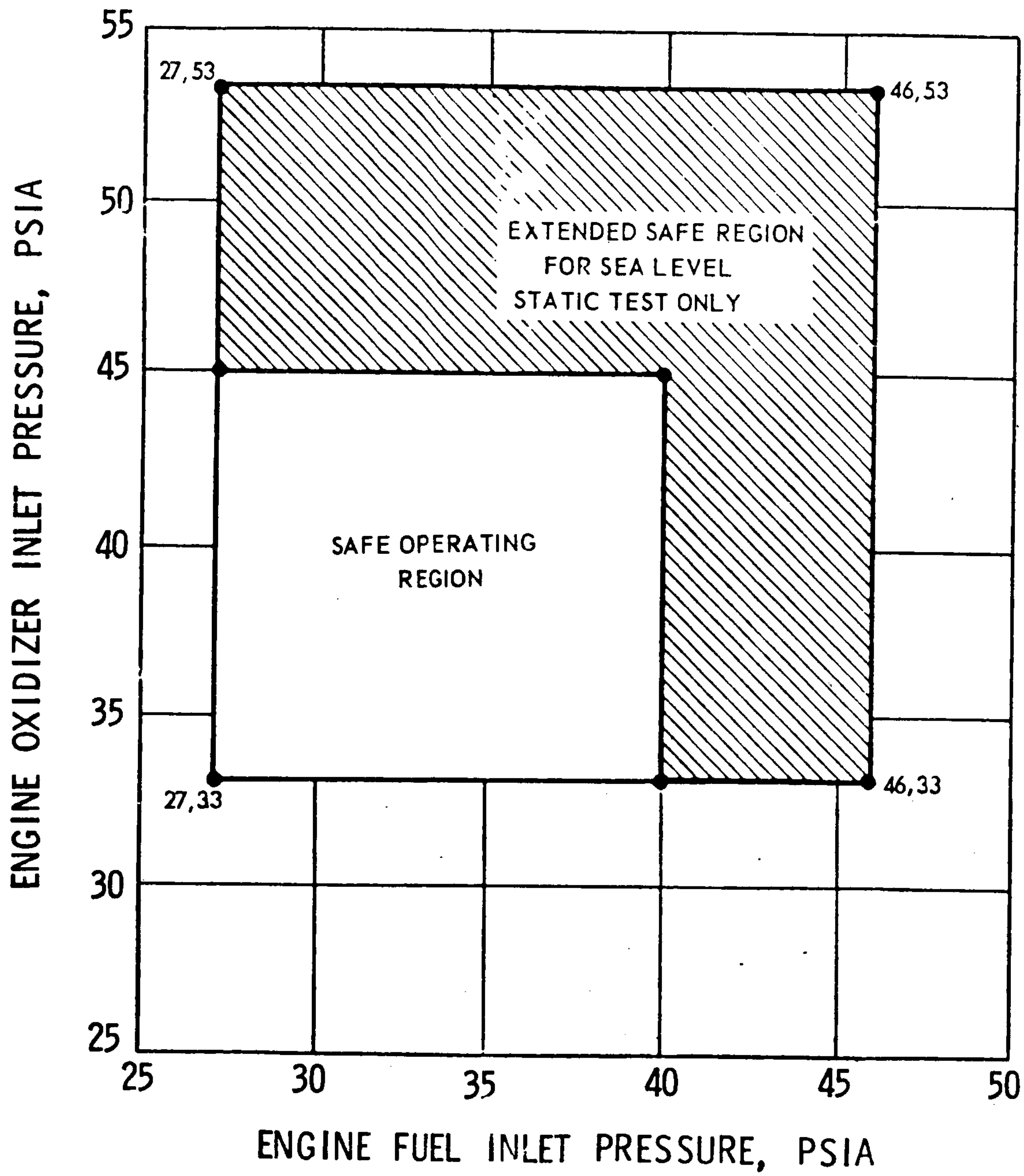
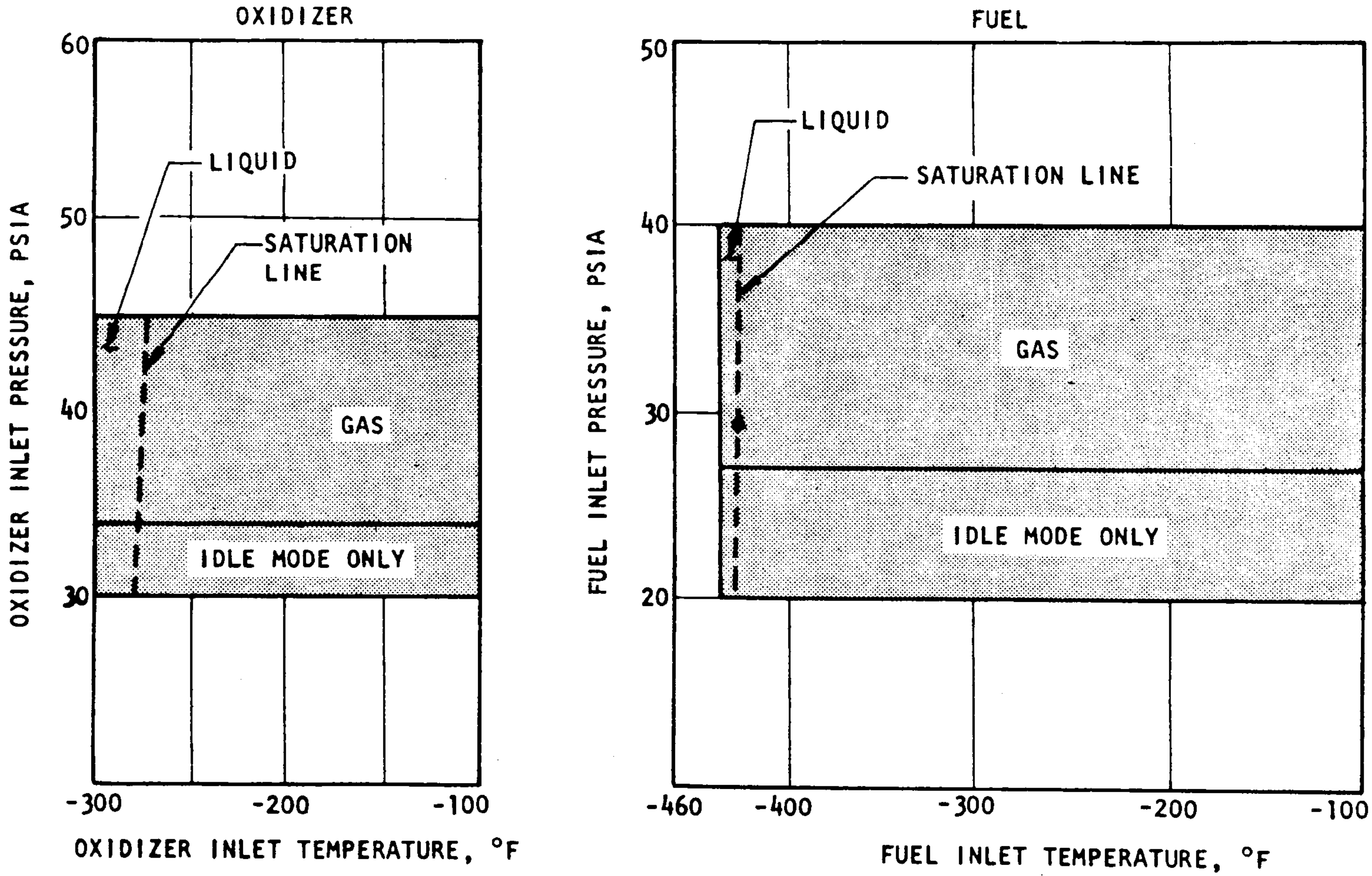
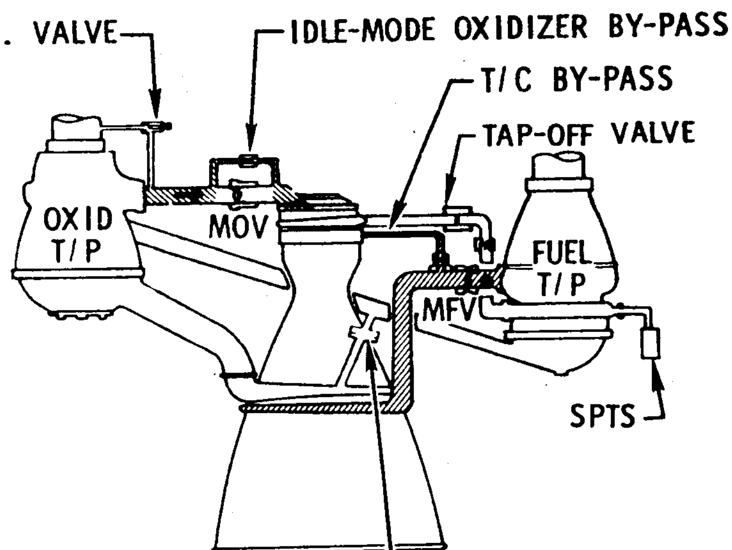


FIGURE 4.0-25 MAINSTAGE START REQUIREMENTS
ENGINE INLET PRESSURE LIMITS



NOTE: At idle mode start, liquid propellants are at tank outlets.

FIGURE 4.0-26 IDLE MODE PROPELLANT SUPPLY CONDITIONS/PRESTART REQUIREMENTS



DEFINITION

- IDLE MODE IS LOW-THRUST OPERATION IN WHICH THE ENGINE TURBOPUMPS ARE NOT IN OPERATION. PROPELLANTS ARE SUPPLIED TO THE ENGINE UNDER TANK PRESSURE. COMBUSTION IS SUPPORTED BY A PORTION OF THE MAIN COMBUSTOR INJECTOR. IDLE MODE CAN BE INITIATED WITH GASEOUS OR LIQUID PROPELLANTS.

IDLE-MODE FUNCTIONS

- TO SETTLE PROPELLANTS IN PREPARATION FOR FULL-THRUST OPERATION, ELIMINATING THE NEED FOR AN AUXILIARY PROPULSION SYSTEM.
- TO PRECONDITION (CHILL) THE ENGINE SYSTEM AND PROPELLANTS, ELIMINATING RECIRCULATION SYSTEMS AND THE ASSOCIATED PUMPS AND HARDWARE.
- LOW-THRUST OPERATION FOR LOW-IMPULSE MANEUVERS.
- POST MAINSTAGE OPERATION

FIGURE 4.0-27 J-2S ENGINE IDLE MODE OPERATION

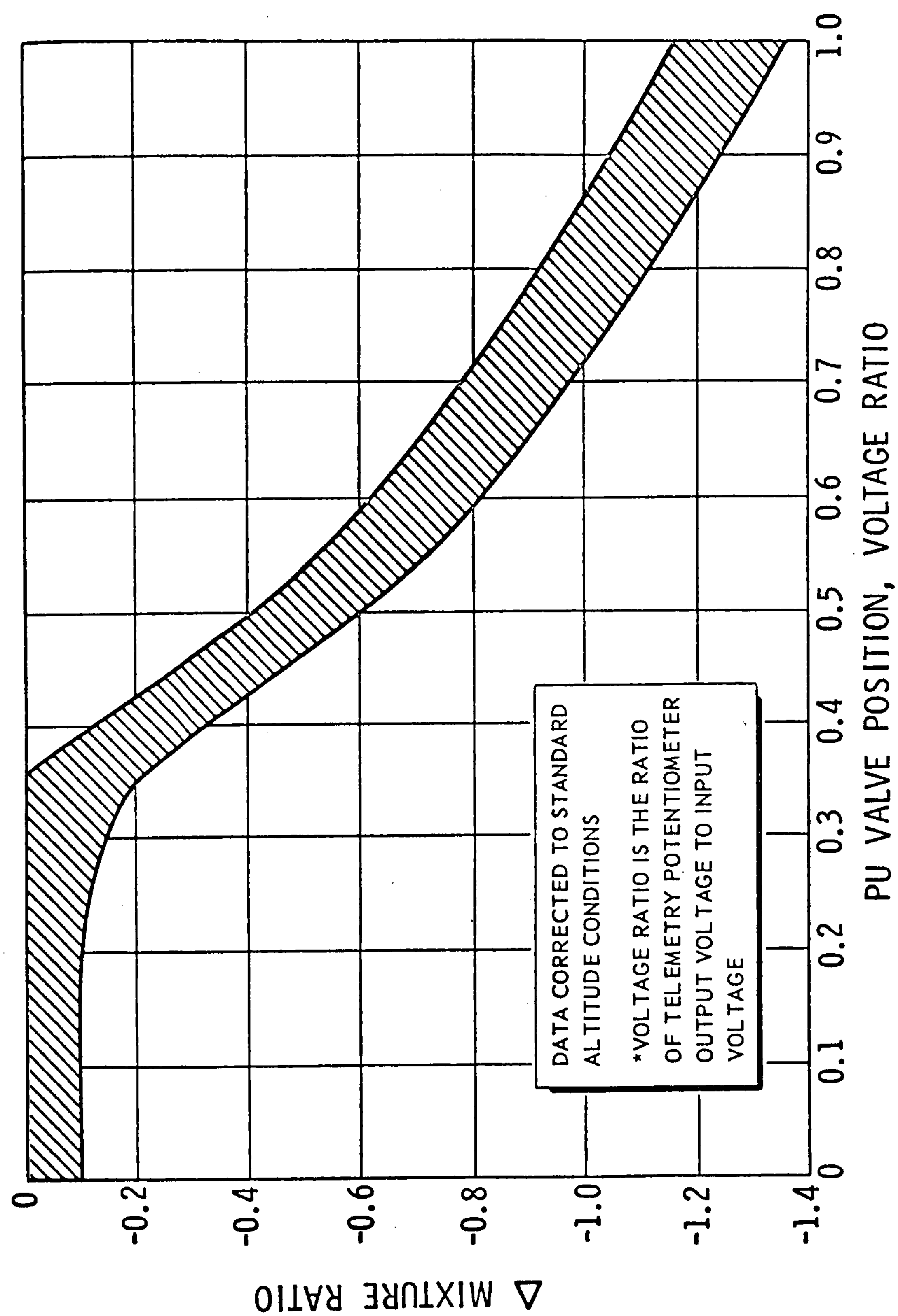


FIGURE 4.0-28 ESTIMATED PU VALVE POSITION VS DIFFERENTIAL ENGINE MIXTURE RATIO

SECTION 5

TRADE STUDIES FOR CONFIGURATION SELECTION

5.0 GENERAL

The trade studies investigated the J-2S thrust calibration range, J-2S engine mixture ratio schedules, J-2S/stage operational sequence requirements, and vehicle/payload configurations to determine the basic vehicle/stage system requirements and structural load criteria.

Missions other than the basic lunar design mission (LOR) were considered. The other mission trades determined vehicle configurations which would optimize the usefulness of the J-2S engine without major structural or systems changes to the existing Saturn V stages or support operations.

The data was used to select the four design vehicle configurations for detail study in Phase II.

5.1 J-2S THRUST TRADES

S-II and S-IVB stage thrust trades were conducted using a LOR flight profile. The primary objective of these trades was to determine the preferred calibration range. At the calibration mixture ratio of 5.5, the J-2S engine is capable of operating between a maximum thrust level of 265,000 pounds and a minimum thrust level of 230,000 pounds. The minimum and maximum thrust calibration range as a function of mixture ratio is shown in Figure 5.1-1. Specific impulse (I_{sp}) is a function of mixture ratio but is independent on the calibrated thrust level.(See Fig. 5.1-2).

For the S-IVB thrust trades, a constant mixture ratio of 5.0 was selected to determine the thrust to payload relationship. Payload weights were determined for several thrust levels between the maximum and minimum calibrated thrust levels (from 237,500 lbs. to 206,500 lbs) at the 5.0 mixture ratio and I_{sp} of 434.5 seconds. The payload gain due to the 31,000 pounds of additional thrust is approximately 90 pounds. The S-IVB, as a LOR mission third stage, is, therefore, not thrust sensitive; and it might be presumed that the lower J-2S calibration thrust level should be selected to avoid impact to the current S-IVB thrust structure. However, other mission applications of the S-IVB should be considered. If a higher thrust is desirable, an assessment of thrust structure impact should be made.

When the 265,000 pound thrust J-2S engine is used on the S-IVB, there is increased flexibility which results in a more useful boost stage. The S-IVB is thrust sensitive when used on a two-stage vehicle. For the low Earth orbit mission, the payload for the INT-20 vehicle (4 F-1 S-IC/S-IVB) is increased 3,200 pounds by increasing the S-IVB thrust 31,000 pounds. It is probable that other launch vehicles, such as the Saturn IB or 260-inch SRM/S-IVB, would also show a payload improvement at higher S-IVB thrust levels. The McDonnell Douglas Corporation made a rough determination of S-IVB thrust

5.1 (Continued)

structure changes required to uprate the structural capability for 265,000 pounds (a thrust increase of 35,000 lbs). The modifications result in a thrust structure weight increase of approximately 75 pounds. Preliminary estimates show that there would be no increase in recurring costs but $56,500 would be required for development to modify the S-IVB thrust structure. This minimal weight and cost increase to uprate the S-IVB thrust structure for the 265K J-2S engine seems acceptable in light of other mission applications.

A LOR payload increase of approximately 4,300 pounds results when the S-II engine thrust is varied between the calibrated thrust levels. This data substantiates conclusions reached from previous studies that maximum thrust is desired in the S-II stage. A 265K calibrated J-2S engine should be used in the S-II.

It was concluded that both the S-IVB and S-II stages should be designed for the 265 K J-2S engine(s).

5.2 J-2S MIXTURE RATIO TRADES

The S-II and S-IVB mixture ratio trades were conducted using a LOR flight profile with 265K thrust J-2S engines in both stages. The primary objective was to determine the optimum mixture ratio (MR) schedule of both stages. The trajectory optimization was based on the standard Saturn V LOR mission flight profile for a 72° launch azimuth and used the same ground rules and assumptions. In all cases, maximum utilization of S-II and S-IVB propellant was achieved. Propellant required for a velocity increment of 109 meters per second was reserved in the S-IVB stage.

The J-2S thrust and specific impulse as a function of MR were obtained from the Rocketdyne J-2S Interface Criteria Document, R-7211, and are shown in Figures 5.1-1 and 5.1-2. In all cases, the J-2S engine required a 5.0 MR burn for 2.5 seconds during the start sequence.

Initially, the S-II and S-IVB stages were analyzed independently to determine the influence of MR in each stage. While one stage was being analyzed, the other was held constant at the LOR MR schedule (S-II MR schedule of 5.0 to 5.5 to 4.7, and S-IV MR constant at 5.0). Finally, the S-II and S-IVB were analyzed simultaneously to determine the influence of any stage interrelationships on MR operating levels.

5.2.1 Independent S-II Stage MR Analysis

During the S-II MR analysis, the S-IVB stage MR was held constant at the Saturn V nominal LOR operating level of 5.0. The S-II was operated for 2.5 seconds at a MR of 5.0 during ignition sequence to satisfy J-2S start requirements.

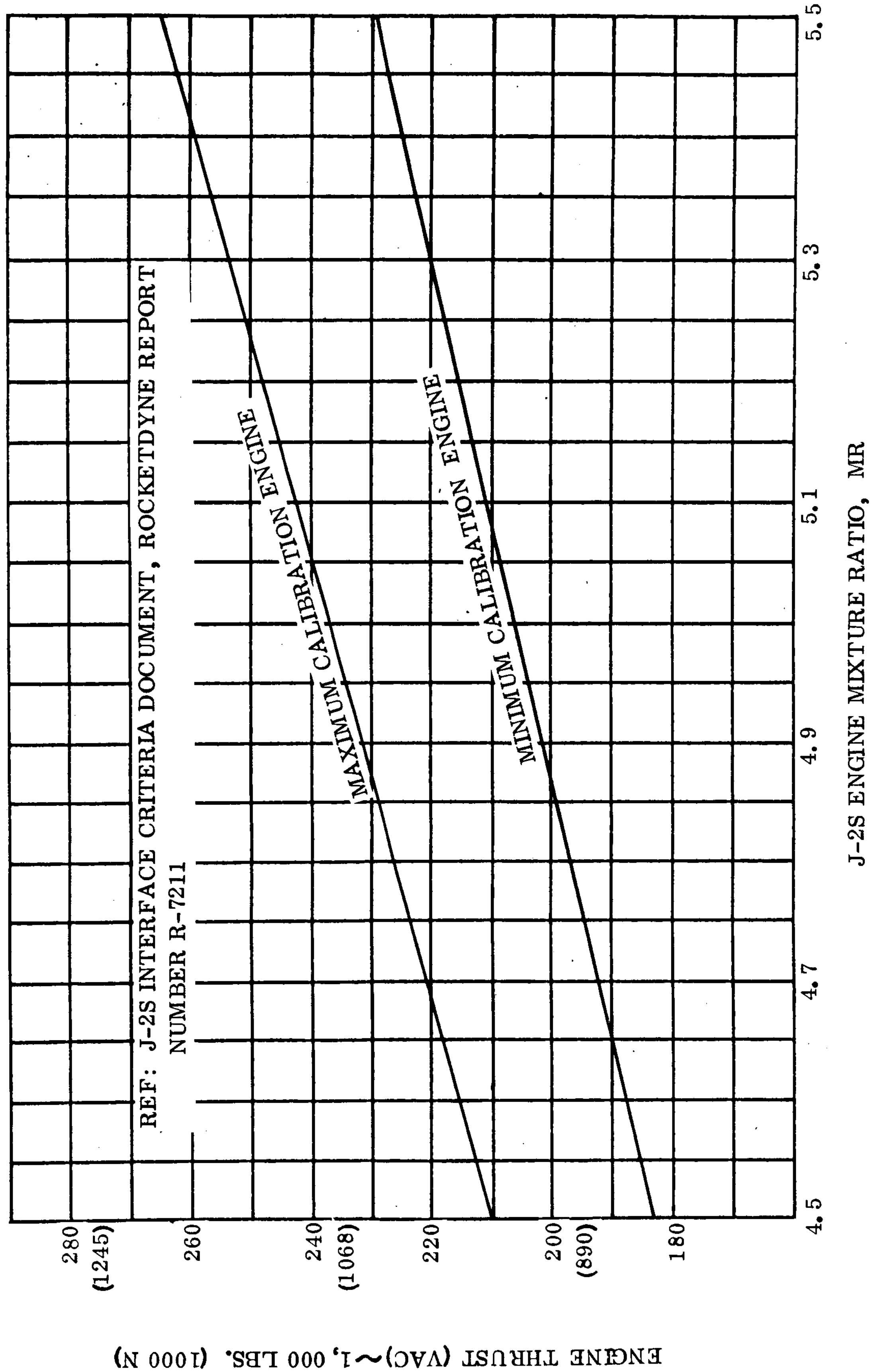


FIGURE 5.1-1 J-2S SINGLE ENGINE THRUST VERSUS MIXTURE RATIO

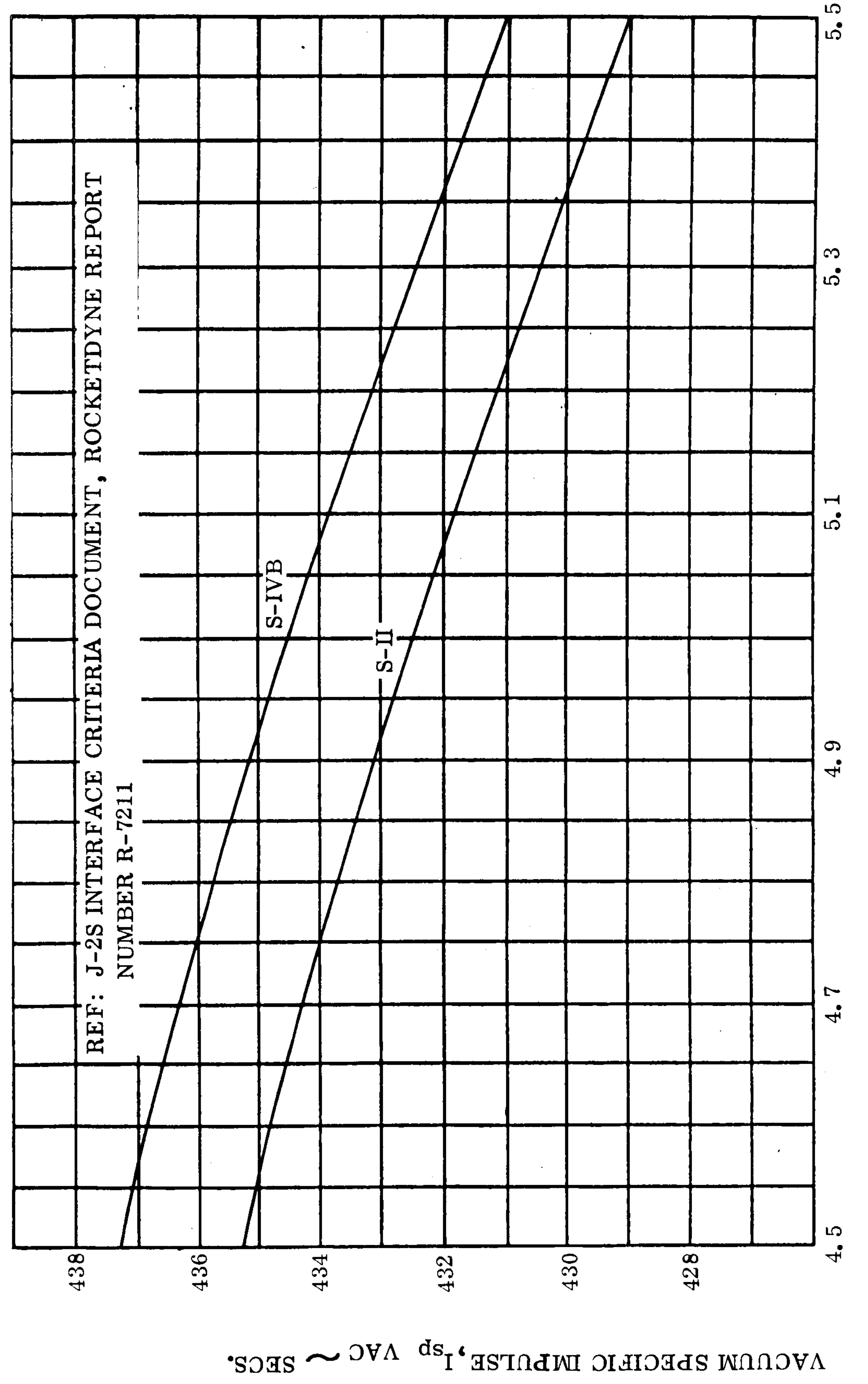


FIGURE 5.1-2 J-2S ENGINE SPECIFIC IMPULSE SIMULATED FOR THE S-II AND S-IVB STAGES

5.2.1.1 S-II MR Schedule

a. Shift times for the LOR S-II MR schedule of 5.0 to 5.5 to 4.7 were optimized. Duration of burn at 5.0 MR was fixed at 2.5 seconds but the burn duration at the 5.5 MR was varied (and, therefore, also the burn time at the 4.7 MR). Since a propellant depletion cutoff was used, only one burn duration at the 5.5 MR results in maximum S-II impulse propellant. Propellant must be off-loaded at all other burn times. Figure 5.2.1-1 presents the relationship between the change in total S-II impulse propellant and the burn duration at 5.5 MR. The relationship between the change in LOR net payload and the burn duration at 5.5 MR is presented in Figure 5.2.1-2. The payload optimized at the maximum impulse propellant loading. This is probably true for all S-II MR schedules similar to the LOR schedule; that is, high second MR, lower third MR.

b. Alternate MR schedule wi th high thrust second MRs and low thrust but higher specific impulse third MRs were examined. Again, the S-II began its burn at 5.0 MR for 2.5 seconds. Second MR values of 5.5 and 5.4 were used and, third MR values ranged from 4.5 to 5.3. The relationship between the change in LOR net payload, the second MR values, and the third MR values is presented in Figure 5.2.1-3. In each case, the shift time was determined such that the maximum S-II impulse propellant was consumed. The optimum S-II MR schedule is seen in Figure 5.2.1-3 to be 5.0 to 5.5 to 4.7.

5.2.1.2 S-II Constant MR

Operation at S-II constant MR operation levels was considered. Any MR other than the impulse propellant tank MR (TMR_I) of 5.26 will require off-loading either LOX or LH_2 propellant. The amount of propellant off-loading required as a function of constant S-II mixture ratio is presented in Figure 5.2.1-4. The relationship between the change in LOR net payload and constant S-II MR is presented in 5.2.1-5. No payload increase over that obtained using the LOR MR schedule was obtained at any constant S-II MR. However, a high MR (maximum thrust) is best when using constant MR even though LH_2 must be off-loaded.

5.2.1.3 S-II MR Analysis Conclusions

The optimum S-II MR operational level, when analyzed independently of the S-IVB, is a MR schedule of 5.0 to 5.5 to 4.7 with the shift time from 5.5 to 4.7 selected to maximize S-II impulse propellant.

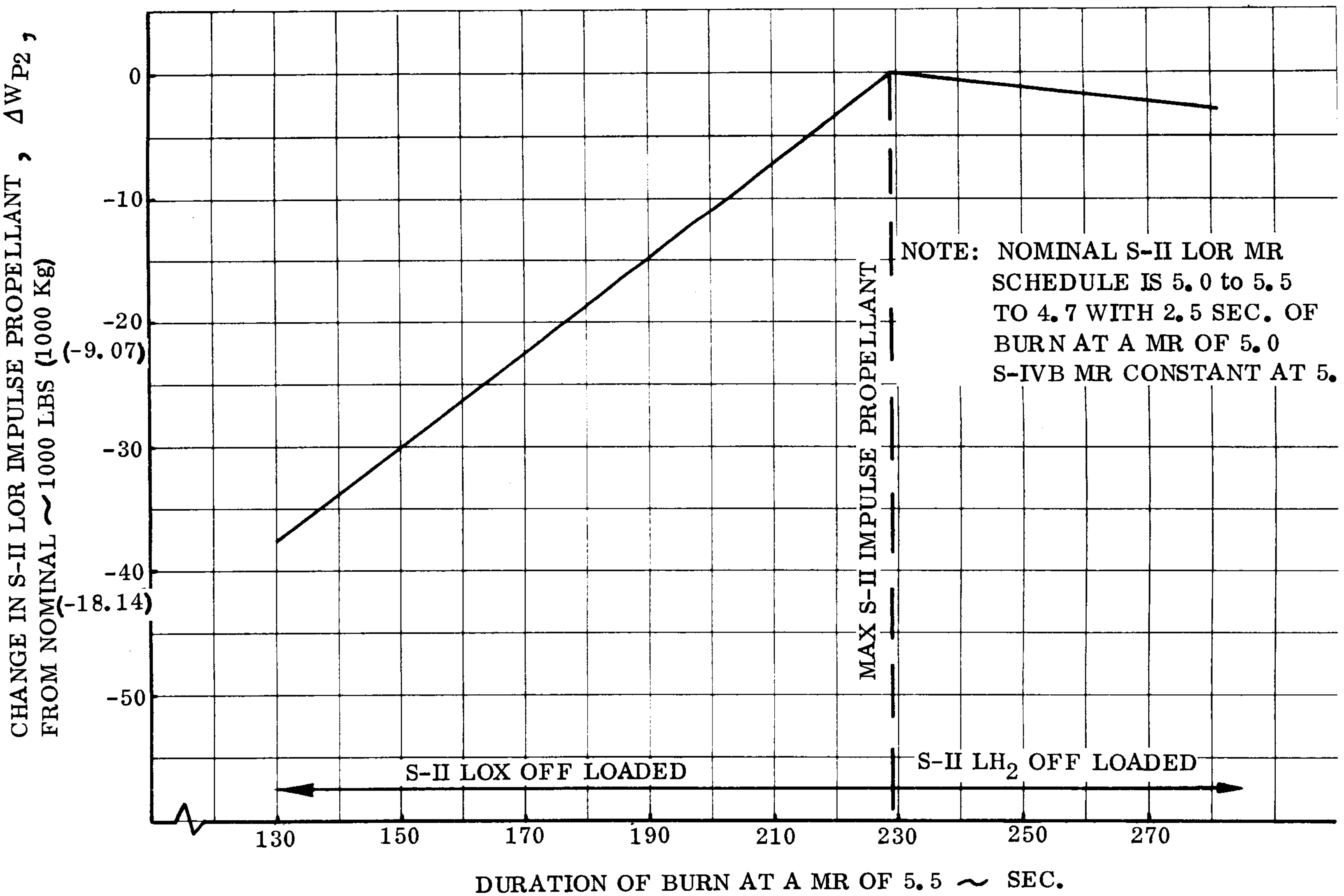


FIGURE 5.2.1-1 S-II SHIFT TIME STUDY - CHANGE IN S-II IMPULSE PROPELLANT WITH BURN DURATION AT A MR OF 5.5

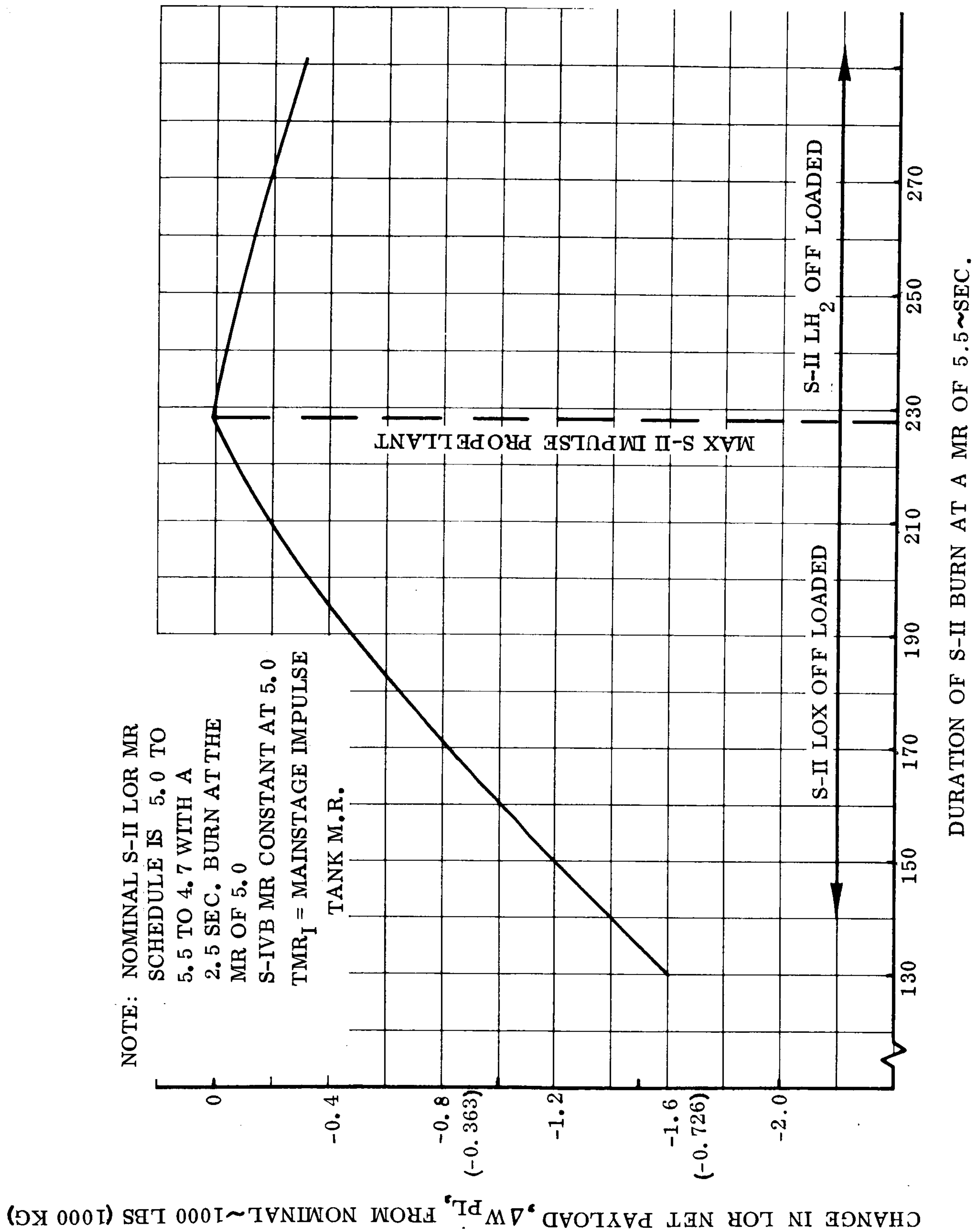


FIGURE 5.2.1-2 S-II SHIFT TIME STUDY - CHANGE IN LOR NET PAYLOAD, ΔW_{PL}, FROM NOMINAL WITH BURN DURATION AT A MR OF 5.5

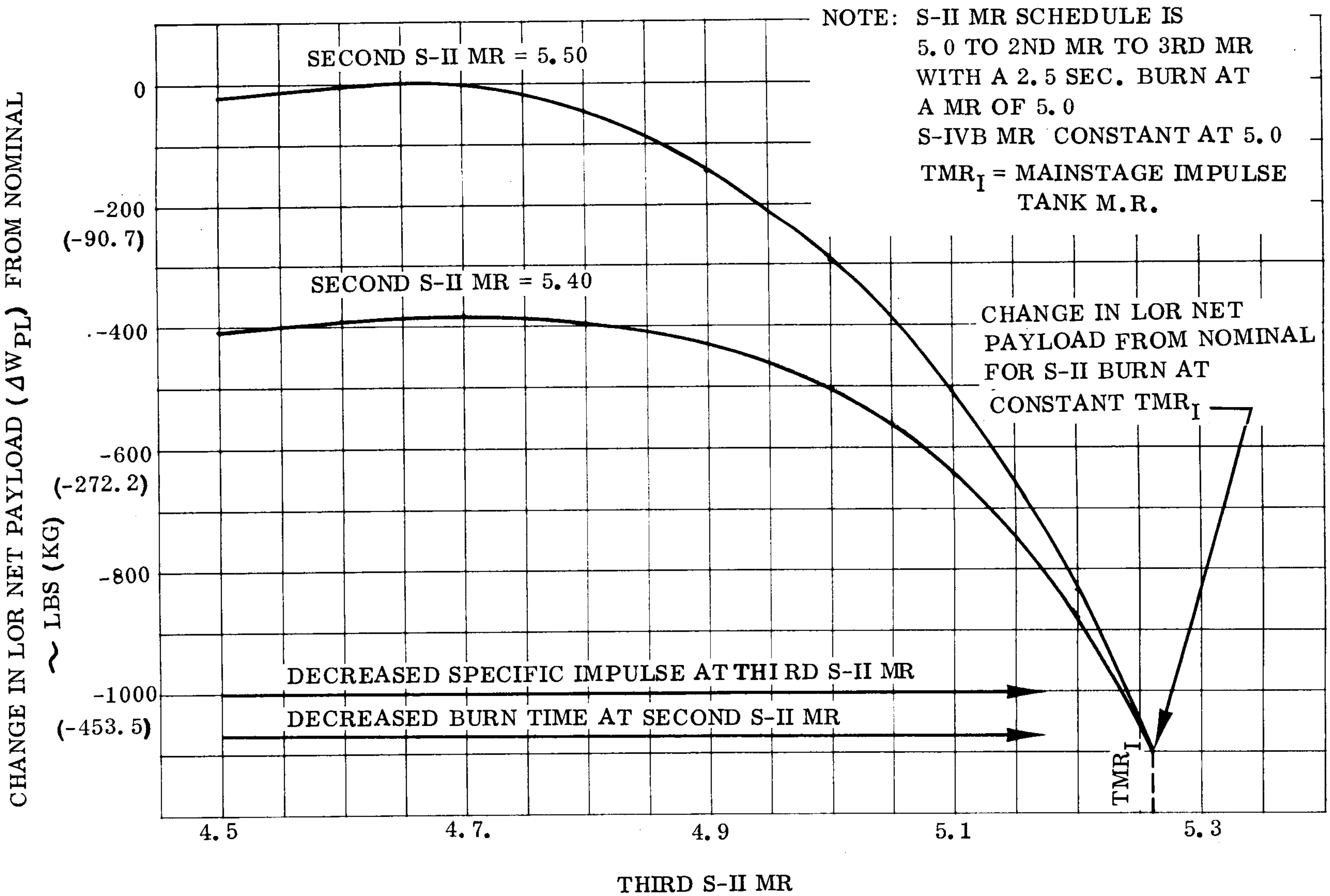


FIGURE 5.2.1-3 S-II MR SCHEDULE STUDY - CHANGE IN LOR NET PAYLOAD FROM NOMINAL (ΔW_{PL}) VS. SPECIFIED S-II THIRD BURN MR'S (WITH S-II SECOND BURN MR HELD CONSTANT AS A PARAMETER)

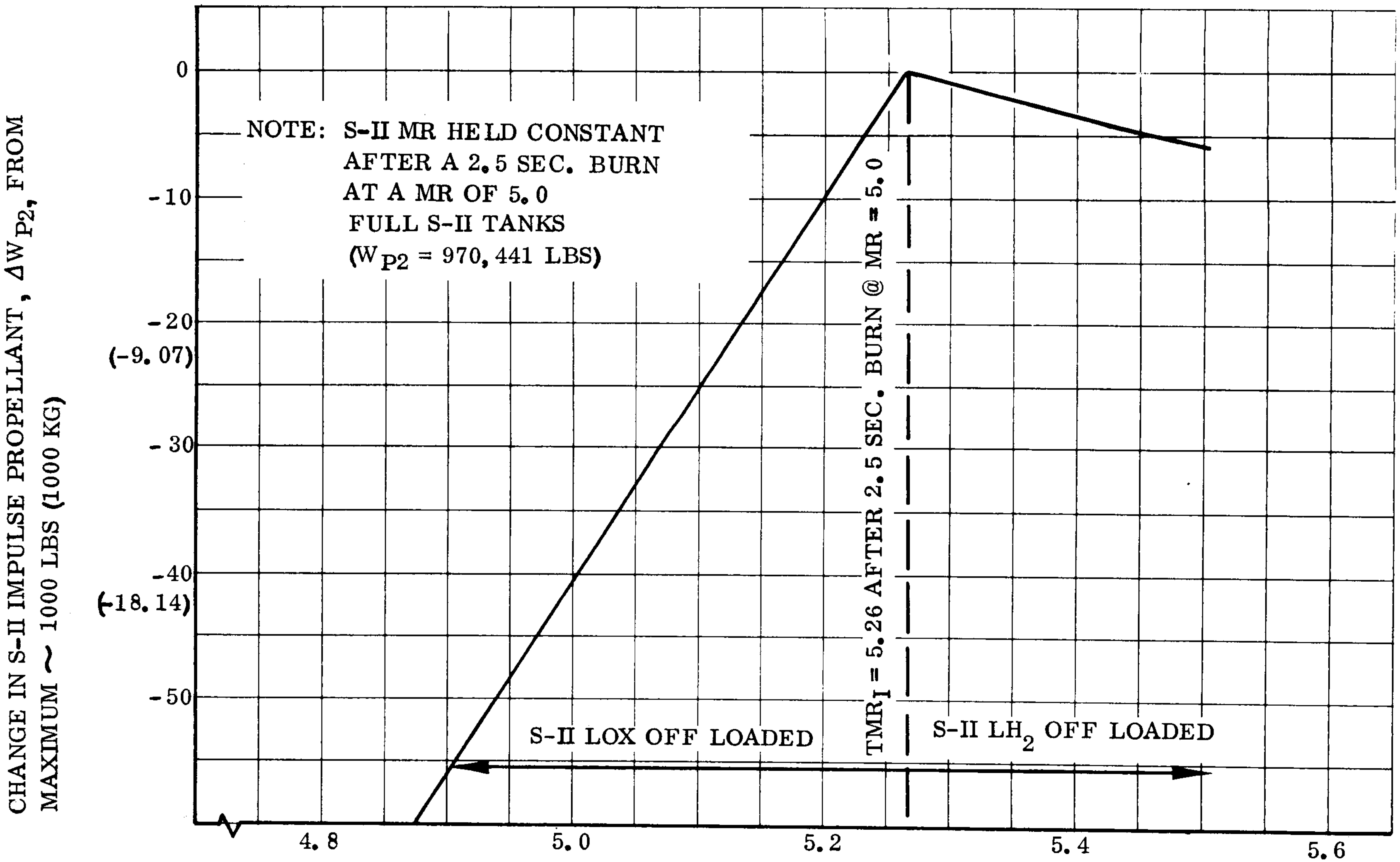


FIGURE 5.2.1-4 S-II CONSTANT MR STUDY - CHANGE FROM MAXIMUM S-II IMPULSE PROPELLANT FOR SPECIFIED CONSTANT S-II MIXTURE RATIOS

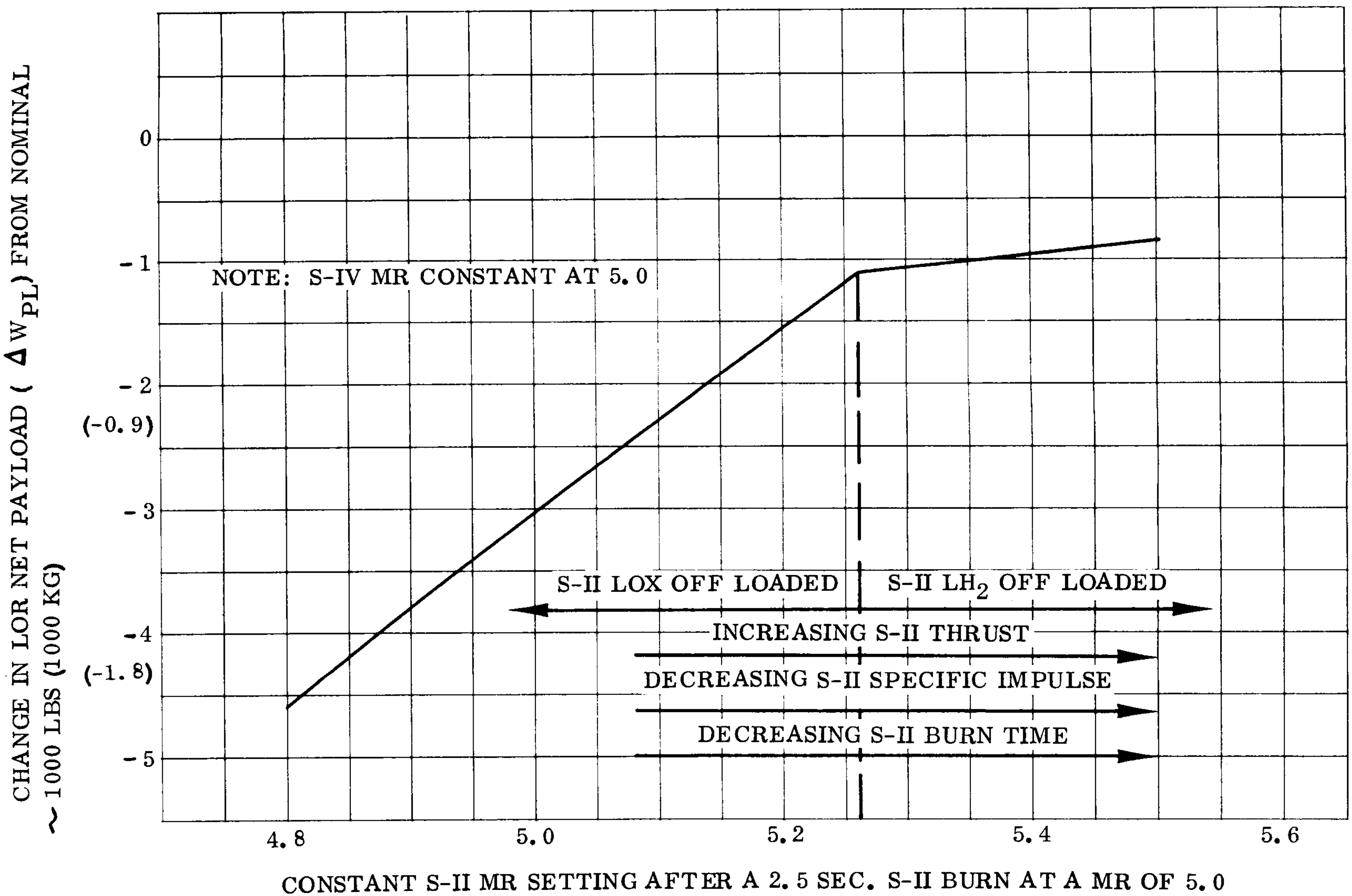


FIGURE 5.2.1-5 S-II CONSTANT MR STUDY - CHANGE IN LOR NET PAYLOAD (ΔW_{PL}) FROM NOMINAL VS. SPECIFIED CONSTANT S-II MIXTURE RATIOS (AFTER A 2.5 SEC. S-II BURN AT A MR OF 5.0)

5.2.2 Independentent S-IVB Stage MR Analysis

During the S-IVB MR analysis, the S-II stage used the Saturn V LOR MR schedule of 5.0 to 5.5 to 4.7. In the simulation, the S-IVB was operated at a constant MR or with a single shift MR schedule.

5.2.2.1 S-IVB Constant MR

When the S-IVB burn is at constant MR, propellant off-loading is required for all MRs except the impulse tank mixture ratio (TMR_I) of 4.79. The relationship between the change in S-IVB impulse propellant and MR is presented in Figure 5.2.2-1. Figure 5.2.2-2 shows the relationship between the change in LOR net payload and S-IVB constant MR. The payload optimized at the MR of 4.79, the maximum impulse propellant loading. The maximum net LOR payload increase was obtained by analyzing the S-II and S-IVB MR schedules simultaneously as reported in Section 5.2.3.

5.2.2.2 S-IVB MR Schedule

S-IVB MR schedules were considered as a means to increase payload.

a. Shift time, in all MR schedules analyzed, was selected to maximize S-IVB impulse propellant. The payload relationship between the various MR schedules analyzed is presented in Figure 5.2.2-3. No gain in payload results from any MR schedule other than the constant 4.79 MR payload value. Mixture ratio schedules which shift down from a high MR to a lower MR produce more payload than when shifting up between the same two MRs.

b. Shift time other than that required for maximum impulse propellant was considered to determine the effect of S-IVB burn time. The MR schedule considered was 5.5 to 4.5. Propellant offloading is required, as shown in Figure 5.2.2-4, when the S-IVB burn duration at 5.5 MR is different from the time at which impulse propellant is maximized (full tanks).

   The relationship between change in LOR net payload and burn duration at the 5.5 MR is presented in Figure 5.2.2-5. No payload gain results from off-loading the S-IVB to vary the burn time. This trend would be expected to hold for any other high to low MR schedules.

5.2.2.3 S-IVB MR Analysis Conclusions

The optimum MR operation level for the S-IVB stage, when analyzed independently of the S-II, is the constant impulse propellant tank MR (TMR_I) of 4.79.

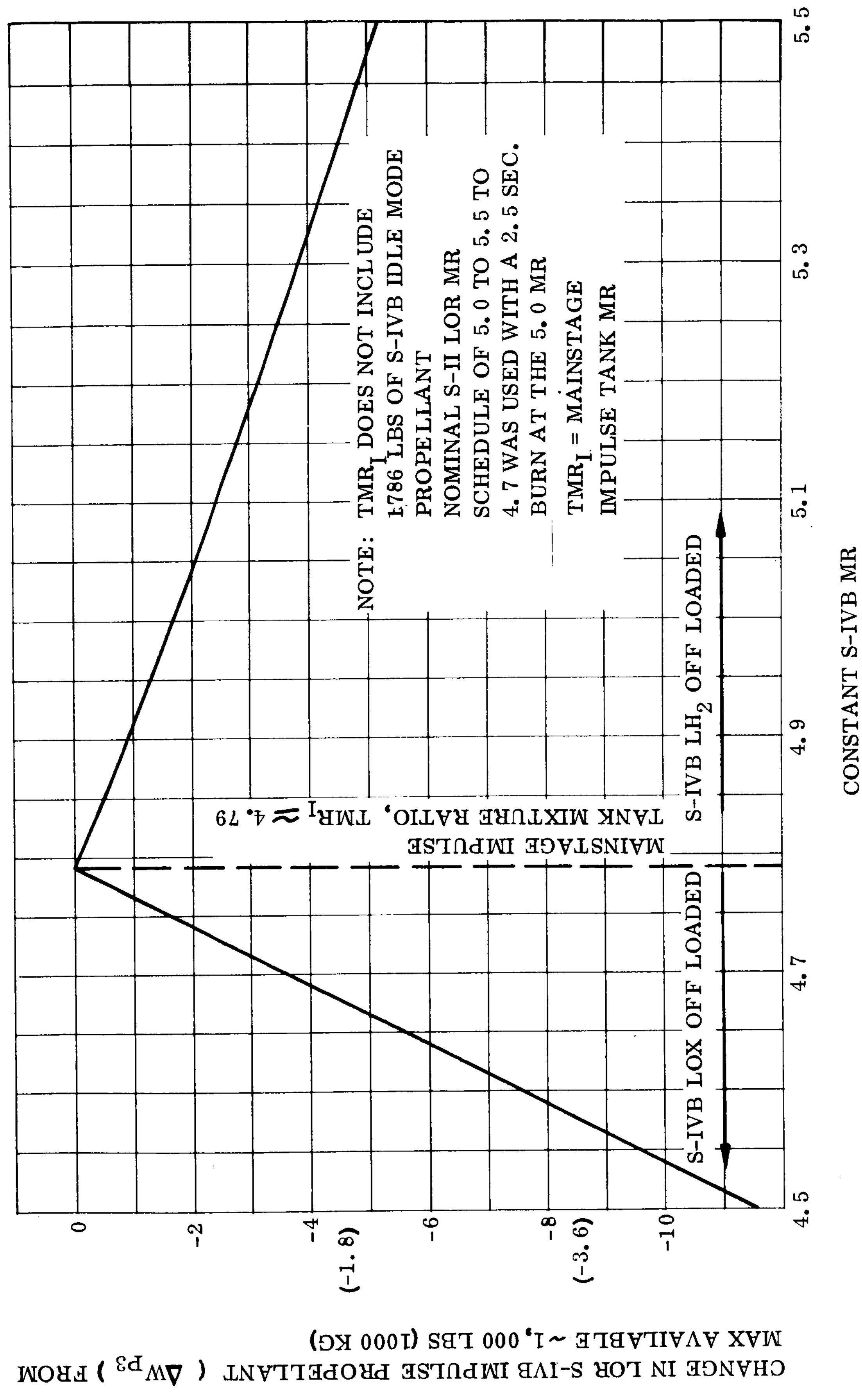


FIGURE 5.2.2-1 S-IVB CONSTANT MR STUDY - CHANGE IN S-IVB IMPULSE PROPELLANT (ΔW_{P3}) FROM MAXIMUM AVAILABLE VS. CONSTANT S-IVB MR

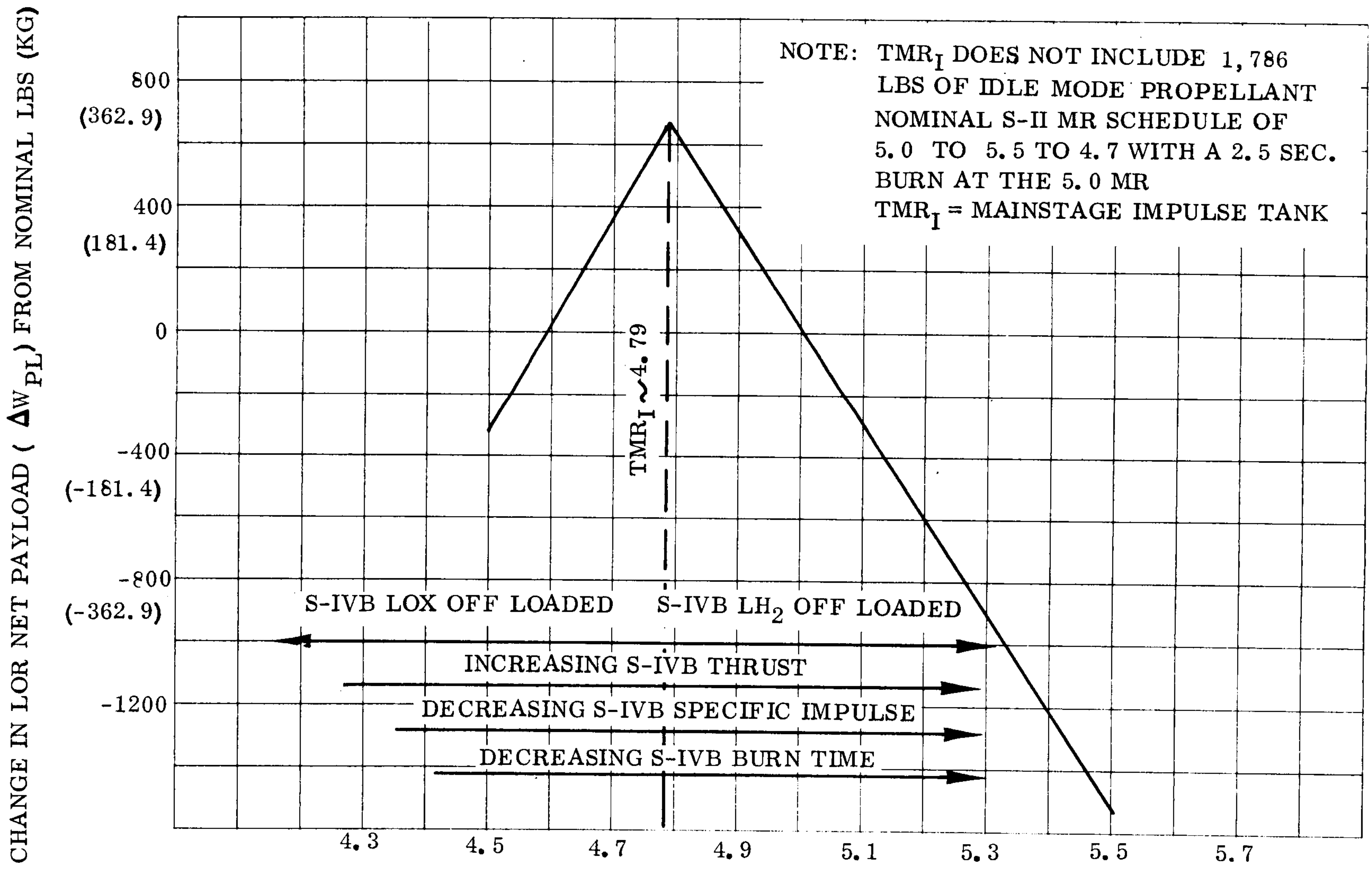


FIGURE 5.2.2-2 S-IVB CONSTANT MR STUDY - CHANGE IN LOR NET PAYLOAD (ΔW_{PL}) FROM NOMINAL VS. CONSTANT S-IVB MR

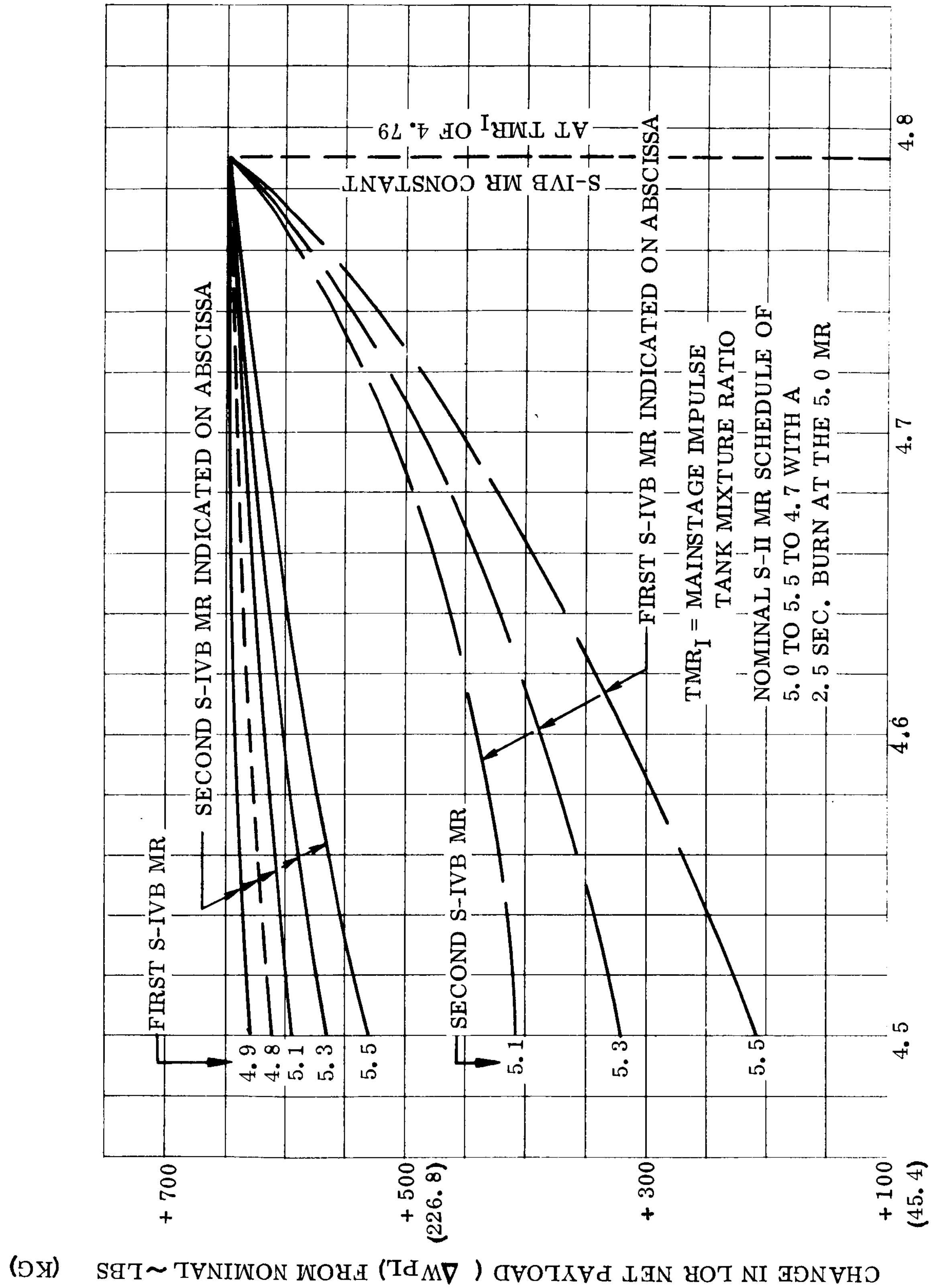


FIGURE 5.2.2-3 S-IVB MR SCHEDULE STUDY - CHANGE IN NET LOR PAYLOAD (ΔW_{PL}) FROM NOMINAL VS. S-IVB MR SCHEDULE

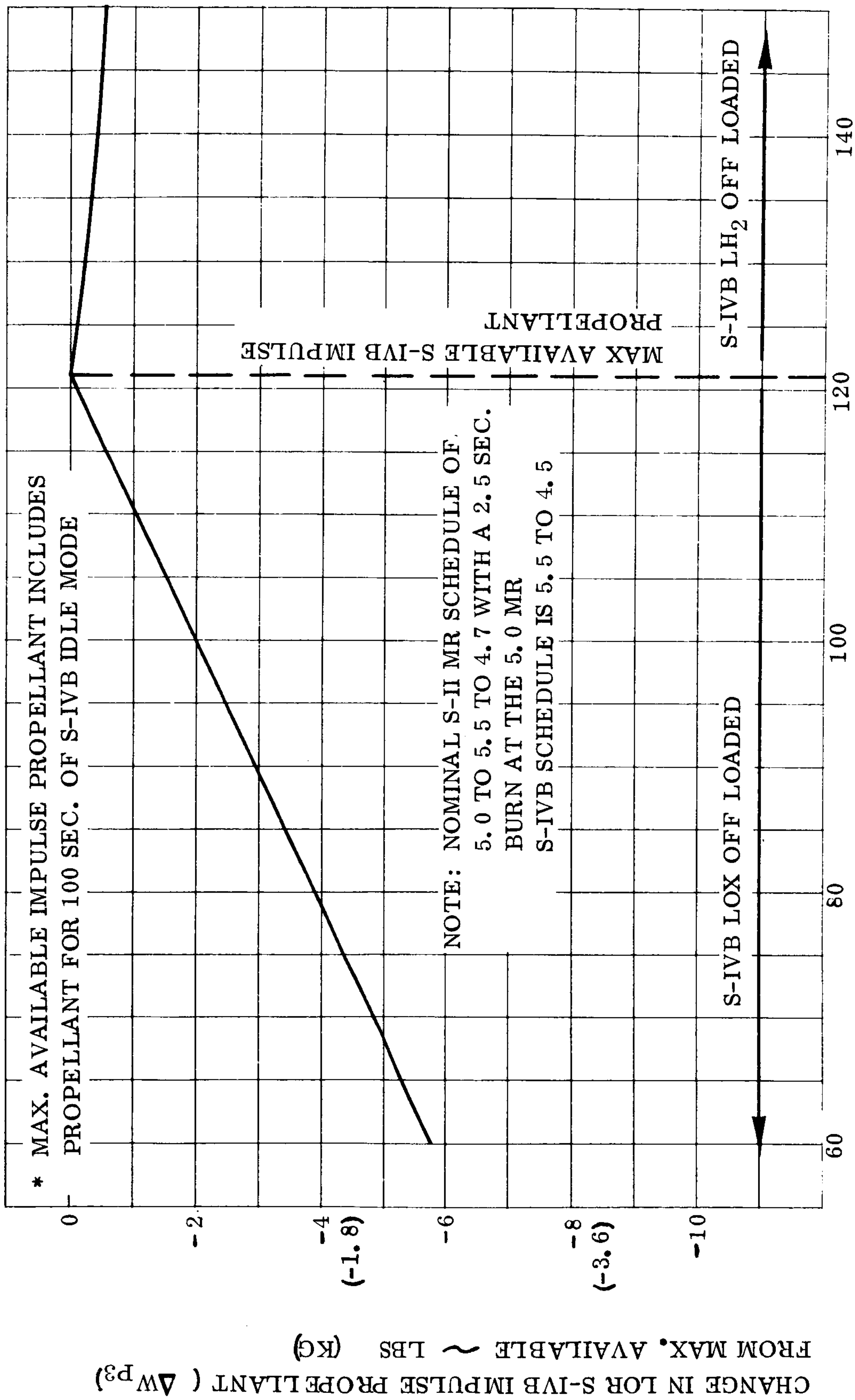


FIGURE 5.2.2-4 S-IVB SHIFT TIME STUDY - CHANGE IN S-IVB LOR IMPULSE PROPELLANT (ΔWP_3) FROM MAX AVAILABLE VS. DURATION OF BURN AT THE 5.5 S-IVB MR

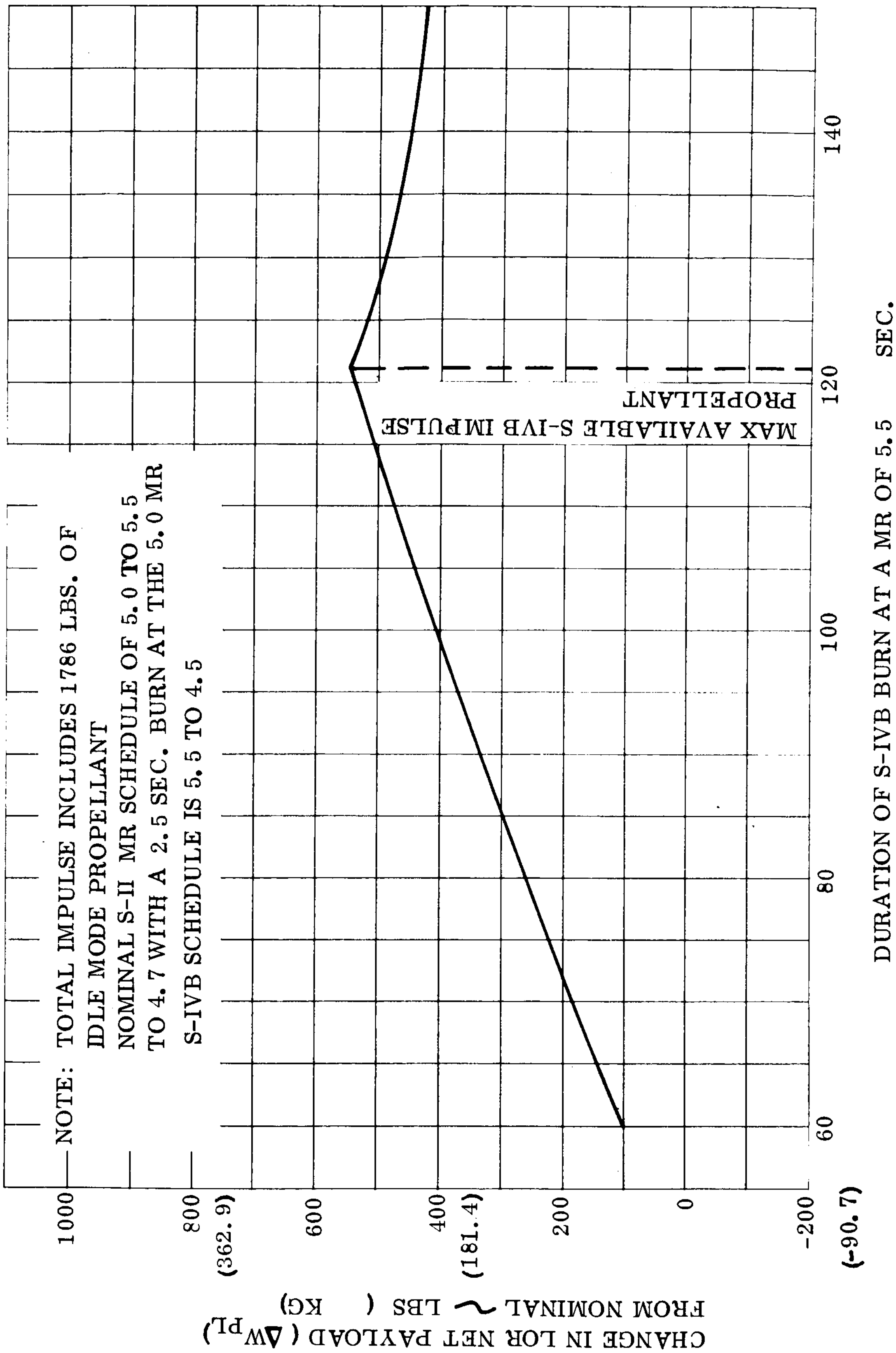


FIGURE 5.2.2-5 S-IVB SHIFT TIME STUDY - CHANGE IN LOR NET PAYLOAD (ΔW_{PL}) FROM NOMINAL VERSUS DURATION OF BURN AT THE S-IVB MR of 5.5

5.2.3 Simultaneous S-II Stage and S-IVB Stage MR Analysis

There is an interrelationship between the S-II MR schedule and the S-IVB constant MR. By analyzing the S-II and S-IVB simultaneously, the optimum MR operating levels were found to be a 5.0 to 5.5 to 4.5 MR schedule for the S-II and a constant 4.79 MR for the S-IVB.

5.3 MISSIONS, FLIGHT PROFILES AND PAYLOAD ENVELOPES TRADES

The basic purpose of these trade studies was to select the basic vehicle/payload configurations and mission flight profiles which would be studied in detail in Phase II of the study. The trade study examined significant mission flight profiles to determine the expected vehicle performance and engine/stage operational sequence requirements, i.e., start, restart, idle mode duration, mainstage duration, coast duration, etc. To establish the basic vehicle load criteria, predicted payload weights and shapes and mission requirements were examined.

5.3.1 Mission Definitions

Predictions of future mission applications were necessary since the J-2S engine/stage operation requirements are dependent on the mission flight profile. The post-Apollo missions for both two and three stage Saturn V launch vehicles during the ten-year period of interest (1972-1981) fall into three major categories: lunar, Earth orbital and planetary.

5.3.1.1 Lunar Missions

Lunar mission plans for the Apollo Program follow-on may include extended astronaut staytimes on the lunar surface. Logistics support will require an increased Apollo spacecraft delivery capacity. Direct unmanned lunar cargo deliveries may be required to support the extended manned exploration. Although the manned delivery systems may retain the Apollo shape, unmanned cargo carrying payloads will probably be a 260-inch cylindrical shape with a nose cap. The nose cap would be of the type designated by MSFC as the modified launch vehicle (MLV) nose cap and is shown in Figure 5.3-1. Future lunar missions require not only increased payload capability but also any structural modifications caused by the alternate payload shape.

5.3.1.2 Earth Orbital Missions

Potential earth orbital missions following the initial post-Apollo spent (or "wet") S-IVB workshop may be divided into two classes: (1) low-Earth orbit and polar orbit missions with orbit altitudes ranging between 100 and 300 N M, and (2) synchronous orbit missions with an orbit altitude of 19,323 N M and inclinations ranging between 0^{o} and 55^{o}.

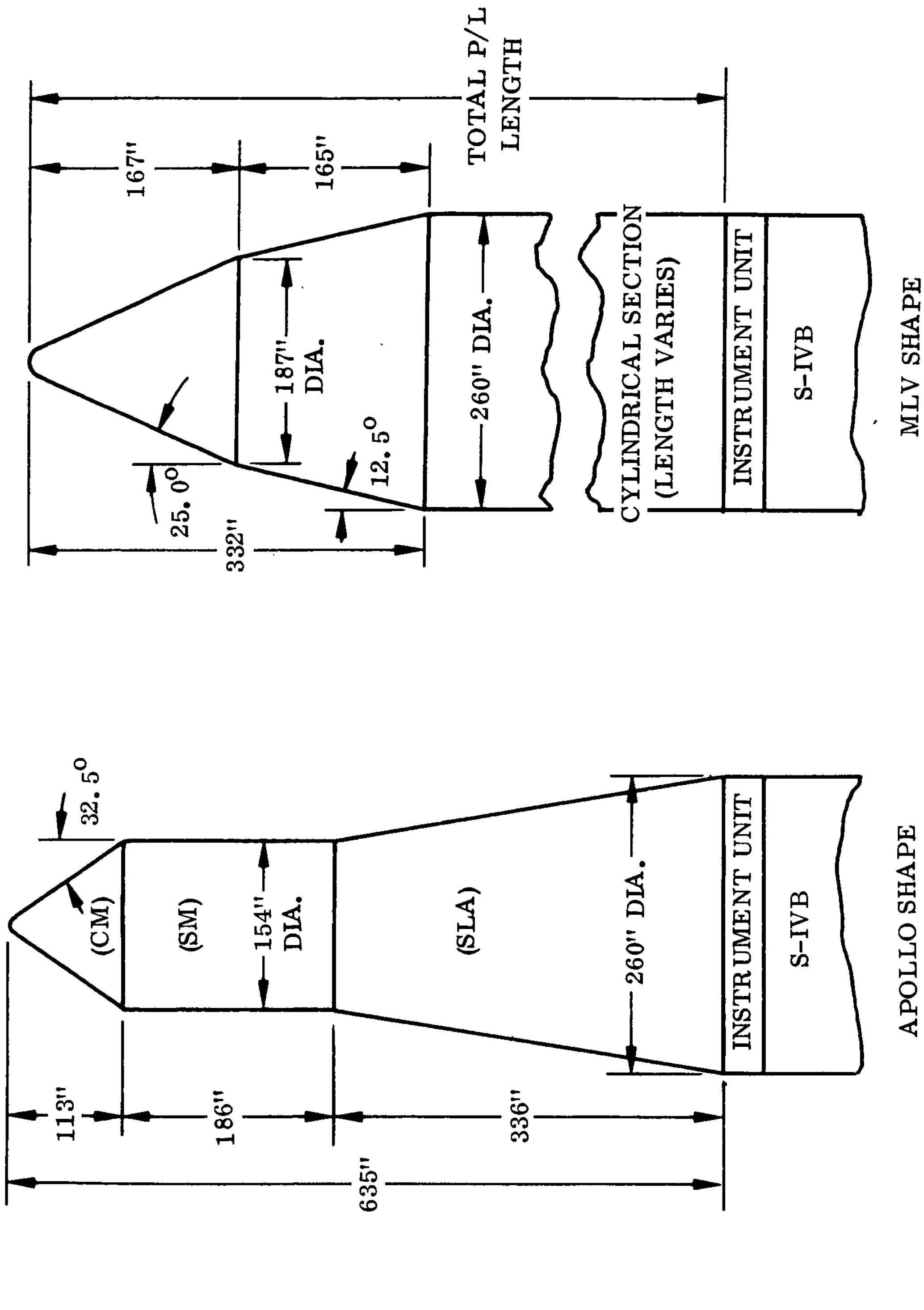


FIGURE 5.3-1 PAYLOAD SHAPES

5.3.1.2 (Continued)

All the low-earth orbit missions were assumed to be manned and used ground fitted space stations with the same basic S-IVB/Apollo shape. For low Earth coplanar missions, a two stage Saturn V (S-IC/S-II) vehicle would be used. For a polar mission, launch vehicle yaw steering is required to avoid spent-stage impact on inhabited land masses. A three-stage vehicle would be preferable from a performance standpoint because of large energy expenditures for the yaw steering maneuver. However, the large payload size required to obtain a reasonable payload density will cause large structural loads.

Synchronous missions would include manned astronomical laboratories, and unmanned, but man-visited communication, navigation and earth resources satellites. The Apollo payload shape would provide sufficient volume to obtain a reasonable density for these payloads with an estimated weight of 66,000 pounds.

5.3.1.3 Planetary Missions

All planetary missions were assumed to be unmanned. Planetary payload weight capability would be equivalent or less than the synchronous orbit mission capability. The Apollo payload shape provides reasonable payload density. Manned missions were not considered since these require large uprated Saturn V vehicles.

5.3.2 Mission Flight Profiles versus Performance and Engine/Stage Systems Impact

There are several possible flight profiles for each mission. The flight profile affects the payload weight and stage systems modifications such as number of engine starts, and the duration of idle mode, mainstage burn and coast periods. Mission requirements were studied on a comparative basis to determine the design profiles that are a reasonable compromise between J-2S engine/stage systems modification and mission peculiar requirements.

The J-2S operating criteria of particular interest to mission flight profiles are:

a. Nominal idle mode duration - 100 seconds

b. Maximum mainstage burn duration - 500 seconds

c. Mainstage cutoff to mainstage restart sequencing cannot be accomplished directly through the idle mode.

d. The required shutdown time between mainstage cutoff and restart signals is 300 seconds.

5.3.2.1 Lunar Flight Profiles

Candidate lunar flight profiles and the J-2S operational sequence requirements are shown in Figure 5.3-2.

The direct injection flight profile may be used on an unmanned mission. Although no parking orbit is used, which restricts the launch window and checkout prior to translunar injection (TLI), this may be acceptable for unmanned missions. Increased payload results from elimination of hydrogen boiloff and not forcing the velocity vector into a circular orbit.

The flight profile for the manned lunar orbital rendezvous (LOR) mission is the same as for the current Apollo/Saturn V. One S-II mainstage burn and two S-IVB mainstage burns are used. The maximum coast time in parking orbit is approximately 4.5 hours. Idle mode prior to engine mainstage start is used for engine/propellant conditioning. The engine start sequence requirements are within the design criteria of the J-2S engine.

The LOR profile was selected as the lunar design flight profile. The use of this flight profile eliminates mission peculiar requirements for engine/stage system changes and allows a direct assessment to be made of engine/stage changes required when J-2 engines are replaced with J-2S engines on the Saturn V vehicle.

5.3.2.2 Earth Orbital Mission Profiles

Low-Earth Orbit Mission Profiles (Coplanar)

Three candidate flight profiles for coplanar orbit missions with a two-stage vehicle (S-IC/S-II) are shown in Figure 5.3-3. The final circular orbit altitude required was assumed to be 300 N. M.

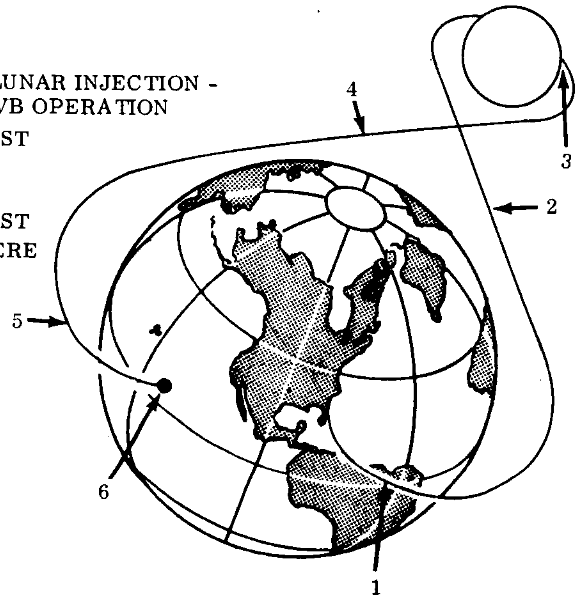
The first profile considered was direct injection to 300 N. M. altitude. This profile would have the least impact on engine/stage systems because only on S-II burn and no coast is required (no S-II repressurization system or coast attitude control system (RCS) needed). However, past studies have shown that direct injection boost profile is an inefficient way to fly to 300 N. M. since the payload delivered is only about two-thirds that of a Hohmann transfer to circular orbit.

The second flight profile considered used a Hohmann transfer from a 100 N. M. parking orbit to the 300 N. M. altitude. This profile would require three S-II burns: (1) mainstage for injection into parking orbit, (2) mainstage for boost out of parking orbit and (3) mainstage or idle mode for orbit circularization. Mainstage restart would require an S-II repressurization system.

THREE STAGE FLIGHT PROFILES

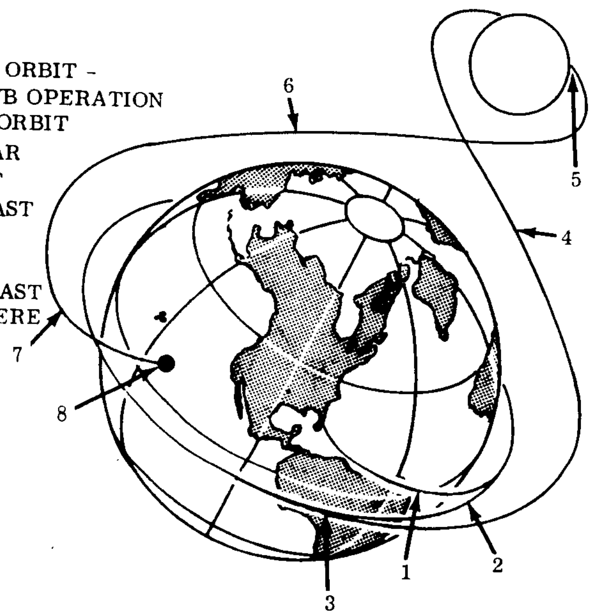
DIRECT

1. BOOST TO TRANSLUNAR INJECTION - S-IC, S-II AND S-IVB OPERATION
2. TRANSLUNAR COAST
3. LUNAR PASSAGE
4. "FREE RETURN" TRANSEARTH COAST
5. EARTH ATMOSPHERE REENTRY
6. SPLASH



LOR

1. BOOST TO EARTH ORBIT - S-IC, S-II and S-IVB OPERATION
2. COAST IN EARTH ORBIT
3. S-IVB TRANSLUNAR INJECTION BOOST
4. TRANSLUNAR COAST
5. LUNAR PASSAGE
6. "FREE RETURN" TRANSEARTH COAST
7. EARTH ATMOSPHERE REENTRY
8. SPLASH



J-2S ENGINE(S) SEQUENCE

DIRECT

S-II

- A) IDLE MODE - 1 SEC
- B) MAINSTAGE TO LOX DEPLETION

S-IVB

- A) IDLE MODE - 1 SEC
- B) MAINSTAGE TO TRANSLUNAR INJECTION
- C) MAINSTAGE CUTOFF

LOR

S-II

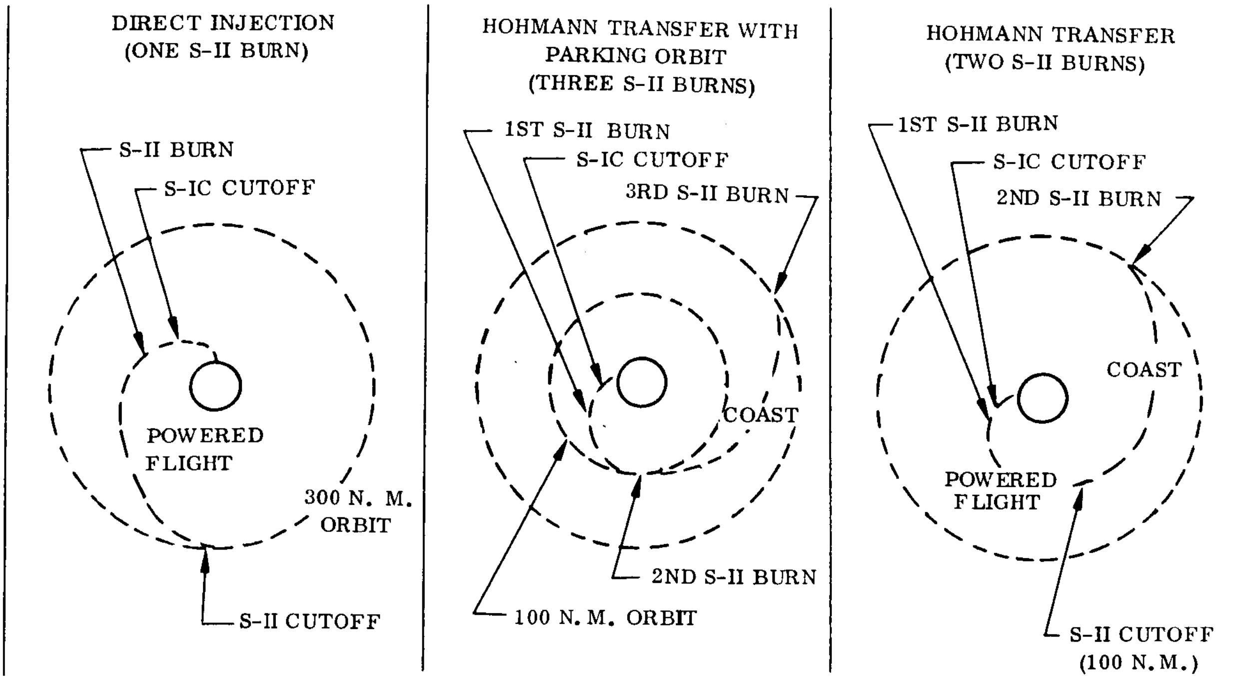
- A) IDLE MODE - 1 SEC
- B) MAINSTAGE TO LOX DEPLETION

S-IVB

- A) IDLE MODE - 1 SEC
- B) MAINSTAGE TO ORBIT
- C) MAINSTAGE CUTOFF
- D) COAST 3 ORBITS - 4.5 HRS.
- E) IDLE MODE - 100 SECS.
- F) MAINSTAGE TO TRANSLUNAR INJECTION
- G) MAINSTAGE CUTOFF

FIGURE 5.3.2 CANDIDATE LUNAR FLIGHT PROFILES

TWO-STAGE FLIGHT PROFILES
300 N. M. LAUNCH AZIMUTH 108° - COPLANAR FLIGHT



J-2S ENGINE SEQUENCE (S-II STAGE)

A) IDLE MODE - 1 SEC. B) MAINSTAGE TO ORBIT INJECTION	A) IDLE MODE - 1 SEC. B) MAINSTAGE TO PARKING ORBIT C) CUTOFF D) COAST 3 ORBITS - 4.5 HRS. E) IDLE MODE - 100 SEC. F) MAINSTAGE TO TRANSFER ORBIT G) CUTOFF H) COAST ~ 3000 SEC. I) IDLE MODE ORBIT CIRCULARIZATION ~150 SEC. OR IDLE MODE - 100 SEC. MAINSTAGE ORBIT CIRCULARIZATION ~2 SEC.	A) IDLE MODE - 1 SEC. B) MAINSTAGE TO TRANSFER ORBIT C) CUTOFF D) COAST~3000 SEC. E) IDLE MODE ORBIT CIRCULARIZATION ~150 SEC. OR IDLE MODE - 100 SEC. MAINSTAGE ORBIT CIRCULARIZATION ~2 SEC.

FIGURE 5.3-3 CANDIDATE TWO-STAGE LOW EARTH ORBIT FLIGHT PROFILES

5.3.2.2 (Continued)

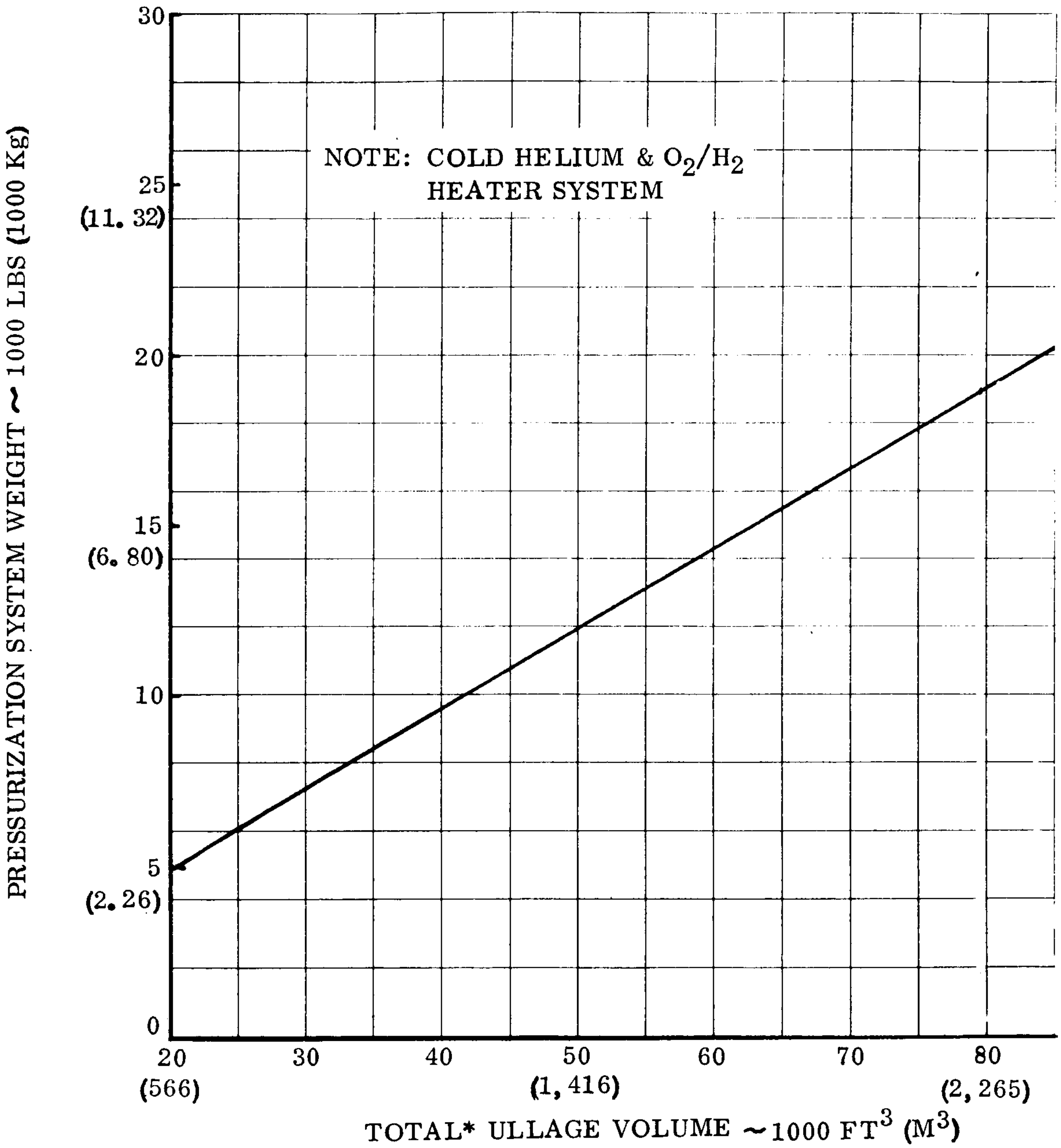
The third profile considered used a Hohmann transfer orbit without a parking orbit. The S-II injects on to the Hohmann transfer orbit at 100 N. M. and coasts to apogee at 300 N. M. (approx. 50 min.). An S-II mainstage burn or idle mode burn is used for orbit circularization. Two S-II burns are required: (1) mainstage for injection on to the Hohmann transfer orbit and (2) mainstage or idle mode for orbit circularization. Mainstage restart requires an S-II repressurization system while idle mode restart does not. An S-II repressurization system weight was estimated to be approximately 12,000 pounds. The payload,consequently, would be reduced 12,000 pounds because, for two-stage vehicles, the exchange ratio between S-II weight and payload is 1 for 1. This estimate was based on a Boeing "in-house" cursory analysis of a S-IVB type cryogenic repressurization design using an O_2/H_2 burner. Systems weights as a function of S-II total ullage volume is shown in Figure 5.3-4. Mainstage circularization requires only two seconds of full mainstage burn in addition to the required 100 second idle mode burn for engine/propellant conditioning. Idle mode circularization requires a total of about 150 seconds idle mode burn. Idle mode circularization should be preferable when considering the payload degradation and the cost of a S-II repressurization system.

The design flight profile then selected was S-II burn to perigee of the Hohmann transfer orbit, coast for approximately 50 minutes to 300 N. M. and orbit circularization using S-II idle mode. The idle mode circularization demonstrates J-2S low impulse maneuver capability. This profile requires minimal S-II modifications since no parking orbit or mainstage restart is used. A reaction control system is required for attitude control during the coast which must be provided by either the payload or the S-II stage. For purpose of future comparative interest, it was decided to include an S-II coast RCS system as a design requirement. Such a system is, of course, a mission peculiar item and not attributable to any J-2S engine change.

Polar Orbit Mission Profiles

The two candidate polar mission profiles are shown in Figures 5.3-5. Both profiles use yaw steering of all boost stages to avoid spent stage impacts on populated land masses. S-IC yaw steering of 1 degree per second is initiated after maximum dynamic pressure time of flight (approx. 100 sec.) to minimize structural loading during the yaw maneuver. The profiles apply to either three-stage or two-stage vehicles.

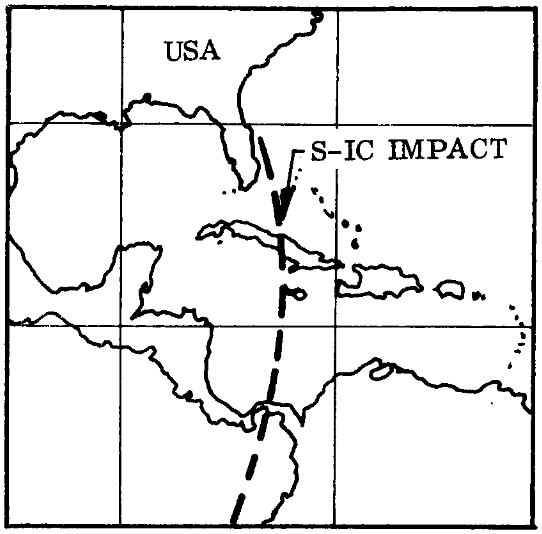
A flight profile around the east coast of South America expends a large amount of energy. The payload weight for a three stage vehicle is only about 50,000 pounds and is much smaller than required for manned activity in polar orbit. Therefore, this profile was not considered further.



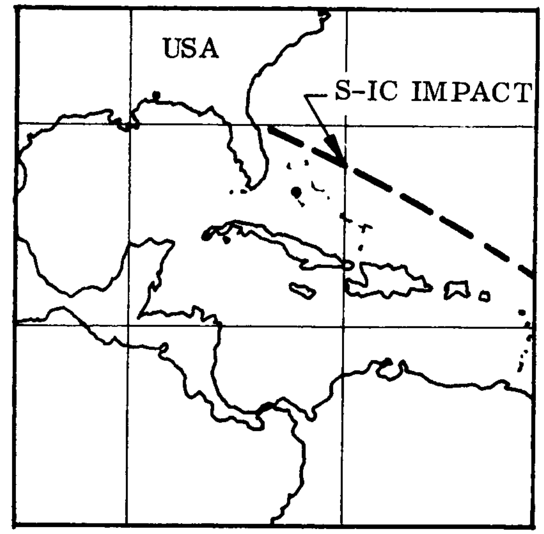
* (LOX + LH_2) 2ND BURN + (LOX + LH_2) 3RD BURN

FIGURE 5.3-4 S-II REPRESSURIZATION SYSTEM WEIGHT

FLIGHT PROFILES
TWO-AND THREE-STAGE
100 N.M. - YAW STEERING ALL STAGES



OVERLAND
(CUBA & PANAMA)



WATER ROUTE

J-2S ENGINE SEQUENCE

S-II

A) IDLE MODE - 1 SEC.
B) MAINSTAGE TO ORBIT INJECTION (2-STG) OR MAINSTAGE TO LOX DEPLETION (3-STG.)

S-IVB (3-STG. APPLICATION)

A) IDLE MODE - 1 SEC.
B) MAINSTAGE TO ORBIT INJECTION

S-II

A) IDLE MODE - 1 SEC.
B) MAINSTAGE TO LOX DEPLETION

S-IVB

A) IDLE MODE - 1 SEC.
B) MAINSTAGE
C) COAST
D) IDLE MODE - 100 SEC.
E) MAINSTAGE TO ORBIT INJECTION

FIGURE 5.3-5 CANDIDATE POLAR ORBIT FLIGHT PROFILES

5.3.2.2 (Continued)

The flight profile over Cuba and the Isthmus of Panama was selected as the design flight profile for the polar mission. This profile is direct injection to 100 N. M. circular orbit. This profile requires only a single burn of each boost stage. Except for the yaw steering guidance commands, no other mission peculiar modifications are required. Flights to altitudes above 100 N. M. would involve the same design requirements as the Low Earth Orbit Mission (coplanar).

Synchronous Mission Profiles

The two candidate synchronous mission profiles are shown in Figure 5.3-6. The first flight profile considered used continuous burn of all three stages into the transfer orbit at 100 N. M. No parking orbit is used. After injection, the S-IVB is cut off and the S-IVB payload coasts in a transfer orbit for approximately 5.3 hours to the synchronous altitude. At the synchronous altitude, the S-IVB is reignited for the second time for circularization and plane change. Since no parking orbit is used, hydrogen boiloff is reduced and a higher payload capability results. However, the elimination of the parking orbit greatly reduces the launch window.

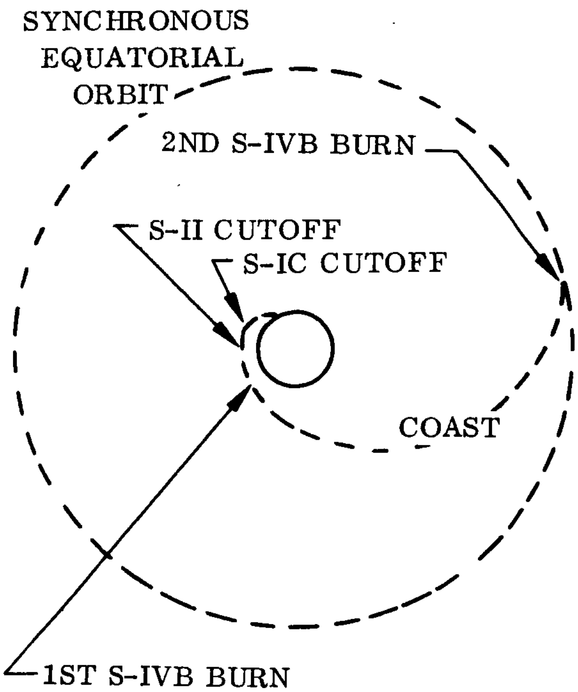
The second flight profile considered used continuous burn of all three stages into a 100 N. M. parking orbit. This portion of the profile is very similar to the LOR profile and imposes no additional requirements on the vehicle. The number of revolutions in the 100 N. M. parking orbit can be varied from one up to five depending on the satellite longitudinal position desired. The S-IVB is reignited for boost out of parking orbit onto the Hohmann transfer orbit. After a coast of approximately 5.3 hours in the transfer orbit, the S-IVB is reignited for orbit circularization and plane change. The five revolution parking orbit gives total satellite longitudinal position flexibility. However, the parking orbit coast of 7.5 hours combined with the transfer coast of 5.3 hours results in significant stage systems impact. The operational lifetime of certain S-IVB and IU systems or components would have to be extended. Also, hydrogen boil-off during the coast periods will be significant and result in less delivered payload than the first profile.

The second profile with the parking orbit coast was selected for the synchronous orbit mission. This flight profile is a "worst case" condition for stage systems modifications and payload capability due to extended operational lifetime, S-IVB three mainstage starts, and large hydrogen boiloff. The profile also gives maximum flexibility in selecting satellite position.

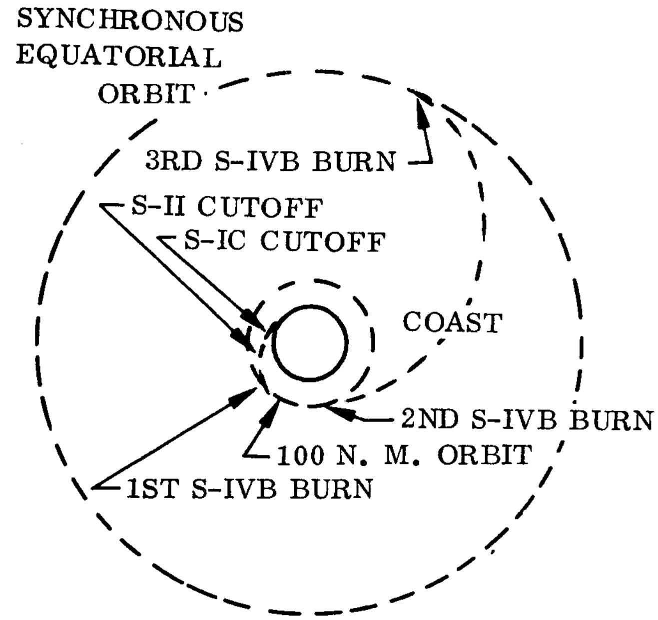
Although the five revolution parking orbit is selected for the design flight profile, the payload obtained with one and three revolutions were determined as a portion of the alternate mission studies.

THREE STAGE FLIGHT PROFILES

TRANSFER WITH NO PARKING ORBIT



HOHMANN TRANSFER WITH PARKING ORBIT



J-2S ENGINE SEQUENCE

Transfer with no parking orbit

S-II
A) IDLE MODE - 1 SEC.
B) MAINSTAGE TO LOX DEPLETION

S-IVB
A) IDLE MODE - 1 SEC.
B) MAINSTAGE TO TRANSFER ORBIT
C) CUTOFF
D) COAST - 5.3 HRS.
E) IDLE MODE - 100 SEC.
F) MAINSTAGE ORBIT CIRCULARIZATION

Hohmann transfer with parking orbit

S-II
A) IDLE MODE - 1 SEC.
B) MAINSTAGE TO LOX DEPLETION

S-IVB
A) IDLE MODE - 1 SEC.
B) MAINSTAGE TO PARKING ORBIT
C) CUTOFF
D) COAST 5 ORBITS - 7.5 HRS.
E) IDLE MODE - 100 SEC.
F) MAINSTAGE TO TRANSFER ORBIT
G) CUTOFF
H) COAST - 5.3 HRS.
I) IDLE MODE - 100 SEC.
J) MAINSTAGE ORBIT CIRCULARIZATION

FIGURE 5.3-6 CANDIDATE SYNCHRONOUS ORBIT FLIGHT PROFILES

5.3.2.3 Planetary Mission Profiles

The planetary mission flight profile is the same as the LOR mission. One burn of the S-II and two burns of the S-IVB are required. The second burn of the S-IVB is used to boost out of the parking orbit to the specified energy level. This design flight profile imposes only the following system modification beyond those required for the design LOR flight profile: Slosh baffles must be moved for each propellant level at engine restart. These levels are mission energy dependent.

5.3.3 Payload Envelope Trades

Saturn V structural design for the LOR mission is based on the payload size and weight reacting under specific wind criteria. Saturn V type vehicles with payloads that have weights and plan form areas significantly different from Apollo can result in structural loads that exceed vehicle stages design capability. The purpose of these trades was to establish payload shapes and sizes for J-2S/Saturn V missions and to predict probable stage structural modifications. Both two-and three-stage vehicles were examined.

The maximum length of the payload is restricted by the mobile launcher hammer-head crane. For a three-stage Saturn V, the maximum payload length above the IU is 109 feet. For a two-stage Saturn V, the maximum payload length above the S-II is 170 feet.

Payload weight has a major influence on the structural loads. It also influences the payload size since a reasonable payload density must be maintained. For space station application, a payload density of approximately 5 pounds per cubic foot was assumed. For Apollo-type missions, a density of about eleven pounds per cubic foot was assumed.

Two payload shapes were considered; the standard Apollo and a 260-inch diameter cylinder with a modified launch vehicle (MLV) nose cap. The MLV nose cap is a configuration originated for advanced studies purposes. The two payload shapes are shown in Figure 5.3-1.

5.3.3.1 Lunar Mission Payload Envelopes

The standard Apollo payload shape was selected for the lunar mission/LOR flight profile. This shape maintains the Saturn V vehicle geometry and gives a direct comparison to the Saturn V with J-2 engines. The increase in LOR payload weight is uniformly distributed within the Apollo spacecraft. Structural loads will increase due to the greater payload weight but should be within SA-511 stage capability.

5.3.3.2 Earth Orbital Payload Envelopes

Low-Earth Orbit Payload Envelopes (Coplanar)

Studies have shown that for manned mission, the S-IVB/Apollo shape can satisfy postulated mission volume requirements. Since this envelope has the same plan form area as the three-stage Saturn V, it would have minimal effect on stage structural loads and probably no stage structural modifications would be required.

The other payload envelope considered was the 260-inch cylinder with the MLV nose cap. Results of previous studies show that structural loads would be larger than with a S-IVB/Apollo envelope due to the large diameter and nose cap. Stage structure would probably need to be strengthened.

The S-IVB/Apollo envelope was selected since it satisfies postulated mission volume requirements with no expected stage structural modifications.

Polar Orbit Payload Envelope

The polar orbit missions considered are manned space stations.

A two-stage polar mission would have payload volume requirements similar to the LEO mission discussed above. For similar reasons, the S-IVB Apollo envelope was selected for the two-stage polar orbit mission.

A three-stage polar mission payload envelope is restrained to a length 109 feet above the IU because of mobile launcher hammerhead crane interference. In order to maintain a reasonable space station volume with the 109 foot height limit, a 260-inch cylinder with a MLV nose cap must be used. This payload envelope results in a total vehicle height considerably greater than nominal Saturn V. The higher structural loads would undoubtedly exceed stage structural capability.

Synchronous Orbit Payload Envelope

Based on past studies, vehicle payload capability to synchronous orbit is expected to be on the order of 40 percent less than that obtained for a LOR mission. The Apollo shape was selected to be the design payload envelope, since at this payload weight it affords a reasonable payload density.

5.3.3.3 Interplanetary Payload Envelopes

The Apollo envelope was selected for interplanetary payload missions. These payloads are undefined and the Apollo shape has sufficient payload volume for all predicted payload weights.

5.3.4 Summary of Trade Studies for Configuration Selection

From the trade studies described, four Saturn V vehicle configurations were selected for defining the impact of incorporating J-2S engines in the upper stages. They are:

a. A three-stage vehicle using the Apollo payload shape and flying the basic lunar orbital rendezvous flight profile.

b. A three-stage vehicle using an Apollo payload shape and flying to the synchronous altitude via a 100-nautical mile parking orbit.

c. A two-stage vehicle (S-IC/S-II) using an S-IVB workshop type space station with the Apollo shape and flying to a 300 N M circular Earth orbit.

d. A two-stage vehicle (S-IC/S-II) using an S-IVB workshop type space station with the Apollo shape and flying to a 100 N M circular polar orbit.

A fifth vehicle, a three-stage vehicle with a 260-inch diameter payload above the S-IVB, flying the polar orbit mission, was given a cursory examination. As suspected, the increase in inert weight caused by major structural modification to all stages reduced the vehicle payload capability below that of a two-stage vehicle. It was, therefore, studied no further. The trade study information is reported in Appendix A of D5-15772-6, Structural Analysis.

Mission Peculiar Aspects

It became apparent as the trade studies developed that a factual comparison between a J-2 vehicle and a J-2S vehicle can, only be made using the LOR mission profile and the three-stage vehicle configuration. Other missions introduce vehicle changes which are not directly attributable to J-2/J-2S engine operating differences, and should not be charged against the J-2S engine. It was, therfore, decided that the basic reference for J-2/J-2S comparison was the three-stage LOR vehicle. Vehicle and stage changes required for other missions are treated as "mission peculiar."

SECTION 6

VEHICLE DESCRIPTION

6.0 GENERAL

This section contains descriptions of the four design vehicle configurations selected in the Phase I trade study activity. Detailed design information was developed for these four design configurations during the Phase II activity. An assessment of total Saturn V vehicle, support operations and KSC launch operations impact was made to arrive at a master schedule and the program costs required to incorporate J-2S engines into the Saturn V vehicle. Two two-stage configurations and two three-stage configurations are defined.

The external geometry of all four design configurations is identical to the Baseline J-2/Saturn V SA-511. "S-IVB Workshop"/Apollo payload envelopes are used for the two-stage configurations. Each configuration is an assembly of SA-511 type stages modified only for installation of the J-2S engine or to withstand increased structural loads. The increase in LOR payload weight was uniformly distributed within the Apollo spacecraft. The payload weight of each of the other configurations was assumed to be uniformly distributed within the payload envelope. Vehicle attitude control during boost was accomplished in the normal Saturn V manner.

The S-IC stage uses five standard F-1 engines which develop 7,610,064 pounds of total thrust. The S-IC total propellant weight used was 4,577,113 pounds. The S-II and S-IVB stages used J-2S engine(s) calibrated at 265,000 pounds thrust each. J-2S idle mode operation, three-start capability, and LOX depletion cutoff were used to enhance mission flexibility and stage simplification. The thrust structure of both the S-IVB and S-II stages was modified for the increased J-2S thrust. The S-II propellant tank capacity used was 819,325 pounds of LOX and 158,195 pounds of LH_2 for a maximum capacity of 977,520 pounds. The S-IVB propellant tank capacity used was 193,273 pounds of LOX and 44,414 pounds of LH_2 for a maximum capacity of 237,687 pounds.

6.1 THREE STAGES - LOR MISSION

The LOR mission design vehicle is the basic Saturn V vehicle as developed for the lunar landing mission. The vehicle consists of the S-IC stage, S-II stage, S-IVB stage, IU and Apollo spacecraft payload. The vehicle configuration is shown in Figure 6-1.

No structural modifications are necessary because the increased structural loads due to the increased payload weight do not exceed J-2/Saturn SA-511 stage structural capability.

Stage systems modifications are only those required for replacement of J-2 engines with J-2S engines.

6.2 THREE STAGE - SYNCHRONOUS MISSION

The Synchronous Mission design vehicle is the three-stage J-2S/Saturn V (Figure 6-1). Structural loads do not exceed structural capability of the stages, therefore, the S-IC and S-II stages are identical to those of the LOR design vehicle. The only modifications to the S-IVB stage were for the three-burn capability and for extended stage lifetime due to the long flight time of the Synchronous Orbit Mission. The lifetime of the Instrument Unit (IU) was also extended. The S-IVB Auxiliary Propulsion System (APS) was used for attitude control during vehicle coast.

6.3 TWO STAGE - LOW EARTH ORBIT MISSION

The low Earth mission design vehicle is a two-stage (S-IC/S-II) configuration. Saturn V/Apollo external geometry is retained by using an "S-IVB workshop"/Apollo payload envelope as shown in Figure 6-2.

Structural loads do not exceed the Baseline J-2/Saturn V SA-511 stage structural capability, therefore, the S-IC is identical to the LOR/S-IC. The S-II has systems modifications for operation during the coast to orbit period. A continuous propellant vent system and a reaction control system* is provided. J-2S idle mode operation is used to provide the low impulse maneuver for orbit circularization. The IU is located between the "S-IVB Workshop" and the Apollo spacecraft to limit the required modifications from Saturn V equipment.

6.4 TWO STAGE - POLAR ORBIT MISSION

The polar mission design vehicle is a two-stage (S-IC/S-II) configuration. Saturn V/Apollo external geometry was retained by using an "S-IVB Workshop"/Apollo payload envelope as shown in Figure 6-2. External geometry is the same as for the two-stage low-Earth orbit mission.

Structural loads were within the vehicle structural capability except in the S-IC RP-1 tank. The beef-up of the S-IC tank side wall and longitudinal stringers increased the stage weight by 74 pounds. It is also possible that load alleviation techniques could be used to lower the structural loads to Baseline S-IC SA-511 capability. For the study, the modification to the tank was assumed. The S-II stage is identical to the LOR/S-II. Since the flight profile is direct ascent (no coast), a continuous propellant vent system or a reaction control system was not required. The IU is located between the "S-IVB Workshop" and the Apollo spacecraft to limit the required modifications from Saturn V equipment. The yaw steering maneuver requires no modification other than those caused by the structural overload condition in the S-IC.

* Attitude control during coast could be provided by either an RCS system on the S-II stage or on the payload (Space Station). For the purpose of future comparative interest, it was decided to include an S-II RCS system as a design requirement.

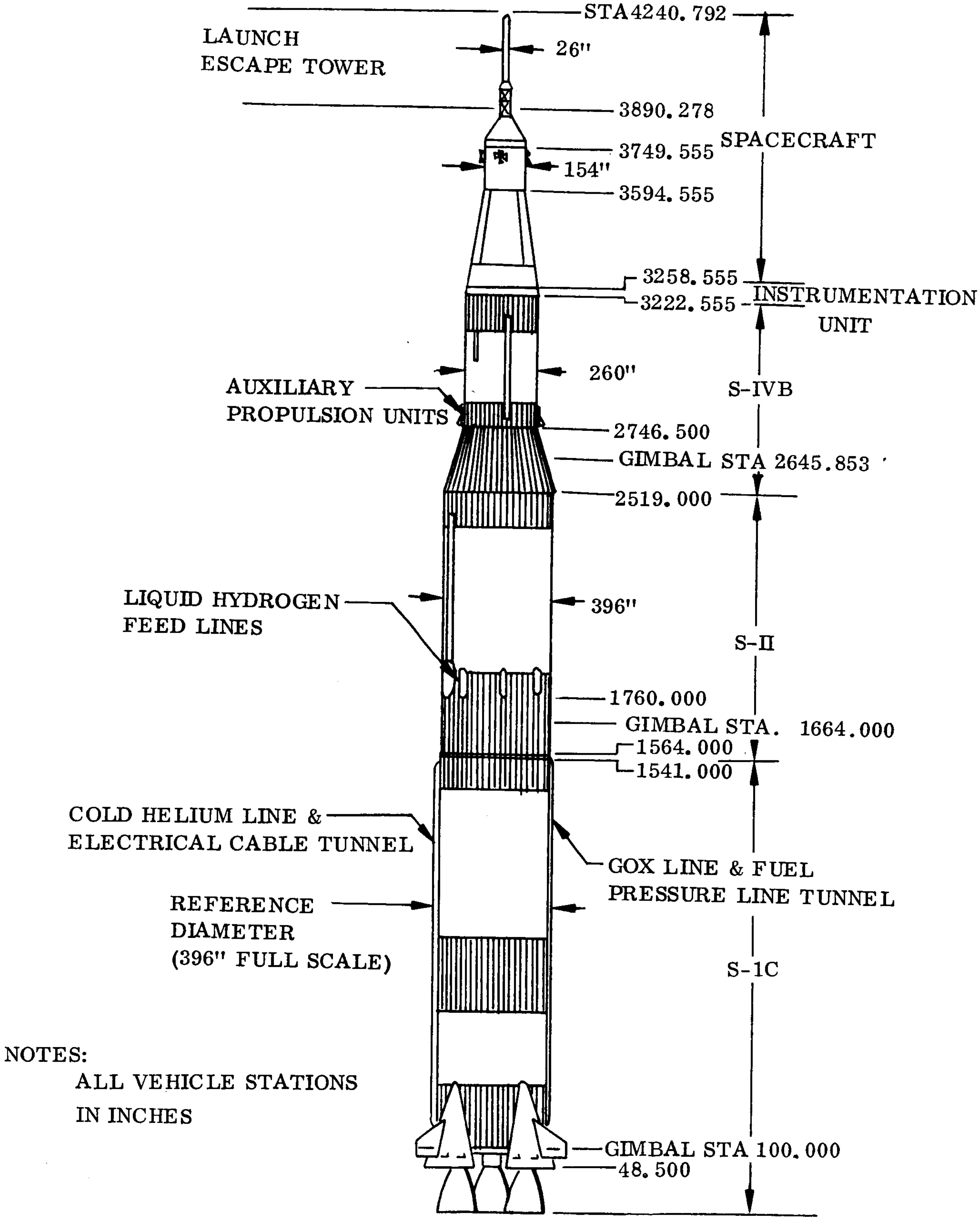


FIGURE 6-1 APOLLO-SATURN V VEHICLE CONFIGURATION

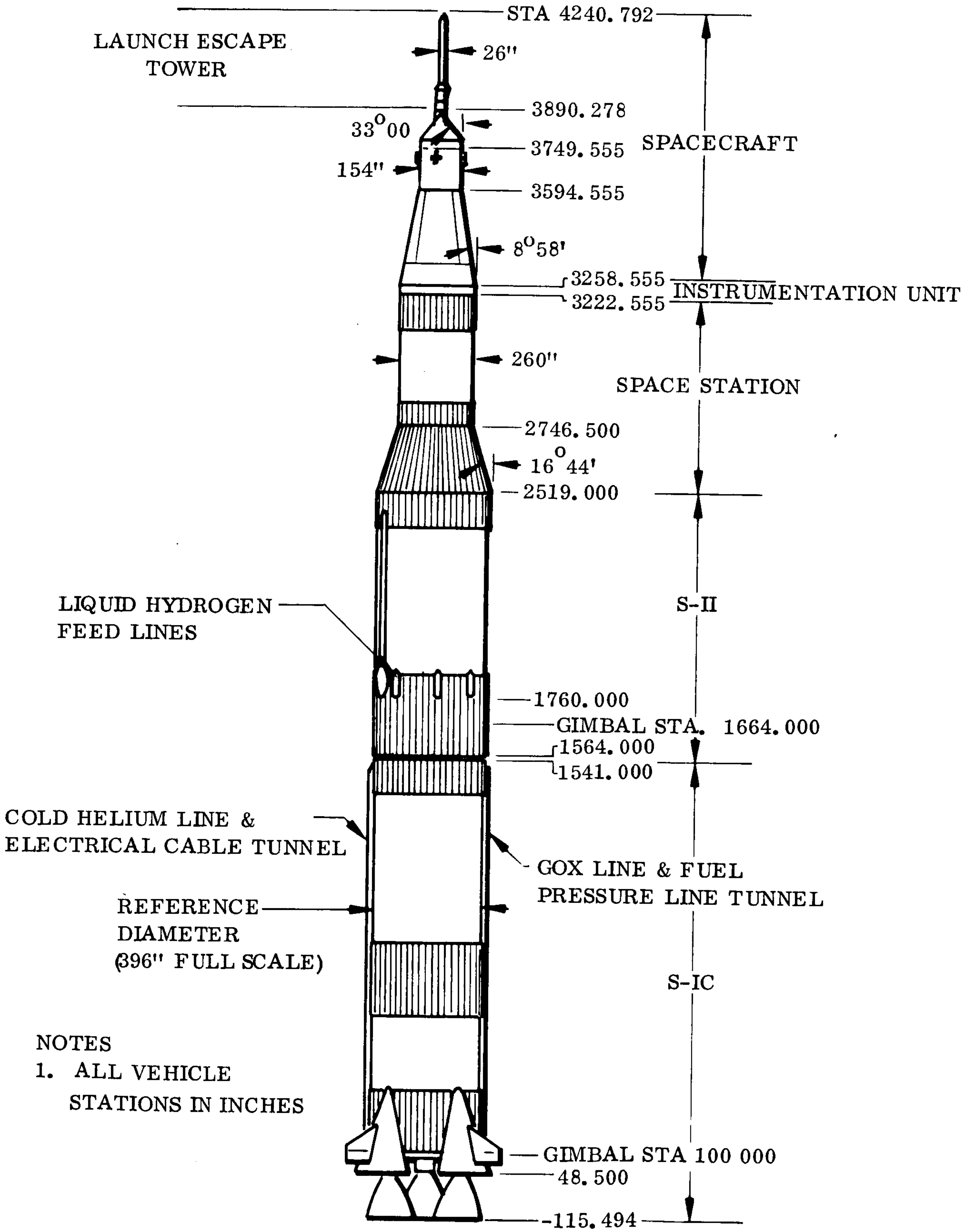


FIGURE 6-2 TWO-STAGE SATURN V VEHICLE CONFIGURATION

SECTION 7

VEHICLE ENVIRONMENTS

7.0 GENERAL

The J-2S/Saturn V performance parameters, nominal design characteristics and requirements are given in this section. These design environments were used by stage contractors in their analysis of stage impact. The final stage designs resulting from the analysis of these environments are given in Section 10.0, Stage/Systems Description.

7.1 BASELINE SA-511 SATURN V VEHICLE

The SA-511 Saturn V launch vehicle with J-2 engines is used in the J-2S Improvement Study as the baseline vehicle. The differences between the Baseline SA-511 vehicle and the J-2S vehicle reflects the effects of incorporating the J-2S into the Saturn V. The Baseline SA-511 vehicle is the basic LOR mission design vehicle as developed for the lunar landing mission. The vehicle consists of the S-IC stage, S-II stage, S-IVB stage, IU, and Apollo spacecraft payload. The vehicle configuration is the same as shown in Figure 6-1. The S-IC stage uses five F-1 engines which develop 7,610,000 pounds of total thrust. The S-II and S-IVB stages use 230,000 pound thrust J-2 engines. The flight event history, vehicle weight description, and propulsion data, are given in Table 7.1-I through 7.1-III. Vehicle design characteristics are shown in Table 7.1-IV. The LOR payload capability is 100,078 pounds.

The J-2S Improvement Study Baseline Launch Vehicle SA-511, dated September 1968, as supplied by MSFC is contained in Appendix A of this document.

7.2 J-2S SATURN V DESIGN TRAJECTORIES

Trajectory simulations were made for the four design mission/flight profiles selected in Phase I trade studies. The design trajectories are nominal trajectories in that they do not include 3σ dispersions. They identify payload capability and design characteristics for structural, control, and heating analyses. The Baseline SA-511 payload capability and design characteristics are shown for comparison.

The boost optimization program used to generate the design trajectories is a "plumb-line/COV" point mass type simulator. The flight simulation begins with a 12 second vertical rise followed by a programmed tilt maneuver terminating at 35 seconds. At this point, a gravity turn is initiated and continued until the COV guidance is initiated. The COV guidance technique is used to optimize the remaining portion of the trajectory. The S-IC center engine cutoff occurs 12 seconds prior to S-IC cutoff. There is a 3.8 second coast between S-IC cutoff and S-II ignition. Thrust decay or thrust buildup is not simulated but the thrust buildup and decay propellants are accounted for in the stage drop weights.

TABLE 7.1-I BASELINE VEHICLE SA-511 FLIGHT EVENT HISTORY (J-2/SATURN V)

Time (sec)	Event
0	Lift-off
12	Initial Tilt Program
35	Terminate Tilt Program
149.486	Inboard Engine Cutoff
153.0	Initiate 'chi freeze' Mode
161.486	Outboard Engine Cutoff (Separation)
165.286	Ignite S-II at MR = 5.0
167.786	Shift MR to 5.5
193.986	Jettison S-IC/S-II Interstage
198.986	Jettison LES, Initiate Guidance
426.312	MR Shift to 4.7
543.832	S-II Cutoff/Separation S-IVB Ignition @ MR = 5.5
676.027	S-IVB Cutoff in 100 N. M. Parking Orbit S-IVB Reignition @ MR = 5.0
993.160	S-IVB Final Cutoff

TABLE 7.1-II BASELINE VEHICLE SA-511 WEIGHT DESCRIPTION (J-2/SATURN V)

I.	S-IC Stage		
	Total Mainstage Propellant		4,577,113 LBS.
	S-IC Stage (dry)		289,409
	S-IC Residuals		67,585
	S-IC Thrust Decay Propellant		9,068
	Inboard Eng T.D.	1914	
	Inboard Eng Exp. Prop.	408	
	Outboard Eng T.D.	6746	
	S-IC/S-II Interstage (small)		1,548
	S-II Ullage Rocket Propellant		1,360
	S-II Thrust Buildup		1,801
	S-IC/S-II Large Interstage		9,435
	Launch Escape System		8,936
II.	S-II Stage		
	Total Mainstage Propellant		970,441*
	S-II Stage (dry)		80,686
	S-II Residuals		14,136
	S-II Thrust Decay Propellant		342
	S-II/S-IVB Interstage and Retro Propellant		8,086
	S-IVB Aft Frame (Sep with Interstage)		48
	S-IVB Ullage Rocket Propellant		117
	S-IVB Thrust Buildup Propellant		440
III.	S-IVB Stage		
	S-IVB Weight Lost in Parking Orbit (100 N. M. for 3 Orbits)		4,411
	S-IVB Thrust Decay Propellant	107	
	LH_2 Vented	3281	
	GOX Vented	170	
	Auxiliary Prop. Losses and Ullage for Restart	438	
	First Burn Propellant	18	
	S-IVB Restart Thrust Buildup Propellant	397	
	S-IVB Mainstage Prop (Incl. FPR and FGR)		227,991**
	S-IVB Stage (dry)		25,033

TABLE 7.1-II BASELINE VEHICLE SA-511 WEIGHT DESCRIPTION (Continued)
(J-2/SATURN V)

S-IVB Residuals	2,508
S-IVB Thrust Decay Propellant	93
S-IVB Second Burn Roll Propellant	34
Instrument Unit	4,183
Flight Geometry Reserves (=39 m/sec)	1,211
Flight Performance Reserves (=70 m/sec)	2,260
Net Payload	100,078 LBS.

* S-II LOX Loading = 815,427
S-II LH_2 Loading = 155,014

** S-IVB LOX Loading = 190,914
S-IVB LH_2 Loading = 37,077

TABLE 7.1-III BASELINE VEHICLE SA-511 PROPULSION DATA
(J-2/SATURN V)

S-IC Stage

Thrust/eng = 1,522,000 lb
$\dot{w}$/eng = 5754.2533 lb/sec
Ae/eng = 9.9313349 m^2/eng
Cross sectional area used for Aero computation = 79,45976 m^2

S-II Stage

MR = 5.0
F = 205052 lb/eng
$\dot{w}$/eng = 481.34272 lb/sec

MR = 5.5
F = 229927 lb/eng
$\dot{w}$/eng = 543.69117 lb/sec

MR = 4.702
F = 190125 lb/eng
$\dot{w}$/eng = 445.25761 lb/sec

S-IVB Stage

MR = 5.5
F = 231012 lb/eng
$\dot{w}$ = 543.68557 lb/sec

MR = 5.0
F = 206012 lb/eng
$\dot{w}$ = 481.33644 lb/sec

The S-IVB stage burns at MR = 5.5 from ignition to orbital insertion. It is restarted and burns to injection at MR = 5.0 .

TABLE 7.1-IV
BASELINE VEHICLE SA-511 DESIGN CHARACTERISTICS
(J-2/SATURN V)

LOR PAYLOAD (LBS)	100,078
LOAD CRITERIA	
MAX Q (LBS/FT^2)	699
G'S AT MAX Q	2.07
MAX G	4.15
HEIGHT (FT)	363.0
CONTROL MODE	GIMBAL ENGS. & APS** DURING S-IVB COAST
HEATING	
AERODYNAMIC (AHI) (FT - LBS/FT^2 - RAD)	46.43 X 10^6
THRUST (LBS) MAX	
S-IC	7,610,064
S-II	1,149,635
S-IVB	231,012

** AUXILIARY PROPULSION SYSTEM

7.2 (Continued)

The S-II stage uses a 5.0 to 5.5 to 4.7 mixture ratio shift schedule. The S-II is ignited at a mixture ratio of 5.0 and shifted to 5.5 after 2.5 seconds. The shift from 5.5 to 4.7 occurs when the propellants remaining in the tanks reach a 4.7 ratio. The remaining flight events are mission peculiar and are identified under each mission design trajectory discussion.

The J-2S thrust and Isp used in generating the trajectories are presented in Figures 7.2-1 and 7.2-2, respectively. The S-II thrust and Isp is degraded, compared to the S-IVB stage, due to engine clustering effects. This method is that used by MSFC to obtain the data supplied in Table XI of Appendix A.

The S-IC propulsion characteristics were simulated in the following manner. Thrust was computed by adding the thrust altitude pressure term, $(P_o - P_a) A_e$, to the sea level thrust. Propellant flow rate was assumed to be a constant equal to the ratio of sea level thrust to sea level specific impulse. The effect of acceleration on thrust was not considered. The one second of idle mode operation required for the start of the J-2S engines was not simulated. However, the 100 seconds of idle mode operation for J-2S restart was simulated. Idle mode thrust was assumed to be 5,000 pounds for the full 100 seconds with an Isp of 280 seconds at a mixture ratio of 2.5.

The aerodynamic heating indicator (AHI) value was calculated for each design trajectory for aeroheating comparison with the Baseline Vehicle SA-511. Detailed definition of the four design trajectories along with computer printouts showing pertinent performance and trajectory data is included in Document D5-15772-3, Mission/Performance Analysis - J-2S Improvement Study.

7.2.1 Three-Stage-Lunar Orbital Rendezvous Mission

The J-2S/Saturn V vehicle with an Apollo payload shape is used for Lunar Orbital Rendezvous (LOR) Missions to transport men and material to the lunar surface. The LOR design flight profile is identical to Baseline SA-511 design flight profile and is shown in Figure 7.2-3.

The launch vehicle phases of the LOR flight profile are as follows:

a. Boost with three stages to 100 NM Earth parking orbit.
b. Three revolution coast in Earth parking orbit.
c. S-IVB boost to translunar injection.
d. Launch vehicle/spacecraft (LV/SC) separation.

The simulated vehicle is launched along a 72^{o} launch azimuth. Maximum dynamic pressure (q) of 709 pounds/foot^2 occurs at 85 seconds after liftoff. Maximum longitudinal acceleration of 4.1 g's occurs at S-IC center engine cutoff (149.5 sec).

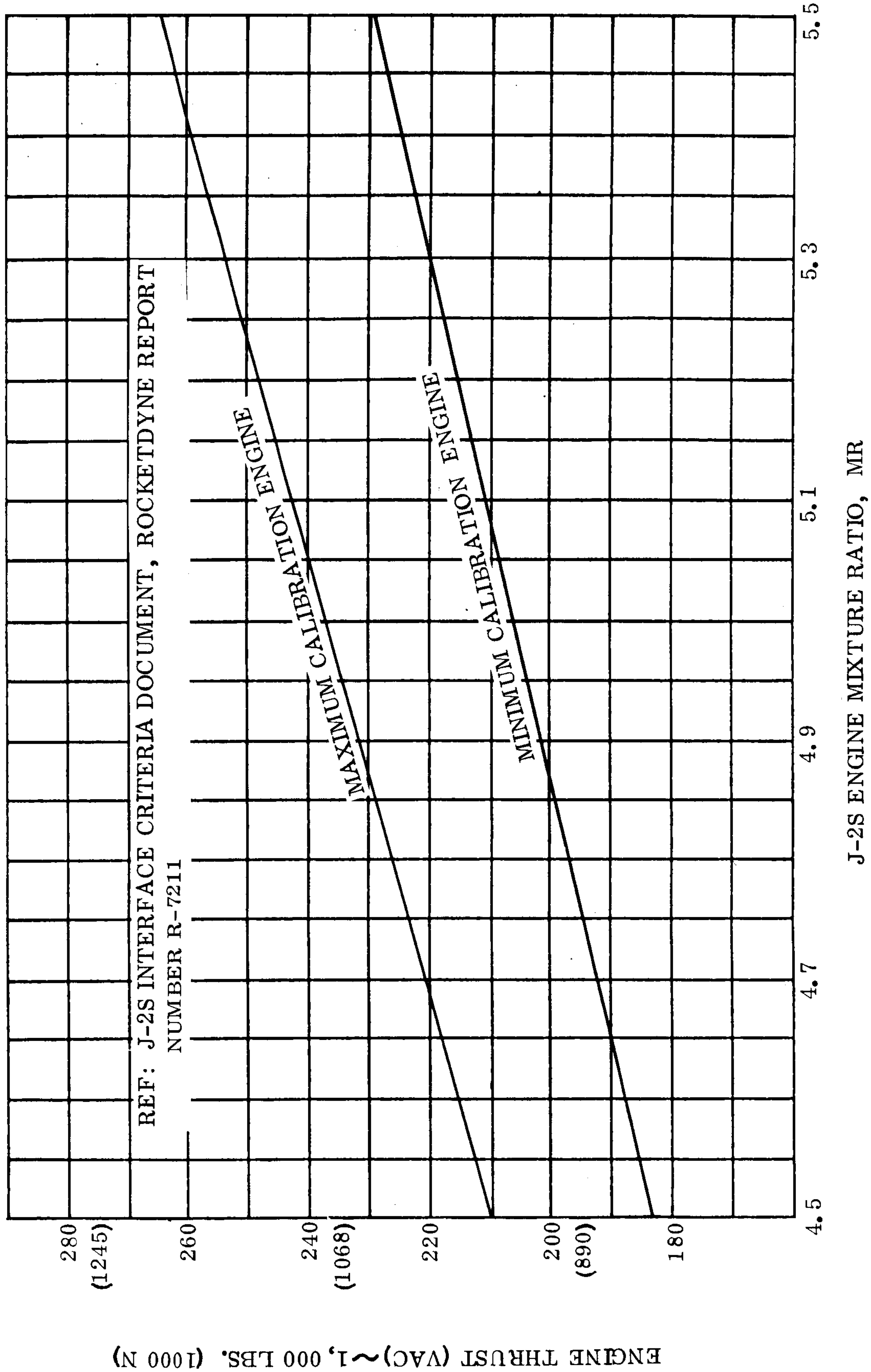


FIGURE 7.2-1 J-2S SINGLE ENGINE THRUST VERSUS MIXTURE RATIO

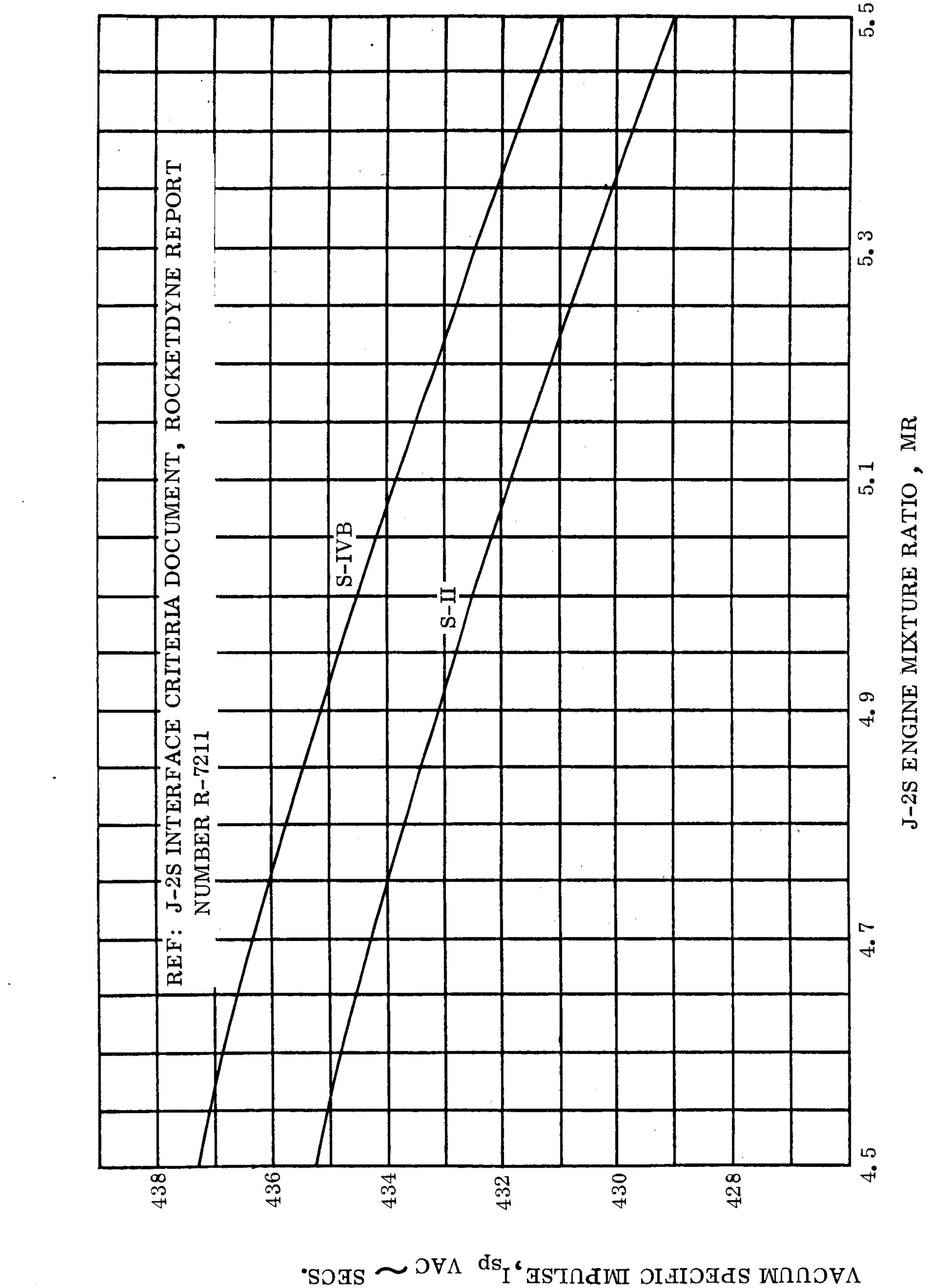


FIGURE 7.2-2 J-2S ENGINE SPECIFIC IMPULSE SIMULATED FOR THE S-II AND S-IVB STAGES

1. BOOST TO EARTH ORBIT - S-IC, S-II, AND S-IVB OPERATION
2. COAST IN EARTH PARKING ORBIT
3. S-IVB TRANSLUNAR INJECTION BOOST
4. TRANSLUNAR COAST
5. LUNAR PASSAGE
6. "FREE RETURN" TRANSEARTH COAST
7. EARTH ATMOSPHERE REENTRY
8. SPLASH

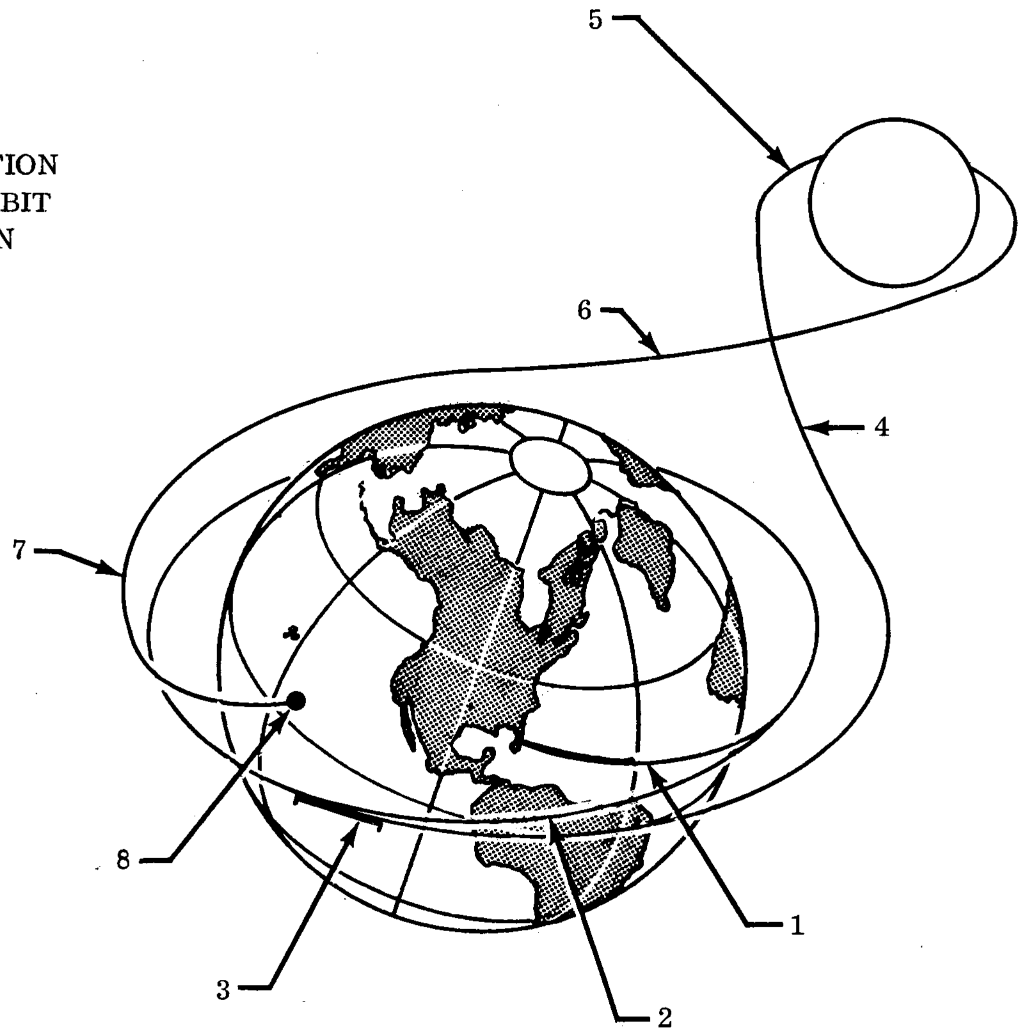


FIGURE 7.2-3 LOR DESIGN FLIGHT PROFILE

7.2.1 (Continued)

A "chi freeze" (vehicle pitch attitude freeze) constraint was initiated 3.5 seconds after S-IC center engine cutoff and continued until Launch Escape System (LES) jettison. "Chi freeze" was used to maintain vehicle attitude during S-IC/S-II first and second plane separation, and LES jettison. The S-IC/S-II large interstage (second plane separation) was jettisoned at 193.986 seconds after lift-off. The LES is jettisoned five seconds later at which time COV guidance is initiated. The S-II stage is cut off at LOX depletion (198.889 seconds). S-II stage cutoff, S-II/S-IVB separation and S-IVB stage ignition occurs simultaneously in the simulation. The S-IVB inserts the vehicle into a 100 NM (185.2 km) circular parking orbit. The three revolution parking orbit coast is not simulated but the nominal trajectory reflects the parking orbit weight loss of 4,446 pounds by an instantaneous weight drop at parking orbit insertion. The S-IVB stage restart in the idle mode is simultaneous with parking orbit insertion. After 100 seconds burn in the idle mode, the S-IVB switches to full thrust and burns to translunar energy cutoff.

The Flight Event History showing the sequence of events during the boost phase is given in Table 7.2-I. The Mission Design Characteristics and Weight Summary containing propulsion data, propellant weights, drop weights, the aerodynamic heating indicator, and other pertinent trajectory data are given in Table 7.2-II. Selected trajectory design data used in determination of the vehicle environments are shown in Figure 7.2-4.

7.2.2 Three-Stage Equatorial Synchronous Orbit Mission

The J-2S/Saturn V vehicle with an Apollo payload shape is used to transport men and material to an equatorial (0^{o}) synchronous orbit. The synchronous orbit design flight profile is shown in Figure 7.2-5.

The launch vehicle phases of the Equatorial Synchronous Orbit flight profile are as follows:

a. Boost with three stages to 100 NM Earth parking orbit.
b. Five revolution coast in Earth parking orbit.
c. S-IVB boost to transfer ellipse.
d. Coast in transfer ellipse.
e. S-IVB boost to Equatorial Synchronous Orbit Injection.

The simulated vehicle is launched along a 90^{o} launch azimuth to minimize parking orbit inclination. The maximum S-IVB propellant utilization was obtained by simulating the S-IVB thrust at the 5.5 mixture ratio level. No "chi freeze" was simulated during the boost to parking orbit. With these exceptions, the boost to parking orbit for the synchronous orbit mission is similar to the LOR boost to parking orbit. Maximum dynamic pressure (q) of 723 pounds/foot^2 is encountered at

TABLE 7.2-I

FLIGHT EVENT HISTORY FOR THE NOMINAL LOR DESIGN TRAJECTORY (J-2S/SATURN V)

TIME (SECS)	EVENT
0	Liftoff
12	Initiate tilt program
35	Terminate tilt program, begin gravity turn
149.486	Inboard engine cutoff
153.0	Initiate chi freeze
161.486	Outboard engine cutoff, S-IC separation
165.286	S-II ignition at a MR = 5.0
167.786	Shift MR to 5.5
193.986	Jettison S-IC/S-II large interstage
198.986	Jettison LES, initiate COV
396.451	Shift MR to 4.7
498.889	S-II cutoff and separation, S-IVB ignition at a MR = 5.0
617.323	Parking Orbit Injection, weight drop, S-IVB reignition in idle mode
717.323	Terminate idle mode, initiate mainstage
1008.046	S-IVB second cutoff, begin LOR transfer

TABLE 7.2-II LOR MISSION DESIGN CHARACTERISTICS AND WEIGHT SUMMARY (J-2S/SATURN V)

S-IC STAGE		
Sea Level Thrust	lbs	7,610,064
Sea Level Specific Impulse	secs	264.5
Liftoff Weight	lbs	6,414,030
Impulse Propellant	lbs	4,577,113
Drop Weight	lbs	370,591
T/W_o		1.186
S-II STAGE		
Vacuum Thrust, M.R. = 5.0	lbs	1,182,035
Vacuum Specific Impulse	secs	432.5
Weight at Ignition	lbs	1,466,326
Impulse Propellant	lbs	970,441
Forward Interstage Jettison Weight	lbs	9,427
Launch Escape System	lbs	8,936
Drop Weight	lbs	102,571
S-IVB STAGE		
Vacuum Thrust, M.R. = 5.0	lbs	237,500
Vacuum Specific Impulse	secs	434.5
Weight at First Ignition	lbs	374,952
Impulse Propellant Consumed to Orbit	lbs	64,737
Cutoff Weight in Orbit	lbs	310,215
Weight Drop in Parking Orbit	lbs	4,446
Weight at Ignition in Orbit	lbs	305,769
Impulse Propellant Consumed to Injection	lbs	160,696
Total Impulse Propellant	lbs	225,433
Cutoff Weight at Injection	lbs	145,073
Drop Weight	lbs	27,160
GROSS PAYLOAD	lbs	117,913
Instrument Unit Weight	lbs	4,183
Propellant Reserves (109 m/sec)	lbs	3,664
NET PAYLOAD	lbs	110,066

TABLE 7.2-II LOR MISSION DESIGN CHARACTERISTICS AND WEIGHT SUMMARY (Continued)

FLIGHT TRAJECTORY DATA

Launch Azimuth	degs	72
Max q (85 secs)	lbs/ft^2	709
Max "g's"		4.14
AHI	$ft\text{-}lbs/ft^2\text{-}rad$	49.09×10^6

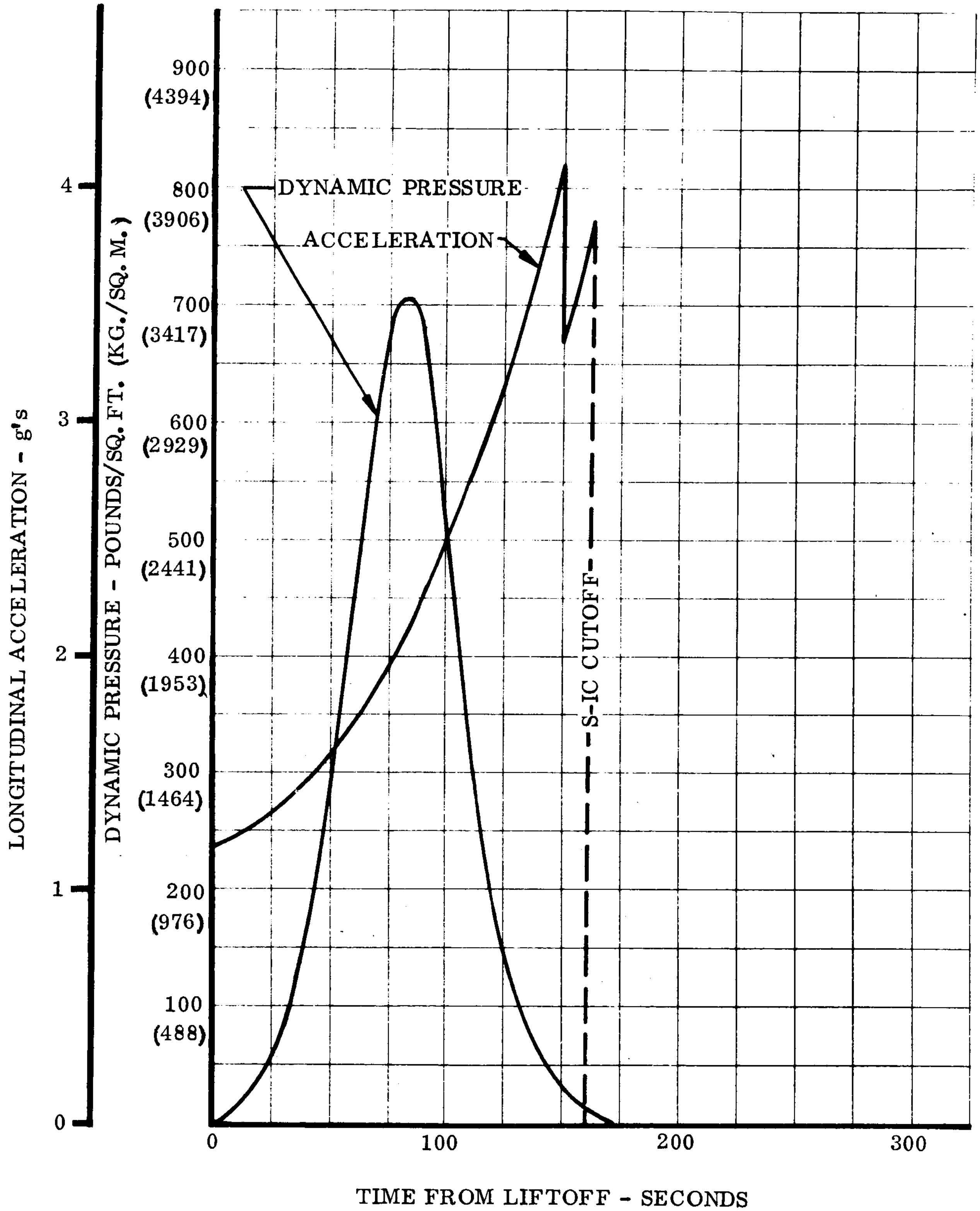


FIGURE 7.2-4 J-2S/SATURN V THREE STAGE - LOR DYNAMIC PRESSURE AND LONGITUDINAL ACCELERATION VERSUS TIME

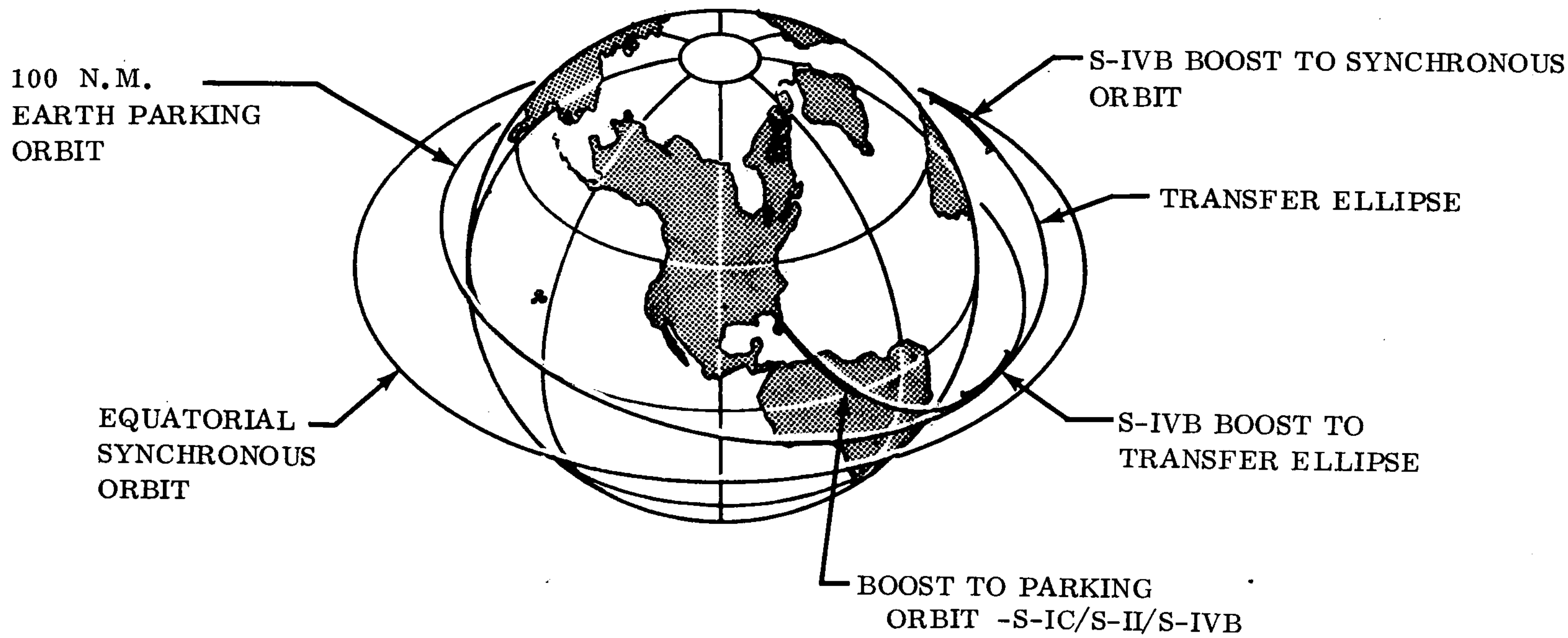


FIGURE 7.2-5 EQUATORIAL SYNCHRONOUS ORBIT DESIGN PROFILE

7.2.2 (Continued)

84 seconds after liftoff. Maximum longitudinal acceleration of 4.23 g's occurs at S-IC center engine cut off (149.5 sec.).

Five revolutions in the 100 NM Earth parking orbit (approximately 7.5 hours) was simulated. The S-IVB stage first restart with a 100 second idle mode burn is initiated at the sixth descending node of the Earth parking orbit. A plane change of 2.1 degrees and insertion into a Hohmann transfer orbit is accomplished with this S-IVB burn. The vehicle coast to synchronous altitude via Hohmann transfer orbit (approximately 5.3 hours) was simulated. At the synchronous altitude, the S-IVB stage was restarted for the second time with a 100 second idle mode burn. This S-IVB third burn is used to circularize the orbit and to perform the plane change required for an equatorial (0^{o}) orbit.

The Flight Event History showing the sequence of events during the boost phase is given in Table 7.2-III. The Mission Design Characteristics and Weight Summary containing propulsion data, propellant weights, drop weights, the aerodynamic heating indicator, and other pertinent trajectory data are given in Table 7.2-IV. Selected trajectory design data used in determination of the vehicle environments are shown in Figure 7.2-6.

7.2.3 Two-Stage Low Earth Orbit Mission (Coplanar)

The two-stage configuration of the J-2S/Saturn V vehicle (S-IC/S-II) is used for the Low Earth Orbit (LEO) mission to transport men and material to 300 N. M. circular Earth orbit. The two-stage vehicle uses a payload envelope having the external dimensions of the S-IVB stage and the Apollo payload. This results in the vehicle having the same external geometry as the LOR vehicle. The LEO design flight profile is shown in Figure 7.2-7.

The launch vehicle phases of the LEO design flight profile are as follows:

a. Boost with S-IC/S-II to perigee of 100 N. M.
b. Coast of S-II/payload to 300 N. M. apogee of the elliptical orbit.
c. S-II start to idle mode for orbit circularization at 300 N. M. altitude.

The vehicle is launched from Cape Kennedy on a 108^{o} launch azimuth. No "chi freeze" was simulated during the boost phase. The boost phase of the S-IC and S-II was similar to the LOR boost phase except for S-II insertion into the 100 N. M. perigee of the Hohmann transfer ellipse. Maximum dynamic pressure (q) of 746 pounds/foot^2 occurs at 83 seconds after lift-off. Maximum longitudinal acceleration of 4.41 g's occurs at S-IC center engine cutoff (149 sec.).

TABLE 7.2-III FLIGHT EVENT HISTORY FOR THE EQUATORIAL (0^{o}) SYNCHRONOUS ORBIT DESIGN MISSION (J-2S/SATURN V)

TIME (SECS)	EVENT
0	Liftoff
12	Initiate tilt program
35	Terminate tilt program, begin gravity turn
149.486	Inboard engine cutoff
161.486	Outboard engine cutoff, S-IC separation
165.286	S-II ignition at a 5.0 M.R.
167.786	Shift to 5.5 M.R.
193.986	Jettison S-IC/S-II large interstage
198.986	Jettison LES
396.451	Shift to 4.7 M.R.
498.889	S-II cutoff and separation, S-IVB ignition at a 5.5 M.R.
545.886	P.O. injection and weight drop, begin five orbit coast
27737.000	Initiate 100 second idle mode burn
27837.000	End idle mode, begin mainstage at a 5.5 M.R.
28046.086	S-IVB second cutoff, begin transfer to synchronous orbit altitude
46830.000	Weight drop, and initiate 100 second idle mode
46930.000	End idle mode, begin mainstage at a 5.5 M.R.
47017.093	S-IVB third cutoff, payload in equatorial synchronous orbit

TABLE 7.2-IV SYNCHRONOUS ORBIT MISSION DESIGN CHARACTERISTICS AND WEIGHT SUMMARY (J-2S/SATURN V)

S-IC STAGE		
Sea Level Thrust	lbs	7,610,064
Sea Level Specific Impulse	secs	264.5
Liftoff Weight	lbs	6,367,342
Impulse Propellant	lbs	4,577,113
Drop Weight	lbs	370,591
T/W_0		1.20
S-II STAGE		
Vacuum Thrust, M.R. = 5.0	lbs	1,182,035
Vacuum Specific Impulse	secs	432.5
Weight at Ignition	lbs	1,419,637
Impulse Propellant	lbs	970,441
Forward Interstage Jettison Weight	lbs	9,427
Launch Escape System	lbs	8,936
Drop Weight	lbs	102,614
S-IVB STAGE		
Vacuum Thrust, M.R. = 5.5	lbs	265,000
Vacuum Specific Impulse	secs	431
Weight at First Ignition	lbs	328,220
Impulse Propellant Consumed to P.O.	lbs	28,896
Cutoff Weight in P.O.	lbs	299,324
Weight at Second S-IVB Start (I.M.)	lbs	292,531
Impulse Propellant Consumed to Injection	lbs	130,342
Weight at Second S-IVB Cutoff	lbs	162,188
Weight at Third S-IVB Start (I.M.)	lbs	158,942
Total Impulse Propellant for Final Burn	lbs	55,335
Cutoff Weight in Orbit	lbs	103,607
Total S-IVB Impulse Propellant (No Reserves)	lbs	214,573
Drop Weight	lbs	28,676
GROSS PAYLOAD	lbs	74,931
Instrument Unit Weight	lbs	4,183
Propellant Reserves	lbs	4,757

TABLE 7.2-IV SYNCHRONOUS ORBIT MISSION DESIGN CHARACTERISTICS AND WEIGHT SUMMARY (Continued)

NET PAYLOAD	lbs	65,991
FLIGHT TRAJECTORY DATA		
Launch Azimuth	degs	90^{o}
Max "q" (84 secs)	lbs/ft^2	723
Max "g's"		4.23
AHI	$ft\text{-}lbs/ft^2\text{-}rad$	50.6×10^6

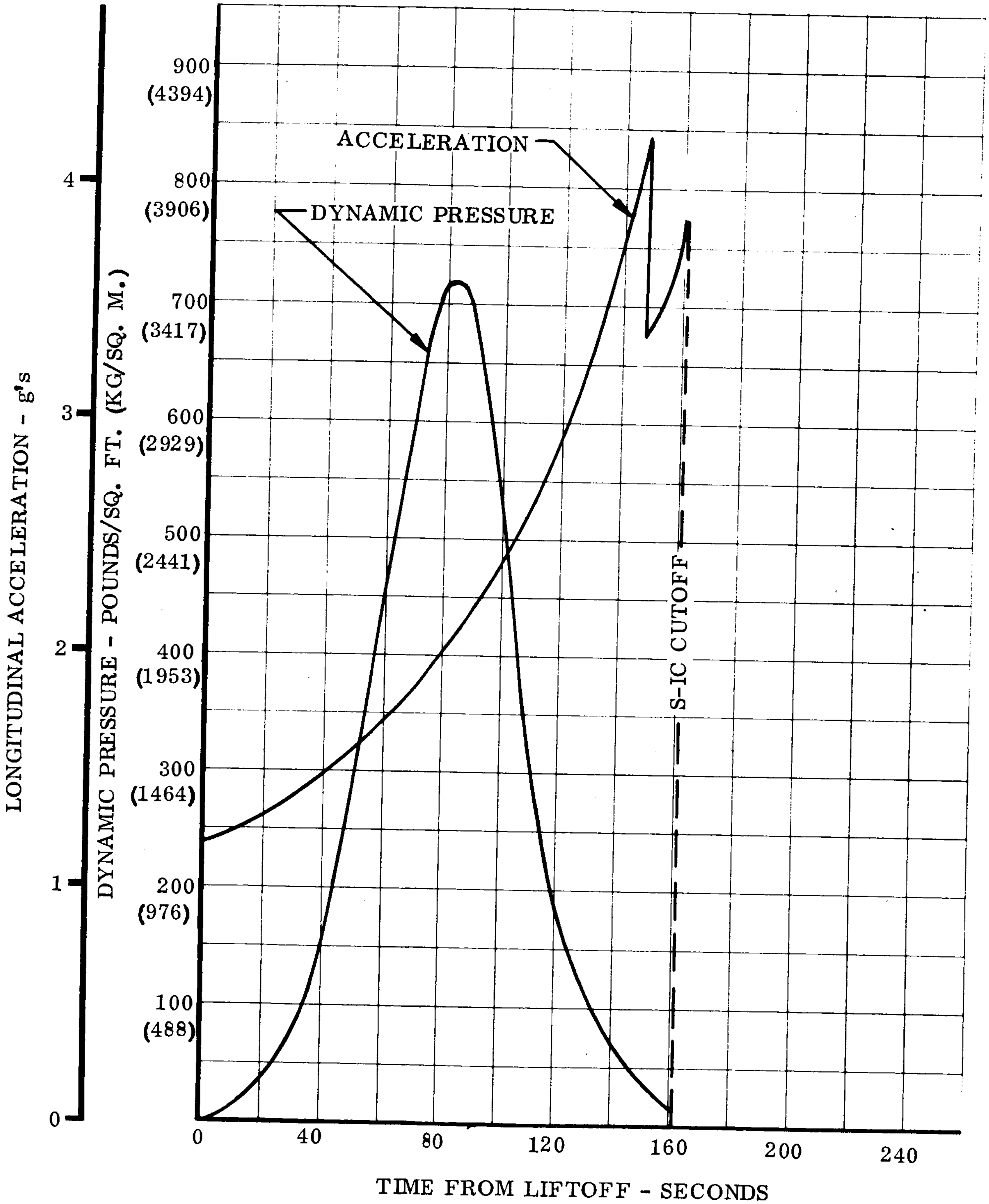
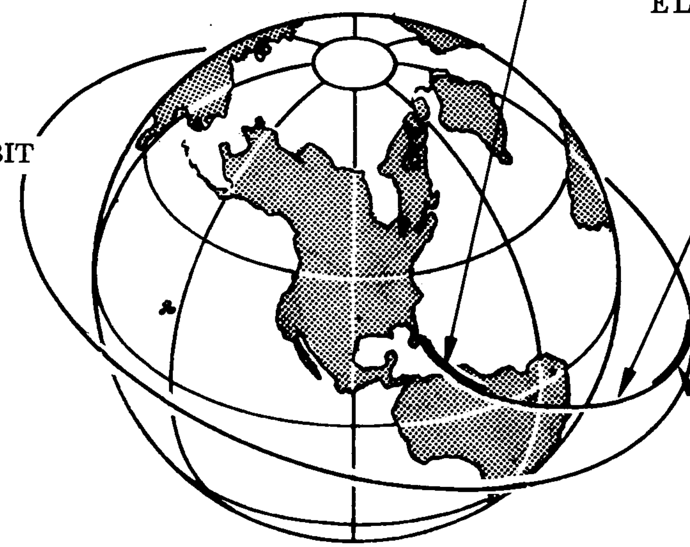


FIGURE 7.2-6 J-2S/SATURN V THREE-STAGE - SYNCHRONOUS MISSION DYNAMIC PRESSURE AND LONGITUDINAL ACCELERATION VERSUS TIME



NOTE: COPLANAR FLIGHT

FIGURE 7.2-7 LEO DESIGN FLIGHT PROFILE

7.2.3 (Continued)

The vehicle coast of approximately 45 minutes in the elliptical orbit was simulated. At the 300 NM apogee, the S-II was started to idle mode for injection into a 300 NM circular orbit. Final orbit inclination is approximately 34°.

The Flight Event History showing the sequence of events during the boost phase is given in Table 7.2-V. The Mission Design Characteristics and Weight Summary containing propulsion data, propellant weights, drop weights, the aerodynamic heating indicator, and other pertinent trajectory data are given in Table 7.2-VI. Selected trajectory design data used in determination of the vehicle environments are shown in Figure 7.2-8.

7.2.4 Two-Stage Polar Orbit Mission

The two-stage configuration of the J-2S/Saturn V vehicle (S-IC/S-II) is used to transport men and material to polar orbit. The two-stage vehicle uses a payload having the external dimensions of the S-IVB stage and the Apollo payload. This results in the vehicle having the same external geometry as the LOR vehicle. The two-stage polar orbit design flight profile is shown in Figure 7.2-9.

The launch vehicle phases of the two-stage polar orbit design flight profile are as follows:

a. Direct injection with S-IC/S-II to 100 N M polar orbit with overflight of Cuba and the Isthmus of Panama.
b. Yaw steering of S-IC initiated after max q.
c. Yaw steering continuous through S-II burn.
d. 100 NM polar orbit insertion during S-II first burn.

The simulated vehicle was launched along a 140 degree launch azimuth and used direct injection boost of both S-IC and S-II to 100 NM polar orbit. The boost phase of the S-IC and S-II was similar to the LOR boost phase except for the yaw maneuver and insertion into the polar orbit with the S-II.

During the boost phase, the launch vehicle executes a yaw maneuver to avoid stage impact on major land masses. Yaw steering was initiated 100 seconds after lift-off. A yaw rate of 1 degree/second was used during S-IC boost. A constant yaw angle of 44.9 degrees was used during S-II boost. The yaw maneuver causes the vehicle to swing around the east coast of Florida and across mid-Cuba. The vehicle passes over the Isthmus of Panama and flies around the west coast of South America. Figure 7.2-10 is a map showing the "instantaneous stage impact" trace. The trajectory shaping was sufficient to avoid stage impact on all major land masses. Maximum dynamic pressure (q) of 698 pounds/foot^2 occurs at 79.2 seconds after lift-off. Maximum

TABLE 7.2-V FLIGHT EVENT HISTORY FOR THE TWO-STAGE LOW EARTH ORBIT (LEO) DESIGN MISSION (J-2S/SATURN V)

TIME (SECS)	EVENT
0	Liftoff
12	Initiate tilt
35	Terminate tilt program begin gravity turn
149.486	Inboard engine cutoff
161.486	Outboard engine cutoff, S-IC separation
165.286	S-II ignition at a 5.0 M.R.
167.786	Shift to 5.5 M.R.
193.986	Jettison S-IC/S-II large interstage
198.986	Jettison LES
416.651	Shift M.R. to 4.7
484.038	Shift from mainstage to idle mode
489.038	Idle mode cutoff, begin transfer orbit
3165.000	Initiate idle mode
3305.858	Idle mode cutoff, injection into final orbit

TABLE 7.2-VI TWO-STAGE LEO MISSION DESIGN CHARACTERISTICS AND WEIGHT SUMMARY (J-2S/SATURN V)

S-IC STAGE		
Sea Level Thrust	lbs	7,610,064
Sea Level Specific Impulse	secs	264.5
Liftoff Weight	lbs	6,282,348
Impulse Propellant	lbs	4,577,113
Drop Weight	lbs	370,591
T/W_0		1.21
S-II STAGE		
Vacuum Thrust, M.R. = 5.0	lbs	1,182,035
Vacuum Specific Impulse	secs	432.5
Weight at Ignition	lbs	1,334,644
Propellant Consumed During Mainstage	lbs	943,358
Forward Interstage Jettison Weight	lbs	9,427
Launch Escape System	lbs	8,936
Propellant Consumed During 5 Sec. IM	lbs	446
Propellant Consumed During Final IM	lbs	12,577
Total Impulse Propellant	lbs	970,122
LH_2 Boiloff in Transfer Orbit	lbs	319
Drop Weight	lbs	93,987
GROSS PAYLOAD	lbs	265,593
Instrument Unit Weight	lbs	4,183
Propellant Reserves		
. S-II Mainstage	lbs	5,602
. Apogee burn (I.M.) (for 1^o plane change)	lbs	8,585
NET PAYLOAD	lbs	247,224
FLIGHT TRAJECTORY DATA		
Launch Azimuth	degs	108
Max "q" (83 secs)	lbs/ft^2	746
Max "g's"		4.41
AHI	$ft\text{-}lbs/ft^2\text{-}rad$	53.3×10^6

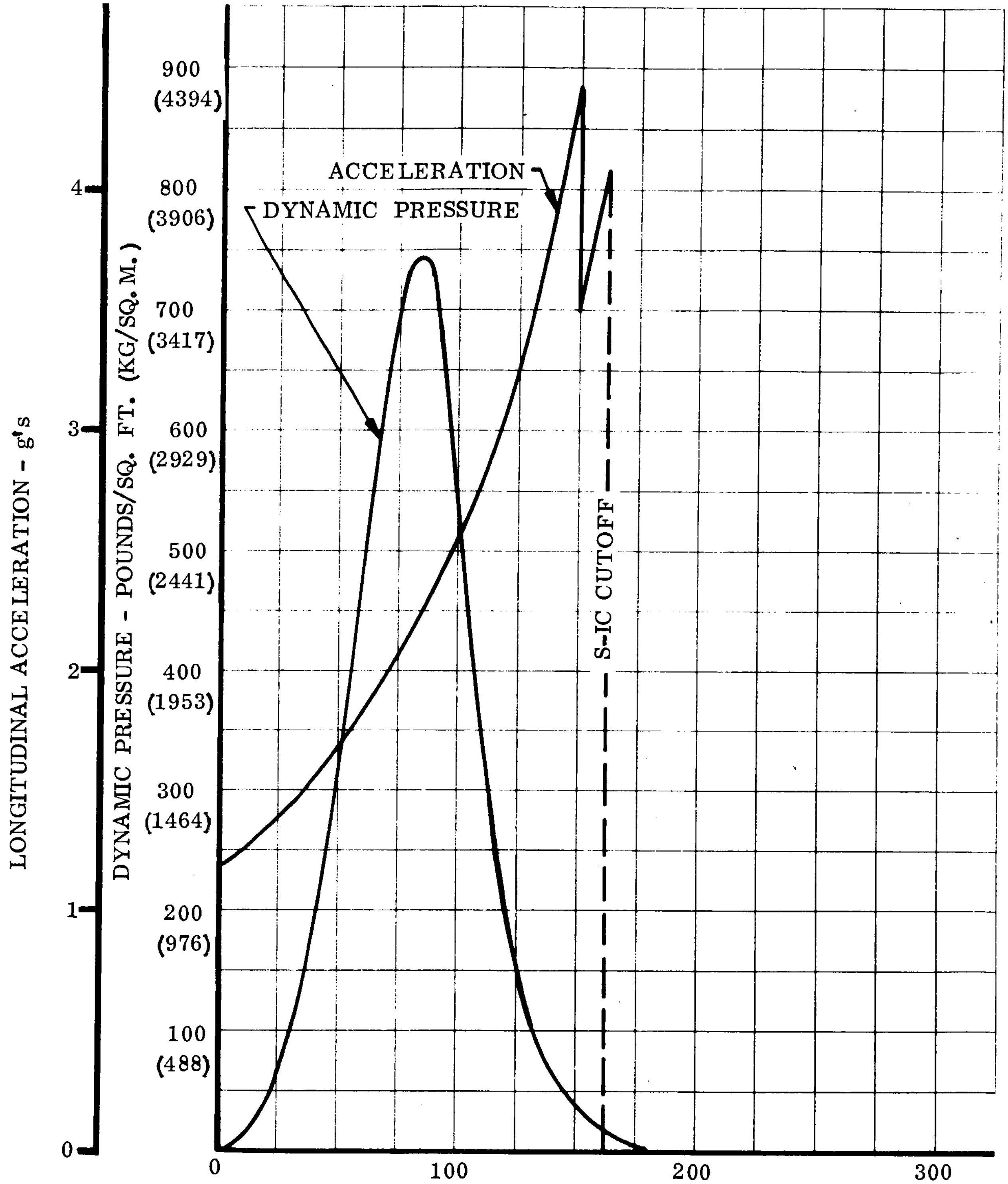


FIGURE 7.2-8 J-2S/SATURN V TWO-STAGE LEO DYNAMIC PRESSURE AND LONGITUDINAL ACCELERATION VERSUS TIME

FIGURE 7.2-9 TWO-STAGE POLAR ORBIT FLIGHT PROFILE

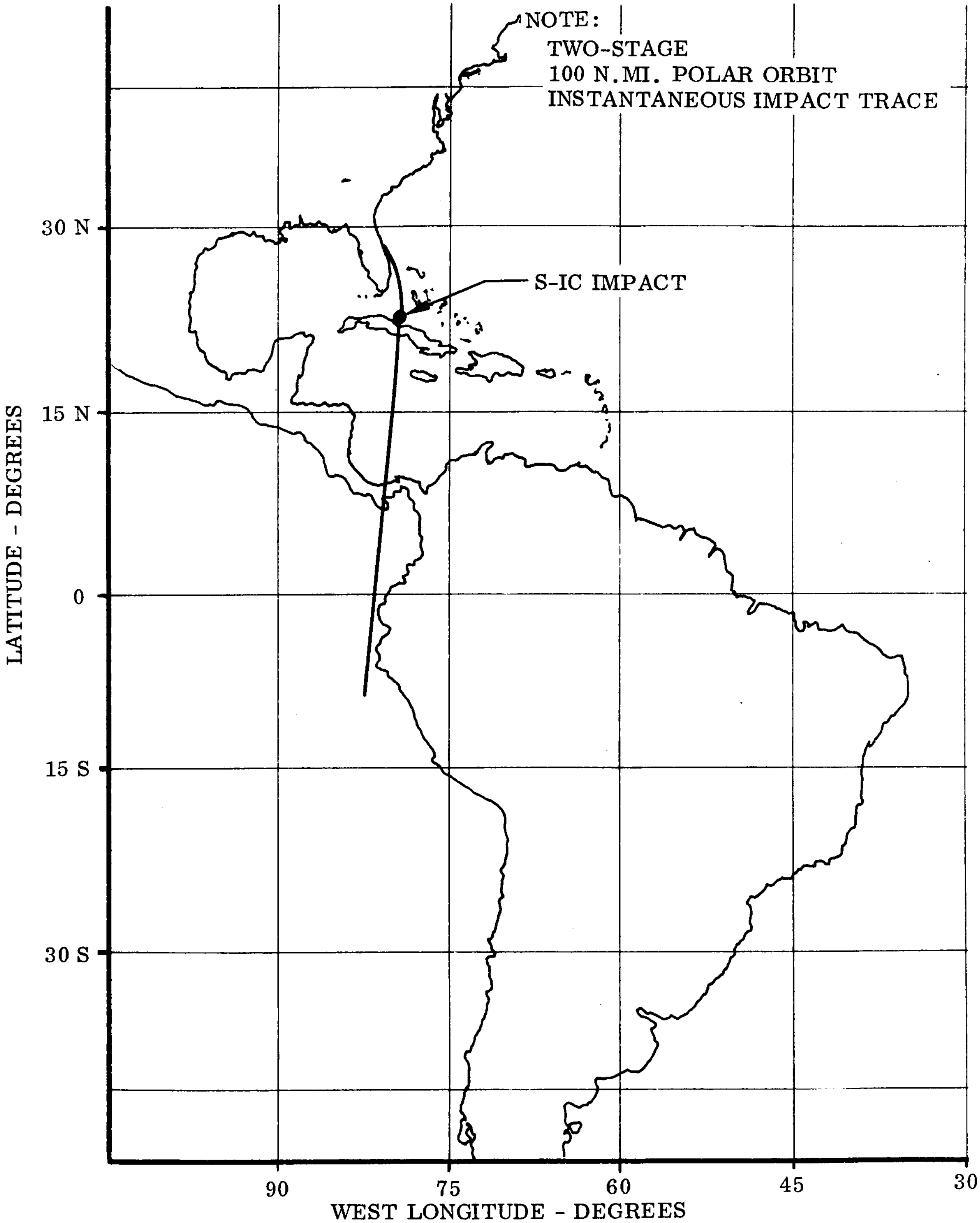


FIGURE 7.2-10 POLAR ORBIT MISSION - INSTANTANEOUS IMPACT TRACE

7.2.4 (Continued)

longitudinal acceleration of 4.5 g's occurs at S-IC center engine cutoff (149.5 seconds).

The Flight Event History showing the sequence of events during the boost phase is given in Table 7.2-VII. The Mission Design Characteristics and Weight Summary containing propulsion data, propellant weights, drop weights, the aerodynamic heating indicator, and other pertinent trajectory data are given in Table 7.2-VIII. Selected trajectory design data used in determination of the vehicle environments are shown in Figure 7.2-11.

TABLE 7.2-VII FLIGHT EVENT HISTORY
DESIGN TWO STAGE POLAR - NO CHI FREEZE (J-2S/SATURN V)

TIME (SEC.)	FLIGHT EVENT
0	Liftoff
12	Initiate Pitch Program
35	Terminate Tilt Program
100	Initiate Yaw (yaw rate = 1^{o}/sec)
149.486	Inboard Engine
153	Terminate S-IC Stage Yaw Rate
161.486	Outboard Engine Cutoff, S-IC Separation
165.286	S-II Ignition at MR = 5.0, Set Thrust Vector Angle in Yaw Plane to 44.9^{o}
167.786	Shift MR to 5.5
193.986	Jettison S-IC/S-II Large Interstage
198.986	Jettison LES
396.451	Shift MR to 4.7
497.220	S-II Cutoff in 100 N. M. Polar Orbit (90^{o} orbit inclination)

TABLE 7.2-VIII TWO STAGE POLAR MISSION DESIGN CHARACTERISTICS AND WEIGHT SUMMARY NO X FREEZE (J-2S/SATURN V)

S-IC STAGE		
Sea Level Thrust	lbs	7,610,064
Sea Level Specific Impulse	secs	264.5
Liftoff Weight	lbs	6,250,497
Impulse Propellant	lbs	4,577,113
Drop Weight	lbs	370,591
T/W_o		1.217
S-II STAGE		
Vacuum Thrust, M.R. = 5.0	lbs	1,182,035
Vacuum Specific Impulse	secs	432.5
Weight at Ignition	lbs	1,302,792
Propellant Consumed During Mainstage	lbs	965,018
Forward Interstage Jettison Weight	lbs	9,427
Launch Escape System	lbs	8,936
Total Impulse Propellant	lbs	970,441
Drop Weight	lbs	93,987
GROSS PAYLOAD		
Instrument Unit Weight	lbs	4,183
Propellant Reserves	lbs	5,424
NET PAYLOAD	lbs	215,818
FLIGHT TRAJECTORY DATA		
Launch Azimuth	degs	140
Max "q" (79.2 sec)	lbs/ft^2	698
Max "g's"		4.49
AHI	$ft\text{-}lbs/ft^2 - rad$	45.66×10^6

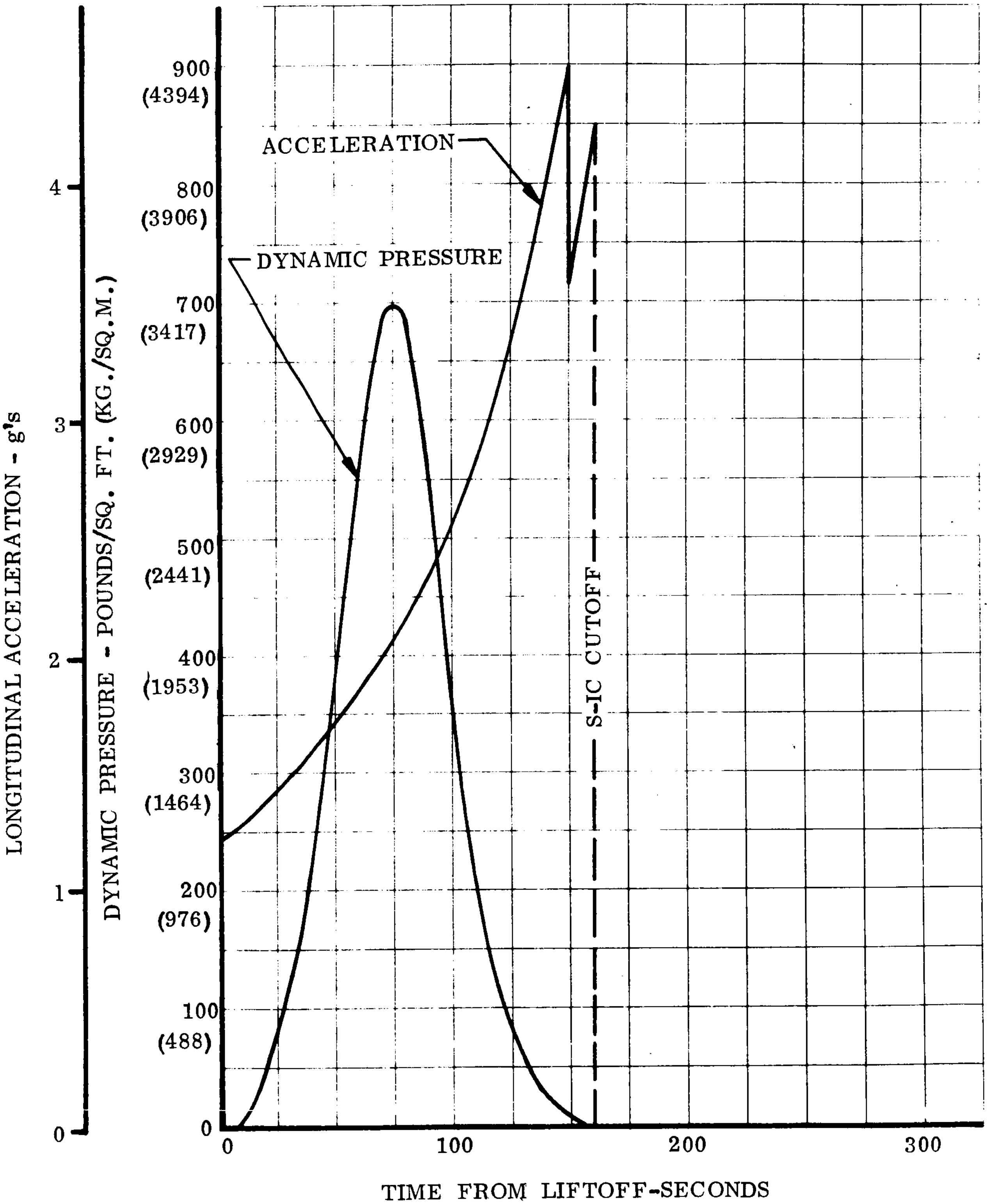


FIGURE 7.2-11 J-2S/SATURN V TWO-STAGE POLAR ORBIT DYNAMIC PRESSURE AND LONGITUDINAL ACCELERATION VERSUS TIME

7.3 PRELIMINARY VEHICLE WEIGHT AND MASS CHARACTERISTICS

Distributed weights for the LOR. LEO, and two-stage polar vehicles are shown in Tables 7.3-I through 7.3-IV. The distributed pitch moments of inertia for the various masses are also shown. These mass characteristics were used to develop the structural loads and dynamics data.

Accumulative weights at lift-off are shown in Table 7.3-V for the LOR, LEO and two-stage polar vehicles. These mass properties were used as input data for the structural design loads computer program. Based on S-IC stage propellant consumption rates, accumulative weights for other first stage flight times were computed by the loads program, as required.

Figure 7.3-1 shows the propellant surface levels for each stage of the design vehicles. This information was also an input to the ground and flight structural design load computations.

Figures 7.3-2 through 7.3-7 provide the weight, center of gravity, and the pitch and roll moments of inertia for the LOR, LEO and two-stage polar vehicles during first stage flight. These mass characteristics were based on preliminary weight estimates supplied by the associate stage contractors. Final weights were compared with the preliminary weights to determine the validity of results based on the preliminary weights. The minor differences in weights do not invalidate any study results.

Drop weights used for the design trajectories are shown in the Mission Design Characteristics and Weight Summary Tables, Paragraph 7.2.

TABLE 7.3-I S-IC STAGE DISTRIBUTED WEIGHT LOR, LEO AND 2-STAGE POLAR VEHICLES

AFT BAY STATION (INCHES)	FORWARD BAY STATION (INCHES)	INERT BAY MASS (KILOGRAMS)	MAINSTAGE PROPELLANT MASS (KILOGRAMS)		BAY CG STATION (METERS) LIFTOFF	BAY CG STATION (METERS) CUTOFF	I_p About Point of Attachment (LIFTOFF) ($KGF\text{-}M\text{-}SEC^2$)	I_p About Point of Attachment (CUTOFF) ($KGF\text{-}M\text{-}SEC^2$)
			OXIDIZER	FUEL				
Inert Mass Cantilevered								
from Station	100.0	43,535.7			1.33	1.33	62.760	62.760
-120.0	220.0	36,340.0			3.80			
220.0	365.0	13,956.7			6.92			
Inert Mass + RP-1								
Cantilevered from Station	365.0	11,245.2		135,570.7	7.96	7.72	34.076	3.940
365.0	420.0	2,479.0		86,911.9	9.97			
420.0	470.0	2,388.8		79,416.0	11.30			
470.0	520.0	2,165.9		78,723.1	12.57			
520.0	560.0	1,844.4		63,382.0	13.72			
560.0	602.0	3,812.9	10,331.8	66,743.6	14.37			
Inert Mass Cantilevered								
from Station	602.0	1,924.0			16.81	16.81	2.475	2.475
602.0	660.0	3,032.0		114,662.8	16.28			
660.0	912.0	8,646.0			19.42			
Inert Mass + LOX								
Cantilevered from Station	912.0	4,511.7	209,502.7		21.79	21.31	45.530	4.720
912.0	960.0	1,289.8	109,101.3		23.77			
960.0	1010.0	1,850.6	112,636.1		25.02			
1010.0	1060.0	1,669.6	112,410.1		26.29			
1060.0	1110.0	1,554.0	112,568.0		27.56			
1110.0	1156.5	1,562.7	103,713.9		28.78			
1156.5	1200.0	1,069.6	100,048.0		29.91			
1200.0	1250.0	1,351.6	113,814.7		31.13			
1250.0	1300.0	1,161.7	113,554.4		32.37			
1300.0	1401.0	4,204.7	229,539.7		34.33			
Inert Mass Cantilevered								
from Station	1401.0	2,177.0			37.13	37.13	3.160	3.160
1401.0	1461.0	2,714.0	121,879.6		36.32			
1461.0	1541.0	2,163.0			38.12			
1541.0	1650.0	3,733.4			40.65			

TABLE 7.3-II S-II STAGE, S-IVB STAGE AND PAYLOAD DISTRIBUTED WEIGHTS LOR VEHICLE

AFT BAY STATION (INCHES)	FORWARD BAY STATION (INCHES)	INERT BAY MASS (KILOGRAMS)	MAINSTAGE PROPELLANT MASS (KILOGRAMS) OXIDIZER	MAINSTAGE PROPELLANT MASS (KILOGRAMS) FUEL	BAY CG STATION (METERS) LIFTOFF	BAY CG STATION (METERS) CUTOFF	I_p About Point of Attachment LIFTOFF (KGF-M-SEC^2)	I_p About Point of Attachment CUTOFF (KGF-M-SEC^2)
1650.0	1787.0	3,307.0			43.82			
Inert Mass Cantilevered from Station	1787.0	17,436.0			42.56	42.56	26,366	26,366
1787.0	1848.0	1,870.5			46.22			
Inert Mass + LOX Cantilevered from Station	1848.0	378,770.0**	369,825		46.68	46.42	266,289	8,211
1848.0	1900.0	2,572.2			47.66			
1900.0	1950.0	2,959.9**		782	49.01			
1950.0	2000.0	6,721.5**		5,738	50.23			
2000.0	2050.0	7,858.6**		7,049	51.44			
2050.0	2117.5	10,603.8**		9,457	52.93			
2117.5	2200.0	12,896.1**		11,617	54.83			
2200.0	2387.0	39,768.4**		35,989	58.92			
Inert Mass Cantilevered from Station	2387.0	1,152.0			62.53	62.53	1.423	1,423
2387.0	2519.0	1,505.0			62.44			
2519.0	2746.0	3,479.8			66.44			
2746.0	2832.0	3,044.4**		55	70.72			
Inert Mass + LOX Cantilevered from Station	2832.0	92,521.0**	84,515		70.69	69.27	19,844	8,586
2832.0	3100.5	21,919.2**		18,925	76.05			
Inert Mass Cantilevered from Station	3100.0	525.0			79.83		811	814
3100.5	3222.5	813.0			81.20			
3222.5	3258.5	1,893.7			82.43			
3258.5	3592.0	19,590.0			85.51			
3592.0	3764.0	23,274.0			93.18			
3764.0	3890.0	7,665.0			96.58			
3890.0	4250.0	3,935.0			103.34			

** PROPELLANT WEIGHT INCLUDED IN INERT BAY MASS

TABLE 7.3-III S-II STAGE AND PAYLOAD DISTRIBUTED WEIGHTS LEO VEHICLE

AFT BAY STATION (INCHES)	FORWARD BAY STATION (INCHES)	INERT BAY MASS (KILOGRAMS)	MAINSTAGE PROPELLANT MASS (KILOGRAMS)		BAY CG STATION (METERS) LIFTOFF	BAY CG STATION (METERS) CUTOFF	I_p About Point of Attachment LIFTOFF ($KGF-M-SEC^2$)	I_p About Point of Attachment CUTOFF ($KGF-M-SEC^2$)
			OXIDIZER	FUEL				
1650.0	1787.0	3,307.0			43.82			
Inert Mass Cantilevered from Station	1787.0	17,370.0			42.56	42.56	26,735	26,735
1787.0	1848.0	1,870.5			46.22			
Inert Mass + LOX Cantilevered from Station	1848.0	379,508.0**	369,825		46.68	46.17	267,485	9,389
1848.0	1900.0	2,572.2			47.66			
1900.0	1950.0	2,959.9**		782	49.01			
1950.0	2000.0	6,721.5**		5,738	50.23			
2000.0	2050.0	7,858.6**		7,049	51.44			
2050.0	2117.5	10,603.8**		9,457	52.93			
2117.5	2200.0	12,896.1**		11,617	54.83			
2200.0	2387.0	39,768.4**		36,516	58.92			
Inert Mass Cantilevered from Station	2387.0	1,152.0			62.53	62.53	1,423	1,423
2387.0	2519.0	1,505.0			62.44			
2519.0	2633.0	20,040			65.33			
2633.0	2746.5	13,076			68.22			
2746.5	2875.0	11,857			71.40			
2875.0	3003.0	11,857			74.65			
3003.0	3131.0	11,857			77.90			
3131.0	3258.5	11,857			81.15			
3258.5	3425.5	12,530			84.73			
3425.5	3592.0	7,512			88.93			
3592.0	3890.0	6,778			93.95			
3890.0	4250.0	3,567			103.34			

** PROPELLANT WEIGHT INCLUDED IN INERT BAY MASS

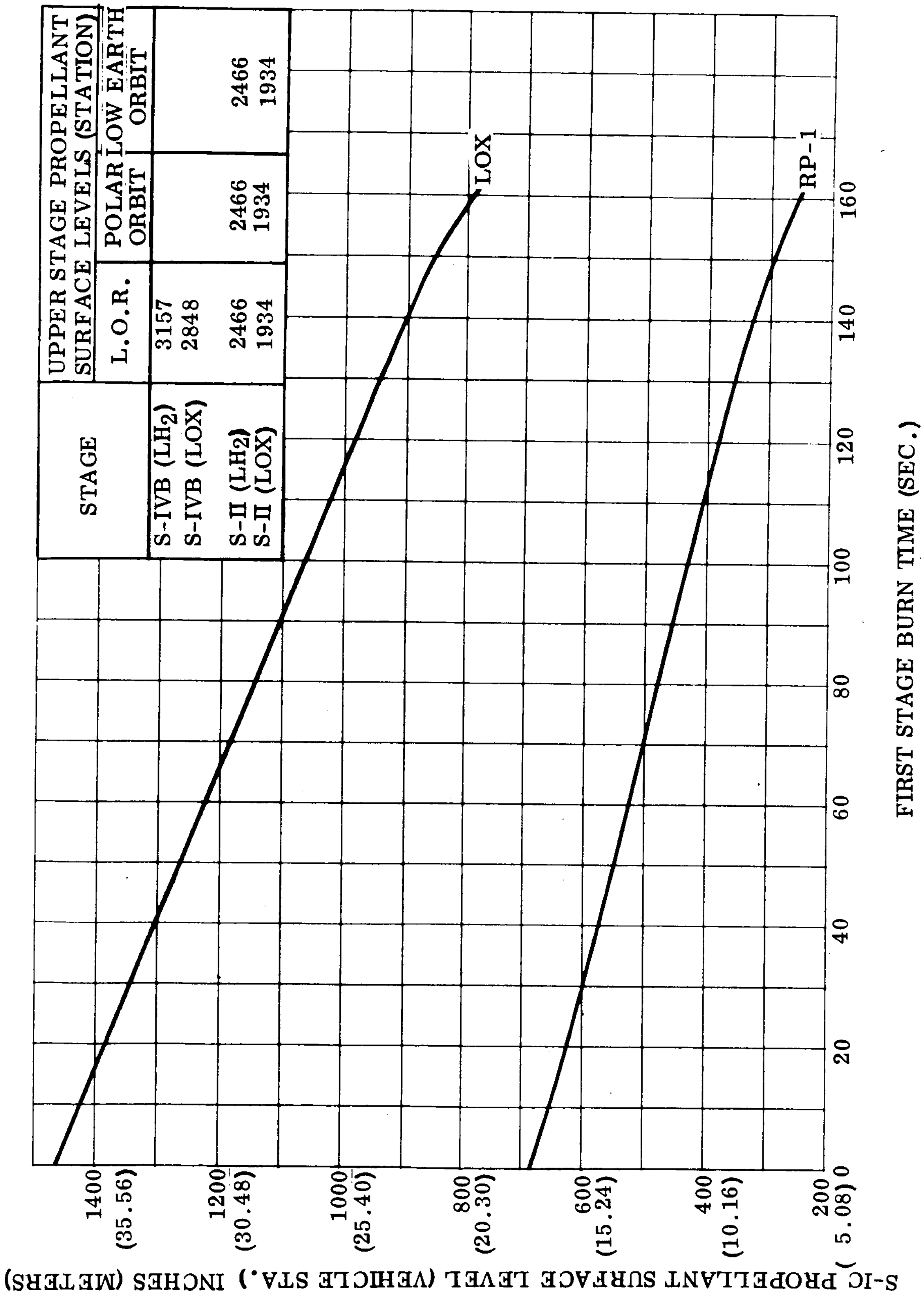
TABLE 7.3-IV S-II STAGE AND PAYLOAD DISTRIBUTED WEIGHTS 2-STAGE POLAR VEHICLE

AFT BAY STATION (INCHES)	FORWARD BAY STATION (INCHES)	INERT BAY MASS (KILOGRAMS)	MAINSTAGE PROPELLANT MASS (KILOGRAMS) OXIDIZER	MAINSTAGE PROPELLANT MASS (KILOGRAMS) FUEL	BAY CG STATION (METERS) LIFTOFF	BAY CG STATION (METERS) CUTOFF	I_p About Point of Attachment LIFTOFF ($KGF\text{-}M\text{-}SEC^2$)	I_p About Point of Attachment CUTOFF ($KGF\text{-}M\text{-}SEC^2$)
1650.0	1787.0	3,307.0			43.82			
Inert Mass Cantilevered from Station	1787.0	17,436.0			42.56	42.56	26,366	26,366
1787.0	1848.0	1,870.5			46.22			
Inert Mass + LOX Cantilevered from Station	1848.0	378,770.0**	369,825		46.68	46.42	266,289	8,211
1848.0	1900.0	2,572.2			47.66			
1900.0	1950.0	2,959.9**		782	49.01			
1950.0	2000.0	6,721.5**		5,738	50.23			
2000.0	2050.0	7,858.6**		7,049	51.44			
2050.0	2117.5	10,603.8**		9,457	52.93			
2117.5	2200.0	12,896.1**		11,617	54.83			
2200.0	2387.0	39,768.4**		35,989	58.92			
Inert Mass Cantilevered from Station	2387.0	1,152.0			62.53	62.53	1.423	1,423
2387.0	2519.0	1,505.0			62.44			
2519.0	2633.0	16,939			65.33			
2633.0	2746.5	11,032			68.22			
2746.5	2875.0	10,000			71.40			
2875.0	3003.0	10,000			74.65			
3003.0	3131.0	10,000			77.90			
3131.0	3258.5	10,000			81.15			
3258.5	3425.5	10,580			84.73			
3425.5	3592.0	6,340			88.93			
3592.0	3890.0	5,730			93.95			
3890.0	4250.0	3,567			103.34			

** PROPELLANT WEIGHT INCLUDED IN INERT BAY MASS

TABLE 7.3-V
VEHICLE ACCUMULATIVE WEIGHTS @ GROUND LIFTOFF

VEHICLE STATION	L.O.R.		2-STAGE POLAR ORBIT		LOW EARTH ORBIT	
	FWD. STA. WEIGHT (KGS)	AFT. STA. WEIGHT (KGS)	FWD. STA. WEIGHT (KGS)	AFT. STA. WEIGHT (KGS)	FWD. STA. WEIGHT (KGS)	AFT. STA. WEIGHT (KGS)
365	2,166,447	2,804,372	2,082,780	2,720,705	2,099,383	2,737,308
602	2,150,313	2,152,263	2,066,646	2,068,596	2,083,249	2,085,199
912	689,546	2,135,025	605,879	2,051,358	622,482	2,067,961
1156.5	681,696	681,696	598,029	598,029	614,632	614,632
1401	673,904	676,094	590,237	592,427	606,840	609,030
1541	670,025	670,025	586,358	586,358	602,961	602,961
1760	663,592	663,592	579,925	579,925	596,528	596,528
1848	192,708	644,318	109,041	560,651	125,644	577,254
2117.5	186,477	186,477	102,810	102,810	119,413	119,413
2387	180,648	181,986	96,981	98,319	113,584	114,922
2519	177,855	177,855	94,188	94,188	110,791	110,791
2746.5	174,411	174,411	66,217	66,217	77,815	77,815
2832	60,692	171,806	59,620	59,620	69,941	69,941
3100.55	57,202	57,695	38,637	38,637	45,116	45,116
3222.55	56,357	56,357	29,027	29,027	33,814	33,814
3258.54	54,464	54,464	26,217	26,217	30,387	30,387
3592.0	34,874	34,874	9,297	9,297	10,345	10,345
3764.0	11,600	11,600	4,787	4,787	5,011	5,011



STAGE	UPPER STAGE PROPELLANT SURFACE LEVELS (STATION)		
	L.O.R.	POLAR ORBIT	LOW EARTH ORBIT
S-IVB (LH_2)	3157		
S-IVB (LOX)	2848		
S-II (LH_2)	2466	2466	2466
S-II (LOX)	1934	1934	1934

FIGURE 7.3-1 LOR/LEO/2-STAGE POLAR ORBIT SATURN V/J-2S PROPELLANT SURFACE LEVELS VS FIRST STAGE BURN TIME

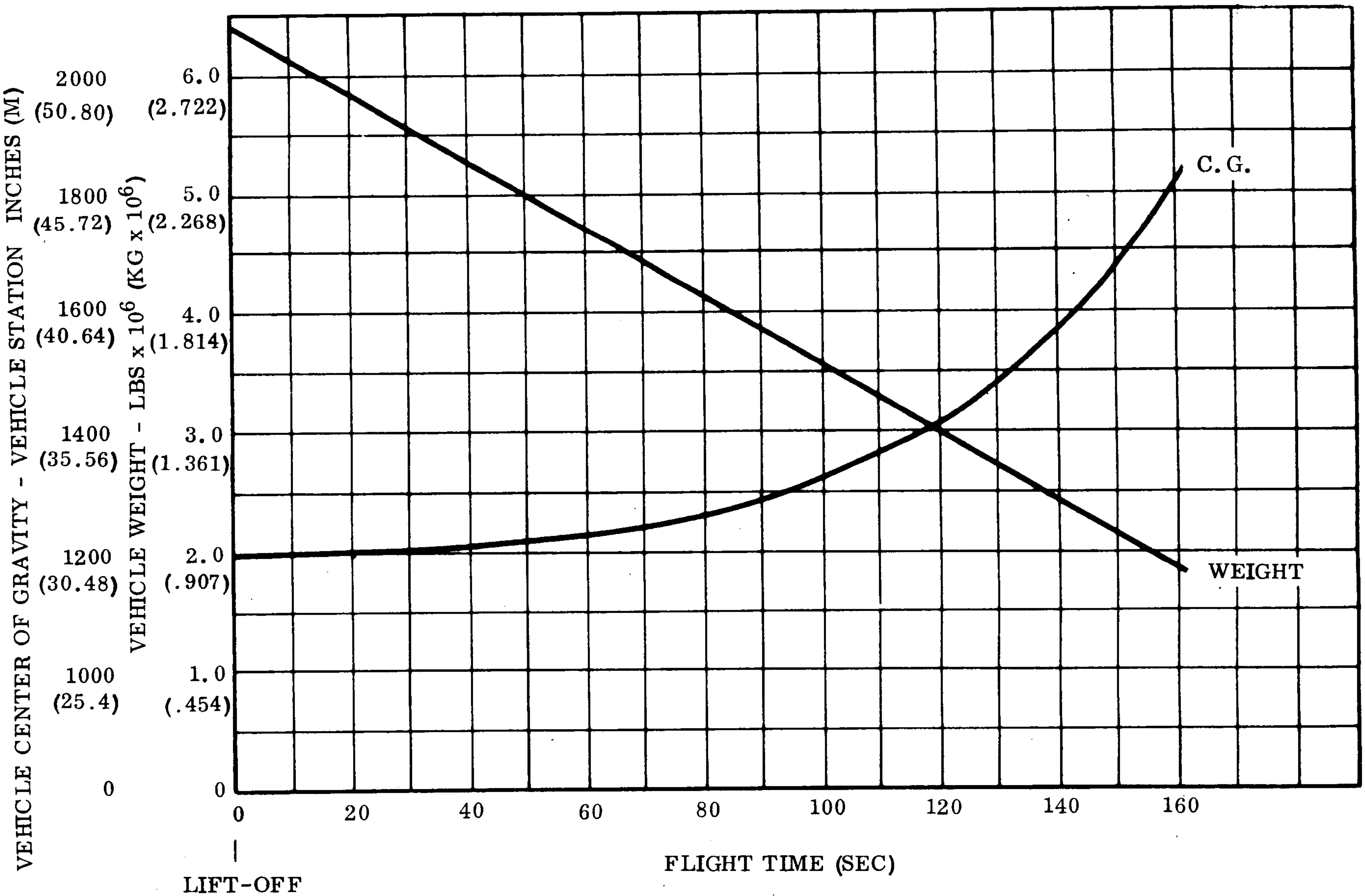


FIGURE 7.3-2 J-2S/SATURN V LOR VEHICLE WEIGHT AND CENTER OF GRAVITY VS. VEHICLE FLIGHT TIME

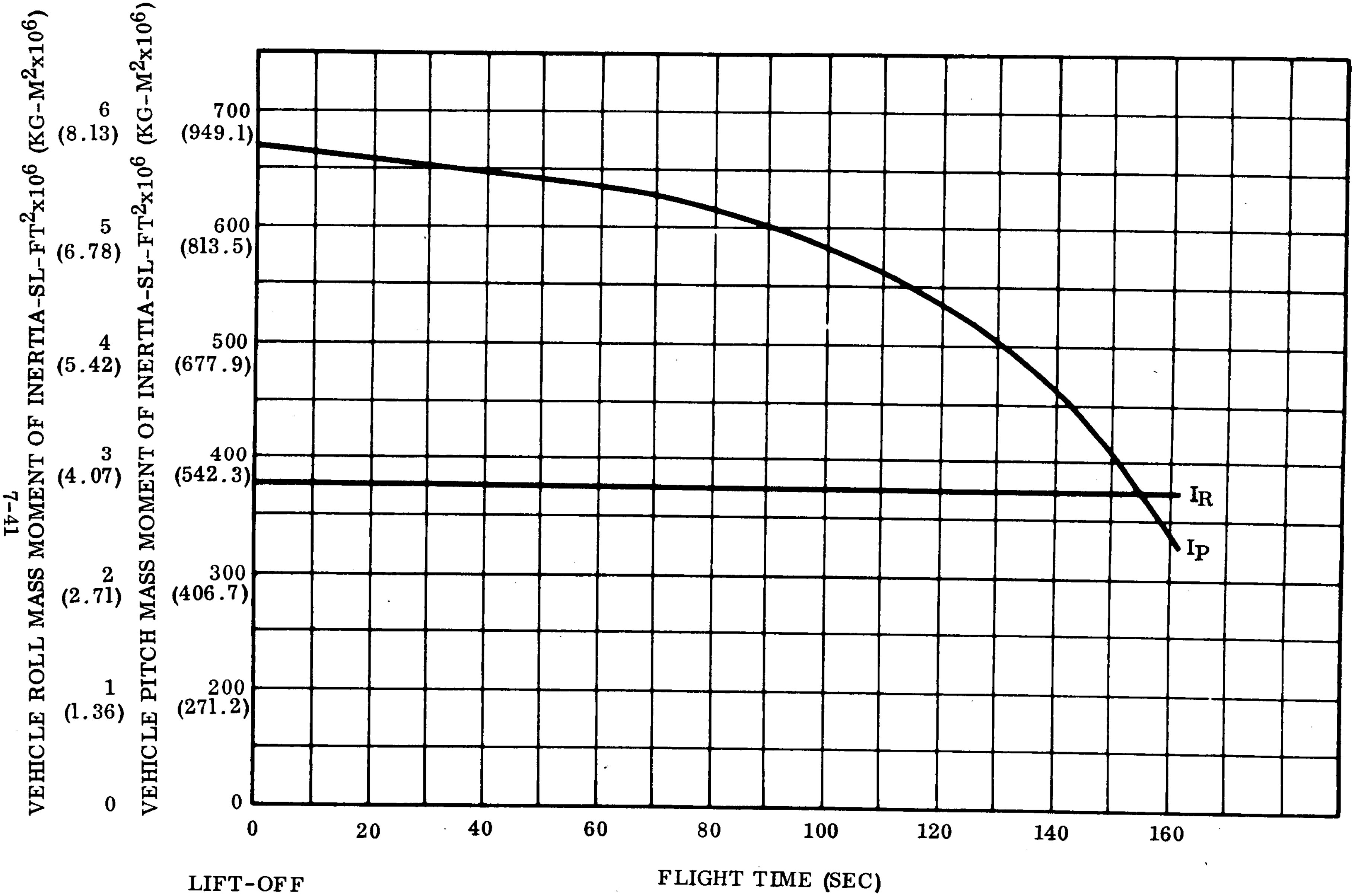


FIGURE 7.3-3 J-2S/SATURN V LOR VEHICLE MASS MOMENT OF INERTIA VS. VEHICLE FLIGHT TIME

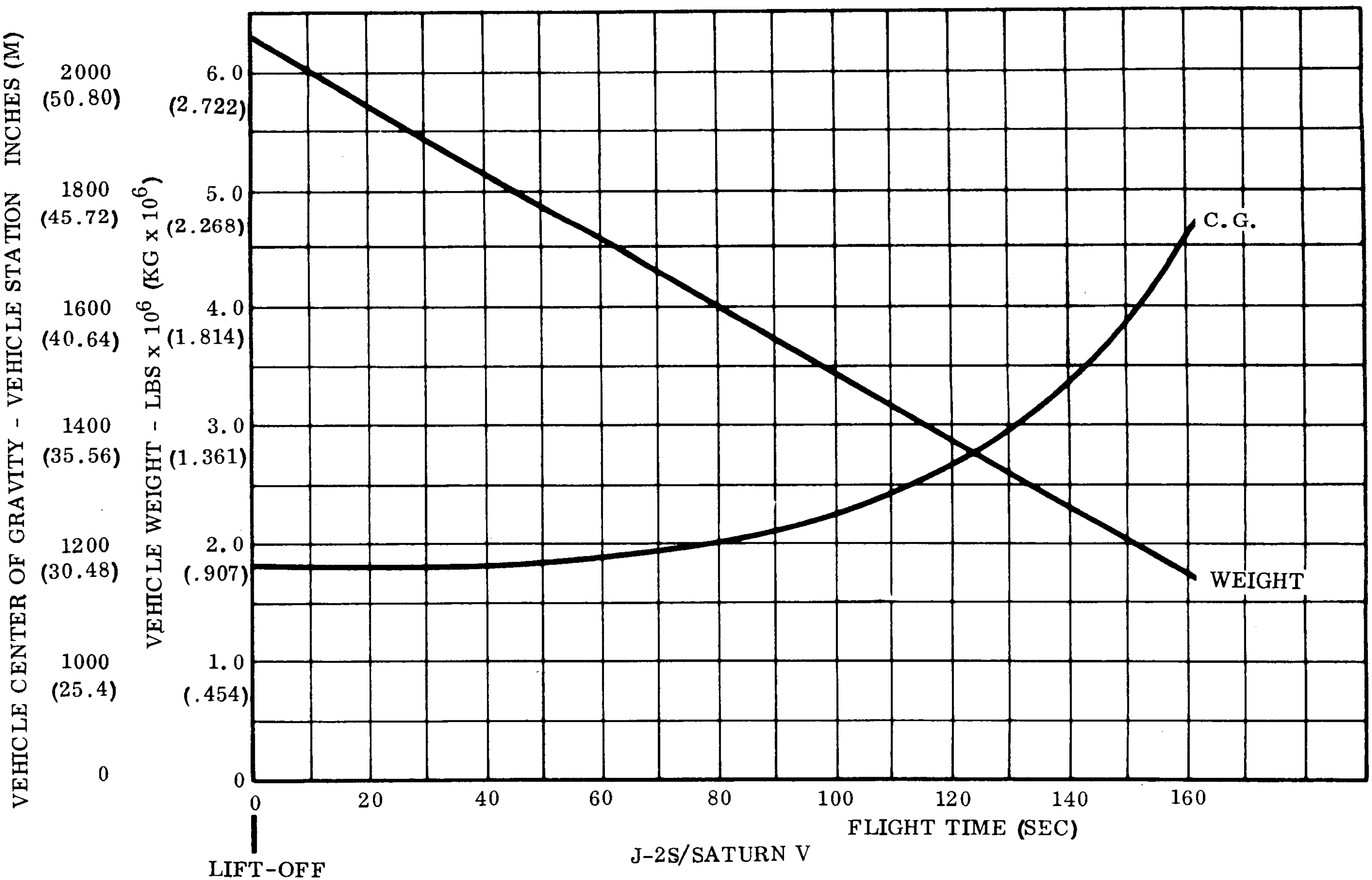


FIGURE 7.3-4 J-2S/SATURN V LOW EARTH ORBIT VEHICLE WEIGHT AND CENTER OF GRAVITY VS VEHICLE FLIGHT TIME

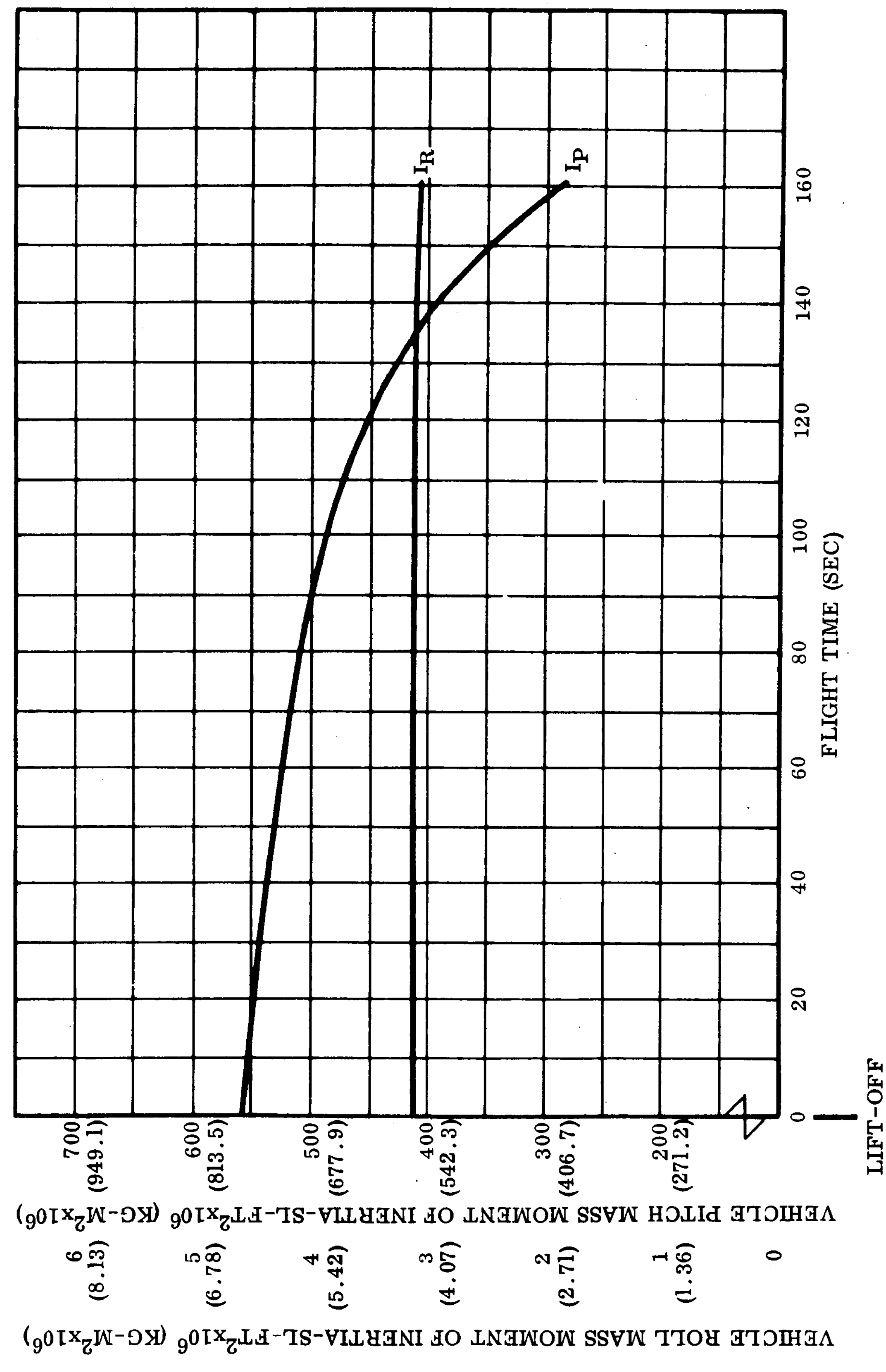


FIGURE 7.3-5 J-2S/SATURN V LOW EARTH ORBIT VEHICLE MASS MOMENT OF INERTIA VS VEHICLE FLIGHT TIME

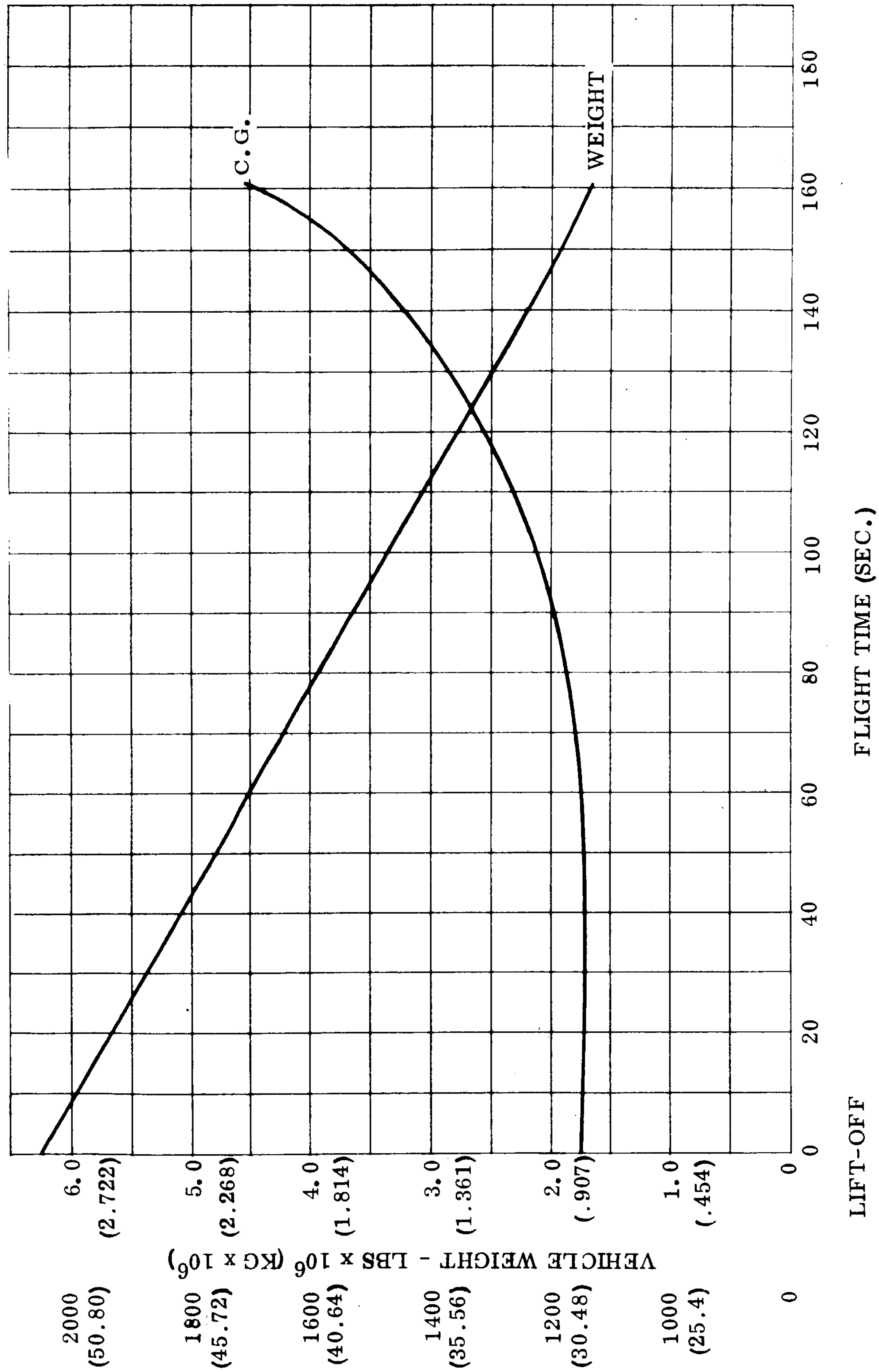


FIGURE 7.3-6 J-2S/SATURN V (2) STAGE POLAR ORBIT VEHICLE WEIGHT AND CENTER OF GRAVITY VS. VEHICLE FLIGHT TIME

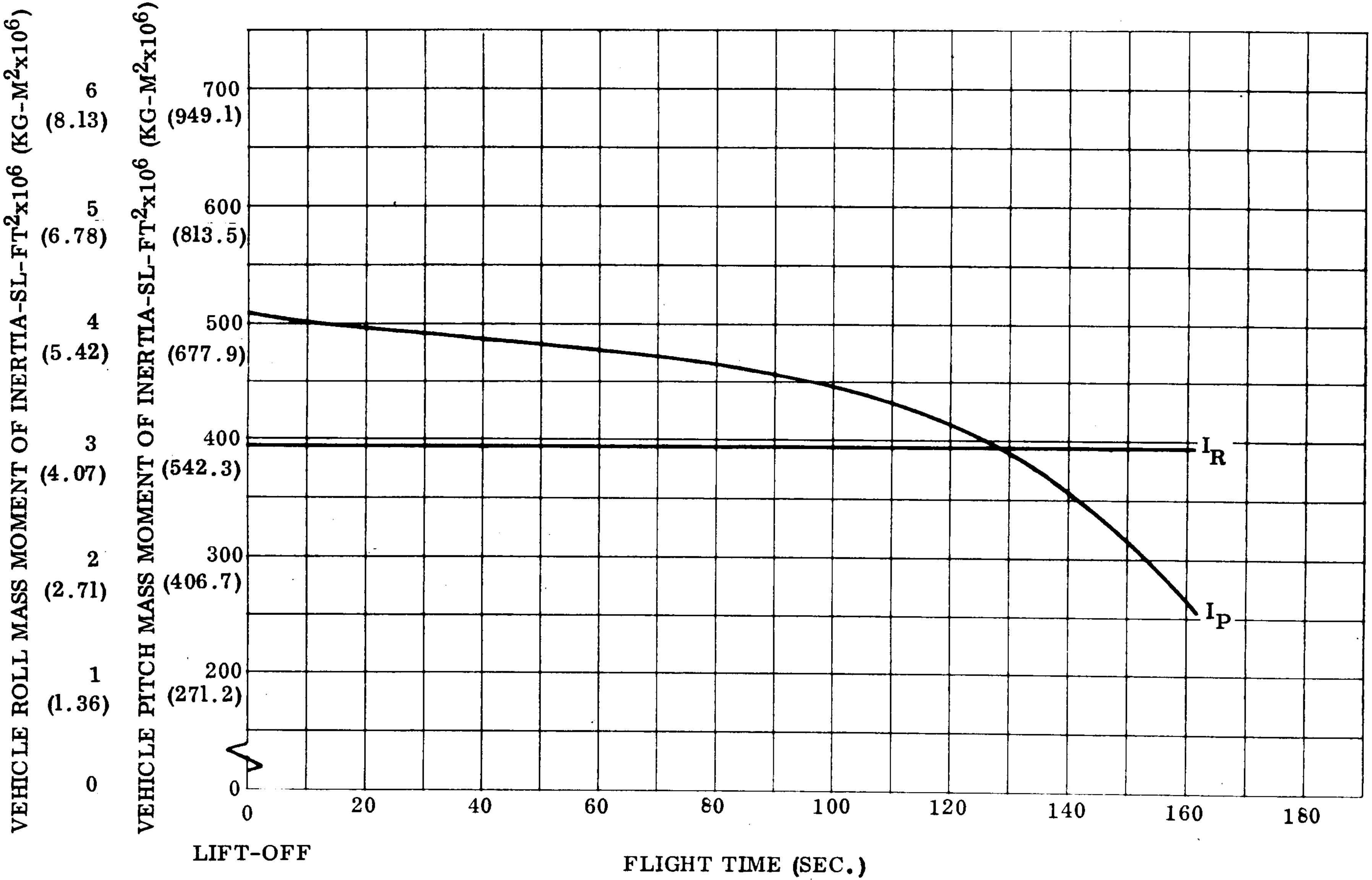


FIGURE 7.3-7 J-2S/SATURN V (2) STAGE POLAR ORBIT VEHICLE MASS MOMENT OF INERTIA VS. VEHICLE FLIGHT TIME

7.4 AERODYNAMICS

The J-2S Improvement Study used the static aerodynamic characteristics of the basic Apollo/Saturn V launch vehicle. These characteristics, provided by MSFC, References 7.4-1 and 7.4-2 were used for performance, control, and basic structural analyses of the J-2S/Saturn V design vehicles. The aerodynamics are for a typical first stage trajectory profile. After first stage flight, the astmosphere effects on the vehicle are negligible. Detailed definition of the aerodynamic properties is given in Document D5-15772-6, Structural Analysis J-2S Improvement Study.

7.4.1 Static Stability

The normal force coefficient gradient with corresponding center of pressure for the total vehicle is presented in Figure 7.4-1. This data includes the effects of protuberances. The normal force coefficient gradient and corresponding center of pressure are good only up to an angle of attack of 2 degrees. The variations for the static stability of the Saturn V vehicle are ± 6 percent on normal force coefficient and ± 0.2 caliber on center of pressure. These variations do not include Reynolds number effects.

The center of pressure is shown for roll angles of 90 degrees and 0 degrees. When the vehicle is rolled 90 degrees, the center of pressure moves forward by 0.15 caliber while the normal force remains essentially the same. For a roll angle of 45 degrees, the center of pressure is 0.1 caliber aft of the center of pressure for zero roll angle.

7.4.2 Axial Force

The variation of vehicle zero angle-of-attack axial force coefficients with Mach number is presented in Figure 7.4-2. The total axial force was determined for power-on conditions.

7.4.3 Normal Force Distribution

Local normal force coefficient distributions at specific angles of attack and Mach numbers are shown in Figures 7.4-3 through 7.4-6.

Negative spikes appear in the local normal force distributions for transonic Mach numbers. At low transonic Mach numbers, the negative spikes caused by local flow separation appear behind the command module and frustums. In the higher transonic Mach number range, the negative spikes appear at the compression corners located at the forward end of the frustums. This is a result of local flow separation, detached shock waves, and shock-wave boundary-layer interaction.

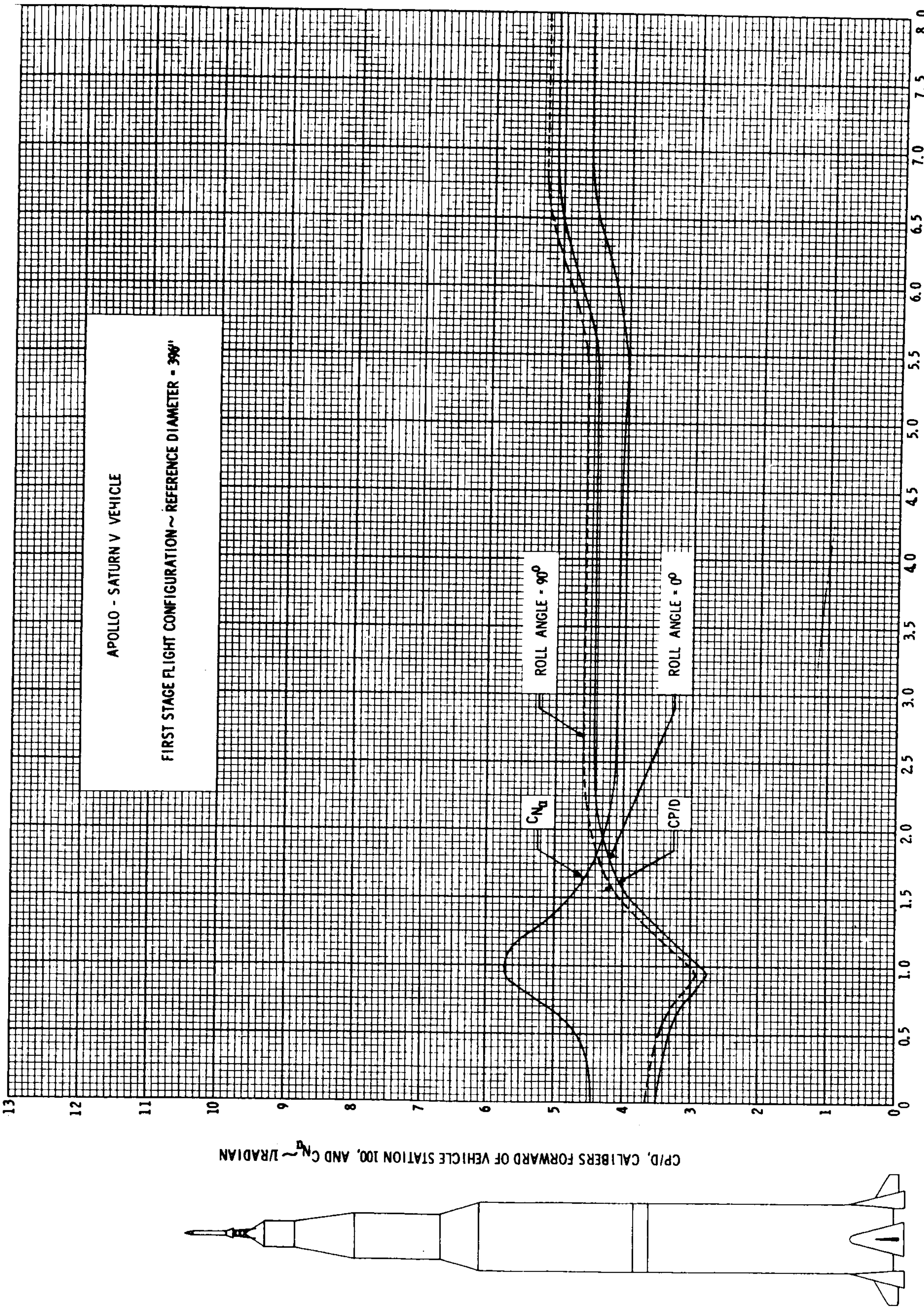


FIGURE 7.4-1 NORMAL FORCE COEFFICIENT GRADIENT AND CENTER OF PRESSURE VERSUS MACH NUMBER.

APOLLO/SATURN V VEHICLE

FIRST STAGE FLIGHT CONFIGURATION REFERENCE DIAMETER = 396"

$\alpha = 0^{o}$

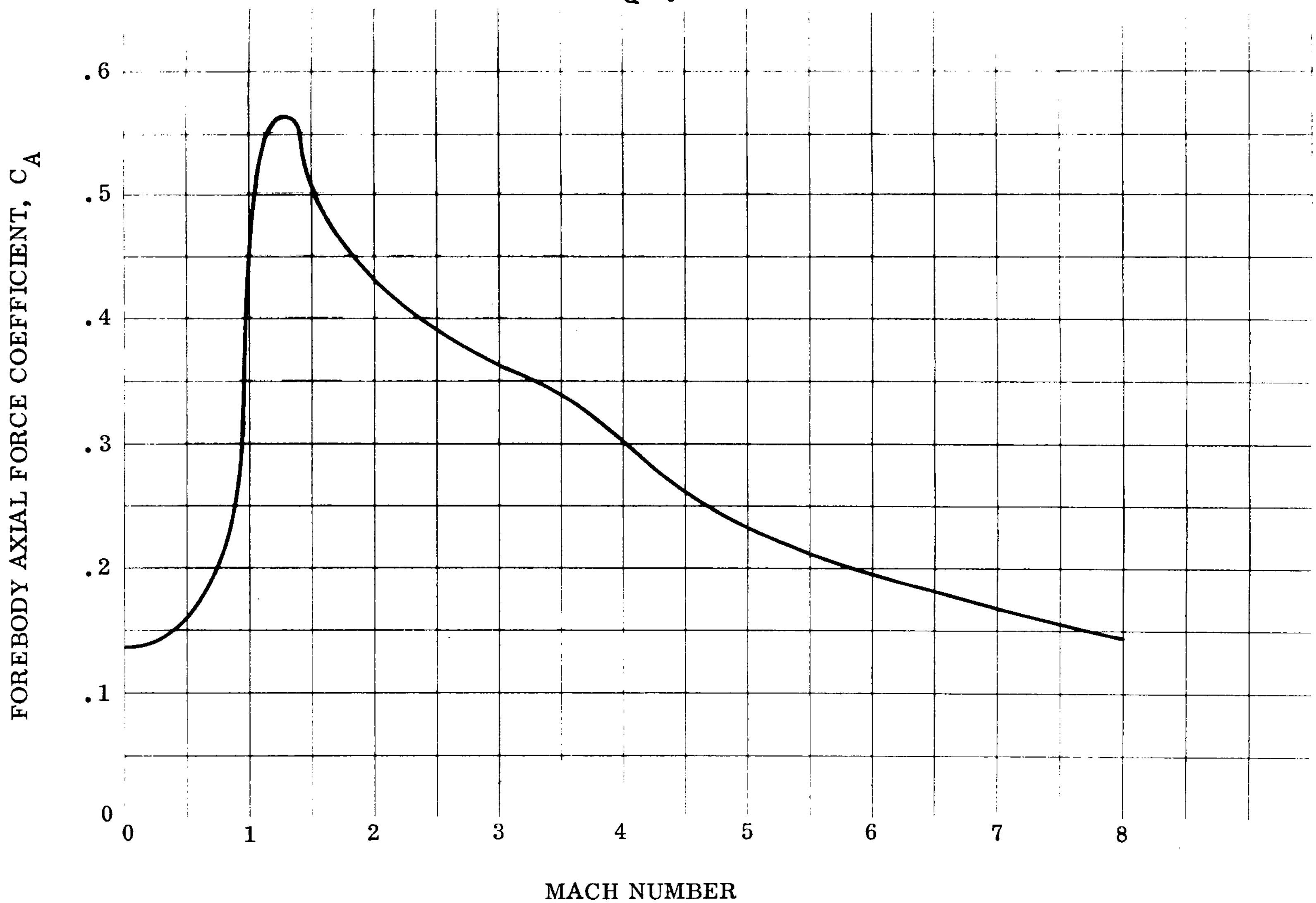


FIGURE 7.4-2 VARIATION OF FOREBODY AXIAL FORCE COEFFICIENT WITH MACH NUMBERS AT $\alpha = 0^{o}$

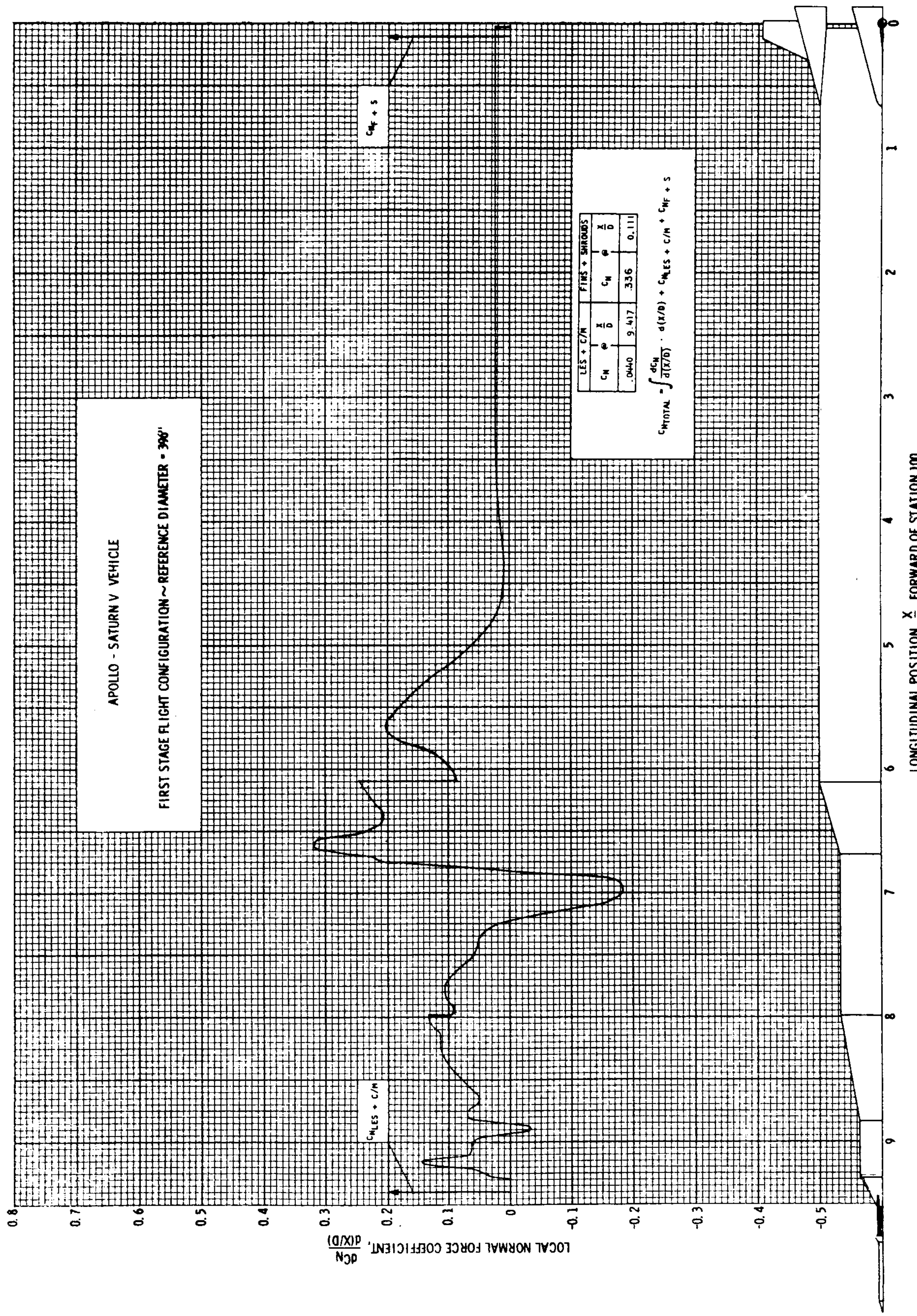
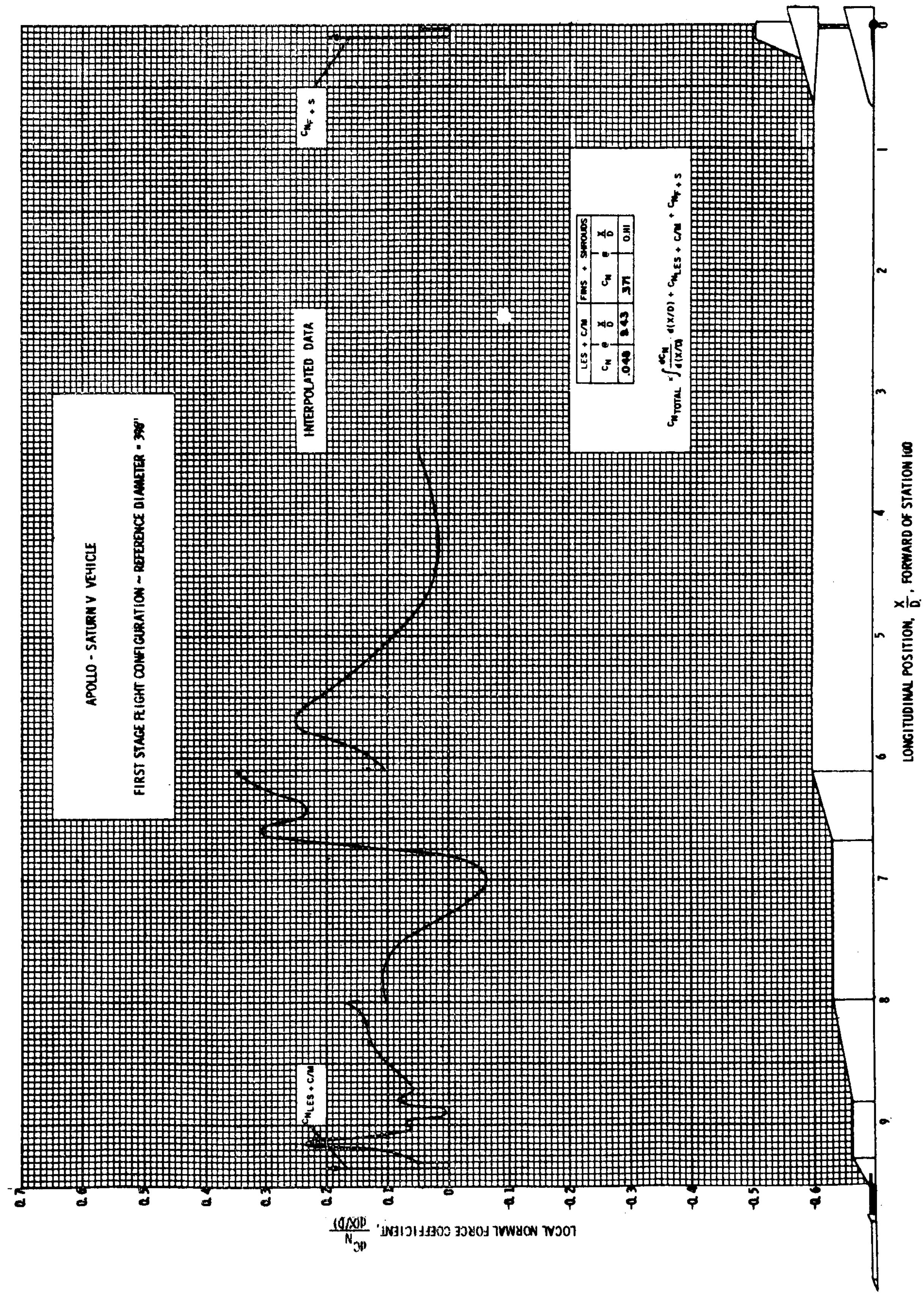


FIGURE 7.4-3 DISTRIBUTION OF LOCAL NORMAL FORCE COEFFICIENT, M = 1.20, α = 10°.



DISTRIBUTION OF LOCAL NORMAL FORCE COEFFICIENT, M = 1.2, α = 12°

FIGURE 7.4-4

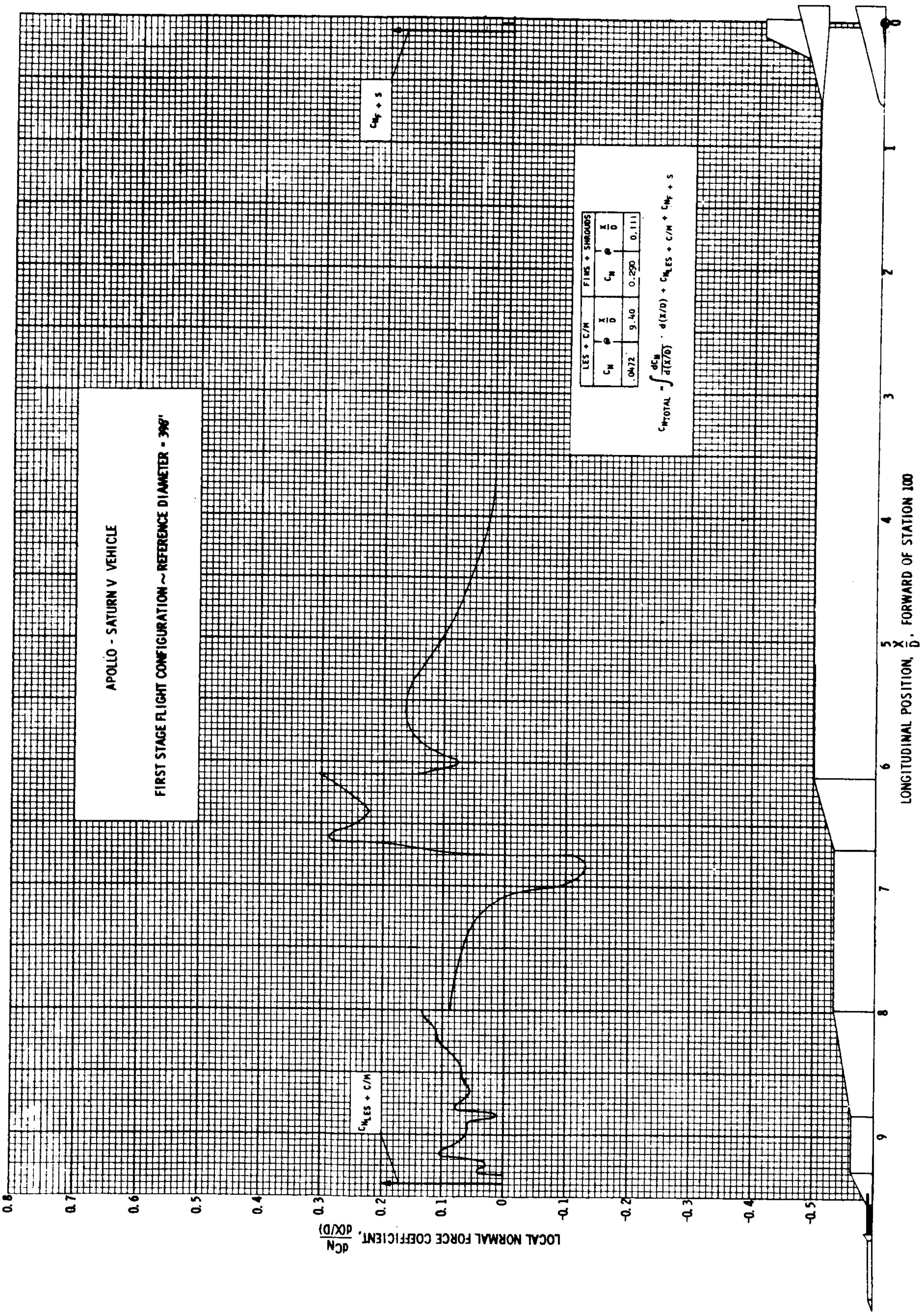


FIGURE 7.4-5 DISTRIBUTION OF LOCAL NORMAL FORCE COEFFICIENT, M = 1.5, α = 10°.

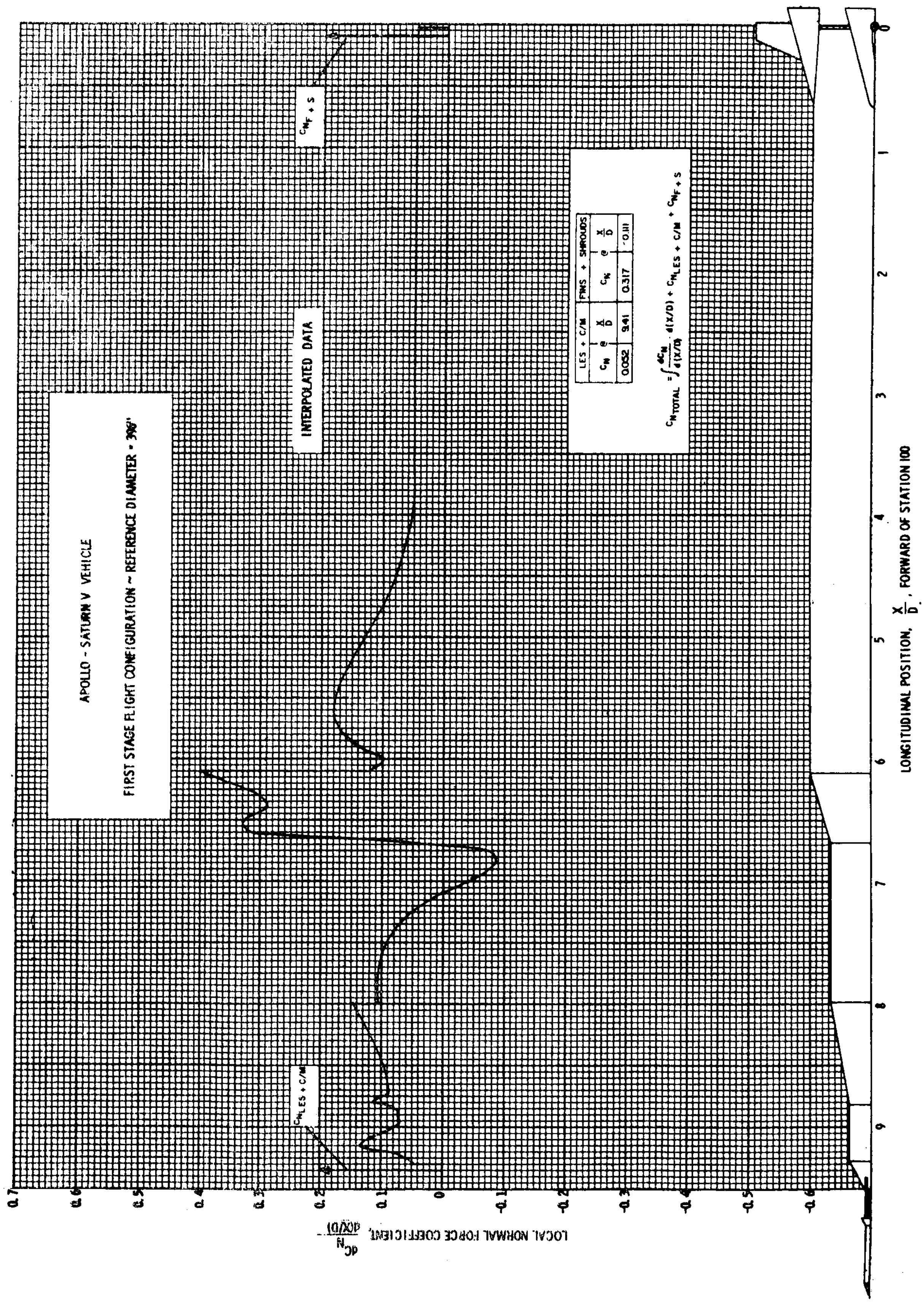


FIGURE 7.4-6

DISTRIBUTION OF LOCAL NORMAL FORCE COEFFICIENT, M = 1.5, α = 12°

7.4.3 (Continued)

The local normal force distributions contain concentrated loads for the contributions of the launch escape system/command module and the fin-shroud combinations. These data are appropriate for total vehicle load analyses, but not for individual component design, since interference and carry over effects are included in these data.

7.4.4 Axial Force Distributions

Distributions of local axial force coefficients are presented at zero angle of attack in Figure 7.4-7 and 7.4-8.

For vehicle design purposes, these data are suitable for vehicle angles of attack less than 10 degrees. Static stability wind tunnel investigations have established approximately a 5 percent increase of vehicle axial force for angles of attack up to 10 degrees.

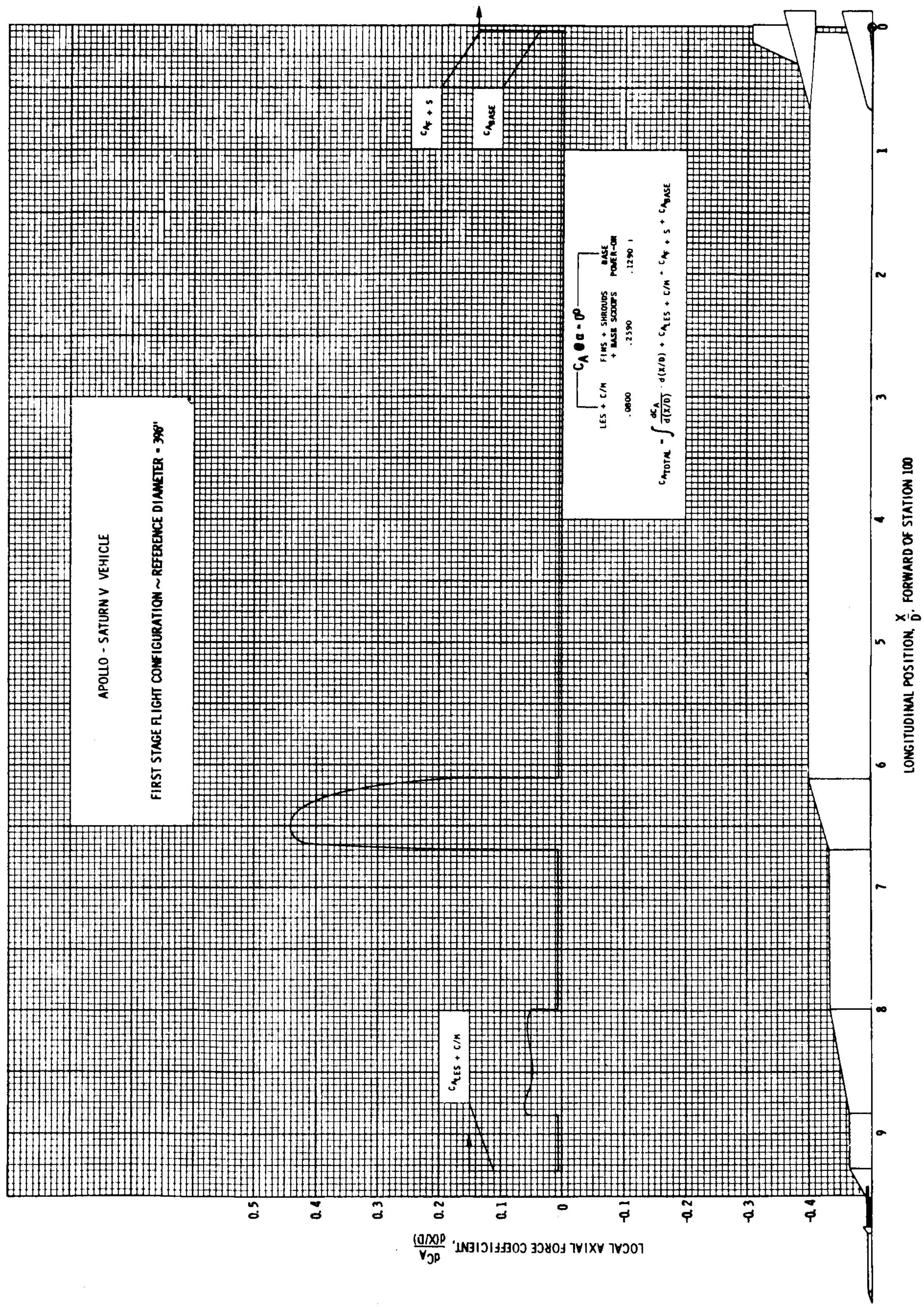


FIGURE 7.4-7 DISTRIBUTION OF LOCAL AXIAL FORCE COEFFICIENT, α = 0°, M = 1.20

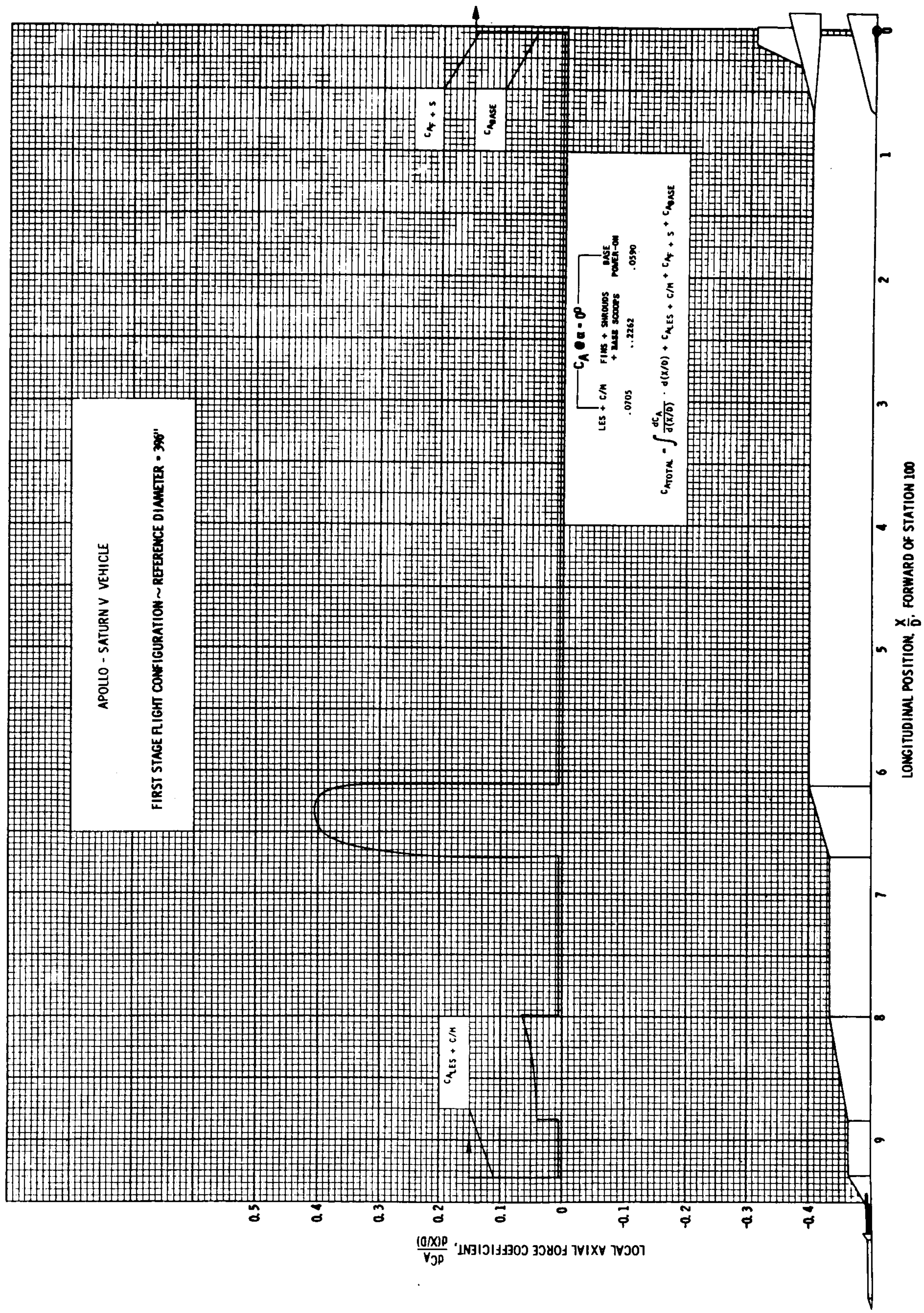


FIGURE 7.4-8 DISTRIBUTION OF LOCAL AXIAL FORCE COEFFICIENT, α = 0°, M = 1.50

7.5 STRUCTURAL CRITERIA AND LOADS

Structural design requirements used by associate contractors in determining stage structural impact are presented in this section. Detailed back-up data are contained in Document D5-15772-6, Structural Analysis.

7.5.1 Environmental Requirements

a. Flight Winds - The flight wind environments were developed using MSFC design wind criteria. The flight wind profile for the nominal case was obtained from a 99 percent probable shear buildup to a 95 percent probable peak wind speed at 10,000 meters altitude. An embedded jet gust was imposed upon the peak of the wind profile. The inflight wind profile is shown in Figure 7.5-1.

b. Acoustics Environments - Preliminary maximum on-pad acoustic environments and inflight fluctuating pressure environments for the J-2S/Saturn V LOR, LEO, synchronous and two-stage polar missions were supplied by MSFC, Reference 7.5-1 and are given in Document D5-15772-6, Appendix B.

   These environments are presented in the form of octave band sound pressure level spectra for the on-pad condition and octave band fluctuating pressure level spectra for the inflight conditions. These spectra are given per vehicle zone. The spectra represent the maximum anticipated environments for each respective zone for the on-pad and inflight conditions.

Although the sound pressure levels differ somewhat from the Saturn V environment, the new acoustic levels do not cause any vehicle modifications.

7.5.2 Design Loads for Nominal Flight

Ultimate compressive design loads for each stage of the LOR, LEO and two-stage polar orbit design vehicles during nominal flight are given in Figures 7.5-2 through 7.5-10. Nominal flight assumes no engine malfunctions. The ultimate combined compressive loads are envelopes of design loads developed for the vehicle design conditions. A tabulation of the ultimate compressive combined loads for each design condition is shown in Tables 7.5-I, 7.5-II and 7.5-III. The ultimate compressive combined loads are determined as follows:

$$N_{C_{ULT}} = \left[\frac{P(X)}{2\pi R(X)} + \frac{BM(X)}{\pi R^2(X)}\right] F.S. - P_{u_{min}} \frac{R(X)}{2}$$

Where:

- $P(X)$ = distributed longitudinal forces including aerodynamic forebody drag
- $BM(X)$ = distributed bending moment
- $R(X)$ = distributed body radius
- $P_{U_{MIN}}$ = minimum ullage pressure (applicable to tank shells only)
- $F.S.$ = ultimate factor of safety of 1.4 for nominal flight loads (manned flight criteria)

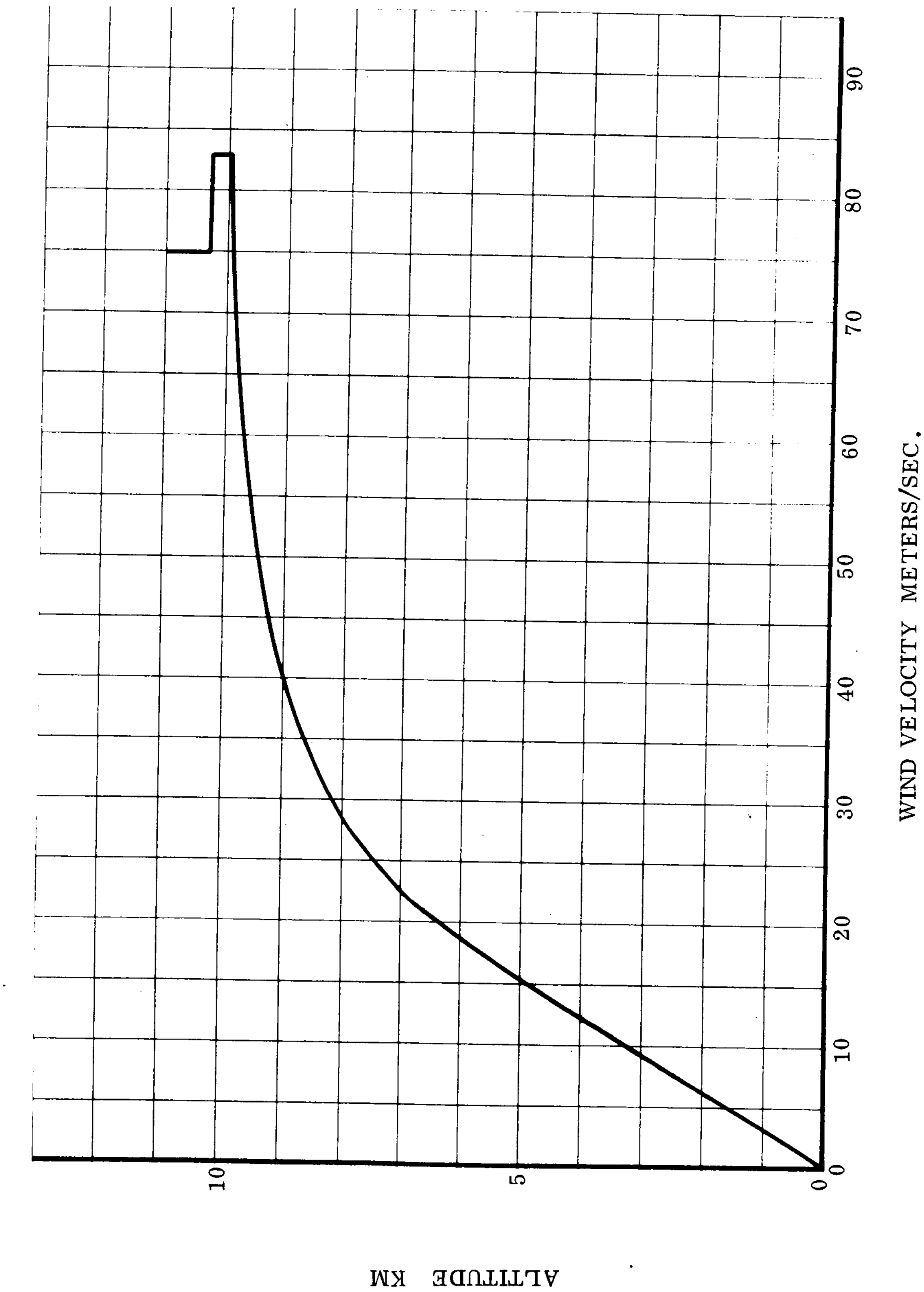


FIGURE 7.5-1 INFLIGHT WIND PROFILE

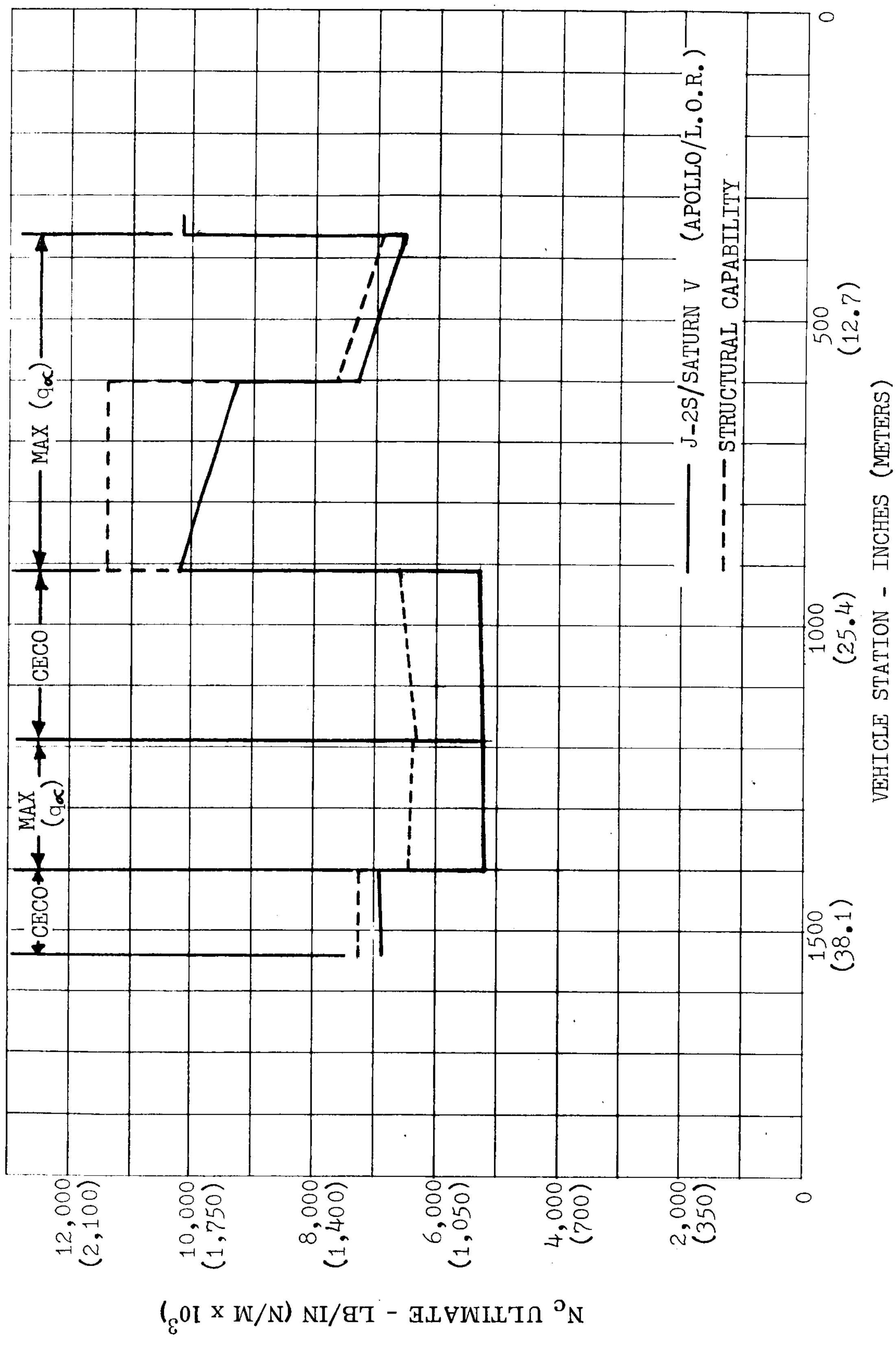


FIGURE 7.5-2 J-2S/SATURN V APOLLO/L.O.R. S-IC COMBINED LOAD DISTRIBUTION

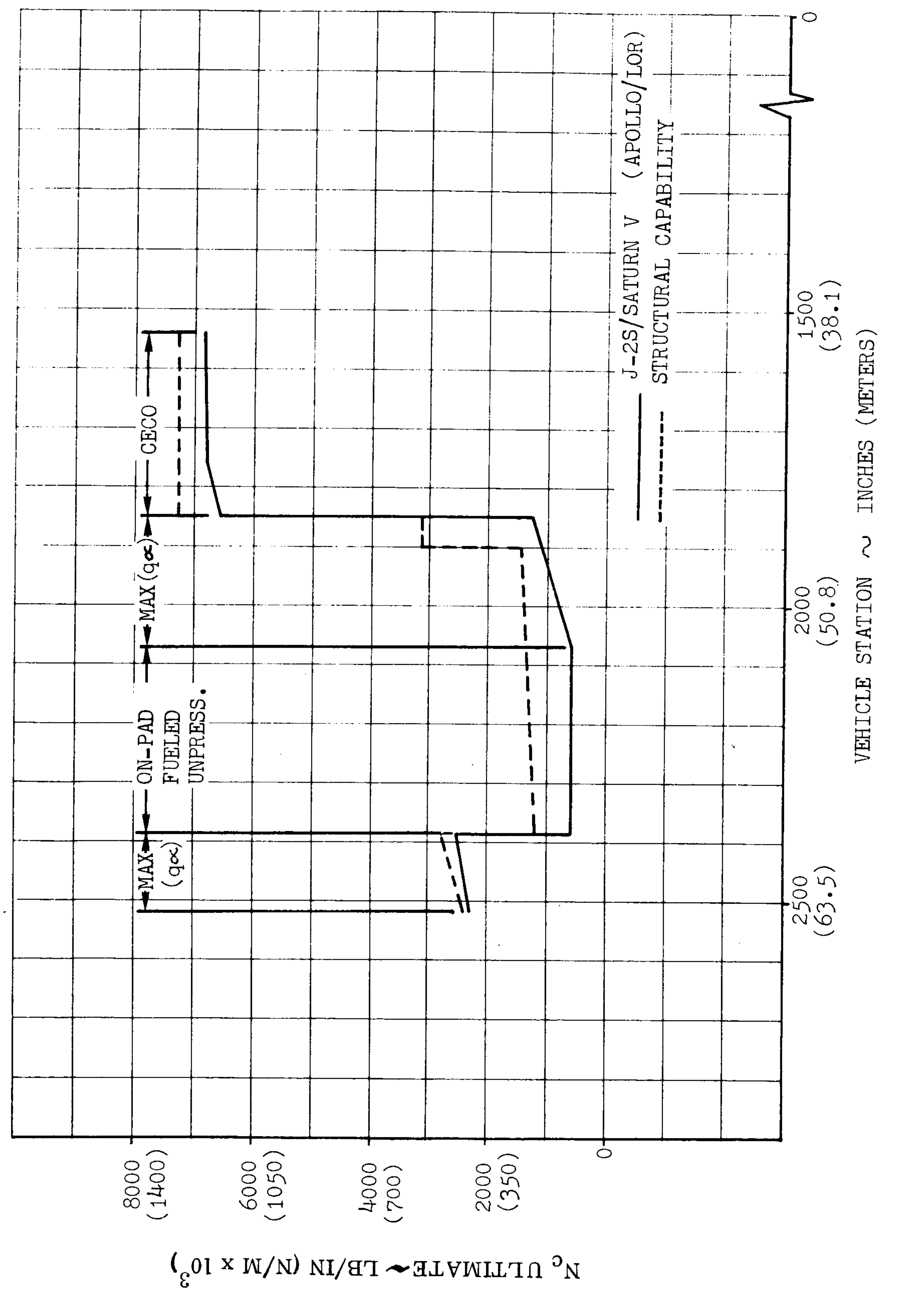


FIGURE 7.5-3 J-2S/SATURN V APOLLO/LOR S-II COMBINED LOAD DISTRIBUTION

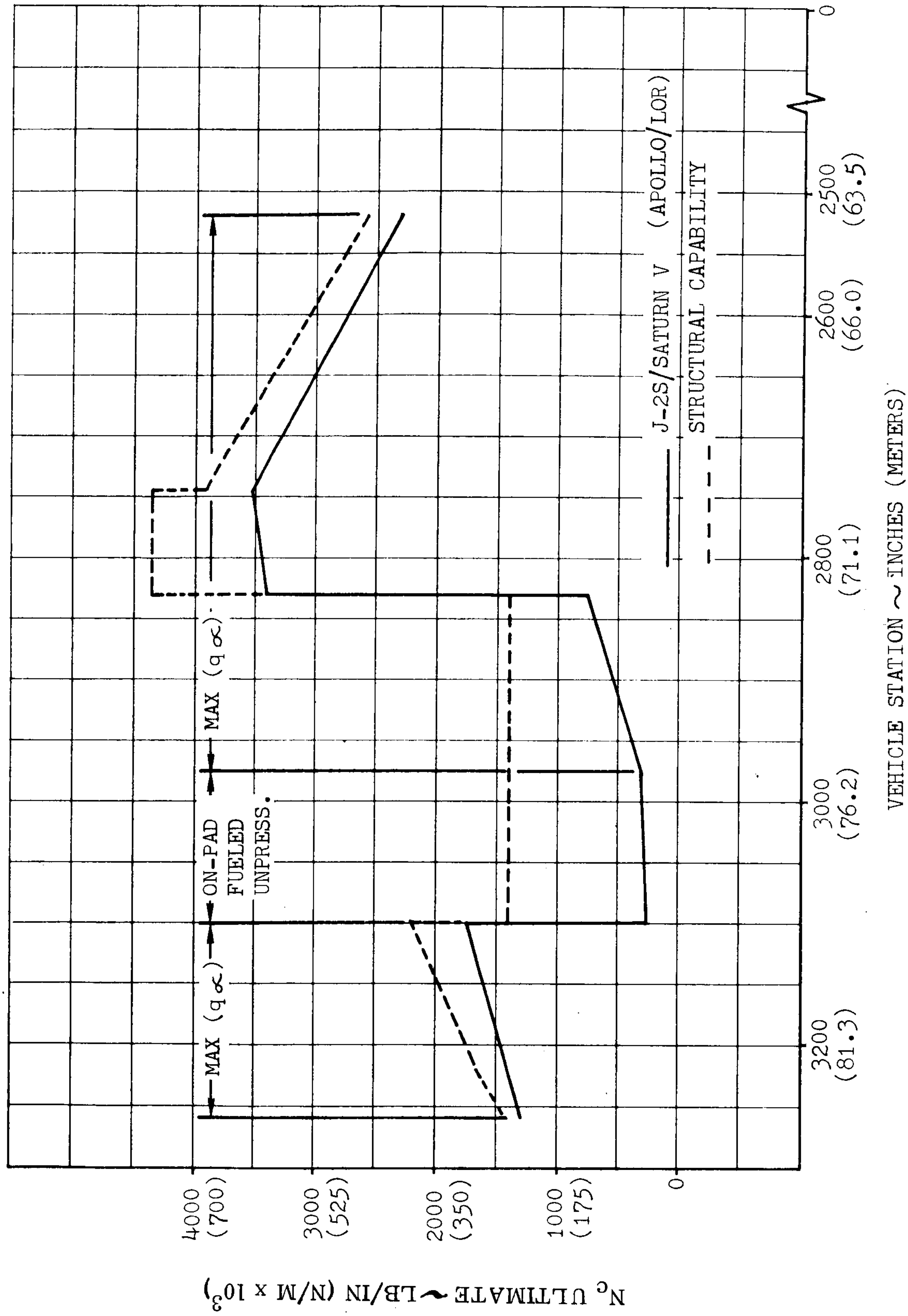


FIGURE 7.5-4 J-2S/SATURN V APOLLO/LOR S-IVB & I.U. COMBINED LOAD DISTRIBUTION

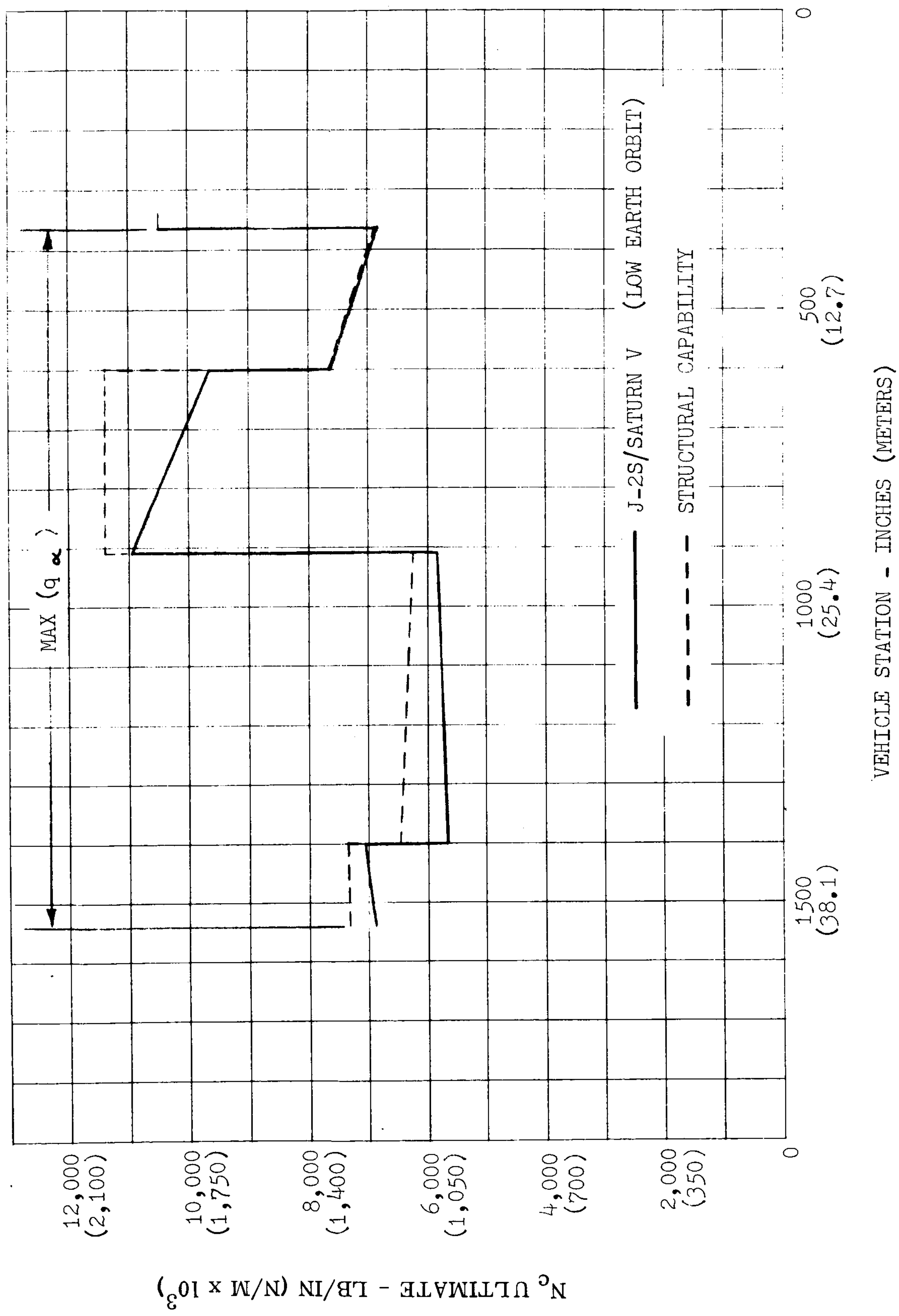


FIGURE 7.5-5 J-2S/SATURN V LOW EARTH ORBIT S-IC COMBINED LOAD DISTRIBUTION

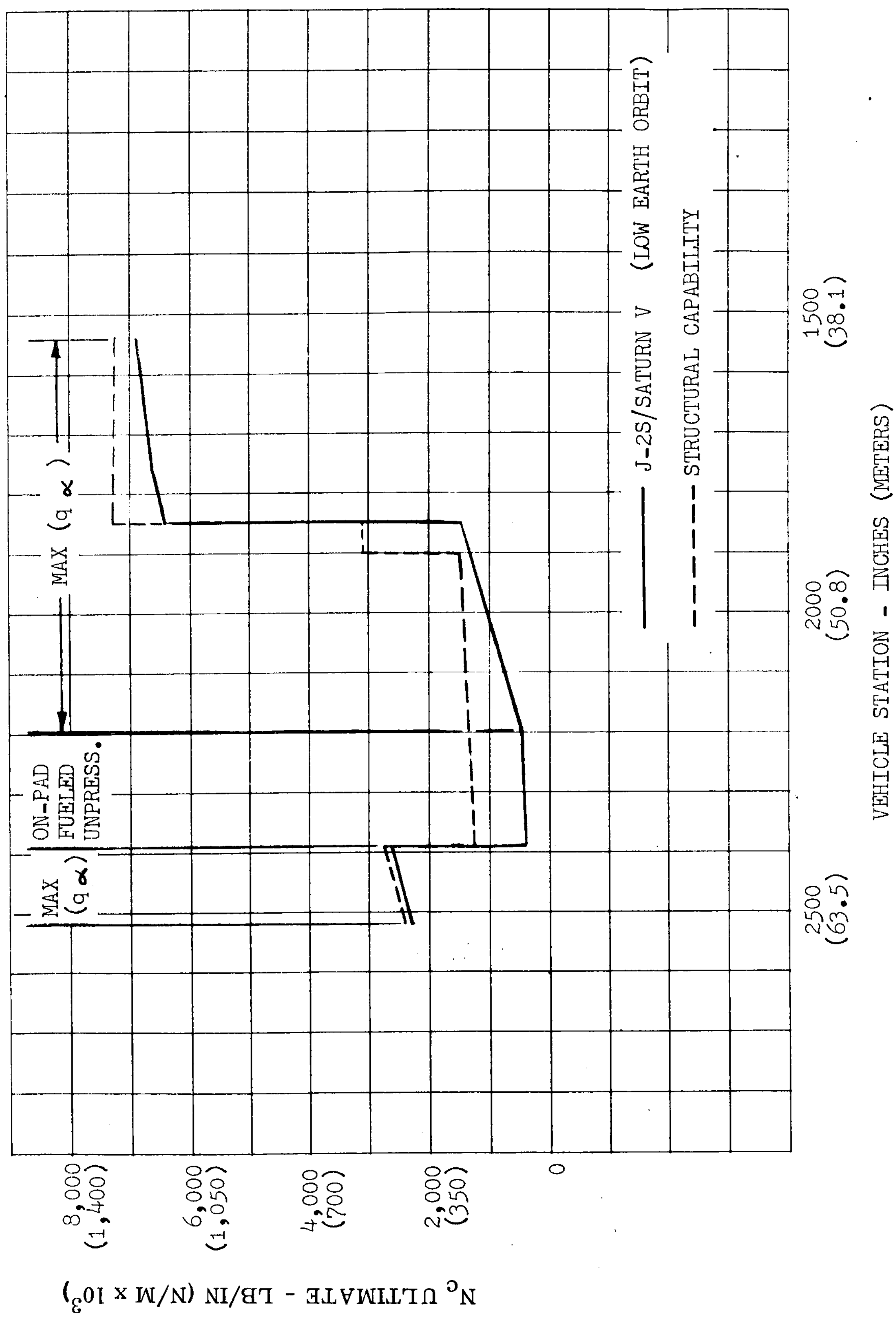


FIGURE 7.5-6 J-2S/SATURN V LOW EARTH ORBIT S-II COMBINED LOAD DISTRIBUTION

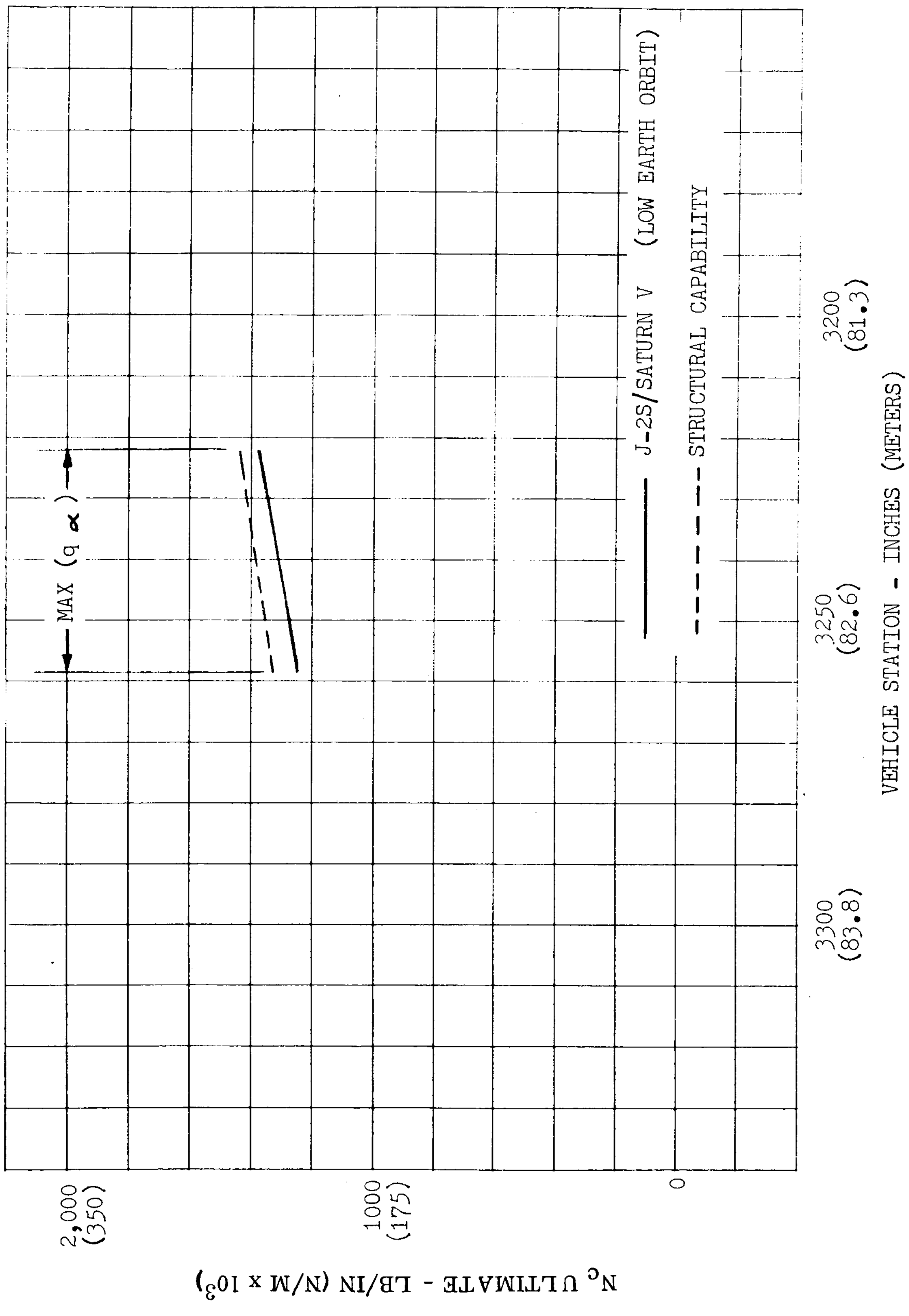


FIGURE 7.5-7 J-2S/SATURN V LOW EARTH ORBIT I.U. COMBINED LOAD DISTRIBUTION

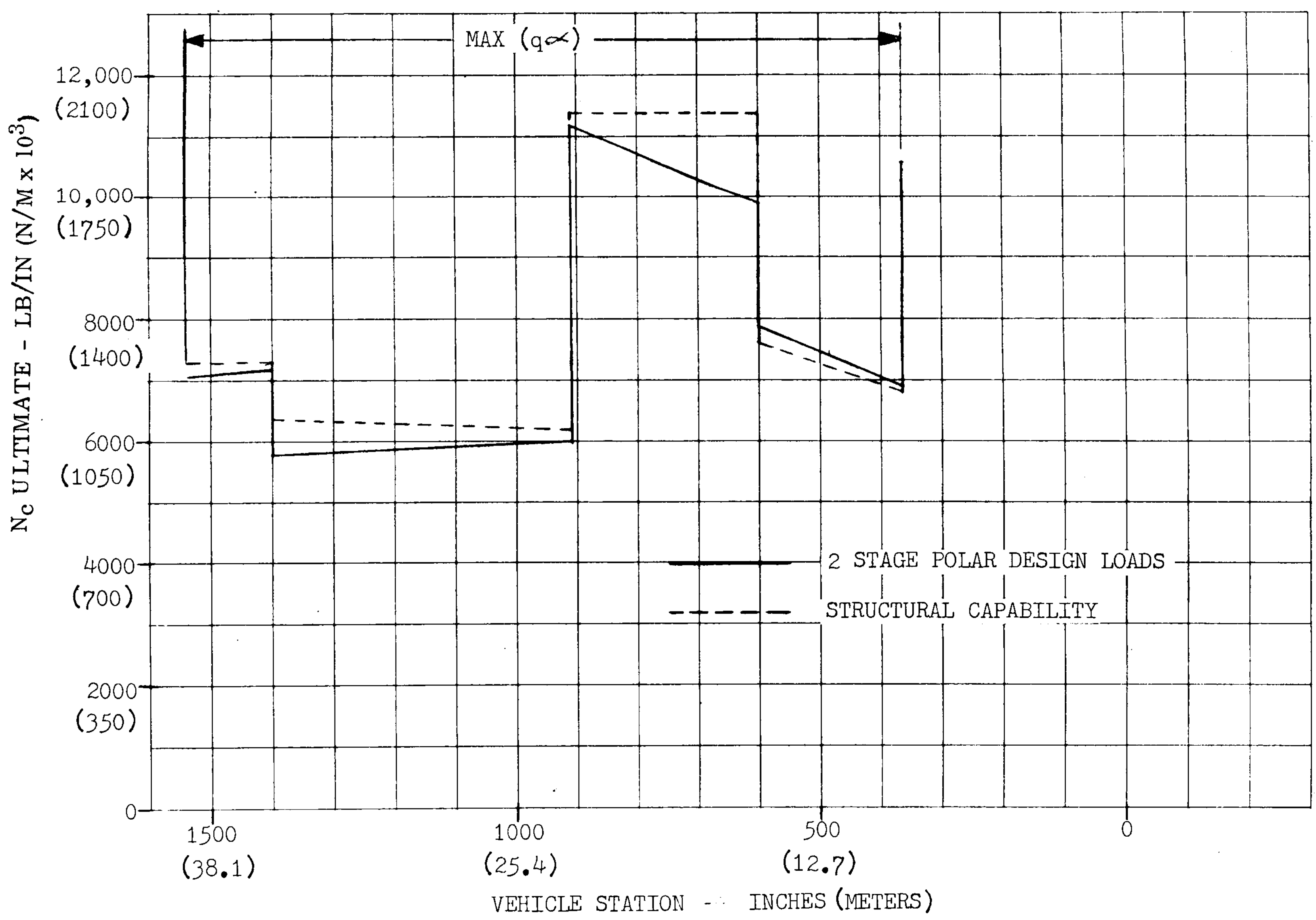


FIGURE 7.5-8 J-2S/SATURN V 2 STAGE POLAR ORBIT S-IC COMBINED LOAD DISTRIBUTION

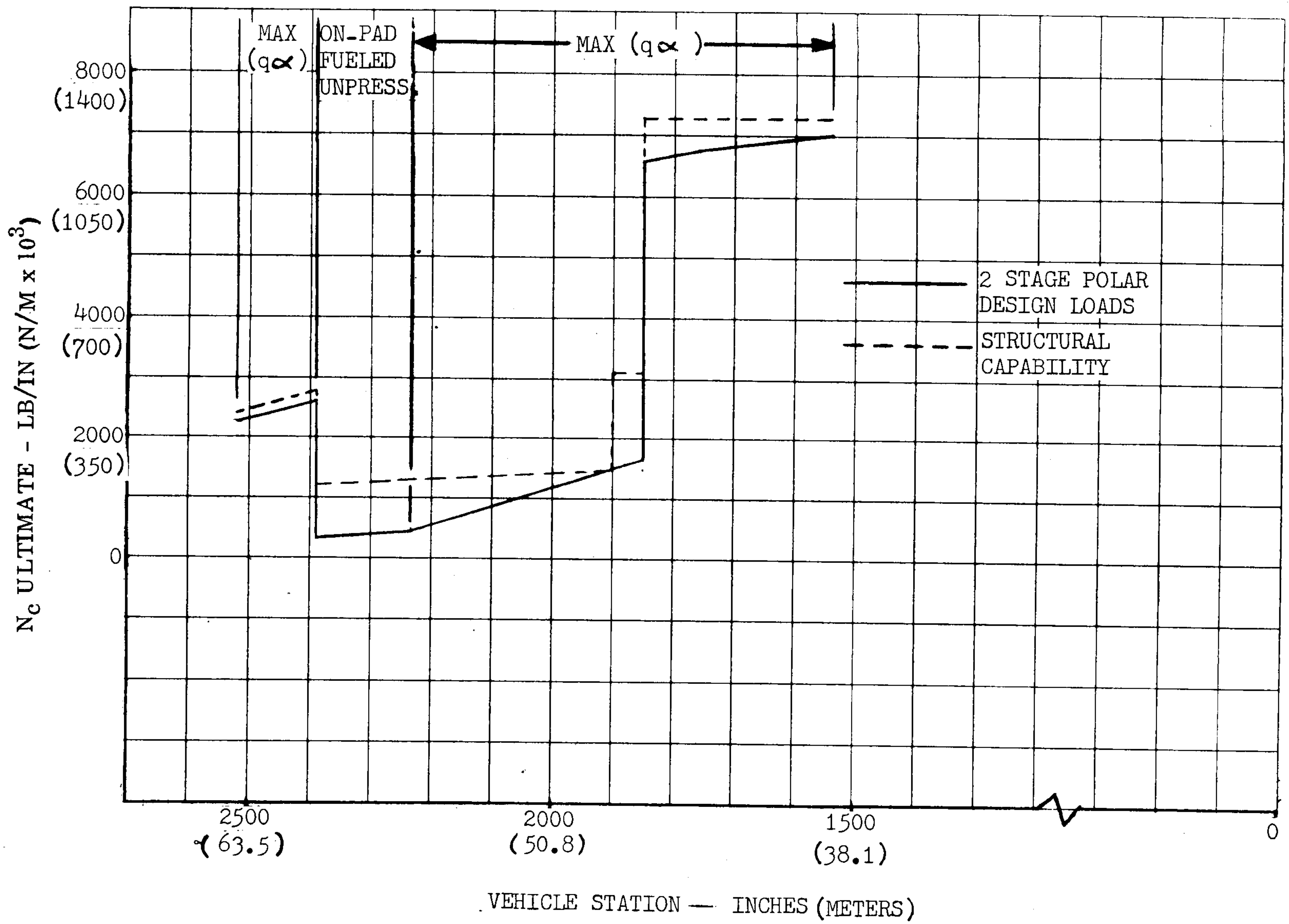


FIGURE 7.5-9 J-2S/SATURN V 2-STAGE POLAR ORBIT S-II COMBINED LOAD DISTRIBUTION

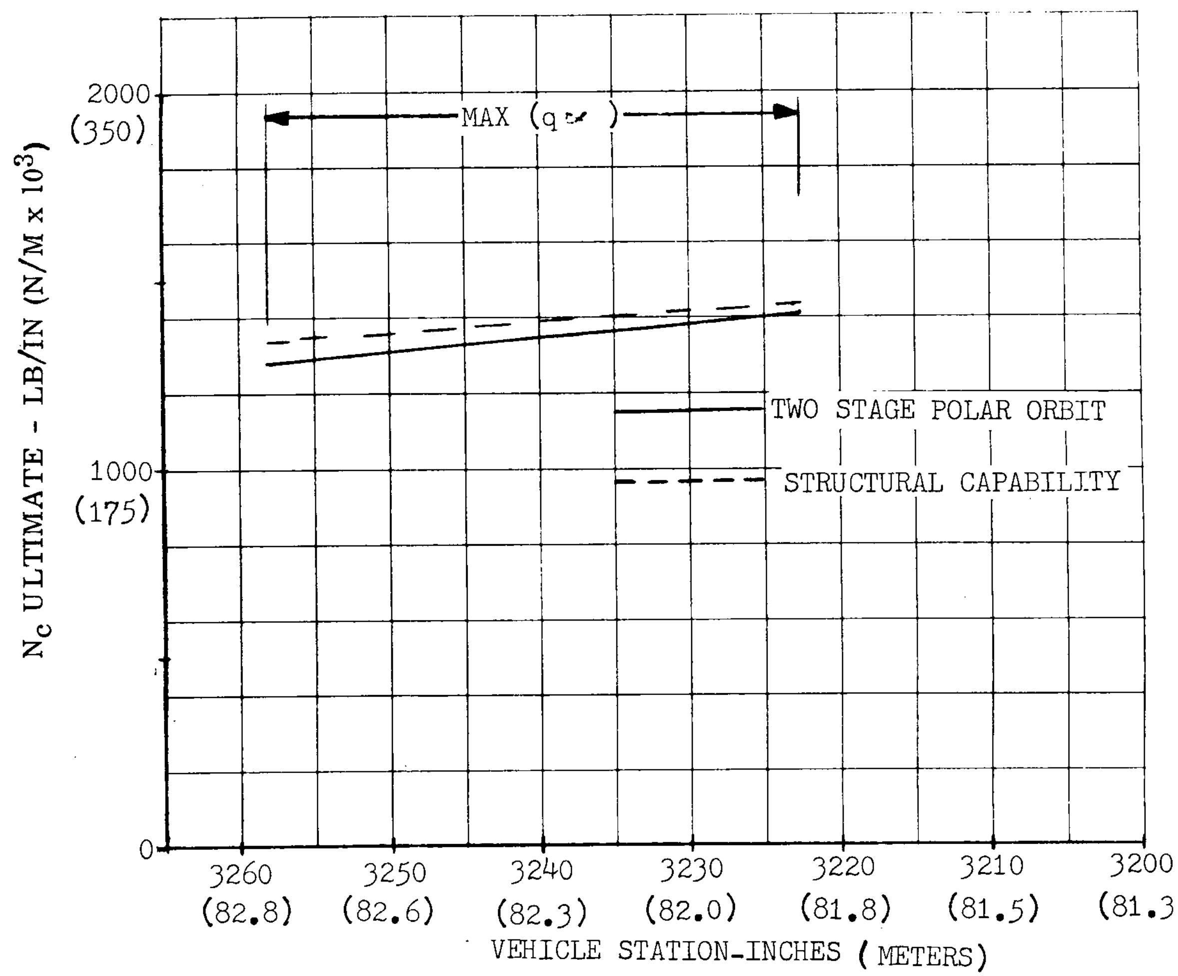


FIGURE 7.5-10 J-2S/SATURN V 2 STAGE POLAR ORBIT I.U. COMBINED LOAD DISTRIBUTION

TABLE 7.5-I N_C ULTIMATE FOR J-2S/SATURN V APOLLO/LOR

STATION (IN)	$M \times 10^{-6}$ (IN-LB)	$M/\pi R^2$ (LB/IN)	$P \times 10^{-6}$ (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN)	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
ON-PAD FUELED UNPRESS.									
365A	106.5	865	6.184	4970	5835	8169	0	0	8169
365F	106.5	865	4.770	3833	4698	6577	0	0	6577
602A	88.5	719	4.746	3815	4534	6348	0	0	6348
602F	88.5	719	4.741	3811	4530	6342	0	0	6342
912A	69.5	564	4.708	3784	4348	6087	0	0	6087
912F	69.5	564	1.520	1222	1786	2500	0	0	2500
1401A	43.0	349	1.491	1198	1547	2166	0	0	2166
1401F	43.0	349	1.486	1194	1543	2160	0	0	2160
1541	37.5	304	1.477	1188	1492	2089	0	0	2089
1760	28.0	227	1.463	1176	1403	1964	0	0	1964
1848A	26.0	211	1.421	1142	1353	1894	0	0	1894
1848F	26.0	211	.425	342	553	774	0	0	774
2387A	11.5	93	.401	323	416	582	0	0	582
2387F	11.5	93	.398	320	413	578	0	0	578
2519	9.5	77	.392	315	392	549	0	0	549
2746.5	6.0	113	.385	471	584	818	0	0	818
2832A	4.5	85	.379	464	549	769	0	0	769
2832F	4.5	85	.134	164	249	349	0	0	349
3100.55A	2.0	38	.127	156	196	274	0	0	274
3100.55F	2.0	38	.126	154	192	269	0	0	269
3222.55	1.5	28	.124	152	180	252	0	0	252
3258.55	1.3	24	.120	147	171	240	0	0	240

TABLE 7.5-I (Continued)

STATION (IN)	$M \times 10^{-6}$ (IN-LB)	$M/\pi R^2$ (LB/IN)	$P \times 10^{-6}$ (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN)	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
REBOUND									
365A	84.0	682	7.260	5836	6518	9125	0	0	9125
365F	84.0	682	5.620	4518	5200	7280	9.3	921	6359
602A	68.0	552	5.560	4469	5021	7029	9.3	921	6108
602F	68.0	552	5.560	4469	5021	7029	0	0	7029
912A	51.0	414	5.520	4437	4851	6791	0	0	6791
912F	51.0	414	1.760	1415	1829	2561	3.3	327	2234
1401A	30.0	244	1.725	1387	1631	2283	3.3	327	1956
1401F	30.0	244	1.725	1387	1631	2283	0	0	2283
1541	25.0	203	1.717	1380	1583	2216	0	0	2216
1760	18.3	149	1.700	1367	1516	2122	0	0	2122
1848A	16.2	132	1.650	1326	1458	2041	0	0	2041
1848F	16.2	132	.489	393	525	735	19.3	1911	-1299
2387A	8.5	69	.461	371	440	616	19.3	1911	-1295
2387F	8.5	69	.458	368	437	612	0	0	612
2519	7.2	58	.458	368	426	596	0	0	596
2746.5	4.8	90	.451	552	642	899	0	0	899
2832A	4.0	75	.443	542	617	864	0	0	864
2832F	4.0	75	.157	192	267	374	16.3	1614	-1240
3100.55A	2.0	38	.149	182	220	308	16.3	1614	-1306
3100.55F	2.0	38	.148	181	219	307	0	0	307
3222.55	1.5	28	.146	178	206	288	0	0	288
3258.55	1.3	25	.143	175	200	280	0	0	280

TABLE 7.5-I (Continued)

STATION (IN)	$M \times 10^{-6}$ (IN-LB)	$M/\pi R^2$ (LB/IN)	$P \times 10^{-6}$ (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN)	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
MAX. ($q\alpha$) NOMINAL FLIGHT									
365A	119	965	7.825	6289	7254	10,157	0	0	10,157
365F	119	965	6.438	5175	6140	8,596	20.38	2018	6,578
602A	188	1523	6.375	5124	6647	9,306	20.38	2018	7,288
602F	188	1523	6.366	5117	6640	9,296	0	0	9,296
912A	276	2239	6.297	5062	7301	10,221	0	0	10,221
912F	276	2239	3.161	2541	4780	6,692	14.38	1424	5,268
1401A	277	2246	3.097	2489	4735	6,629	14.38	1424	5,205
1401F	277	2246	3.088	2482	4728	6,619	0	0	6,619
1541	268	2176	3.069	2467	4643	6,500	0	0	6,500
1760	255	2073	3.039	2443	4516	6,322	0	0	6,322
1848A	252	2049	2.956	2376	4425	6,195	0	0	6,195
1848F	252	2049	1.029	827	2876	4,026	27.5	2722	1,304
2387A	128	1043	.976	785	1828	2,559	27.5	2722	- 162
2387F	128	1043	.971	780	1823	2,552	0	0	2,552
2519	107	870	.957	769	1639	2,295	0	0	2,295
2746.5	81	1524	.819	1002	2526	3,536	0	0	3,536
2832A	77	1443	.806	987	2430	3,402	0	0	3,402
2832F	77	1443	.332	407	1850	2,590	27.38	1780	810
3100.55A	45	855	.316	387	1242	1,739	27.38	1780	- 41
3100.55F	45	855	.314	384	1239	1,735	0	0	1,735
3222.55	33	626	.309	378	1004	1,406	0	0	1,406
3258.55	30	569	.300	367	936	1,310	0	0	1,310

TABLE 7.5-I (Continued)

STATION (IN)	$M \times 10^{-6}$ (IN-LB)	$M/\pi R^2$ (LB/IN)	$P \times 10^{-6}$ (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN)	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
CENTER ENGINE CUT-OFF (MAX. ACCELERATION)									
365A	0	0	7.781	6254	6254	8812	0	0	8812
365F	0	0	7.403	5950	5950	8330	24.0	2376	5954
602A	0	0	7.273	5847	5847	8186	24.0	2376	5810
602F	0	0	7.256	5832	5832	8165	0	0	8165
912A	0	0	7.116	5720	5720	8008	0	0	8008
912F	0	0	6.282	5050	5050	7070	18.0	1782	5288
1401A	0	0	6.159	4951	4951	6931	18.0	1782	5149
1401F	0	0	6.140	4935	4935	6909	0	0	6909
1541	0	0	6.104	4907	4907	6870	0	0	6870
1760	0	0	6.045	4859	4859	6803	0	0	6803
1848A	0	0	5.870	4718	4718	6605	0	0	6605
1848F	0	0	1.760	1415	1415	1981	27.5	2723	- 742
2387A	0	0	1.662	1336	1336	1870	27.5	2723	- 853
2387F	0	0	1.650	1326	1326	1856	0	0	1856
2519	0	0	1.625	1306	1306	1828	0	0	1828
2746.5	0	0	1.590	1947	1947	2726	0	0	2726
2832A	0	0	1.567	1918	1918	2685	0	0	2685
2832F	0	0	.555	680	680	952	31.0	3069	-2117
3100.55A	0	0	.528	646	646	904	31.0	3069	-2165
3100.55F	0	0	.523	641	641	897	0	0	897
3222.55	0	0	.516	631	631	883	0	0	883
3258.55	0	0	.498	610	610	854	0	0	854

TABLE 7.5-II N_C ULTIMATE FOR J-2S/SATURN V LOW EARTH ORBIT VEHICLE

STATION (IN)	M x 10^{-6} (IN-LB)	$M/\pi R^2$ (LB/IN)	P x 10^{-6} (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN)	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
ON-PAD, FUELED, UNPRESSURIZED									
365A	106.5	865	6.036	4852	5717	8004	0	0	8004
365F	106.5	865	4.629	3721	4586	6420	0	0	6420
602A	88.5	719	4.598	3696	4415	6181	0	0	6181
602F	88.5	719	4.594	3693	4412	6177	0	0	6177
912A	69.5	564	4.560	3665	4229	5921	0	0	5921
912F	69.5	564	1.373	1104	1668	2335	0	0	2335
1401A	43.0	349	1.343	1080	1429	2001	0	0	2001
1401F	43.0	349	1.338	1075	1424	1994	0	0	1994
1541	37.5	304	1.330	1069	1373	1922	0	0	1922
1760	28.0	227	1.315	1057	1284	1798	0	0	1798
1848A	26.0	211	1.273	1023	1234	1728	0	0	1728
1848F	26.0	211	.277	223	434	608	0	0	608
2387A	11.5	93	.253	203	296	414	0	0	414
2387F	11.5	93	.250	201	294	412	0	0	412
2519	9.5	77	.244	196	273	382	0	0	382
2746.5	6.0	113	.172	211	324	454	0	0	454
2832	4.5	85	.154	189	274	384	0	0	384
3100.55	2.0	38	.099	121	159	223	0	0	223
3222.55	1.5	28	.075	92	120	168	0	0	168
3258.55	1.3	24	.067	82	106	148	0	0	148

TABLE 7.5-II (Continued)

STATION (IN)	$M \times 10^{-6}$ (IN-LB)	$M/\pi R^2$ (LB/IN)	$P \times 10^{-6}$ (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN)	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
REBOUND (EMERGENCY SHUTDOWN)									
365A	84	682	7.080	5691	6373	8922	0	0	8922
365F	84	682	5.440	4373	5055	7077	9.3	921	6156
602A	68	552	5.380	4324	4876	6826	9.3	921	5905
602F	68	552	5.375	4320	4872	6821	0	0	6821
912A	51	414	5.340	4292	4706	6588	0	0	6588
912F	51	414	1.590	1278	1692	2369	3.3	327	2042
1401A	30	244	1.555	1250	1494	2092	3.3	327	1765
1401F	30	244	1.553	1248	1492	2089	0	0	2089
1541	25	203	1.547	1243	1446	2024	0	0	2024
1760	18.3	149	1.530	1230	1379	1931	0	0	1931
1848A	16.2	132	1.480	1190	1322	1851	0	0	1851
1848F	16.2	132	.319	256	388	543	19.3	1911	-1368
2387A	8.5	69	.291	234	303	424	19.3	1911	-1487
2387F	8.5	69	.288	231	300	420	0	0	420
2519	7.2	58	.288	231	289	405	0	0	405
2746.5	4.8	90	.201	246	336	470	0	0	470
2832	4	75	.180	220	295	413	0	0	413
3100.55	2	38	.116	142	180	252	0	0	252
3222.55	1.5	28	.089	109	137	192	0	0	192
3258.55	1.3	25	.080	98	123	172	0	0	172

TABLE 7.5-II (Continued)

STATION (IN)	$M \times 10^{-6}$ (IN-LB)	$M/\pi R^2$ (LB/IN)	$P \times 10^{-6}$ (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN)	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
MAX ($q\alpha$) NOMINAL FLIGHT									
365A	149	1211	7.813	6280	7491	10,487	0	0	10,487
365F	149	1211	6.358	5111	6322	8851	20.39	2019	6832
602A	228	1851	6.294	5059	6910	9674	20.39	2019	7655
602F	228	1851	6.285	5052	6903	9664	0	0	9664
912A	350	2844	6.214	4995	7839	10,975	0	0	10,975
912F	350	2844	2.924	2350	5194	7272	14.39	1425	5847
1401A	340	2761	2.859	2298	5059	7083	14.39	1425	5658
1401F	340	2761	2.849	2290	5051	7071	0	0	7071
1541	325	2639	2.830	2275	4914	6880	0	0	6880
1760	307	2489	2.800	2251	4740	6636	0	0	6636
1848A	296	2403	2.715	2182	4585	6419	0	0	6419
1848F	296	2403	.759	610	3013	4218	27.5	2722	1496
2387A	161	1307	.705	567	1874	2624	27.5	2722	-98
2387F	161	1307	.699	562	1869	2617	0	0	2617
2519	134	1091	.685	551	1642	2299	0	0	2299
2746.5	96	1803	.413	506	2309	3233	0	0	3233
2832	87	1639	.378	463	2102	2943	0	0	2943
3100.55	52	986	.267	327	1313	1838	0	0	1838
3222.55	38	720	.216	265	985	1379	0	0	1379
3258.55	34	649	.201	246	895	1253	0	0	1253

TABLE 7.5-II (Continued)

STATION (IN)	$M \times 10^{-6}$ (IN-LB)	$M/\pi R^2$ (LB/IN)	$P \times 10^{-6}$ (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN)	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
CECO (MAXIMUM ACCELERATION)									
365A	0	0	7.711	6198	6198	8677	0	0	8677
365F	0	0	7.306	5873	5873	8222	24.0	2376	5846
602A	0	0	7.167	5761	5761	8065	24.0	2376	5689
602F	0	0	7.148	5746	5746	8044	0	0	8044
912A	0	0	6.998	5625	5625	7875	0	0	7875
912F	0	0	6.106	4908	4908	6871	18.0	1782	5089
1401A	0	0	5.974	4802	4802	6723	18.0	1782	4941
1401F	0	0	5.952	4784	4784	6698	0	0	6698
1541	0	0	5.914	4754	4754	6656	0	0	6656
1760	0	0	5.851	4703	4703	6584	0	0	6584
1848A	0	0	5.662	4551	4551	6371	0	0	6371
1848F	0	0	1.237	994	994	1392	27.5	2723	-1331
2387A	0	0	1.132	910	910	1274	27.5	2723	-1449
2387F	0	0	1.118	899	899	1259	0	0	1259
2519	0	0	1.091	877	877	1228	0	0	1228
2746.5	0	0	.765	937	937	1312	0	0	1312
2832	0	0	.688	842	842	1179	0	0	1179
3100.55	0	0	.444	544	544	762	0	0	762
3222.55	0	0	.334	409	409	573	0	0	573
3258.55	0	0	.300	367	367	514	0	0	514

TABLE 7.5-III N_C ULTIMATE FOR J-2S/SATURN V TWO STAGE POLAR ORBIT VEHICLE

STATION (IN)	$M \times 10^{-6}$ (IN-LB)	$M/\pi R^2$ (LB/IN)	$P \times 10^{-6}$ (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN)	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
ON PAD FUELED UNPRESS.									
365A	106.5	865	5.999	4,825	5690	7966	0	0	7966
365F	106.5	865	4.593	3,694	4559	6383	0	0	6383
602A	88.5	719	4.561	3,668	4387	6142	0	0	6142
602F	88.5	719	4.557	3,665	4384	6138	0	0	6138
912A	69.5	564	4.523	3,637	4201	5881	0	0	5881
912F	69.5	564	1.336	1,074	1638	2293	0	0	2293
1401A	43.0	349	1.306	1,050	1399	1959	0	0	1959
1401F	43.0	349	1.301	1,046	1395	1953	0	0	1953
1541	37.5	304	1.293	1,040	1344	1882	0	0	1882
1760	28.0	227	1.279	1,029	1256	1758	0	0	1758
1848A	26.0	211	1.236	994	1205	1687	0	0	1687
1848F	26.0	211	.240	193	404	566	0	0	566
2387A	11.5	93	.217	175	268	375	0	0	375
2387F	11.5	93	.214	172	265	371	0	0	371
2519	9.5	77	.208	167	244	342	0	0	342
2746.5	6.0	113	.146	179	292	409	0	0	409
2832	4.5	85	.131	160	245	343	0	0	343
3100.55	2.0	38	.085	104	142	199	0	0	199
3222.55	1.5	28	.064	78	106	148	0	0	148
3258.55	1.3	24	.058	71	95	133			133

TABLE 7.5-III (Continued)

STATION (IN)	$M \times 10^{-6}$ (IN-LB)	$M/\pi R^2$ (LB/IN)	$P \times 10^{-6}$ (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN)	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
REBOUND									
365A	84.0	682	7.043	5665	6347	8886	0	0	8886
365F	84.0	682	5.403	4346	5028	7039	9.3	921	6118
602A	68.0	552	5.344	4298	4850	6790	9.3	921	5869
602F	68.0	552	5.344	4298	4850	6790	0	0	6790
912A	51.0	414	5.304	4266	4680	6552	0	0	6552
912F	51.0	414	1.546	1243	1657	2320	3.3	327	1993
1401A	30.0	244	1.512	1216	1460	2044	3.3	327	1717
1401F	30.0	244	1.511	1215	1459	2043	0	0	2043
1541	25.0	203	1.503	1209	1412	1977	0	0	1977
1760	18.3	149	1.486	1195	1344	1882	0	0	1882
1848A	16.2	132	1.436	1155	1287	1802	0	0	1802
1848F	16.2	132	.277	223	355	497	19.3	1911	-1414
2387A	8.5	69	.249	200	269	377	19.3	1911	-1534
2387F	8.5	69	.246	198	267	374	0	0	374
2519	7.2	58	.243	195	253	354	0	0	354
2746.5	4.8	90	.171	209	299	419	0	0	419
2832	4.0	75	.154	189	264	370	0	0	370
3100.55	2.0	38	.100	122	160	224	0	0	224
3222.55	1.5	28	.075	92	120	168	0	0	168
3258.55	1.3	25	.069	85	110	154	0	0	154

TABLE 7.5-III (Continued)

STATION (IN)	$M \times 10^{-6}$ (IN-LB)	$M/\pi R^2$ (LB/IN)	$P \times 10^{-6}$ (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN)	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
MAX ($q\alpha$) NOMINAL FLIGHT									
365A	158.3	1285	7.815	6282	7567	10,594	0	0	10,594
365F	158.3	1285	6.339	5095	6380	8,932	20.43	2023	6,909
602A	252.1	2047	6.274	5043	7090	9,926	20.43	2023	7,903
602F	252.1	2047	6.265	5036	7083	9,916	0	0	9,916
912A	371.7	3018	6.195	4980	7998	11,197	0	0	11,197
912F	371.7	3018	2.855	2295	5313	7,438	14.43	1429	6,009
1401A	361.0	2931	2.790	2243	5174	7,244	14.43	1429	5,815
1401F	361.0	2931	2.781	2235	5166	7,232	0	0	7,232
1541	345.1	2802	2.762	2220	5022	7,031	0	0	7.031
1760	325.8	2645	2.732	2196	4841	6,777	0	0	6,777
1848A	316.7	2572	2.646	2127	4699	6,579	0	0	6,579
1848F	316.7	2572	.682	548	3120	4,368	27.5	2722	1,646
2387A	171.5	1392	.628	505	1897	2,656	27.5	2722	-66
2387F	171.5	1392	.622	500	1892	2,649	0	0	2,649
2519	143.2	1163	.608	489	1652	2,313	0	0	2,313
2746.5	102.7	1935	.363	444	2379	3,331	0	0	3,331
2832	93.0	1752	.332	407	2159	3,023	0	0	3,023
3100.55	55.9	1052	.238	291	1343	1,880	0	0	1,880
3222.55	40.8	769	.194	238	1007	1,410	0	0	1,410
3258.55	36.8	694	.181	222	916	1,282	0	0	1,282

TABLE 7.5-III (Continued)

STATION (IN)	$M \times 10^{-6}$ (IN-LB)	$M/\pi R^2$ (LB/IN)	$P \times 10^{-6}$ (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN)	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
MAXIMUM ACCELERATION (CECO)									
365A	0	0	7.697	6187	6187	8662	0	0	8662
365F	0	0	7.268	5842	5842	8179	24.0	2376	5803
602A	0	0	7.127	5729	5729	8021	24.0	2376	5645
602F	0	0	7.107	5713	5713	7998	0	0	7998
912A	0	0	6.957	5592	5592	7829	0	0	7829
912F	0	0	6.009	4830	4830	6762	18.0	1782	4980
1401A	0	0	5.876	4723	4723	6612	18.0	1782	4830
1401F	0	0	5.853	4705	4705	6587	0	0	6587
1541	0	0	5.815	4674	4674	6544	0	0	6544
1760	0	0	5.751	4623	4623	6472	0	0	6472
1848A	0	0	5.560	4469	4469	6257	0	0	6257
1848F	0	0	1.084	871	871	1219	27.5	2722	-1503
2387A	0	0	.977	785	785	1099	27.5	2722	-1623
2387F	0	0	.963	774	774	1084	0	0	1084
2519	0	0	.936	752	752	1053	0	0	1053
2746.5	0	0	.658	805	805	1127	0	0	1127
2832	0	0	.592	725	725	1015	0	0	1015
3100.55	0	0	.384	470	470	658	0	0	658
3222.55	0	0	.288	353	353	494	0	0	494
3258.55	0	0	.261	319	319	447	0	0	447

7.5.2 (Continued)

The compressive structural capabilities as supplied by the associate stage contractors are shown with the design loads. Structural loads are less than capability for all J-2S/Saturn V design vehicles except in the S-IC fuel tank for the two-stage polar orbit mission. Structural "beef-up" could be used to increase the fuel tank strength or load alleviation techniques could be used to reduce ultimate compressive loads. "Beef-up" would be accomplished by revising mill tapes to increase structure cross-section. The S-IC stage weight would increase approximately 74 pounds.

Critical load parameters of the synchronous orbit vehicle are bounded by the LOR and LEO parameters. As the loading for these latter vehicles did not exceed the structural capability of their individual stage, it was considered unnecessary to perform a loads analysis for the synchronous mission.

The design conditions considered in determining the ultimate compressive combined loads are as follows:

a. Shear Distribution

The ground wind shear distribution for a 99.9 percent probable prelaunch wind and a 99 percent probable launch wind were provided by MSFC, References 7.5-2 and 7.5-3.

b. Ground Wind Bending Moment Distributions

The ground wind bending moment distributions for a 99.9 percent probable pre-launch wind and a 99 percent probable launch wind were also provided by MSFC, References 7.5-2 and 7.5-3.

c. Inflight Bending Moment Distributions

The maximum inflight bending moment distributions were determined from a flight simulation of the first stage boost phase of each mission. The digital simulation used six degrees of freedom to invesitgate flexible body vehicle responses. Included in the simulation were: distributed nonlinear aerodynamics, four flexible modes, fuel slosh motions, wind penetration capability, and linear third-order engine actuators. The flight wind profile (shown in Figure 7.5-1) was considered to act in the vehicle yaw plane.

d. On-Pad Longitudinal Force Distributions

For the on-pad ground wind fueled-unpressurized condition, the longitudinal force at any vehicle station is equal to the vehicle weight forward of that station (load factor equal 1 g).

7.5.2 (Continued)

e. Longitudinal Force Distributions for Emergency Shutdown Condition

Longitudinal force at any vehicle station for the emergency shutdown (rebound) condition is equal to the product of vehicle weight forward of that station times the rebound load factors. The rebound distributed load factors were provided by MSFC, Reference 7.5-3.

f. Longitudinal Force Distributions for Nominal Maximum $q\alpha$ Product condition

The longitudinal force distribution for the nominal maximum $q\alpha$ product conditions is defined as follows:

$$P(X) = \eta W(X) + D(X)$$

Where:
- $P(X)$ = longitudinal force at any vehicle station
- η = acceleration load factor at design condition
- $W(X)$ = vehicle weight forward of any vehicle station
- $D(X)$ = drag force effective at any vehicle station

g. Longitudinal Force Distribution for Maximum Acceleration Condition

The longitudinal force at any vehicle station for the maximum acceleration (first stage center engine cut-off) condition is equal to the product of the vehicle weight forward of that station times the acceleration load factor.

7.5.3 Structural Dynamics

7.5.3.1 Technical Approach

Lateral vibration properties to be used in the structural loads and vehicle control analyses were computed. Free-free mode shapes, slopes and frequencies for the first through the fourth mode were computed at liftoff, maximum $q\alpha$ product, first stage cutoff, upper stage ignition, and upper stage burnout conditions. Calculations were based upon lumped weights, cantilevered weights with their equivalent rotary inertias, and vehicle bending and shear stiffness supplied by associate stage contractors.

7.5.3.2 Results

The natural frequencies and generalized masses for the first four free-free natural modes for the LOR, LEO and two-stage polar orbit vehicle are shown in Tables 7.5-IV, 7.5-V and 7.5-VI, respectively. Modal deflections and modal slopes for these three J-2S/Saturn V vehicles are presented in Document D5-15772-6, Appendix C.

TABLE 7.5-IV FREQUENCY AND GENERALIZED MASS FOR J-2S/ SATURN V LOR VEHICLE

FLIGHT CONFIGURATION	MODE 1		MODE 2		MODE 3		MODE 4	
	FREQ. (CPS)	GEN.MASS $\frac{LB\text{-}SEC^2}{IN}$	FREQ. (CPS)	GEN. MASS $\frac{LB\text{-}SEC^2}{IN}$	FREQ. (CPS)	GEN. MASS $\frac{LB\text{-}SEC^2}{IN}$	FREQ. (CPS)	GEN. MASS $\frac{LB\text{-}SEC^2}{IN}$
S-IC Lift-Off	.9203	114.0141	1.6757	58.2133	2.4830	29.7206	3.4104	13.0961
S-IC MAX ($Q\alpha$)	.9868	92.6615	1.8541	48.8295	2.5551	34.7858	3.4131	13.1617
S-IC Cut-Off	1.0713	79.5311	2.1130	27.9572	3.3367	11.5396	6.2437	72.1828
S-II Ignition	1.6304	28.3400	3.2147	9.7200	6.0986	42.7400	7.5146	92.2600
S-II Burn-Out	2.2296	145.025	6.1208	87.75	10.7220	28.86	14.3625	4,812.27
S-IVB Ignition	4.7502	74.2590	10.4829	26.6840	13.8948	1008.2930	33.2317	547,959.5
S-IVB Burn-Out	5.2938	75.1640	10.7233	26.0350	22.7804	309,300.0	49.4051	3.628×10^6

TABLE 7.5-V FREQUENCY AND GENERALIZED MASS FOR J-2S/ SATURN V LOW EARTH ORBIT VEHICLE

FLIGHT CONFIGURATION	MODE 1		MODE 2		MODE 3		MODE 4	
	FREQ. (CPS)	GEN.MASS LB-SEC2/IN	FREQ. (CPS)	GEN.MASS LB-SEC2/IN	FREQ. (CPS)	GEN.MASS LB-SEC2/IN	FREQ. (CPS)	GEN.MASS LB-SEC2/IN
S-IC LIFT-OFF	1.0971	76.5611	1.8826	33.6196	2.6222	17.6597	4.4876	45.7632
S-IC Max ($Q\alpha$)	1.1866	58.8544	2.0554	29.9809	2.6578	20.4182	4.6005	23.3179
S-IC CUT-OFF	1.2922	47.1344	2.3989	13.7265	4.4044	22.4417	6.6970	88.7977
S-II IGNITION	2.0320	12.6920	3.9770	21.4830	7.0618	31.9160	10.2707	44.3370
S-II BURN-OUT	3.5597	44.6530	7.5995	36.6550	12.8620	60.1250	18.0159	79.4080

TABLE 7.5-VI FREQUENCY AND GENERALIZED MASS FOR J-2S/SATURN V TWO STAGE POLAR ORBIT VEHICLE

FLIGHT CONFIGURATION	MODE 1		MODE 2		MODE 3		MODE 4	
	FREQ. (CPS)	GEN. MASS $\frac{LB-SEC^2}{IN}$	FREQ. (CPS)	GEN. MASS $\frac{LB-SEC^2}{IN}$	FREQ. (CPS)	GEN. MASS $\frac{LB-SEC^2}{IN}$	FREQ. (CPS)	GEN. MASS $\frac{LB-SEC^2}{IN}$
S-IC LIFT-OFF	1.1351	66.0239	1.9011	33.1088	2.6842	17.4517	4.5568	158.0583
S-IC MAX ($Q\alpha$)	1.2245	50.8604	2.0719	31.4971	2.7123	19.5473	4.8539	24.5440
S-IC CUT-OFF	1.3299	40.6101	2.4644	13.6932	4.6315	24.0970	6.8074	110.6553
S-II IGNITION	2.0709	11.6410	4.1722	17.8031	7.4679	25.1369	10.8112	42.5721
S-II BURN-OUT	3.8186	37.0190	8.1880	31.8730	13.8314	49.9260	19.3246	71.2917

7.5.4 Engine-Out Design Requirements (Malfunction Flight)

The effects of an engine-out malfunction during the maximum $q\alpha$ product region were determined for the LOR, LEO and polar vehicles. Two cases of this type were investigated: (a) all engines out; and (b) one engine out.

All engines out during this flight region will cause an obvious loss of vehicle control, vehicle tumbling and certain structural failure. This condition was studied to determine the time-to-failure (or mission abort capability) after the malfunction occurs. A specific amount of time is required to perform the sequence of events which successfully separates the Command Module from the vehicle after a malfunction occurs. Therefore, a successful mission abortion requires that the time-to-failure of the vehicle structure after a malfunction be greater than this minimum. Failure was defined as the condition when the structural factor of safety was reduced to one.

One engine-out during this flight time may, or may not, cause a loss of vehicle control. This condition was studied to see if single engine-out flight loads for controllable flight became a critical design parameter. No effort was made in the study to impose these one-engine-out flight loads upon the design load envelopes of the study vehicles.

The loss of an outboard engine when the vehicle is in the maximum $q\alpha$ product region causes a reduction of engine compressive forces. At this time, the vehicle is also in a state of high moment loading due to angle of attack, slosh, flexibility and control forces; and the bending moments are further increased when the malfunction occurs. This combination of increasing bending moments and decreasing engine compressive forces may cause the vehicle tension loads to exceed the structure tensile capability.

The inflight wind profile shown in Figure 7.5-1 was used in the analysis of both the nominal and malfunction flight conditions. This 95 percentile design wind was assumed to act in the vehicle yaw plane. A 1.1 factor of safety was applied to the malfunction flight loads. The most critical time for the malfunction to occur was determined to be 0.2 seconds before peak wind speed is reached.

7.5.4.1 One Engine-Out Malfunction at Maximum $q\alpha$ Product

Uncontrollable Flight

This study has determined that the loss of either engine number one or number two will cause all of the study vehicles to experience a loss of control followed by tumbling and structural failure. The loss of engine number one at maximum $q\alpha$ product will cause the most severe one engine-out instability condition.

7.5.4.1 (Continued)

Engine number one malfunction time-to-failure for the two vehicle stations (2746.5 and 2832 AFT) that experience the earliest tension failure are shown in Table 7.5-VII. Time-to-failure is defined as the time from start of malfunction to the time when the structure factor of safety is reduced to one. Also shown is the time from malfunction to the time Launch Escape Vehicle (LEV) angle of attack limit is reached. Abort limits for S-IC flight are described in terms of the LEV angle of attack limit at the altitude at which abort occurs. This constraint is due to the limiting over-pressures on the command module after separation from the launch vehicle.

Controllable Flight

Failure of upwind engines number three or number four will not cause a loss of vehicle control. The loss of engine number three will produce the maximum inflight malfunction loads for this controllable flight condition. The ultimate compression combined loads and ultimate tensile combined loads for number three engine out malfunction were determined as follows:

a. Ultimate Compressive Combined Loads

Ultimate compressive combined loads are determined as follows:

$$N_{C_{ULT}} = \left[\frac{P(X)}{2\pi R(X)} + \frac{BM(X)}{\pi R^2 (X)} \right] F.S. - P_{u_{min}} \frac{R(X)}{2}$$

Where:

P(X)	=	distributed longitudinal forces including aerodynamic dynamic forebody drag
BM(X)	=	distributed bending moment
R(X)	=	distributed body radius
$P_{U_{MIN}}$	=	minimum ullage pressure (applicable to tank shells only)
F.S.	=	ultimate factor of safety of 1.4 for nominal flight loads and 1.1 for one-engine-out flight loads

Ultimate compressive combined loads for a one-engine-out malfunction are not a design condition for any of the J-2S design vehicles.

b. Ultimate Tensile Combined Loads

Ultimate tensile combined loads are determined as follows:

TABLE 7.5-VII FAILURE TIMES AFTER ENGINE MALFUNCTION (MAXIMUM $q\alpha$ PRODUCT FLIGHT REGION WITH A UNCONTROLLABLE VEHICLE)

J-2S STUDY VEHICLE	NUMBER ONE ENGINE OUT MALFUNCTION TIME-TO-FAILURE AFTER MALFUNCTION ~ SEC. TENSION FAILURE @ STA. 2746.5	TENSION FAILURE @ STA. 2832 AFT	LEV. ANGLE OF ATTACK LIMIT REACHED
LOR	3.47	3.55	2.50
LEO	2.49	2.53	1.95
POLAR	1.78	2.15	1.70

J-2S STUDY VEHICLE	ALL ENGINES OUT MALFUNCTION TIME-TO-FAILURE AFTER MALFUNCTION ~ SEC. TENSION FAILURE @ STA. 1564	TENSION FAILURE @ STA. 1760	LEV. ANGLE OF ATTACK LIMIT REACHED
LOR	0.97	1.15	2.30
LEO	0.31	0.79	1.75
POLAR	0.25	0.25	1.50

7.5.4.1 (Continued)

$$N_{T_{ULT}} = \left[\frac{BM(X)}{\pi R^2(X)} - \frac{P(X)}{2\pi R} + P_{u_{max}} \frac{R(X)}{2} \right] F.S.$$

Where:

P(X)	=	distributed longitudinal forces including aerodynamic forebody drag
BM(X)	=	distributed bending moment
R(X)	=	distributed body radius
$P_{U_{MAX}}$	=	maximum ullage pressure (applicable to tank shells only)
F.S.	=	ultimate factor of safety of 1.4 for nominal flight loads and 1.1 for one-engine flight loads.

The loss of the upwind engine number 3 at 0.2 seconds before the peak wind speed gave the maximum tension load for a one-engine-out malfunction with a controllable vehicle.

A tabulation of the one-engine-out tension design criteria for the critical vehicle stations is shown in Table 7.5-VIII. These data show the critical stations, the structural tensile capability at those stations, the maximum limit tensile combined load, the factor of safety associated with each of the maximum limit loads, and the time from the start of the malfunction to structural overload (structural overload occurring when the factor of safety is less than one). The data show that vehicle station 1564 and vehicle station 1760 exceed tension capability. The other critical stations maintain a factor of safety greater than 1.1 which is the safety factor for one-engine-out malfunction flight loads. The controllable, one-engine-out malfunction at maximum ($q\alpha$) product is a tension design requirement.

7.5.4.2 All Engine-Out Malfunction at Maximum $q\alpha$ Product

An all-engines-out malfunction at maximum $q\alpha$ product resulted in vehicle tumbling and structural failure. Vehicle stations 1564 and 1760 experienced the earliest tension failure due to the malfunction. The time-to-failure after malfunction along with the time to reach Launch Escape Vehicle (LEV) angle of attack limits are shown in Table 7.5-VII.

TABLE 7.5-VIII TENSION DESIGN CRITERIA (NUMBER THREE ENGINE OUT MALFUNCTION AT MAXIMUM $q\alpha$ PRODUCT WITH A CONTROLLABLE VEHICLE)

LOR				
STATION	MAXIMUM N_T LIMIT (LB/IN)	TENSION CAPABILITY (LB/IN)	FACTOR OF SAFETY	TIME FROM MALFUNCTION TO STRUCTURAL FAILURE
1541	1319	2040	1.547	—
1564	1310	1200	0.916	0.83 Sec.
1760	1071	1250	1.167	—
2746.5	1641	1858	1.132	—
2832A	1490	1811	1.215	—

LEO				
STATION	MAXIMUM N_T LIMIT (LB/IN)	TENSION CAPABILITY (LB/IN)	FACTOR OF SAFETY	TIME FROM MALFUNCTION TO STRUCTURAL FAILURE
1541	1550	2040	1.316	—
1564	1541	1200	0.779	0.50 Sec.
1760	1300	1250	0.962	0.60 Sec.

	POLAR			
STATION	MAXIMUM N_T LIMIT (LB/IN)	TENSION CAPABILITY (LB/IN)	FACTOR OF SAFETY	TIME FROM MALFUNCTION TO STRUCTURAL FAILURE
1541	1640	2040	1.244	—
1564	1629	1200	0.736	0.43 Sec.
1760	1390	1250	0.899	0.56 Sec.

7.6 HEATING ENVIRONMENT

Thermal design criteria for the J-2S/Saturn V design vehicles were provided by MSFC, Reference 7.6-1. A comparison of the relative aerodynamic heating for different launch vehicles is provided by the equation:

$$AHI = \int_{o}^{t} \frac{qv}{\frac{\pi}{2} - \alpha} dt$$

where:

AHI = aerodynamic heating indicator

q = dynamic pressure

v = velocity

α = angle of attack in radians

t = flight time

The aerodynamic heating indicator is a term developed to give relative aerodynamic heating effects on geometrically similar vehicle with dissimilar trajectories. It is further defined as being proportional to the time integrated heat input. Therefore, for two vehicles with the same AHI, the total heat input is assumed to be equal and the structural temperatures generally are assumed to be within the same temperature range.

The design trajectories for the design J-2S/Saturn V vehicles presented in Paragraph 7.2 include the nominal AHI histories. Design maximum AHI's were obtained by increasing the nominal AHI values 13 percent. Figure 7.6-1 compares these maximum AHI's with those of the SA-501 to 503 vehicles and SA-504 to 510 vehicles.

The SA-501 to 503 vehicle thermal environment was used for the LOR, LEO and synchronous missions and the SA-504 to 510 vehicle thermal environment was used for the two-stage polar mission. Protuberance heating factors for the study vehicles was provided by MSFC, Reference 7.6-2.

Upper stage contractors assessed their stage for thermal impact based on the heating environments defined above. No upper stage aerodynamic heating problems were encountered. Per MSFC direction, no heating analysis of the S-IC stage was made.

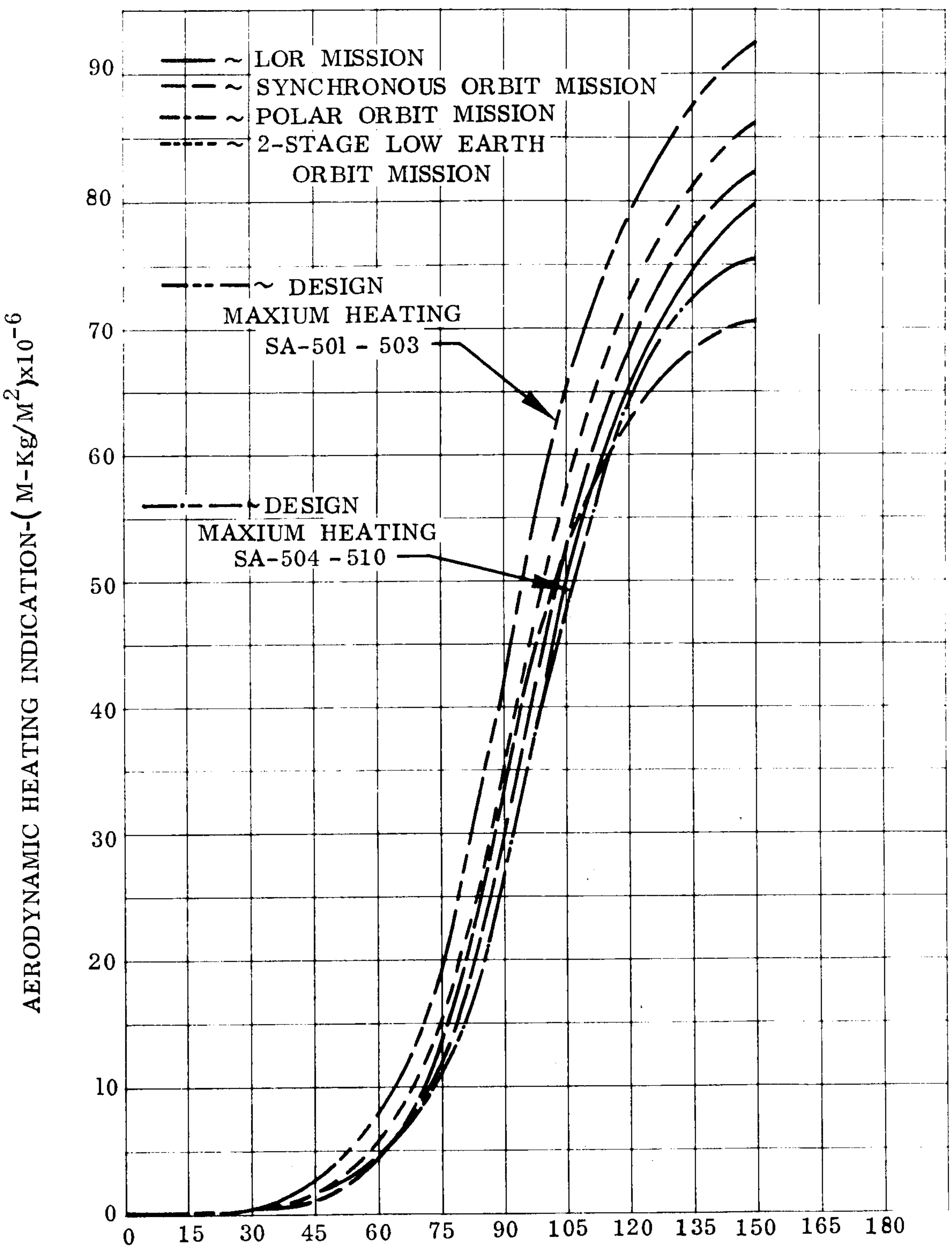


FIGURE 7.6-1 AERODYNAMIC HEATING INDICATOR HISTORIES FOR J-2S IMPROVEMENT STUDY MISSIONS (MAXIMUM HEATING)

7.7 VEHICLE CONTROL

Rigid body control requirements and control system dynamic characteristics were determined for the J-2S Lunar Orbit Rendezvous (LOR), Low Earth Orbit (LEO), and Polar Orbit (PO) vehicle configurations. Vehicle weight and mass characteristics required to perform these control analyses are given in paragraph 7.3. Structural dynamics characteristics are described, in detail, in D5-15772-6, Structural Analyses - J-2S Improvement Study. The S-IC and S-II outboard engines are gimbaled to provide vehicle control. A gimbaled engine and auxiliary propulsion system provide control during S-IVB stage operation.

7.7.1 Rigid Body Controls

Flight control system gains for first stage flight were calculated using a mathematical model of a rigid vehicle employing the attitude/attitude-rate control mode. Root summed square (RSS) thrust deflections (β) have been determined for the maximum dynamic pressure (q) portion of the trajectory. The 95 percent Apollo design wind was used in obtaining these results.

7.7.1.1 Gains

Figures 7.7-1 through 7.7-3 present the flight control system gains. The gains were chosen to yield a control frequency of 0.15 hertz and a damping ratio of 0.7. The low control frequency was chosen to avoid control system bending mode coupling.

7.7.1.2 Root Summed Square (RSS) Thrust Deflections

Figures 7.7-4 through 7.7-6 present the RSS thrust deflections (β) required to control the vehicles in the region of maximum dynamic pressure. The curves are envelopes of RSS thrust deflection obtained by applying the maximum inflight wind encountered by the vehicle at each time point along the trajectory. In actuality, the maximum inflight wind would be encountered only once during a flight. The curves present the maximum thrust deflection at whatever time the peak wind is encountered. The maximum thrust deflection requirements are well within the Saturn V capability of 5.15 degrees.

The RSS values were obtained by determining the additional thrust deflections required because of the inclusion of scatter parameters. These additional deflections were then added to the nominal value using the RSS technique. The following scatter terms were applied:

Thrust Imbalance	$\pm$ 1.5%
Thrust Misalignment	$\pm$ 0.122 degrees
Axial CG Offset	$\pm$ 7 inches

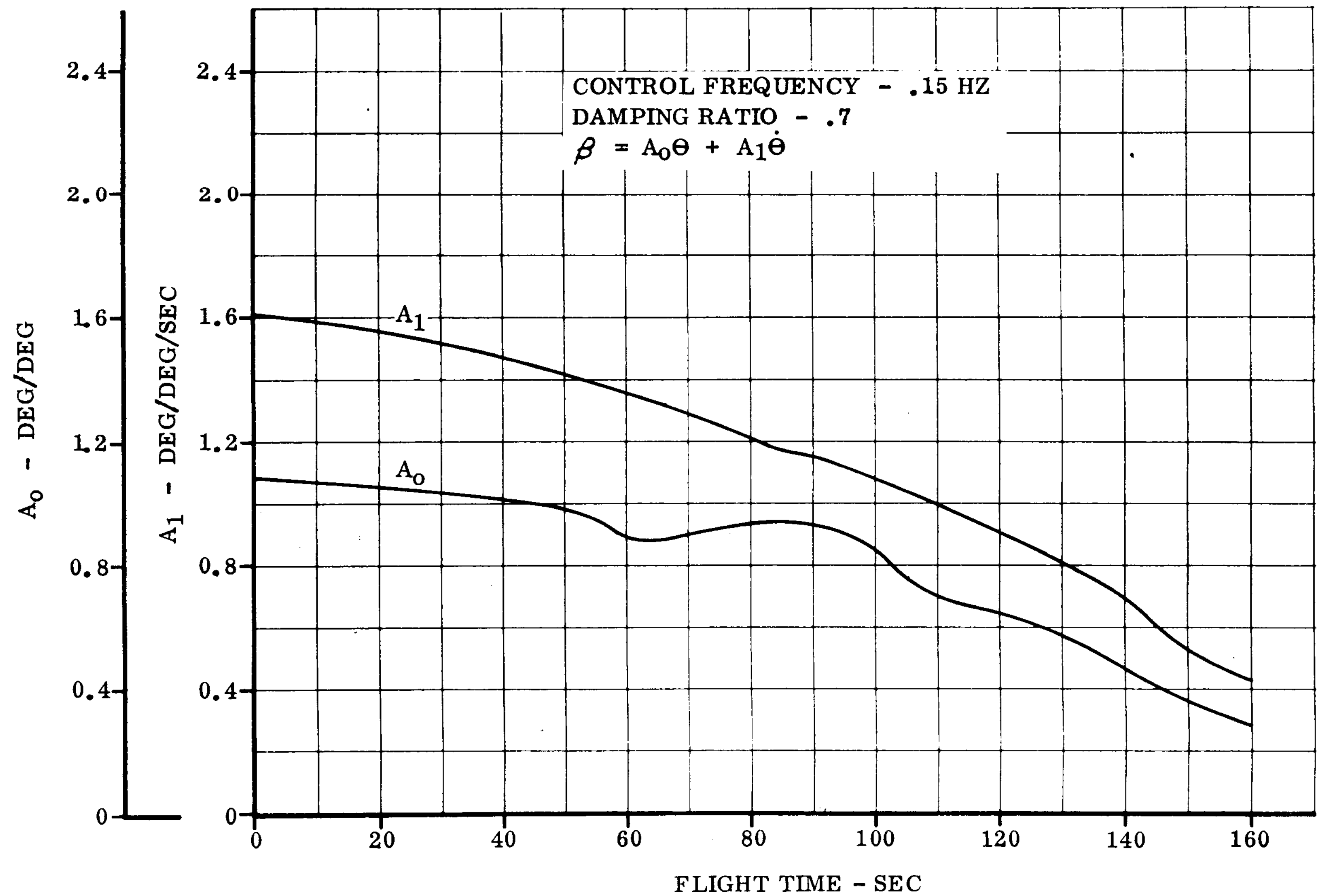


FIGURE 7.7-1 FLIGHT CONTROL SYSTEM GAINS FOR THE J-2S/LOR VEHICLE

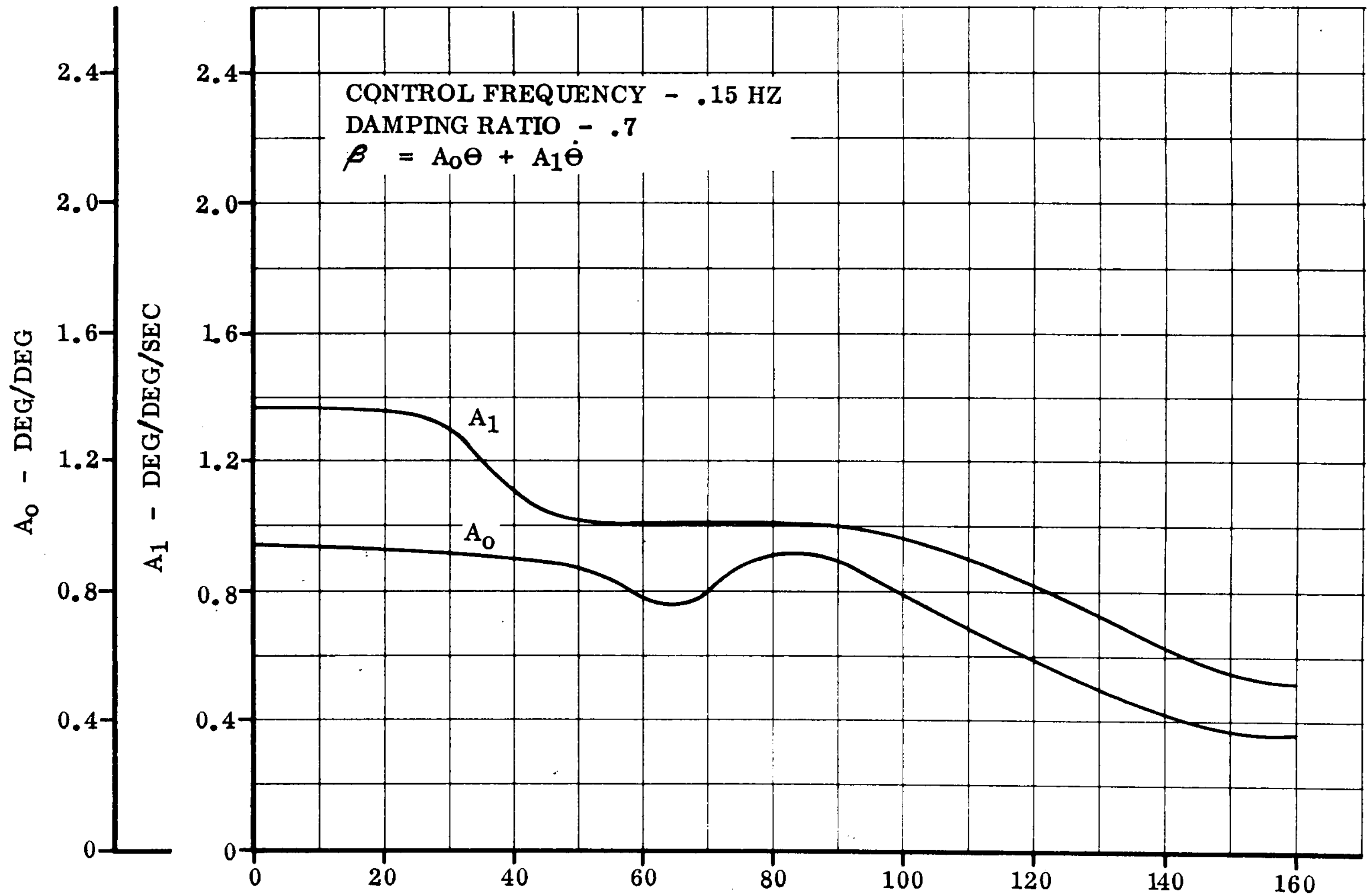


FIGURE 7.7-2 FLIGHT CONTROL SYSTEM GAINS FOR THE J-2S/LOW EARTH ORBIT VEHICLE

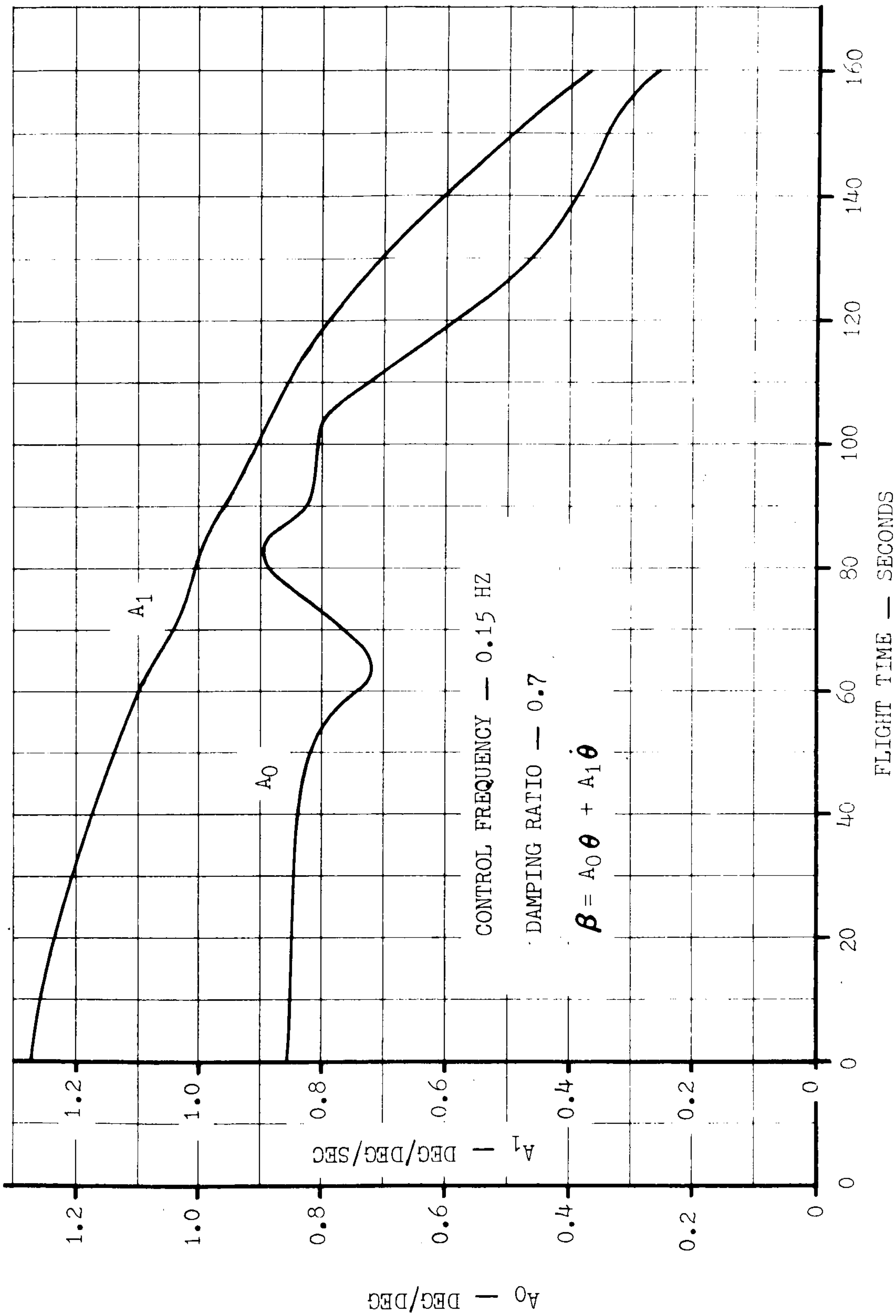


FIGURE 7.7-3 FLIGHT CONTROL SYSTEM GAINS FOR THE J-2S/POLAR ORBIT VEHICLE

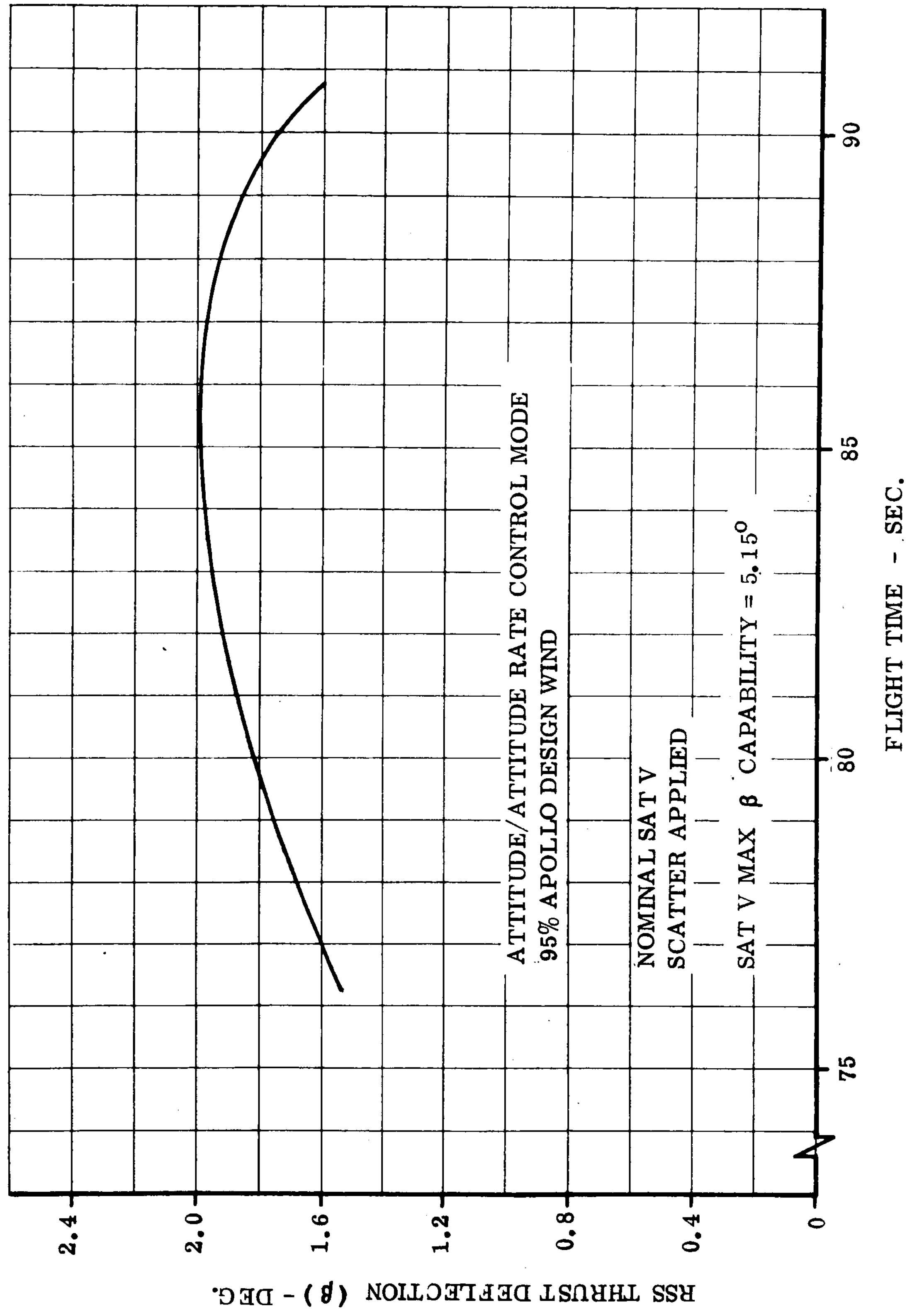


FIGURE 7.7-4 RSS THRUST DEFLECTION FOR THE J-2S/LOR VEHICLE

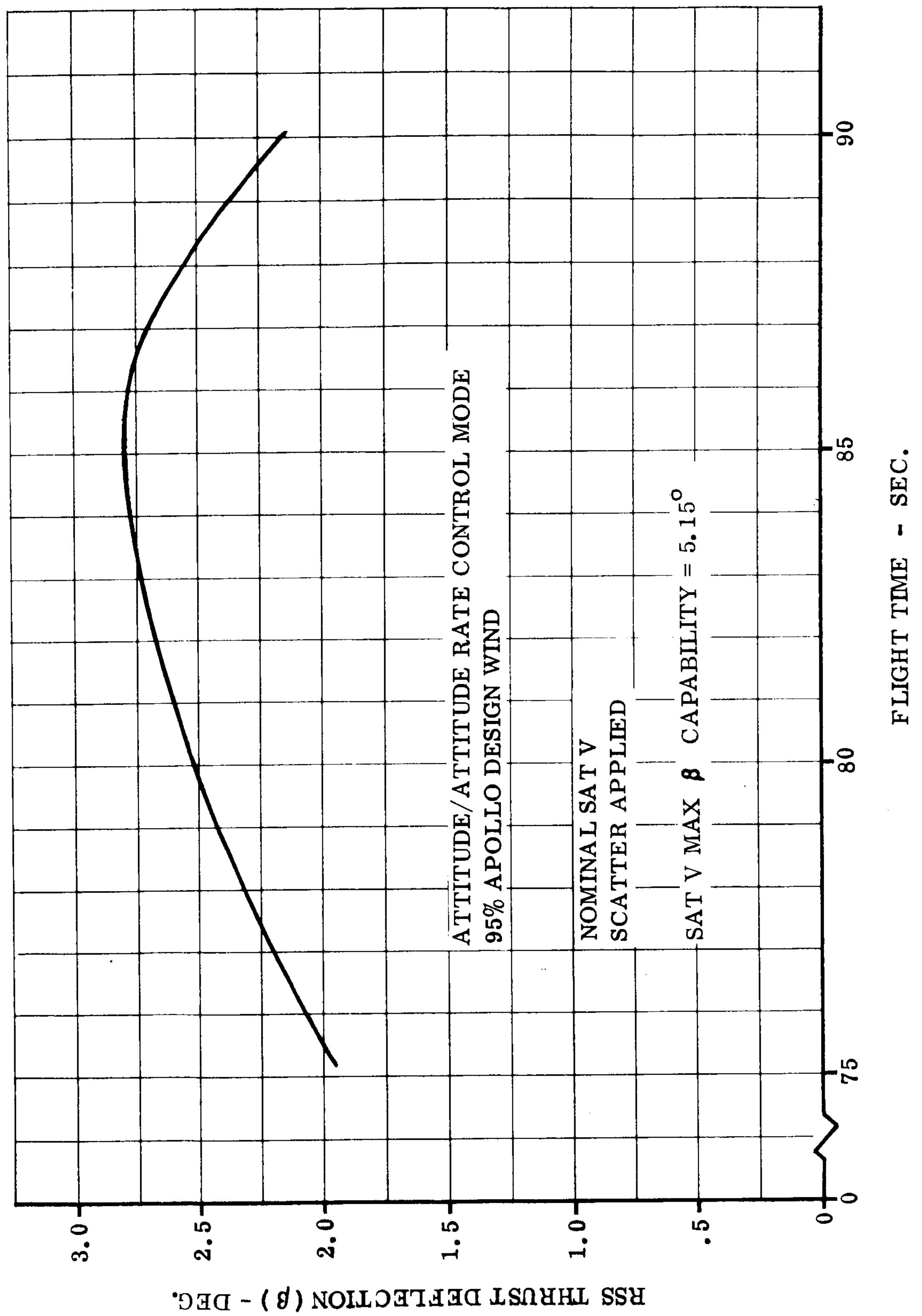


FIGURE 7.7-5 RSS THRUST DEFLECTIONS FOR THE J-2S LOW EARTH ORBIT VEHICLE

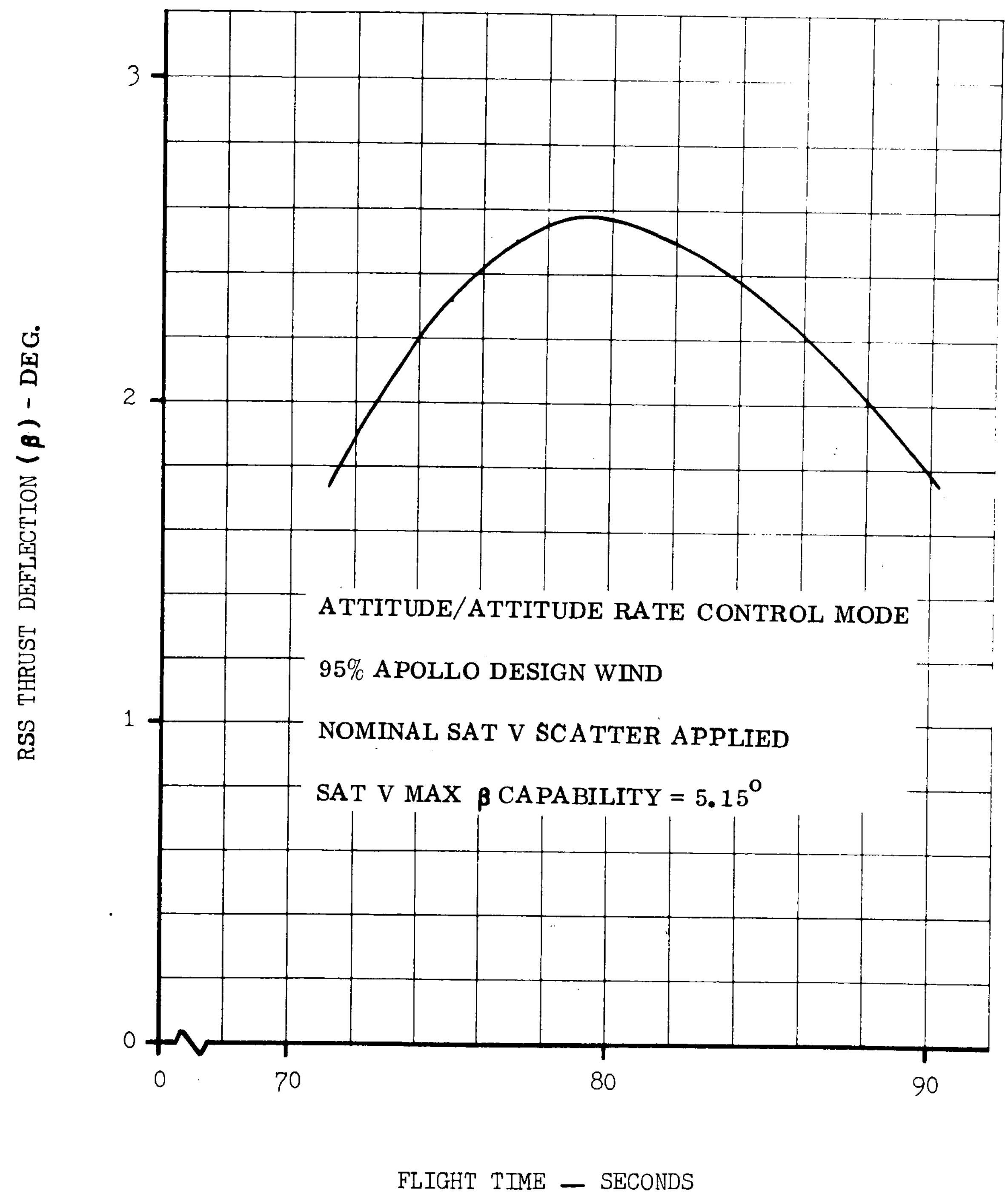


FIGURE 7.7-6 RSS THRUST DEFLECTION FOR THE J-2S POLAR ORBIT VEHICLE

7.7.1.2 (Continued)

Lateral CG Offset	± 2 inches
Normal Force Coefficient Variation	± 6%
Center of Pressure Variation	± 79 inches
Control System Gains Variation	± 10%

7.7.2 Control System Dynamics

A flight control system stability analysis was made to determine the difficulty of designing the compensator networks for the flight control system of the J-2S design vehicles. The difficulty is determined by comparing the uncompensated J-2S/Saturn V results with those of a typical J-2/Saturn V uncompensated system.

A frequency response for each vehicle was obtained for the following flight times as applicable.

a. First stage at q max

b. S-II Ignition

c. S-II Burnout

d. S-IVB Ignition

e. S-IVB Burnout

These time points represent the flight times of the most severe disturbances.

The basic attitude/attitude-rate feedback control system is used exclusively for these analyses. To determine the relative effects of the attitude and attitude rate feedbacks, a frequency response was obtained for attitude feedback only, attitude-rate feedback only, and for the combined attitude and attitude-rate feedback. The assumptions used are compatible with those used for Saturn V control system studies. Propellant slosh data were obtained from the respective stage contractors.

Uncompensated control system frequency response Nyquist and Bode plots were generated. These were compared to uncompensated Saturn V frequency response characteristics to indicate the degree of difficulty in designing compensator networks for the J-2S vehicles. The stability parameters compared were: aerodynamic gain margin; rigid body phase margin; slosh magnitude peak; first bending mode phase margin; first bending mode magnitude peak; second bending mode magnitude peak;

7.7.2 (Continued)

third bending mode magnitude peak; and fourth bending mode magnitude peak.

Comparison of the frequency responses for the J-2S and Saturn V vehicles shows that no difficulties will be encountered in designing compensators networks for the J-2S vehicles; i.e. compensator design for J-2S will be no more difficult than it was for Saturn V. In fact, it appears that some of the Saturn V compensators may be used for the J-2S vehicles with little or no modifications.

Detailed analyses of the control system dynamics including the Nyquist and Bode plots are contained in D5-15772-5, Vehicle Control and Control System Dynamic Characteristics, J-2S Improvement Study.

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SECTION 8

PERFORMANCE/MISSION ANALYSIS

8.0 GENERAL

This section contains general performance for the four J-2S/Saturn V design vehicles described in Section 6. Comparisons are made with the Baseline SA-511 Saturn V (J-2 engines).

8.1 LOR PERFORMANCE COMPARISONS

The initial J-2S/Saturn V LOR design trajectory, reported in Section 7.2, was based upon initial best estimates of stage changes required to accommodate the J-2S engines. This "nominal" design trajectory provided the payload performance and vehicle design characteristics for detailed analyses of loads, structures, acoustics, thermal conditions and control dynamics, etc. After all stage changes resulting from use of J-2S had been determined, a "final" design trajectory was run. Changes in vehicle design characteristics were so small that study results would not change. Performance changes are shown in Table 8.1-I. The final LOR payload is 108,953 pounds compared to 110,066 pounds for the nominal LOR design trajectory. The difference is due to changes in drop weight items, usable propellant loadings and mixture ratio schedules as shown in the table.

The mixture ratio trade indicated that (1) the optimum operation of the S-IVB stage is obtained with a constant mixture ratio which gives maximum usable impulse propellant, and (2) the optimum operation of the S-II stage is obtained with a mixture ratio schedule of 5.0 to 5.5 to 4.5. The 5.0 MR burn duration is 2.5 seconds. The 5.5 MR burn duration is such that maximum S-II impulse propellant will be consumed.

The drop weight differences reflect hardware and operational changes resulting from associate contractor detailed analyses of their stage systems. Results of these analyses are reported in Section 10.0, Stage/System Description.

Table 8.1-1 also shows the LOR performance of the Baseline SA-511 vehicle (100,078 lbs). The use of J-2S engines in the Saturn V vehicle yields a LOR net payload increase of approximately nine percent.

As discussed in Section 7.2, Design Trajectories, the S-II specific impulse was degraded by two seconds to account for the engine clustering effects. Flight data from the SA-501, SA-502 and SA-503 Saturn V vehicle indicates that S-II specific impulse is not degraded. The J-2S/Saturn LOR performance will increase approximately 900 pounds with no S-II specific impulse degradation. The LOR Baseline SA-511 would likewise increase with the same assumption.

TABLE 8.1-I COMPARISONS OF SATURN V LOR DESIGN CHARACTERISTICS AND WEIGHT SUMMARIES

		Baseline LOR J-2 Engines	Nominal LOR Design J-2S Engines	Final LOR Design J-2S Engines
S-IC STAGE				
Sea Level Thrust	lbs	7,610,064	7,610,064	7,610,064
Sea Level Specific Impulse	secs	264.5	264.5	264.5
Liftoff Weight	lbs	6,404,883	6,414,630	6,416,022
Impulse Propellant	lbs	4,577,113	4,577,113	4,577,113
Drop Weight	lbs	370,771	370,591	370,266
T/W_O		1.118	1.186	1.186
S-II STAGE				
MR Schedule		5.0 to 5.5 to 4.7	5.0 to 5.5 to 4.7	5.0 to 5.5 to 4.5
Vacuum Thrust, M.R. = 5.0	lbs	1,025,260	1,182,035	1,182,035
Vacuum Specific Impulse	secs	426.0	432.5	432.5
Weight at Ignition	lbs	1,456,999	1,466,326	1,468,643
Impulse Propellant	lbs	970,441	970,441	970,441
Forward Interstage Jettison Weight	lbs	9,435	9,427	8,336
Launch Escape System	lbs	8,936	8,936	8,936
Drop Weight	lbs	103,855	102,571	104,610
S-IVB STAGE				
First Burn Mainstage MR		5.5	5.0	4.905
First Burn Mainstage Vac. Thrust	lbs	231,012	237,500	232,314
First Burn Mainstage Specific Impulse	secs	424.9	434.5	435.1
Weight at First Ignition	lbs	364,331	374,952	376,320
Impulse Propellant Consumed to Orbit	lbs	71,872	64,737	66,990
Cutoff Weight in Orbit	lbs	292,459	310,215	309,330
Weight Drop in Parking Orbit	lbs	4,411	4,446	4,604
Weight at Ignition in Orbit	lbs	288,048	305,769	304,726
Second Burn Mainstage MR		5.0	5.0	4.905
Impulse Propellant Consumed from PO to Injection (Excluding Reserves)	lbs	152,648	160,696	159,870

TABLE 8.1-I COMPARISONS OF SATURN V LOR DESIGN CHARACTERISTICS AND WEIGHT SUMMARIES (Continued)

		Baseline LOR J-2 Engines	Nominal LOR Design J-2S Engines	Final LOR Design J-2S Engines
Total Impulse Consumed (Excluding Reserve)	lbs	224, 520	225, 433	226, 860
Cutoff Weight at Injection	lbs	135, 400	145, 073	144, 856
Drop Weight	lbs	27, 668	27, 160	27, 955
GROSS PAYLOAD				
Instrument Unit Weight	lbs	4, 183	4, 183	4, 301
Propellant Reserves (109 m/sec)	lbs	3, 471	3, 664	3, 647
NET PAYLOAD		100, 078	110, 066	108, 953
FLIGHT TRAJECTORY DATA				
Launch Azimuth	degs	72	72	72
Max q (85 secs)	lbs/ft^2	699	709	708
Max "g's"		4.15	4.14	4.12
AHI	$ft\text{-}lbs/ft^2\text{-}rad$	46.43×10^6	49.09×10^6	48.77×10^6

8.2 SYNCHRONOUS ORBIT PERFORMANCE COMPARISONS

Synchronous orbit inclination has a significant impact on payload capability. With no plane change during boost phase, a 90 degree launch azimuth down the Air Force Eastern Test Range (AFETR) results in a 28.5 degree orbit inclination. Boost at a 90 degree launch azimuth makes maximum use of the Earth's rotational velocity and results in maximum payload delivered to orbit. By varying the launch azimuth within the AFETR limits, a slightly reduced payload can be delivered to orbit without plane change during boost. However, to obtain some orbital inclinations (for example, 0^o and 55^o) boost turning is required in addition to launch azimuth variation within the AFETR limits.

Synchronous orbit performance for several orbit inclinations were generated using the design synchronous orbit trajectory ground rules. The net payload as a function of synchronous orbit inclination is presented in Figure 8.2-1. The payload for the 28.5 degree inclination orbit is 10,000 pounds larger than the payload for the equatorial (0^o) inclination orbit, a 15 percent increase.

For the synchronous design mission, a large amount of S-IVB LH_2 is unusable for mainstage impulse due to boiloff during the parking orbit coast (7.5 hours) and the transfer coast (5.3 hours). Even at the J-2S maximum mixture ratio, 5.5, the loaded S-IVB LOX was less than LOX tank capacity. Operating at the 5.5 MR (lowest J-2S specific impulse), maximizes the S-IVB stage total impulse. Lower MRs require larger LOX offloadings and further reduce the stage total impulse. By reducing the number of parking orbit revolutions (reduced hydrogen boiloff) the S-IVB LOX tank loading can be increased and a mixture ratio less than 5.5 can be used. Since the lower mixture ratios yield higher specific impulse, the S-IVB stage impulse is increased for the one and three parking orbit cases. Figure 8.2-2 shows the equatorial synchronous payload as a function of revolutions in parking orbit. The one revolution parking orbit case yields 4,562 pounds (7 percent) additional payload over the design trajectory (five parking orbit revolutions).

The Baseline SA-511 vehicle performance for the equatorial synchronous orbit design mission is shown in Figures 8.2-1 and 8.2-2. The J-2S/Saturn V equatorial synchronous orbit payload (66,000 lbs) is 11 percent higher than the J-2 Saturn V payload (59,500 lbs).

8.3 LOW EARTH ORBIT (LEO) PERFORMANCE COMPARISONS

Figure 8.3-1 is a comparison of two LEO orbit circularization methods for the Hohmann transfer flight profile. These two circularization methods are: (1) S-II idle mode burn and (2) S-II 100 second idle mode burn with S-II mainstage burn. The data is based on a 90 degree launch azimuth and an S-II mixture ratio schedule of 5.0 to 5.5 to 4.7. At 300 NM, the main stage burn circularization gives 7,000 pounds more payload if the S-II/J-2S mainstage restart repressurization system weight is not accounted for. Boeing parametric studies indicate such a

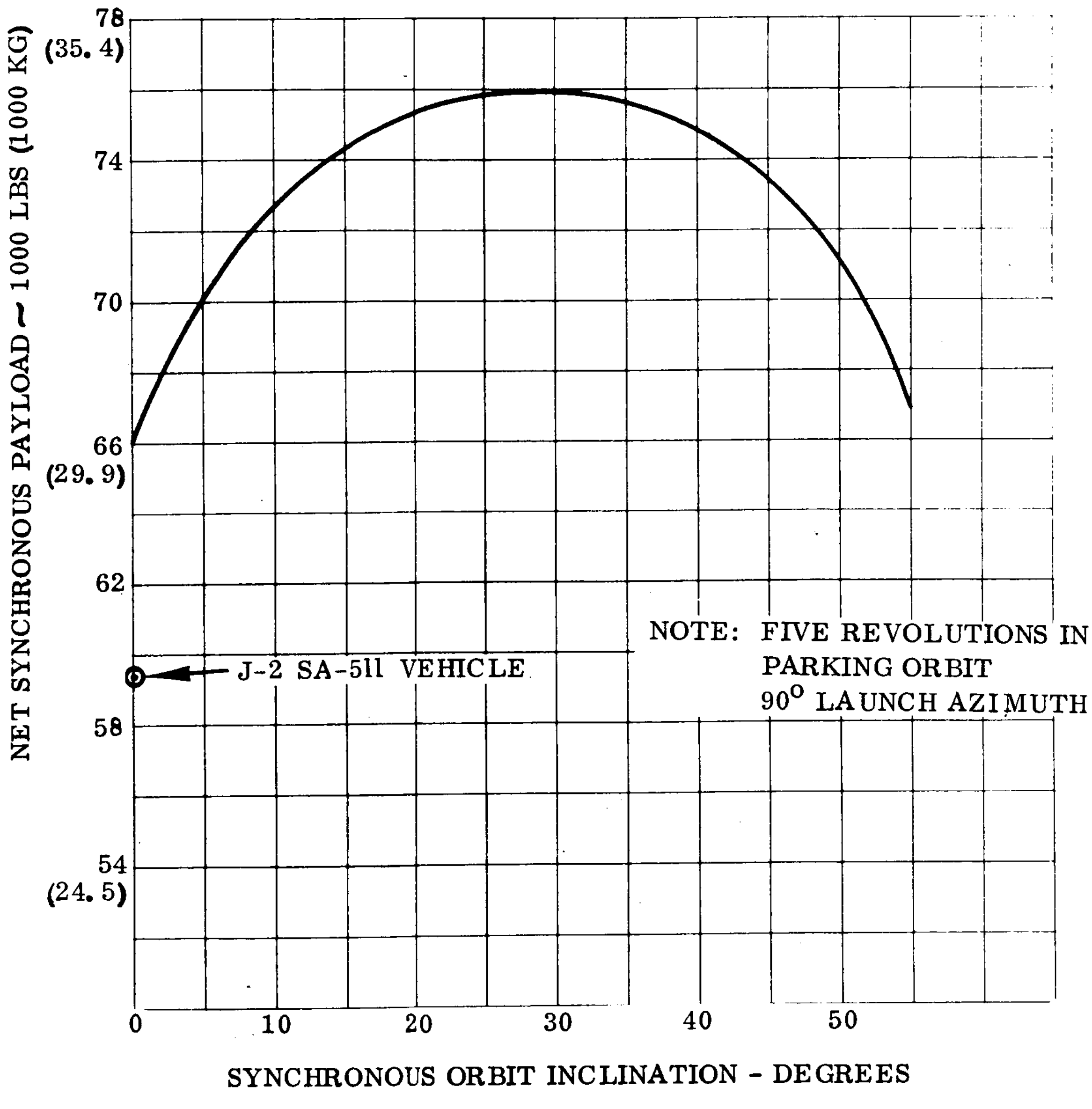

FIGURE 8.2-1 J-2S/SATURN V SYNCHRONOUS ORBIT PERFORMANCE

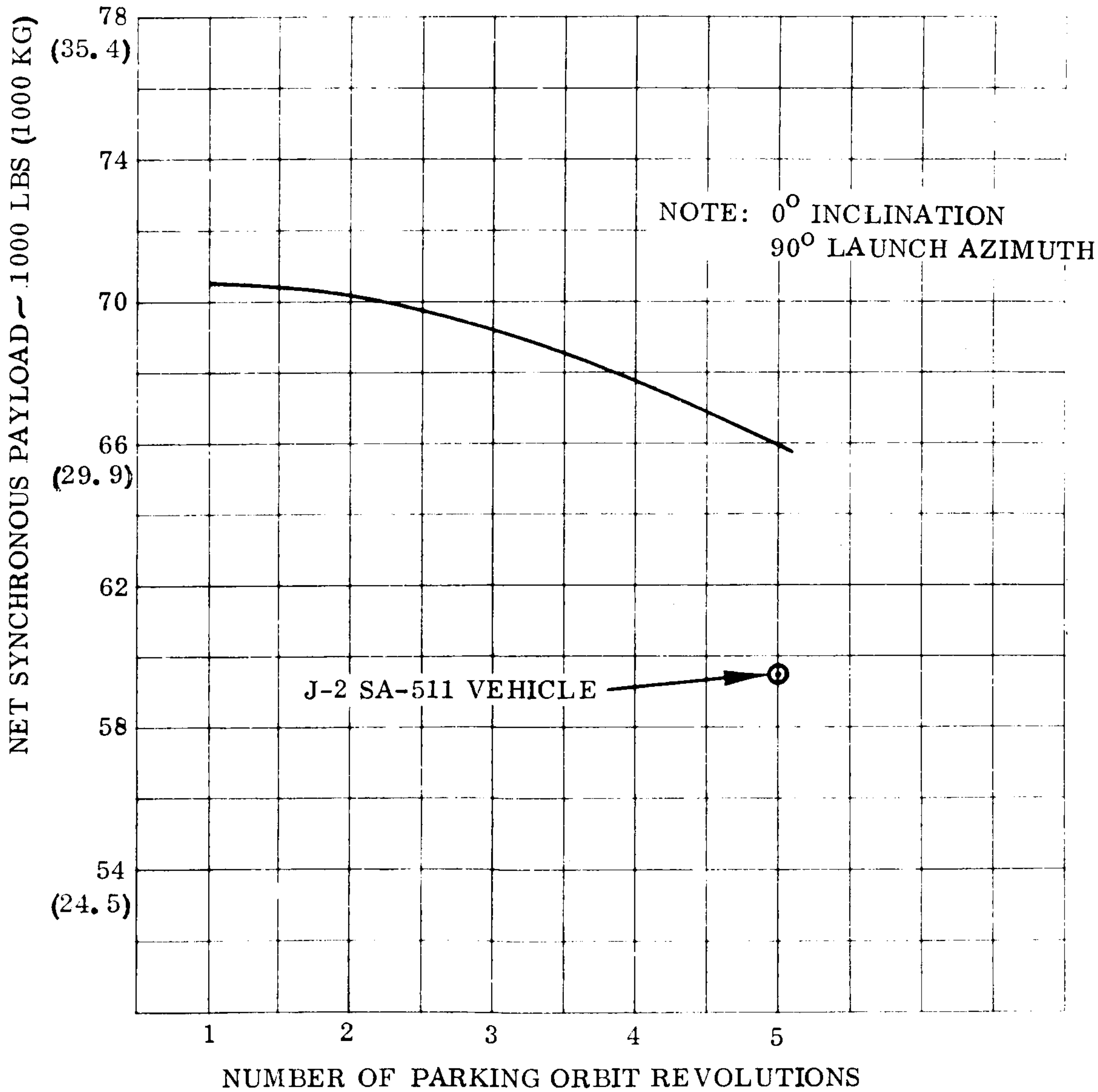

FIGURE 8.2-2 J-2S/SATURN V SYNCHRONOUS ORBIT PERFORMANCE

8.3 (Continued)

pressurization system would weigh 12,000 pounds (See Paragraph 5.3.2.2). After adjusting the mainstage burn data, the idle mode performance is shown to be better for altitudes up to 600 NM. The S-II propellant reserve for a 1° plane change (allowance for rendezvous) at Hohmann transfer apogee is not included in Figure 8.3-1. However, this propellant allowance is included in LEO design trajectory payload computation.

Comparison of direct injection payload with the Hohmann transfer payload (using idle mode injection) for the two-stage J-2S/Saturn V is shown in Figure 8.3-2. Direct injection type mission is not as attractive as the LEO Hohmann transfer design mission. At 300 NM circular orbit, direct injection payload is approximately 50 percent less than the Hohmann transfer payload.

A comparison of the J-2 and J-2S engines for the LEO direct injection two-stage vehicle is given in Figure 8.3-2. The J-2 payload is greater at the higher altitudes due to a shorter S-II burn time with J-2S engines (J-2S has a higher propellant flow rate with constant stage propellant tank capacity). The gravity losses during the J-2/S-II burn are less than the gravity losses during the J-2S/S-II burn because a larger fraction of the total J-2/S-II impulse is delivered at high altitude. The data was generated using 265K thrust J-2S engines. No attempt was made to optimize the thrust calibration level as a function of orbit altitude. Lower thrust calibration levels would increase payload performance at the higher orbit altitudes.

Launch azimuth effect on the LEO direct injection performance is shown in Figure 8.3-3. The data is for launch azimuths of 90°, 60°, and 30° for circular orbit altitudes up to 500 NM.

8.4 POLAR ORBIT PERFORMANCE COMPARISONS

Direct injection polar orbit missions to 100, 200 and 300 NM with a two-stage J-2S/Saturn V vehicle were simulated. This data is shown in Figure 8.4-1. For comparison, the 100 NM polar orbit direct injection payload for a two-stage vehicle with J-2 engines is shown. The payload with J-2S engines is some seven percent greater than with J-2 engines.

A Hohmann transfer from 100 NM to 200 NM and 300 NM was considered. At apogee, S-II idle mode burn was used for circularization. The payloads comparison of Hohmann transfer and direct injection are shown in Figure 8.4-1. Direct injection payload capability decreases rapidly with orbit altitude as compared to the Hohmann transfer profiles.

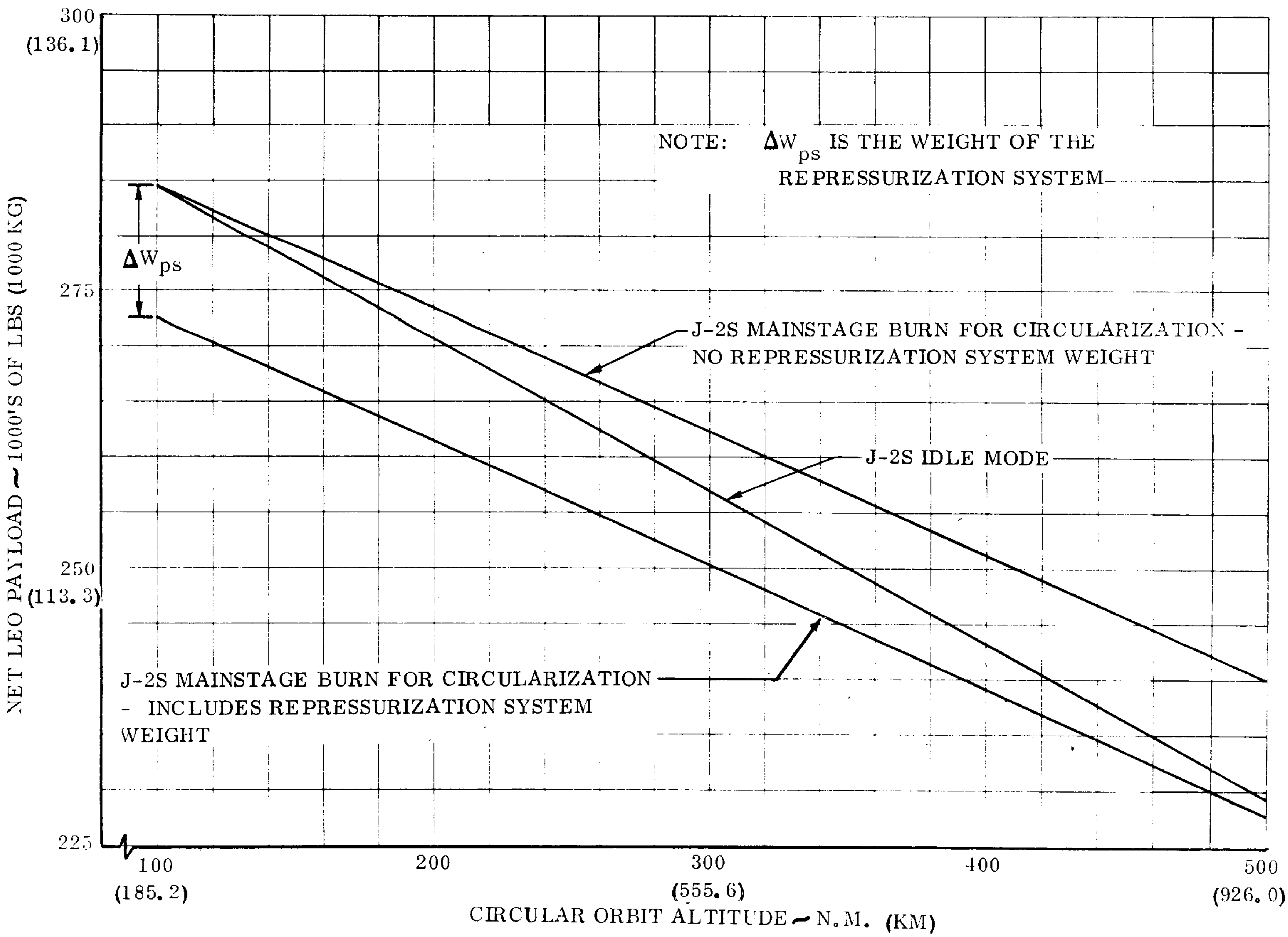

FIGURE 8.3-1 LOW EARTH ORBIT PERFORMANCE S-IC/S-II TWO STAGE VEHICLE MAINSTAGE VS IDLE MODE FOR ORBIT CIRCULARIZATION

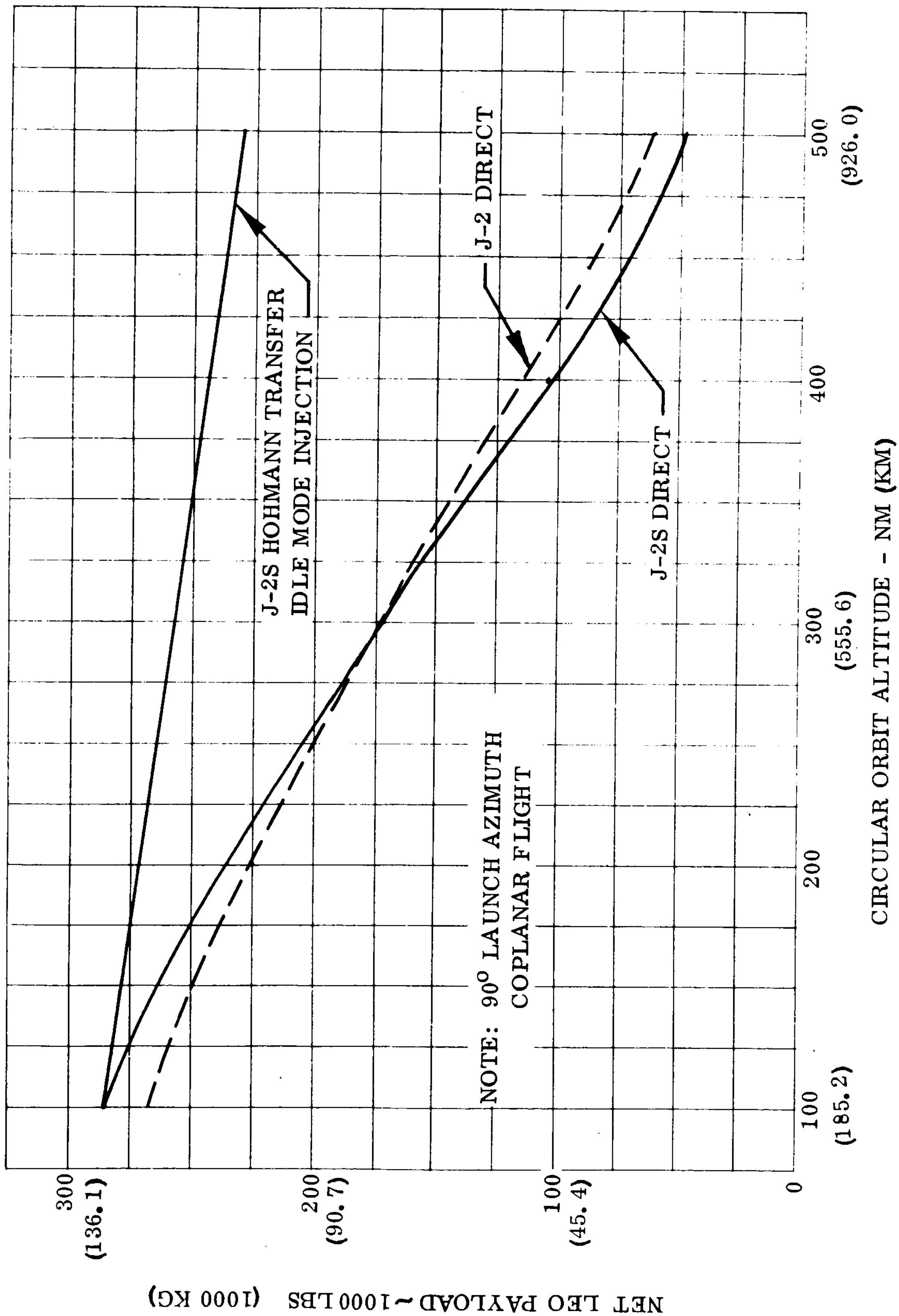

FIGURE 8.3-2 LOW EARTH ORBIT PERFORMANCE S-IC/S-II TWO STAGE VEHICLE

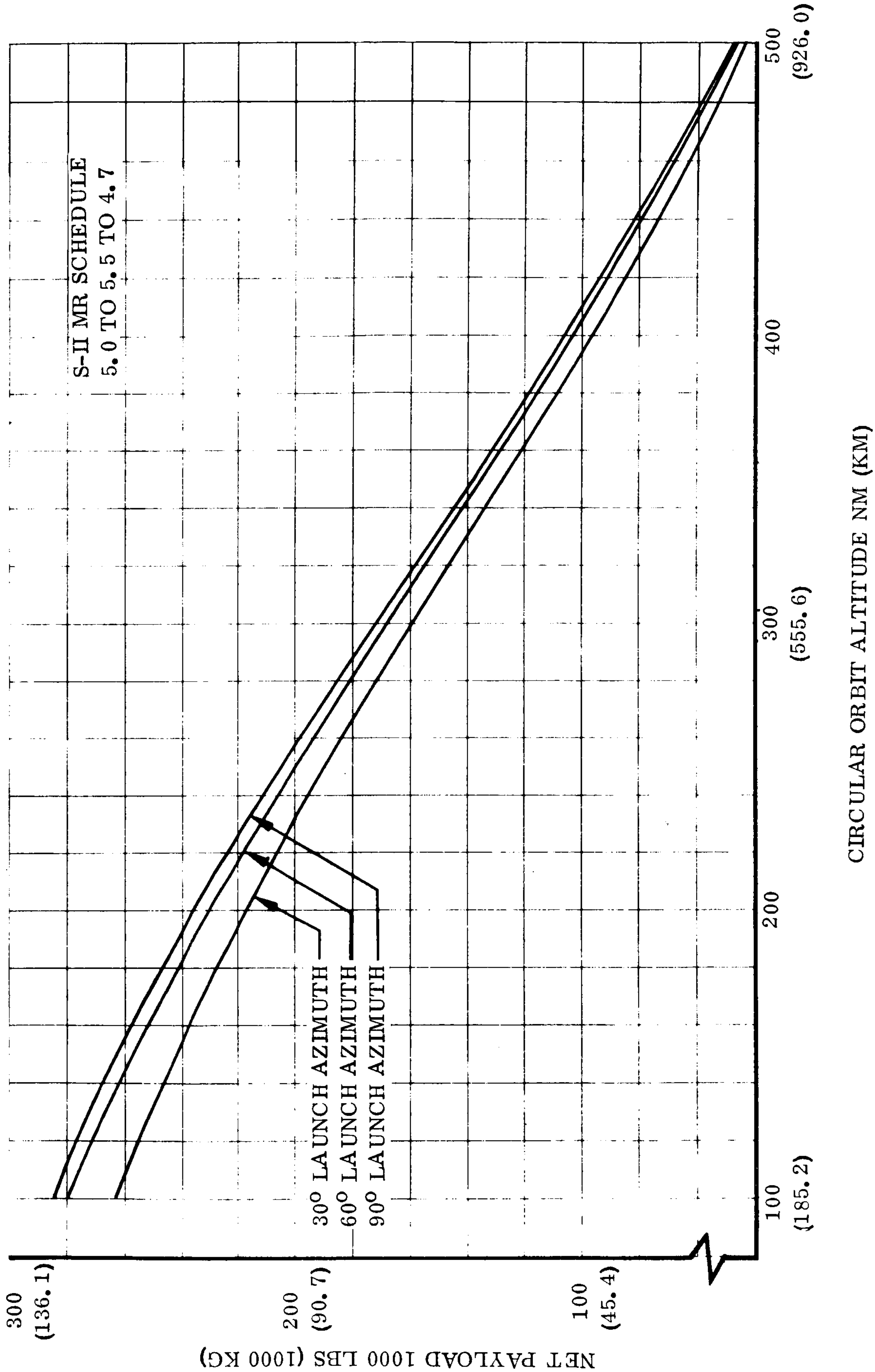

FIGURE 8.3-3 TWO-STAGE J-2S/SATURN V DIRECT ASCENT TO CIRCULAR ORBIT

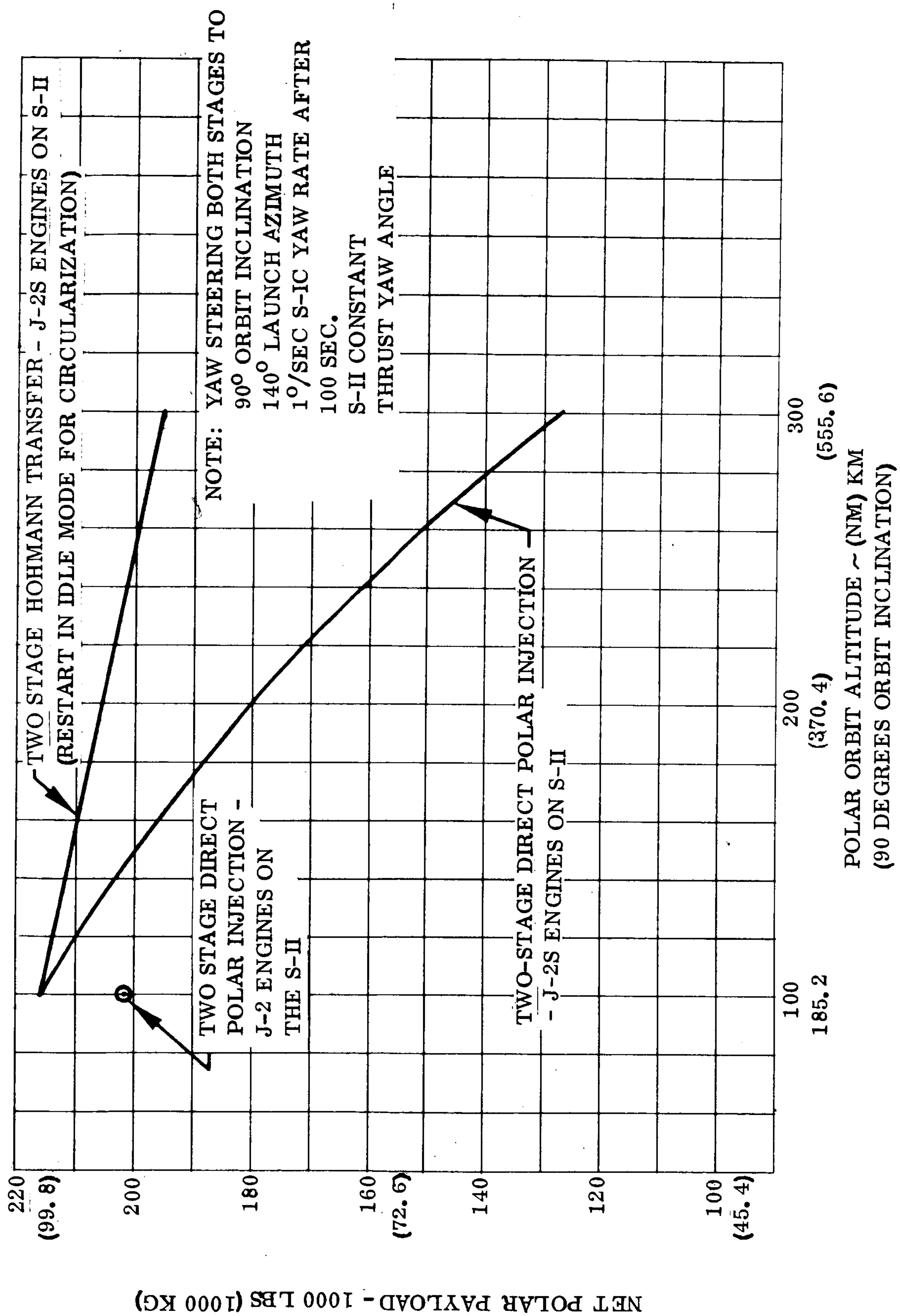

FIGURE 8.4-1 SATURN V POLAR ORBIT PERFORMANCE S-IC/S-II TWO-STAGE VEHICLE

8.5 ALTERNATE MISSION PERFORMANCE ANALYSIS

This performance analysis is idealized in that drop weights are based on a LOR J-2S/Saturn V vehicle. No vehicle loads were determined or assessment made of additional structural weight. Also the Apollo payload shape was assumed in all cases; thus certain high payloads assume high payload densities.

8.5.1 Direct Injection Lunar Mission - Three Stage J-2S/Saturn V Vehicle

The direct injection lunar mission is characterized by continuous burn to injection on the lunar transfer ellipse. No low Earth parking orbit is used.

To determine the optimum direct injection payload, trajectories were simulated for various sets of cutoff conditions (altitude, velocity and flight path angle). For each set of cutoff conditions, the liftoff weight and the boost profile were optimized.

For the optimum trajectory, the minimum altitude during S-IVB burn is 40.6 NM (75.2 km) where the dynamic pressure is 38 pounds per square foot. The net payload of 115,320 pounds represents an increase of approximately 6,400 pounds over the J-2S/Saturn V final LOR design payload of 108,953 pounds. Payload capability would decrease for higher dip altitudes. This optimum trajectory also assumes a zero launch window; that is the launch must occur at the exact planned time.

8.5.2 High Energy Missions - Three Stage J-2S/Saturn V Vehicle

The flight profile and ground rules for the high energy missions are the same as for the normal LOR design mission.

For a C_3 of 95 km^2/sec^2 (C_3 is twice the specific energy), the S-II can insert the fully loaded S-IVB plus payload into a 100 NM parking orbit. At higher C_3's, there exists a parking orbit altitude for which no S-IVB suborbital burn is required.

Net payloads for C_3's ranging from -20 km^2/sec^2 to $+120$ km^2/sec^2 and for 72^o and 90^o launch azimuths are presented in Figures 8.5-1 and 8.5-2. Representative energy requirements for various missions; i.e., Mars capture, .5 AU Solar Probe, etc. are noted on the plots.

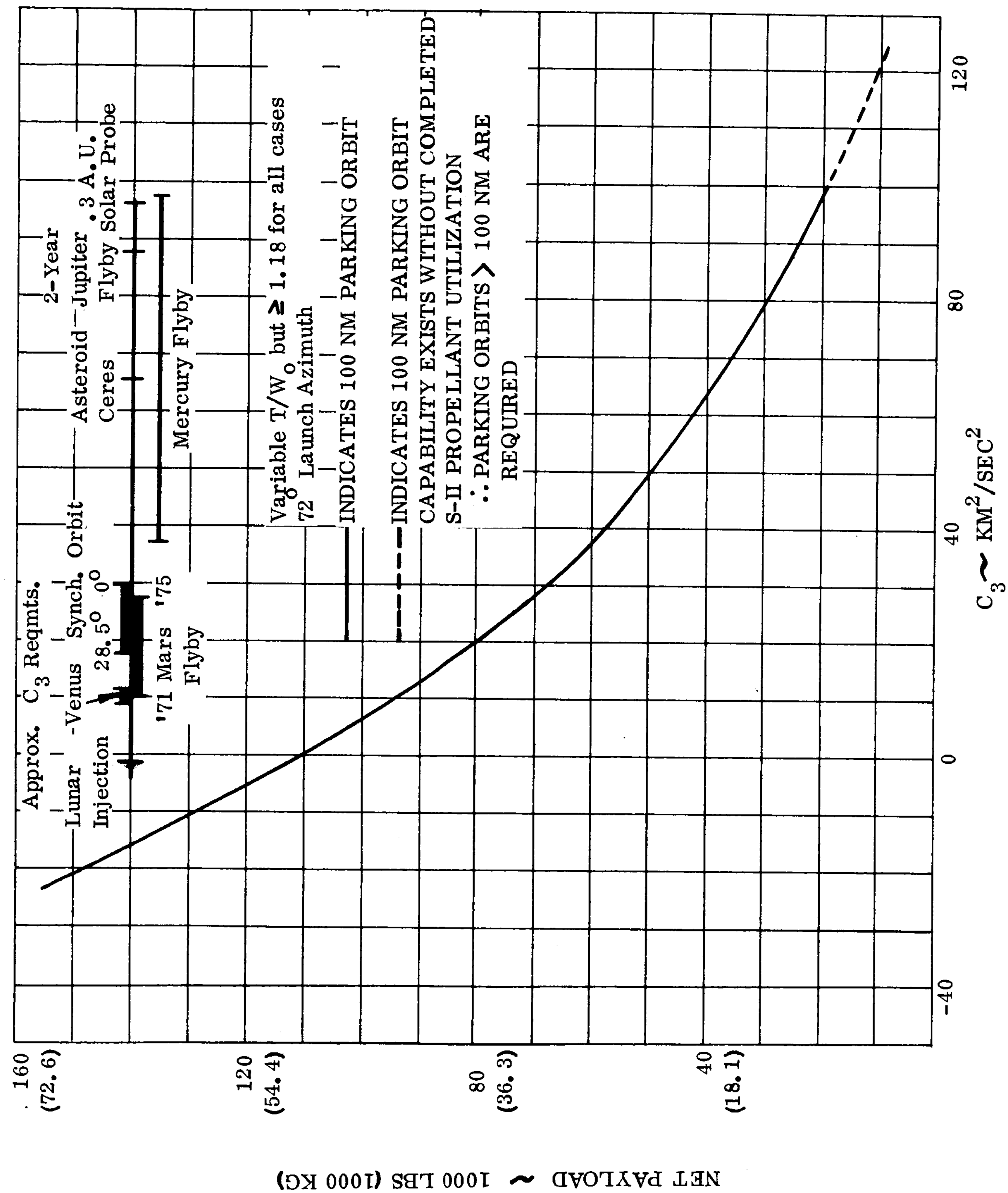

FIGURE 8.5-1 THREE STAGE HIGH ENERGY MISSION FROM EARTH PARKING ORBIT WITH J-2S ENGINES

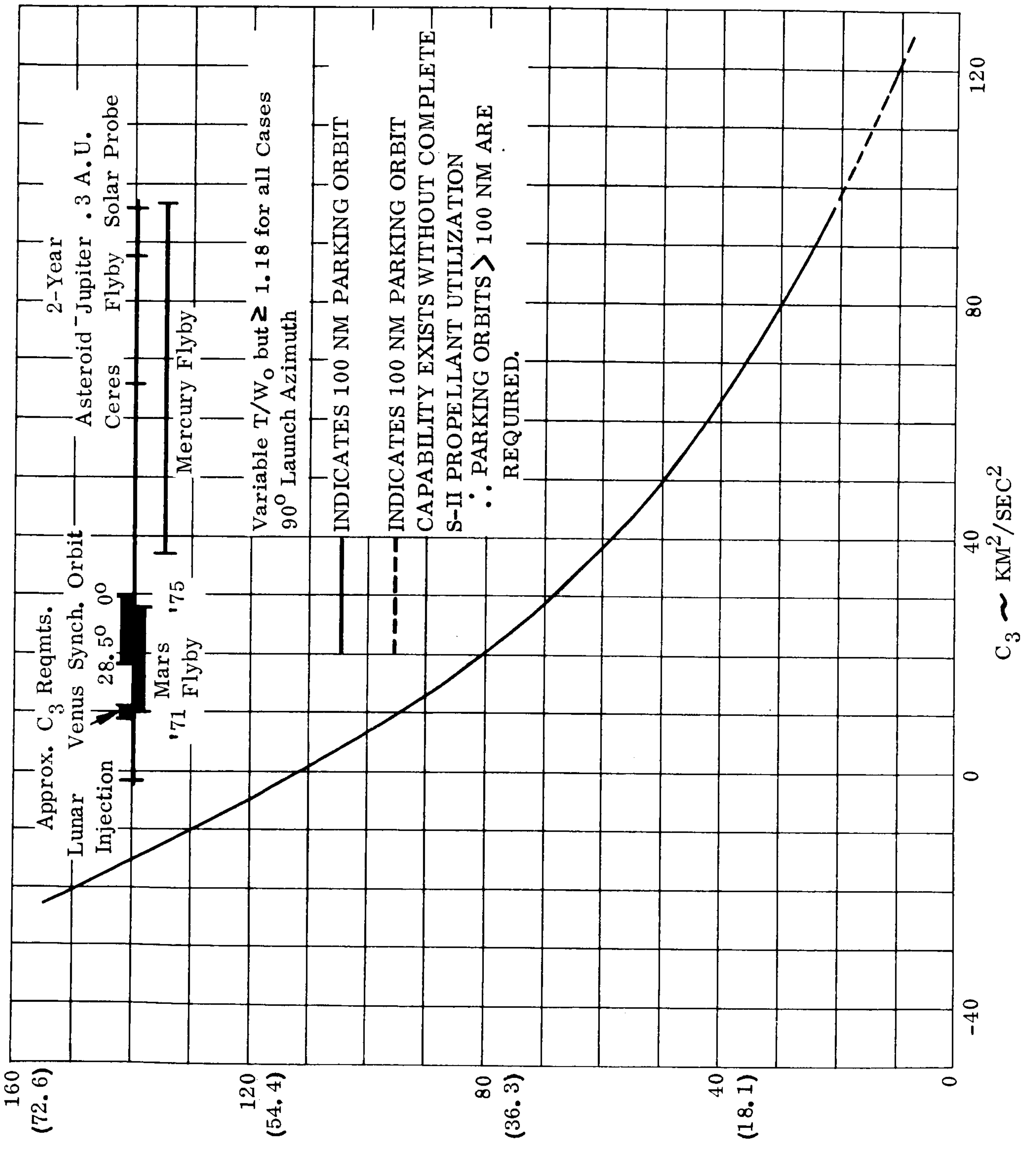

FIGURE 8.5-2 THREE STAGE HIGH ENERGY MISSION FROM EARTH PARKING ORBIT WITH J-2S ENGINES

8.5.3 Coplanar Hohmann Transfer Mission

Two-stage and three-stage J-2S/Saturn V Hohmann transfer performance data has been generated for circular orbit altitudes from 100 NM to 30,000 NM. The flight profile was direct ascent to the Hohmann transfer ellipse at 100 NM and coast to the desired orbit altitude. At apogee of the Hohmann transfer ellipse, circularization is achieved by restarting in the idle mode for 100 seconds and switching to mainstage. For the two-stage vehicle, the use of S-II in idle mode only for the circularization maneuver was also considered. These data are shown in Figure 8.5-3.

8.5.4 Direct Ascent to Circular Orbit - Three Stage J-2S/Saturn V Vehicle

Payload capability of the three-stage J-2S/Saturn V as a function of orbit altitude has been determined for the direct ascent mission. The propellant loadings in all three stages were optimized. The direct ascent and Hohmann transfer performance are compared in Figure 8.5-4. Direct ascent payload drops off rapidly with altitude. At 700 NM, three-stage Hohmann transfer performance is 60 percent greater than direct ascent performance.

8.5.5 Polar Orbit Missions - Three Stage J-2S/Saturn V Vehicle

The polar orbit mission data is based on a 140° launch azimuth, a fixed rate yaw steering in the first stage, and a fixed inertia yaw angle in the upper stages, overflight of Cuba and the Isthmus of Panama, and a first equatorial crossing just west of South America. For all data presented, the S-II and S-IVB operates at the thrust, specific impulse and mixture ratio schedule used for the nominal LOR design trajectory.

The polar net payload as a function of orbit altitude is shown in Figure 8.5-5.

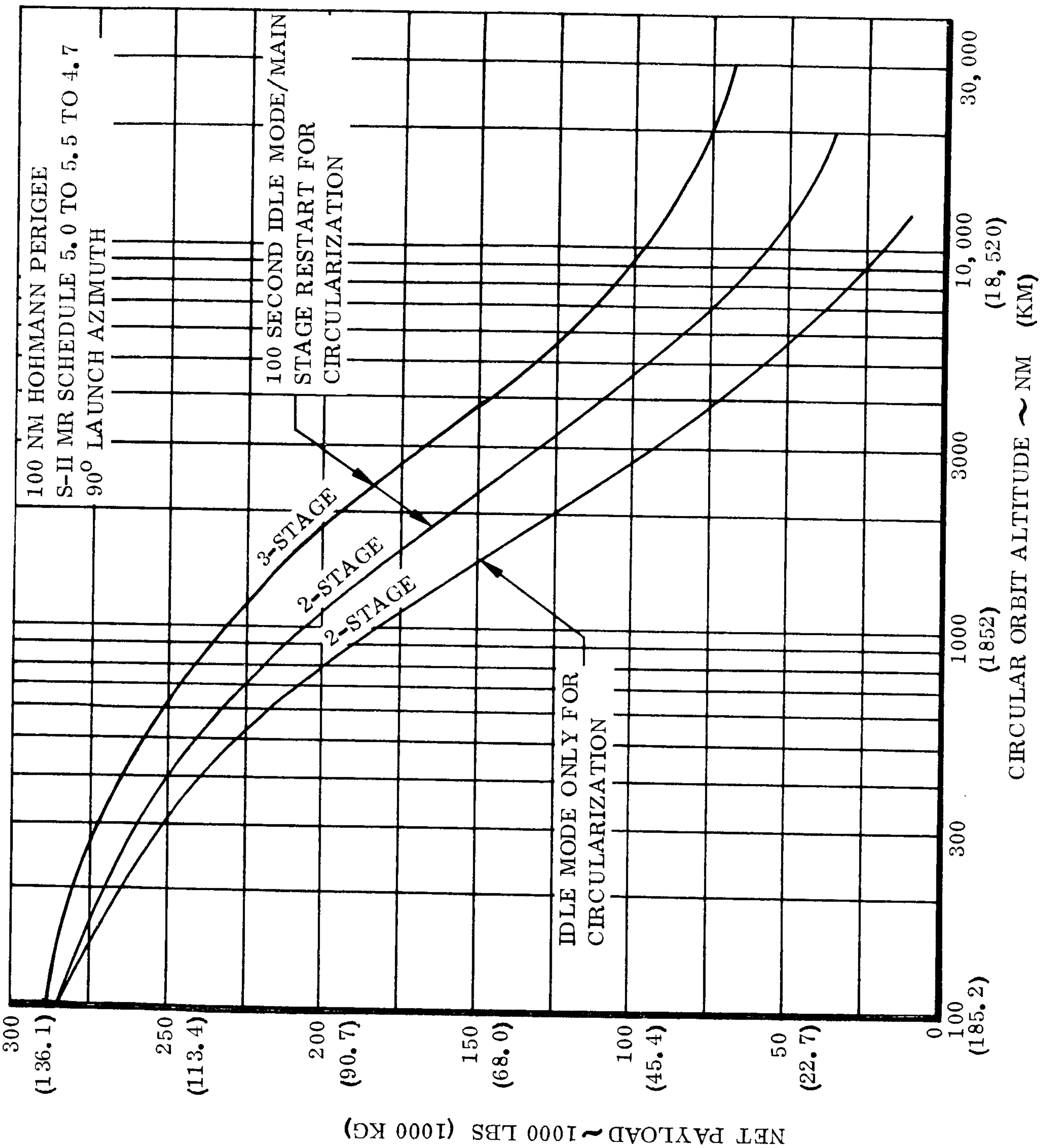

FIGURE 8.5-3 J-2S/SATURN V HOHMANN TRANSFER TO CIRCULAR ORBIT

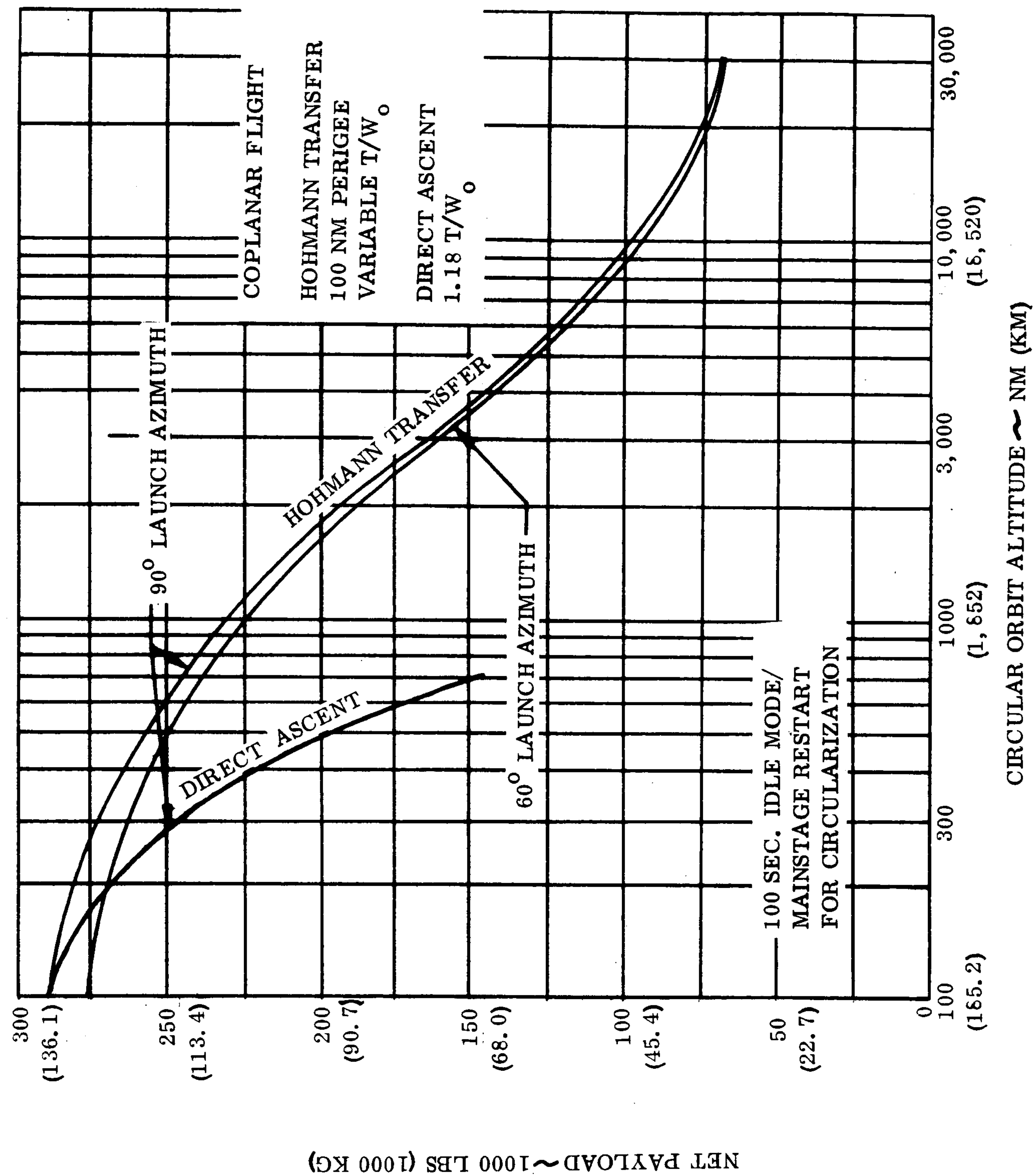

FIGURE 8.5-4 THREE-STAGE J-2S/SATURN V PERFORMANCE TO CIRCULAR ORBIT

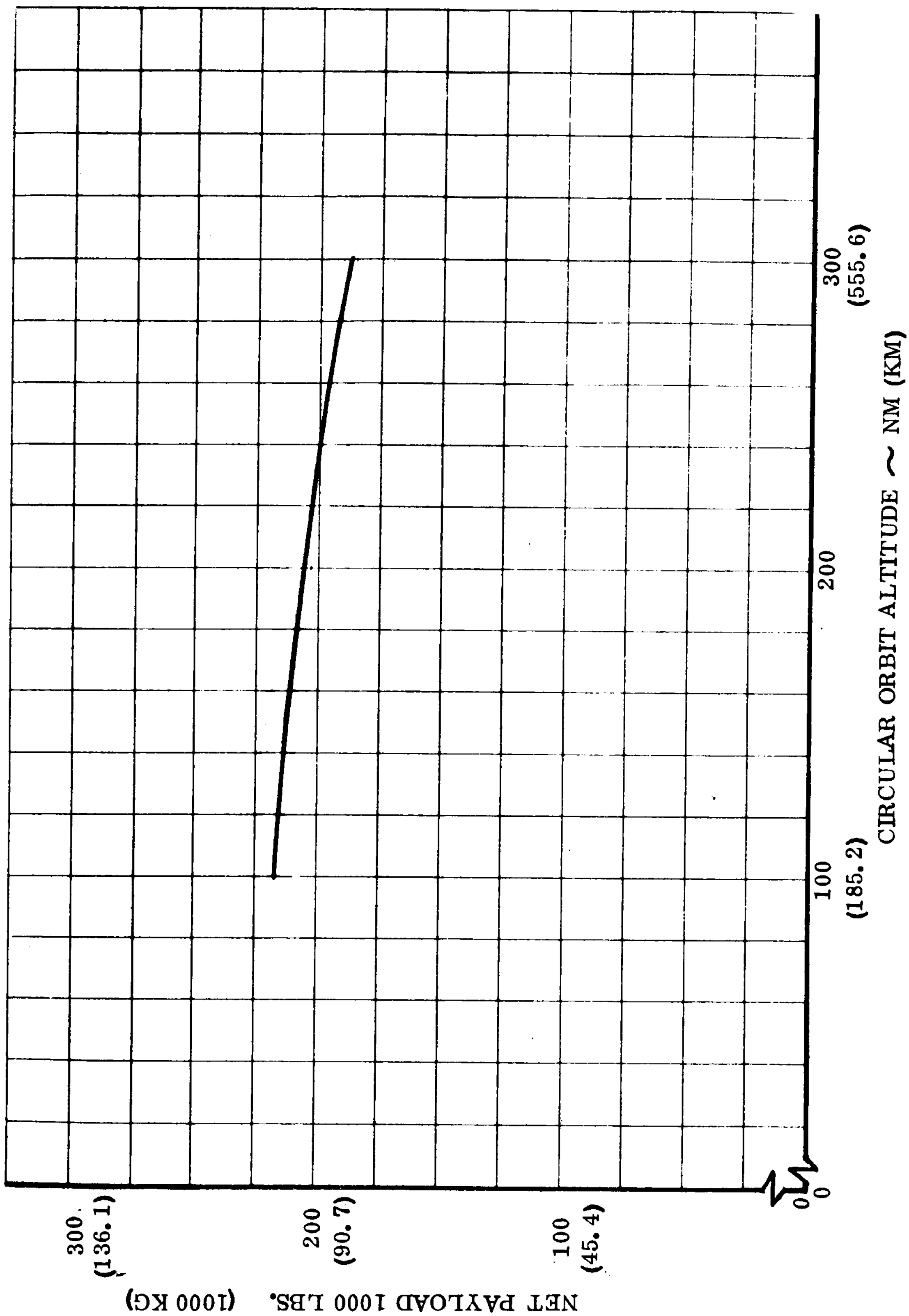

FIGURE 8.5-5 THREE-STAGE J-2S/SATURN V POLAR ORBIT PERFORMANCE

SECTION 9

PAYLOAD EXCHANGE RATIOS FOR J-2S/SATURN V VEHICLES

9.0 GENERAL

This section contains the payload exchange ratios for the LOR and 100 NM LEO missions (Table 9-I). Exchange ratios provide a means for determining payload performance changes from small variations in vehicle nominal design parameters such as stage weight, propellant weight, thrust and specific impulse. As an example, J-2S engine tests conducted after release of J-2S engine characteristics (Rocketdyne J-2S Interface Criteria Document, R-7211) indicate a possible five second Isp increase. The payload performance gain due to the five second Isp increase can be obtained by the use of payload exchange ratios as follows:

From Table 9-I

$$\frac{\Delta PL}{\Delta I_{sp_2}} = 397.4 \text{ lbs/sec; S-II Stage}$$

$$\frac{\Delta PL}{\Delta I_{sp_3}} = 347.8 \text{ lbs/sec; S-IVB Stage}$$

$$\Delta PL (I_{sp2}) = \frac{\Delta PL}{\Delta I_{sp2}} \times \Delta I_{sp2} = 397.4 \frac{\text{lbs}}{\text{sec}} \times 5 \text{ sec.} = 1,987 \text{ lbs}$$

$$\Delta PL (I_{sp3}) = \frac{\Delta PL}{\Delta I_{sp3}} \times \Delta I_{sp3} = 347.8 \frac{\text{lbs}}{\text{sec}} \times 5 \text{ sec.} = 1,739 \text{ lbs}$$

$$\text{Total } \Delta PL \text{ from } I_{sp} \text{ increase} = \Delta PL (I_{sp2}) + \Delta PL (I_{sp3})$$

$$= 1,987 \text{ lbs} + 1,739 \text{ lbs}$$

$$= 3,726 \text{ lbs}$$

Revised LOR Payload = 108,953 + 3,726

= 112,679 lbs

Similarily, payload performance increases would be obtained for the synchronous orbit, low-Earth orbit, and polar orbit missions.

9.1 DISCUSSION

The payload exchange ratios for both two-stage and three-stage J-2S/Saturn V vehicles are presented in Table 9-I. The two-stage vehicle mission is direct ascent to 100 NM circular orbit. The three-stage vehicle mission is LOR. The exchange ratio for a parameter is determined by perturbing the parameter about its nominal value. The resulting changes in net payload are plotted as a function of the perturbed parameter. The exchange ratio (rate of change of payload with respect to the perturbed variable) is the slope of the curve at zero perturbation. A number of these parameters are coupled which requires the perturbation of a second parameter in order to keep a third parameter constant. For example, if first stage inert weight is changed, liftoff weight must be changed to keep first stage propellant weight constant.

TABLE 9-I J-2S/SATURN V EXCHANGE RATIOS

STAGE	PERTURBED PARAMETER		LOR THREE-STAGE EXCHANGE RATIO PAYLOAD/PARAMETER	100 NM TWO-STAGE EXCHANGE RATIO PAYLOAD/PARAMETER
S-IC	Thrust,	F_1	.008530 lbs/lb	.02075 lbs/lb
	Specified Impulse,	I_{SP1}	546.6 lbs/sec	1420 lbs/sec
	Propellant Weight,	W_{P1}	.008472 lbs/lb	.02310 lbs/lb
	Drop Weight,	W_{D1}	-.0664 lbs/lb	-.1788 lbs/lb
S-II	Thrust,	F_2	.01851 lbs/lb	.02675 lbs/lb
	Specific Impulse,	I_{SP2}	397.4 lbs/sec	1217 lbs/sec
	Propellant Weight,	W_{P2}	.02812 lbs/lb	.1077 lbs/lb
	Drop Weight,	W_{D2}	-.3098 lbs/lb	-1.0 lbs/lb
S-IVB	Thrust,	F_3	.002775 lbs/lb	
	Specific Impulse,	I_{SP3}	347.8 lbs/sec	
	Propellant Weight,	W_{P3}	.1173 lbs/lb	
	Drop Weight,	W_{D3}	-1.0 lbs/lb	
	Weight Loss in Parking Orbit,	W_{KO}	-.39125 lbs/lb	

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SECTION 10
STAGE/SYSTEMS DESCRIPTION

10.0 GENERAL

This section describes system and operational changes required when J-2S engines are incorporated in Saturn V. Changes required to replace J-2 engines with J-2S engines for the LOR mission are identified separately from mission peculiar changes required to adopt the J-2S/Saturn V for other missions. These changes are based on the J-2S/Saturn V performance parameters, design characteristics and requirements defined in Section 7. Each stage of the design vehicle was evaluated and documented by the contractor responsible for the system. The descriptions were used in definition of resource requirements and associated schedules and costs for implementation of the vehicle system.

10.1 S-IC STAGE

10.1.1 S-IC Stage Description

The S-IC stage, as shown in Figure 10.1-1, is a cylindrical structure designed to provide the initial boost for the Saturn V Apollo vehicle and is identical to baseline S-IC SA-511. This booster is 138 feet long and 33 feet in diameter. The basic structures of the S-IC are the thrust structure, fuel (RP-1) tank, intertank section, LOX tank, and the forward skirt. Attached to the thrust structure are the five F-1 engines which produce a combined nominal sea level thrust of 7,610,000 pounds. Four of these engines are placed equidistantly around a circle with a diameter of 30.33 feet. The four outboard engines are attached so they have a gimbaling capability. Each outboard engine can move in a 5 degree, 9 minute square pattern to provide pitch, yaw, and roll control. The fifth engine is fixed mounted at the stage centerline. In addition to supporting the engines, the thrust structure also provides support for the base heat shield, engine accessories, engine fairings and fins, propellant lines, retro motors, and environmental control ducts. The intertank structure provides structural continuity between the LOX and fuel tanks, which provide propellant storage; and the forward skirt is necessary for interfacing with the S-II stage.

Propellants are supplied to the engine turbopumps by 15 suction ducts: 5 from the LOX tank, and 10 from the fuel tank. The fuel tank is a semimonocoque cylindrical structure closed at each end by an ellipsoidal bulkhead. Antislosh ring baffles are located on the inside wall of the tank, and an antivortex cruciform baffle is located in the lower bulkhead area. The configuration of the LOX tank is basically the same with the exception of capacity. The LOX tank will provide storage for 47,405 cubic feet including ullage. The fuel tank will hold approximately 29,221 cubic feet including ullage. The mixture ratio between LOX and RP-1 is approximately 2.27:1 (LOX to RP-1).

10.1.1 (Continued)

The LOX and fuel pressurization systems provide and maintain the Net Positive Suction Pressure (NPSH) required for the LOX and fuel turbopumps during engine start and mainstage. These systems also provide protection from high pressures which might occur in the LOX and fuel tanks. Before engine ignition, the LOX and fuel tanks are pressurized from a ground helium supply. During flight, LOX pressurization is accomplished by gaseous oxygen obtained by using F-1 engine heat exchangers to convert oxygen tapped from the engines, from liquid to gas. The fuel tank is pressurized by gaseous helium supplied by helium bottles located in the LOX tank. The LOX and fuel feed systems contain sensors for LOX and fuel depletion for purposes of engine cutoff during flight.

Eight solid propellant retro motors that provide separation thrust after S-IC burnout are attached externally to the thrust structure, but located inside the four outboard engine fairings. The S-IC and S-II stages are severed by linear shaped charges, and the retro motors supply the necessary deceleration force to provide separation. Each retro motor is pinned securely to the vehicle support and pivot support fittings at an angle of 7.5 degrees from stage centerline.

Additional systems on the S-IC include:

a. The Environmental Control System (ECS) which protects the S-IC stage from temperature extremes, excessive humidity, and hazardous gases.

b. The hydraulic system which provides power to operate the engine valves and thrust vector control system.

c. The pneumatic control pressure system which provides a pressurized nitrogen supply for command operations of various pneumatic valves, and a purge for TV camera lenses.

d. The electrical system which distributes and controls the stage electrical power.

e. The instrumentation system which monitors functional operation of the stage systems and provides signals for vehicle tracking during S-IC burn.

10.1.2 S-IC Stage Structures

a. LOR, LEO and Synchronous Missions

The environments and ultimate design loads for the LOR, LEO and synchronous missions were less than the vehicle structural capability. Therefore, the S-IC stage for these missions is identical to the Baseline SA-511 first stage configuration.

b. Two-Stage Polar Missions

Due to flight profile peculiarities, the maximum qα product was greater for the two-stage polar mission than it was for the other three missions. This condition indicates higher vehicle bending moments which, in turn, produced ultimate combined compressive loads that were greater than the S-IC fuel tank sidewall capability.

The tank wall capability is presented in D5-13829, Capabilities of the Major S-IC Structural Components, and is based on structural tests of the S-IC fuel tank under combined internal pressure and axial compressive loading. These capabilities are conservative to some extent since the tank was not tested to failure.

No structural modification of the lower segment of the fuel tank (Station 365.0 to 483.5) is recommended. The critical design loads for the lower segment exceed published capability by less than 0.8 percent; e.g., at Station 365, the design load is 6,909 lb/in combined with an internal local pressure of 27.5 psig while the tank wall capability for this pressure is 6,856 lb/in. The design load at Station 483.5 is 7,430 lb/in combined with 20.8 psig internal pressure. The minimum capability at this location is 7,550 lb/in at 20.8 psig.

The upper segment of the tank (Station 483.5 to 602.0) should be modified to provide structural capability equal to the design loads. A proved analytical method presented in D5-13272, Analysis of Stability Critical Orthotropic Cylinder Subject to Axial Compression, was used to resize the upper segment for the new design condition which is approximately 3.3 percent in excess of capability at Station 602. Existing stringer spacing and ring frame size and spacing was used in defining a minimum skin/stringer modification. The required modifications are to increase the longitudinal stringer cap width by 0.1 inches from Station 483.5 to Station 602.0, and the skin thickness by 0.005 inches from Station 544.26 to Station 602.0. These changes are shown in Figure 10.1-2. The increase in S-IC weight is approximately 74 pounds.

10.1.2 (Continued)

Rather than modify the fuel tank, load alleviation techniques such as trajectory shaping, restricted launch availability, and restricted design criteria could be used. Trajectory shaping would consist of optimizing yaw steering start time, yaw steering rate and launch azimuth. This would change the S-IC tilt program and consequently, result in changes to the load criteria parameters such as dynamic pressure (q) and acceleration (g). Launch availability could be reduced by restricting launches to months of the year other than the windy months such as March as now used for design criteria. The design criteria could be reduced by accepting a reduction in the current 95 percent probability wind design criteria. The 1.4 design criteria for manned flight could be reduced. The factor of safety resulting from the S-IC overload condition is 1.36. This mission may be unmanned; therefore, a factor of safety of 1.25 would be used as design criteria and the stage structural capability would be satisfactory.

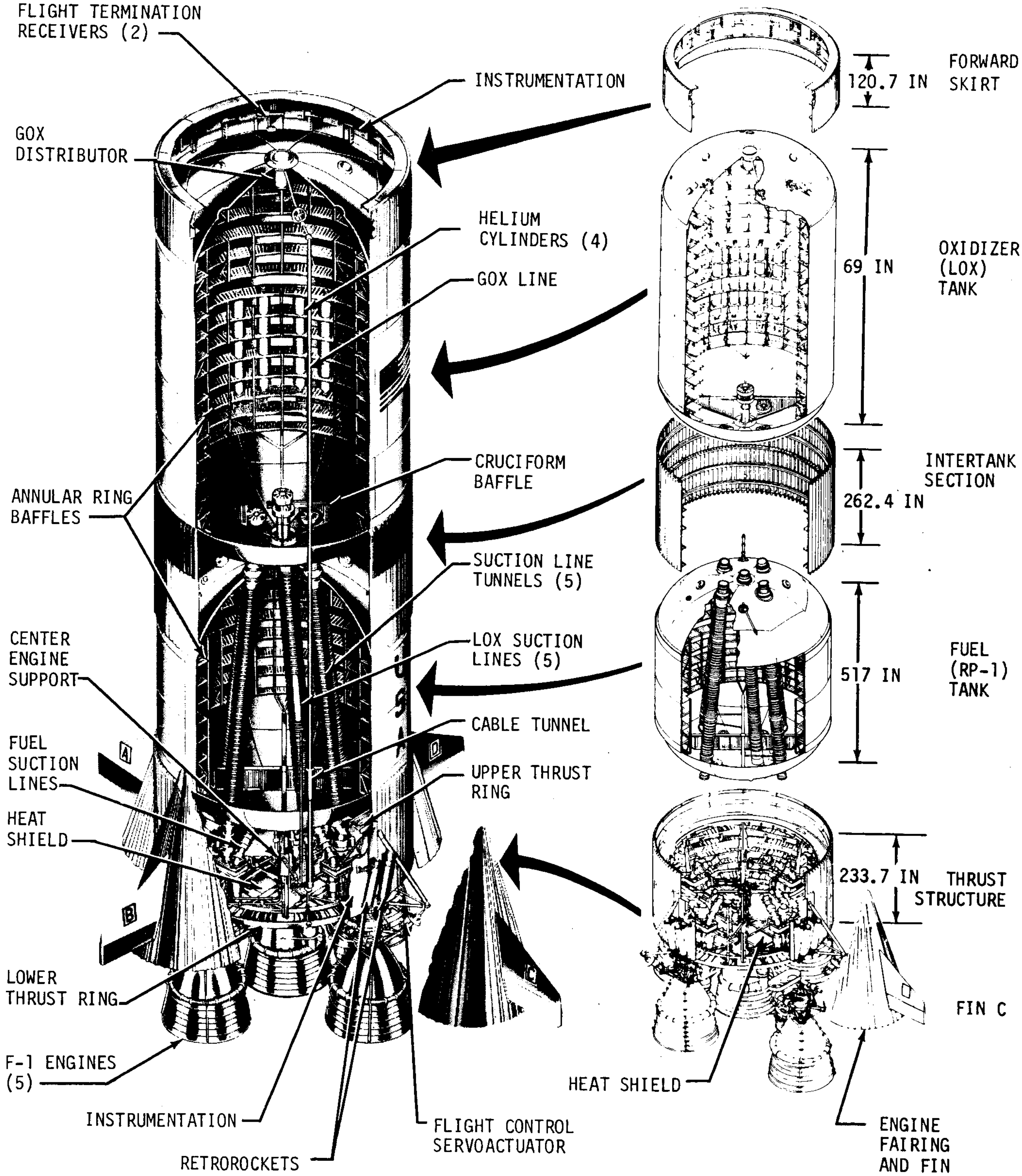
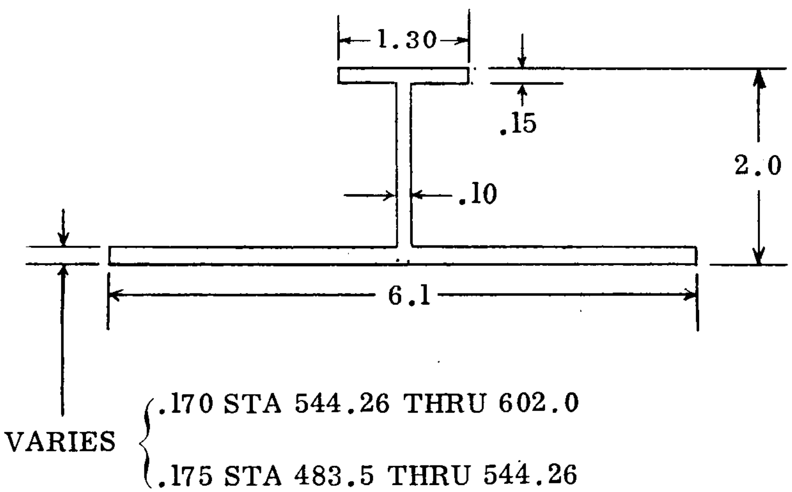
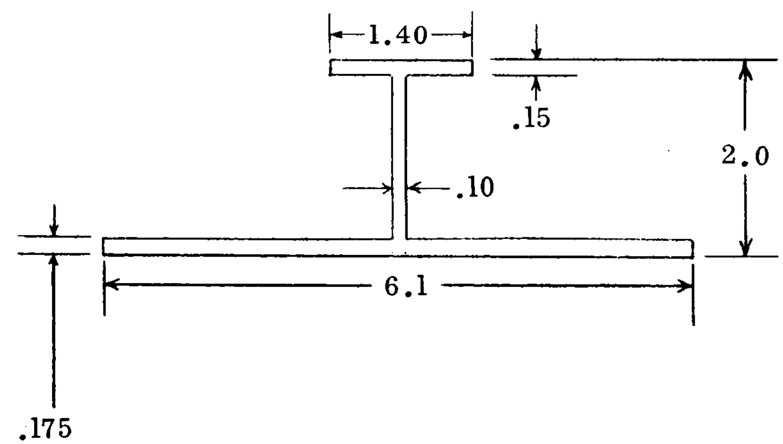


FIGURE 10.1-1 S-IC STAGE CONFIGURATION



EXISTING CONFIGURATION
STATION 483.5 THRU 602.0
BASELINE WEIGHT



MODIFIED CONFIGURATION
STATION 483.5 THRU 602.0
Δ WEIGHT = + 74 LB.

NO MODIFICATION REQUIRED
BELOW STA. 483.5

NOTE:
ALL DIMENSIONS IN INCHES

FIGURE 10.1-2 SATURN V + J-2S TWO-STAGE POLAR ORBIT S-IC FUEL TANK MODIFICATIONS

10.2 S-II/J-2S STAGE/SYSTEM DESCRIPTION

This section contains the technical definition of the changes required to the S-II stage for incorporation of the J-2S engine for the LOR, SO, PO, and LEO missions. In all cases, the change definition for the LOR mission is treated as basic, with changes for the other missions being described as additive to the basic changes. The technical definition of stage and GSE systems is supported by analysis and by discussion of alternate approaches. The corresponding operational changes, test program requirements, and reliability assessments are included.

10.2.1 STRUCTURAL SYSTEMS

This section describes the structural systems design changes, stage interface changes, and stage stiffness properties for the several missions studied. The most significant changes are in the thrust structure complex, and are caused by the increased thrust of the J-2S engines.

10.2.1.1 Design Changes

Design changes are itemized below for the LOR and LEO missions and also for the PO missions within the limitations of the study.

10.2.1.1.1 LOR Mission

A comparison of the S-II stage primary body structural capability with the LOR mission design loads envelope, supplied by TBC, is shown in Figure 7.5-3.

The S-II structural capability exceeds the design loads. Therefore, no change to the basic airframe (body) structure is required except as noted below for J-2S engine installation or to accommodate functional systems modification, deletion, or additions.

a. Thrust Structure Complex. The increased thrust of the J-2S engine requires strengthening of the entire thrust structure to maintain structural integrity and/or to comply with present (S-II-11) stiffness requirements. The stiffness requirement must be maintained to conserve use of the present (J-2) feedlines, engine actuation system, engine servicing system, and engine-to-stage interface hardware without requalification or redesign.

   The approach taken in strengthening the thrust structure is to maintain the location of component parts and type of material, and to increase the thickness of detail elements. This approach was selected on the basis of minimum cost or impact on associated hardware, tooling, manufacturing, or engineering.

   Detail changes are as follows:

   1. Thickness of thrust cone skin, stringers, longerons, and frame caps and webs will be increased. (Figures 10.2-1 through 10.2-7).

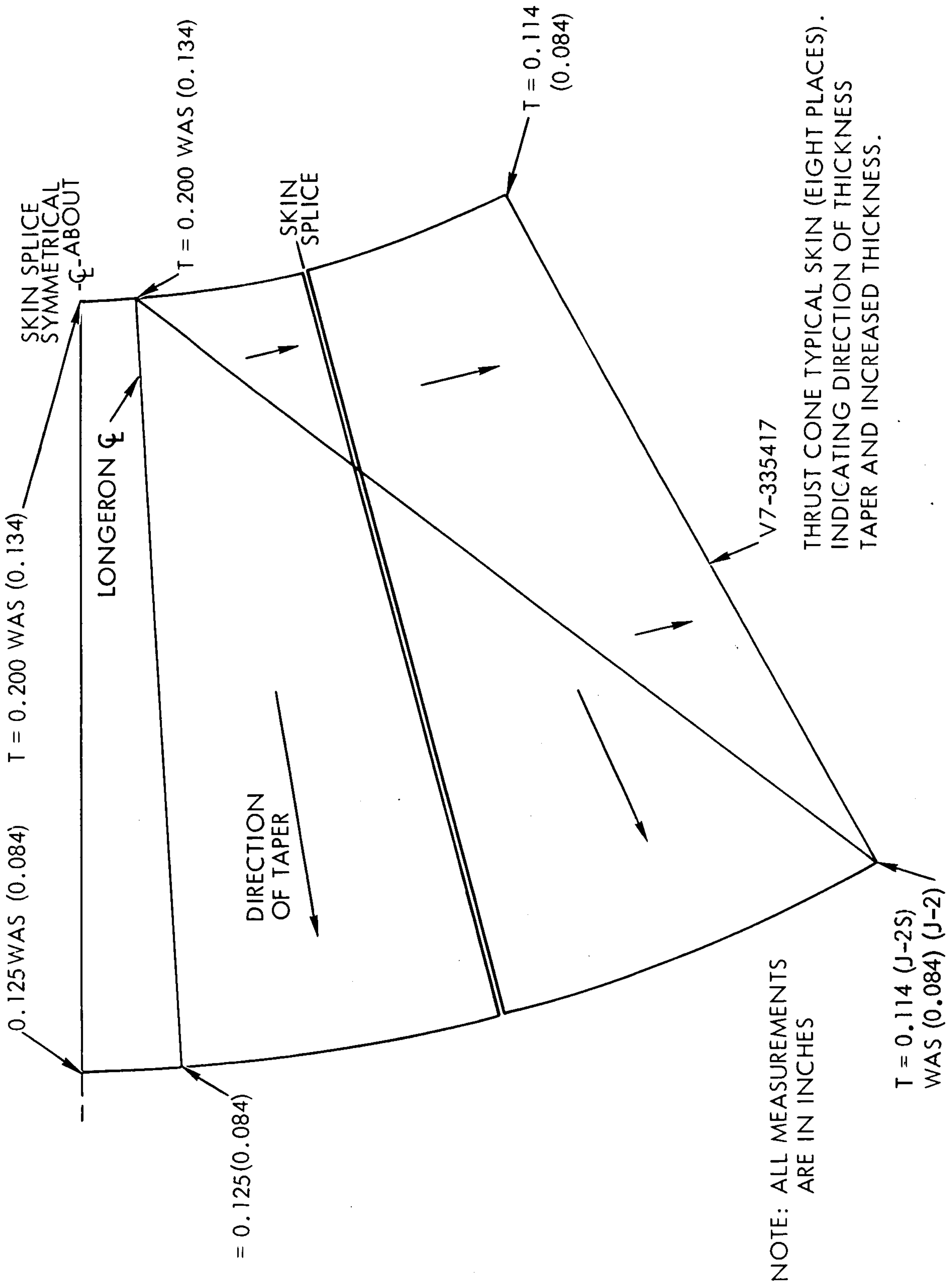
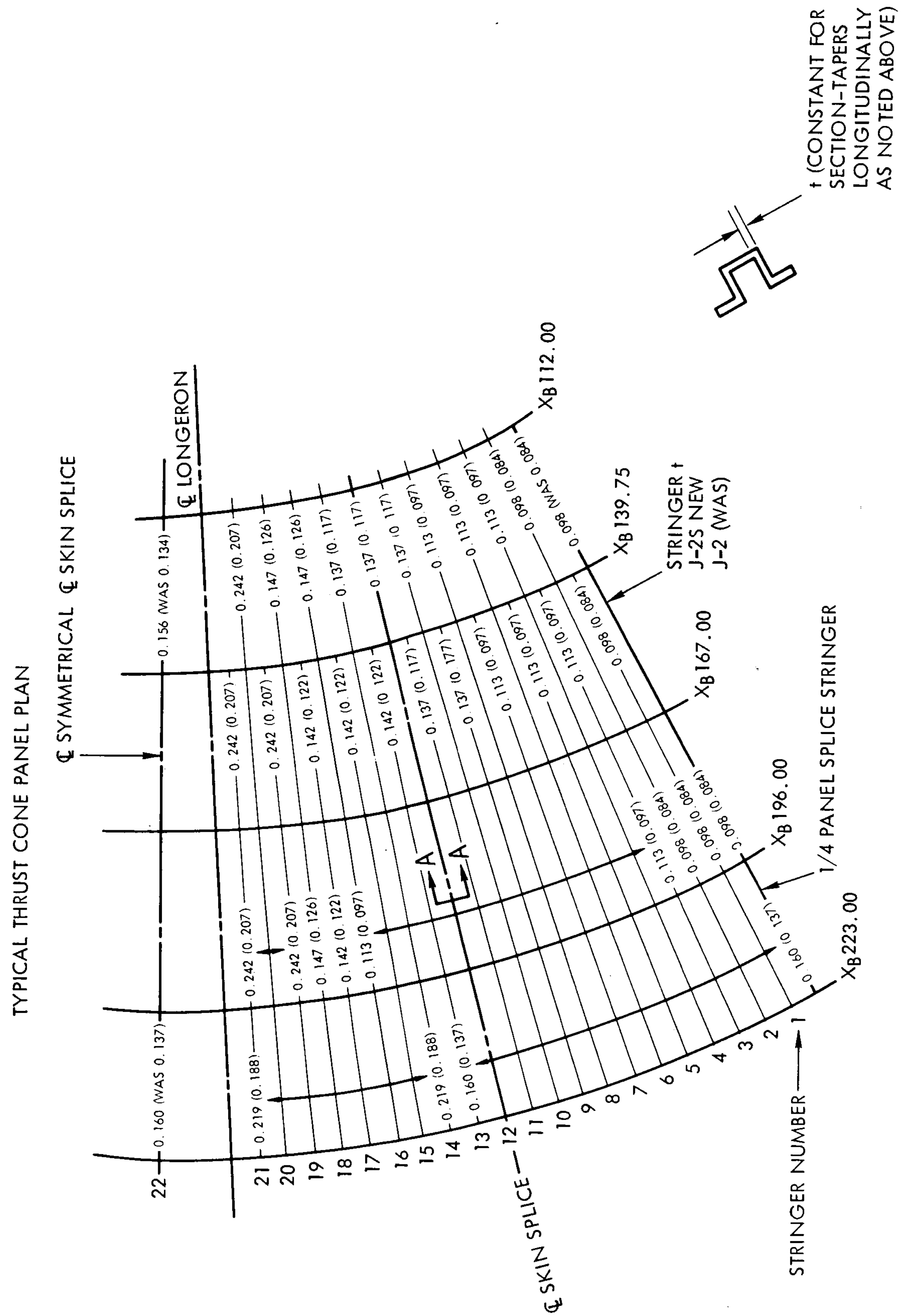
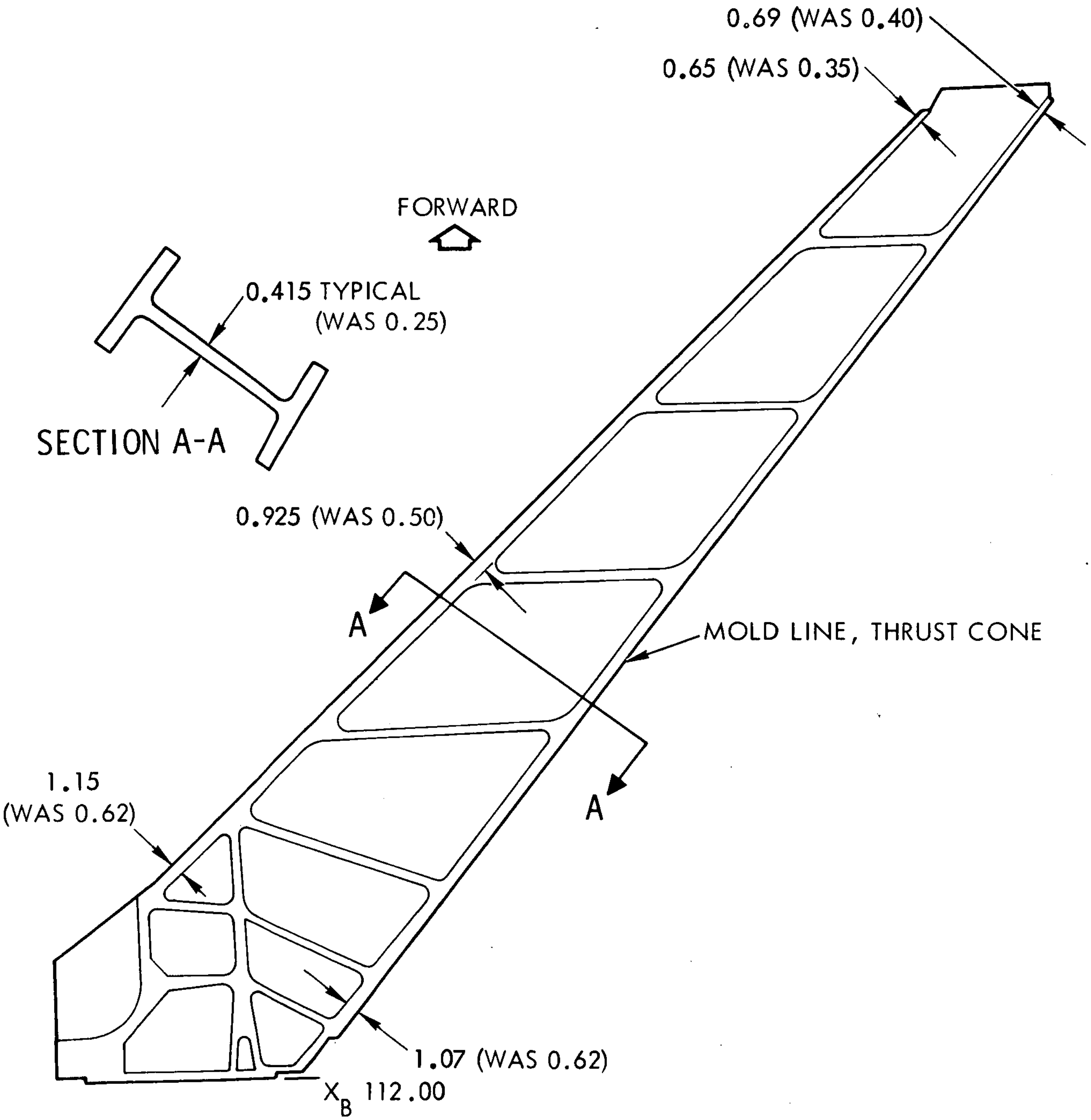


Figure 10.2-1. Thrust Cone Skin Panel Modification



NOTE: ALL MEASUREMENTS ARE IN INCHES
ALL STRINGERS INCREASED APPROXIMATELY 16.5% IN THICKNESS

Figure 10.2-2. Thrust Cone Stringer Modification



NOTE: ALL MEASUREMENTS ARE IN INCHES

Figure 10. 2-3. Thrust Longeron Modification

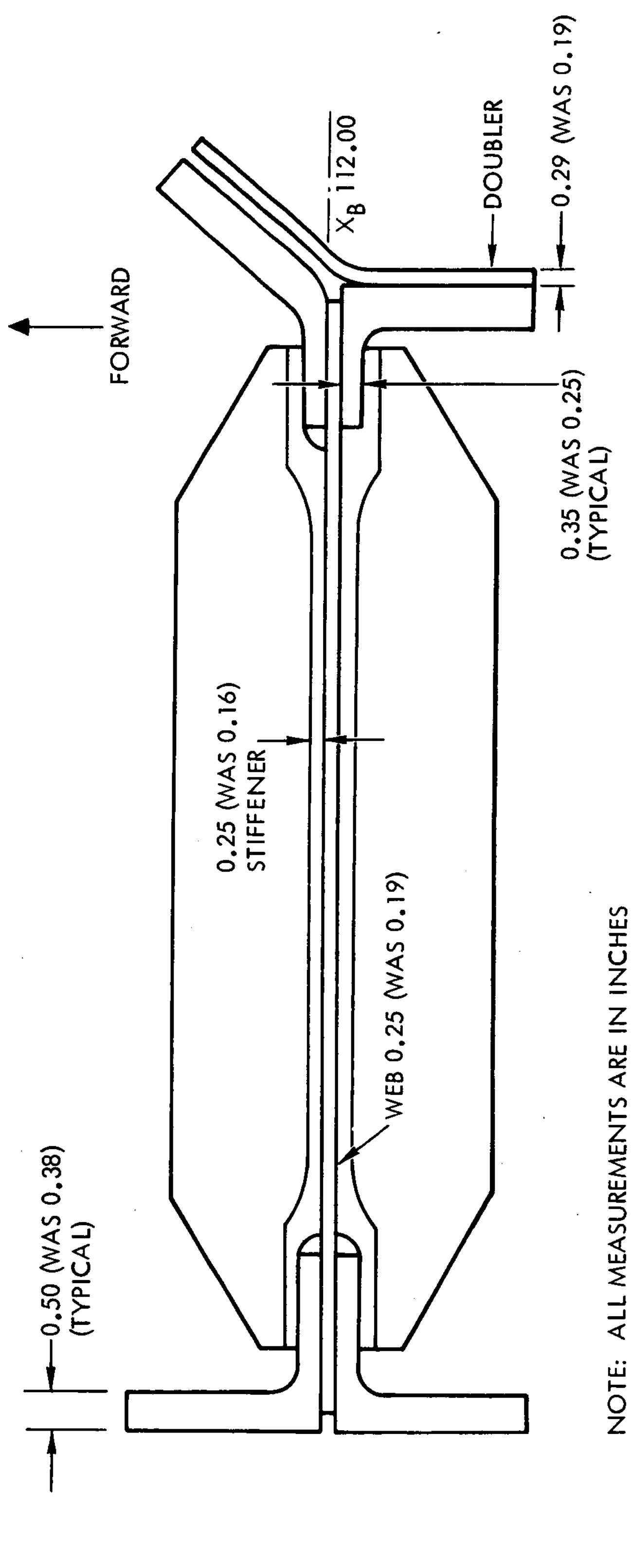


Figure 10. 2-4. Frame X_B 112 Modification

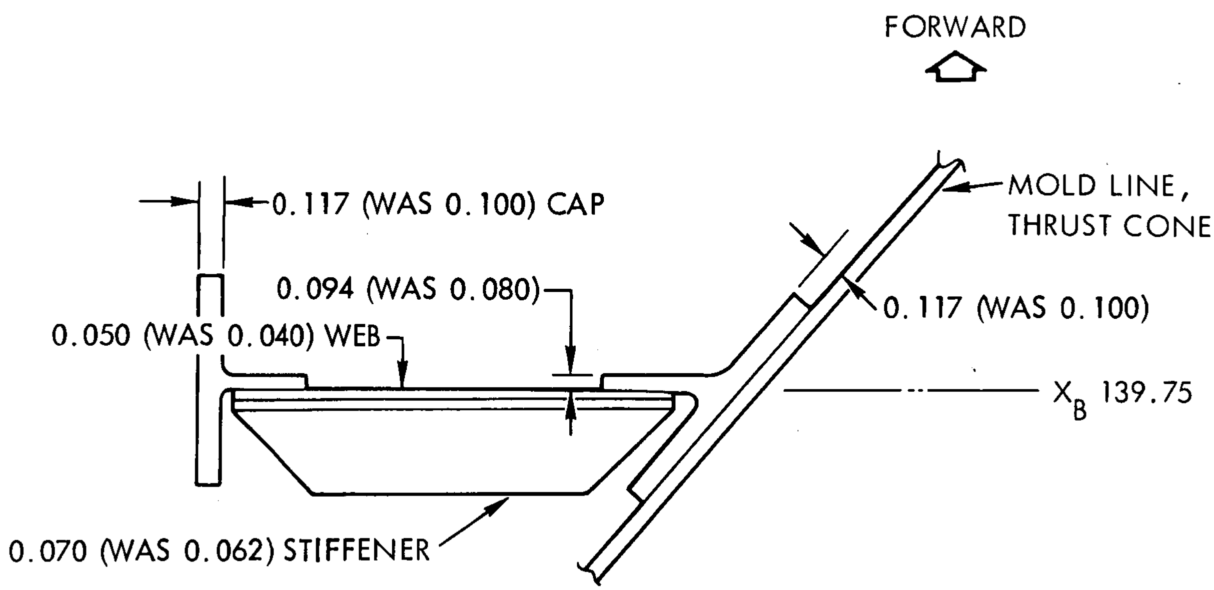


Figure 10.2-5. Frame X_B 139.75 Modification

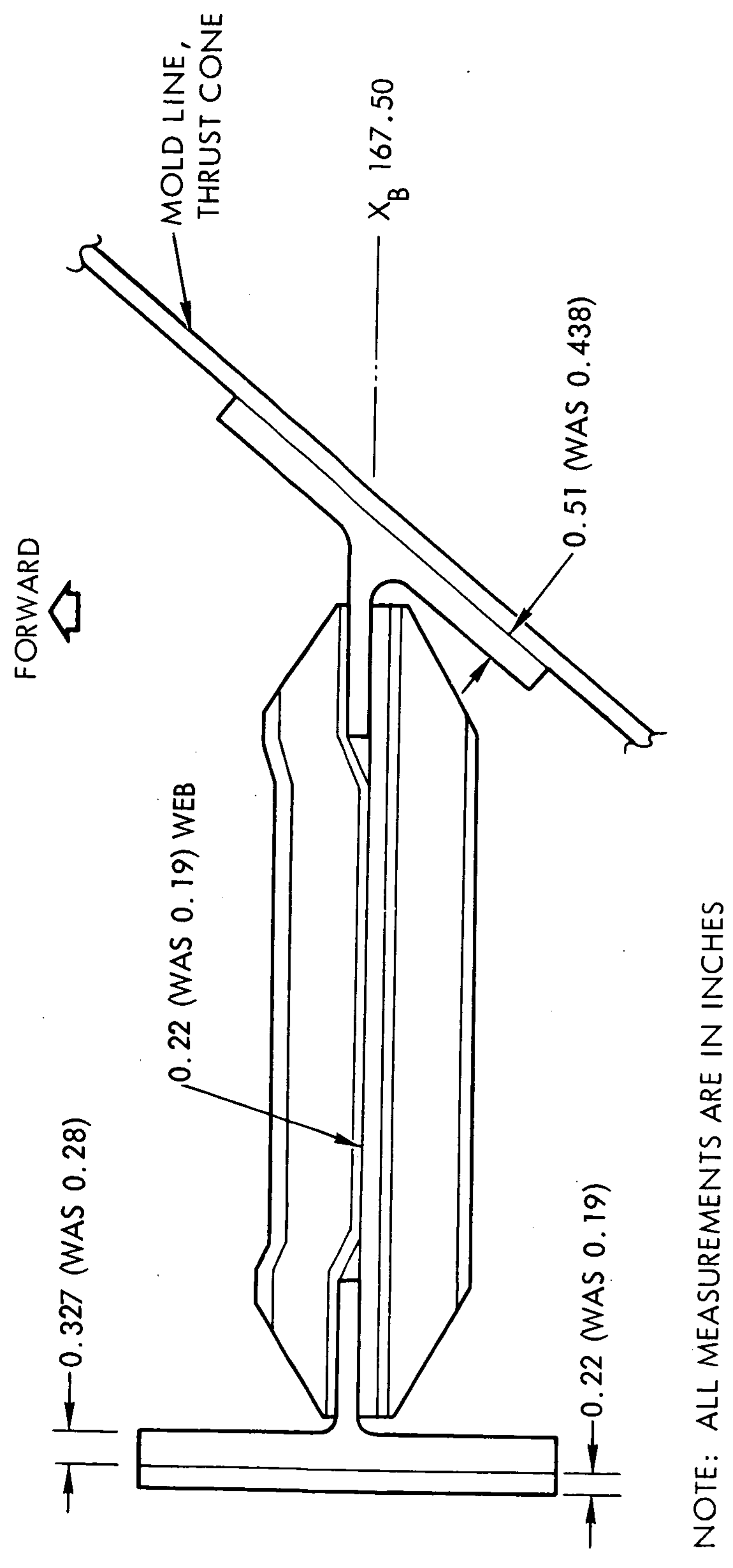


Figure 10.2-6. Frame X_B 167.50 Modification

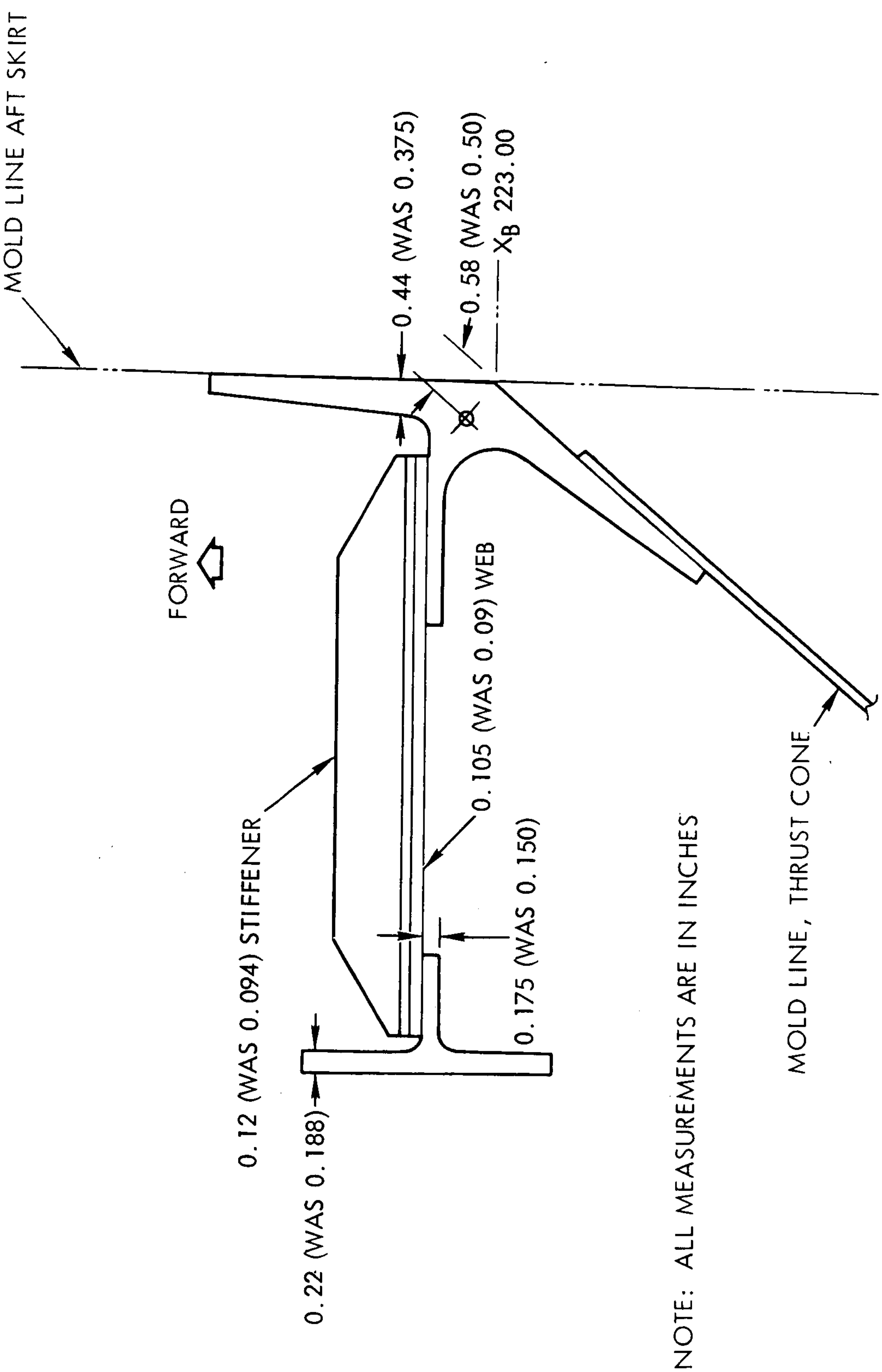


Figure 10.2-7. Frame X_B 223 Modification

2. Cap and web thickness of center engine support beam will be increased. The overall depth of beam is maintained (Figure 10.2-8).

3. Outboard engine thrust block will be locally redesigned to include additional bolts for reacting higher shear loads (Figure 10.2-9).

4. Brackets for deleted engine servicing, LOX and LH_2 recirculation, helium injection, and valve actuation systems will be removed as applicable.

5. Support brackets will be added for new LH_2 feed line purge system.

b. Base Heat Shield. Increased clearance requirements and the increased thermal environment of the J-2S engine require redesign of the rigid panel portion of the base heat shield. Minor modification of support struts, flexible curtains, and associated hardware are required as a result of the redesigned panel.

The basic configuration, material, and method of support for the J-2 heat shield is used in the redesign in order to minimize development cost and impact on associated systems.

Detail changes are as follows:

1. Rigid panel will be redesigned to include: clearance for larger J-2S engine configuration, approximately 1/2 inch greater aft (hot side) core thickness to increase content of the enclosed thermal resistive material (siloons), and addition of a stainless steel wire mesh screen over the entire aft surface for greater resistance to the combined effects of temperature and vibration (Figures 10.2-10 through 10.2-14).

2. Support brackets will be relocated on the forward side of the heat shield to maintain edge distance (Figure 10.2-10).

3. Flexible curtains and attachments will be modified as necessary to match engine hat band and rigid panel changes (Figure 10.2-13).

4. Support tubes and fittings will be modified to accommodate rigid panel revisions.

c. LOX Tank Aft Bulkhead. Two special LOX tank gore panels are replaced with plain gore panels to eliminate holes and bosses no longer required (because of deletion of the LOX recirculation system).

d. LH_2 Tank Skins. The LH_2 tank cylinder number two is simplified by eliminating holes and bosses (not required because of deletion of the LH_2 recirculation pump, by-pass lines, return line, and fairing).

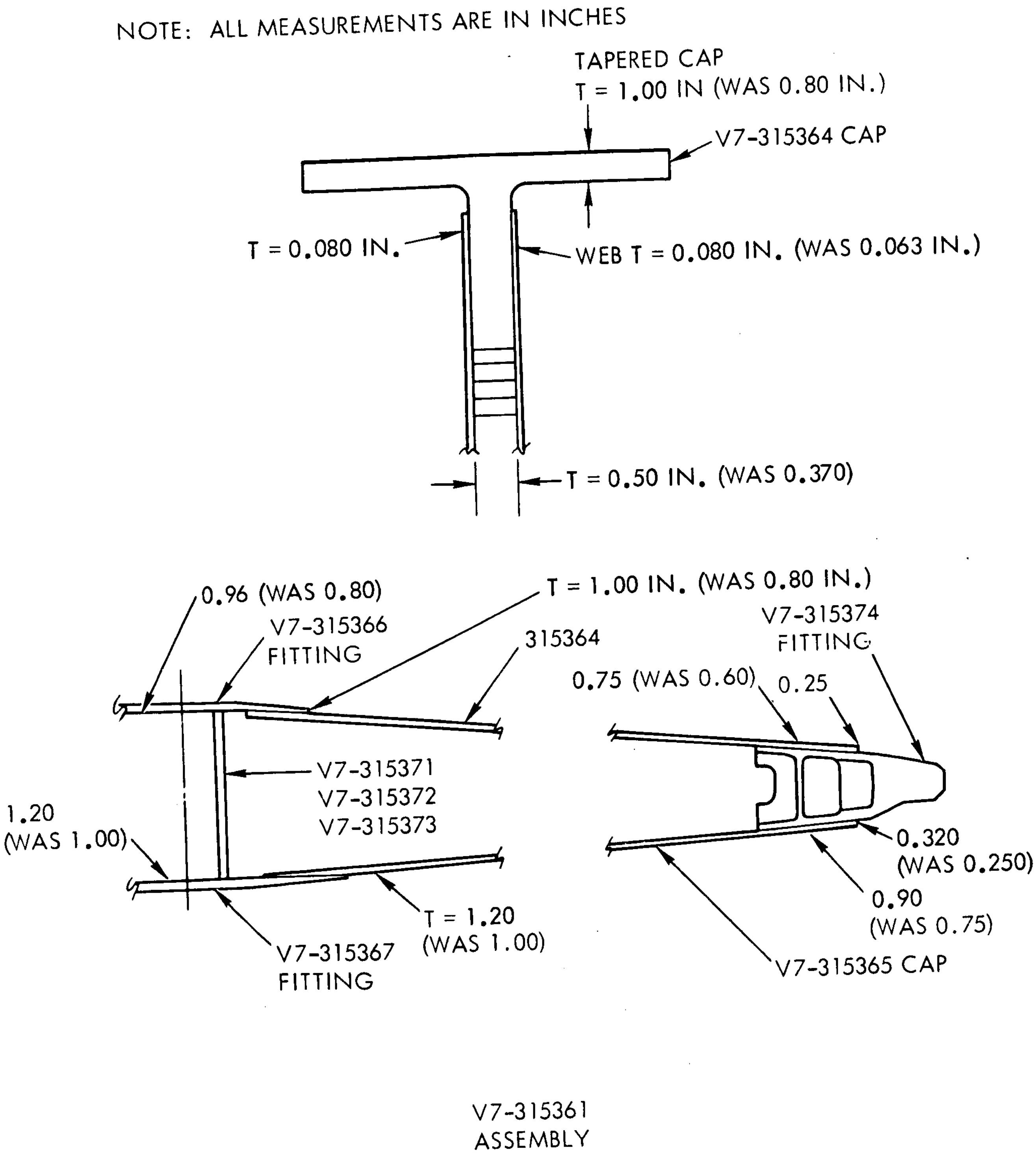


Figure 10.2-8. Center Engine Support Beam Modification

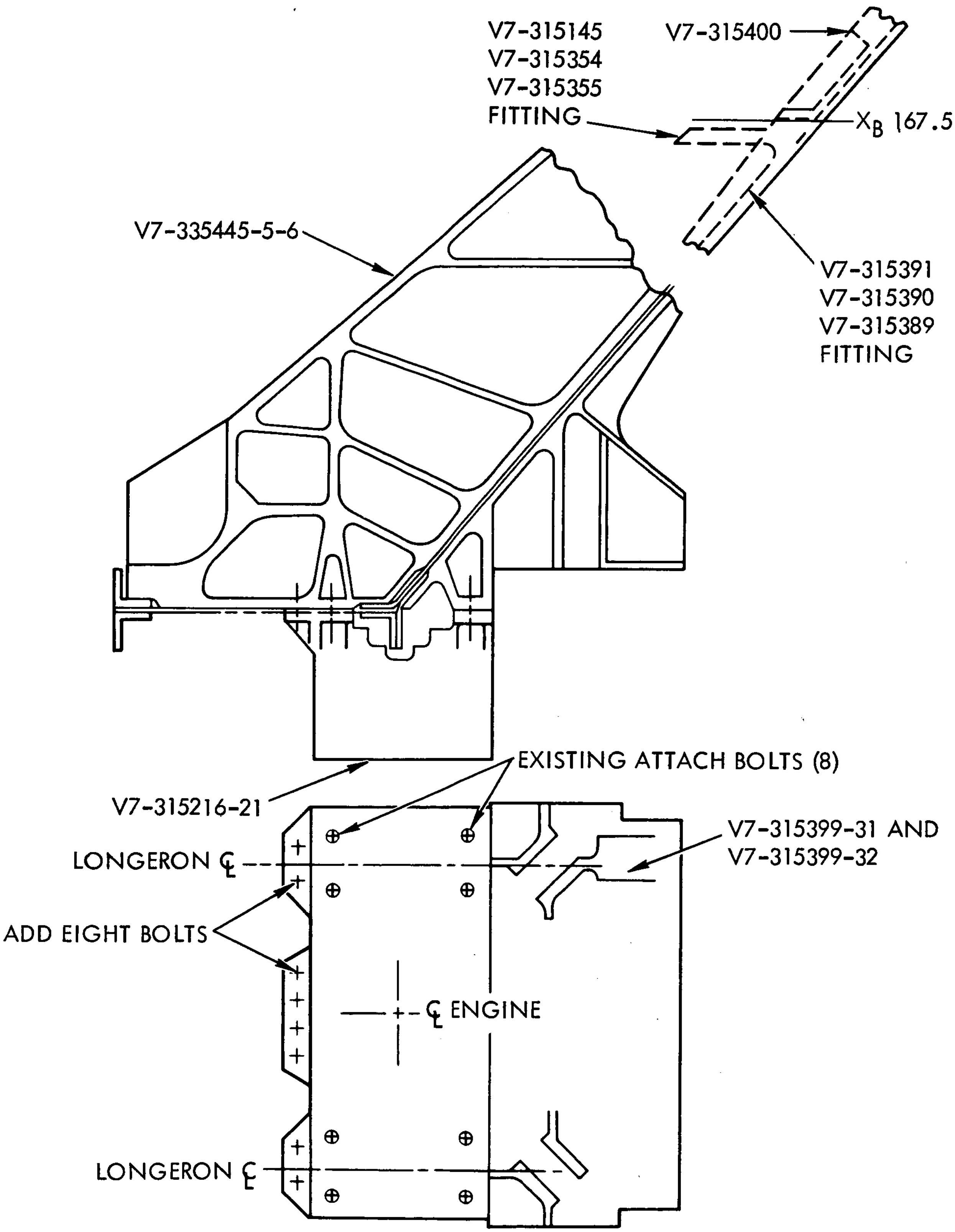


Figure 10. 2-9. Outboard Engine Thrust Block Modification

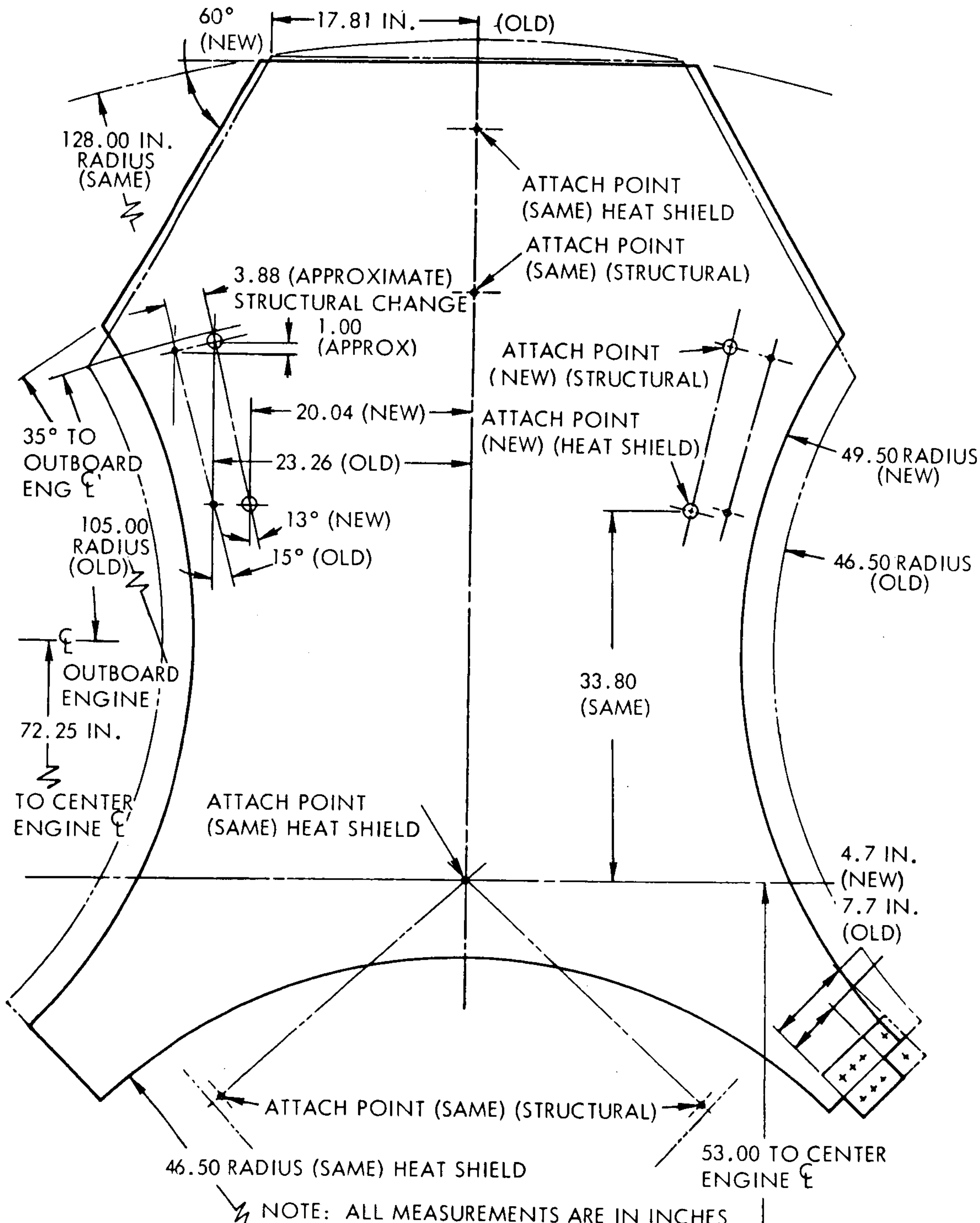


Figure 10. 2-10. Heat Shield Quarter-Panel Modification

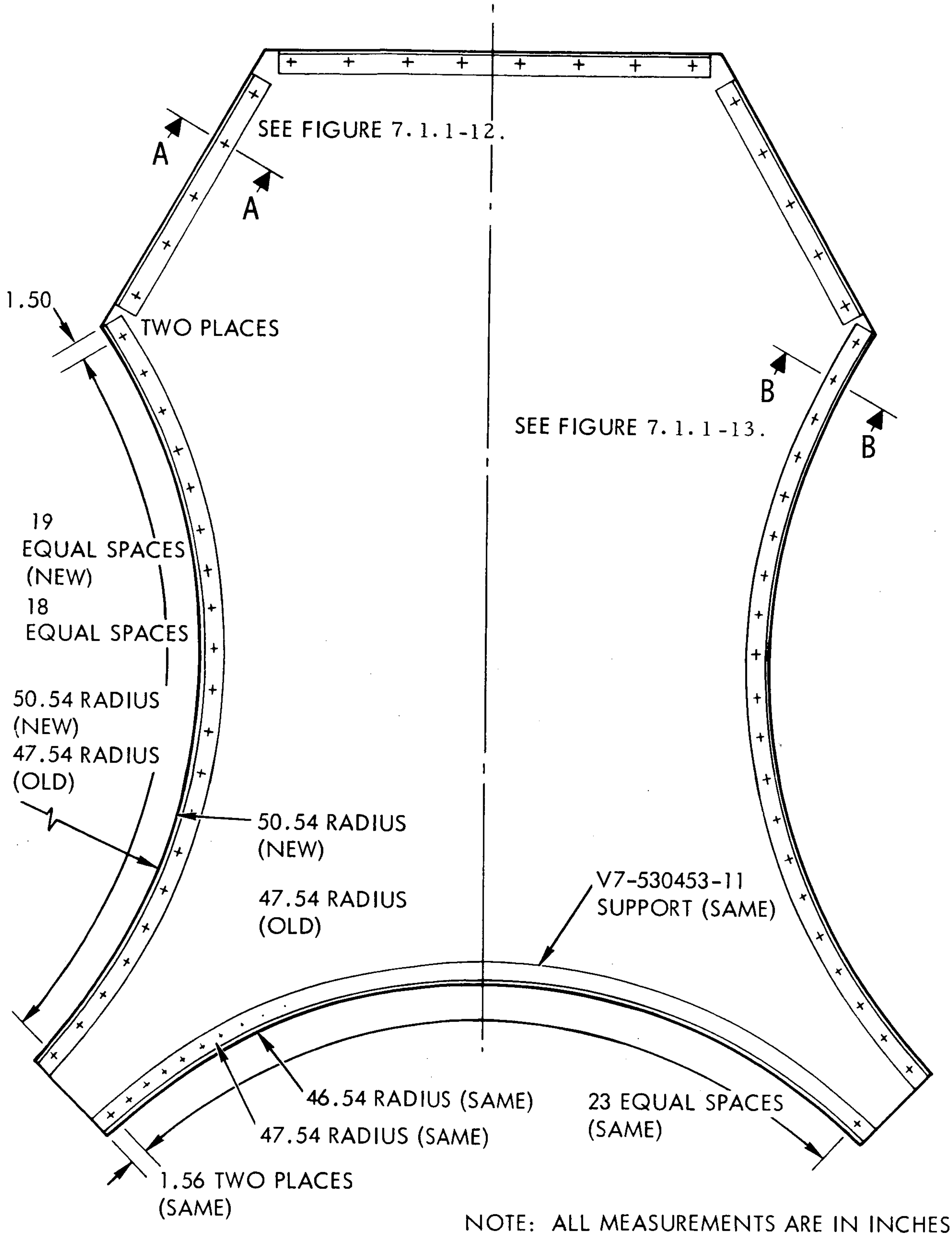


Figure 10.2-11. Heat Shield

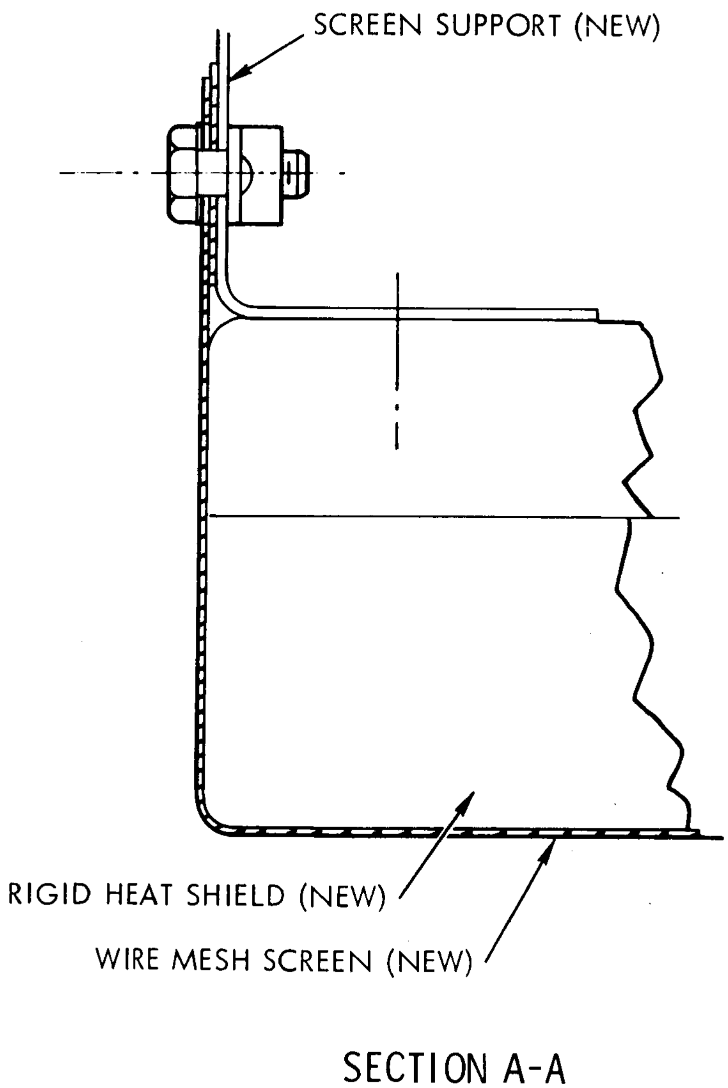


Figure 10. 2-12. Heat Shield

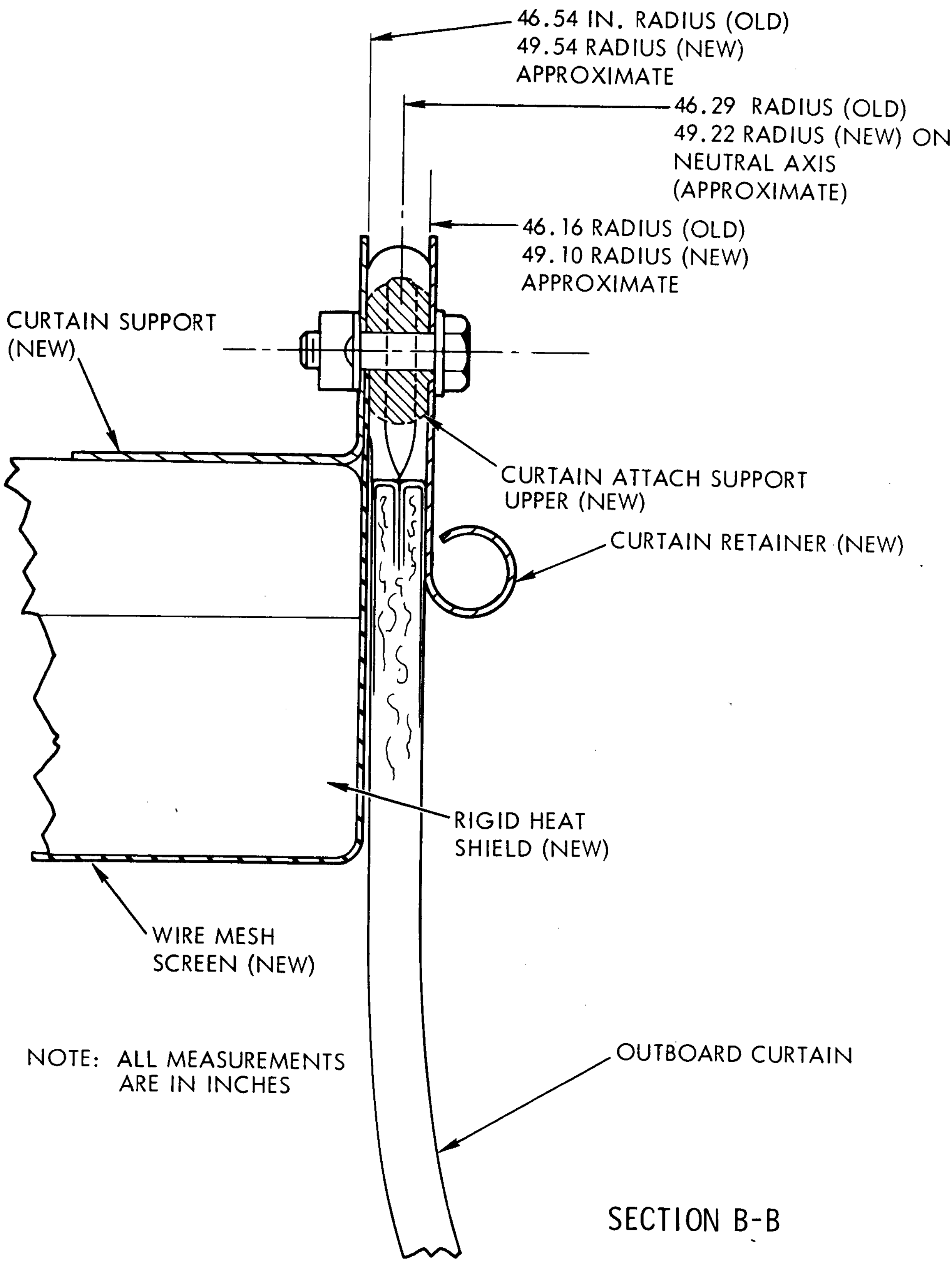


Figure 10.2-13. Heat Shield

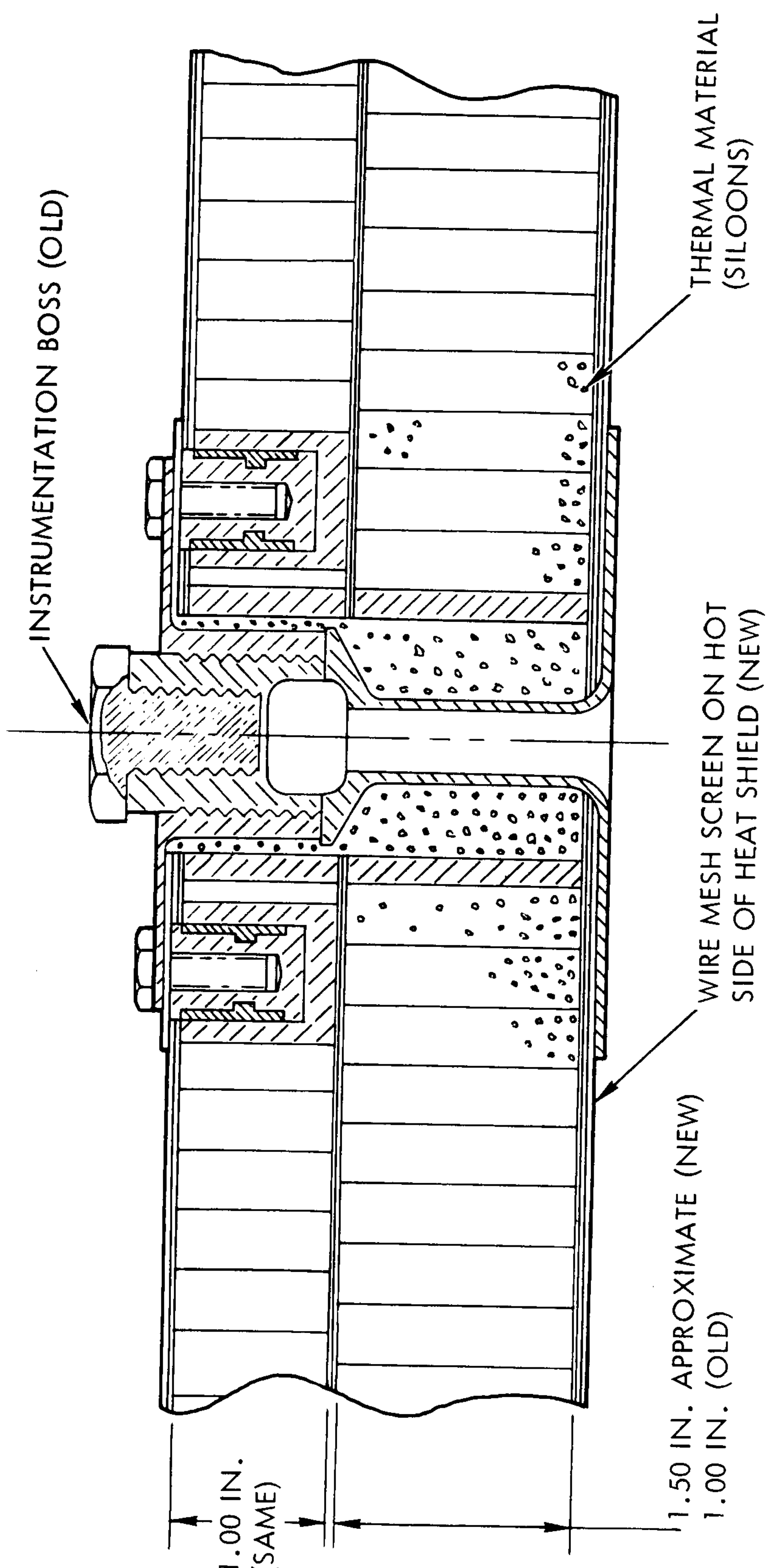


Figure 10.2-14. Heat Shield

e. Insulation. The LH2 tank sidewall insulation is simplified on the number two cylinder in the area of the recirculation system, fairing, and removed brackets.

f. Aft Interstage.

1. The ullage motor attachment fittings and structural reinforcement (not required because of ullage motor removal) are deleted.

2. External cork insulation to stringers and skin in the area of ullage motor removal is added for thermal protection of the structure (Figure 10.2-15).

3. The two LH_2 recirculation batteries and the associated support structure are removed due to system deletion.

10.2.1.1.2 PO Mission (Three-Stage Vehicle)

Study of the three-stage PO mission study was terminated at MSFC direction based upon trade studies performed by TBC (with inputs from the associate stage contractors), which showed that this mission would best be accomplished using a two-stage Saturn V (S-IC plus S-II).

Figure A-31, D5-15772-6, Structural Analysis, showed a comparison of S-II stage primary structural capability with the PO mission design envelope profile supplied by TBC. It is noted that the S-II structural capability is exceeded for the forward skirt, LH_2 tank sidewall, aft skirt and aft interstage structures. Therefore, structural strengthening of these areas would be required for the three-stage PO mission vehicle.

A preliminary study was performed prior to study termination to obtain rough-cut weight increases required to sustain the three-stage PO mission design (body) loads. Results of this study indicate approximately 11,500 pounds weight increase to the stage dry weights (8800 pounds, S-II stage dry weight increase, plus 2700 pounds of S-IC/S-II interstage weight increase).

10.2.1.1.3 PO Mission (Two-Stage Vehicle)

Figure 7.5-9 showed a comparison of the basic S-II stage structural capability with the polar orbit (two-stage) mission design envelope profile supplied by TBC. It is noted that the S-II structural capability exceeds the design loads. Therefore, no change to the basic airframe (body) structure is required except to accommodate functional systems modification, deletion, or additions as needed to implement J-2S engines. Study of the two-stage PO mission was also terminated at MSFC direction.

10.2.1.1.4 LEO Mission

The S-II structural capability exceeds the design loads. Therefore, no change to the basic airframe (body) structure is required except to accommodate functional systems

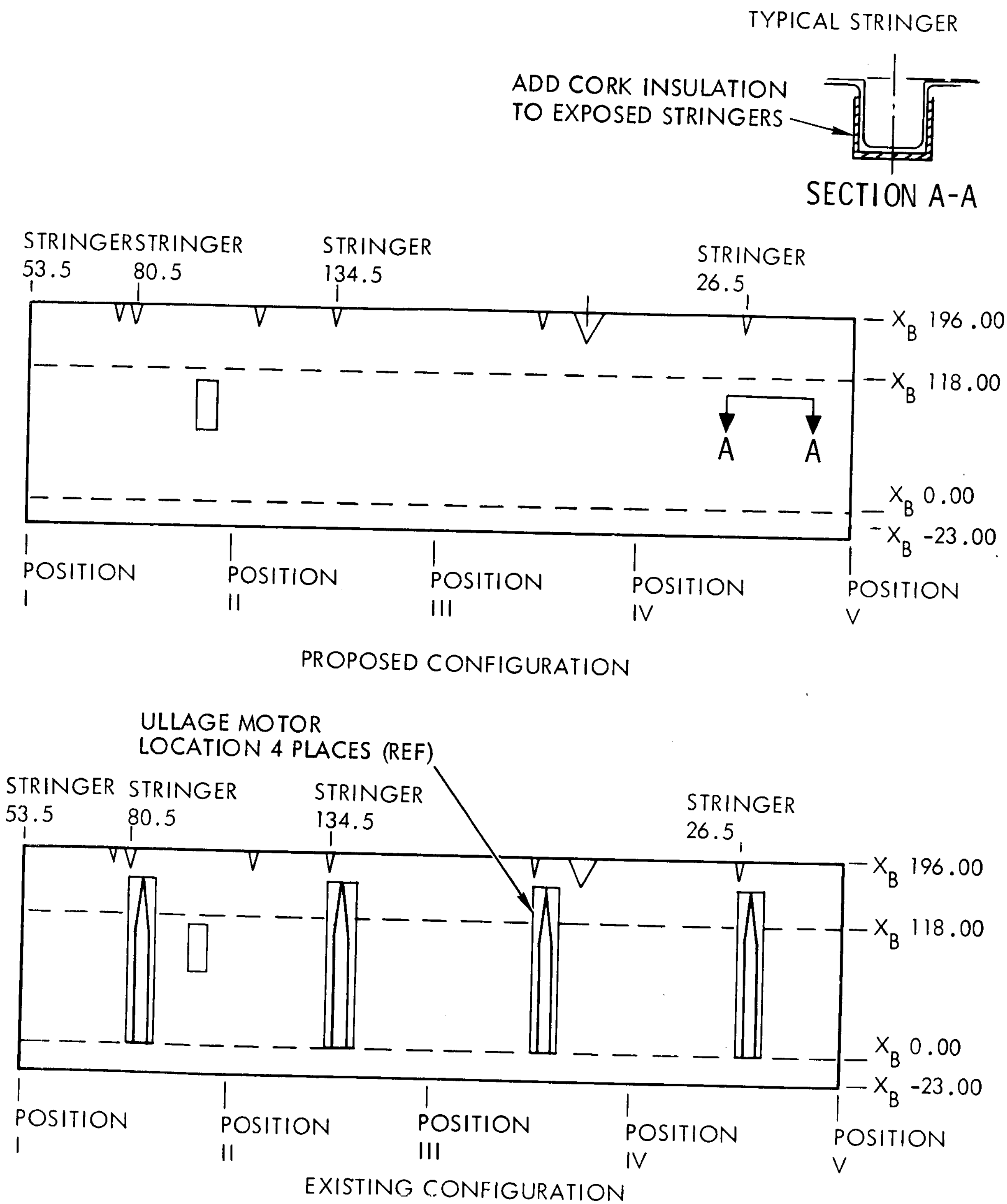


Figure 10. 2-15. Insulation Modification-Ullage Motor Removal

modification, deletion, or addition. Detail changes to structures in addition to those required for the LOR mission structures are listed below.

a. Forward Skirt. Addition of the new LH_2 tank balanced vent system requires the following changes:

   1. Add two holes and associated doublers to provide support for vent line nozzles (Figure 10.2-16).

   2. Add support brackets as necessary to support new one-inch diameter vent line system and two new vent valves.

b. LH_2 Tank and Forward Bulkhead. Addition of the new LH_2 tank balanced vent system requires the following changes:

   1. Modify the forward bulkhead gore panel to provide mounting provisions for new balanced vent line system (Figure 10.2-17).

   2. Modify forward bulkhead insulation to include insulation of the new balanced vent line outlet flange at the existing electrical interface outlet (Figure 10.2-17).

   3. Relocation of the LH_2 low-level sensors to Station 410 requires relocating five holes in the LH_2 tank Cylinder 2 to Station 390 (was Station 349).

c. Aft Skirt. Addition of the reaction control system requires the following changes in the aft skirt area. (Figures 10.2-18 and 10.2-23 show the general location of the two RCS modules).

   1. Add holes and associated doublers in the aft skirt skin for RCS module electrical, helium pressure, and thermal conditioning systems (Figure 10.2-19).

   2. Add reinforcement to the aft skirt structure as shown in Figures 10.2-20 and 10.2-21 to provide load paths to S-II structural frames at Stations 223, 224, and 283, and to provide adaptation for the RCS module to S-II stage interface utilizing present S-IVB module attachment locations and fittings. These fittings are retained since their design incorporates insulation to prevent heat transfer, and allows relative motion for thermal expansion between the module and the stage structure.

   3. Relocate holes to new locations for three LOX pump seal drain outlets.

d. Slosh Control System. Add a new conical slosh control baffle in the LOX tank to maintain settled propellant during low "g" coast flight and thus ensure J-2S engine restart capability (Figure 10.2-22).

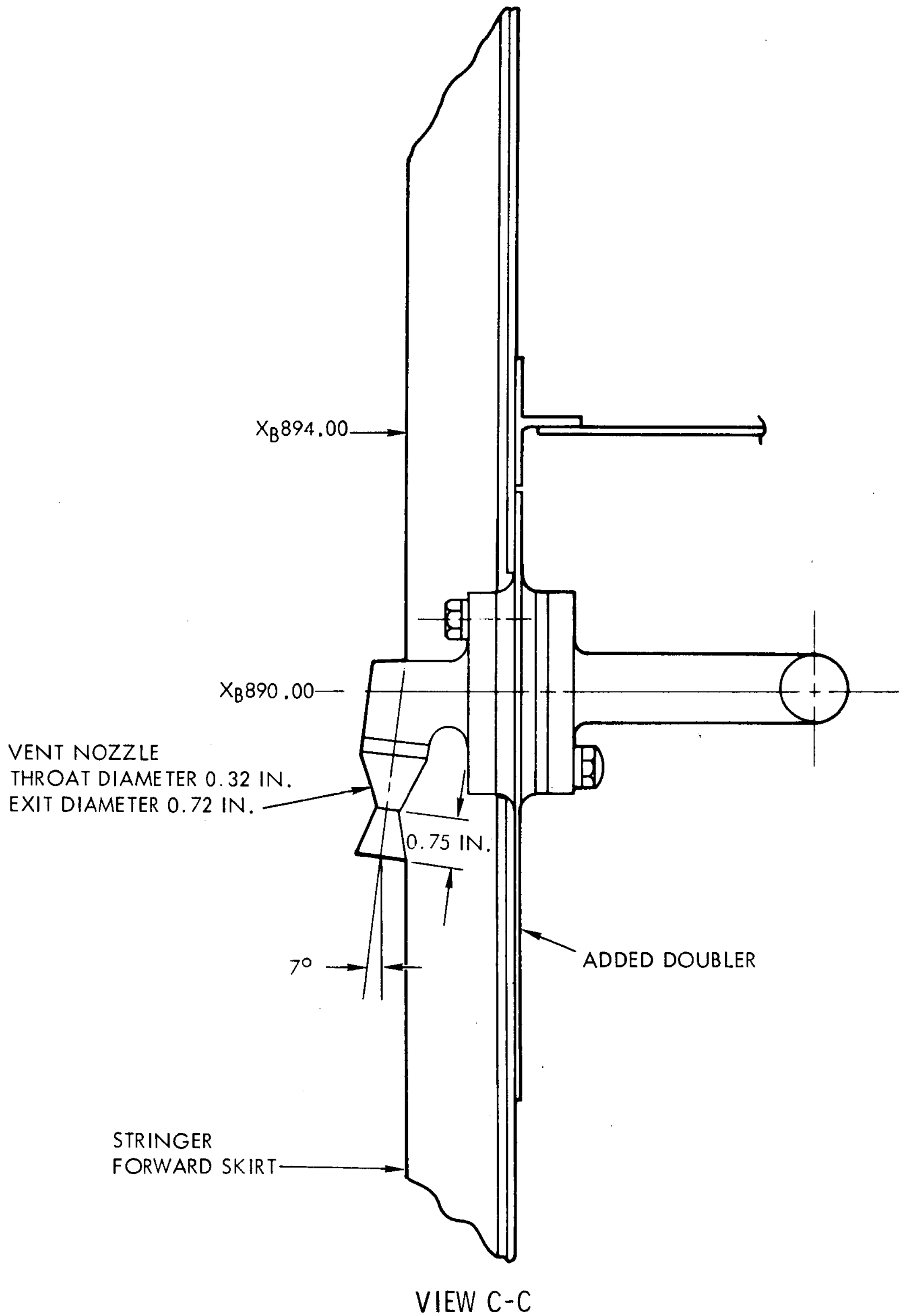


Figure 10.2-16. Forward Skirt Modification

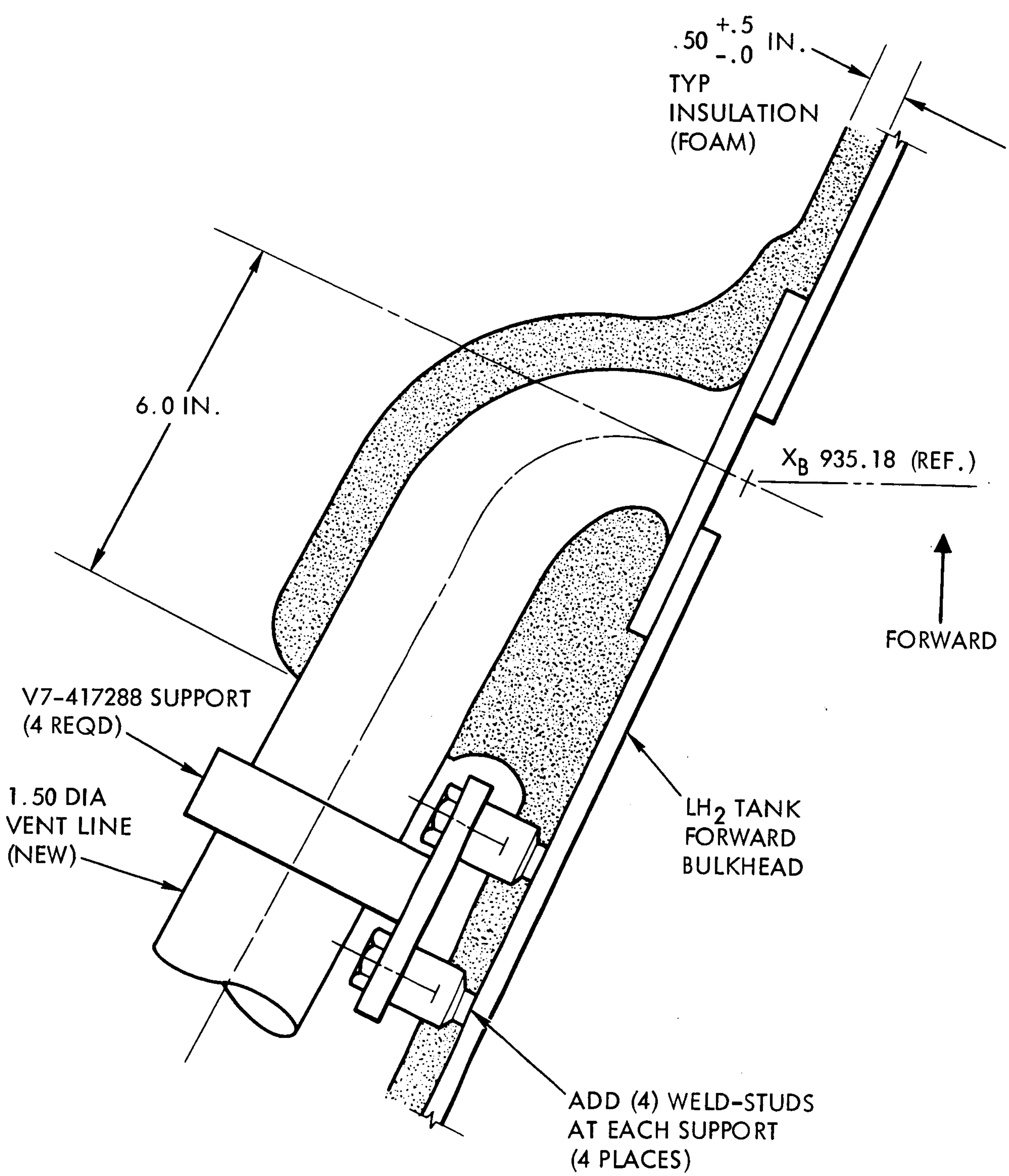


Figure 10.2-17. Forward Bulkhead & Insulation Modification

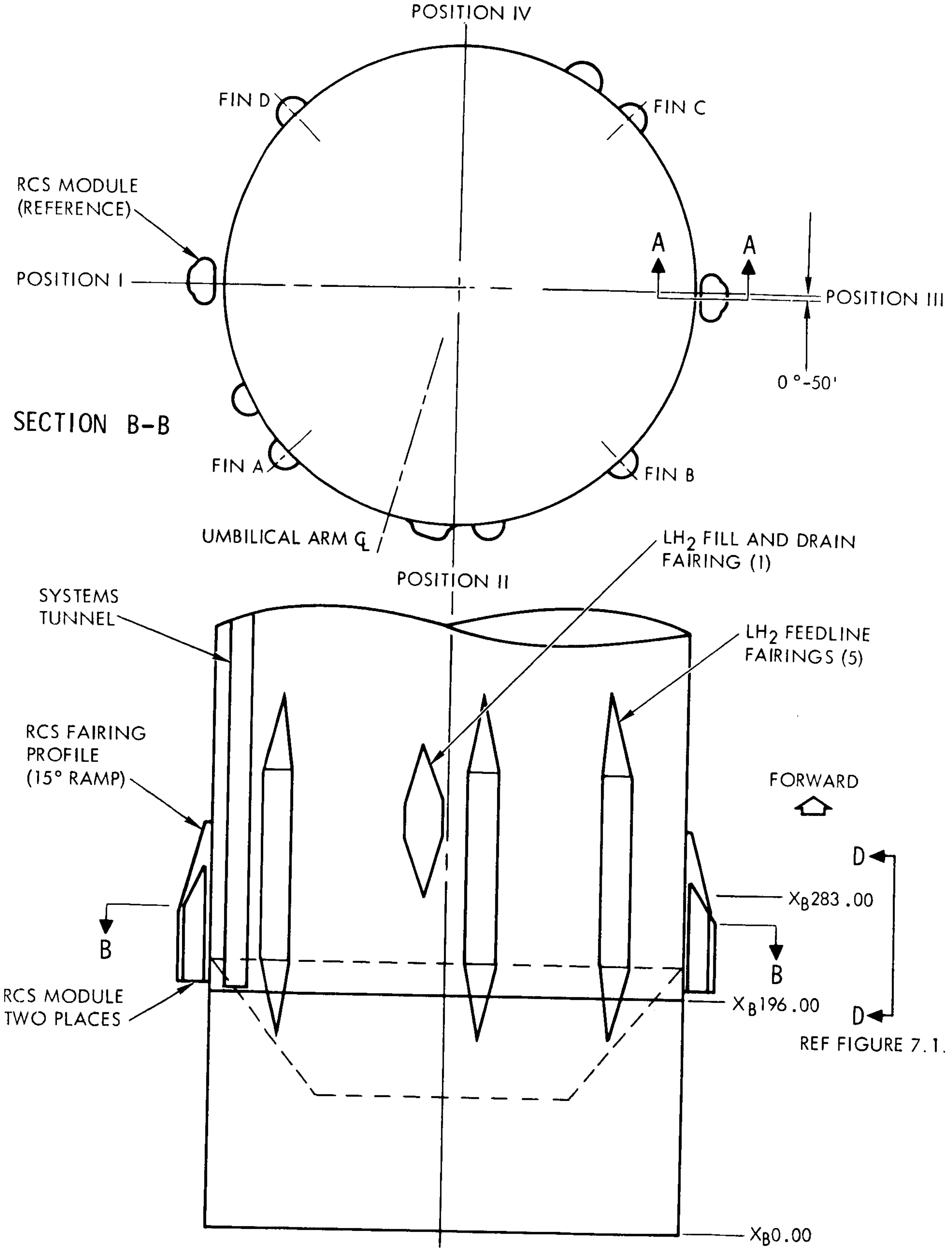


Figure 10.2-18. RCS Module Location ~ Aft Skirt

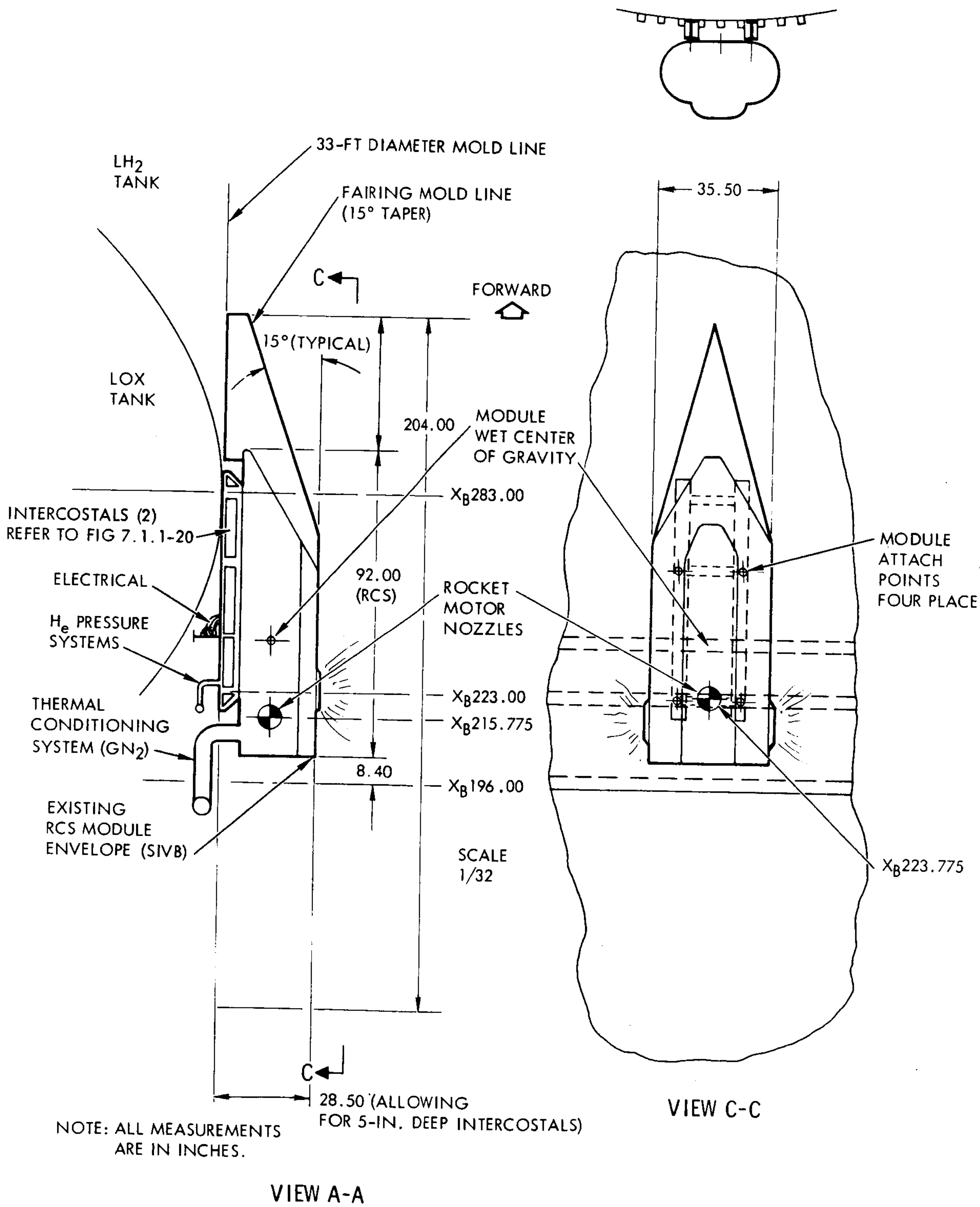


Figure 10. 2-19. RCS Module Provisions ~ Aft Skirt

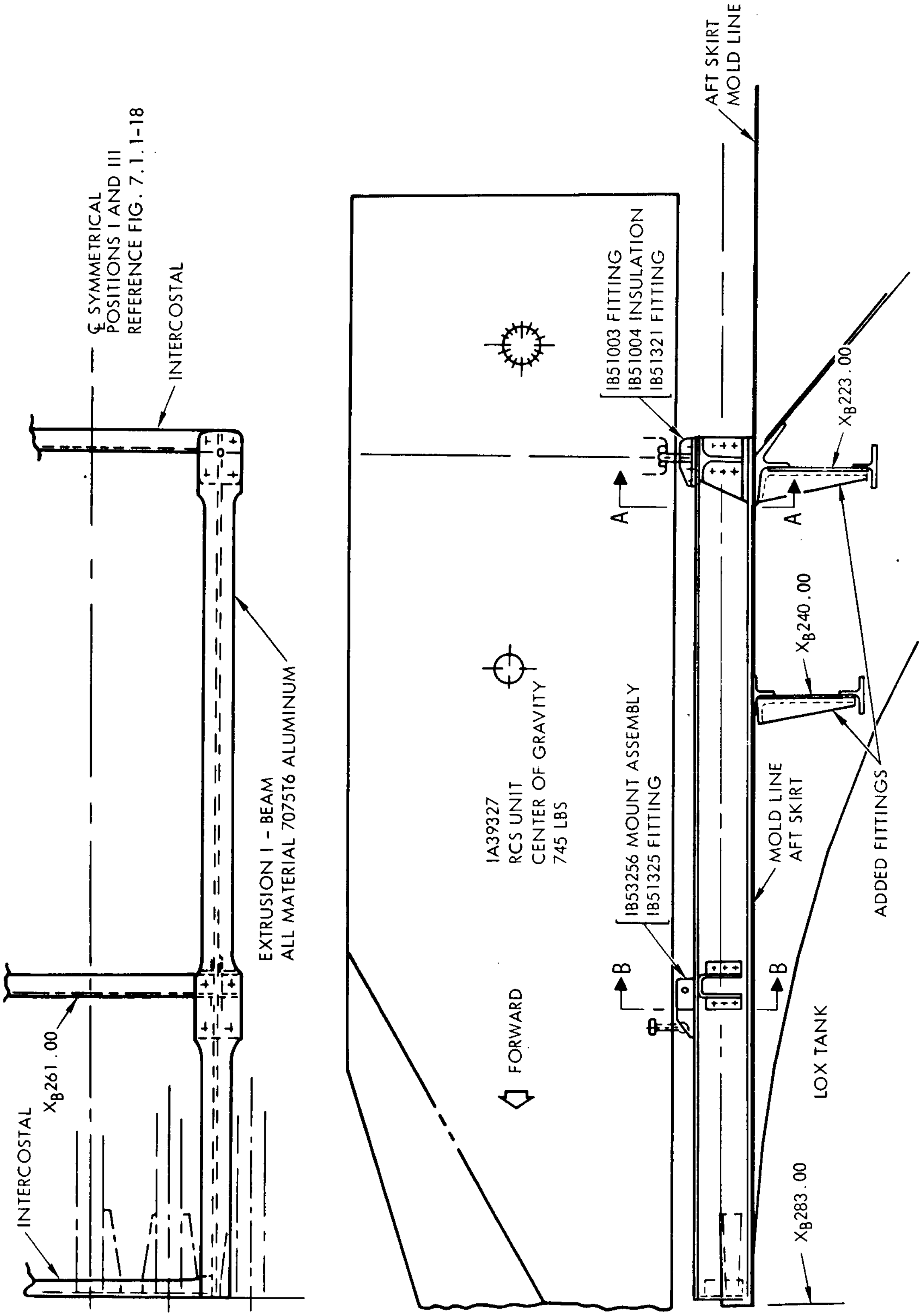


Figure 10. 2-20. RCS Module Support ~ Aft Skirt

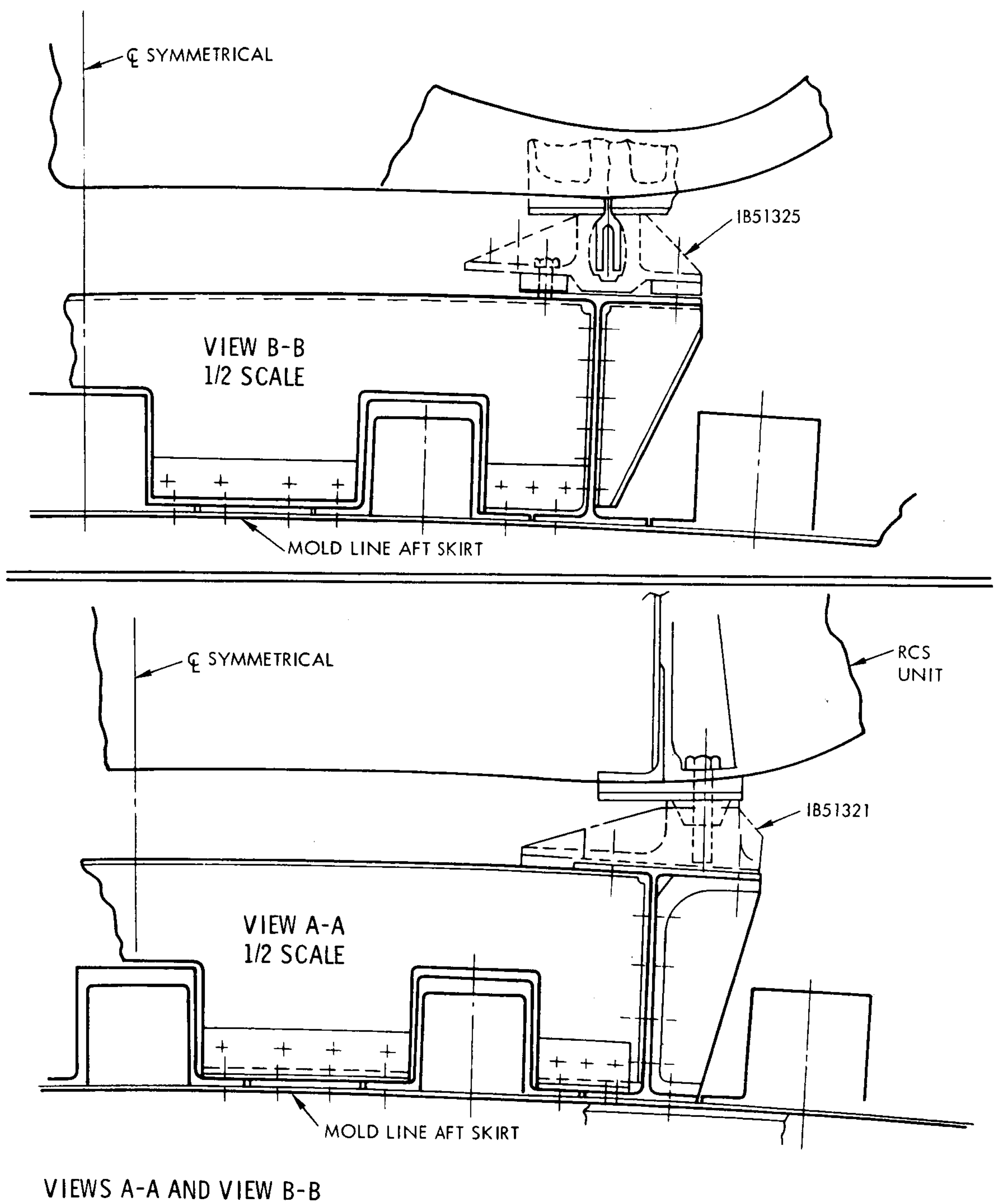


Figure 10.2-21. RCS Module Support ~ Aft Skirt

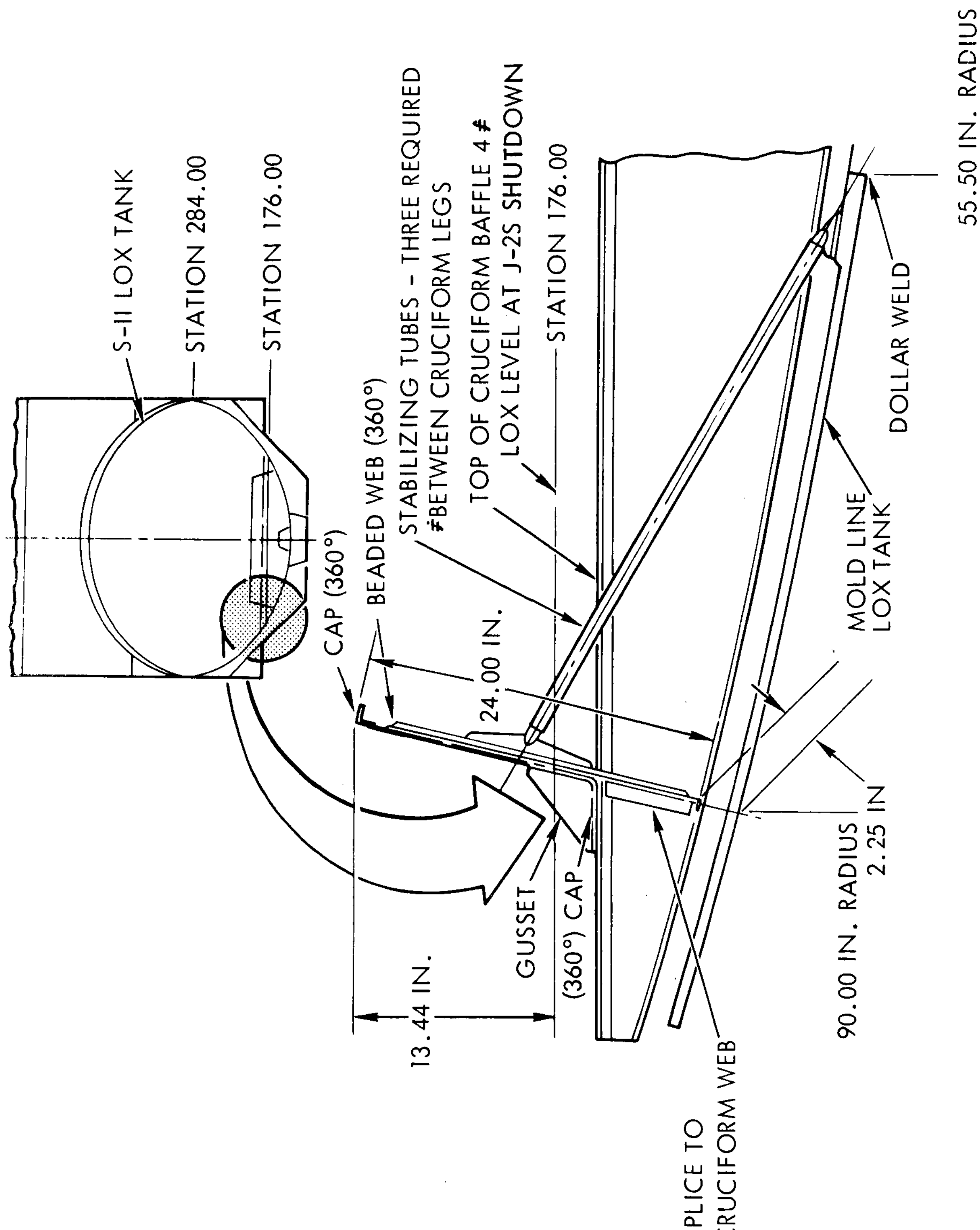
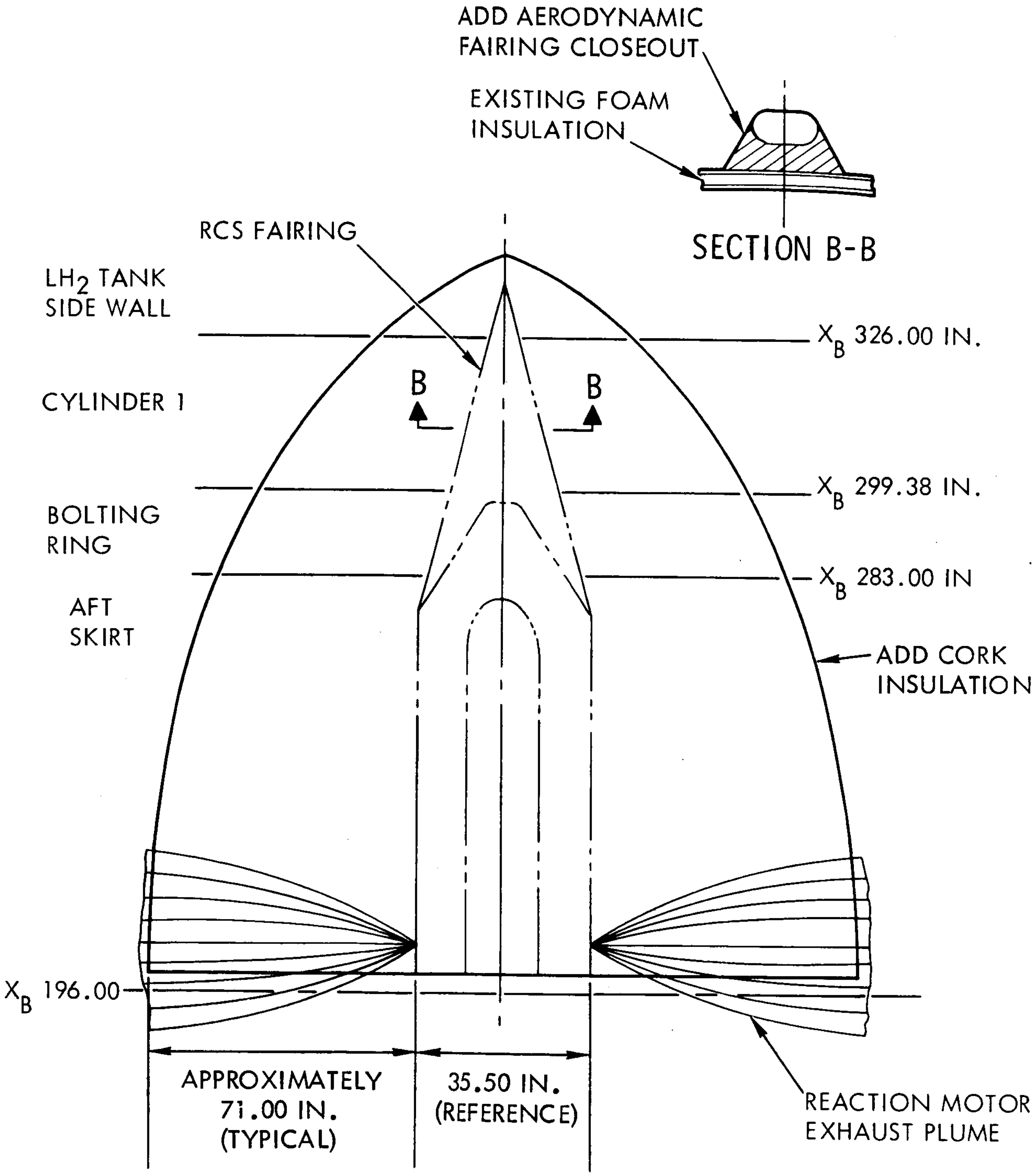


Figure 10.2-22. Slosh Control Baffle ~ LOX Tank



VIEW D-D

(SEE FIGURE 7.1.1-18)

Figure 10.2-23. RCS Module Provisions-LH_2 Tank Sidewall Insulation

e. Thrust Structure.

1. Provide supports on the thrust cone for one new battery container. The container is shock-mounted similar to existing electrical containers, and contains four batteries with associated power transfer switches to provide on-stage power for the hydraulic pumps of the thrust vector control system during coast flight (Figure 10.2-24).

2. The forward mounting bolts for the modified ARMA package require relocation. The package is extended approximately 6.5 inches forward to accommodate the new DC auxiliary motor pump.

3. Add mounting provision for new 10-inch diameter bottle (TVC air supply) (Reference figure 10.2-25).

f. Insulation. The following changes to stage and electrical container insulation are required:

1. Increase cork insulation thickness and density on all S-II electrical containers to maintain internal environment within present operational design limits during coast flight.

2. Add external cork insulation in the vicinity of the RCS module as shown in Figures 10.2-18 and 10.2-23 to protect the LH_2 tank sidewall insulation against aerodynamic heating or RCS thruster impingement.

3. Modify LH_2 tank sidewall insulation at the RCS module locations to provide an aerodynamic fairing closeout between the module and the sidewall insulation (Section B-B of Figure 10.2-23).

4. Modify the LH_2 tank sidewall insulation to include new flange and electrical connector at Station 390 (five places).

10.2.1.2 Structural Interface Changes Between Stages

The primary body structural capability of the S-II stage does not exceed the design envelopes for the LOR, LEO, and two-stage PO missions. Therefore, no structural interface changes between stages will be required for those missions. Interface changes would be necessary for the three-stage PO mission, but study of this mission was terminated.

10.2.1.3 Stage Stiffness Properties

The S-II stage axial, shear, bending, and torsional stiffness properties are shown in Table 10.2-I for a sequence of vehicle station increments. Because no design changes to the primary body (shell) structure are required for the LOR, LEO, and two-stage PO missions, these stiffness properties are identical to those for the S-II-11 vehicle. Stiffness properties for the three-stage PO mission were not developed.

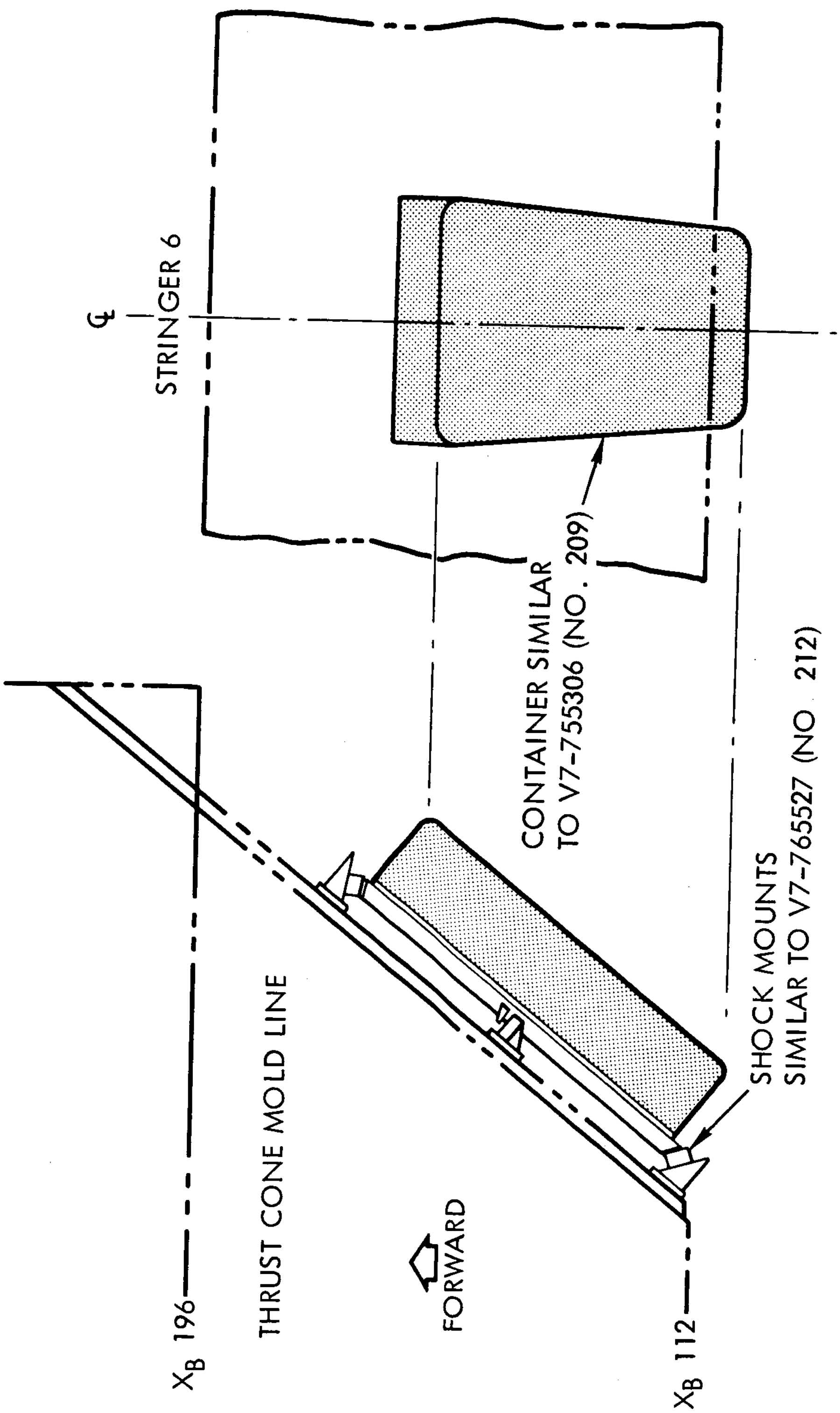


Figure 10.2-24. New Battery Container Support-Thrust Cone

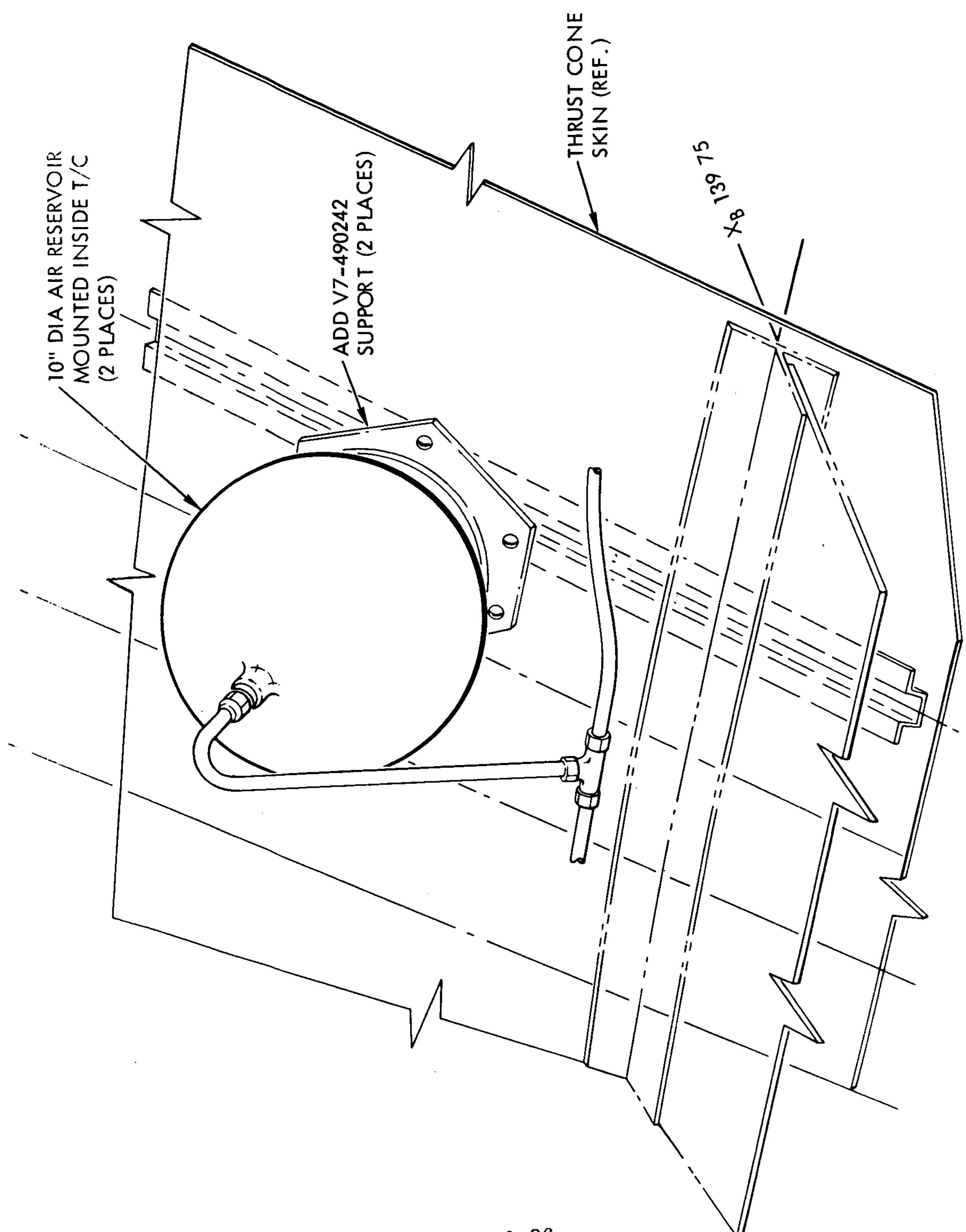


Figure 10.2-25. Installation of Thrust Vector Control System Air Reservoir

Table 10.2-I. S-II Stage Stiffness Properties

Vehicle Station	Axial EA pounds x 10^{-6}	Torsion GJ pound x $inches^2$ x 10^{-12}	Bending EI pound x $inches^2$ x 10^{-12}	Shear KAG pounds x 10^{-6}
1541 - 1564	2945	13.7	58	174
1564 - 1760	2797	13.7	55	174
1760 - 1804	2550	13.7	50	174
1804 - 1847	4120	13.7	81	174
1847 - 1863	3590	32.4	70	414
1863 - 1890	3270	29.0	64	370
1890 - 1989	2170	29.0	43	370
1989 - 2088	2170	29.0	43	370
2088 - 2187	2190	29.2	43	374
2187 - 2286	2200	29.5	43	376
2286 - 2385	2215	29.7	43	379
2385 - 2402	2576	24.1	50	307
2402 - 2430	2053	16.4	40	209
2430 - 2458	2426	11.2	48	142
2458 - 2494	2465	11.8	48	150
2494 - 2519	2348	11.4	46	145

NOTES: 1. Properties are based on S-II-11 stage (Baseline vehicle for J-2S study).

2. Data is applicable for J-2S LOR and LEO mission stages.

10.2.2 MECHANICAL SYSTEMS

This section describes the changes to the S-II stage mechanical systems to accept a J-2S engine in place of the J-2. The systems concerned are the engine system, propellant feed system, propellant management system, pressurization system, valve actuation system, flight control system, and the environmental control system. An indication of the simplification afforded by the J-2S incorporation may be gained by comparing the mechanical systems launch redlines for a J-2/S-II and a J-2S/S-II stage in Tables 10.2-II and 10.2-III, respectively.

10.2.2.1 Engine System

This section describes those stage changes associated with the higher performance of the J-2S engine, the revised engine start system using a solid propellant turbine spinner, the relaxation of required propellant inlet conditions at start command, revised servicing requirements, and idle mode operation of the engine.

10.2.2.1.1 LOR Mission

a. Engine Performance. The nominal steady state altitude thrust level of the J-2S engine is 265,000 pounds (stage thrust of 1,325,000 pounds) as compared to 230,000 pounds (stage thrust of 1,150,000 pounds) for the J-2 engine. This increase in thrust and an associated increase in specific impulse (427 versus 421 seconds minimum) are achieved by operating the engine at a higher thrust chamber pressure level and use of a higher expansion ratio nozzle (40:1 versus 27.5:1).

   The thrust buildup characteristics of the J-2S as installed in the S-II stage are presented in Figure 10.2-26. The primary fluid requirements to assure this buildup are propellants in the engine for a minimum of one hour prior to engine start command, LH_2 inlet pressures between 27 and 40 psia, LOX inlet pressures between 33 and 45 psia, and liquid at the tank outlets.

   Engine circuitry is such that the engine start command initiates idle mode operation after which a mainstage start command is given to bring the engine to full operation. During idle mode, the engine is operating by virtue of the tank ullage pressure only, with no turbopump operation. The S-II application limits this time period between engine start command and mainstage command to 1.0 seconds because of second stage separation criteria and J-2S starting requirements.

   The engine start sequence is shown in Figure 10.2-27 (with the J-2 start sequence included for comparison).

   As noted, for S-II usage the J-2S engine will be calibrated to the nominal 265,000 pounds $\pm$ 3 percent thrust level at a 5.5 $\pm$ 2 percent engine mixture ratio (EMR). During the start transient the propellant utilization (PU) valve will be commanded to the null or 5.0 EMR position. At this EMR each engine will produce a nominal thrust of 237,500 pounds. At mainstage start command plus 4.5 seconds the PU valve will be commanded to the 5.5 EMR position. The

Table 10.2-II. Countdown Redlines With J-2 Engine

Description	Values		Time Period Notes
	Minimum	Maximum	
Temperature, Engine LOX Pump Discharge (Five measurements for Engines 1 through 5)	None	-284 F	Commit at T-22 seconds
Temperature, Engine LH_2 Pump Inlet (Five measurements for Engines 1 through 5)	None	-420.5 F	Commit at T-22 seconds NOTE: The margin between the measured engine inlet LH_2 temperatures and the operational limit may be less than the instrumentation error. A redline correction procedure is recommended when a value of 0.2 degrees error is evident, based on saturation pressure and temperature conditions.
Pressure, LOX Tank Ullage, psia	Ambient	30	During propellant loading
	None	43	From initiation of LOX tank pressurization (T-187 seconds) until T-30 seconds
	36	43	From LOX tank pressurization complete unitl T-30 seconds
Pressure, LH_2 Tank Ullage, psia	Ambient	30	During propellant loading
	None	37	From initiation of LH_2 tank pressurization (T-97 seconds) until T-30 seconds
	33	37	From LH_2 tank pressurization complete until T-30 seconds

Table 10.2-II. Countdown Redlines With J-2 Engine (Cont)

Description	Values		Time Period Notes
	Minimum	Maximum	
Pressure, Valve Actuation Helium Bottle, psia	2800	3450	Commit at T-19 seconds
Pressure, Valve Actuation Regulator Outlet, psia	690	815	From bottle pressurization complete until start of automatic sequence (T-187 seconds)
Pressure, Engine LH_2 Pump Inlet, (Five measurements for Engines 1 through 5), psia	41.0	None	From LH_2 pressurization complete until T-30 seconds
Pressure, Helium Injection Bottle, psia	2800	3450	From start of helium injection (T-30 minutes) to T-30 seconds
Pressure, Common Bulkhead Internal, psia	None	3	Commit at T-187 seconds
Pressure, Engine Helium Tank (five measurements), psia	2800	3450	Commit at T-187 seconds
Pressure, Engine Start Tank (five measurements), psia	*	*	From start of auto sequence (T-187 seconds) until T-30 seconds
Pressure, Hydraulic Accumulator Gas (four measurements), psia	3000	-	From lock-up until 3 minutes after lock-up
Pressure, Helium Injection Orifice Outlet, (five measurements), psia	200	300	From start of He injection until T-15 minutes

Table 10.2-II. Countdown Redlines With J-2 Engine (Cont)

Description	Values		Time Period Notes
	Minimum	Maximum	
Pressure, LOX Vent Valve Actuation, psia	-	350	At T-22 seconds
Pressure, LH_2 Vent Valve Actuation, psia	-	350	At T-22 seconds
Temperature, Engine Start Tank (five measurements)			From start of auto sequence (T-187 seconds) until T-30 seconds
Temperature, Thrust Chamber Jacket (five measurements)	-	-170	Commit at T-187 seconds
Temperature, Reservoir Outlet Fluid (four measurements)	*	*	Commit at T-187 seconds
Temperature Fill and Drain Coupling	-250	-	At T-33 seconds
PU Valve Position (five measurements)	Null $\pm 2^{o}$ Figure	Null $\pm 2^{o}$ Figure	From T-10 minutes until T-187 seconds
Hydraulic Reservoir Piston Position (four measurements)			Commit at T-187 seconds

*See launch mission rules

Table 10.2-III. Countdown Redlines With J-2S Engine

Description	Values		Time Period Notes
	Minimum	Maximum	
Pressure, LOX Tank Ullage, psia	Ambient	30	During propellant loading
	None	43	From initiation of LOX tank pressurization (T-187 seconds) until T-30 seconds
	36	43	From LOX tank pressurization complete until T-30 seconds
Pressure, LH_2 Tank Ullage, psia	Ambient	30	During propellant loading
	None	37	From initiation of LH_2 tank pressurization (T-97 seconds) until T-30 seconds
	33	37	From LH_2 tank pressurization complete until T-30 seconds
Pressure, Valve Actuation Helium Bottle, psia	700	750	From bottle pressurization complete until T-19 seconds
Pressure, Common Bulkhead Internal, psia	None	3	Commit T-187 seconds
Pressure, Engine Helium Tank (five measurements), psia	2800	3450	Commit at T-187 seconds

Table 10.2-III. Countdown Redlines with J-2S Engine (Cont)

Description	Values		Time Period Notes
	Minimum	Maximum	
Temperature, Reservoir Outlet Fluid (four measurements)	*	*	Commit at T-187 seconds
Pressure, Hydraulic Accumulator Gas (four measurements), psia	3000	-	From lock-up until 3 minutes after lock-up.
*See launch mission rules			

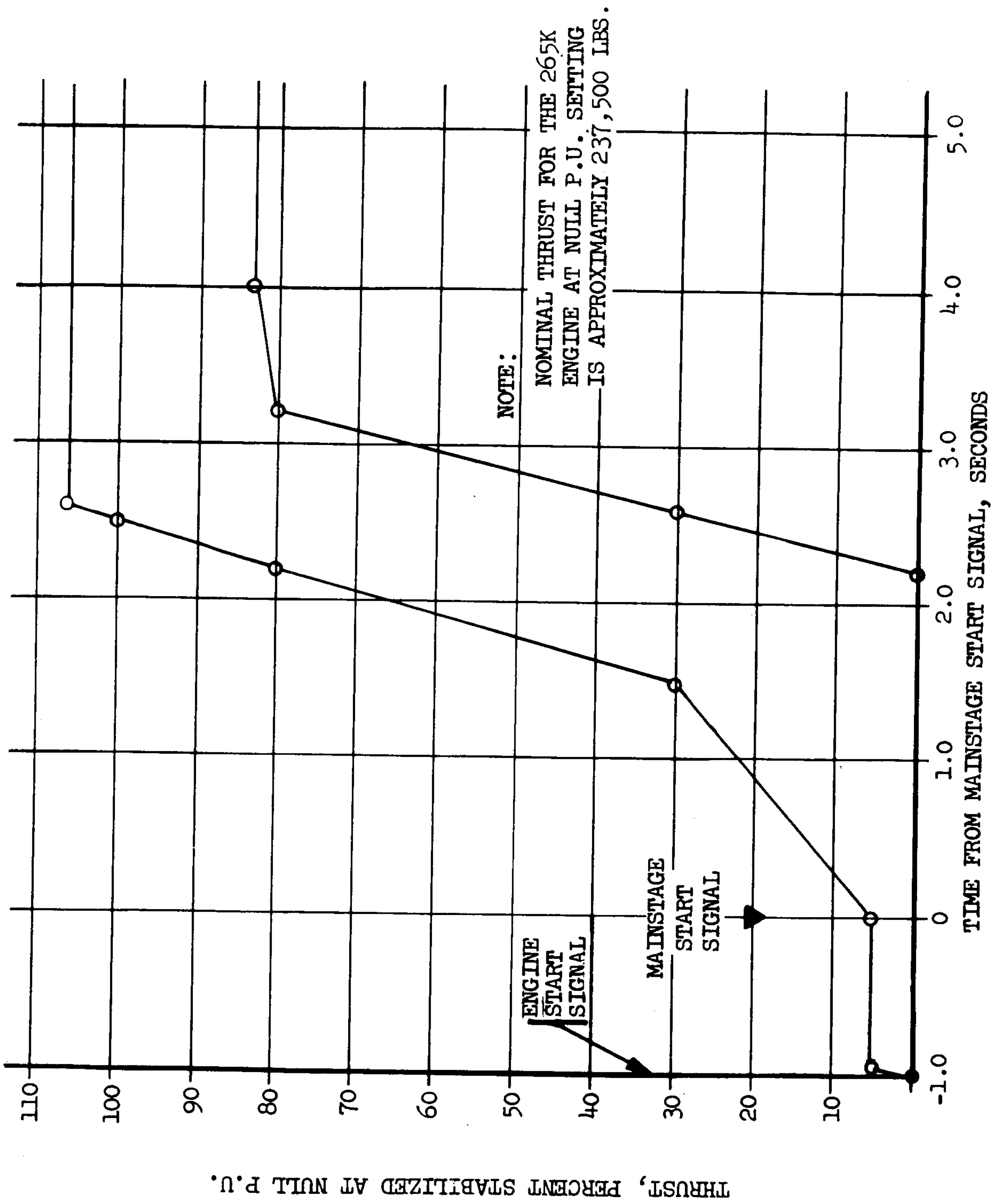


Figure 10.2-26. Proposed J-2S Thrust Buildup Profile (Rocketdyne R-7211)

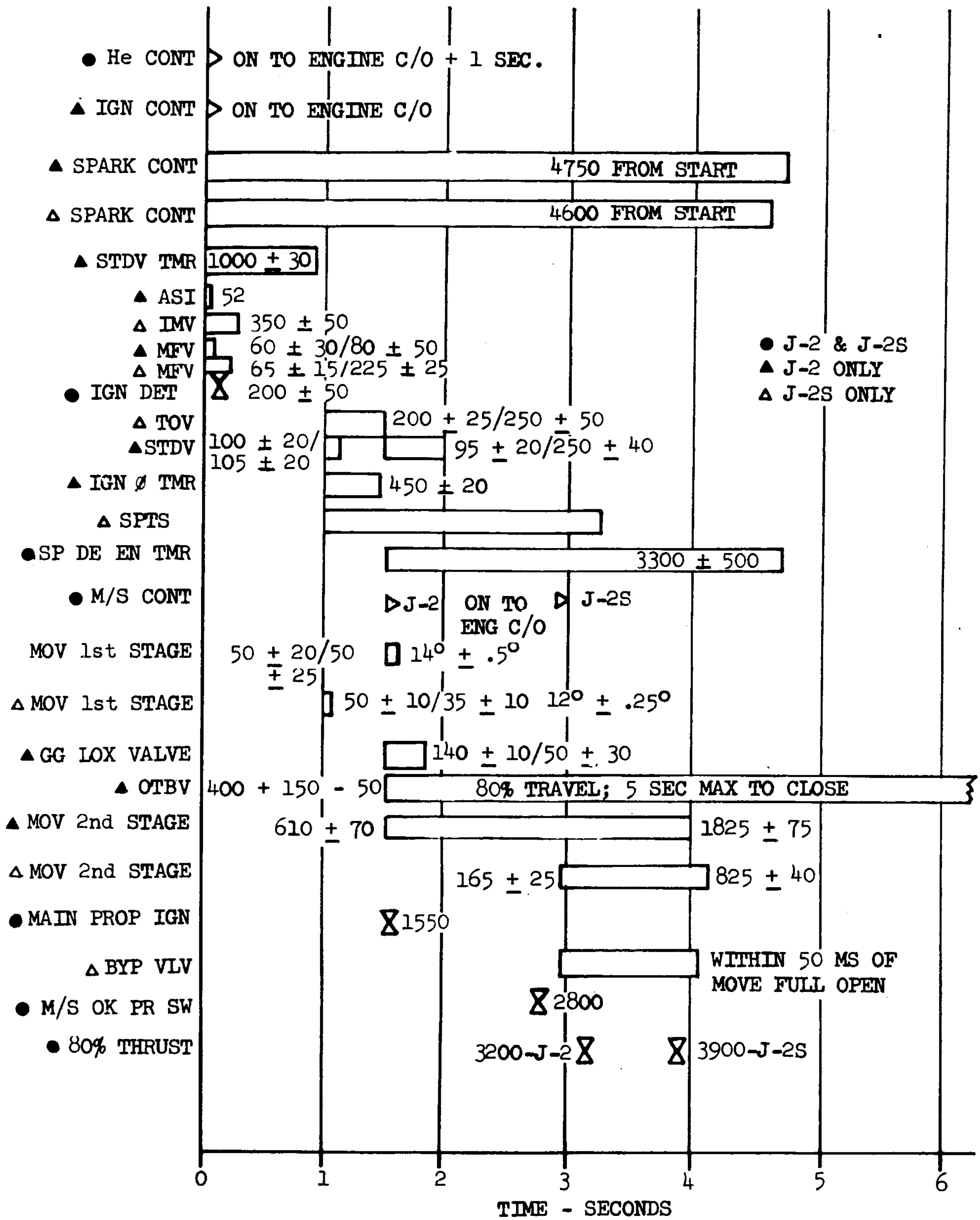


Figure 10.2-27. J-2S and J-2 Engine Sequence

engine will operate at this high mixture ratio until the PU computer commands a shift to a lower mixture ratio (approximately 4.65). The effect on the S-II stage of this operational mode is shown as the thrust versus time curve Figure 10.2-28.

The engine cutoff sequence is shown in Figure 10.2-29 with the J-2 cutoff sequence included.

Figure 10.2-30 illustrates the J-2S engine thrust decay with cutoff from mainstage, and provides an estimate of the maximum impulse and thrust decay time characteristics.

A LOX exhaustion cutoff, wherein the all-engine cutoff command is initiated by dropout of the Mainstage OK pressure switches of any of the five engines, is to be employed for the LOR mission. Utilizing data derived from S-II Battleship testing and transposing to expected J-2S conditions, the stage cutoff thrust characteristics are shown in Figure 10.2-31.

b. Engine Start System. The J-2S engine uses a solid propellant turbine starter (SPTS) in place of the present J-2 engine gaseous hydrogen start tank system. Figure 10.2-32 identifies the location of the SPTS on an engine. As illustrated, the starter is located on the fuel pump side of the engine. There is ample clearance between the starter and the stage structure. The minimum clearance is between the starter and other engine components.

The SPTS ready for installation on the engine weighs approximately 50 pounds, and will not require special or auxiliary Space Division handling equipment. Operation of the SPTS is controlled completely within the J-2S engine control circuitry. The SPTS will be installed just prior to static firing at MTF and concurrently with other ordnance equipment at KSC.

Checkout of the SPTS prior to static firing or launch will consist of resistance check of ignition detector links, electrical checkout of the exploding bridge wire (EBW) initiators (two required per SPTS) prior to installation, and checkout of the EBW firing units by use of pulse sensors (to be performed as part of the engine sequence test).

Resistance and electrical checkout will require electrical checkout equipment similar to that currently being used at KSC for existing S-II stage ordnance equipment. Checkout of the EBW units will require drag-on cables for operation of the pulse sensors. The pulse sensors (two per engine) will be supplied as part of the engine checkout equipment. The checkout consists of the actual firing of EBW's into the pulse sensors during engine sequence tests with proper operation indicated by the pick-up of the installed test pressure switches. The SPTS drag-on cables and pulse sensors will be removed at the same time as the existing S-II stage ordnance cables and pulse sensors are removed.

Use of the SPTS causes a new requirement for ordnance storage facilities and checkout equipment at MTF, and the attendant training of personnel in ordnance handling.

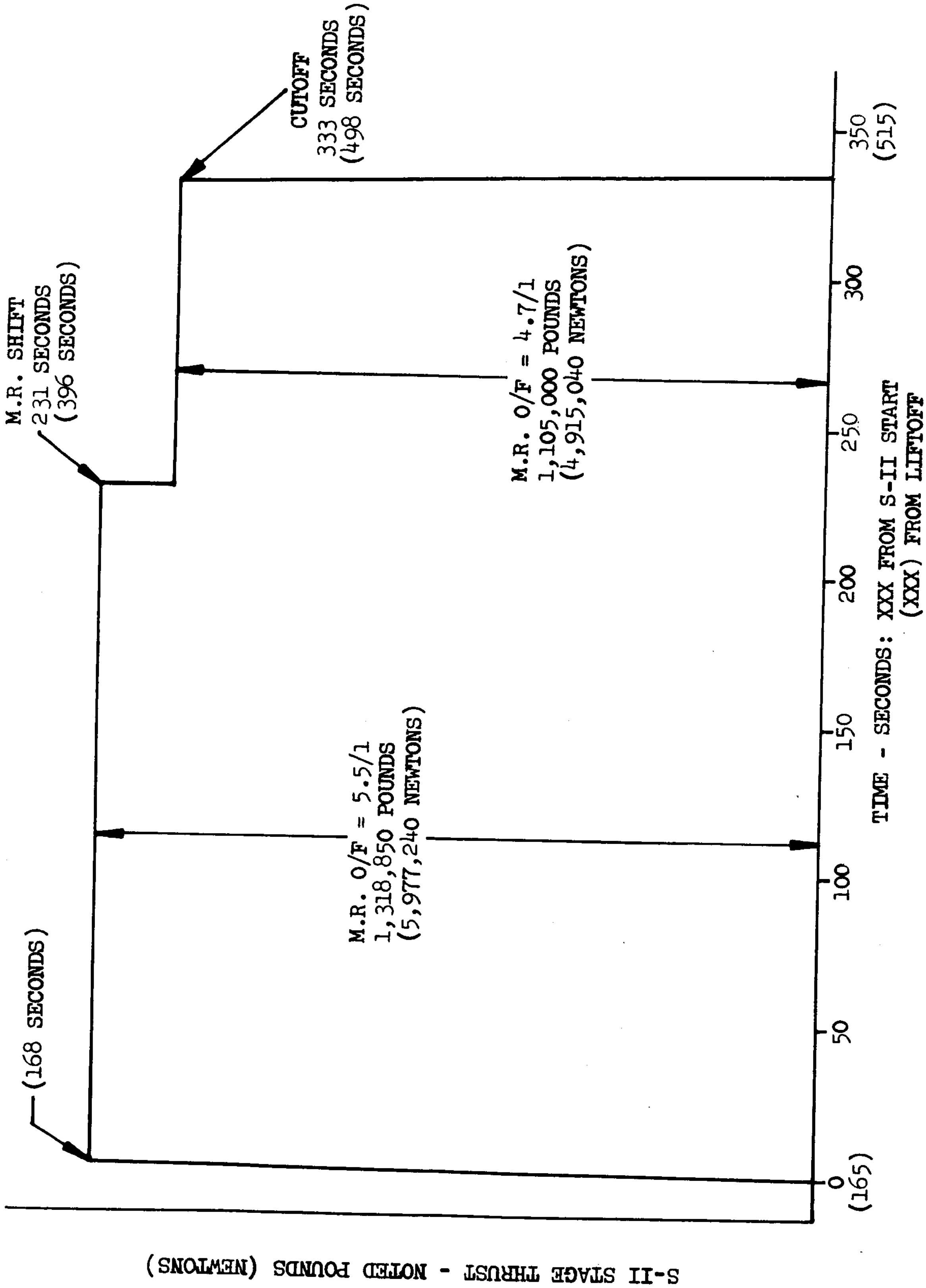


Figure 10.2-28. Thrust Versus Time Profile, LOR Mission (S-II Stage With 265,000-Pound Thrust Engines)

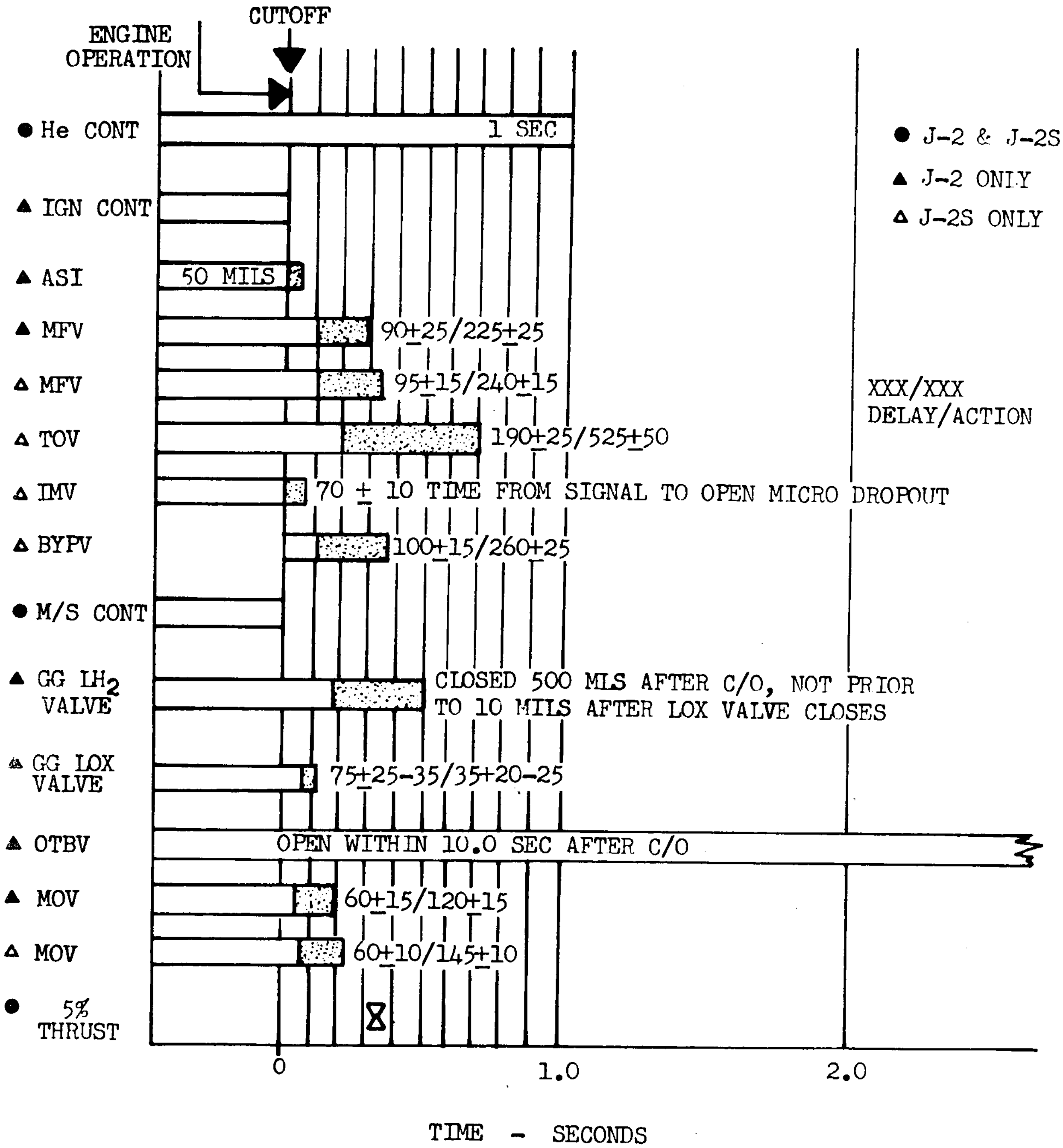


Figure 10.2-29. J-2S and J-2 Engine Checkout Schematic

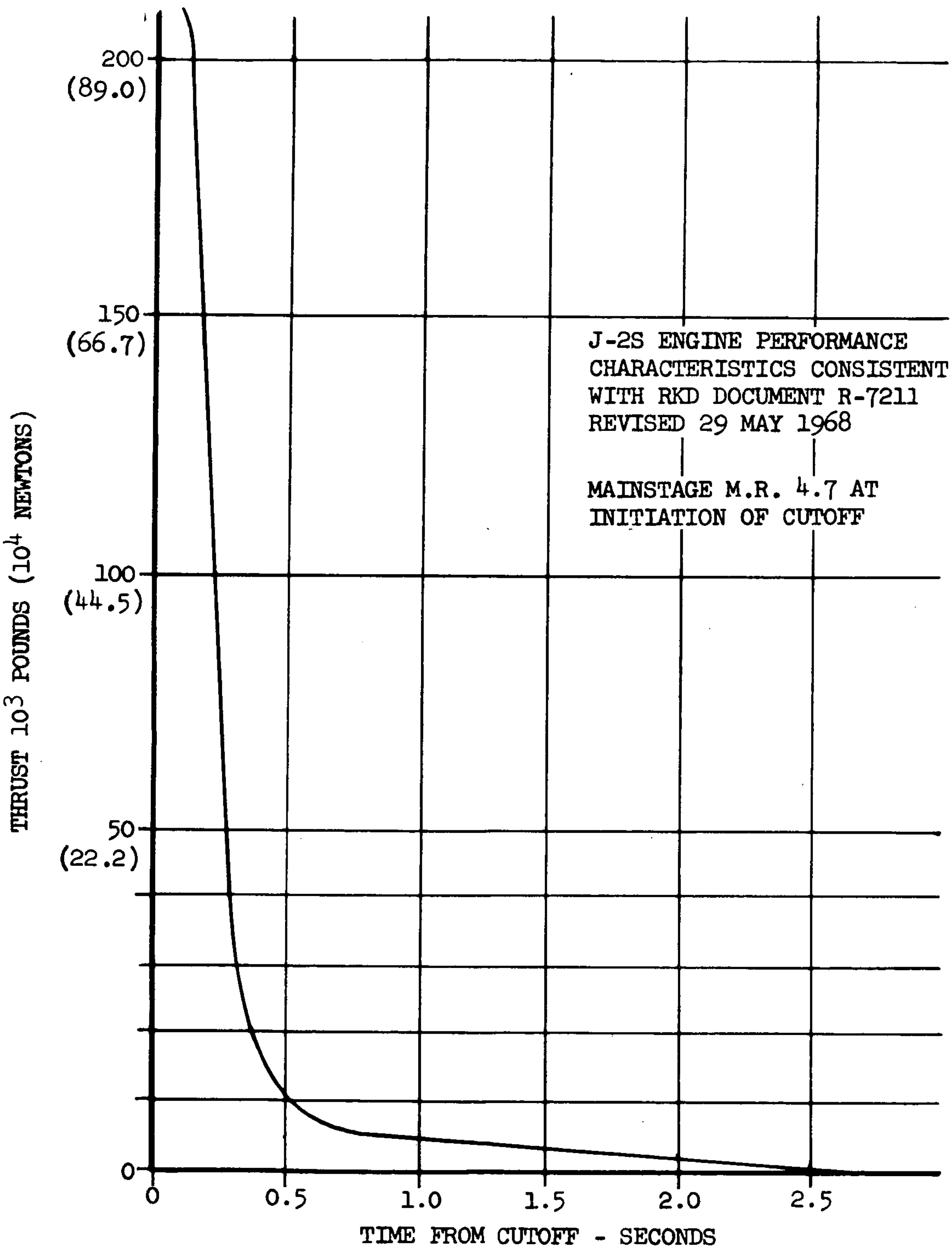


Figure 10.2-30. Estimated J-2S Thrust Decay Characteristics

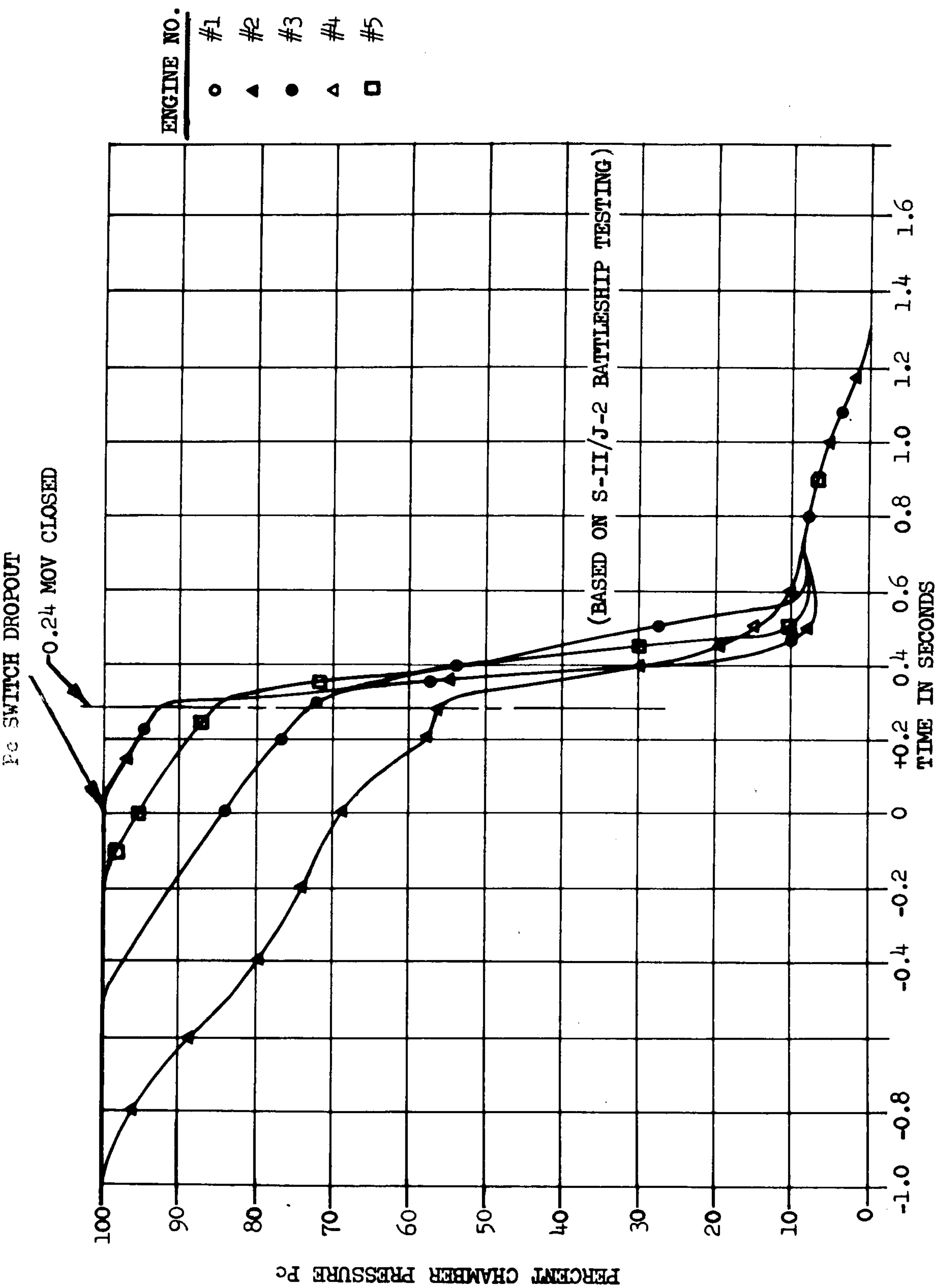


Figure 10.2-31. Predicted S-II Engines Chamber Pressure From LOX Exhaustion (P_c) Cutoff

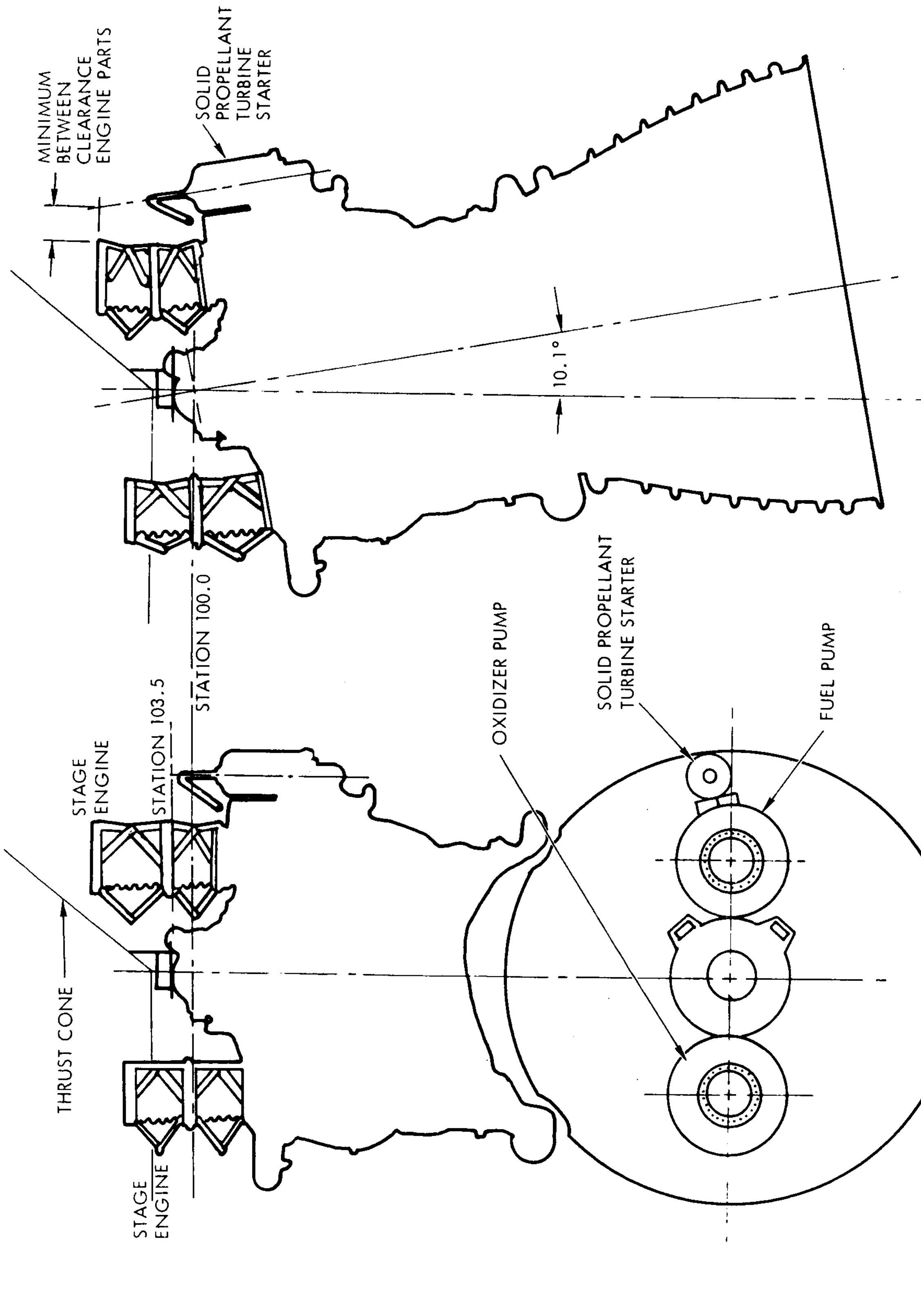


Figure 10.2-32. Solid Propellant Turbine Starter Location on J-2S Engine

c. Propellant Inlet Conditioning.

1. Functional Changes - The starting characteristics of the J-2S engine permit deletion of the propellant Recirculation Systems. The propellant inlet pressure and temperature requirements for J-2S engine starting are:

Pressure

LOX 33 psia minimum
LH_2 27 psia minimum
(Figure 10.2-33)

Temperature

LOX -100 F to -300 F
LH_2 -100 F to -425 F
(Figure 10.2-34)

These minimum pressure requirements are unchanged from the J-2 application. The temperature requirements, however, have been relaxed from subcooled liquid conditions at the engine inlets for the J-2 engine to liquid conditions at the tank outlets for the J-2S engines. Additional requirements are that propellants shall be supplied to the engines for a minimum of one hour prior to start command, and that the feed ducts of the S-II adhere to the following:

LOX Duct

Maximum Length	95 inches
Maximum Volume	(21 gallons)
Maximum Heat Input	3500 Btu/hr ($\approx$1 Btu/sec)

LH_2 Duct

Maximum Length	390 inches
Maximum Volume	(85 gallons)
Maximum Heat Input	18,000 Btu/hr ($\approx$5 Btu/sec)

These specific lengths and volumes permit use of the current S-II feed ducts. The heat input through the present vacuum jacketed lines has been determined from test data evaluation to range from 2.5 to 4.5 Btu/sec on the LH_2 ducts, well within the maximum requirement. The LOX ducts will require additional insulation of the LOX feed duct adapters, spools, and prevalves. With the added insulation, analysis indicates the heat leak of the LOX ducts will be approximately 0.46 Btu/sec.

2. System Changes - The deletion of the fuel and LOX recirculation system is one of the major advantages of incorporating the J-2S engine on the S-II stage. The reduction in stage complexity can be seen by referring to Figures 10.2-35, -36, -37 and -38. With the deletion of the recirculation system, there would be no stage hardware requirement for engine preconditioning and the following items would be deleted:

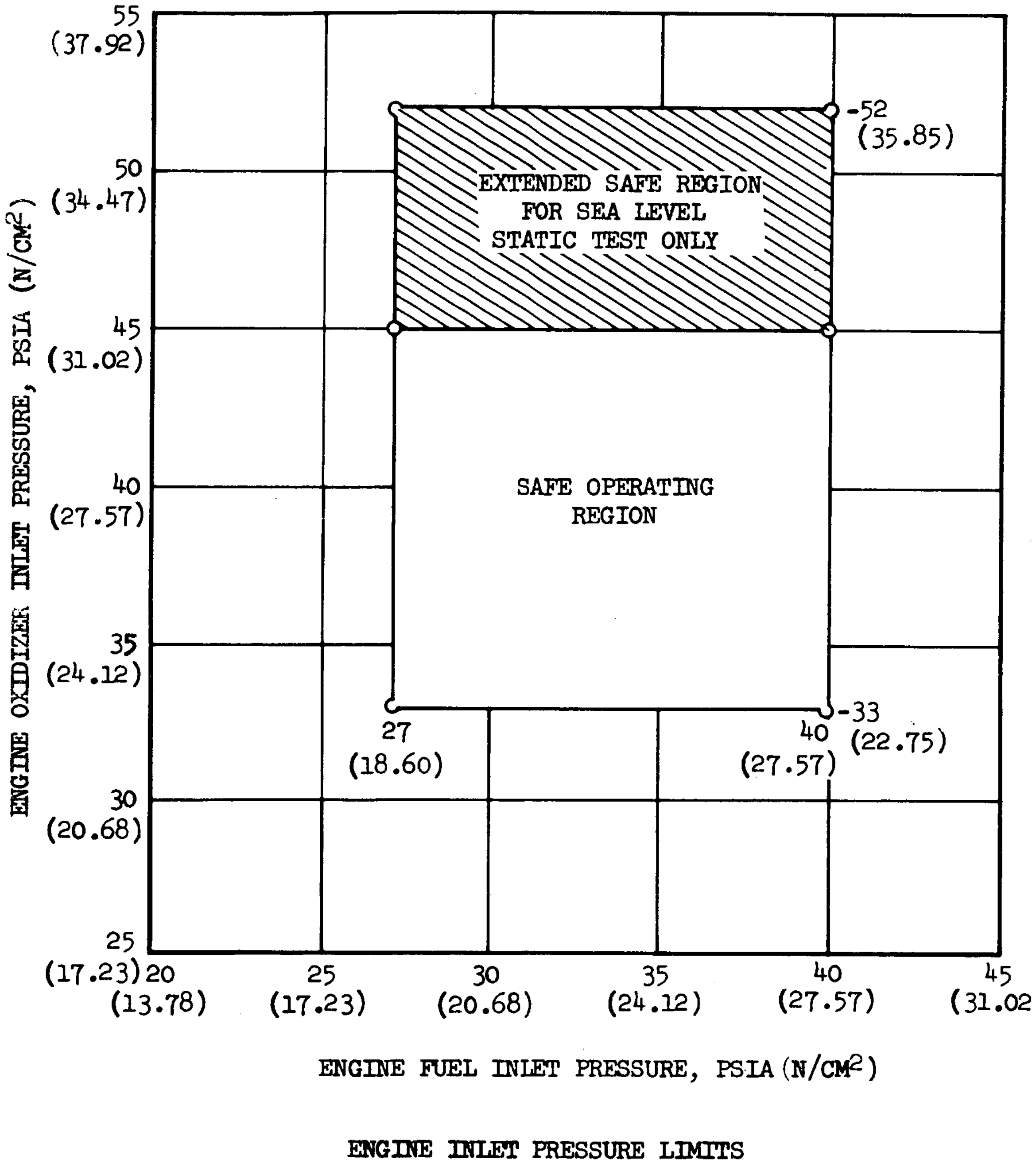
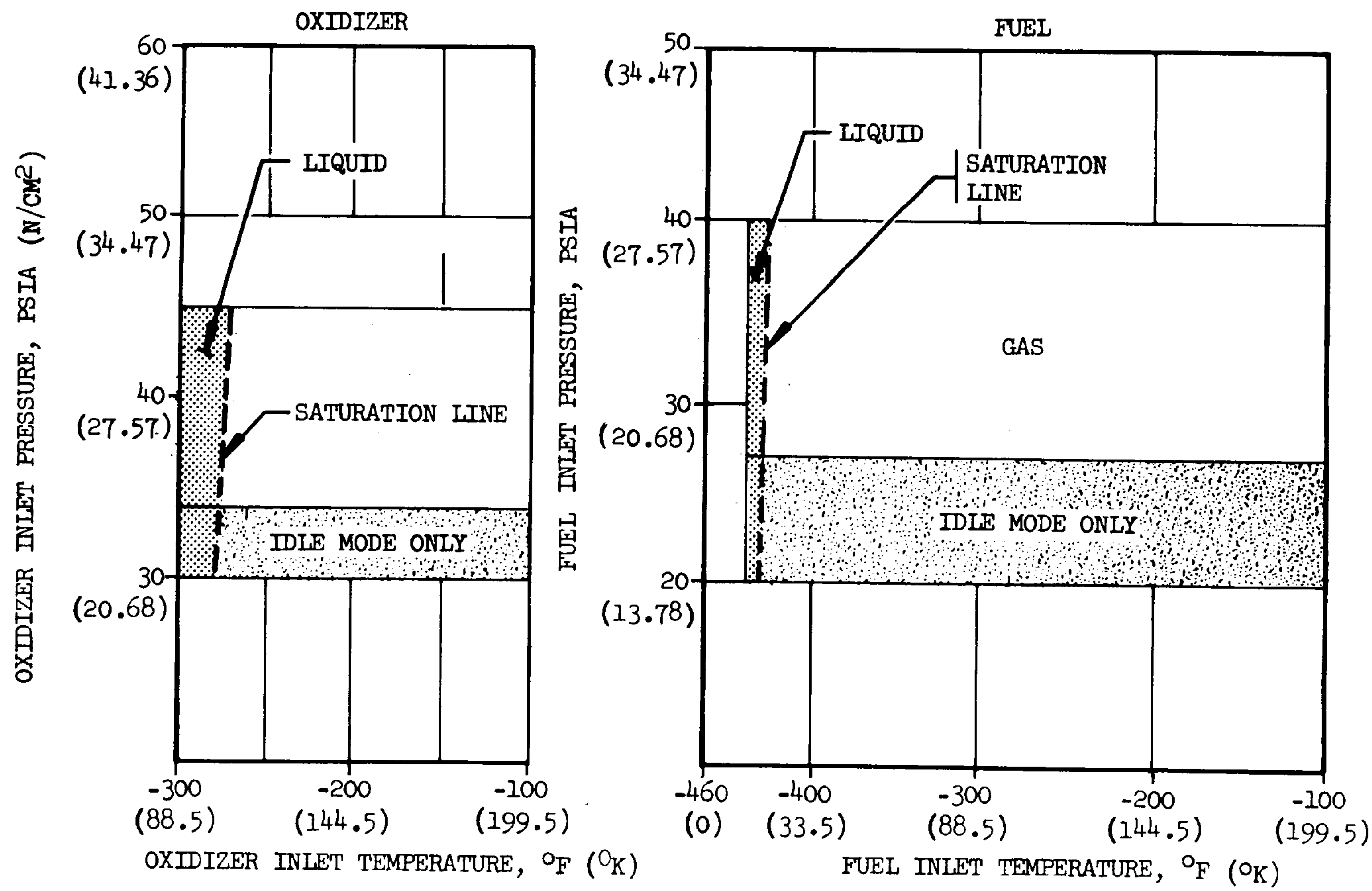


Figure 10.2-33. Mainstage Start Requirements, Engine Inlet Pressure Limits



NOTE: AT IDLE MODE START, LIQUID PROPELLANTS ARE AT TANK OUTLET

Figure 10.2-34. Pressure and Temperature Requirements for J-2S Engine Start

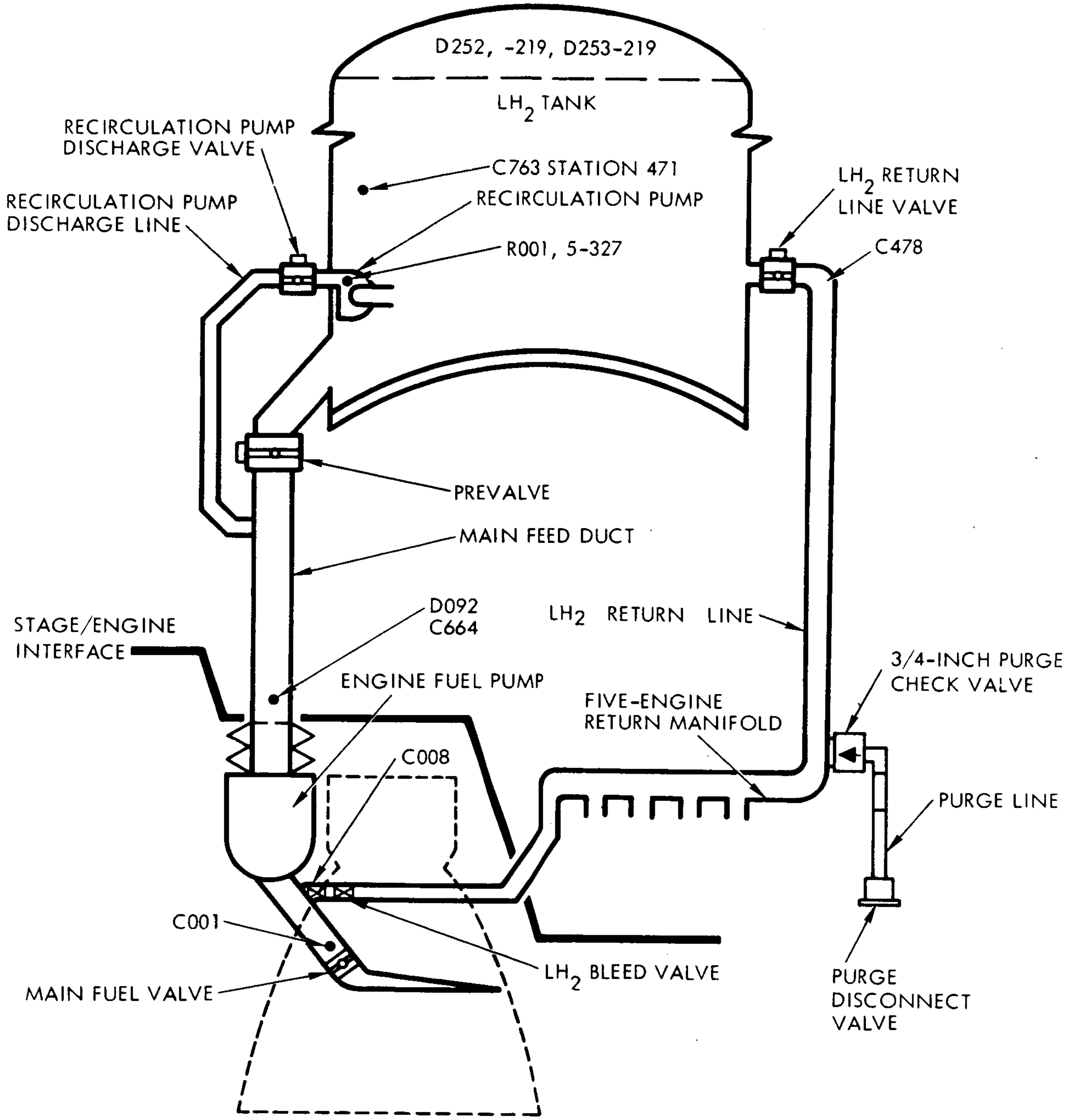


Figure 10.2-35. J-2 LH_2 Recirculation System Schematic

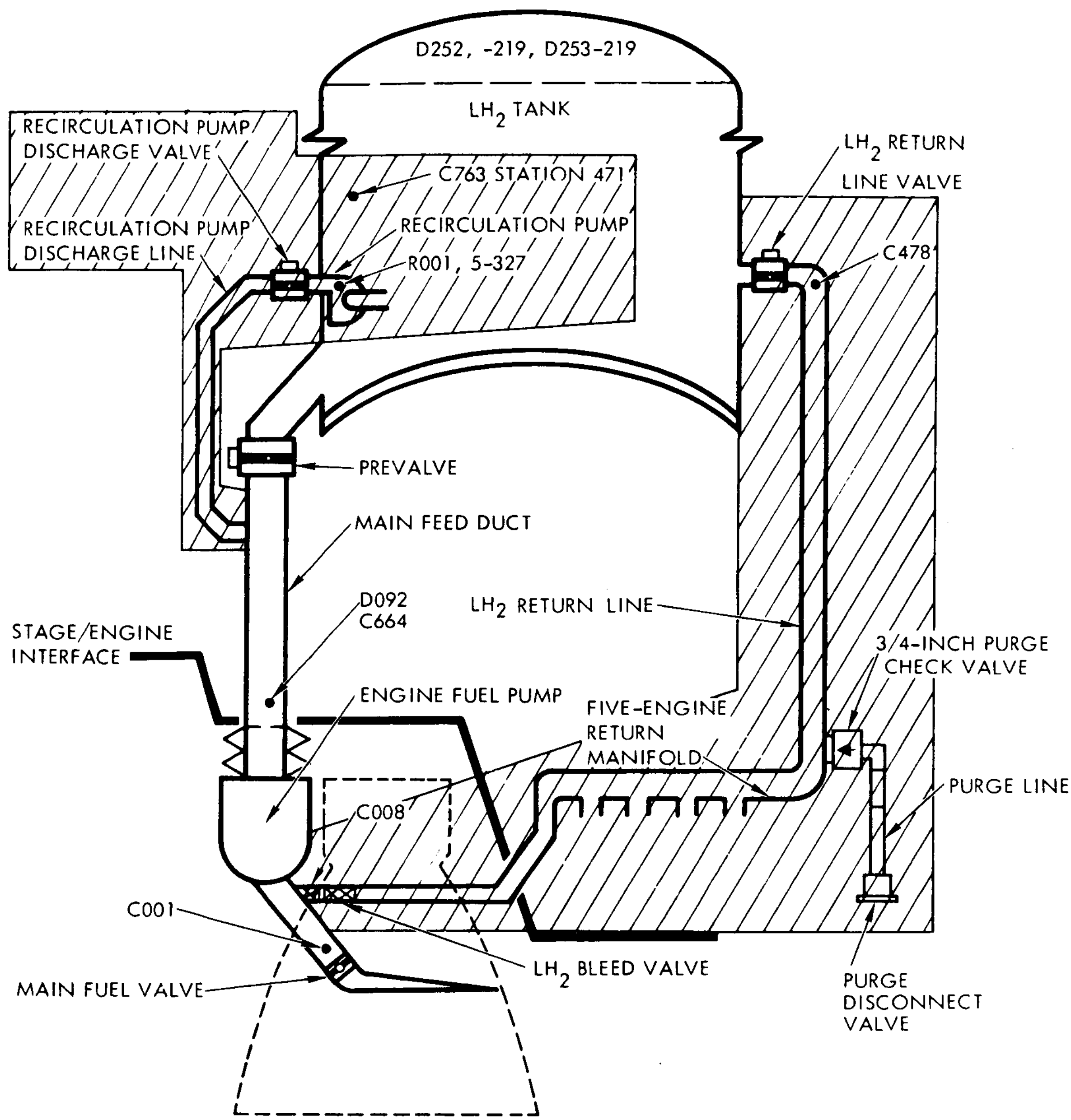


Figure 10.2-36. J-2S LH_2 Recirculation System Schematic

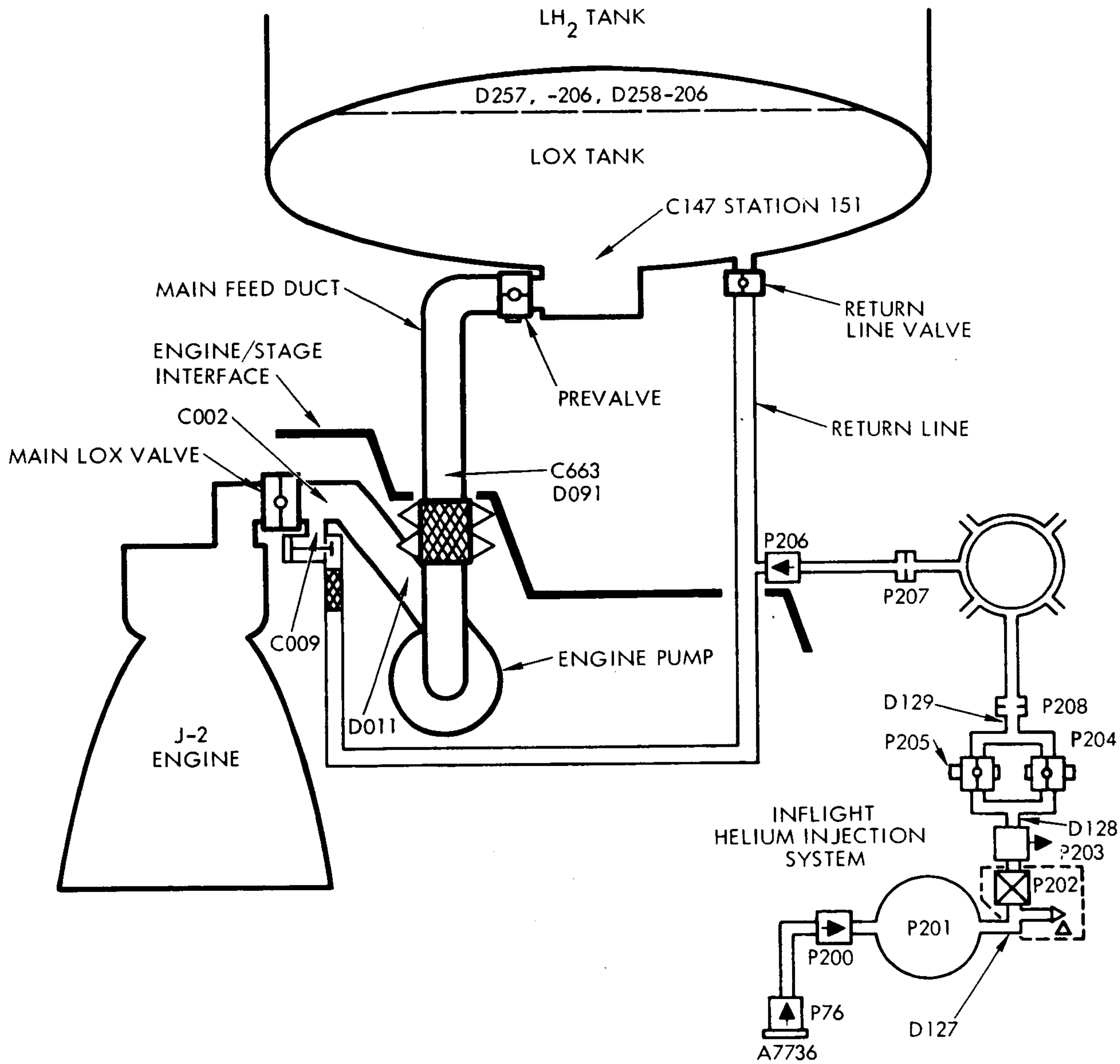


Figure 10.2-37. J-2 LOX Recirculation System Schematic

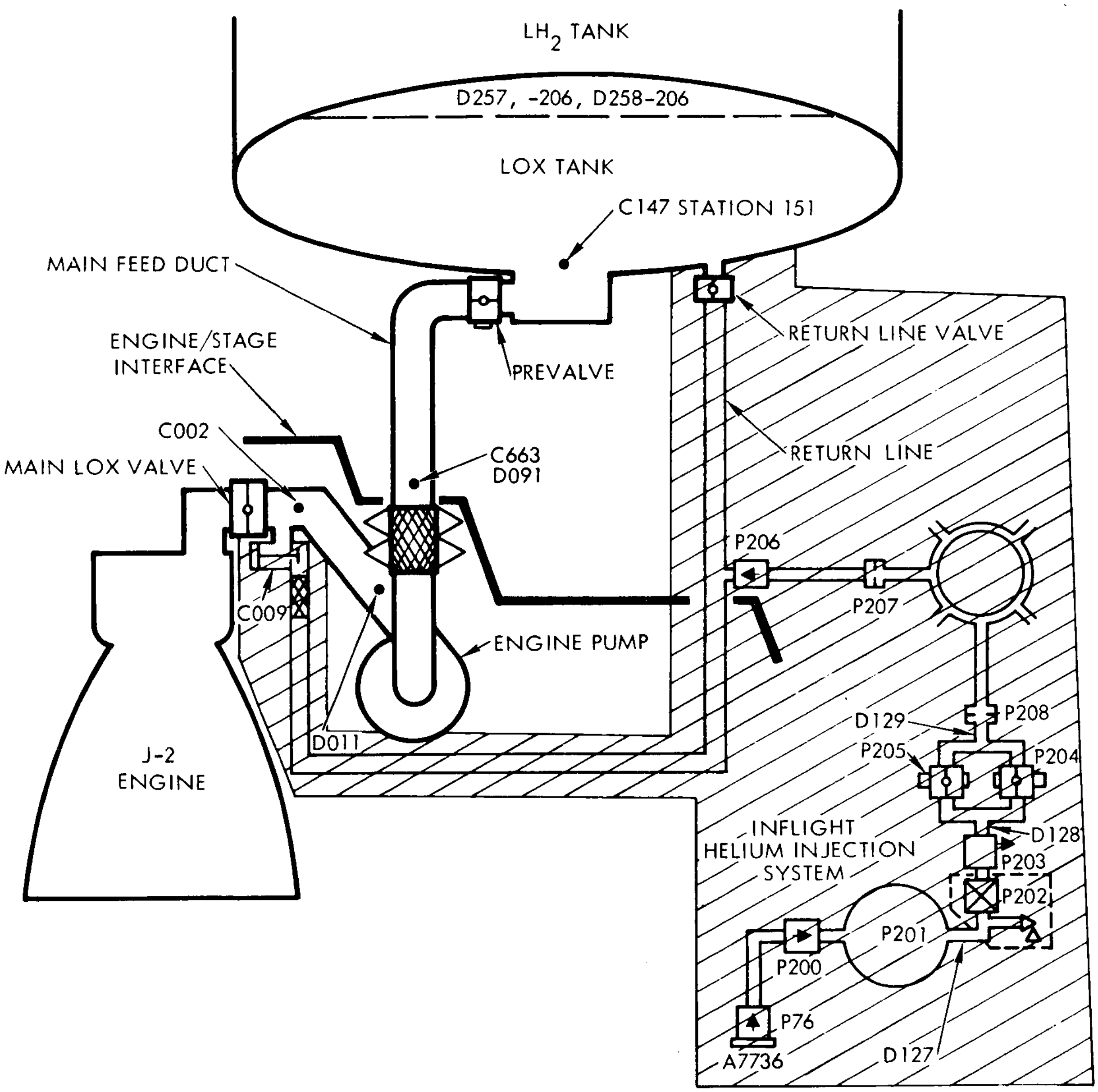


Figure 10. 2-38. J-2S LOX Recirculation System Schematic

(a) Five recirculation pumps and associated electrical power and control equipment

(b) Five recirculation pump discharge valves and associated pneumatic valve actuation and electrical control equipment

(c) Vacuum jacketed recirculation pump discharge lines

(d) Vacuum jacketed LH_2 return lines and return line manifold

(e) Uninsulated LOX recirculation return lines

(f) Six LOX and LH_2 return line valves and associated pneumatic valve actuation and control equipment

(g) Helium injection subsystem

The existing single-point LH_2 feed and recirculation purge system would be replaced by a manifold system connected to each feed duct below the prevalve (utilizing existing pump discharge line ports) refer to Propellant Feed System, Section 10.2.2.2.

3. Verification - The principal configuration change requiring major stage test verification is the elimination of the engine propellant conditioning (recirculation) requirement. The testing which has been completed by the engine contractor has partially verified the capability of the J-2S engine to start satisfactorily within the specified maximum and minimum limits. This has included satisfactory starts made with saturated conditions at the LH_2 engine inlet. Engine starts with saturated conditions at the LOX engine inlet have not yet been completed. Satisfactory completion of the stage evaluation of engine start characteristics following deletion of the recirculation system requires test verification of two primary areas; the ability of the J-2S engine to clear any saturated fluid from the feed duct which may be generated by loss of head at S-IC cutoff (this clearing must be accomplished in connection with the use of a one-second idle mode premainstage operation), and the nature of the thermodynamic processes experienced by the fluid in the feed duct during this time. The possibility of additional gas generating in the feed duct and engine pump inlets as the result of a pressure drop associated with propellant flow during the start transient must also be considered. Ground tests to date have not simulated the S-II stage feed duct configuration or flight simulated effects. Both analytical simulation and test data evaluation are also difficult because of the complexity of extrapolation of the fluid dynamics involved to near zero load factors, and because of the rapid pressure transients experienced during actual flight conditions.

The following is a description of the thermodynamic mechanism expected in flight, at the time when the acceleration head is reduced to nearly zero as a result of cutoff of the S-IC engines. The potential problem described will occur only if the propellant in the S-II feed ducts reached saturation as a result of this loss in head. When the fluid pressure drops, the temperature will remain nearly constant until the pressure is reduced to the saturation value corresponding to the initial temperature of the fluid. At constant entropy, temperature in the subcooled or liquid phase is insensitive to pressure changes of the magnitude involved as shown in Figure 10.2-39. When saturation pressure is reached, gas will start to form and significant temperature changes will be experienced as expansion at constant entropy occurs. However, pressure can drop only at the rate volume is provided for the gas to form as indicated by the quality lines on the temperature-entropy diagram. The gas "bubbles" will be at the same pressure as the liquid forming their boundaries (saturation) and liquid must be displaced to provide the volume needed for further expansion to lower pressures. The rate at which the expansion will proceed to the final saturation conditions as established by the ullage pressure will be finite and will depend on the pressure (or temperature) at which saturation is first reached and upon fluid flow characteristics of the ducts (flow versus Δ pressure). Of course, as the pressure drops, more gas is formed and so on until the final saturation condition is reached.

The quality characteristics as indicated on the "T-S" diagrams are percent by weight functions; the fluids will be contained in the ducts and will flow through the engine turbopumps and propellant feed system primarily as a volumetric function. Therefore, Figures 10.2-40 and 10.2-41 quality versus percent gas by volume for saturated oxygen at 33 psia, and saturated hydrogen at 27 psia, respectively, have been prepared to facilitate convenient appraisal of this relationship.

The predicted engine inlet pressures for LOX and LH_2 are presented in Figures 10.2-42 and 10.2-43, respectively. It is anticipated that maximum simulation of these pressure profiles will be accomplished during design verification testing.

d. Engine Servicing System. The engine servicing system transfers the required fluids at the proper conditions of pressure, temperature, and flow rate from the umbilical connections to the appropriate engine system. It also provides ducting to vent overboard various seals and drains from the engines. The J-2S engine is shown schematically in Figure 10.2-44 with the stage/engine interfaces for engine servicing identified. Figures 10.2-45 and 10.2-46 show the routing of the engine servicing systems used with the J-2 and J-2S engines respectively. The J-2S engine servicing subsystems are the same as used with the J-2 engine except as noted below.

1. Start Tank/Fuel Pump Drain. The start tank system is not required, as the J-2S engine utilizes a solid propellant turbine starter (SPTS) in place of the gaseous hydrogen system of the J-2 engine. Based upon this engine change, the start tank fill and the start tank vent control subsystems will be deleted from the stage. The start tank vent and relief subsystem will be retained to function as, and be identified as, the fuel pump drain system. The stage/engine interface for this identification remains physically the same.

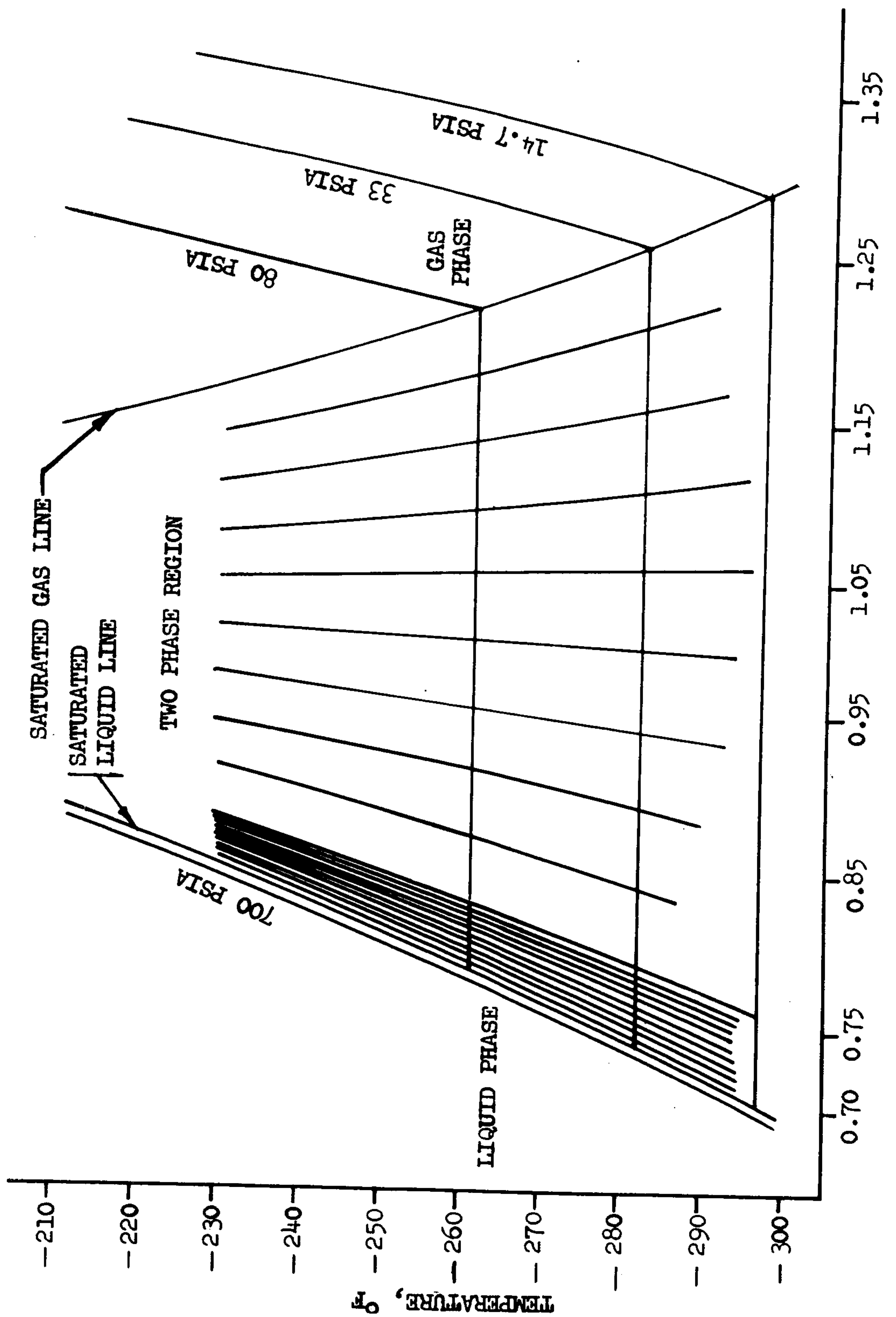


Figure 10.2-39. Oxygen Temperature-Entropy Diagram

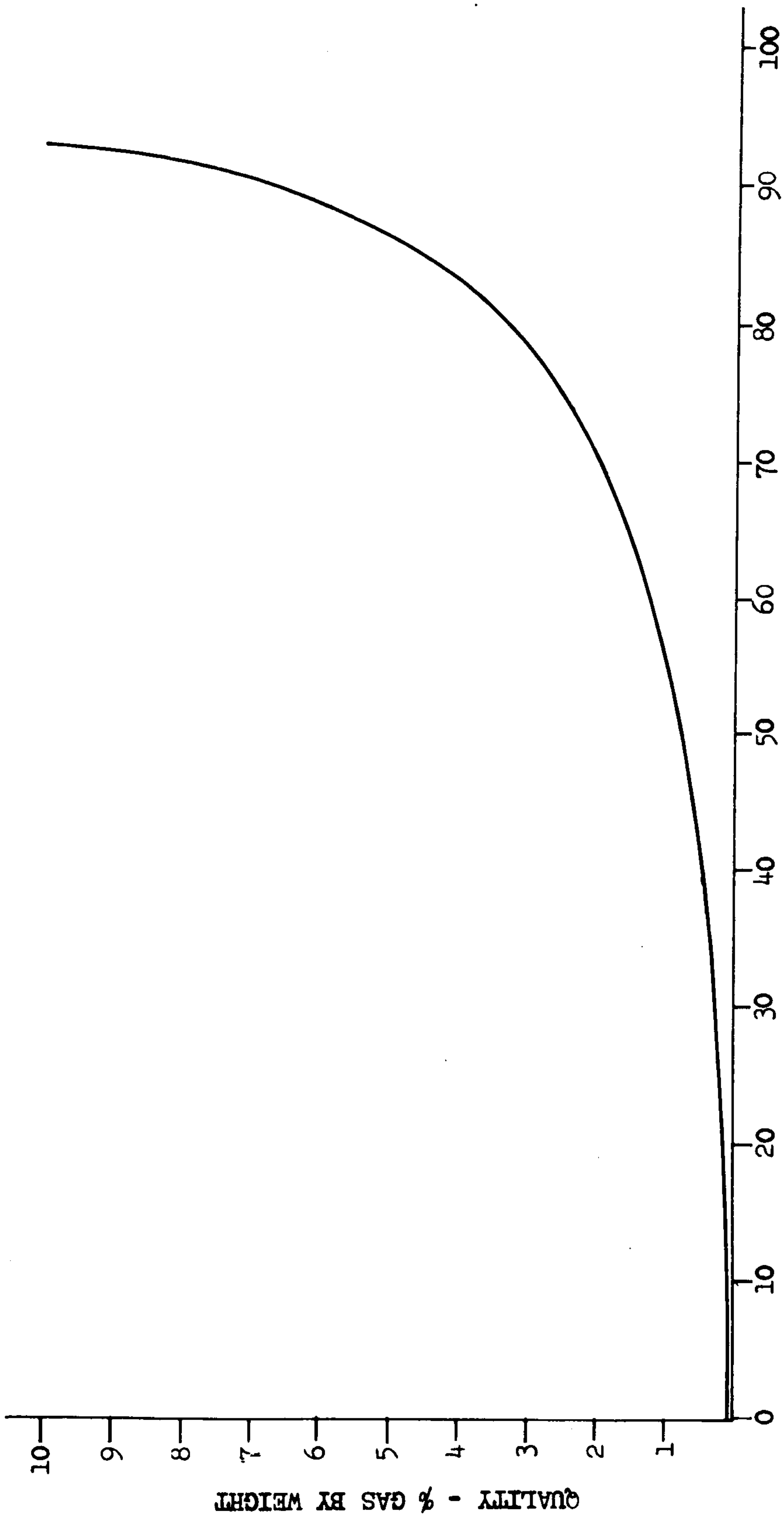


Figure 10.2-40. Quality Versus Percent Gas by Volume, Saturated Oxygen at 33 psia

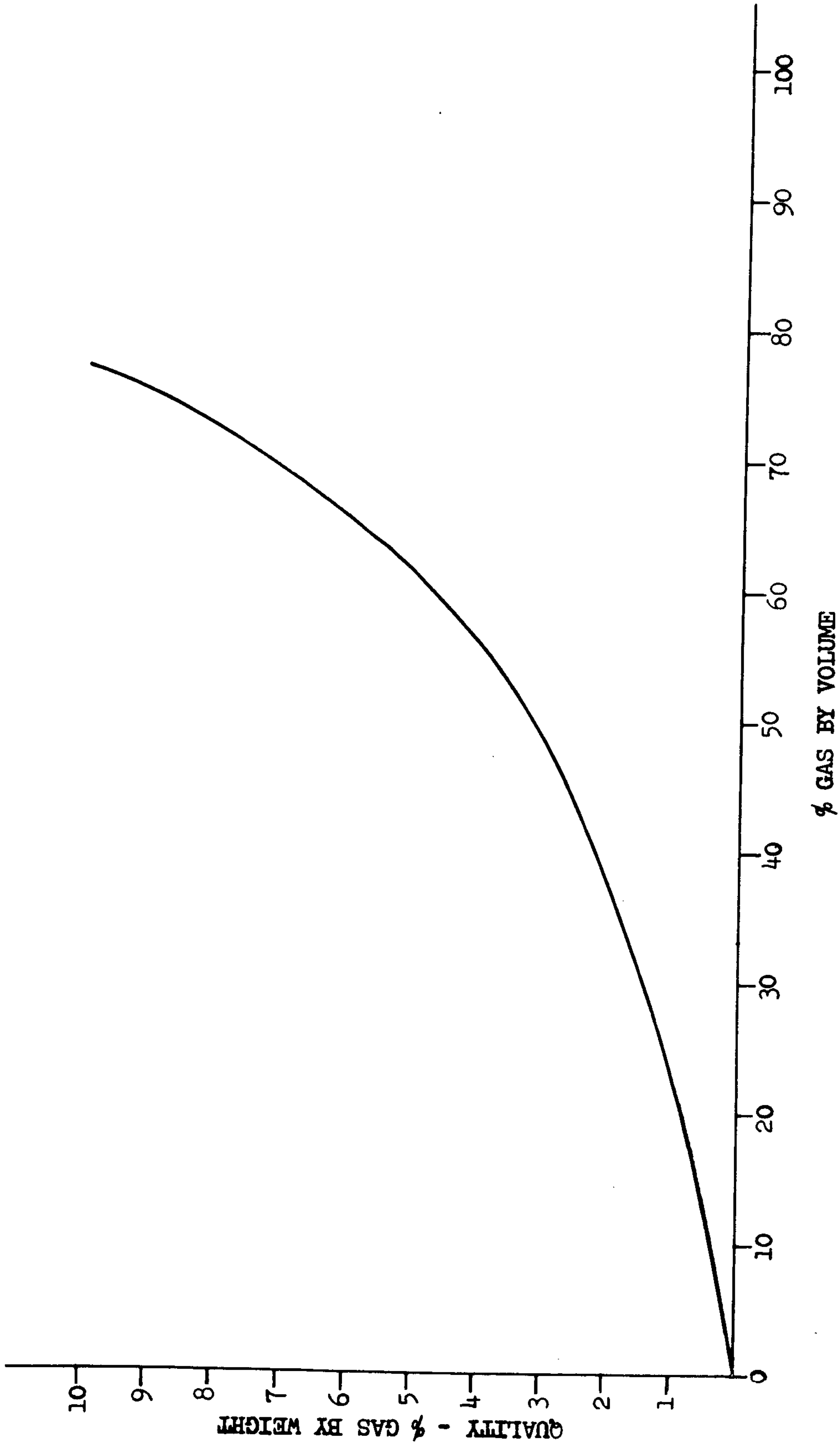


Figure 10. 2-41. Quality Versus Percent Gas by Volume, Saturated Hydrogen at 27 psia

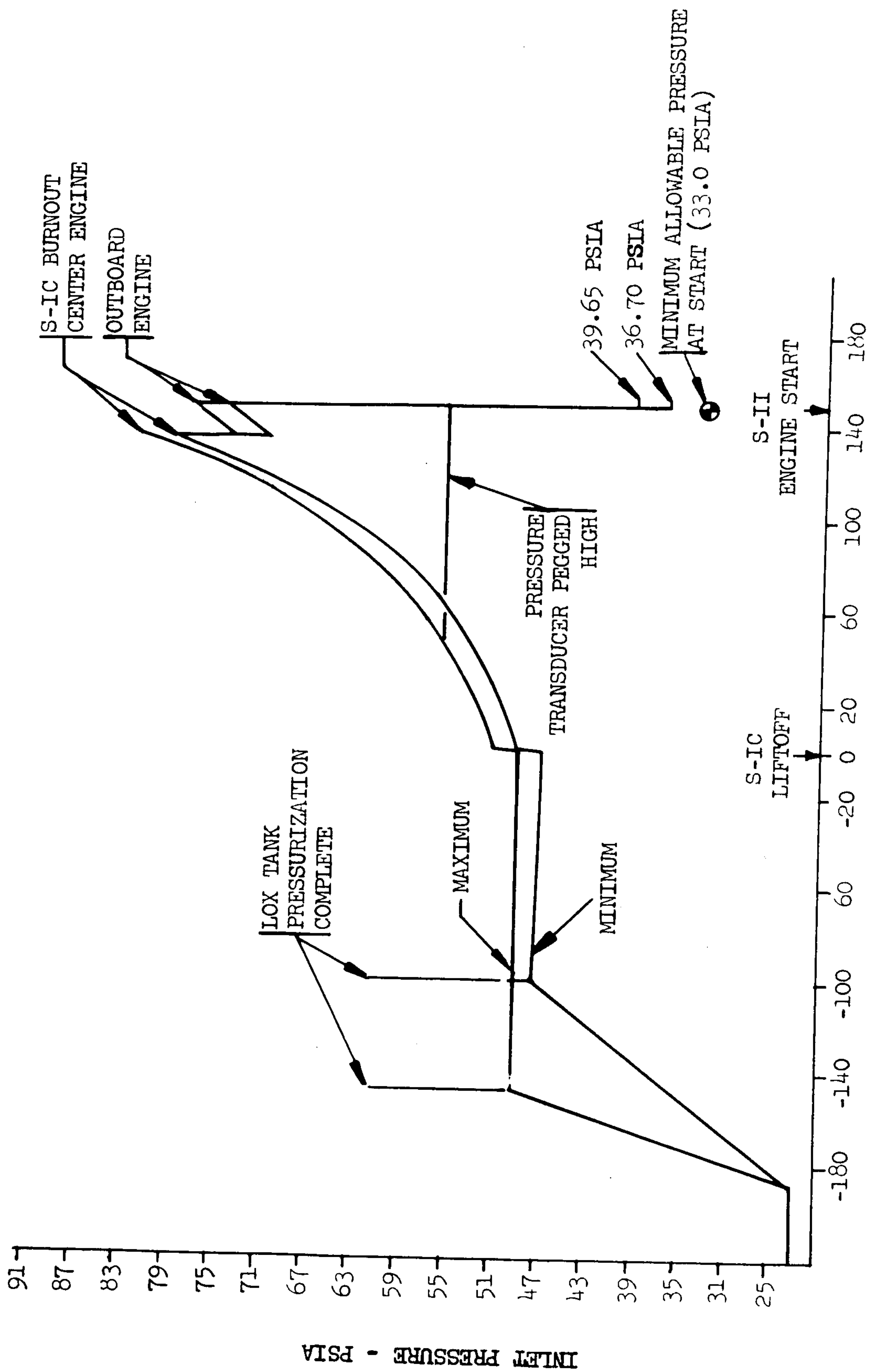


Figure 10.2-42. J-2S/S-II Predicted LOX Engine Inlet Pressure

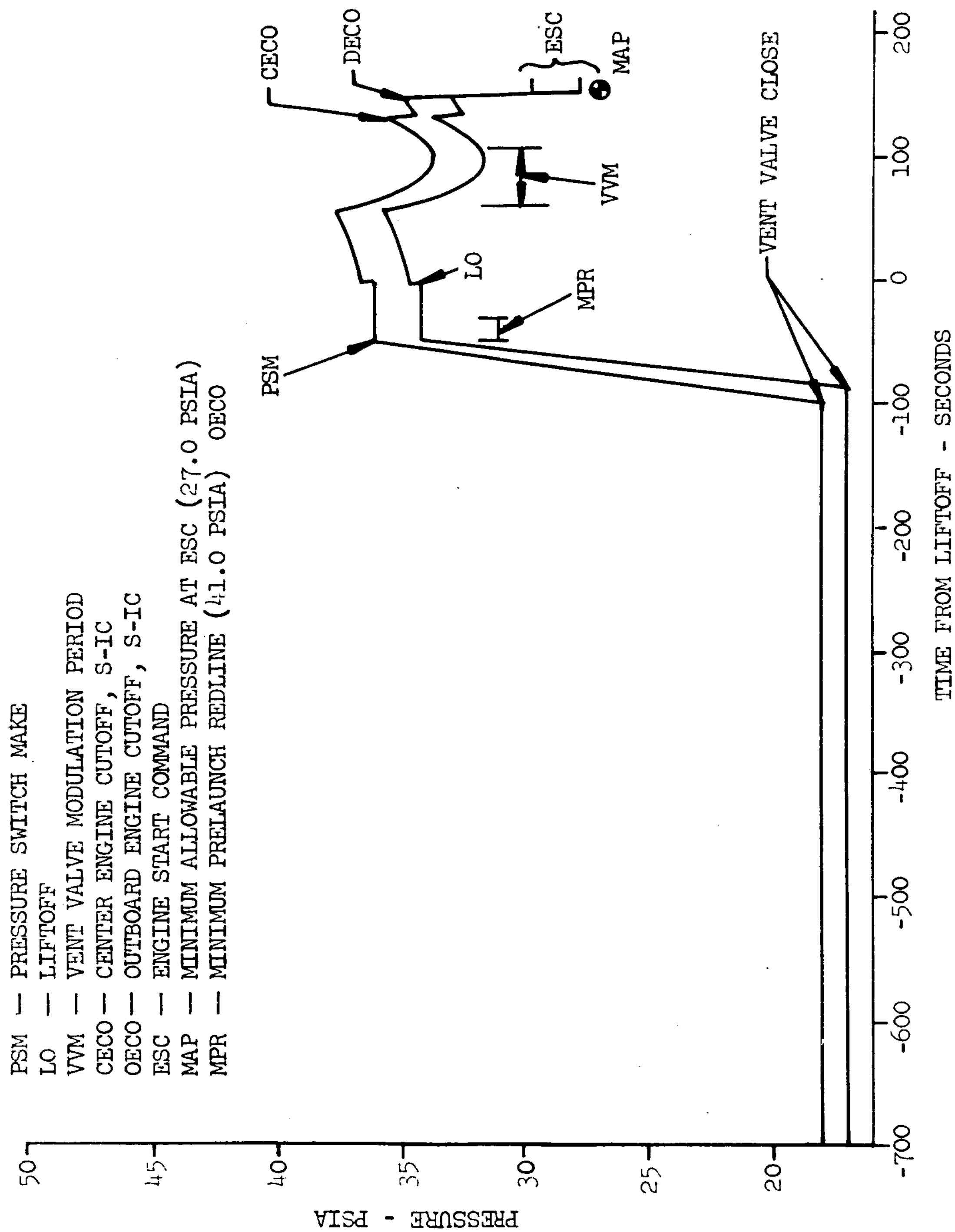


Figure 10.2-43. J-2S/S-II Predicted LH_2 Engine Inlet Pressure Band

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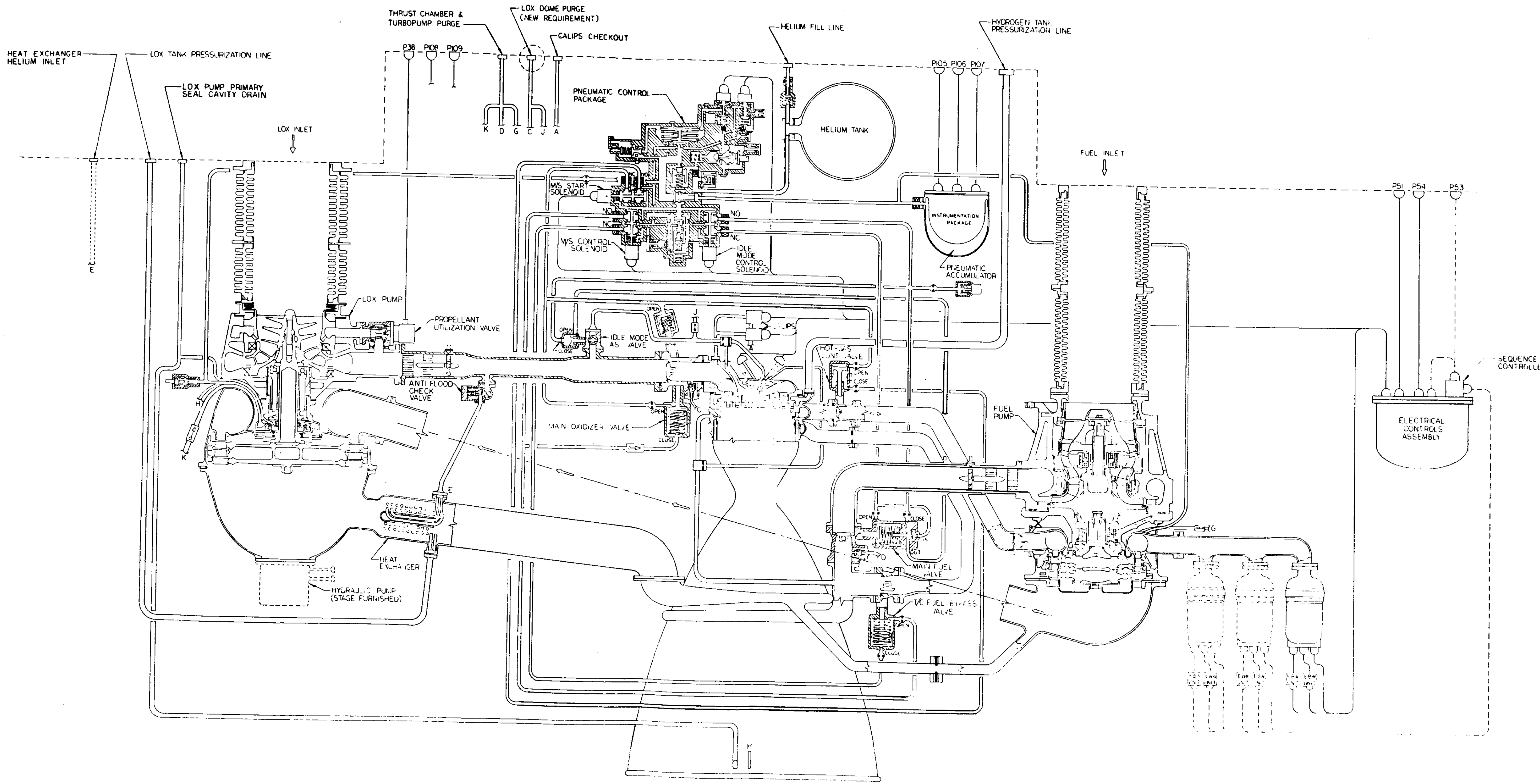


Figure 10. 2-44. J-2S Engine Schematic

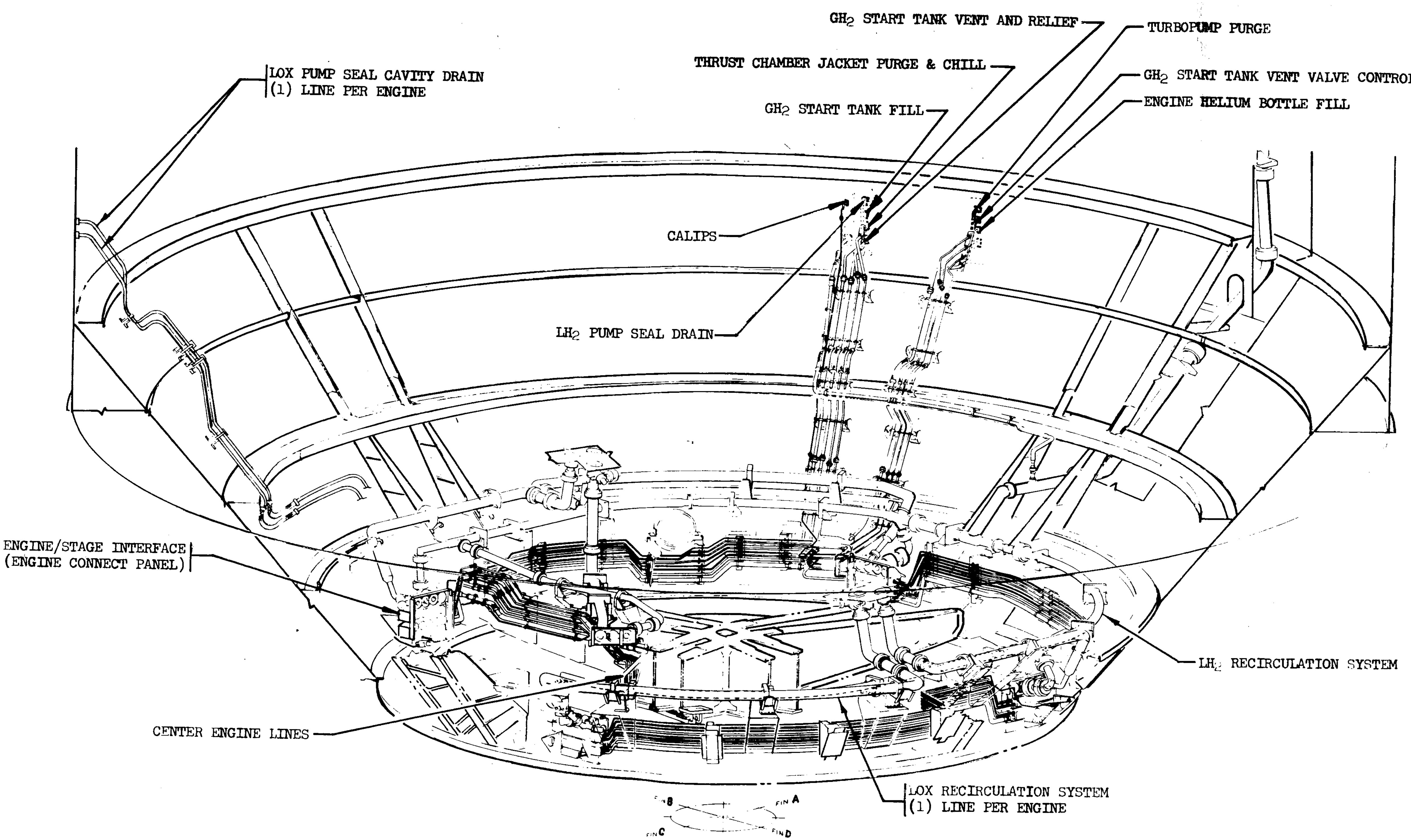


Figure 10.2-45. Installation of J-2 Engine Service Lines

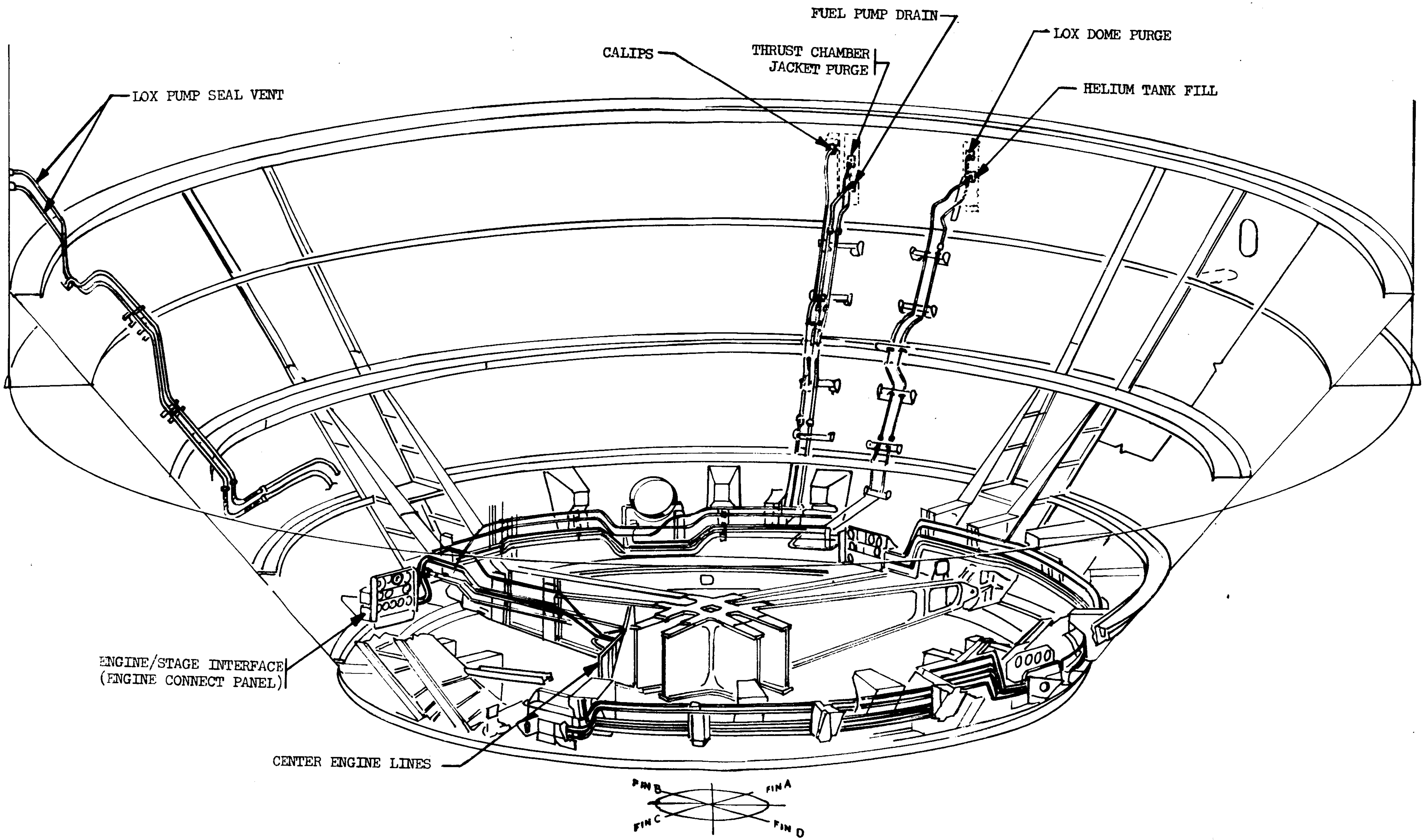


Figure 10.2-46. Installation of J-2S Engine Service Lines

2. Thrust Chamber Purge and Chill/Thrust Chamber-Turbopump Purge. The J-2S engine does not require a prestart thrust chamber chill as does the J-2 engine, but the purge requirements are retained. The purge capability of this system will also meet the purge requirements of the turbopump. The routing of lines to perform purging of the thrust chamber and turbopumps is accomplished upon the engine; therefore, the only change involved on the stage is reidentification of the interfaces. A comparison of J-2S and J-2 thrust chamber purge requirements is presented in Table 10.2-IV.

3. Turbopump Purge/LOX Dome Purge. The turbopump purge on the J-2S engine is included as a part of the thrust chamber purge, thereby eliminating a system on the stage; however, a new LOX dome purge requirement is added. Therefore, the old turbopump purge system will be reidentified and used as the LOX dome purge system. The LOX dome purge requirements are listed in Table 10.2-V.

4. Mainstage OK Pressure Switch. This system will be retained as is for the J-2S implementation. This system permits remote checkout of these pressure switches. An increase in operational checkout pressure from 500 psig to 700 psig will be required.

5. LOX Pump Seal Drain. This system will be retained as is for the J-2S provided the maximum flow, when established, does not exceed the J-2 design flow. The drain system is designed to provide individual overboard venting of gaseous oxygen from each engine LOX pump seal cavity at a maximum flow rate of 36,000 scim per engine.

Figure 10.2-47 is a typical cross-section through the engine servicing systems used with the J-2S engine. A comparison can be made between this figure and the J-2 engine lines shown in Figure 10.2-48, as to which systems are retained, deleted, or reidentified. The support brackets for the systems shown in Figure 10.2-48 will be the same as those used with the J-2 engine lines.

Figure 10.2-49 illustrates typical segments of tubing used within the thrust chamber purge system with the J-2 and J-2S engines. The line routing of this system is the same with both engines, but the Teflon insulation used on the J-2 engine will not be required with the J-2S engine.

Figure 10.2-50 illustrates the connect panel to be used at the outboard engine positions for the J-2S engine. This is physically the same panel used with the J-2 engine but positions 1, 3, 5, 8 and 9 will not be used. Holes for ports 5, 8 and 9 will be omitted. Position 6, turbopump purge, will be reidentified, as shown, as LOX dome purge. Position 11, GH_2 start tank vent and relief, is reidentified as fuel pump drain.

Figure 10.2-51 illustrates the routing and installation of the engine systems used with the J-2S center engine. A single sheet metal connect panel will replace the three connect panels used with the J-2 engine. This J-2S center engine interface is the result of replacing engine-provided flex hoses with stage-provided flex hoses of a different configuration. The stage hard lines extending from the

Table 10.2-IV. Thrust Chamber Purge Requirements

Operation	Gas		Stage/Engine Fluid Interface Requirements				Time (minutes)		Flow (scim)		Remarks
			J-2		J-2S						
	J-2	J-2S	Temperature (F)	Pressure (psig)	Temperature (F)	Pressure (psig)	J-2	J-2S	J-2	J-2S	
Static Firing Preloading	N_2	He	60-200	150-200	50-150	125-175	(A)	(B)	100	125	(A) From 1 minute preload to He purge (T-13 min.) (B) From 30 minutes preload to engine start
Static Firing Auto Sequence	He	--	Ambient	40-185	--	--	5	-	95	--	
Static Firing Cutoff	N_2	He	100-200	150-200	50-150	125-175	10	15	100	125	
Launch Preloading	He	He	50-200	40-185	50-150	125-175	15	(C)	95	125	(C) For 30 minutes before propellant loading
Launch Prechill	He	--	Ambient	40-185	--	--	5	-	95	--	

Table 10.2-V. Oxidizer Dome Purge - J-2S

Operation	Gas	Stage/Engine Fluid Interface Requirements		Time	Flow	Remarks
		Temperature (F)	Pressure (psig)	(minutes)	(scim)	
Static Firing Preloading	He	50-150	375-425	(A)	150	(A) for 30 minutes before propellant loading
Static Firing Cut-off	He	50-150	375-425	15	150	
Launch	He	50-150	375-425	(A)	150	

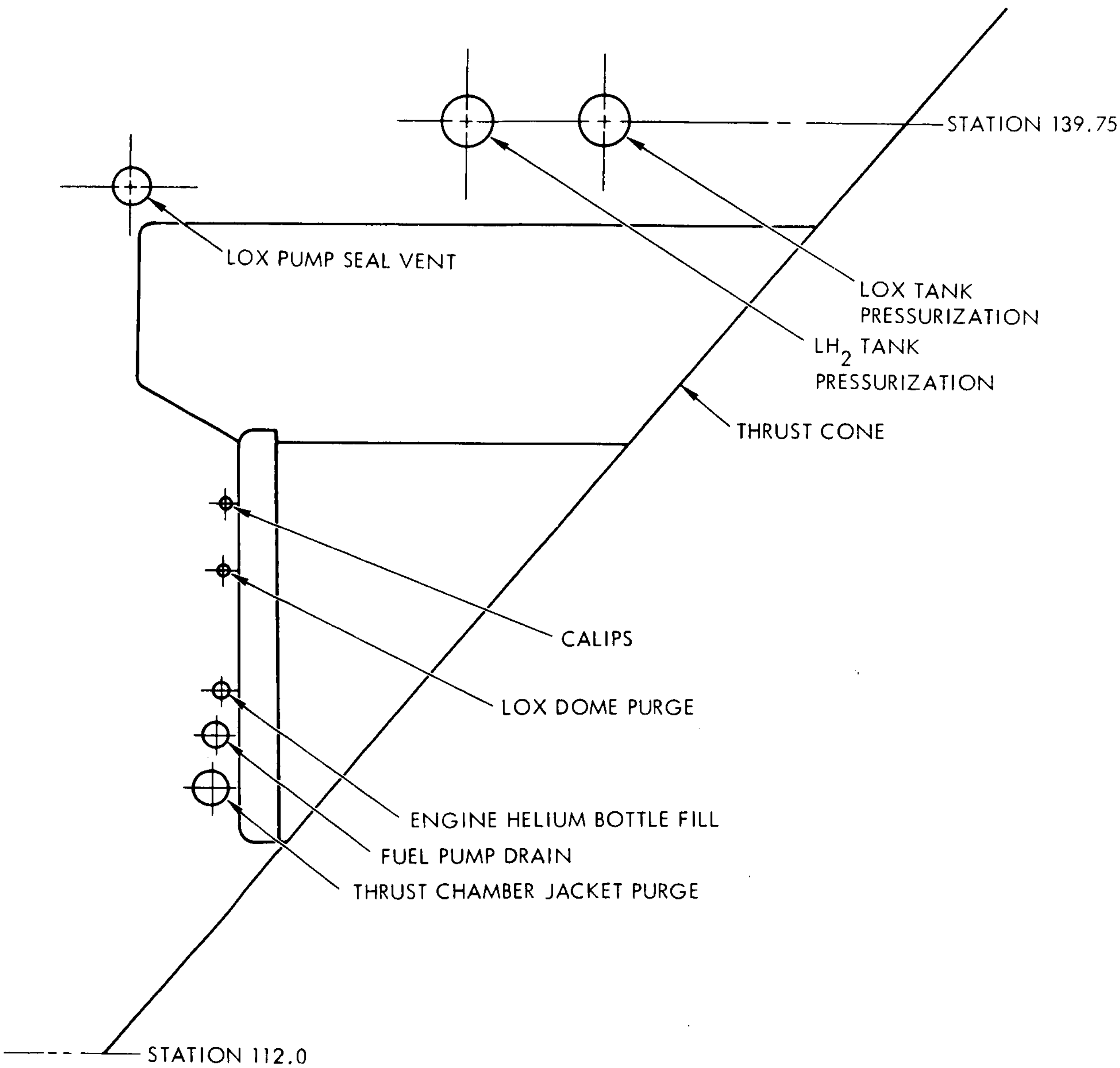


Figure 10.2-47. Installation of J-2S Engine Service Lines (Section B-B)

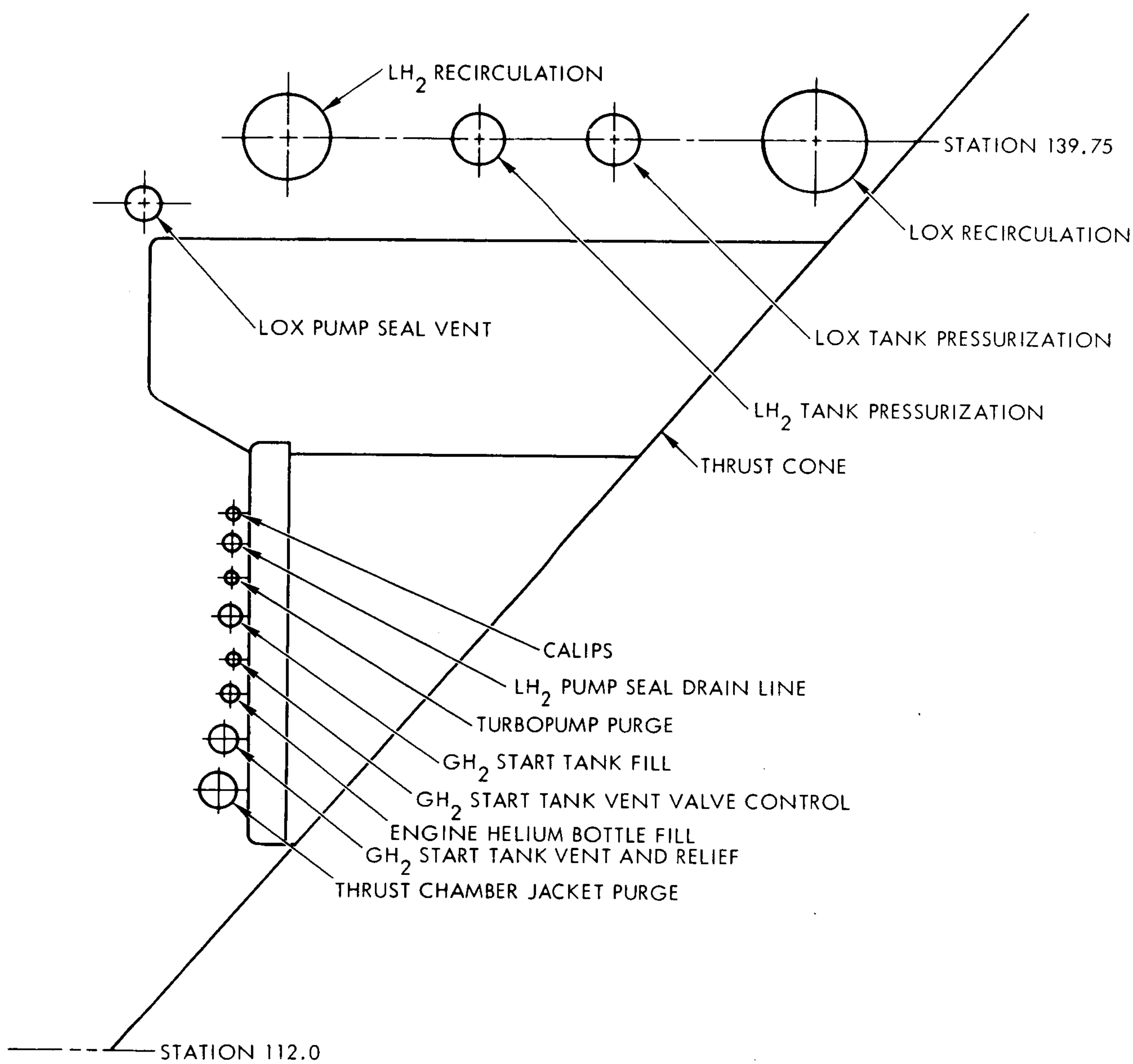


Figure 10.2-48. Installation of J-2 Engine Service Lines (Section A-A)

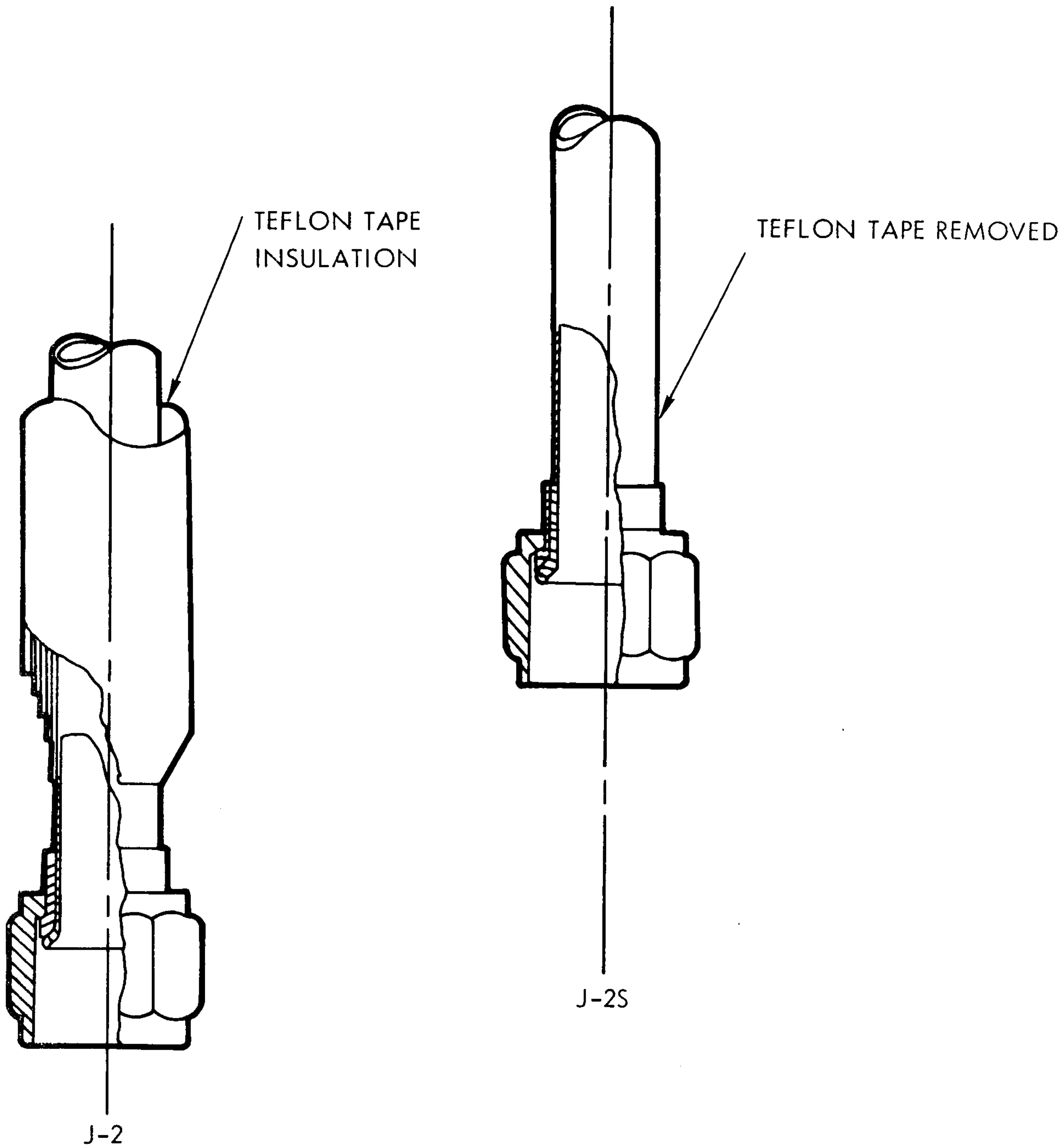


Figure 10. 2-49. Deletion of Thrust Chamber Chill Requirements

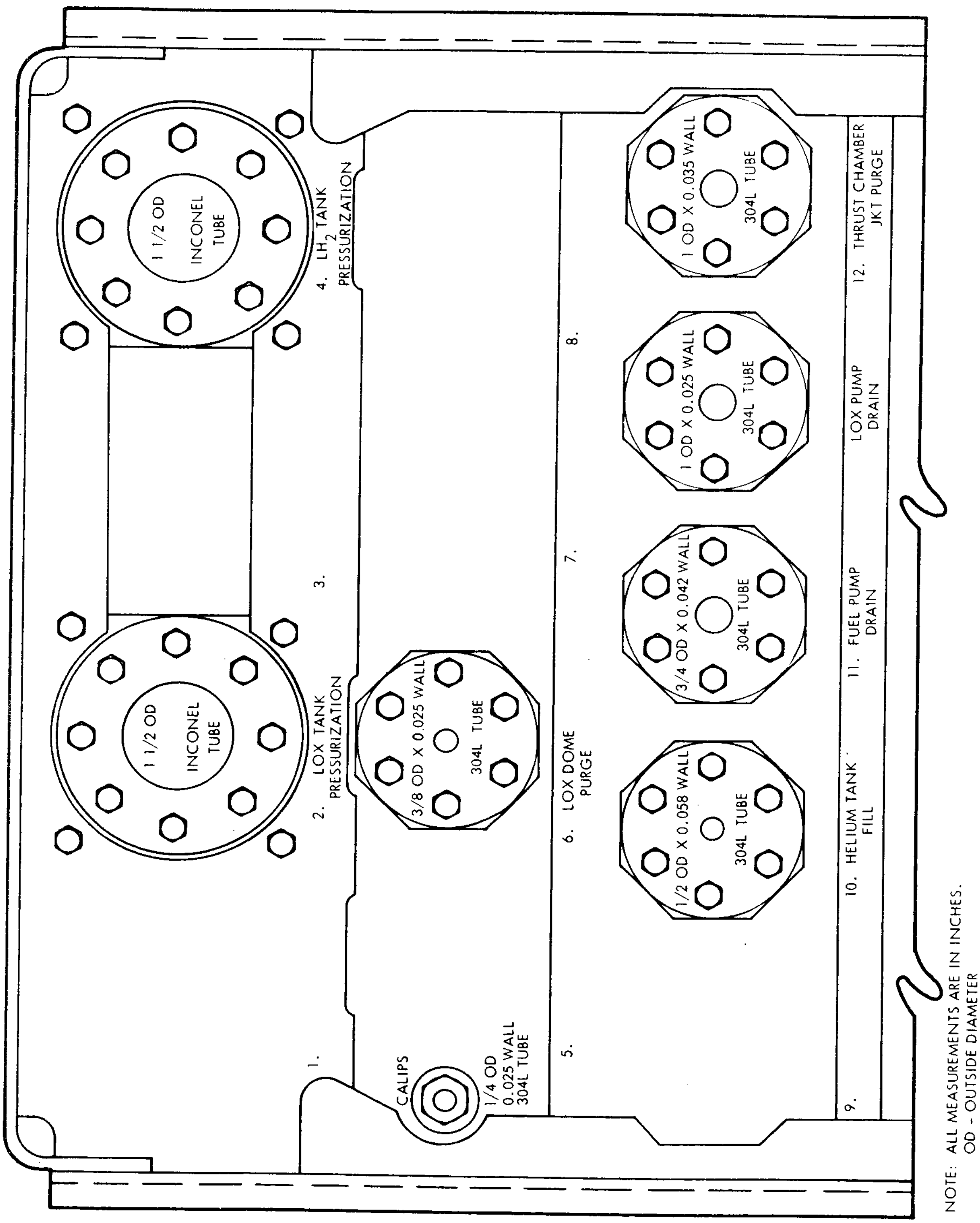


Figure 10.2-50. J-2S/S-II Outboard Engine Interface

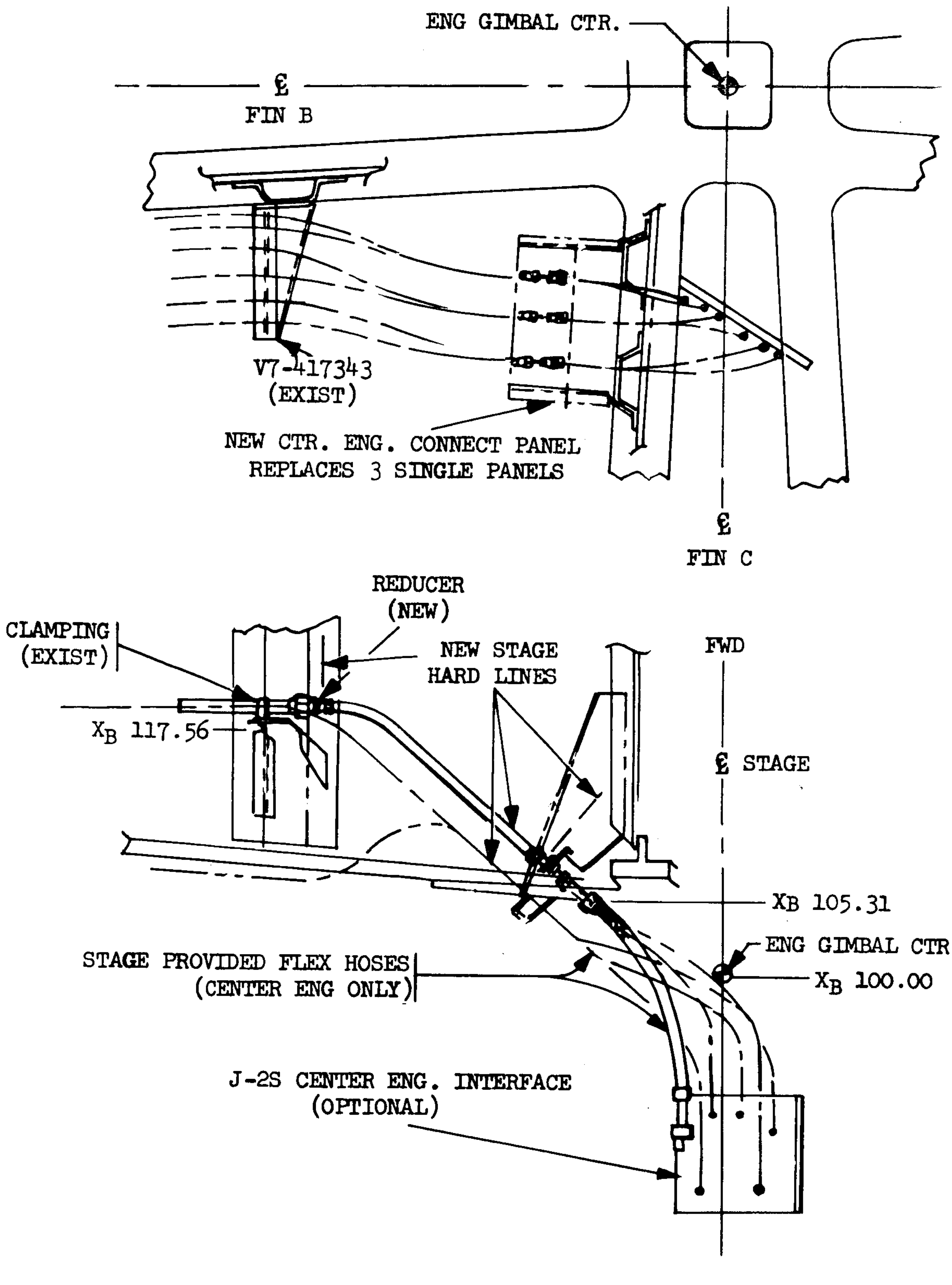


Figure 10.2-51. J-2S Center Engine Interface (Optional)

circular manifold (Figure 10.2-46) out along the center engine beam to a bracket will be the same as used with the J-2 engine. Reducers will be installed in the lines at this bracket to reduce the stage lines down to fit the engine line size. Short sections of stage hard lines (new) will be installed from the reducer to the connect panel where the flex hoses from the engine are attached.

Figure 10.2-52 illustrates the single sheet metal panel to be used at the center engine position with the J-2S engine. Flex hoses from the engine are joined to hard lines at the noted positions on this panel. Bulkhead type unions and "B" nut couplings will be used instead of the flanged joints used with the J-2 engine. Tube sizes listed are for the lines on the stage and engine side of the interface.

Figure 10.2-53 shows the configuration of the umbilical panels used with the J-2 engine and identifies the engine systems. These panels are located at the stage mold line and provide for mating the engine servicing lines extended from the engine to the umbilical carrier plate.

Figure 10.2-54 illustrates the umbilical panels used with the J-2S engine. A comparison can be made between this figure and the J-2 umbilical panels shown in Figure 10.2-53 as to which systems are retained, deleted, or reidentified.

10.2.2.1.2 LEO Mission

a. Engine Performance. Analysis of the low earth orbit mission has resulted in an operational requirement for the S-II stage to fire a normal full-thrust burn with cutoff by the IU followed by a coast (unpowered) period of approximately 45 minutes and then operation of the five J-2S engines in idle mode for approximately 164 seconds to circularize the vehicle orbit. Analysis of the S-II stage characteristics has shown that cutoff of the first burn (full thrust) through approximately 14 seconds of idle mode operation will greatly attenuate propellant slosh characteristics. Figure 10.2-55 illustrates the vehicle thrust versus time profile for the LEO mission.

The estimated thrust decay characteristics for a mainstage to idle mode cutoff is shown in Figure 10.2-56. This estimate is based upon the general cutoff characteristics of the J-2 engine mounted in an S-II stage and time-impulse relationships established in the Rocketdyne interface criteria document (R-7211).

The estimated thrust buildup characteristics to idle mode operation following a 45-minute coast period are shown in Figure 10.2-57. This estimate is based upon pressures at the time of restart (refer to Coast Pressurization, below) of 30 psia in the LH_2 tank and 32.5 psia in the LOX tank. It was assumed that subcooled liquid conditions would exist at each propellant tank outlet and that the feed ducts contained saturated gas.

The estimated idle mode cutoff thrust decay characteristics of the J-2S is shown in Figure 10.2-58. This estimate is based upon the thrust, impulse, and time relationships established in the interface criteria document.

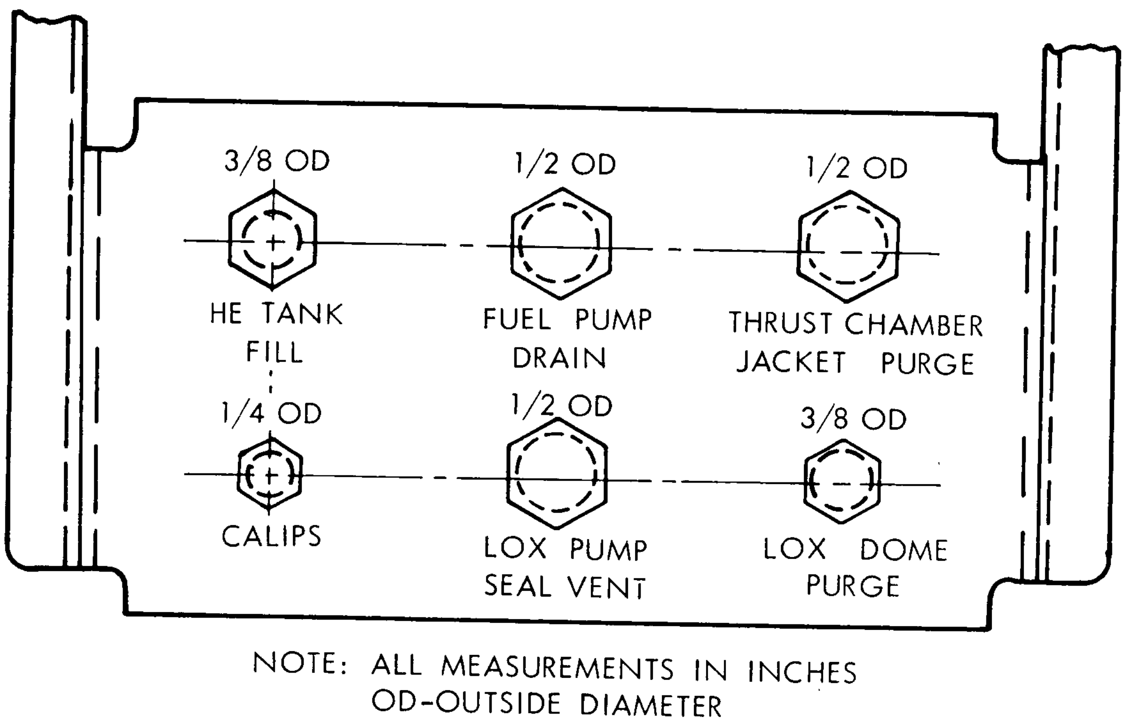


Figure 10.2-52. J-2S/S-II Center Engine Panel (Lower Fin C)

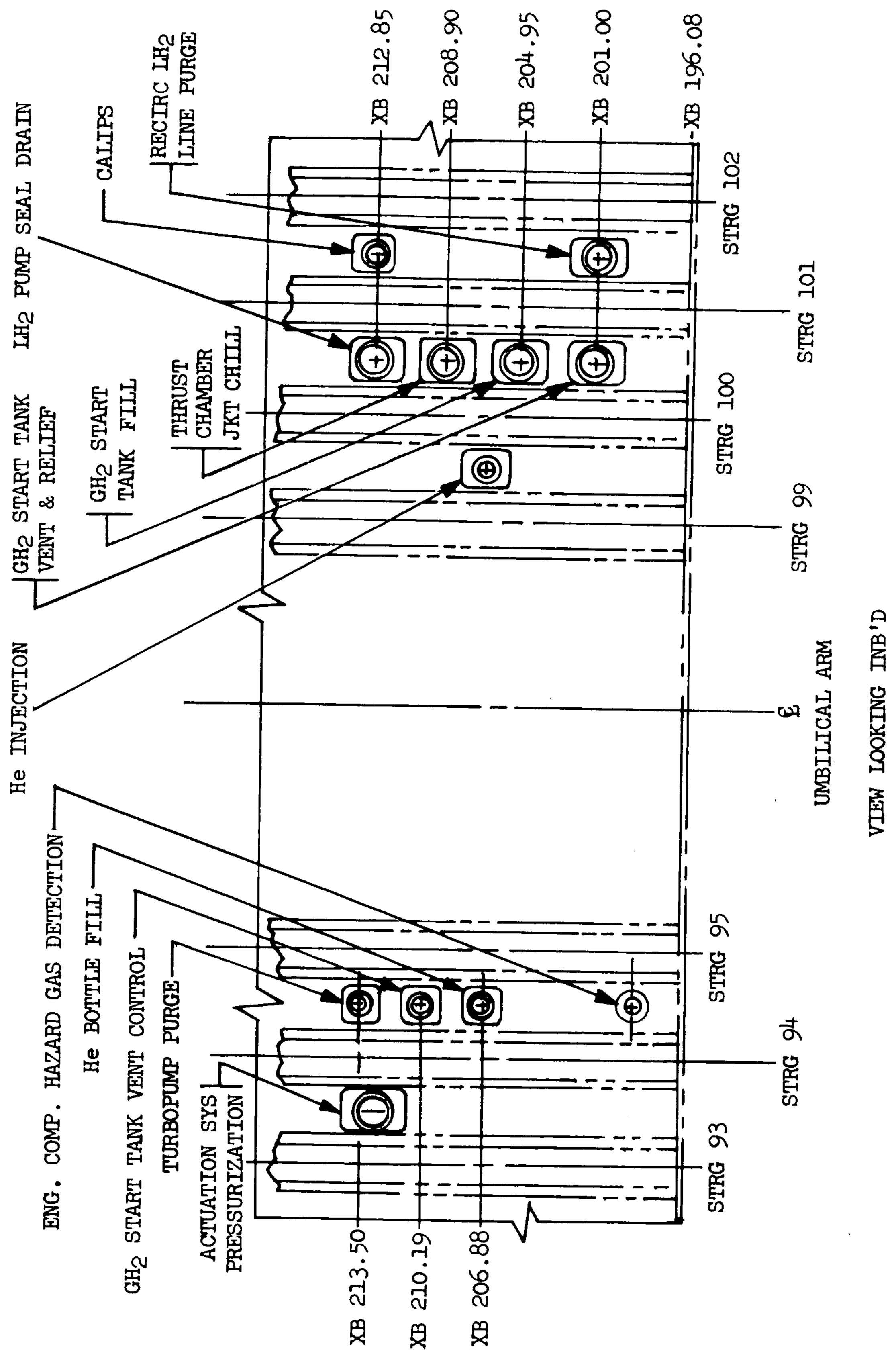


Figure 10.2-53. J-2 Engine Requirements

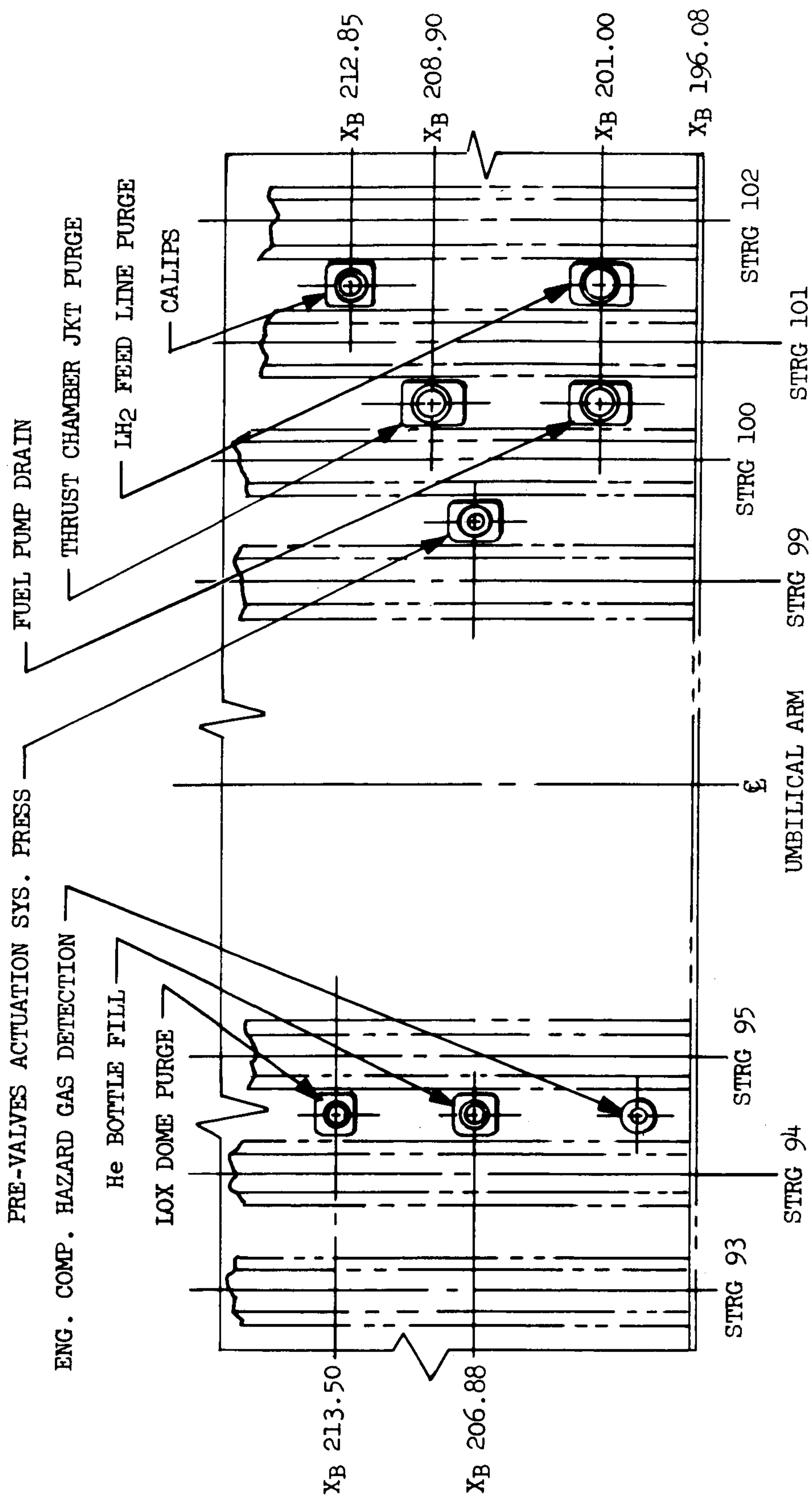


Figure 10.2-54. J-2S Engine Requirements

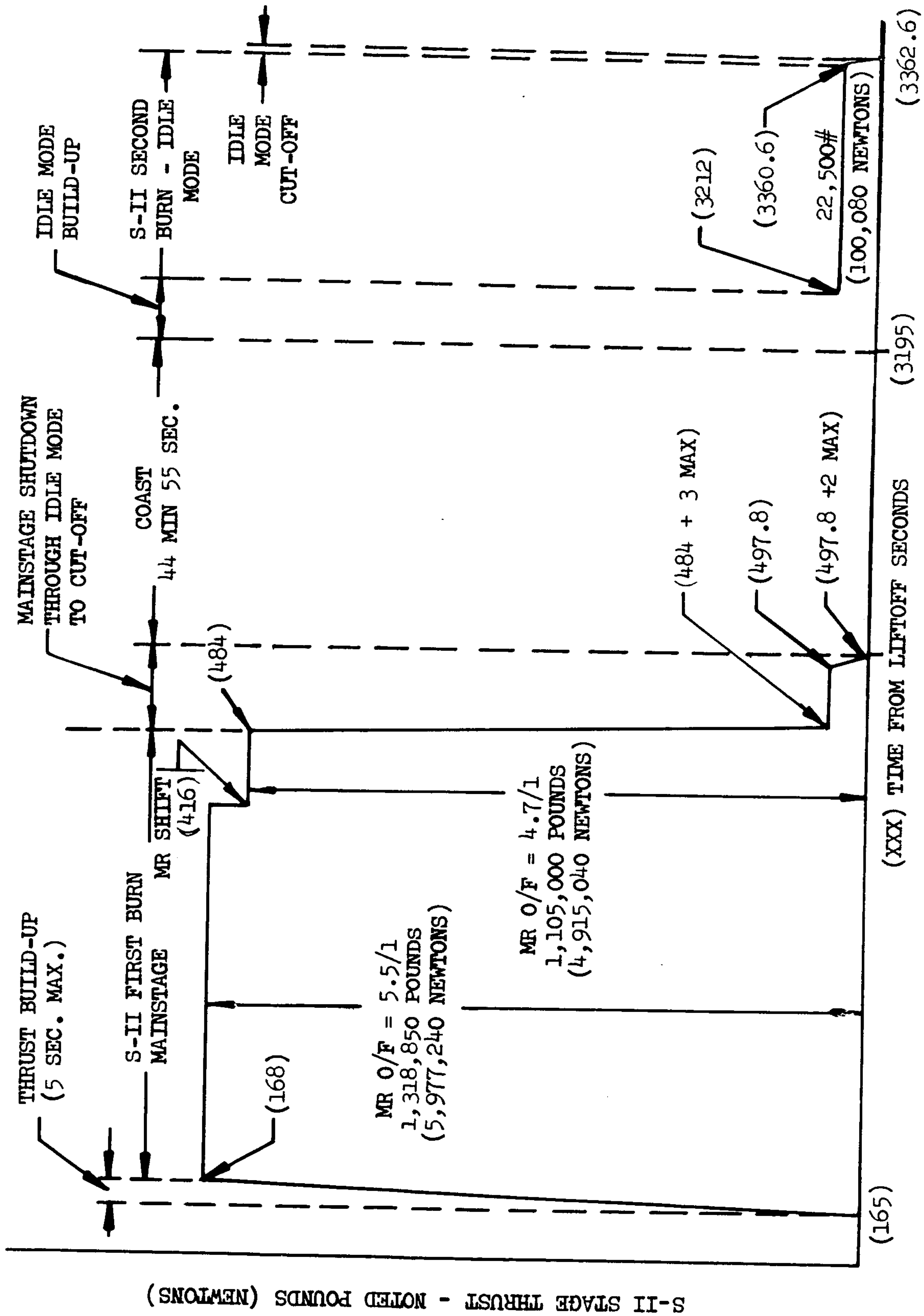


Figure 10.2-55. J-2S/S-II Thrust Versus Time-LEO Mission

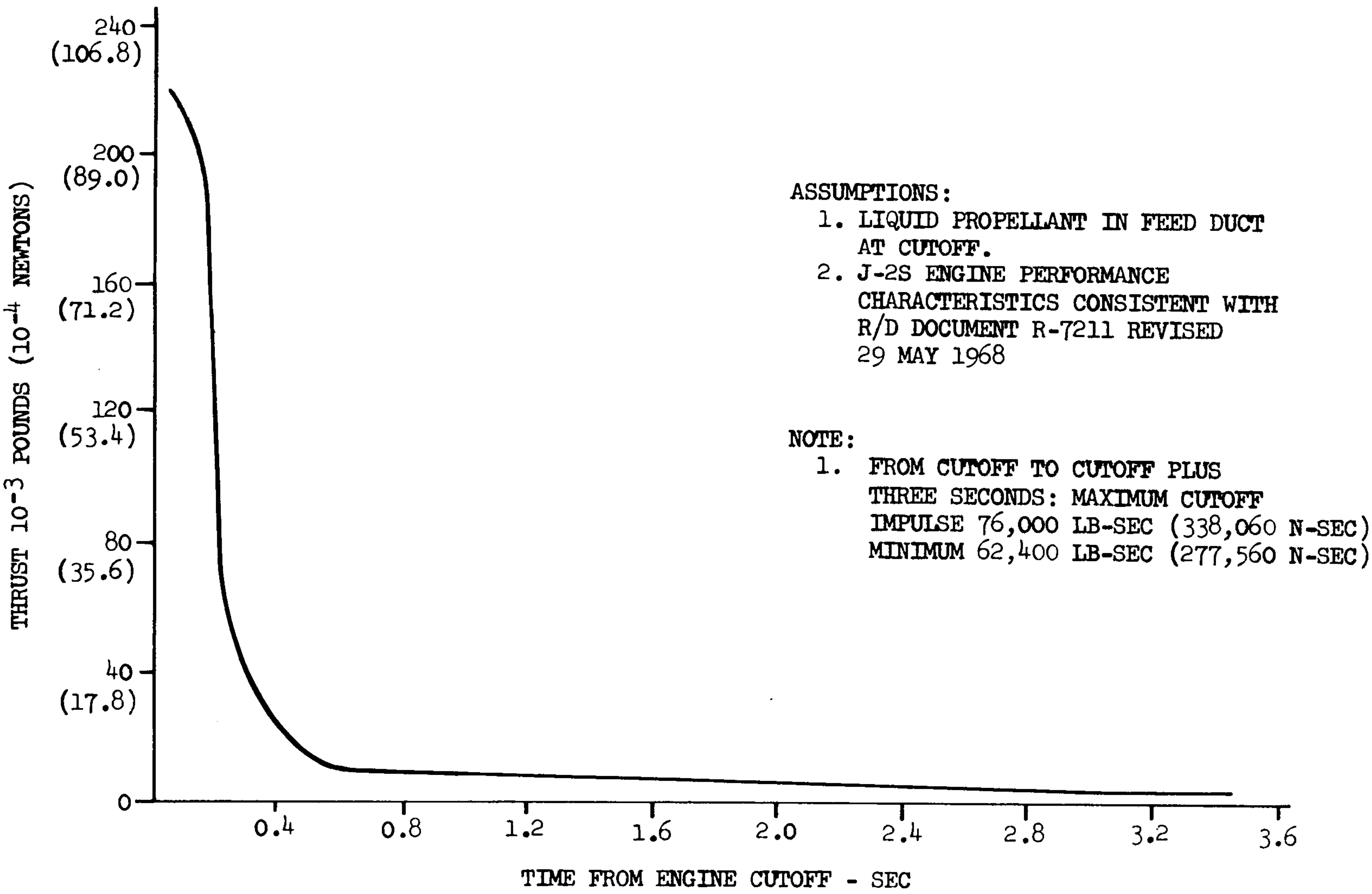


Figure 10.2-56. Estimated J-2S Thrust Decay Characteristics--Mainstage to Idle Mode at 4.7 EMR

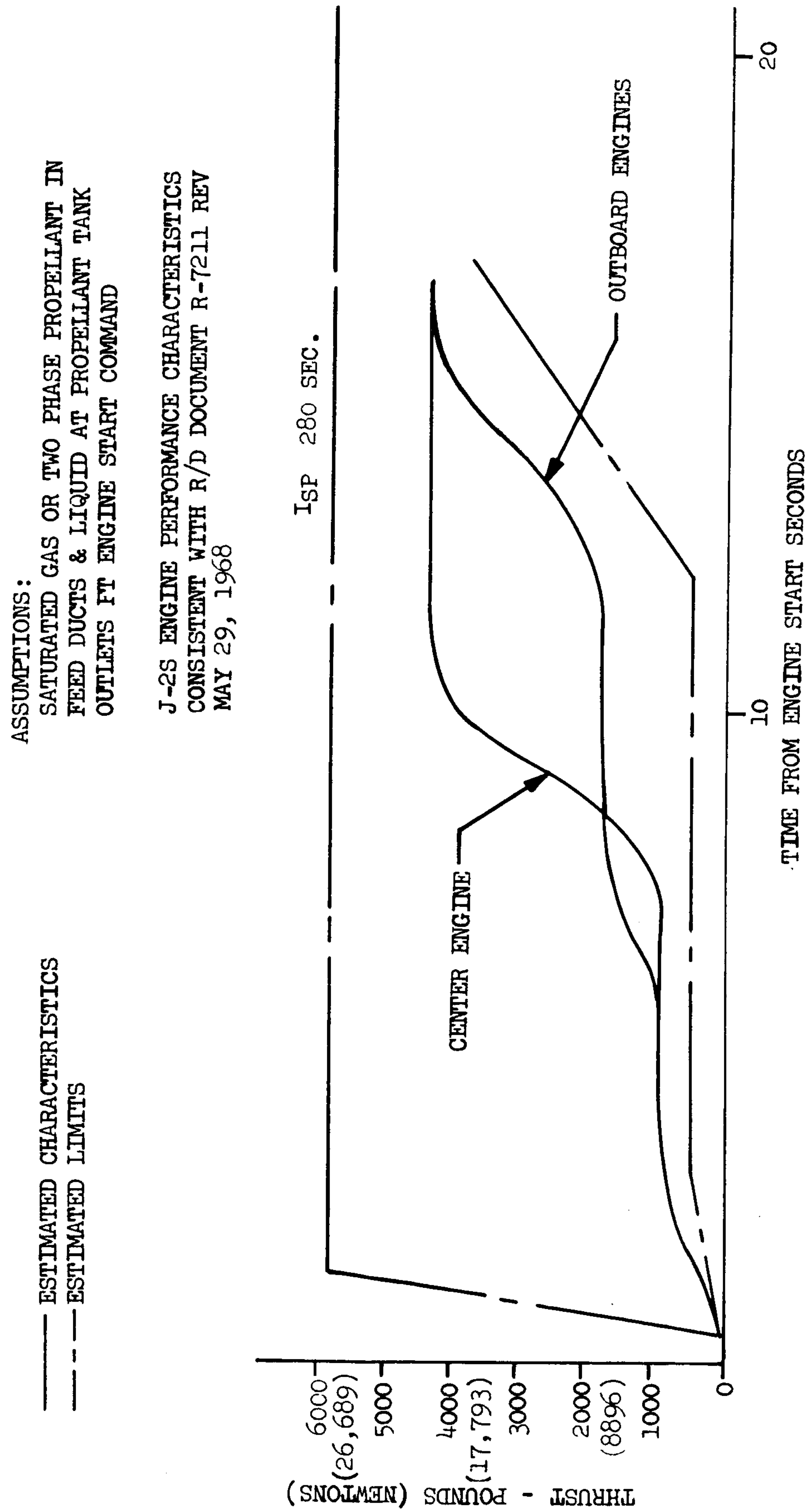


Figure 10.2-57. Estimated J-2S/S-II Idle Mode Thrust Buildup

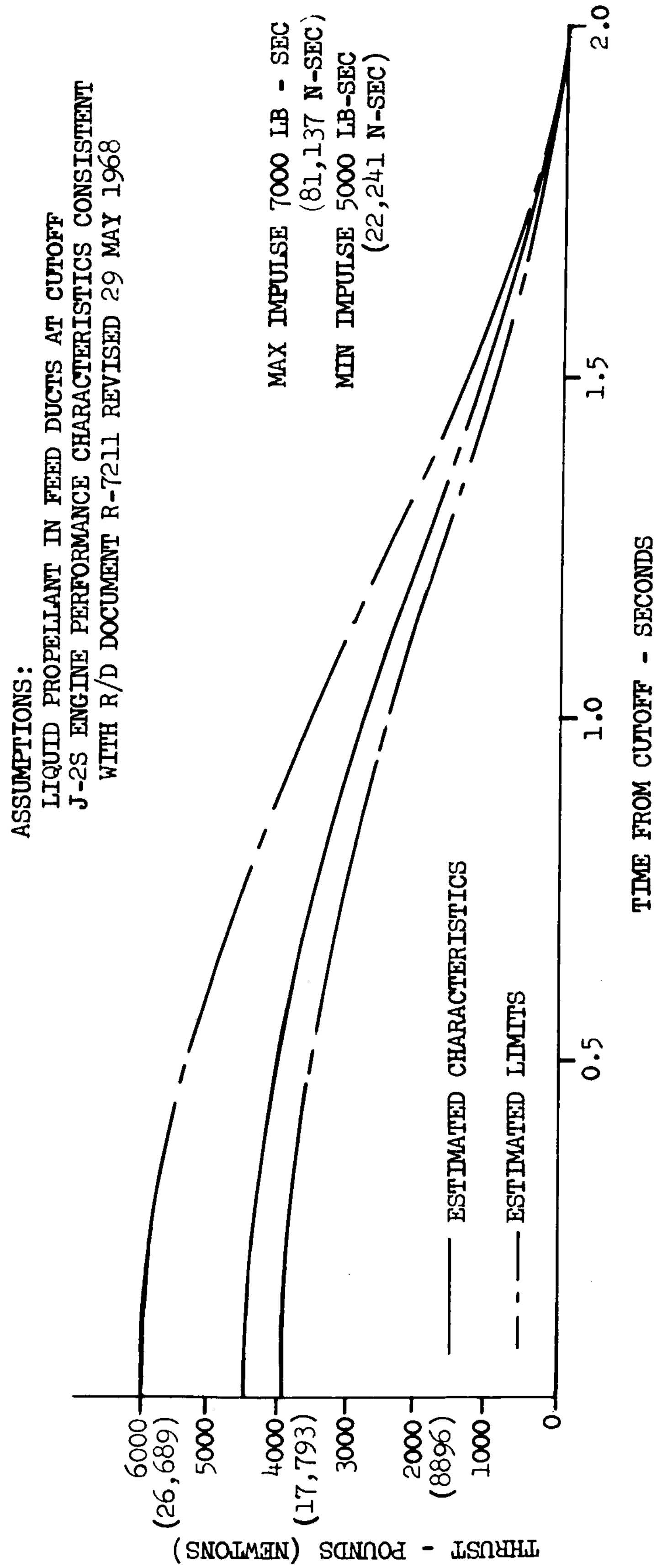


Figure 10.2-58. Estimated J-2S Engine Cutoff Characteristics--S-II LEO Idle Mode Cutoff (First and Second Burn)

b. Propellant Inlet Conditions. The propellant inlet condition requirements for first start on the LEO mission are identical to those for the LOR mission. Additional requirements for the LEO mission exist at first burn shutdown through idle mode to cutoff, restart to idle mode after the 45-minute coast, and idle mode burn of approximately 164 seconds. For idle mode operation, the propellant inlet temperature and pressure requirements, both liquid and gas, are as shown in Figure 10.2-34. To obtain the maximum idle mode thrust, it is essential to operate with propellants in the liquid state which is to the left of the saturation line shown on the figure.

At the initiation of mainstage cutoff to idle mode, the inlet conditions are subcooled liquid as attained during mainstage burn. Because idle mode operation will maintain a positive acceleration, it is concluded that the propellants at the engine inlets will remain liquid through cutoff from idle mode.

During the 45-minute coast period, a low g thrust is applied by balanced venting of the LH_2 tank. With the prevalves remaining open after first burn and with the settling thrust applied, a liquid condition will exist at the tank feed duct exits. It is further expected that the liquid condition will exist in portions of the feed ducts because of the low thrust with a resultant gaseous/liquid mixture at saturation temperatures.

Variable quantities of heat will be transferred into the feed ducts during coast: initial conduction from the engines and radiation from various sources (engines, stage, sun, etc.). It is predicted that the feed duct propellants will reach an equilibrium mixed phase (saturated liquid and gas) condition during coast. The low g thrust forces some liquid down into the ducts and permits some gas to rise into the tank.

The predicted saturated conditions adequately meet the second start to idle mode temperature requirements of a maximum of -100 F for both propellants. A conservative estimate based upon having the ducts full of saturated gas was used in predicting engine thrust buildup to idle mode.

The rate of heat transfer is expected to be low enough for liquid inlet conditions to exist for second burn once the initial duct gas is burned with its added thrust.

c. LOX Pump Seal Drain. When the S-II is modified for the LEO mission, line routing for the LOX pump seal drain requires three of the vents to be rerouted away from positions I and III. The new locations are shown in Figure 10.2-59. This relocation is caused by the requirement for a reaction control system on the LEO mission located at positions I and III.

The primary objective in choosing the new locations for the exit fittings is to place them in an area where they will receive the least aerothermal disturbance. These positions also require minimum redesign and remocking of lines.

The J-2S LOX pump primary seal leakage drain system will be the same as that currently on the J-2 engine. This system consists of a stage-furnished overboard

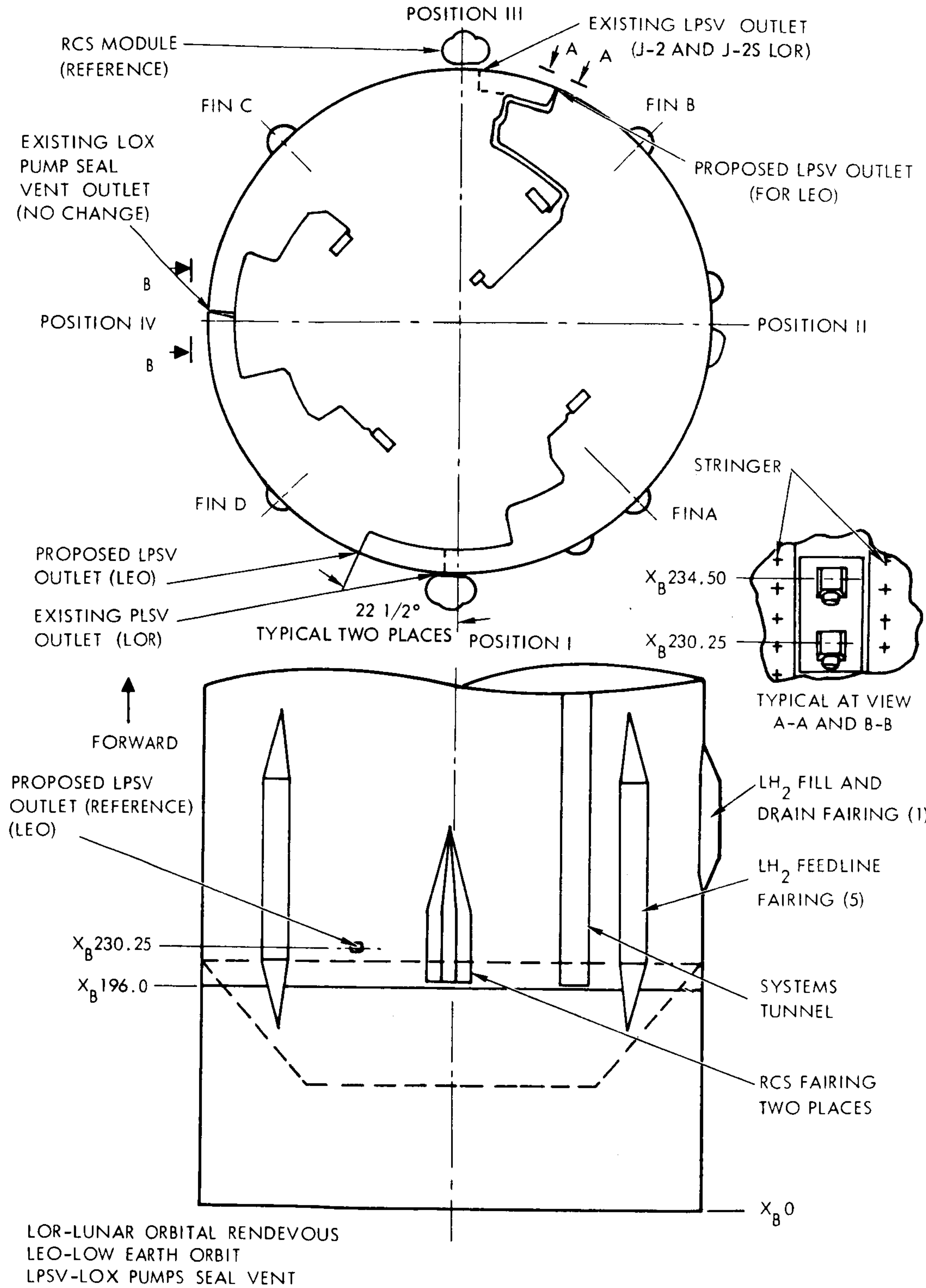


Figure 10. 2-59. J-2 and J-2S LOX Pump Seal Vent Outlet Locations

vent system and an engine-furnished positive drain system. The vent and drain systems are interconnected at the engine connect panel when the engines are installed on the stage.

During acceptance test static firing at MTF, the LOX pump seal leakage is ducted overboard into the engine jet plume and flame bucket by the engine-furnished positive drain line. This line contains a burst diaphragm housing, but the diaphragm is omitted for static firings. After static firing, the diaphragm is installed which precludes leakage entering the S-II engine compartment during KSC tanking tests, prelaunch and first-stage boost because this leakage is then vented overboard by the stage vent system.

When the engine(s) reaches mainstage, the diaphragm(s) is designed to rupture at a nominal 21 psid, and any leakage is drained by the positive drain line.

During LEO mission coast, the leakage is via the positive drain mode. The nominal leakage varies per engine from zero to a maximum value of 36,000 scim. If this engine-to-engine variation in leakage rates produces a significant flight control disturbance, then a thrust spoiling device, illustrated in Figure 10.2-60, will be added to the engine drain line and installed at the same time as the burst diaphragm.

10.2.2.2 Propellant Feed System

This section discusses a design change to the propellant feed system to accomplish system purging prior to propellant loading.

10.2.2.2.1 LOR Mission

a. Fuel Feedline Purge System. The J-2S, like the J-2, will require purging of the fuel feedlines prior to propellant loading. The purging procedures will remain essentially the same for the J-2S, which consists of a feedline helium pressure (30 psig) vent cycle operation.

   A system design change will be required for J-2S. The change in design is required because of the deletion of the fuel recirculation system which provided a common single-point purge at the fuel return line. Figure 10.2-61 schematically shows the new fuel feedline purge system for the J-2S engine installation. The system consists basically of a circular manifold that connects to each engine fuel feedline through a check-valve, and terminates at an umbilical disconnect.

   Figure 10.2-62 illustrates a partial line routing of the fuel feedline purge system. The system consists of a series of formed metal rigid tubes joined together with threaded fittings or by welding. Starting at the umbilical panel interface in the aft skirt, a one-inch diameter line is routed downward in the area between the thrust cone and the aft skirt, and ties into a circular manifold (also one-inch in diameter) just below Station 196. The circular manifold is common to all five engine fuel feedlines. The manifold is routed in a circular configuration 270 degrees around the outside of the thrust cone from Fin C to Fin D. Individual lines, 3/4-inch in diameter, with check valves, extend upward from the circular manifold and connect to the fuel feedlines just below the prevalve through the unused recirculation bypass port. These lines will be routed adjacent to the feedlines through the aft skirt and within the aerodynamic fairings.

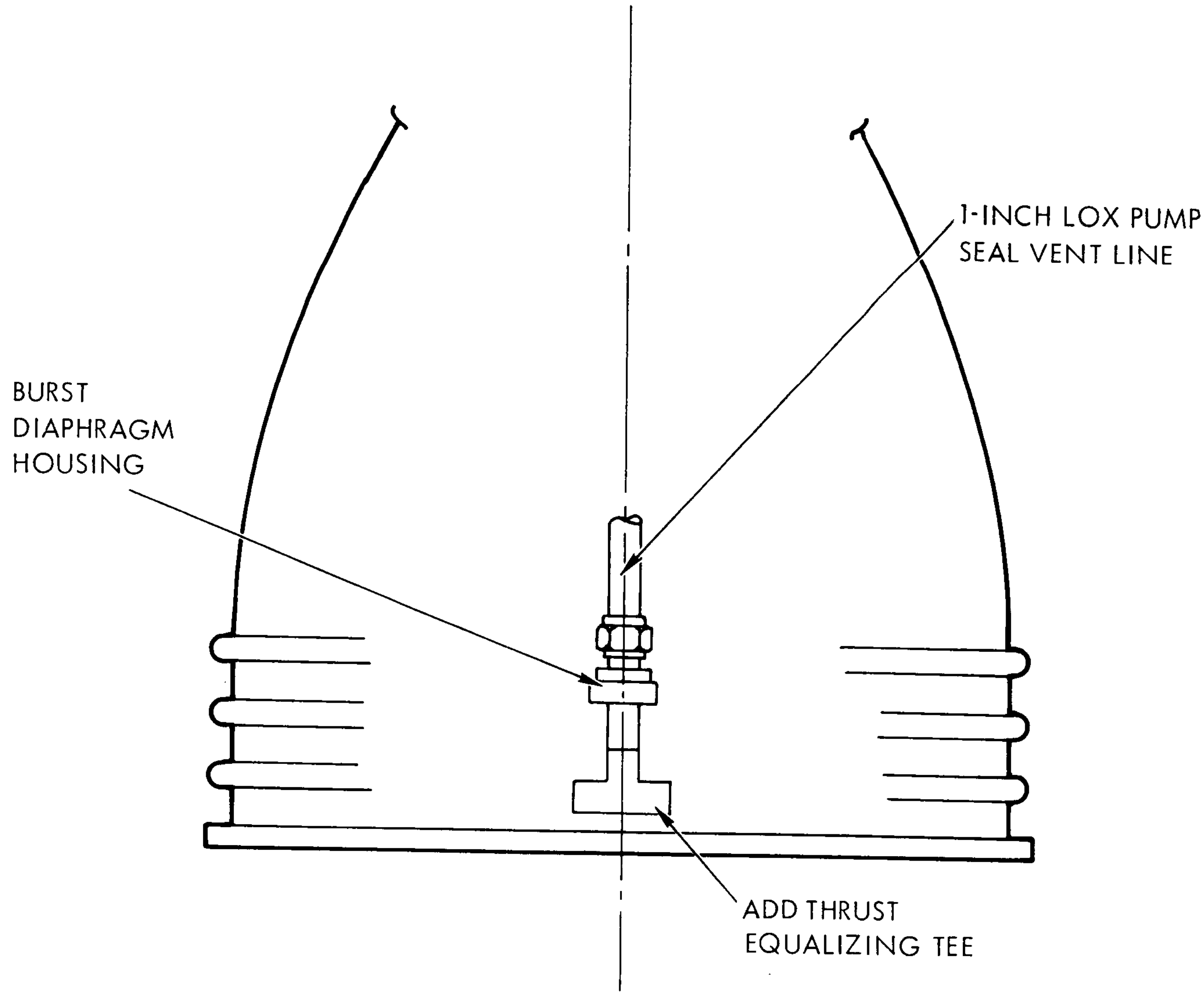


Figure 10.2-60. LOX Pump Seal Drain Zero Thrust Outlet (LEO Mission)

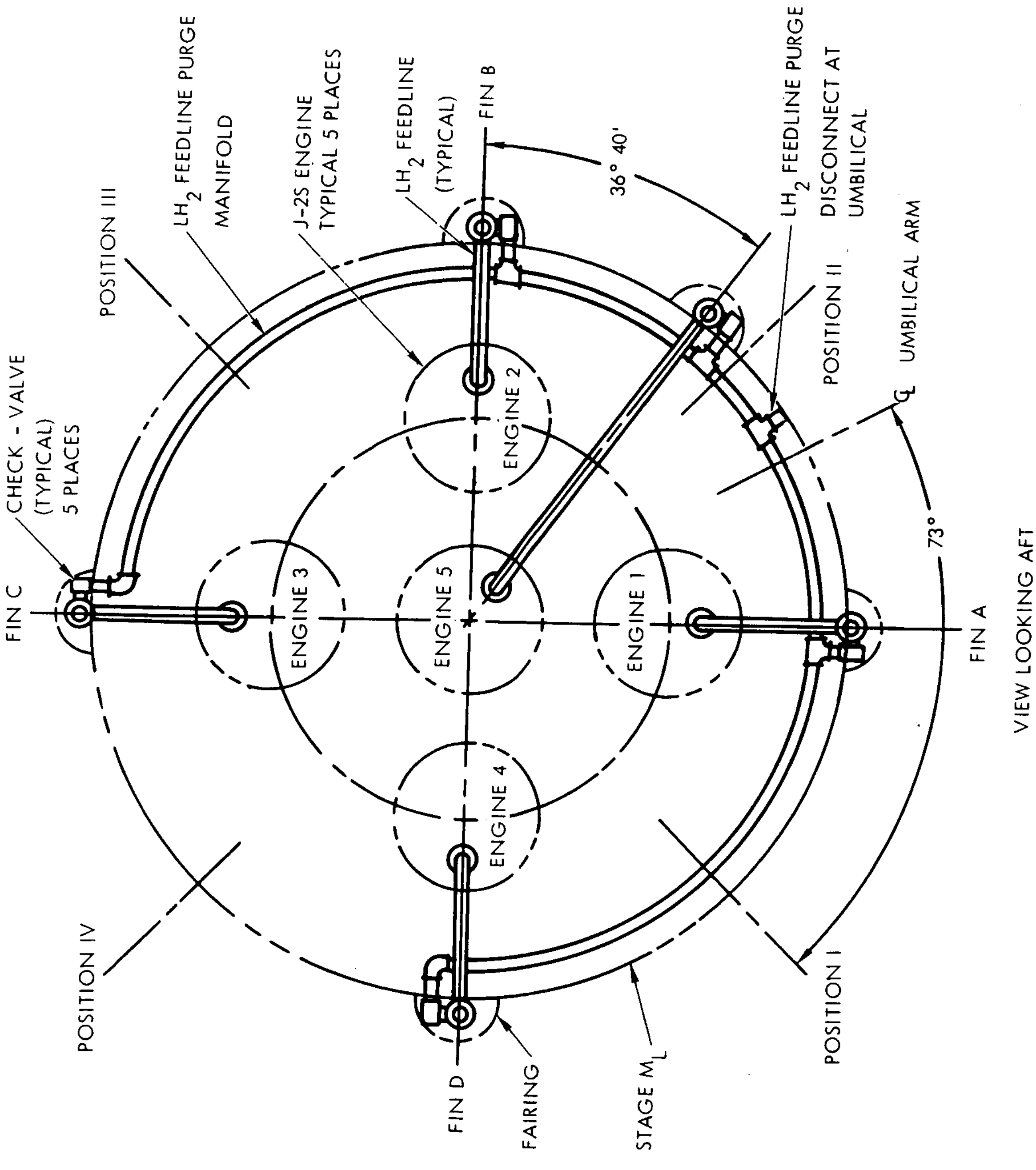


Figure 10.2-61. J-2S Engine LH_2 Feedline Purge System

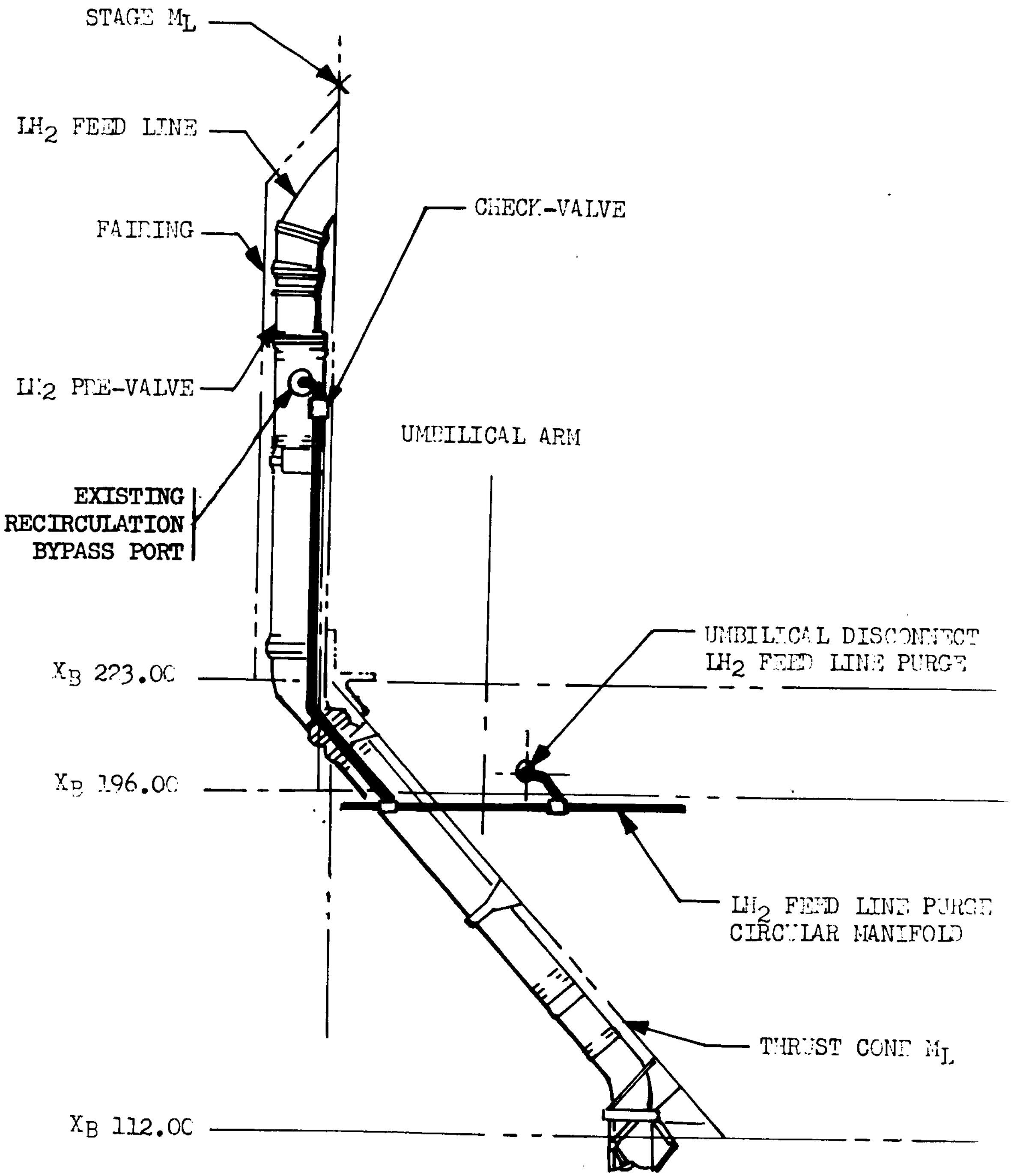


Figure 10.2-62. J-2S LH_2 Feedline Purge (Partial Elevation View)

10.2.2.2.2 LEO Mission

There is no change from that described for the LOR mission.

10.2.2.3 Propellant Management System

The propellant management system changes as described below are concerned with the engine cutoff sequencing to be used on the missions under study.

10.2.2.3.1 The LOR Mission

a. Functional Changes. The propellant management system will be revised to initiate all-engine cutoff at LOX exhaustion (mainstage OK pressure switch dropout). The present S-II stage uses a set of engine cutoff sensors (propellant level) in the LOX tank in conjunction with a timer to cause J-2 engine cutoff.

   The higher flowrates associated with the J-2S engine at 265,000 pounds of thrust will cause a more severe drawdown of the LOX surface near the end of S-II boost, even though g forces are higher than the present S-II. It is estimated that these characteristics with LOX exhaustion cutoff will result in an increase of 1500 pounds of LOX residual over that expected with a J-2 cluster using the engine cutoff sensors and a 1.5-second delay cutoff sequence. The estimated total LOX residual for the J-2S installed S-II stage is 3000 pounds in the tank and sump at cutoff command.

b. System Changes. The LOX depletion engine cutoff system installation in the LOX tank will remain intact for the J-2S equipped stage. The electronics associated with the five LOX tank ECO transducers also will remain intact, but will be disconnected from the engine cutoff circuit and used only to supply an arming signal to a mainstage OK pressure switch engine cutoff circuit. This mainstage OK cutoff circuit will provide a cutoff signal to all five engines upon dropout of the first engines pressure switches. Circuitry will be provided to ignore any engine-out condition that occurs prior to arming of the cutoff system. The LOX depletion ECO circuit will be armed continuously, and the existing switch selector signal to arm that circuit will not be required.

   Further studies will be conducted to determine the optimum cutoff system for maximum stage reliability and payload. The use of an idle mode burn phase after mainstage cutoff offers the possibility of reducing the propellant and ullage gas residuals by as much as 8000 pounds and increasing LOR escape payload by approximately 1500 pounds, but further system complications, such as unbalanced thrust will reduce this value. A total system review will therefore be initiated at program implementation.

10.2.2.3.2 LEO Mission

a. Engine Cutoff System. The LEO mission will require three normal cutoff commands as compared to the single command at LOX exhaustion on the LOR mission. A mainstage cutoff command will be issued by the IU when the vehicle reaches a predetermined velocity, which will result in the engines' continuing operation in the idle mode. An all-engines cutoff command will be impressed upon the engines a specified number of seconds later, which will terminate idle mode operation and put the vehicle in a coasting condition. Following completion of the orbital circulization maneuver, accomplished by restart of the J-2S cluster to idle mode, an all-engine cutoff command will be issued by the IU.

A backup mainstage engine cutoff capability will be provided by retention of the present depletion cutoff systems. The present (S-II-3) system will be retained as is, except that the five LOX low-level sensors will be raised from the sump area at Station 157 to the top of the cruciform baffle at Station 170 to ensure that adequate LOX remains to permit execution of the second idle mode burn for an orbital circulization maneuver. For the same reason, the LH_2 low-level sensors will be raised from their present location at Station 369 to Station 410. This backup cutoff circuit will only be able to command mainstage cutoff, and therefore will not affect idle mode operation.

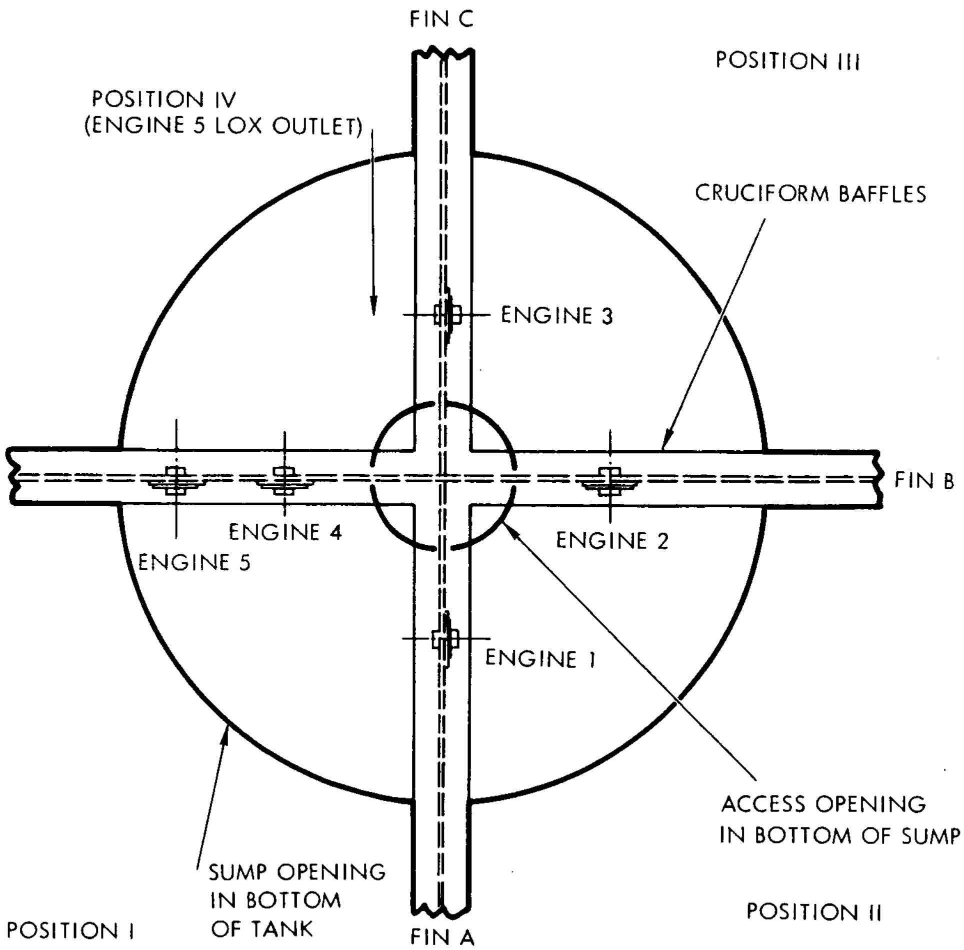
Upon receipt of a mainstage cutoff command from either source (IU or propellant depletion sensors), the stage circuitry will lock out the capability to close the prevalves. A failure assessment associated with this action and time period indicates no catastrophic failure modes. Further studies associated with propellant conditions in the feed ducts during the coast and restart time periods may alter this lockout capability.

b. Depletion Cutoff Sensor Relocation. Figure 10.2-63 depicts the new location requirements for the LOX tank low-level sensors (X_B170). The five sensors will be mounted on the cruciform baffle webs which lie on the four outboard engine center lines. Each sensor will be bolted to a machined pad which, in turn, will rivet to the cruciform web. The existing wire harness assembly for the sensors will have to be lengthened due to the 13-inch forward movement (X_B170 - X_B157 = 13 inches). The existing sensor-mounting pedestal will be deleted.

Figure 10.2-64 represents a typical sensor mounting pad.

Figure 10.2-65 shows the new location of the LH_2 tank low-level sensors at Station X_B410 (was X_B369). The access cover plate and mounting flange have been raised accordingly.

c. Propellant Residuals for Second Burn. Due to the very low propellant flowrates in idle mode operation, it is estimated that the LOX residual weight trapped in the tank and sump will be reduced to approximately 1000 pounds (despite lower g forces) at the start of gas ingestion at the engine. The weight of fuel trapped in the LH_2 tank at the start of gas ingestion is estimated to be 1200 pounds, allowing for thrust misalignment. Studies will be conducted to verify that any gas ingestion and resulting thrust unbalance can be tolerated by the flight control and reaction control systems. The potential increase in payload from burning all available liquid and part of the ullage gases may be as great at 4000 pounds. During second burn idle mode, considerable uncertainty exists concerning the engine mixture ratio, because it is quite sensitive to ullage pressures. To accommodate an assumed mixture ratio of $1.45^{+1.35}_{-0.60}$, a second burn mixture ratio shift (MRS) allowance of 1830 pounds LOX and 1690 pound LH_2 is provided. This is over and above the mainstage flight performance reserve.



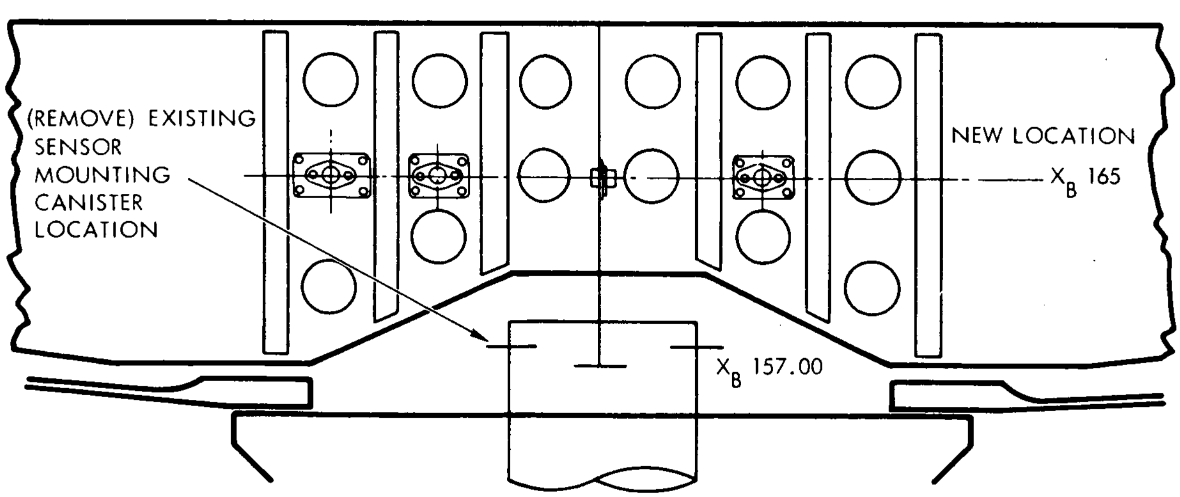


Figure 10.2-63. J-2S LOX Tank Low Level Sensor Locations

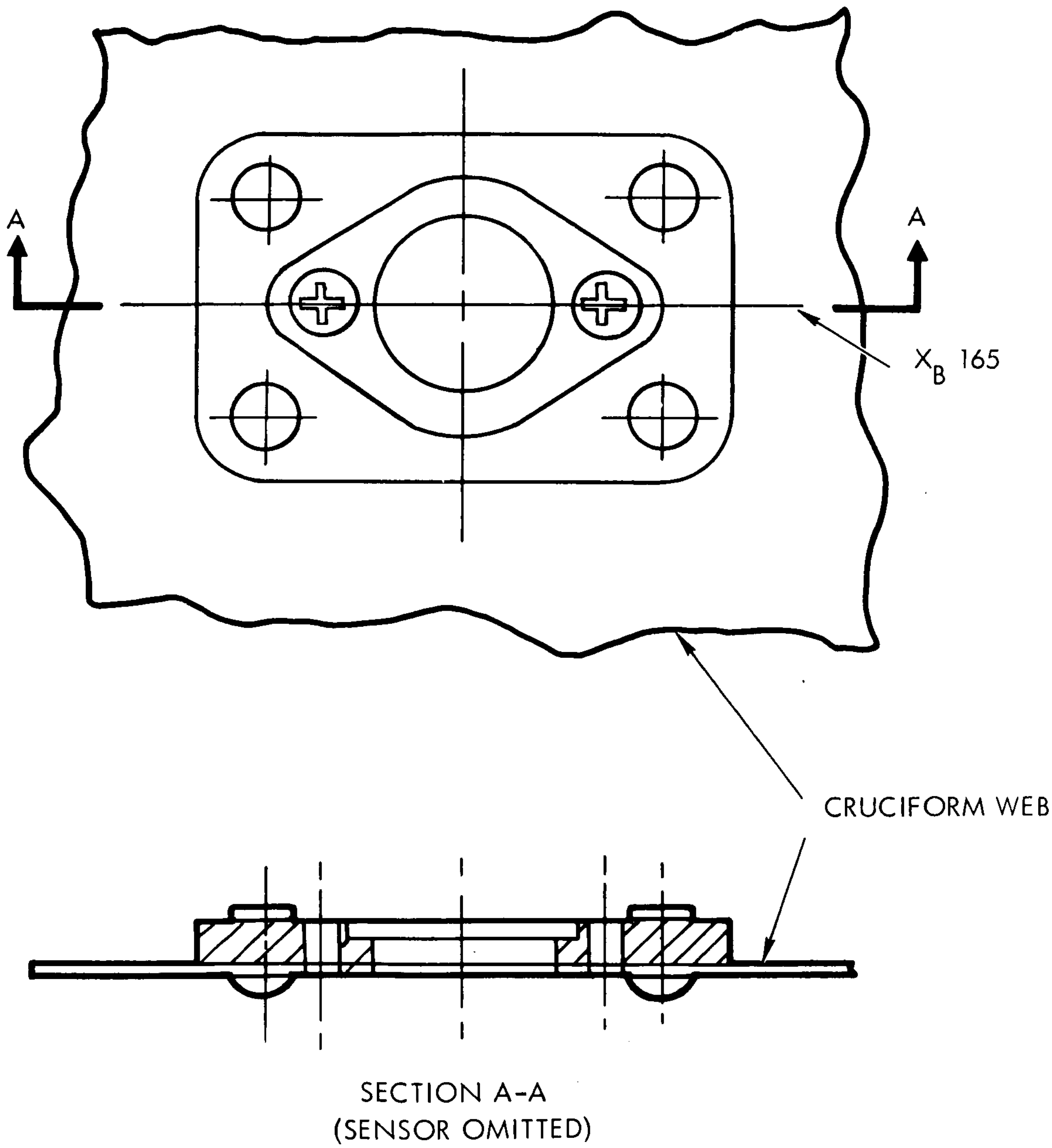


Figure 10.2-64. LOX Depletion Sensor Fitting

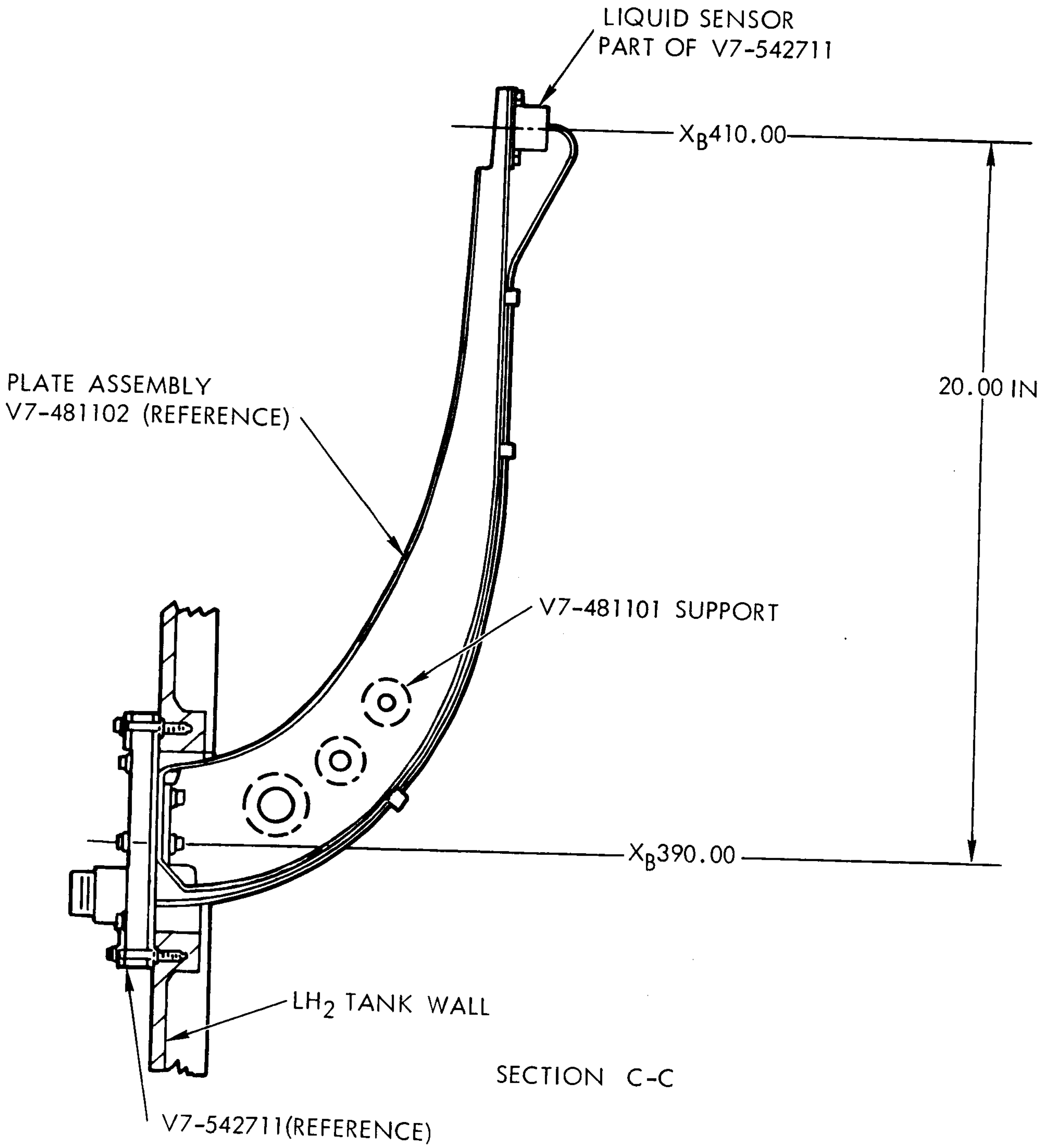


Figure 10.2-65. J-2S LH_2 Tank Low Level Sensor Locations

A list of the propellant items which comprise the nominal propellant masses at mainstage cutoff is presented below:

Itemization of Propellant Mass at Mainstage Cutoff
(Tanks and Sump Only)

Item	LOX (Pounds)	LH_2 (Pounds)
Mainstage Flight Performance Reserve (FPR)	4755	1025
First Burn Idle Mode Consumption	619	565
PU Bias	-	530
Boiloff in Coast	100	250
Nominal Second Burn Consumption (1.45 EMR)	7490	5160
Second Burn Mixture Ratio Shift Allowance	1830 (2.8 EMR)	1690 (0.85 EMR)
Geometry Trapped Residuals	1000	1200
Nominal Propellant Remaining at MECO	15,794	10,420

The minimum propellant required at MECO to ensure circulization of the orbit in second burn is obtained by subtracting the mainstage flight performance reserve and PU bias from the nominal mass at MECO, yielding 11,039 pounds LOX and 8,865 pounds LH_2. These minimum required masses dictated raising of the depletion cutoff level to a minimum station (room temperature) of 168.0 in the LOX tank and 407.6 in the LH_2 tank. To allow adjustability about this level by means of the ECO time delay modules, the ECO point sensors will be installed somewhat higher at station 170 in the LOX tank and station 410 in the LH_2 tank.

d. Anti-Slosh Baffles. Predicted vehicle acceleration and associated first mode sloshing frequencies in the propellant tanks are presented below for three significant times.

First Mode Lateral Slosh Frequencies

Mission Phase	LH2 Tank (cps)	LOX Tank (cps)	Slosh Amplitude Magnification Factor
Main Thrust Just Prior to MECO $a = 3.8 g_o$	0.23	0.38	1.0
Idle Mode $a = 0.07 g_o$	0.034	0.054	54
Propulsive Vent Coast $a = 2 \times 10^{-5} g_o$	0.0006	0.0009	190,000

The possibility exists that MECO will occur when the mainstage slosh kinetic energy is at its maximum (liquid surface is perpendicular to the tank axis). When this energy is converted to potential energy at the lower acceleration values which follow MECO, a higher peak slosh amplitude can result that is inversely proportional to the ratio of acceleration. For example, a 1-inch maximum amplitude sloshing prior to MECO could result in a 54-inch maximum wave height in first idle mode burn, as indicated by the amplification factors presented in the table above. For the coast period, the theoretical amplitude factor has a value of 190,000. This indicates that even a small remaining slosh velocity at first idle mode cutoff could carry the LOX to the forward part of the tank and result in excessive boiloff from contact with the warm sidewalls or in collapse of the tank pressure due to cooling of the ullage gas from contact with finely divided LOX. To minimize the chance of large amplitude excursions in coast, first idle mode cutoff is programmed to occur at 13.8 seconds after MECO, which would catch the worst case residual slosh wave at or near its point of zero velocity. This would also allow time for the damping action of the cruciform baffles to dissipate much of the slosh energy.

For further slosh control, it is tentatively proposed that a retention baffle be added at the LOX level expected at first idle mode cutoff. Such a baffle would also help to prevent excursion of the LOX due to RCS attitude corrections and maneuvers which might uncover the LOX sump before restart. This baffle, shown on Figure 10.2-66 would consist of a 24-inch-wide conical ring perpendicular to the tank wall and quarter immersed at MECO. During the design implementation phase a study will be made of the feasibility of controlling propellant sloshing through special restrictions on the operating cycle of the

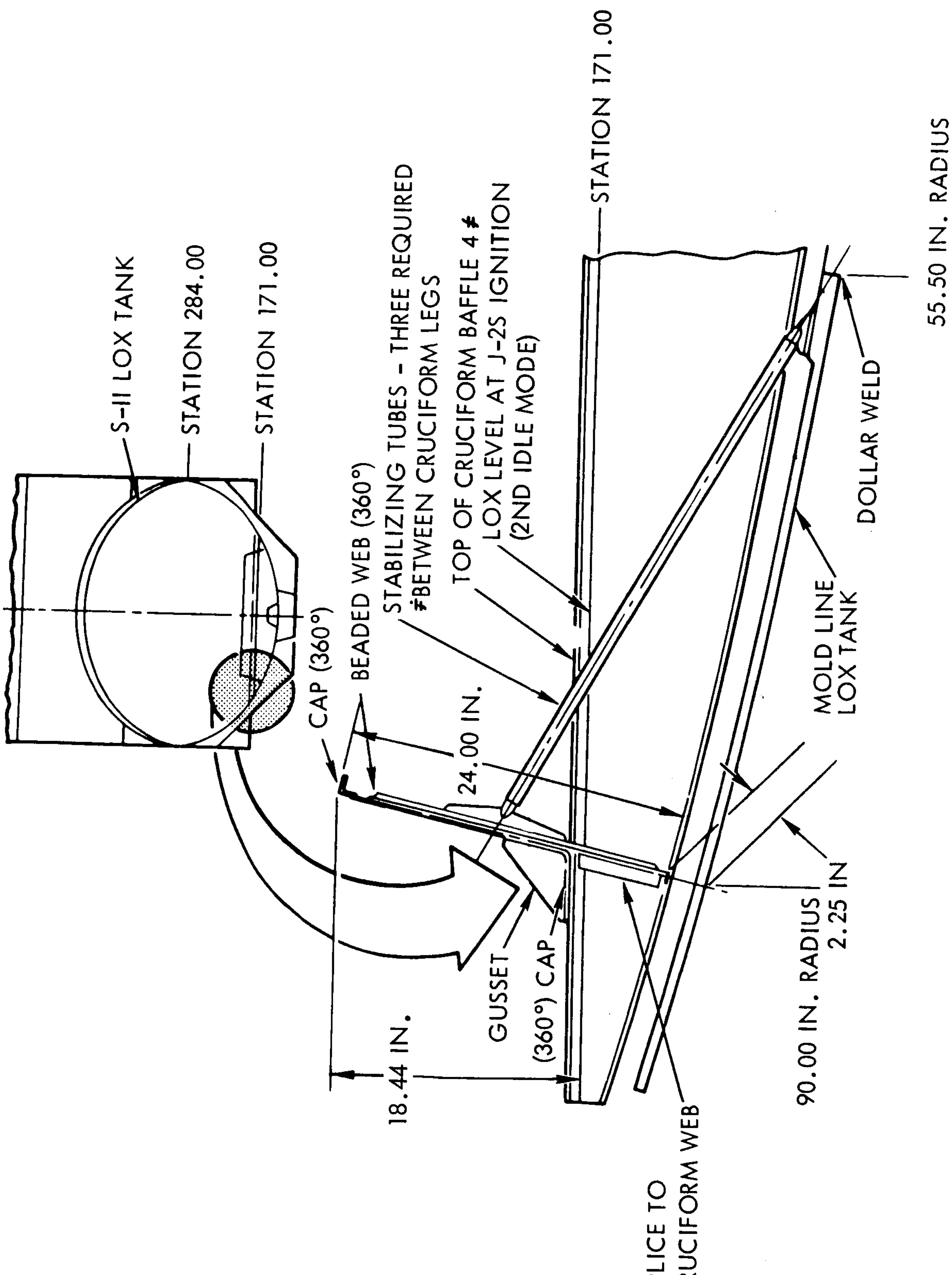


Figure 10.2-66. LOX Tank Slosh Baffle

RCS system, thereby minimizing the need for the retention baffle. No retention baffle is planned for the LH_2 tank since the wall frames provide considerable impedance to large amplitude liquid excursions. The GH_2 propulsive vent provides a vehicle acceleration of 2 x 10^{-5}g during coast which is sufficient to settle any liquid particles in 370 seconds on a free fall basis.

10.2.2.4 Pressurization System

10.2.2.4.1 LOR Mission

a. Functional Changes.

1. LOX Tank Pressurant - Because the oxygen heat exchanger outlet pressure is higher for the J-2S engine (1700 psia) than for the J-2 engine (1100 psia), it is necessary to orifice the manifolded pressurant line upstream of the stage regulator.

   Consideration was made of orificing the outlet side of the heat exchangers adjacent to the interface between the engines and the stage. This approach, which held the maximum regulator inlet pressure to the present 1100 psi at low flowrate, was discarded due to the excessively large pressure loss at high pressurant flow rates which result in starving of the pressurant system.

   Analysis has shown that, with the higher temperature of the pressurant gas of the GOX heat exchangers, the output of four heat exchangers is adequate to meet all LOX tank pressurant requirements. The temperatures are 620 F maximum out of the J-2S heat exchangers versus 440 F maximum out of the J-2 engines. Therefore, to simplify the customer-connect panel to the center engine, the pressurant line from that engine will be removed for the J-2S-equipped stage.

2. LH_2 Tank Pressurant - The hydrogen pressurant tapoff pressure is also higher for the J-2S engine than it is for the J-2 engine. Normally, the J-2S is capable of an output of 1300 psia maximum and the J-2 maximum is 900 psia. This means that the stage hydrogen pressurant lines will have to be orificed, and the lines between the engine hydrogen tapoff and the orifice will have to be requalified to the new pressure levels.

b. System Changes.

1. Figure 10.2-67 illustrates the installation of the LOX and LH_2 pressurization manifolds for the J-2 and J-2S engines. The manifolds extend from the engine connect panels to the LOX regulator and to the LH_2 tank pressurization line, respectively. These manifolds are constructed of Inconel 719 and consist of several sections, some of which contain gimbal joints to allow for expansion and contraction due to temperature and structural motion. The manifold systems are identical for both engines with the exception of the center engine LOX manifold which is not required with the J-2S engines.

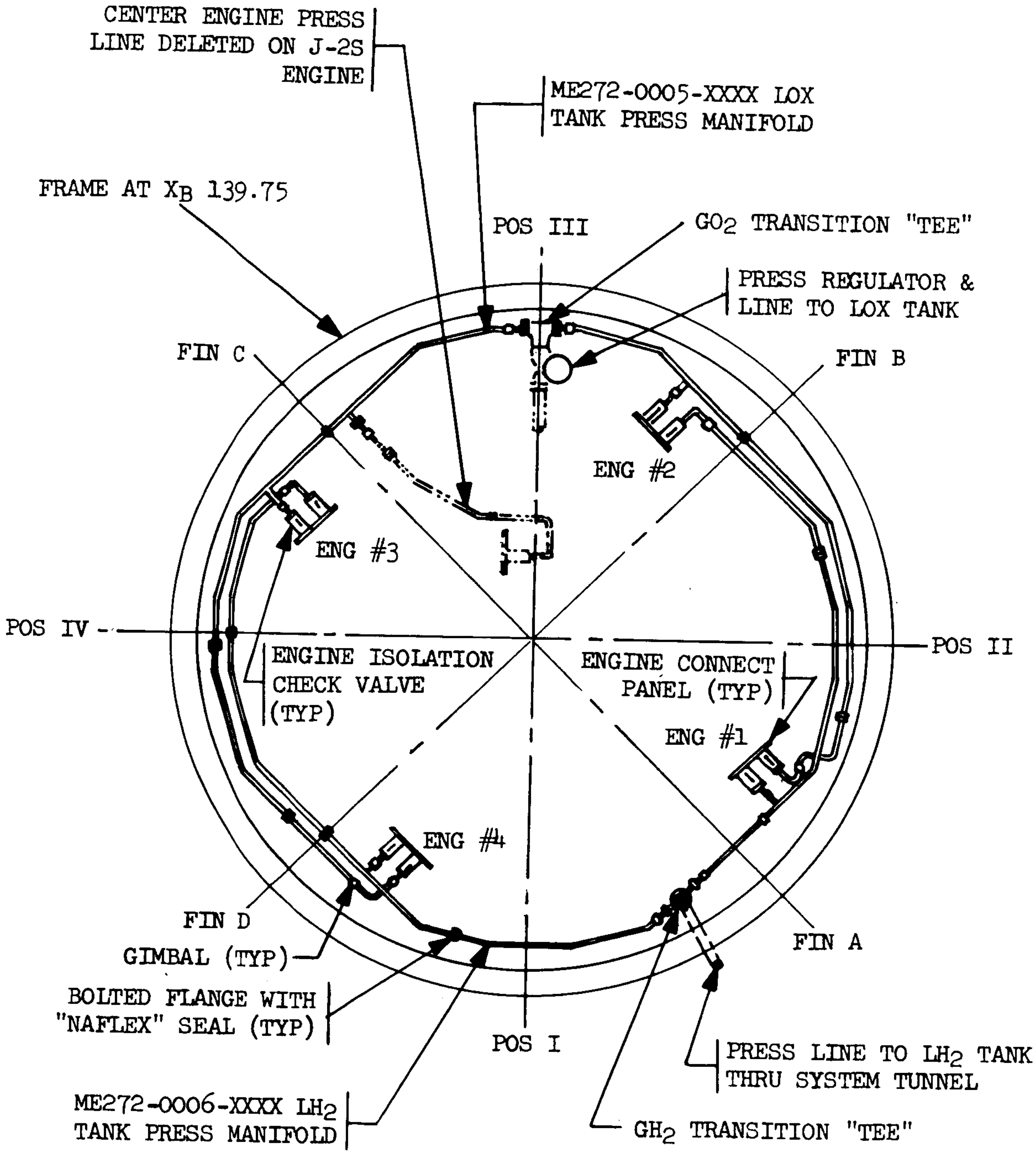


Figure 10. 2-67. J-2 and J-2S Engine LOX and LH_2 Tank Pressurization Manifolds

2. Orifices will be installed upstream of the pressurant regulator to reduce the higher pressure outputs of pressurant gas from the J-2S engine. The size and location of the orifices will be determined following a total system analysis. Figure 10.2-68 shows the pressurization system presently being used on the S-II stage. The anticipated installation of orifices is also shown for adaptation of the system to the J-2S engine.

3. Table 10.2-VI is a listing of procured elements in the present pressurization system and the effect on them of J-2S engine incorporation.

4. Partial requalification of the entire pressurization inlet to regulator systems is required for the higher tap-off pressure output of the J-2S engine.

5. The prepressurization and venting systems will remain identical to the present configurations.

c. System Analysis. The initial study ground rules defined an increase in the mainstage fuel NPSH for the J-2S engine. The NPSH requirements for the J-2S are shown in Figure 10.2-69. The analysis presented below was conducted on this basis with the following conclusions:

1. The present S-II LH_2 tank two-stage pressure level is sufficient to meet the NPSH requirements of the 265,000-pound thrust J-2S engine.

2. The present S-II LOX tank pressure level is sufficient to meet the NPSH requirements of the 265,000-pound thrust J-2S engine.

Subsequently, the J-2S NPSH requirement was lowered to the existing J-2 requirement; a reinforced conclusion therefore can hence be drawn that current tank pressure levels are adequate.

The factors that have the greatest influence on propellant tank pressure levels are engine NPSH requirements, propellant temperature, propellant feedline pressure losses, and fluid head at the pump inlet. In establishing tank pressure levels, some ground rules and assumptions must be made regarding these factors. For this study, the current S-II stage design criteria and assumptions were used and are as follows:

	LOX Tank	LH_2 Tank
Vent system back pressure	15.5 psia	16.2 psia
Initial bulk temperature corresponding to vent system back pressure	163.1 deg (R)	37.1 deg (R)
Design heat loads	108,000 Btu	209,000 Btu
Temperature stratification	1.5 deg	1.9 deg

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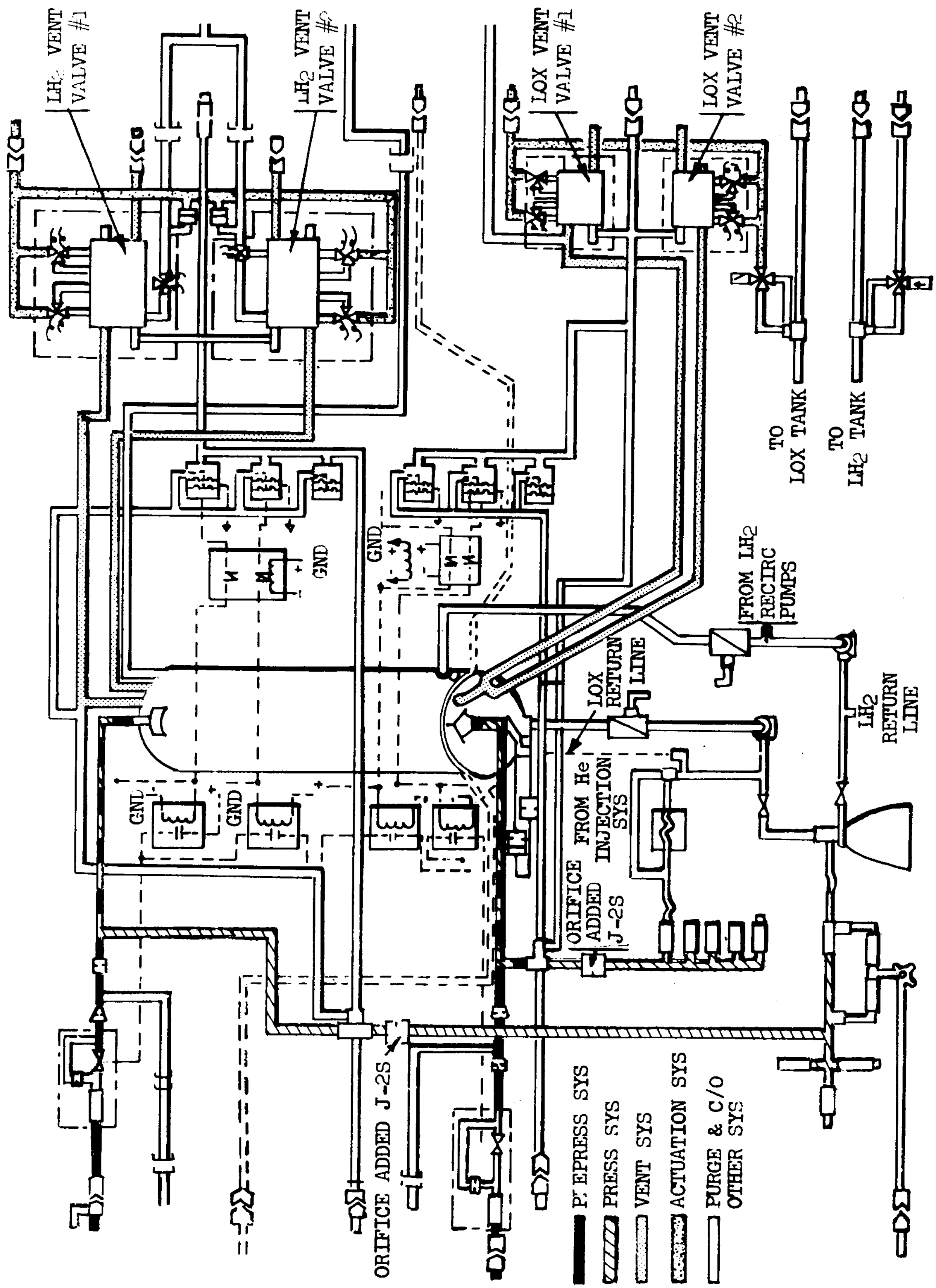


Figure 10.2-68. J-2 Pressurization System

Table 10.2-VI. LOR Mission

System Hardware and Components Affected by J-2S Incorporation into the S-II Stage (Procured Items Only)

Subsystem Affected	Major Hardware and/or Component Affected	Part Number and Quantity Required	Comments
LOX Tank Pressurization System	Pressure Manifold (Engine Isolation Check Valve to Press Reg)	ME272-0005-0018 1	Repressure test
		-0019 1	Requalify
		-0020 1	Repressure test
		-0021 1	Repressure test
		-0022 1	Requalify
		-0023 1	Requalify
		-0024 1	Requalify
		-0026 1	Requalify
	Pressure Regulator	ME284-0161-0010 1	Requalify
LH_2 Tank Pressurization System	Pressure Manifold (Engine Isolation Check Valve to 3-inch Line Through Systems Tunnel)	ME282-0006-0019 1	Repressure test
		-0020 1	Requalify
		-0021 1	Repressure test
		-0022 1	Repressure test
		-0023 1	Requalify
		-0024 1	Requalify
		-0025 1	Requalify
		-0026 1	Requalify
	Pressure Line (Manifold "Tee" to Press Regulator)	ME271-0026-0101 1	Requalify
		-0102 1	Requalify
		-0103 2	Repressure test
		-0109 1	Requalify
	Pressure Regulator	ME284-0161-0005 1	Requalify

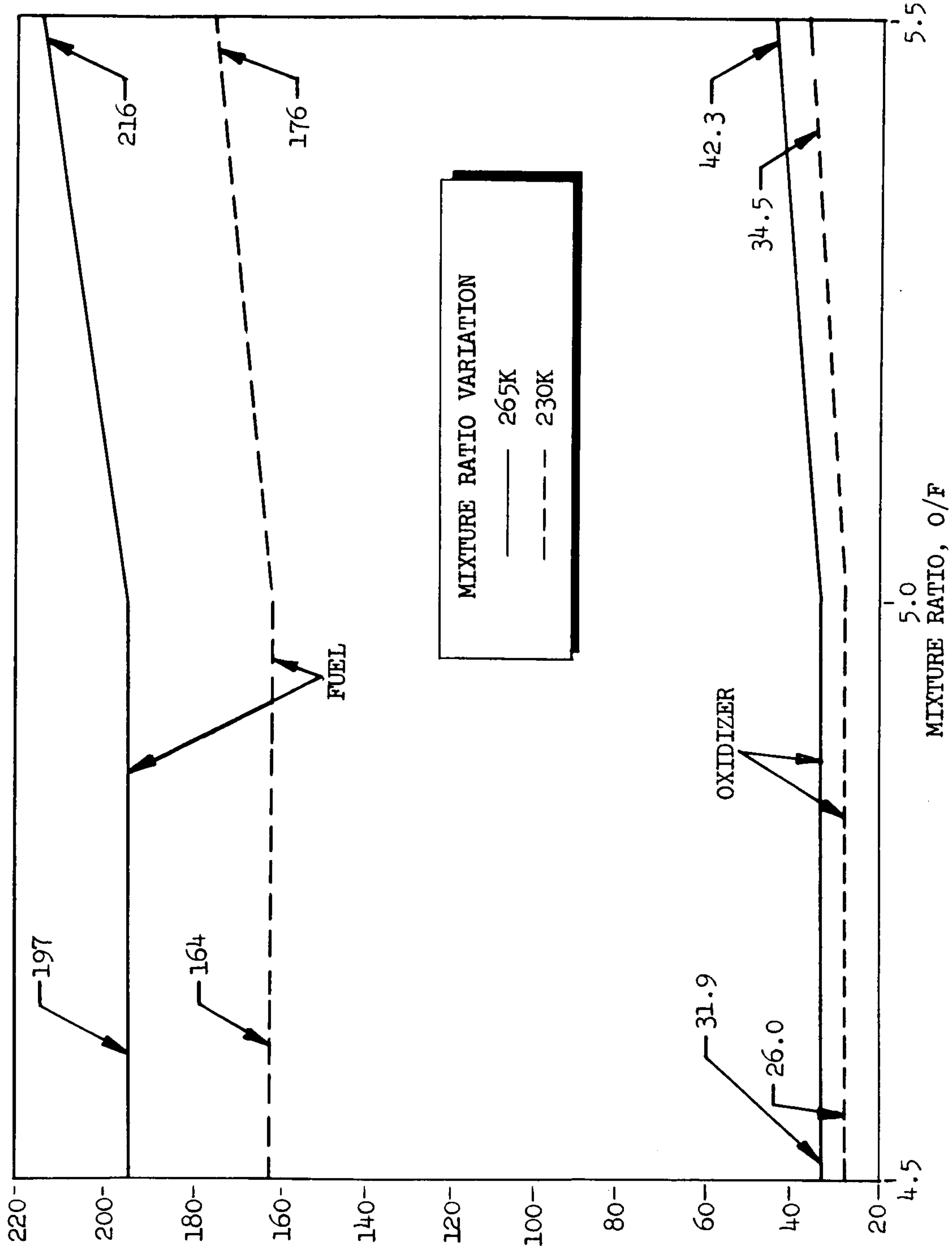


Figure 10.2-69. Mainstage NPSH Requirements

	LOX Tank	LH_2 Tank
Engine inlet temperature at cutoff	164.6 deg (R)	39.0 deg (R)
Vapor pressure at cutoff	16.9 psia	21.65 psia
Fluid head at cutoff (no engine out - 4.7 EMR)	5.6 psi	1.4 psi
Propellant feedline pressure losses (5.5 EMR LOX and 5.0 EMR LH_2)	3.75 psi (472 lb/sec)	2.2 psi (85.6 lb/sec)

The conditions at cutoff are used in establishing tank pressure levels because the conditions at cutoff are more severe.

The NPSH requirements and propellant flow rates for 265,000 pounds were obtained from Rocketdyne document R-7211, J-2S Interface Criteria.

	LOX	LH_2
NPSH at 5.5 EMR	42.3 ft (20.8 psi)	216 ft (6.6 psi)
NPSH at 5.0 EMR	31.9 ft (15.7 psi)	197 ft (6.02 psi)
Engine flow rate at 5.5 EMR	525 lb/sec	95 lb/sec
Engine flow rate at 5.0 EMR	450 lb/sec	88 lb/sec

It will be assumed that the flow rates given in the Rocketdyne document are nominal and do not include tolerances due to engine-to-engine and pump-to-pump variations. The flow rate tolerances for the J-2 engines are 5.5 percent for the fuel and 4 percent for LOX. These tolerances plus the 3.0 pounds per second stage pressurant requirements will also be assumed for the J-2S. Therefore, the propellant feed line pressure losses are as follows:

EMR 5.0	EMR 5.5
LH_2 system delta P = 2.76	delta P = 3.18
LOX system delta P = 3.75	delta P = 5.07

The minimum required ullage pressure is determined by adding the NPSH requirement to the vapor pressure of the fluid in the line and the feedline pressure drop and then subtracting the fluid head from the total.

Table 10.2-VII lists these values for two engine mixture ratios: 5.0 and 5.5 for both propellant tanks. Normal mixture ratio at cutoff will be 4.7, which will result in a satisfactory margin in both tanks.

A one-engine-out mission was also investigated. The only effect of one-engine-out is a 20 percent reduction in the fluid head.

Table 10.2-VII. Ullage Pressure Factors

LH2 TANK				
	5.0 EMR at Cutoff		5.5 EMR at Cutoff	
	No Engine Out	One Engine Out	No Engine Out	One Engine Out
NPSH	6.02 psi	6.02 psi	6.60 psi	6.60 psi
Vapor Pressure at Cutoff	21.65 psia	21.65 psia	21.65	21.65
Propellant Feed-line Losses	2.76 psi	2.76 psi	3.18	3.18
Fluid Head	1.4 psi	1.12 psi	1.4	1.12
Min required ullage pressure	29.03 psia	29.31 psia	30.03 psia	30.31 psi
Min supplied ullage pressure	30.50 psi	30.50 psia	30.50	30.50
Margin	1.47 psi	1.19 psi	0.47	0.19
*Factor of safety	1.27	1.218	1.086	1.035

*Note: By definition, the factor of safety for the LH_2 tank is:

$$SF = \frac{\text{Vapor Pressure at Cutoff} - \text{Initial Bulk Vapor Pressure} + \text{Margin}}{\text{Vapor Pressure at Cutoff} - \text{Initial Bulk Vapor Pressure}}$$

LOX TANK				
	5.5 EMR at Cutoff		5.0 EMR at Cutoff	
	No Engine Out	One Engine Out	No Engine Out	One Engine Out
NPSH	20.8	20.8	15.7	15.7
Vapor Pressure	16.9	16.9	16.9	16.9
Propellant Feed-line Losses	5.07	5.07	3.75	3.75
Fluid Head	5.6	4.48	5.6	4.48
Min required ullage pressure	37.17	38.29	30.75	31.87

Table 10.2-VII. Ullage Pressure Factors (Cont)

LOX TANK				
	5.5 EMR at Cutoff		5.0 EMR at Cutoff	
	No Engine Out	One Engine Out	No Engine Out	One Engine Out
Min Supplied Ullage Pressure	36.00	36.00	36.00	36.00
Margin	-1.17	-2.29	+5.25	+4.13
Max Tank Pressure	42.00 psia	42.00 psia	42.00 psia	42.00 psia

Figure 10.2-70 shows the various study missions altitude versus time predictions. The values in this table were used in determining the predicted ullage pressure in the LH_2 tank during a launch. Figure 10.2-71 is a composite curve showing the predicted ullage pressures that will be available during the S-IC boost portion of the flight. Note that Figure 10.2-71 is based on the use of existing components.

Since the specified temperature of the LOX pressurant gas coming out of the heat exchanger is higher for the J-2S than for the J-2 engine, an analysis was performed to determine the effect this warmer gas would have on the common bulkhead aft facing sheet. The primary consideration is the temperature on the common bulkhead aft facing sheet exceeding the maximum structural design limit. Figure 10.2-72 shows the result of this analysis. The predicted temperature for the common bulkhead aft facing is lower than the maximum structural limit specified. Figure 10.2-73 shows the LOX heat exchanger performance curve provided by Rocketdyne. This heat exchanger curve was used in the predictions for common bulkhead aft facing sheet temperature.

10.2.2.4.2 LEO Mission

a. Coast Pressurization. Earlier studies of the S-II tank pressurization requirements for J-2S engine restart to the idle mode after 45 minutes of coast indicated that no repressurization appeared needed for the hydrogen tank, but that repressurization appeared needed for the oxygen tank, based on an approximate heat balance method and use of Computer Program 6N-992. If the hydrogen ullage gas is vented to supply a propellant settling thrust, the additional pressure loss must be allowed for. Therefore, additional analysis was conducted, including runs with Program 7N-992 with improved heat load inputs.

Figure 10.2-74 shows the engine inlet oxidizer and fuel pressures required for J-2S operation in the idle mode and results of the 7N-992 study plotted at points A, B, and C. Since the large propellant supply lines are designed for mainstage engine operation, a negligible pressure drop exists for the low flow rates required during idle mode operation. Since the propellant acceleration heads are also negligible during the near zero-G coast period, the required engine inlet pressures may be assumed the same as ullage gas pressures in each tank.

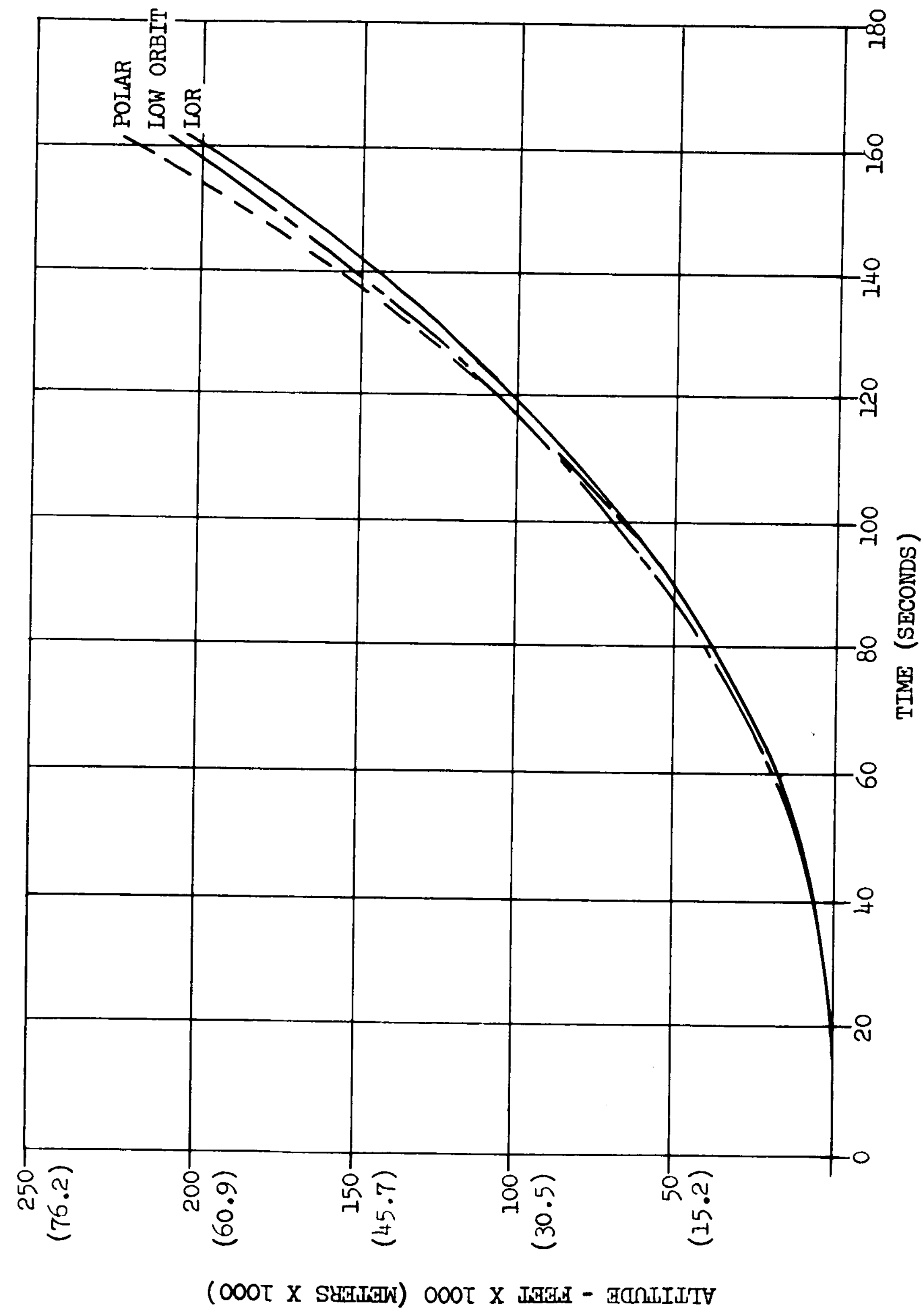


Figure 10.2-70. J-2S Study Missions (Altitude Versus Time)

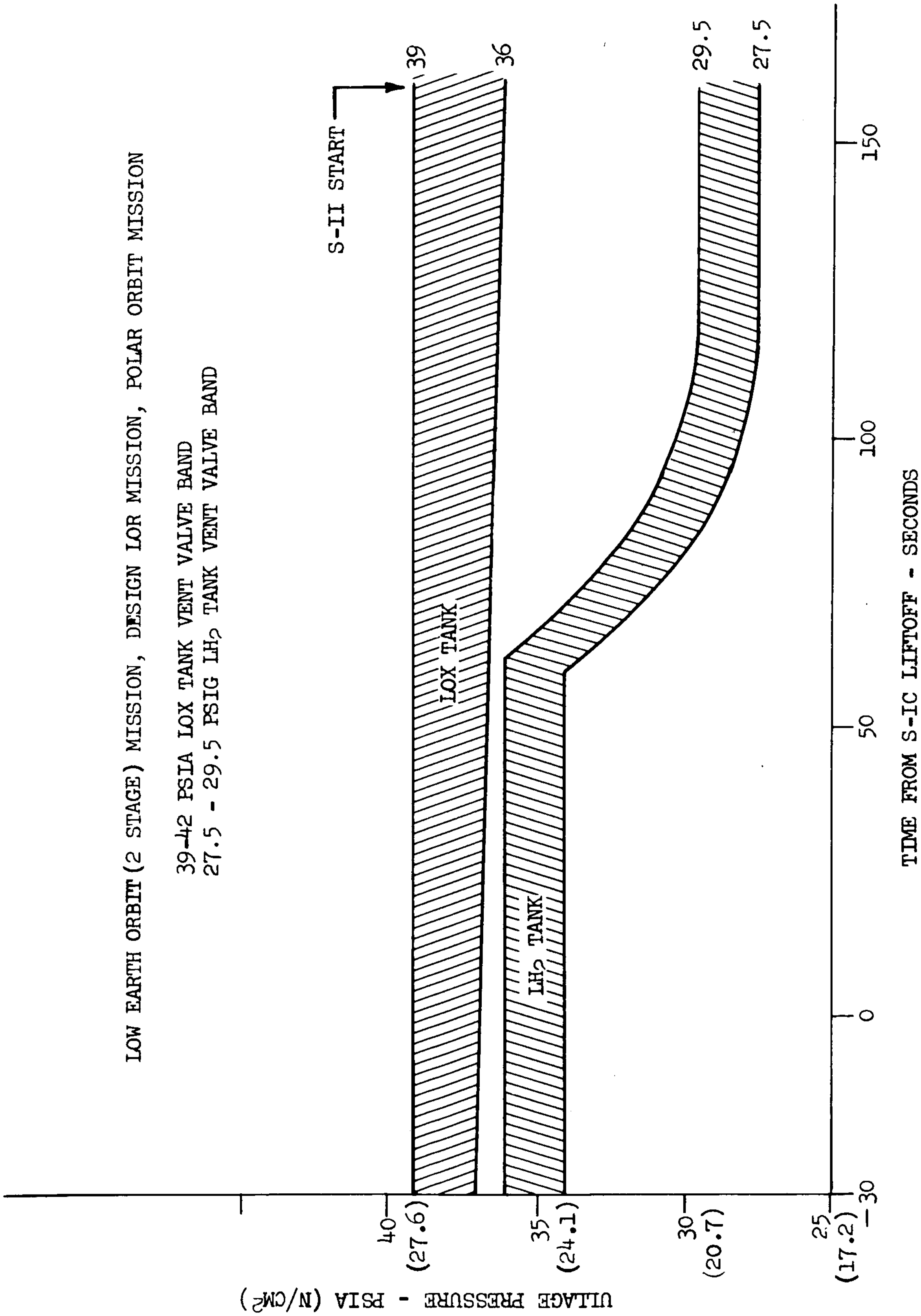


Figure 10.2-71. J-2S/S-II Propellant Tank Ullage Pressure During S-IC Boost

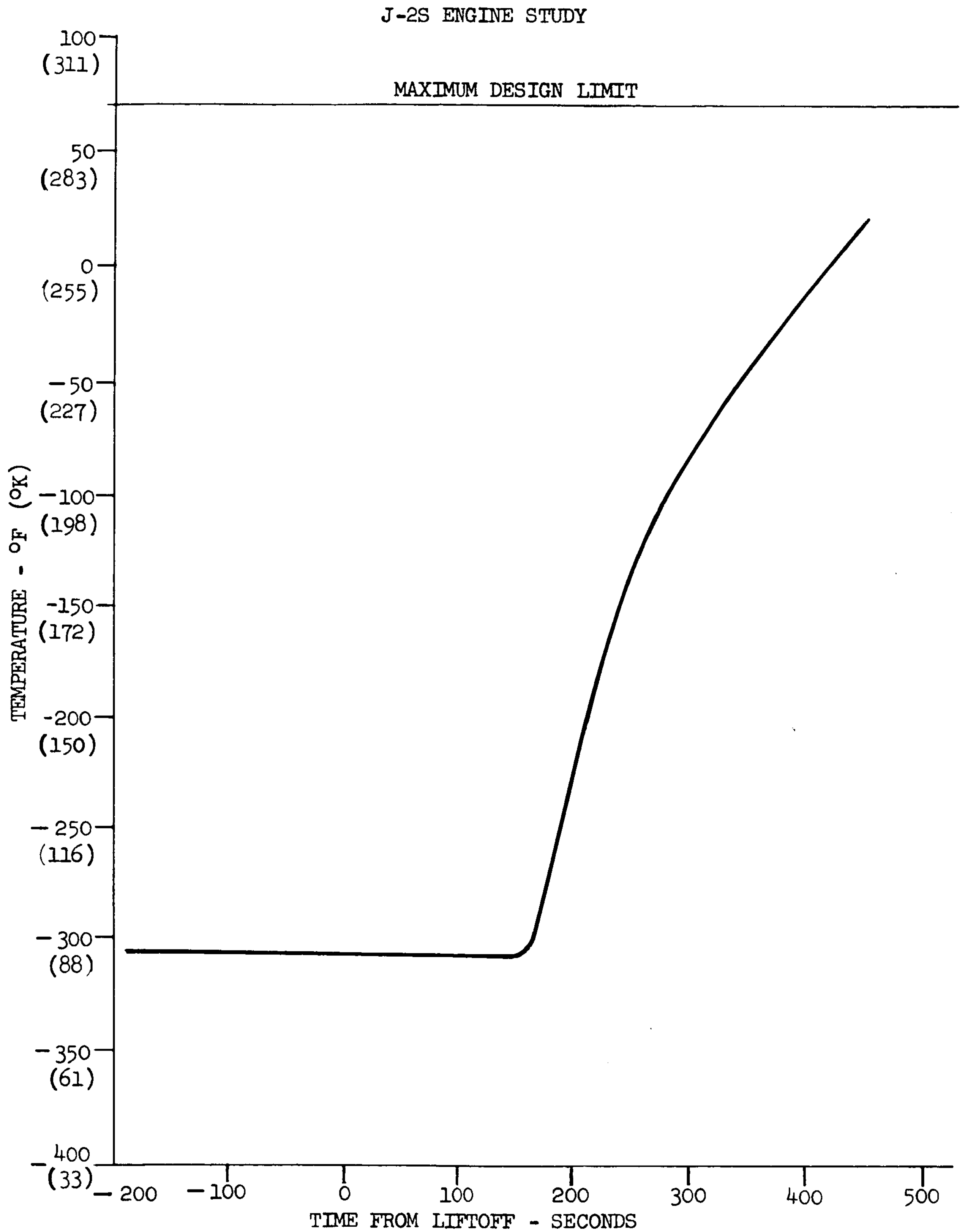


Figure 10.2-72. Common Bulkhead Aft Facing Sheet Temperature at Station 394

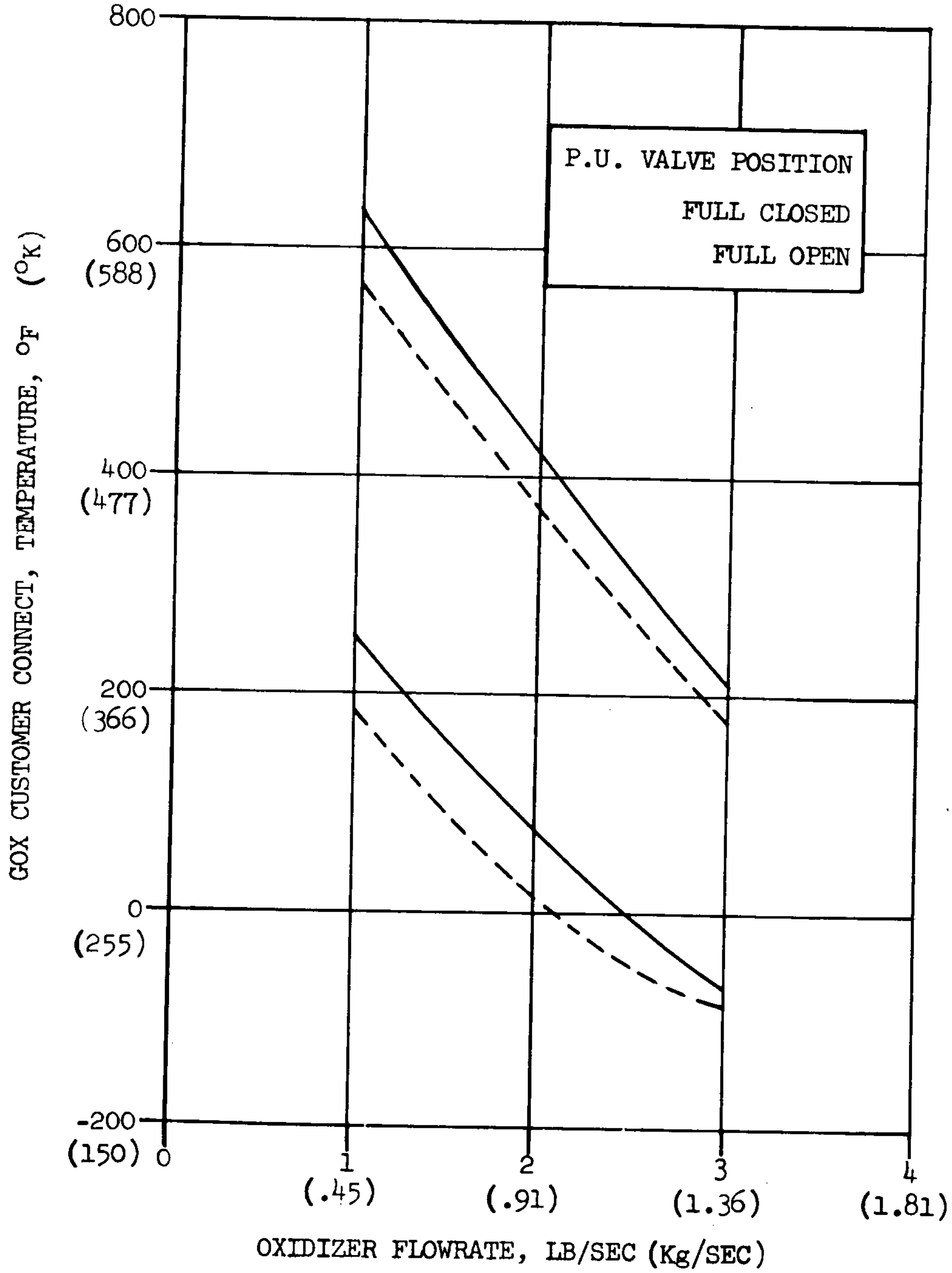


Figure 10. 2-73. Heat Exchanger GOX Temperature Versus Oxidizer Flowrate

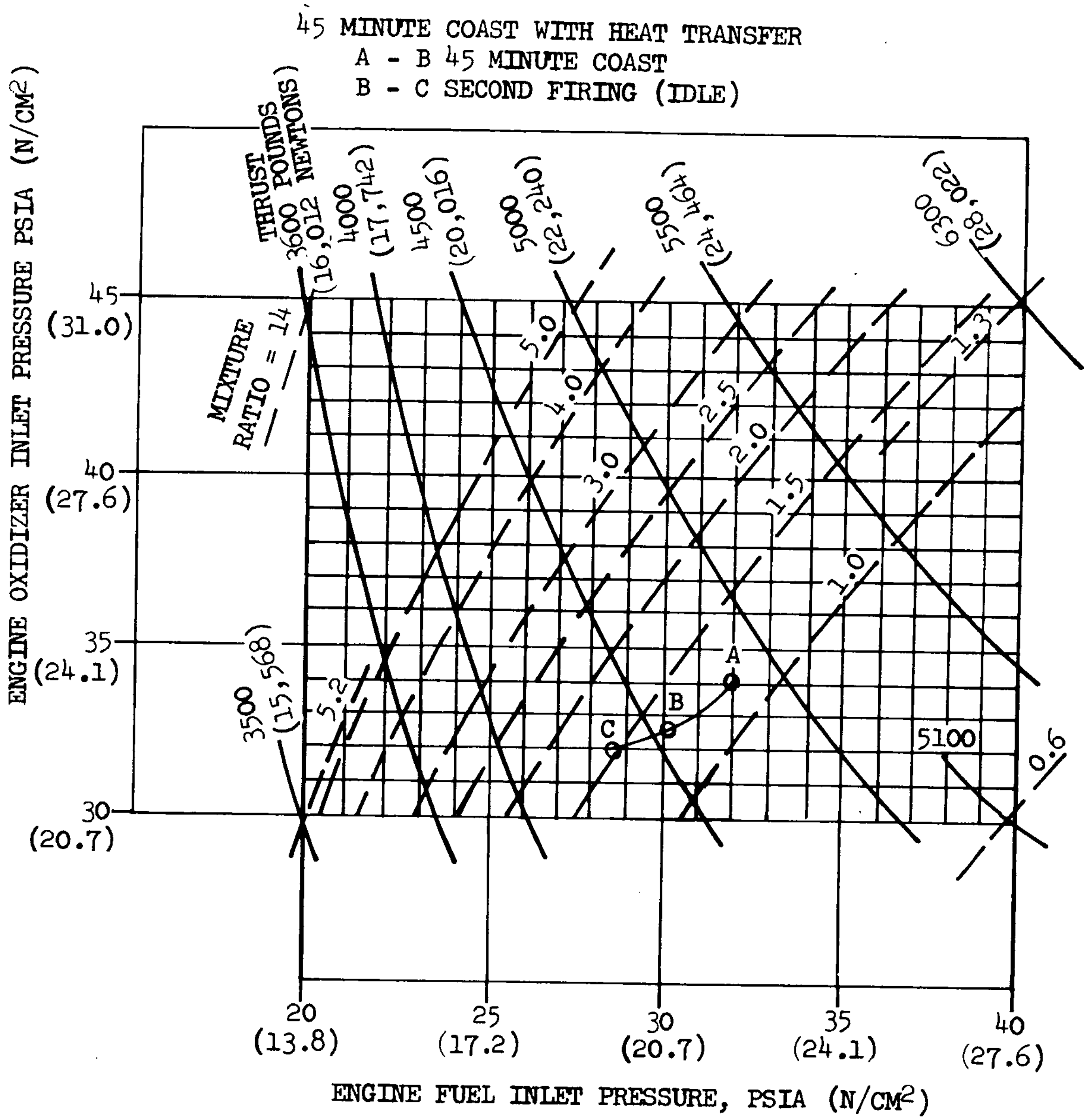


Figure 10.2-74. J-2S/Pressures Versus Restart Requirements

At the termination of S-II stage boost thrust (which includes a period of idle mode operation to reduce cutoff slosh disturbances), the initial conditions for the 45-minute coast period are estimated to be as shown in Table 10.2-VIII. Location of the various heat inputs is shown in Figure 10.2-75 with resulting pressure-time curves from the existing tank pressurization program 6N-992 shown in Figure 10.2-76.

Points A in Figure 10.2-76 show the starting pressures (at mainstage cutoff) of 34 and 32 psia for the LOX and LH_2 tank ullage gases respectively. After coasting for 2700 seconds (45 minutes), the computer program results show the oxygen pressure has dropped to 33 psia (point B') with 3.6 pounds of helium gas makeup added, and to 32.5 psia (point B extrapolated) with no helium makeup. Similarly, the hydrogen ullage gas pressure apparently remained constant, due to the larger heat input (point B'), but because the vehicle will be constantly venting hydrogen gas for the low-g thrust, the pressure actually drops to point B at restart time.

The computer program results show both pressures drop further during the J-2S engine firing time; the oxygen to about 32 psia (point C) and the hydrogen to 28.5 psia (also point C). The pressures at points A, B, and C are plotted in Figure 10.2-74 to show the shift during coast and firing times.

The principal effect of the pressure change is to locate the oxidizer/fuel mixture ratio at about 1.45 in Figure 10.2-74 with the corresponding thrust level averaging about 4400 pounds. The tanked quantities in Table 10.2-VIII reflect this mixture ratio for the 11,000 pounds of propellant needed for circularization, together with typical residuals.

Based on the analysis described above, restart capability of the J-2S engines in the idle mode will exist after a 45-minute coast period without the necessity of repressurization of either the LH_2 or the LOX tank, assuming a low-G settling thrust is provided.

It is also concluded that the hydrogen ullage gas may be used as propellant for the propellant settling requirements.

An additional study was conducted to extend the coast period from the previous 45 minutes to 2 hours. In addition, the tank pressurization computer program 7N-992 was used with and without heat transfer to the ullage gas for both the LH_2 and LOX tanks. Hydrogen ullage gas again was used to provide the required low-g settling thrust during coast, but this time with the correct vent area to eliminate superimposing the result of a separate calculation of its effect on the remaining ullage gas pressure.

Figure 10.2-77 shows the available ullage gas feed pressure plotted on the Rocketdyne diagram for the required J-2S engine feed pressures. The two cases considered were:

1. Without heat transfer to the ullage gas, comprising the most severe (and conservative) condition

2. With heat transfer to the ullage gas

Table 10.2-VIII. Tank Conditions, LEO Mission

Parameter	MECO	Restart	End
Time, seconds	0	2,700	2,837
LH_2 Tank			
Ullage gas pressure, psia	32.0	30.0	28.5
Average gas temperature, deg R	159	139	131
Volume of gas, ft^3	35,500	35,600	36,600
LH_2 Propellant (including residuals), lb	8,000	7,600	3,100
Ullage gas weight (including 20 lb helium), lb	1,340	1,440	1,450
LH_2 temperature, deg R	38.5	39.5	41.0
GH_2 vented for thrust, lb	0	150	151
LOX Tank			
Ullage gas pressure, psia	34.0	32.5	32.0
Average gas temperature, deg R	310	288	285
Volume of gas, ft^3	12,350	12,350	12,450
LOX propellant, including residuals, lb	11,000	10,900	4,400
Ullage gas weight, (including 80 lb helium), lb	3,490	3,590	3,600
LOX temperature, deg R	165	167	169
Engine Performance			
Number of engines	2	5	5
Thrust, each engine, lb	7	4,500	4,300
Specific impulse (vac), sec	125	280	280
Mixture ratio, by weight	0	1.3	1.6
Assumed Heat Inputs, Btu/hr			
To LH_2 through LOX bulkhead		10,000	
To LH_2 through wall		22,000	
To LH_2 through boattail		42,000	
To LH_2 tank sidewall		104,000	
To LOX and GOX through aft bulkhead		6,000	

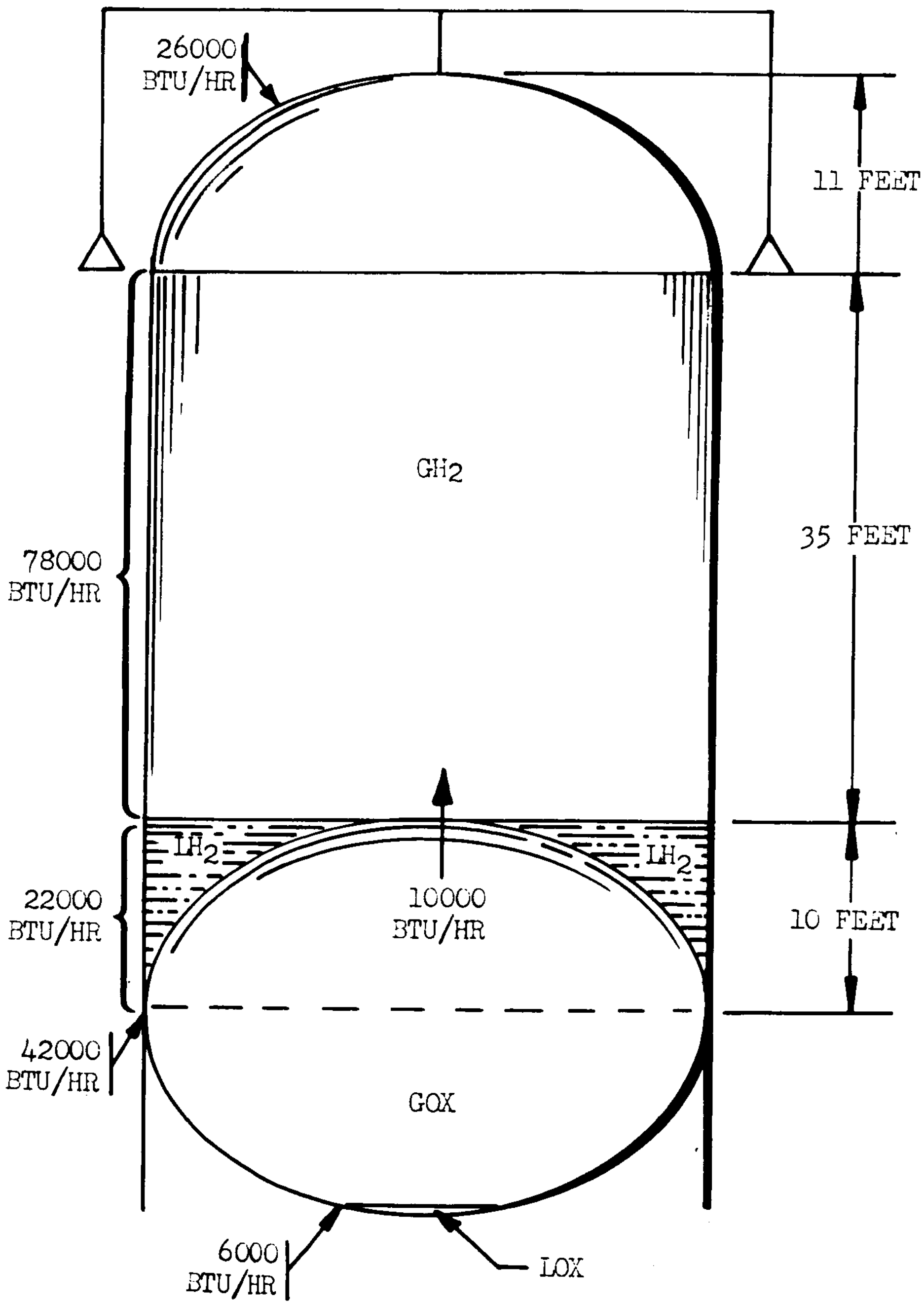


Figure 10.2-75. J-2S Heat Transfer Paths

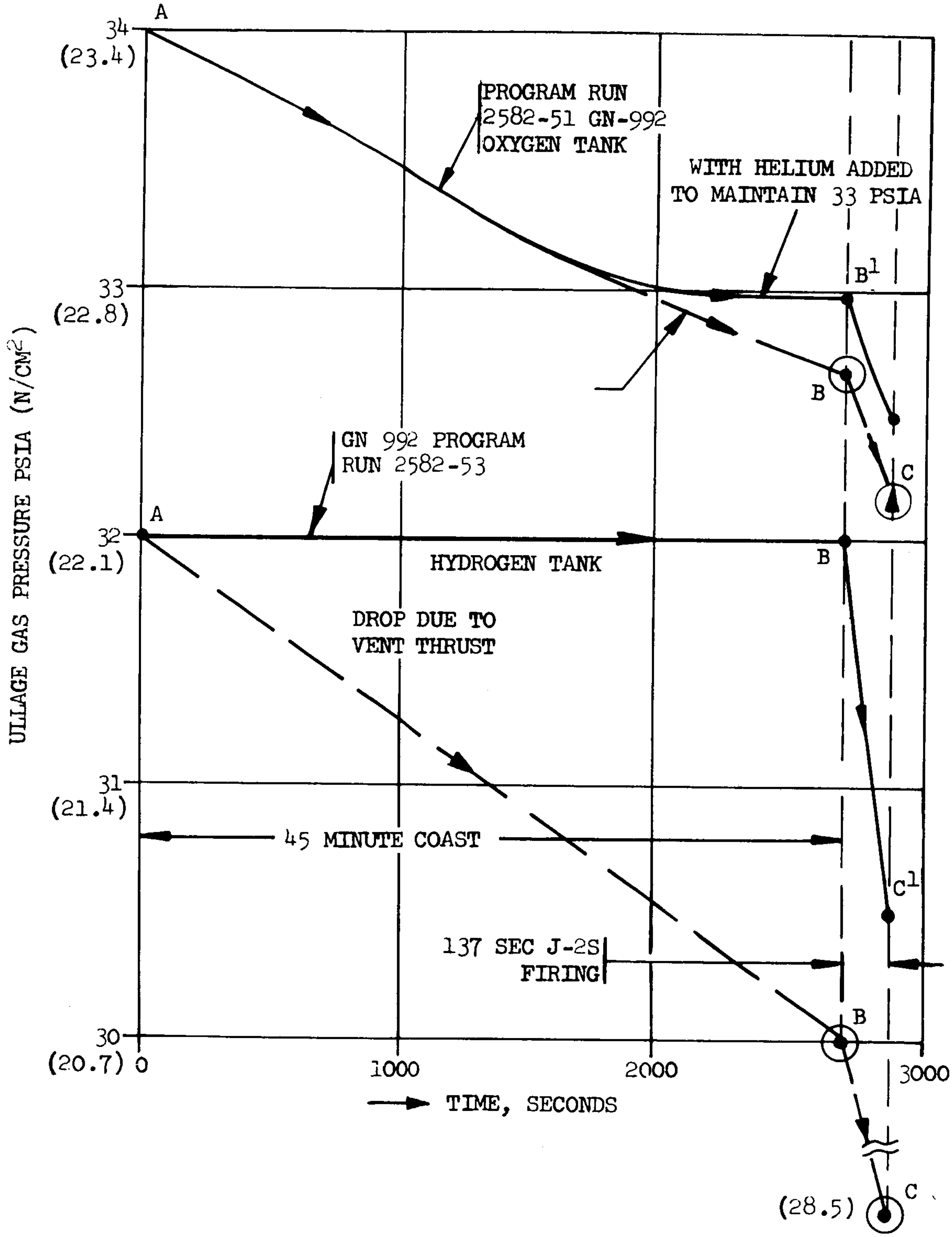
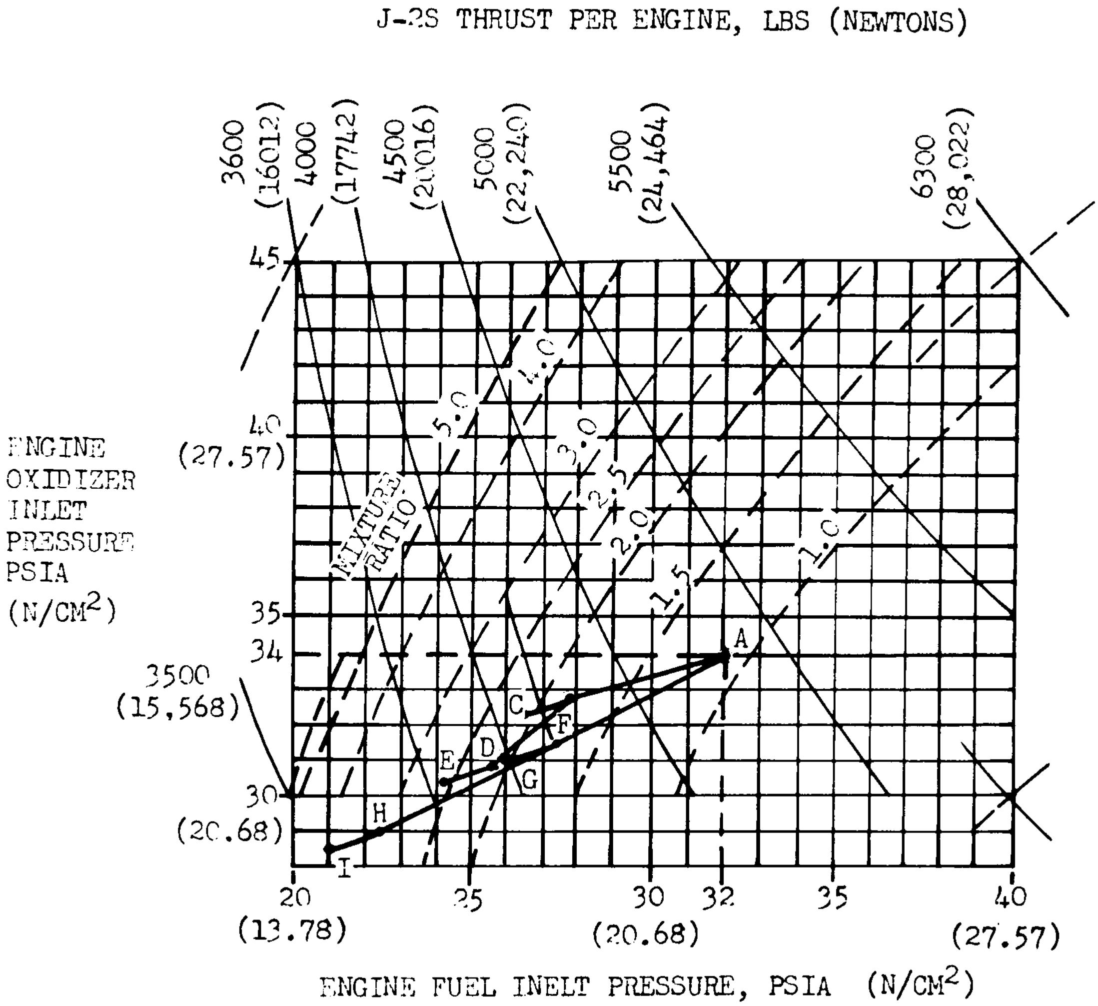


Figure 10. 2-76. J-2S Pressure Versus Time (45 Minutes)



TIME PERIOD	WITH HEAT TRANSFER	WITHOUT HEAT TRANSFER
45 MIN. COAST	AB	AF
2ND BURN-IDLE MODE	BC	FG
2 HOUR COAST	ABD	AFH
2ND BURN-IDLE MODE	DE	HI

Figure 10. 2-77. J-2S Pressure Versus Restart Requirements (45 Minutes and 2 Hours)

In each case, a total hydrogen ullage gas vent area of 0.16-square-inches was provided during the entire coast period to supply an initial $2 \times 10^{-5} g_o$ acceleration for propellant control at 32-psia ullage pressure and 0.057 pound per second flowrate. After 45-minute and 2-hour coast periods, this acceleration reduces to 1.7×10^{-5} and 1.4×10^{-5}, respectively, which are still ample compared to $4 \times 10^{-6} g_o$ required to maintain settled propellants.

No changes were made in the estimated tank conditions at mainstage cutoff, J-2S restart, and end of firing from those shown in Table 10.2-VIII.

Table 10.2-IX lists the available hydrogen and oxygen tank ullage feed pressure, and related engine thrust and oxidizer/fuel mixture ratio data consistent with the engine performance curves of Figure 10.2-77.

Table 10.2-X shows the final computer program printout data for ullage gas/pressures, weights, evaporation rates, total evaporation, vent rate and total weight vented, for time, 0, 2700, 5000, 2700, and 7370 seconds, with and without heat transfer to the ullage gas.

The analysis indicates that the ullage pressure will be sufficient for restart of the J-2S engines after either 45 minutes or 2 hours of coasting when heat transfer to the ullage gas is considered.

Table 10.2-IX. Tank Pressures, Thrust and Mixture Ratio Versus Time (LEO Mission)

Location Symbol (Figure 10.2-74)	Time (Seconds)	Tank Ullage Pressure, psia		Total Thrust (lb)	Mixture Ratio O/F(wt)	Remarks
		LH_2	LOX			
A	0	32.0	34.0	7	(GH_2)	
B	2700	27.9	32.8	4306	1.8	Case (1) with heat transfer to gas
C	2845	26.5*	32.3	4106	2.2	
D	7200	25.5	30.9	3905	2.3	
E	7360	24.1	30.4	3705	2.7	
F	2700	27.4	31.5	4206	1.8	Case (2) without heat transfer to gas
G	2848	26.0*	31.0	4006	2.1	
H	7200	22.4	29.0*	3555	3.2	
I	7373	21.0	28.5*	3505	4.0	

*Extrapolated from program printout data

Table 10.2-X. Tank(s) Parameters/Conditions Versus Time
LEO Mission

Run No.	Parameters	Time, Sec → 0	2700	5000	7200	7370	Remarks
LH_2 TANK							
0541-61 0541-63	PU psia	32.0 32.0	27.4 27.9	24.6 26.2	22.4 25.5	21.1 24.1	Without ΔHg With ΔHg
0541-61 0541-63	W_G lbs	1133 1133	1222 1223	1256 1258	1276 1279	1275 1278	Without ΔHg With ΔHg
0541-61 0541-63	WDOTE lb/sec	0.116 0.116	0.067 0.068	0.0546 0.0555	0.0477 0.0490	0.0144 0.0140	Without ΔHg With ΔHg
0541-61 0541-63	WE lbs	0 0	229 230	367 370	479 483	484 489	Without ΔHg With ΔHg
0541-61 0541-63	MDOTV lb/sec	0.056 0.056	0.048 0.048	0.0434 0.0436	0.0399 0.0412	0.0380 0.0392	Without ΔHg With ΔHg
0541-61 0541-63	WV lbs	0 0	140 140	245 245	336 337	343 344	Without ΔHg With ΔHg
LOX TANK							
0541-62 0541-51	PU psia	34.0 34.0	31.5 32.5*	30.1 31.6*	29.0* 30.9*	28.5* 30.4*	Without ΔHg With ΔHg
0541-62 0541-51	W lbs	5081 5154	5080 5155*	5078 -	5090 -	(4500)X -	Without ΔHg With ΔHg
0541-62 0541-51	WE lbs	0 0	-1.6 1.2	-2.9 -	-4.2 -	-12.3 -	Without ΔHg With ΔHg

*Extrapolated; program "escort" error needs debugging.

Assuming that LH_2 tank vent valve actuation could be detrimental to the mission, two methods of preventing this occurrence were considered. A second assumption was that heat rates to the hydrogen propellant may be high enough to cause this pressure increase. The methods considered were vent valve lock-up and elimination of step pressurization. Elimination of step pressurization was selected because of its simplicity compared to the hardware and controls for locking of the vent valve. Without step pressurization, the final LH_2 tank pressure will be approximately 3 psi less than shown on Figure 10.2-77 points C and E, which remains within the performance limits when heat transfer to the tank exists.

b. Propellant Settling Thrust System (Balanced Vent). To maintain propellants at the tank outlets during the coast period and at idle mode restart, a settling thrust system will be added to the stage. The analysis of Figure 10.2-77 utilized a vehicle acceleration to local gravity (a/g_o) ratio of 2×10^{-5} to provide this propellant orientation. With an assumed vehicle weight of 350,000 pounds, the required thrust is:

$$F = \frac{a}{g} W = 7.0 \text{ lbf}$$

or 3.5 pounds each for two axially directed balanced nozzles.

A thermal analysis has been conducted of the LH_2 tank during the proposed flight profile and the results show that the LH_2 ullage may be used as a gas source to provide the settling thrust. Assuming isentropic expansion from an average ullage gas temperature of 149 R to the solid hydrogen temperature of 25 R, the nozzle efficiency

$$= 1 - Te/T_c = 0.832$$

and the thrust coefficient is

$$C_F = \emptyset$$

$$= 0.97 \times 1.62 \times .832$$

$$= 1.43$$

making the throat area of each nozzle

$$A_t = F/C_F\,Pc = 3.5/1.43 \times 31^{29}$$

$$= 0.079 \text{ in}^2$$

with a throat diameter of 0.32 inch. The area expansion ratio corresponding to the temperature ratio is approximately 5 to 1 resulting in an exit diameter of 0.72 inch. Figure 10.2-78 illustrates the nozzle configuration.

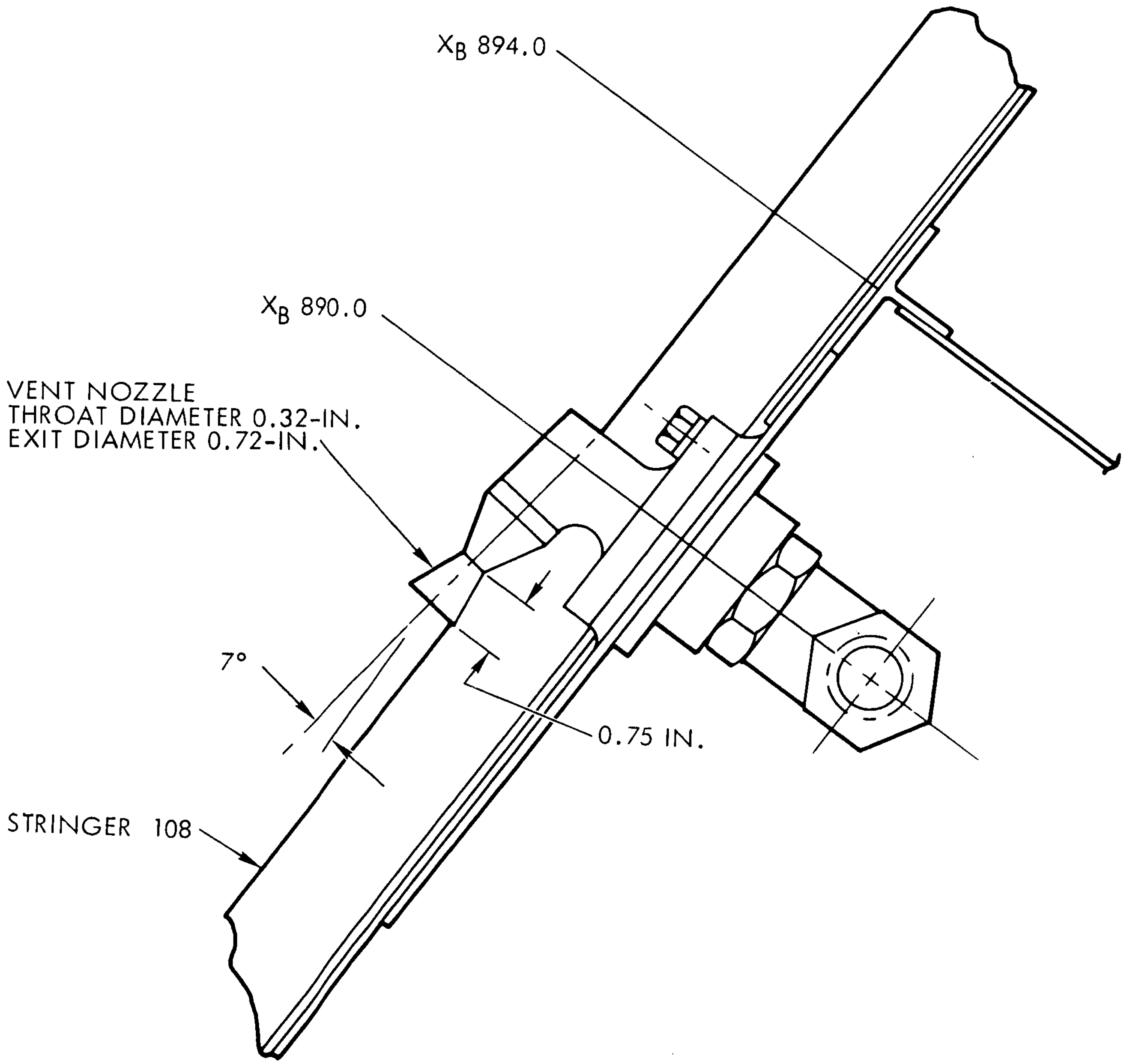


Figure 10.2-78. Balanced Vent Thrust Nozzle Configuration

The flowrate required to provide necessary thrust using an estimated specific impulse of 125 seconds is 2 x 3.5/125, or 0.056 pounds per second. Total vented hydrogen in 2700 seconds is then about 150 pounds. (Net increase of gas mass at restart with about 250 pounds evaporated LH_2 is thus 100 pounds.) A computation of the expansion of remaining ullage gas to replace the lost volume indicates a total pressure drop of about 2 psi as shown at point B on Figure 10.2-74.

The propellant settling thrust system is shown schematically in Figure 10.2-79. The system consists of two parallel mounted normally closed solenoid valves, instrumentation ports, two nozzles mounted on the stage external skin, interconnecting tubing and fittings, and a purge system including two check valves.

Figures 10.2-80 and 10.2-81 show the proposed LH_2 Tank propellant settling thrust system installation. The system consists of two normally-closed solenoid valves mounted on the forward skirt at X_B884 and 27 degrees 15 minutes clockwise from position IV, and two diametrically opposed exhaust nozzles mounted on the forward skirt 0 degrees 30 minutes counterclockwise from position I and III at X_B890. The nozzles will be connected to the valves through a series of 1-inch diameter tubes. A 1.50-inch diameter supply line including flex hoses to compensate for the forward bulkhead thermal excursions connects the valves to the forward bulkhead at an unused electrical port located at X_B935.18, R_{BHD}110.47 and θ = 260 degrees 45 minutes.

A helium purge line is connected from a tee in the existing LH_2 tank vent valve actuation system to each side of the solenoid valves. This arrangement will provide purging both upstream and downstream of the valves to eliminate hazardous conditions during ground operations. The following configurations were also studied:

1. A thrust system consisting of the two diametrically opposed valves mounted on the forward bulkhead and connected to the exhaust nozzles with flex hoses was examined and discarded as one valve failure would create an unbalanced vent system.

2. A third thrust system was investigated which consisted of two valves mounted on the forward bulkhead manhole cover and two diametrically opposed exhaust nozzles mounted on the forward skirt. The nozzles are connected to the valve through a series of 1-inch diameter tubing mounted on the forward bulkhead. This system was eliminated because it required breaking into the thrust system to remove the manhole cover, produced a colder environment for the valves, and exceeded the S-II forward ICD line of constraint.

10.2.2.5 Valve Actuation System

10.2.2.5.1 LOR Mission

a. Functional Changes. With removal of the recirculation systems, the inflight valve actuation system function is reduced to that of prevalve actuation at engine cutoff. Those portions of the system required to provide actuation pressure to the recirculation system valves will be deleted.

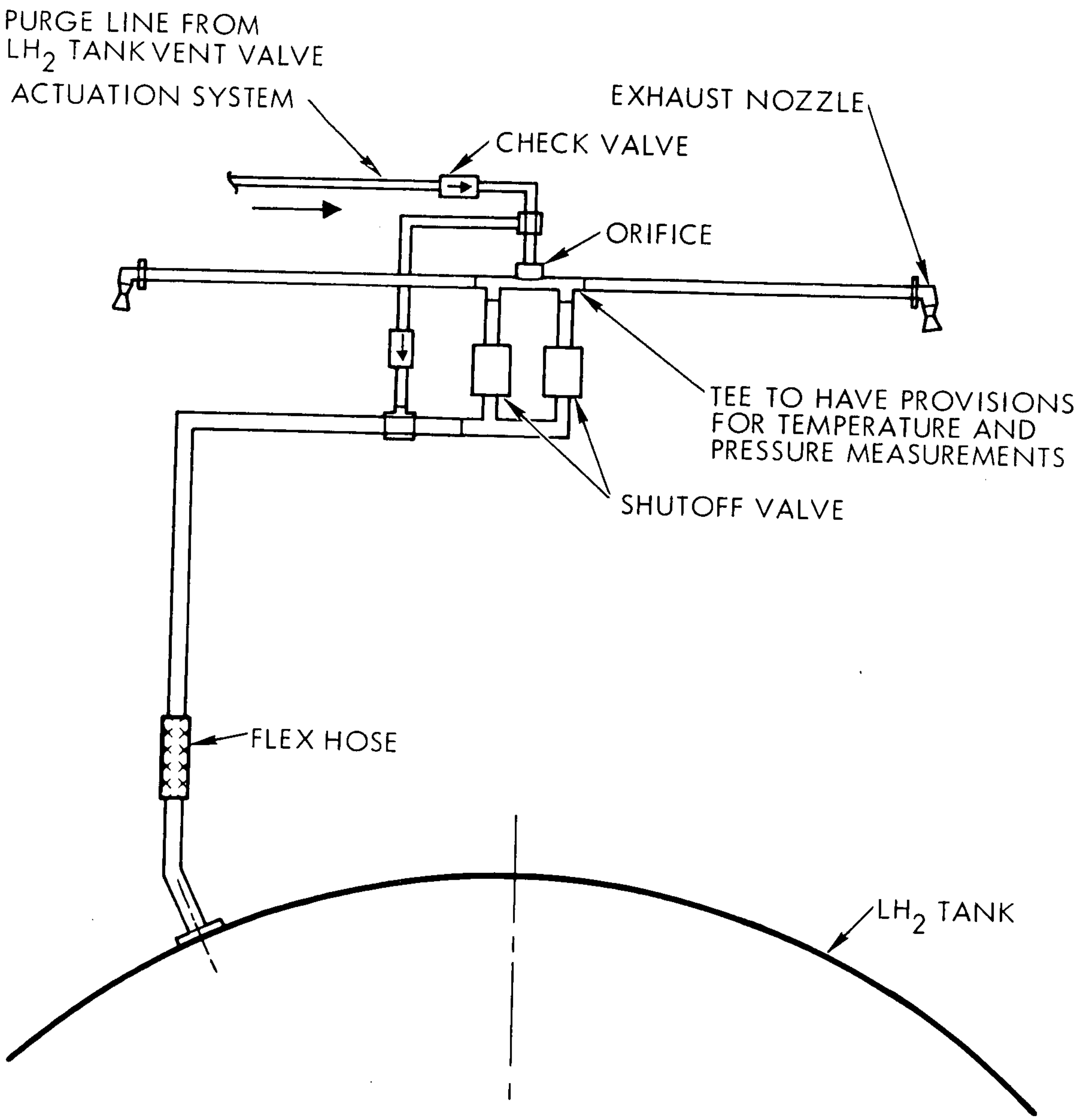


Figure 10.2-79. Propellant Settling Thrust (Balanced Vent) System (Schematic)

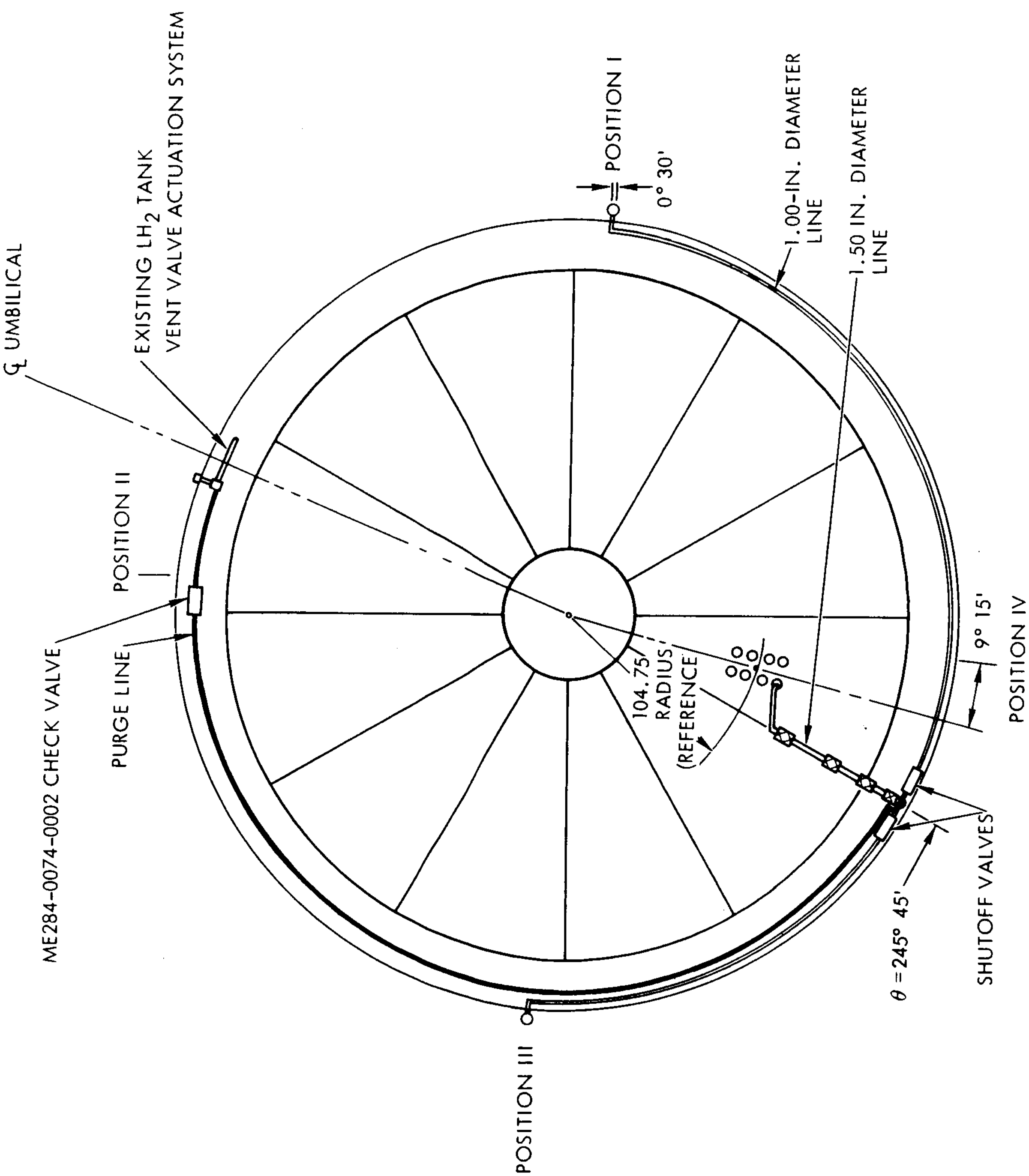


Figure 10. 2-80. Propellant Settling Thrust System (Plan View)

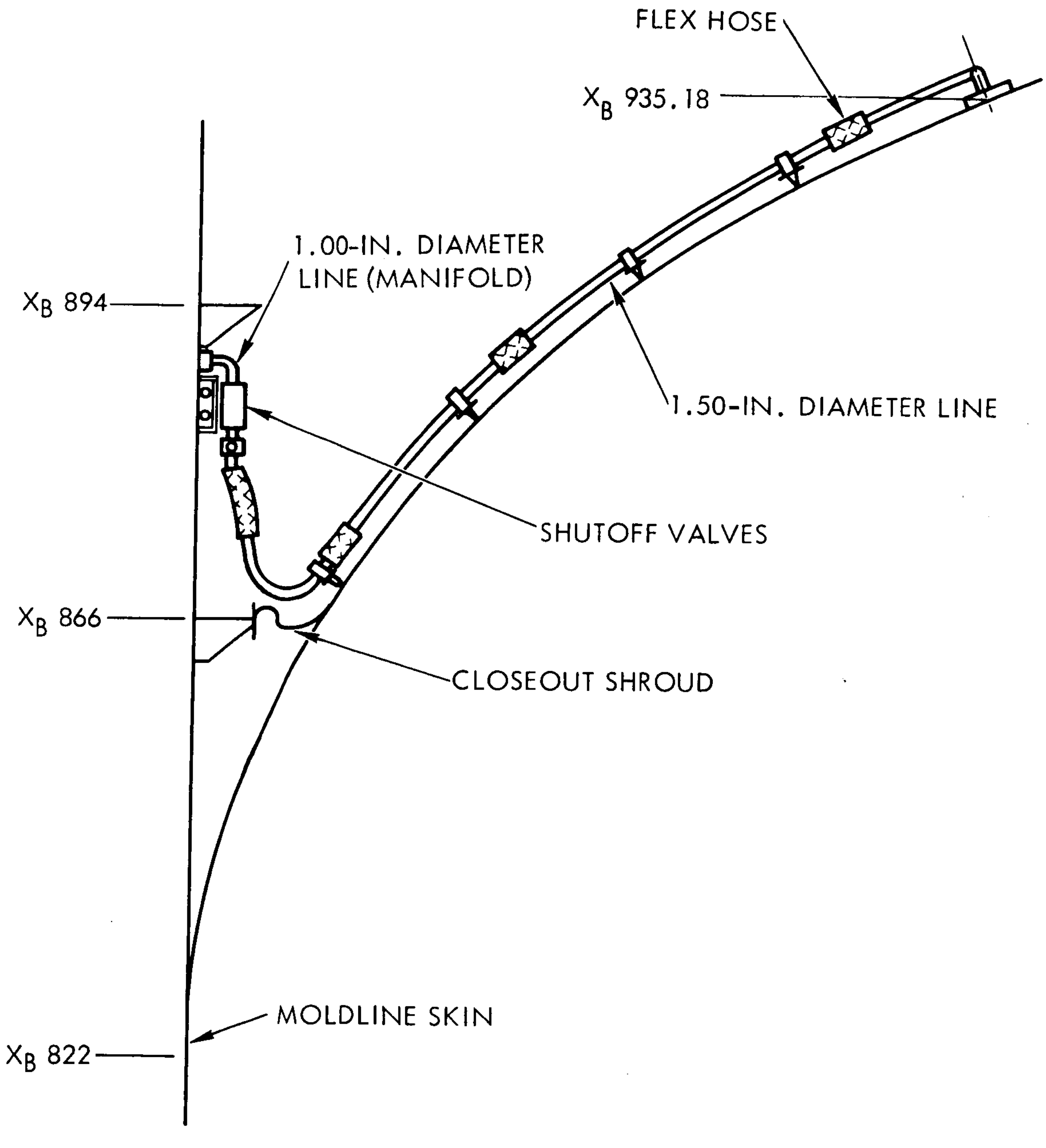


Figure 10.2-81. Propellant Settling Thrust System (Elevation View)

b. System Description. A schematic of the revised prevalve actuation system is shown in Figure 10.2-82. The new system will utilize the existing 1.5 cubic foot helium bottle. However, the system will only be pressurized to 725 ± 25 psig so that the pressure regulator will not be required. The bleed solenoid and one relief valve will be retained. The helium injection system disconnect will be utilized instead of the original valve actuation system disconnect because it is located close to the system components and allows for a cleaner system package. A pressure transducer (DXXXX) is added for static firing evaluation purposes.

Figure 10.2-83 illustrates the routing of the prevalve actuation system for use with the J-2S engine. The actuation lines extend from stage umbilical panel 3A (A7736) to the helium storage bottle located at Fin B, then to the LOX prevalve solenoids located at XB 112 and LH_2 prevalve solenoids at XB 167, and finally from the respective solenoids to the LOX prevalves on the LOX sump and to the LH_2 prevalves in the LH_2 feedlines at XB 315. The lines are constructed of CRES tubing per MB0160-007 and are connected together with B-nut type fittings. No requalification of components will be required.

The GSE S7-41 servicing console helium supply pressure to the stage valve actuation system shall be changed from 3000 + 250/-0 psig to 725 ± 25 psig.

Figure 10.2-84 illustrates the original valve actuation system. The following list of components are deleted.

Schematic Number	NR Specification	Description	Number Required
P45	ME273-0013-0014	Recirculation System Helium Fill Disconnect	1
	ME273-0013-0005	Ground Half of P45	1
P47	ME284-0158-0003	Recirculation System Actuation Regulator	1
P51	ME284-0159-0002	LOX Return Line Solenoid Valve	1
P53	ME284-0159-0001	LH_2 Recirculation Pump Valve and Return Line Valve Solenoid Valve	1
P66	ME284-0270-0001	LOX Recirculation Actuation Relief Valve	1
P67	ME284-0294-0001	LH_2 Recirculation Actuation Relief Valve Test Connect	1

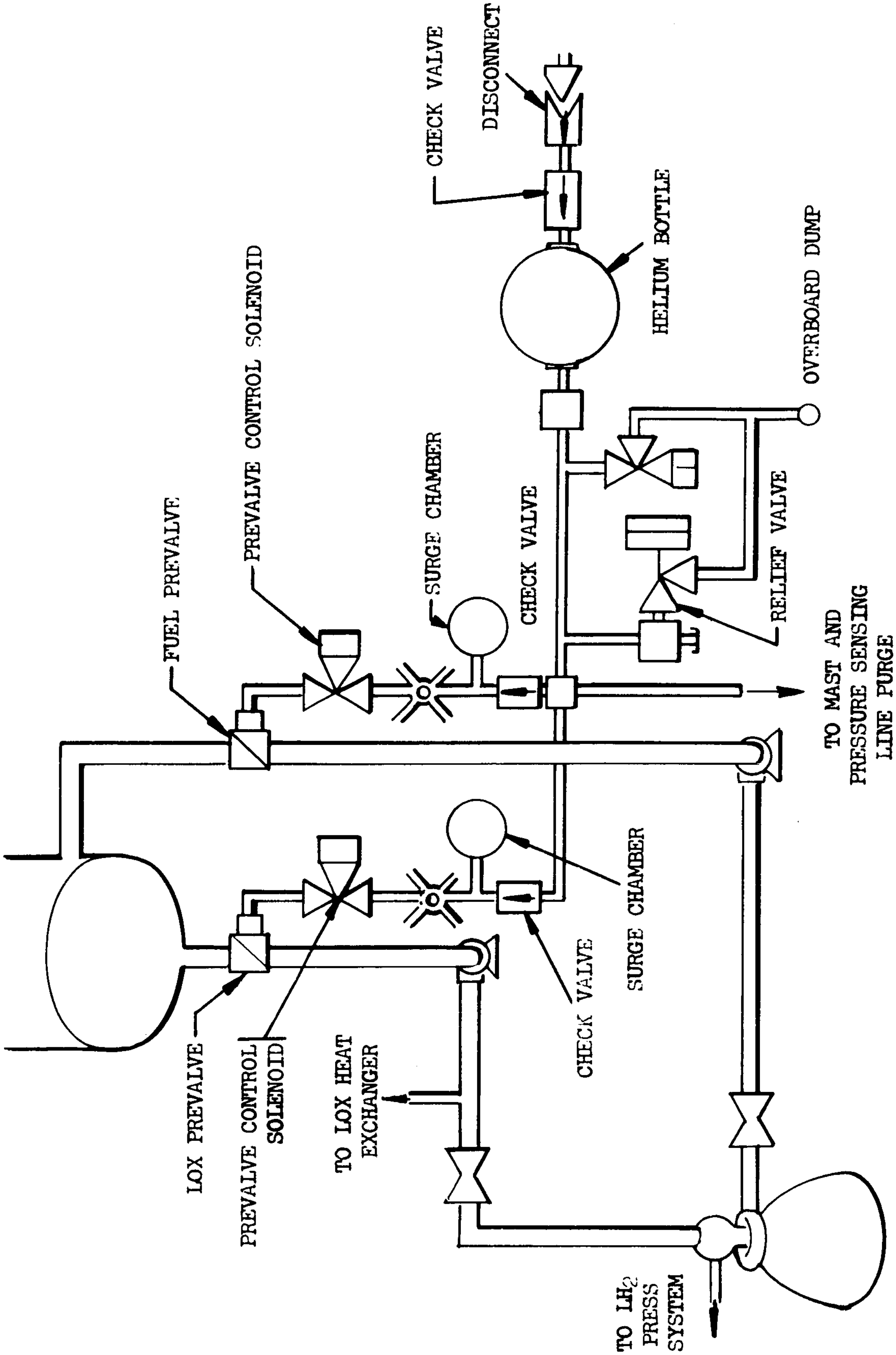
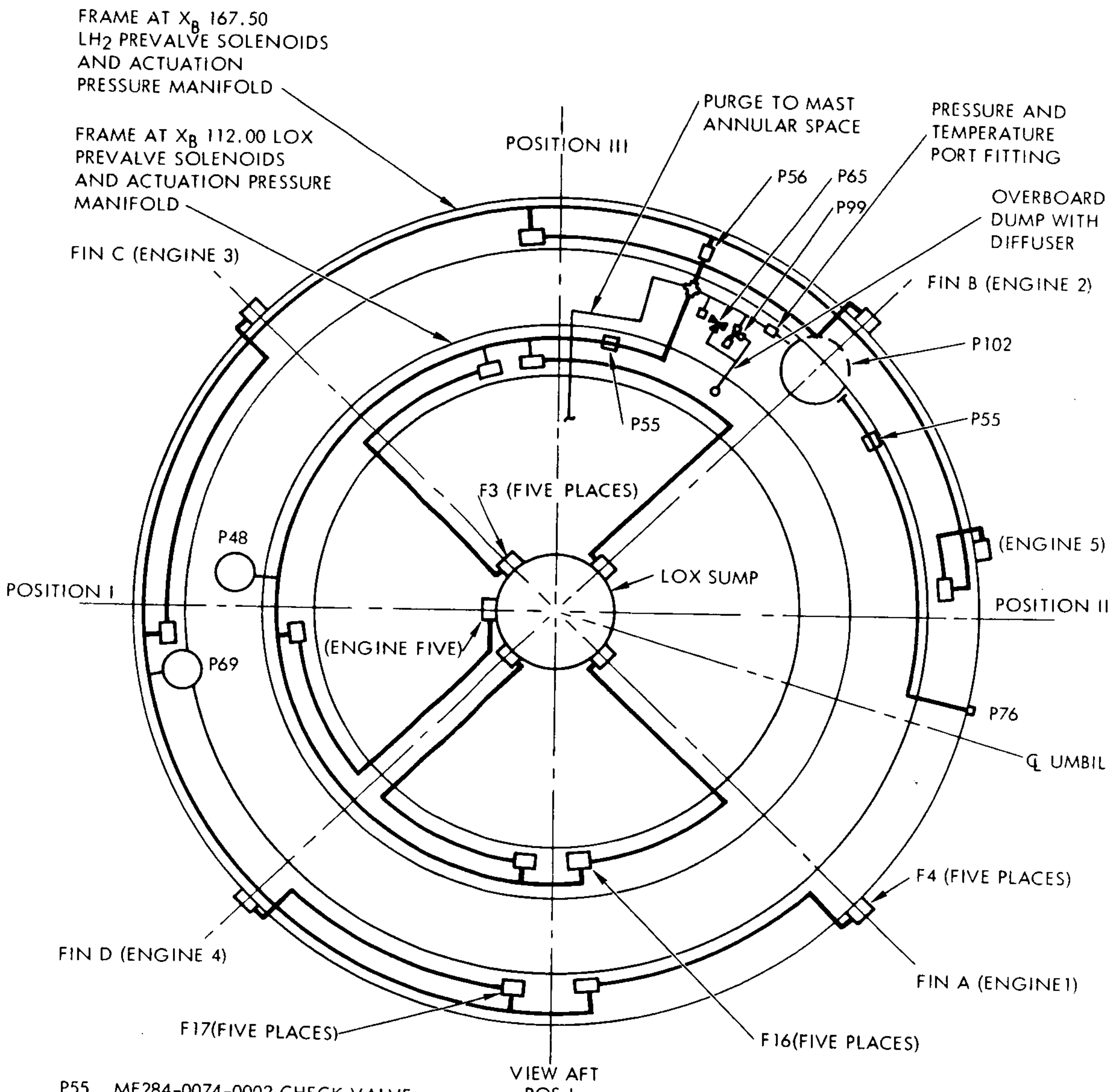


Figure 10.2-82. Engine Valve Actuation System



P55	ME284-0074-0002 CHECK VALVE
P65	ME284-0270-0001 RELIEF VALVE
P76	ME144-0010-0011 DISCONNECT
P99	ME284-0159-0003 BLEED SOLENOID
P102	ME282-0010-0002 HELIUM RECEIVER
P48	ME282-0036-0002 SURGE CHAMBER (LOX SYSTEM)
P56	ME284-0074-0002 CHECK VALVE
F16	ME284-0356-0001 LOX PREVALVE SOLENOIDS
F17	ME248-0356-0001 LH_2 PREVALVE SOLENOIDS
F3	LOX PREVALVE
F4	LH_2 PREVALVE
P69	ME282-0036-0002 SURGE

Figure 10.2-83. Engine Valve Actuation System (Aft View)

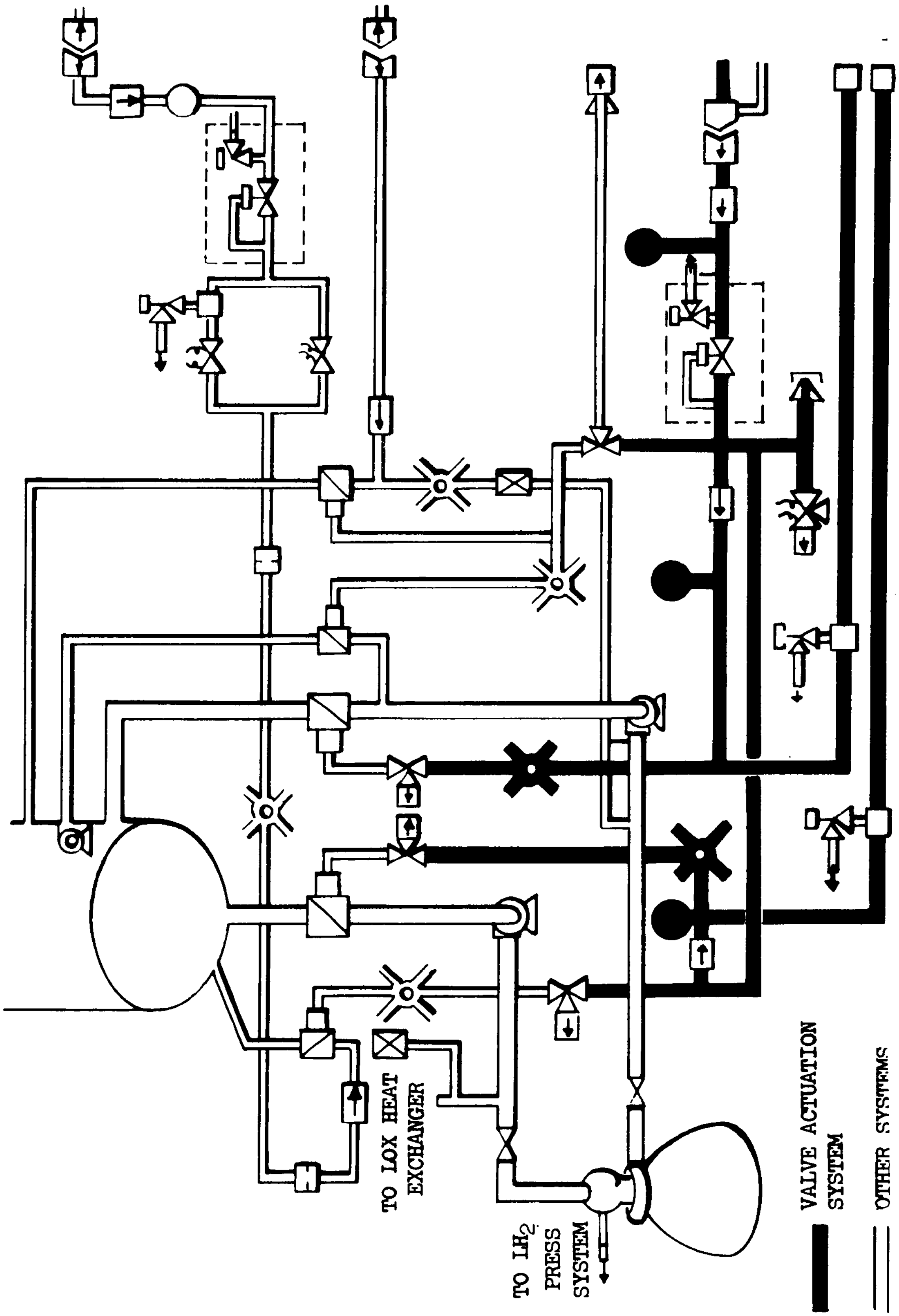


Figure 10.2-84. Existing Valve Actuation System, S-II With J-2 Engine

Schematic Number	NR Specification	Description	Number Required
P68	ME284-0294-0001	LOX Recirculation Actuation Relief Valve Test Connect	1
	ME144-0010-0012	Recirculation System Emergency Actuate Disconnect (LH_2)	1
P72	ME144-0010-0001	Ground Half of P73	1
P94	ME284-0294-0001	Recirculation System Emergency Actuate Connect	1
P105	ME284-0353-0001	Emergency Actuate Check Valve	1

c. Analysis of Change. The following table shows a comparison of the valve actuation system requirements between the J-2 and J-2S systems:

	J-2	J-2S
Stored Gas Capacity	1-1/2 cubic feet at 3000 psig	1-1/2 cubic feet at 725 psig
Regulated Pressure	675 - 750 psig	None
Prevalves	Actuate 10 Valves at ECO	Actuate 10 Valves at ECO
	Actuate 5 Fuel Prevalves at Recirculation STOP	None
11 Recirculation Valves	Hold Closed from S-II Start to S-II ECO	None
Emergency Actuation	6 LH_2 Recirculation Valves	None
Prevalve Surge Chambers	2 at 750 psig	2 at 725 psig

With the removal of the recirculation valves from the stage, the valve actuation system is greatly simplified. The prevalve design is a normally open valve that requires pressure only in the closed position. The emergency actuation system functioned only to operate the LH_2 recirculation valves. With the removal of all recirculation valves, the emergency actuation system is no longer required.

Analysis determined that the stored pressure could be reduced to 725 $\pm$ 25 psig. The prime factors considered were the reduced requirement for stored energy due to the improved prevalves, the removal of 11 recirculation valves that required actuation pressure throughout the flight, and the requirement to actuate the fuel prevalve in flight.

10.2.2.6 Flight Control System

10.2.2.6.1 LOR Mission

a. Engine Actuation System. The nominal drive speed of the engine-driven hydraulic pump as governed by the LOX turbopump will be increased from the present 8,000 rpm to a value of 10,200 rpm. Consistent with the increased speed, and unchanged hydraulic flow requirements, the pump hanger ("wobble plate") angular travel limit will be changed to limit the flow to the existing 8-gpm requirement at the new rated speed. Because the reduced angular travel will produce less radial displacement of the piston shoes relative to their hold-down plate, the diameter of the piston shoe cutouts in the plate will be reduced accordingly. This will provide greater bearing area in an axial thrust direction between the piston shoe flanges and the hold-down plate to cope with the increased, speed-induced, load.

Two revised pumps will be subjected to the following qualification tests at the new speed conditions:

1. Fluid immersion
2. Calibration
3. Endurance
4. Heat rejection

Early studies, based on a J-2S configuration employing a 27.5-to-1 chamber, indicated that an interference would exist between the main hydraulic pump and the thrust chamber. The current J-2S with a 40-to-1 chamber will eliminate the major interference, but, it may be necessary to rotate the case bypass solenoid valve to eliminate a potential minor interference of the connecting wire harness with the engine.

The only other changes required to adapt to the J-2S would be minor revisions to hydraulic hose routing due to differences in the hose bracket attach provisions between the J-2 and J-2S engines.

The actuator phase lag during static firing is expected to be slightly higher than with present J-2 engines. It is, therefore, probable that the existing static firing phase lag requirements will have to be relaxed and/or the supplier's phase lag acceptance test requirements tightened. Characteristics during flight are not affected and are satisfactory.

The existing actuator and existing operating pressure produce a maximum stall force of 48,000 pounds. This is below the 52,000 pounds specified in the J-2S Interface Criteria, document R-7211, which is assumed to be the engine limitation and not a design load for the actuator.

10.2.2.6.2 LEO Mission

a. Engine Actuation System Modifications. The LOX turbopump, which drives the main hydraulic pump, does not rotate when the J-2S is operating in the idle mode. Therefore, because the engines must be gimbaled during the idle mode operation, the hydraulic power for gimbaling must come from an auxiliary motor pump (AMP). The present S-II stage employs an ac-powered AMP in each of the four hydraulic systems. The AMP is used only on the ground for system checkout, and for prestatic firing and prelaunch preparations. Use of the ac-powered AMP in flight would require addition of batteries, a 3-phase 400 Hz inverter/transformer and possible modifications to the motor to cope with the near-vacuum environment in flight. To avoid the development effort required to adapt the present ac system, it is proposed that the existing AMP be replaced with a modified S-IVB dc-powered AMP.

In the S-IVB application, all of the inlet fluid to the AMP is taken from the case of the main pump and an inlet filter is provided to prevent the AMP from ingesting wear particles from the main pump. Also, because of the filter, the case drain flow from the AMP is ducted back to the upstream side of the filter and recirculated within the AMP package. In the S-II application, the AMP inlet will be taken from the prefiltered fluid in the system reservoir and will not require additional filtration. Therefore, because the AMP filter is not required and is, in fact, detrimental in that the S-II reservoir pressure may be insufficient to overcome the filter pressure drop, the filter will be deleted. In addition, a relief valve and differential pressure indicator, paralleling the filter and a quick-disconnect fitting (used for ground servicing on the S-IVB) will be deleted. Elimination of the filter will require that a case drain connection be added to the AMP so that the case drain can be routed back to a point upstream of the existing system return filter.

Figure 10.2-85 shows the installation of the existing accumulator reservoir manifold assembly (ARMA), the AMP, the ARMA/AMP mounting panel and adjacent equipment and structure. Figure 10.2-86 shows the same installation with the new S-IVB type AMP and the revised ARMA/AMP mounting panel. The line routing between the existing AMP and the ARMA, and that of the new AMP and existing ARMA, are shown schematically in Figures 10.2-87 and 10.2-88, respectively.

Electrical power for the four AMP's will be supplied from two 56-volt batteries. Each battery will power the systems on diametrically opposed engines so that if one power supply fails, the thrust vector control symmetry will be retained. Additional discussion of the electrical power system is contained below.

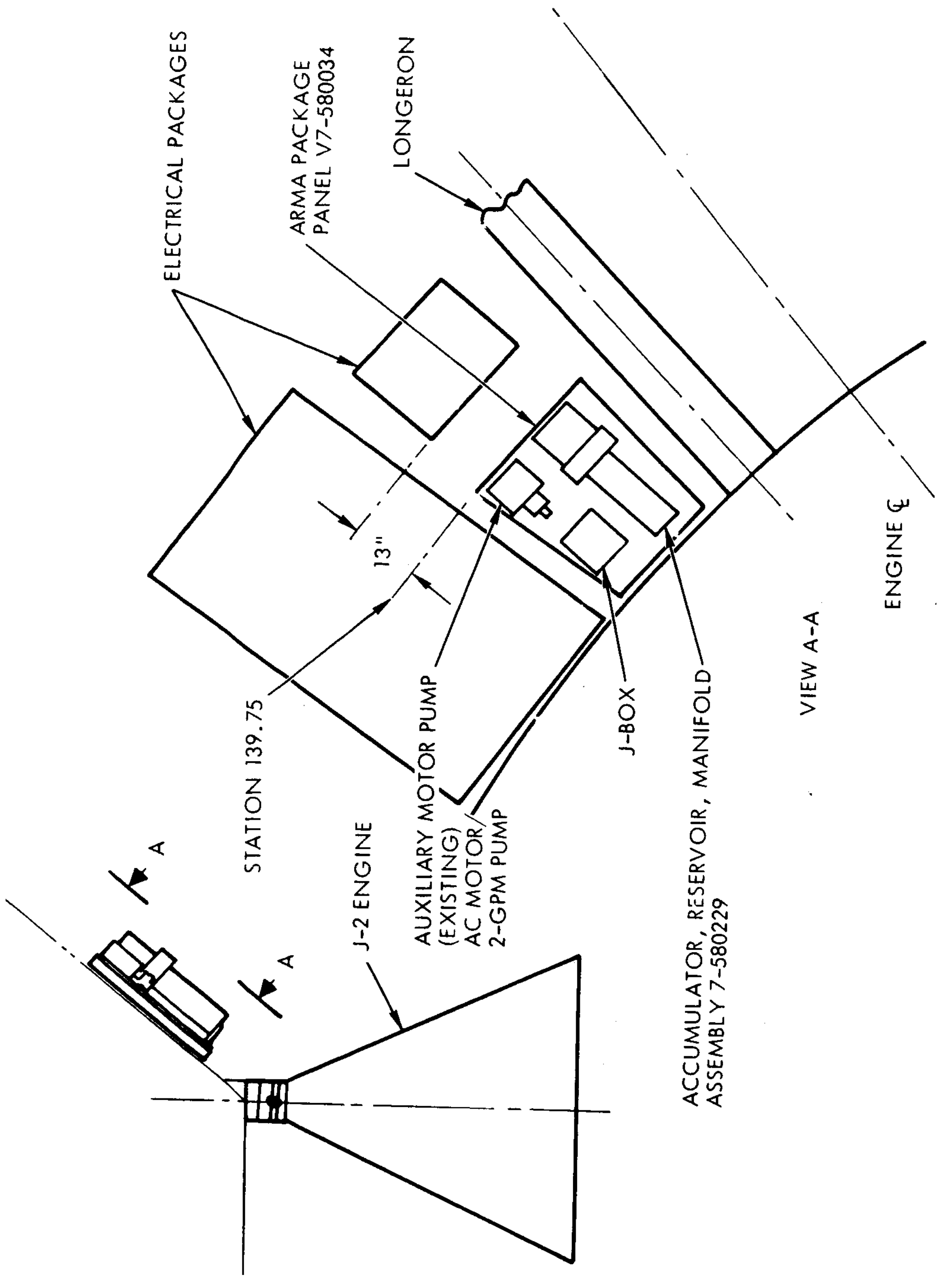


Figure 10.2-85. Accumulator Reservoir Manifold Assembly and Auxiliary Motor Pump Location

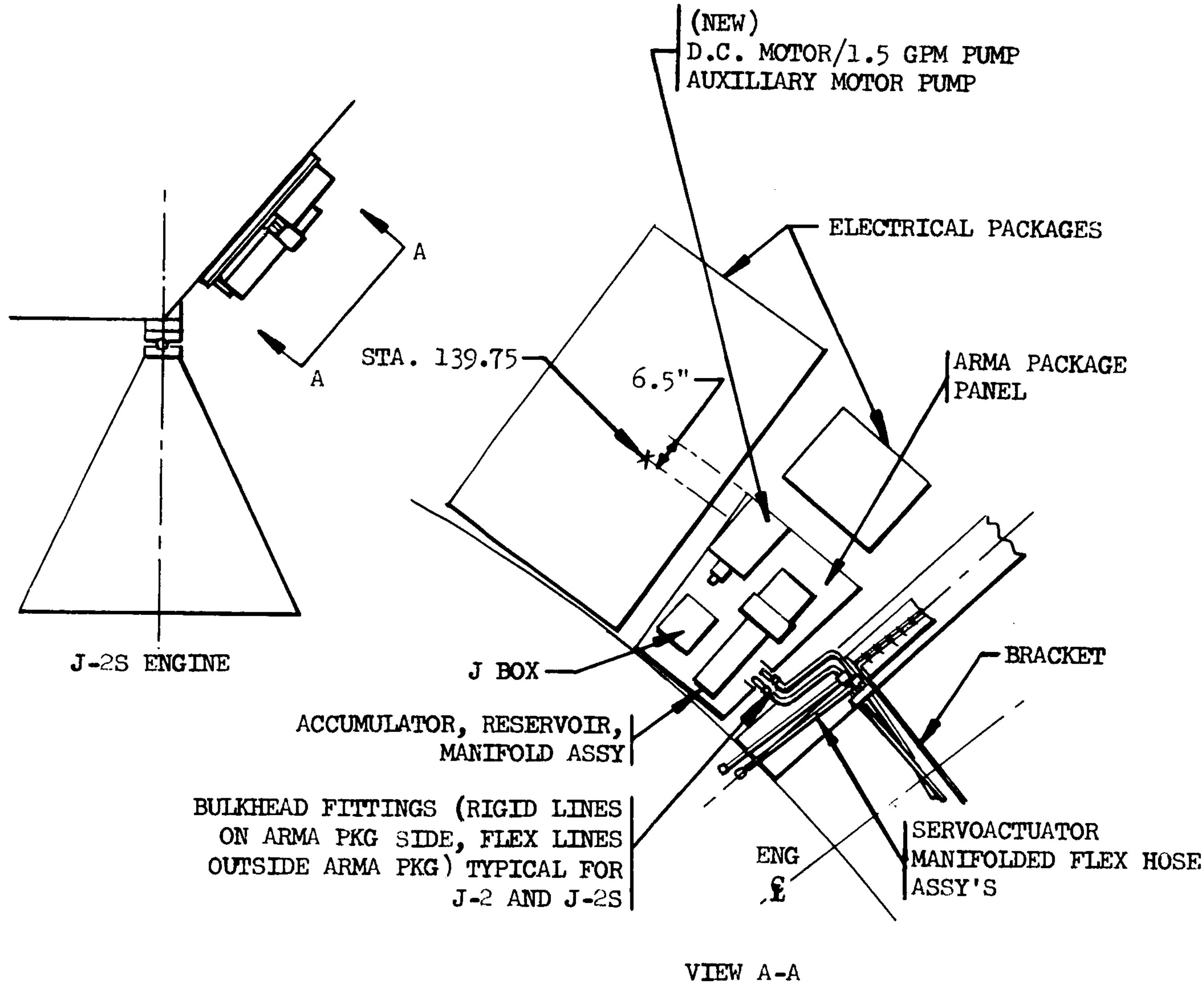


Figure 10.2-86. J-2S/S-II ARMA and AMP Location, LEO Mission

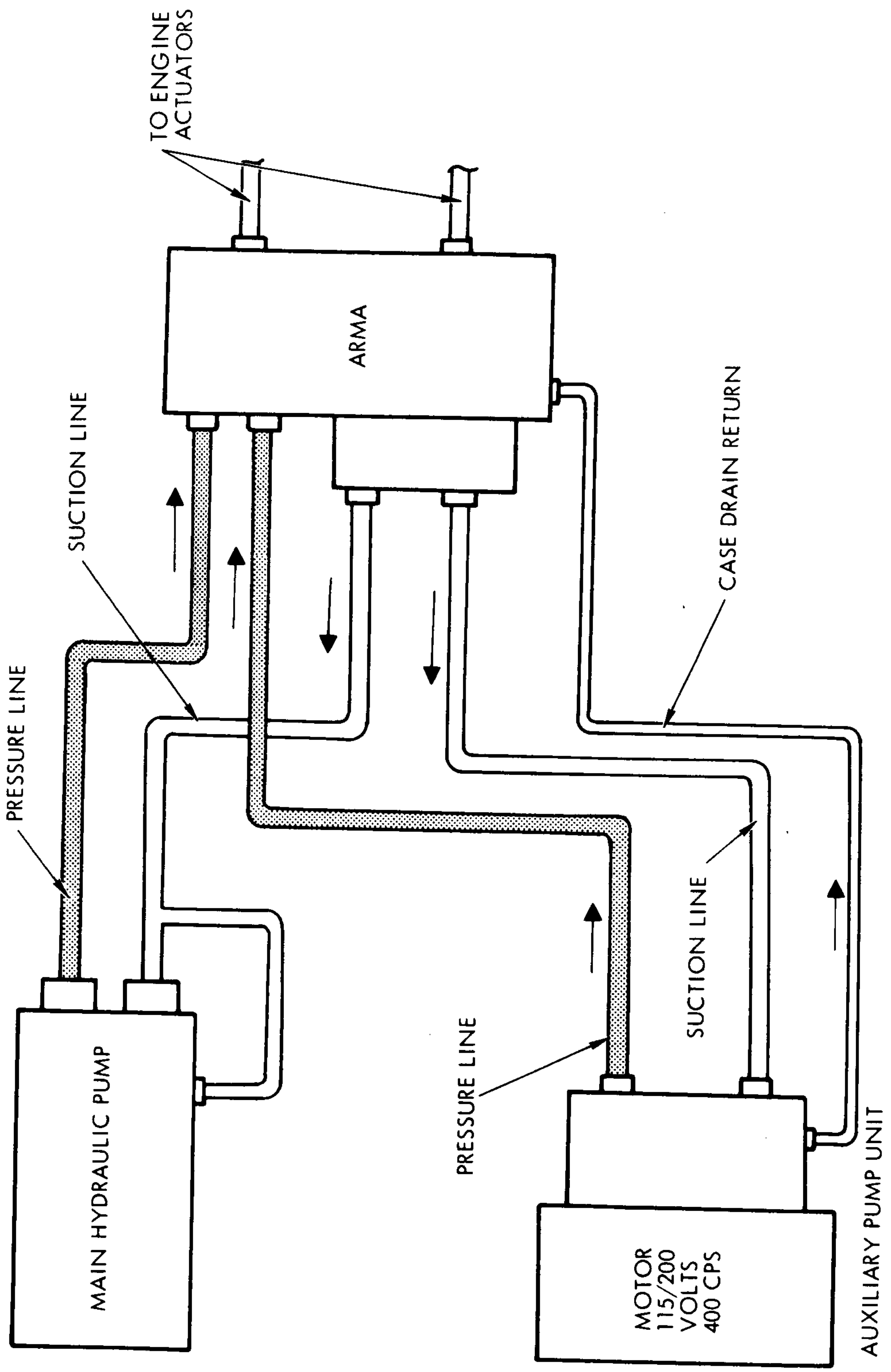


Figure 10.2-87. Existing ARMA to AMP Line Routing

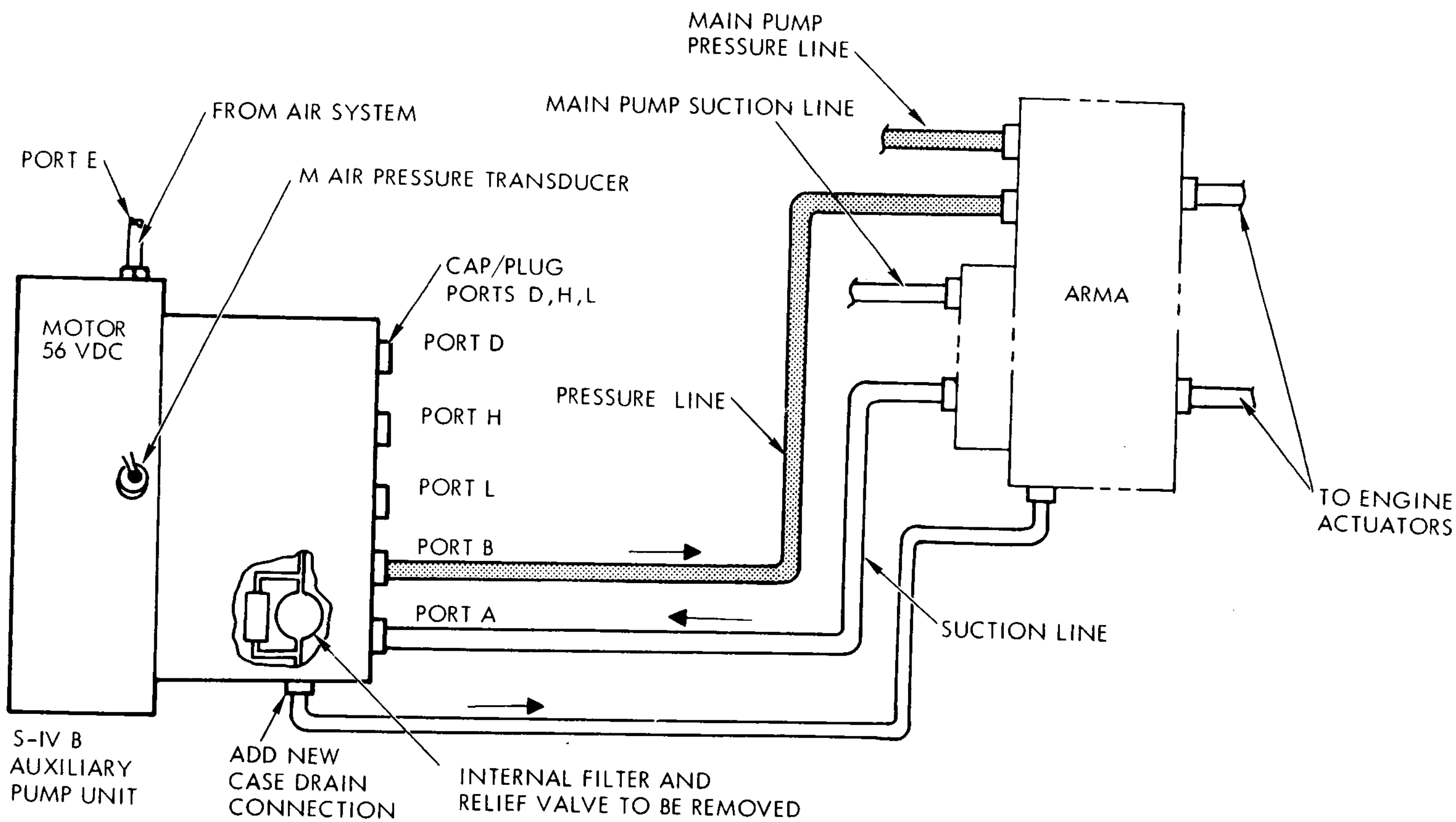


Figure 10.2-88. J-2S/S-II ARMA to AMP Line Routing, LEO Mission

A positive pressure must be maintained in the AMP motor to minimize brush/commutator arcing and to serve as a medium for transferring heat to the outside shell and to the hydraulic fluid where it can be rejected by radiation. Development tests suggest that the motor commutator and brush surfaces rely on an oxide film for lubrication. Air was therefore determined to be the best pressurizing medium.

To maintain the required ambient within the motor, two 525-cubic-inch receivers precharged to a nominal pressure of 470 psig will be utilized. Each receiver will support diagonally opposed systems in a manner similar to the electrical power system distribution described above.

Operation of the engine actuation system from liftoff through S-II boost will be the same as presently employed. The accumulator will be locked-up at termination of the idle-mode following the main stage cutoff. At this time, it is predicted that the accumulators will have discharged approximately 70 cubic inches each through gimbaling and valve leakage, and will have approximately 45 cubic inches remaining.

The systems will remain dormant during the subsequent coast until approximately 3 minutes prior to idle mode start. At this time the AMP's will be started and the accumulators unlocked in preparation for gimbaling control during idle burn. The pumps will recharge the accumulator oil volumes in approximately 34 seconds, thus assuring full accumulator energy at engine start.

The peak demand will occur at initiation of thrust vector control. At this time, there could be residual 2.5-degree engine position error. This will cause a net accumulator depletion of only 8 cubic inches even if the position error exists in both pitch and yaw.

Subsequent to this transient the expected gimbaling plus servovalve leakage will require an average flow of 1.2 gpm, which is 0.5 to 0.6 gpm less than the AMP capability.

If coast periods in excess of one hour are anticipated, it is likely that the AMP's would have to be started periodically to maintain satisfactory fluid temperatures.

The temperature of the gas in the LOX turbopump exhaust hood at engine cutoff is estimated to be 675 F for the J-2, and 730 F for the J-2S. This indicates that the temperatures within the hydraulic pump will be slightly higher after cutoff of the J-2S engine than those presently experienced with the J-2 installation. The highest temperatures of interest will be those of the O-rings in the shaft seal. However, the expected temperatures will be less than the maximum allowable for the "Viton" material employed.

b. EAS Auxiliary Air System. During restart idle mode operation of the J-2S engines, vehicle attitude will be maintained by gimbaling the J-2S engines for thrust vector control. Hydraulic power for this gimbaling must be provided by an auxiliary hydraulic pump driven by a direct-current electric motor as during idle mode operation the J-2S engines propellant pumps and therefore the main hydraulic pump are not operated. The dc motor requires an air atmosphere around the brush-commutator to eliminate excessive sparking wear, and erosion of the brushes during operation at altitude (vacuum).

The air requirement is for missile-clean dry air meeting the requirements of MSFC specification 164 at a pressure of 470 $\pm$ 45 psia. A storage capacity of 1050 cubic inches at this pressure and ambient temperature will satisfy all requirements including the maximum anticipated leak rates.

To improve the system reliability, the auxiliary air system shall be provided as two identical subsystems. Each subsystem shall supply two auxiliary hydraulic systems that are diametrically opposite: one subsystem will operate in conjunction with Engines 1 and 3 and the other subsystem with Engines 2 and 4.

As an air source is not presently available on the S-II vehicle an additional system must be incorporated into the S-II stage.

The auxiliary air system will consist of a single umbilical connection and two subsystems of identical components including a reservoir (525 cubic inches), relief valve, check valve, vent valve (solenoid), instrumentation fittings, interconnecting tubing, and fittings.

Figure 10.2-89 illustrates the routing of the air atmosphere system lines for the J-2S engine auxiliary hydraulic pumps. The system lines extend from stage umbilical panel 3A to the two air storage reservoirs located inside the thrust cone above X_B 139.75 frame adjacent to Position I and III and to the four auxiliary hydraulic pumps located outside the thrust cone adjacent to the fins at X_B 139.75. The lines are constructed of CRES tubing per MIL-T-8808 and are connected together with "B" nut type fittings.

The auxiliary hydraulic pump/motor package, as shown schematically in Figure 10.2-90 includes a connection to the air system, a relief valve, instrumentation ports, and a regulator to reduce the stored 470 psia to 15 psig within the motor compartment.

All components within the air system are presently available as qualified hardware with the exception of the storage bottle relief valve. A partial qualification test program of an existing valve at the new relief setting of 530 to 570 psig will be required.

10.2.2.7 Environmental Control System

10.2.2.7.1 LOR Mission

a. ECCS and TCS. No hardware or procedural changes will be required for the thermal control or engine compartment conditioning systems. The absence of thrust chamber childown and the increased base heating associated with the J-2S engines will result in boattail and thermal control container temperatures which will be slightly higher than presently experienced, but within acceptable limits.

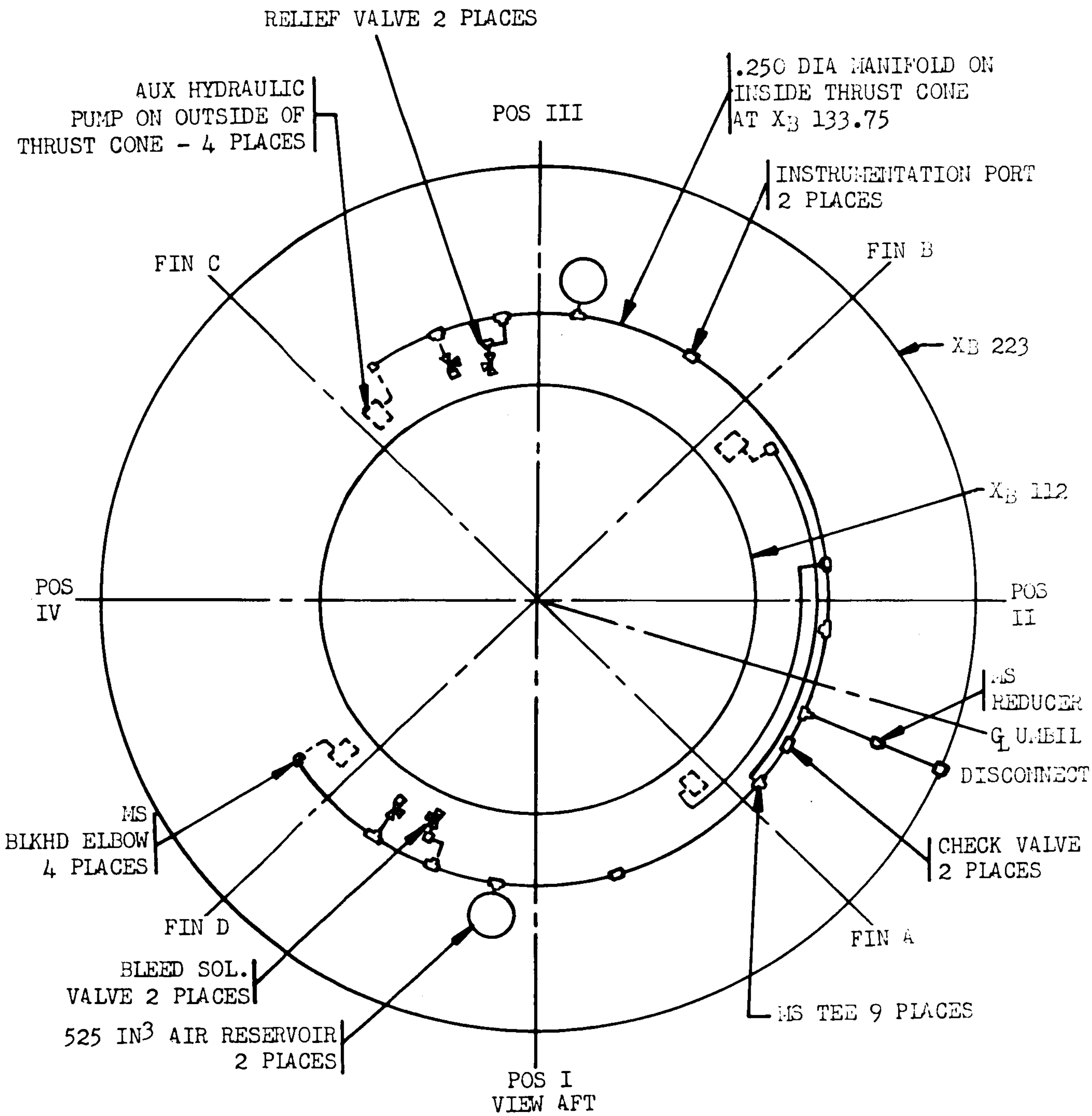


Figure 10.2-89. Air Atmosphere System Line Routing

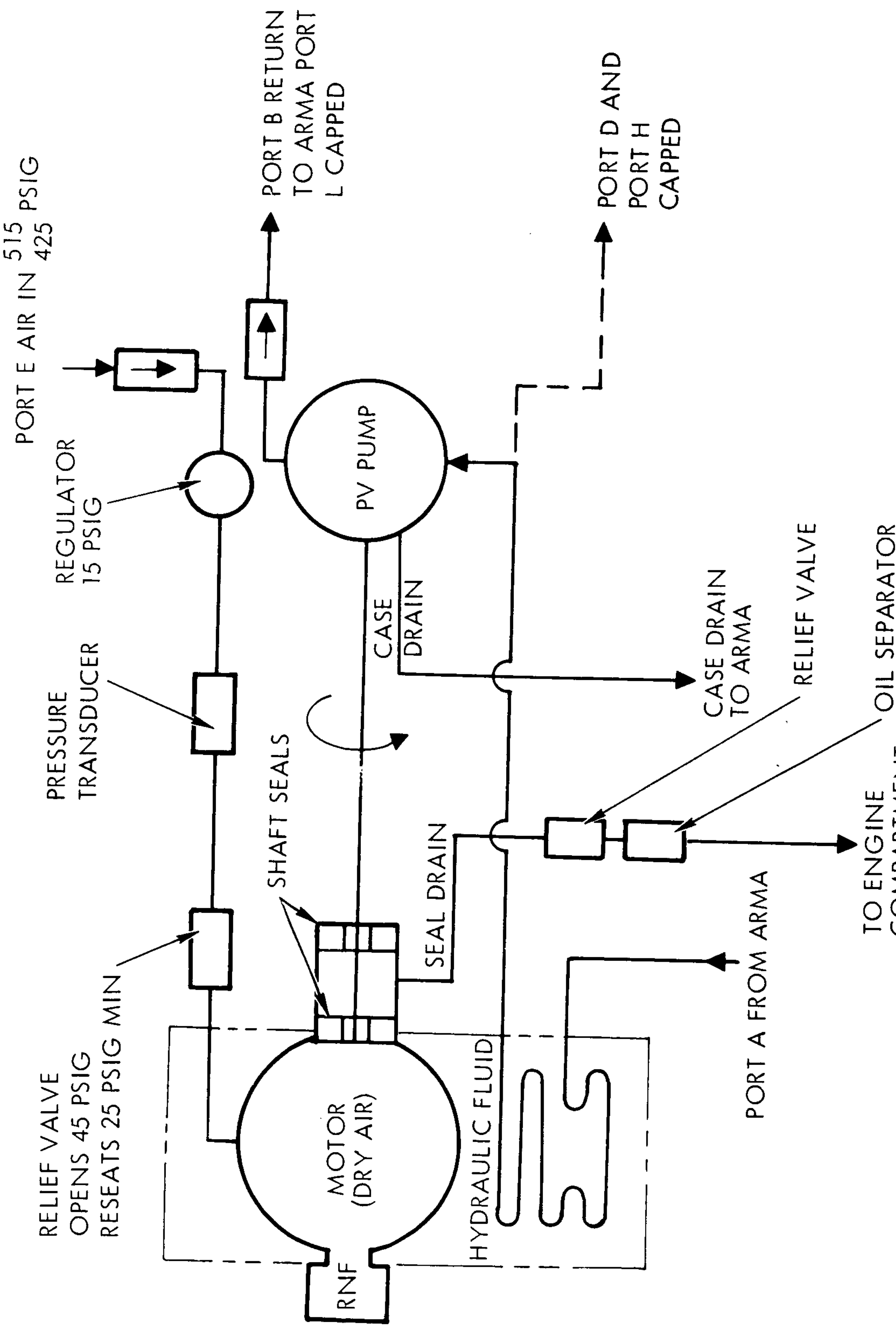


Figure 10.2-90. Engine Actuation System Auxiliary Motor Pump Fluid Schematic

10.2.2.7.2 LEO Mission

a. Coast Requirements. An analysis of one forward and one aft thermal control system container (each typical of the present design) showed that the internal temperatures after a 50-minute coast period would be -27 and -26 F, respectively. The typical minimum design temperature is 0 F.

A maximum heating condition, caused by base heating during S-II boost and solar radiation during coast, was also considered for an aft system container. In this case, the internal temperature was predicted to be 275 F. The upper limit for most containers is 140 F.

Preliminary studies show that the present internal temperature limits can be maintained on both forward and aft containers by use of 1-inch-thick cork insulation on the internal surfaces of the cover and sides and a 1/2-inch-thick cork on the internal surfaces of the ends and external surface of the base. This study was based upon the use of cork with a specific weight of 30 pounds per cubic foot. Additional studies will be required to optimize the insulation design and minimize the weight penalties. If orbital coast periods in excess of 50 minutes are to be considered, additional studies will be required. It is conceivable that there is some duration of coast which would require an active environmental control system.

10.2.2.8 Ullage Motors

The S-II stage equipped with J-2S engines does not require ullage motors; the four ullage motors and their associated electrical controls and structural supports will therefore be deleted.

The requirement for ullage motors on the J-2-equipped S-II stage was to ensure that settled propellants are maintained during the time period from S-IC engine cutoff and to provide a slight acceleration to assist in satisfying engine start NPSH requirements through S-II engine start. Studies have shown that ullage motors are not required to ensure that propellants remain settled for scheduled separation times. As the J-2S engine can be started with zero NPSH conditions and does not require LOX pump discharge propellants to be subcooled, the ullage motors will be deleted.

10.2.3 ELECTRICAL SYSTEM

This section describes changes to the S-II stage electrical system required for incorporation of the J-2S engine. The electrical system includes the following subsystems: electrical power, electrical control, instrumentation, and ordnance. Both design changes and operational changes are described. The principal system changes are the result of elimination of the LH_2 recirculation system, elimination of the ullage motors, and addition of the reaction control system for the LEO mission.

10.2.3.1 Electrical Power System

The electrical power system elements of concern include the recirculation batteries, main battery, instrumentation battery recirculation system inverters, and the associated switches and buses.

10.2.3.1.1 Design Changes

a. LOR Mission. The stage electrical power system for the LOR mission is shown on Figure 10.2-91 with deletions indicated for the recirculation batteries and inverters. Ignition power is shown provided by the main battery instead of by the recirculation system batteries.

An estimated main bus load profile based on the above deletions and on J-2S engine requirements is shown in Figure 10.2-92.

1. Engine System. Each of the five J-2S engines contains a solid propellant turbine starter (SPTS) that is used in starting the engine. Each engine contains two SPTS ordnance initiating chains, consisting of an ordnance initiator device that is fired by a high energy pulse from an exploding bridge-wire (EBW) firing unit. Both ordnance initiators are connected to the solid propellant turbine starter. The main stage bus will provide power for each engine including the ignition and SPTS devices. Dual pulse sensors are installed on the engine to be used in place of the ordnance initiators for checkout. The dual pulse sensors will be powered by the engine bus.

2. Fuel Recirculation System. Because the fuel recirculation system is not required for J-2S engines and will be deleted, the engine ignition load will be added to the main bus and 15 components will be deleted from the electrical power system as listed below. The telemetry and hardwire 28 vdc supplied by the instrumentation bus for the five fuel recirculation pump valve position indicator switches will also be deleted.

Components Deleted

Part Name	Part Number	Number Deleted
Inverter Assembly - Recirculation Pump	ME495-0006-0004	5
Recirculation Power Transfer Switch	ME452-0026-0011	1
Recirculation Batteries	ME461-0015-0001	2
Filter	ME495-0006-0006	5
Recirculation Power Distribution	V7-540277	1
Engine Ignition Power Distributor	V7-540285	1

3. LOX Recirculation System. The LOX recirculation system is also not required for J-2S engines and will be deleted. Therefore, three electrical components taking power from the main bus will be deleted. The telemetry and hardwire 28 vdc supplied by the instrumentation bus for the five LOX recirculation return valve position indication switches will also be deleted.

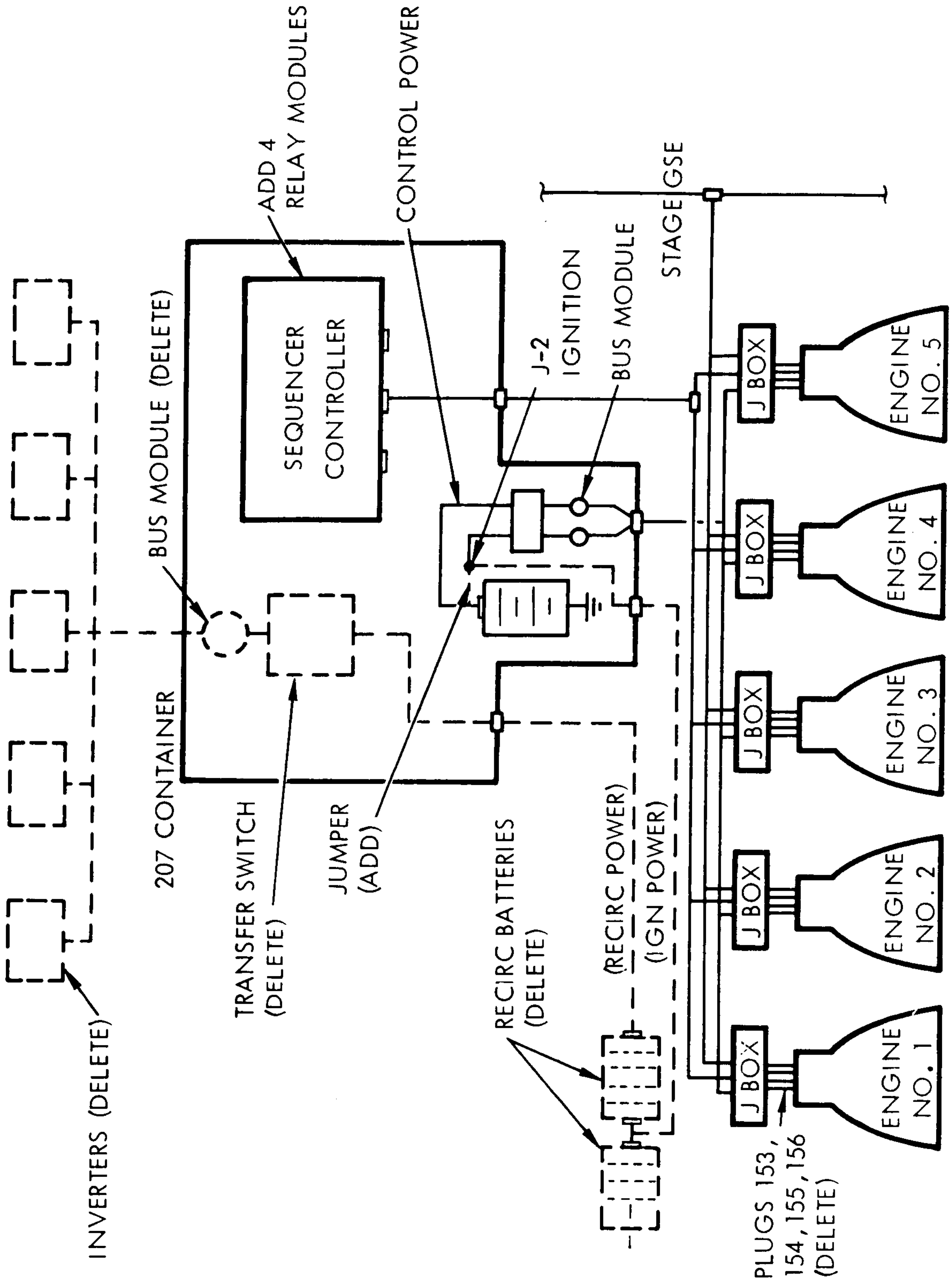


Figure 10.2-91. Electrical Power System Deletions (LOR Mission)

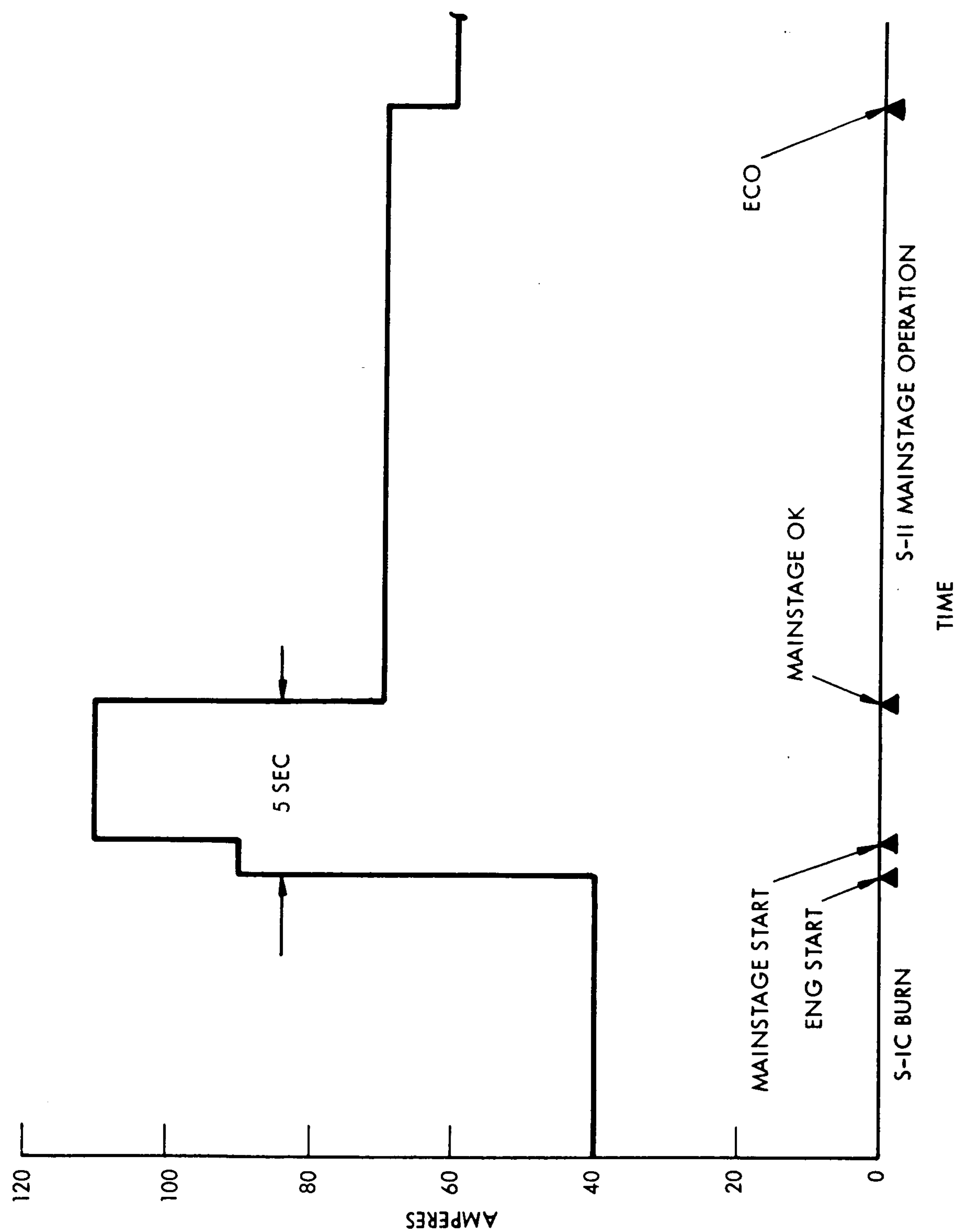


Figure 10. 2-92. Main Bus Current Profile (LOR Mission)

4. Separation System. The separation system EBW firing units 1A and 1B, which are used for the ullage motors, will not be required because the ullage motors will be deleted from the S-II stage. Therefore, the 28 vdc power required to operate these EBW firing units will be deleted from the main and instrumentation bus, and the 28-vdc power required to operate the associated pulse sensors will be deleted from the main bus.

b. LEO Mission. The LOR mission requirements described previously will have the following additional requirements for the LEO mission.

1. Engine System. The J-2S engine requirements for the LEO mission include the capability to terminate mainstage operation and return to idle mode operation prior to the 45-minute coast period. J-2S engine idle mode restart and operation capability will be required at the end of the coast period to complete the LEO mission. Since the main and instrumentation batteries will be required to provide additional power necessary for the 45-minute coast period, restart, and idle mode operation, new batteries will be required for this mission. The battery selected for these applications will have the following characteristics:

Voltage:	26 to 31 volts for load currents from 40 to 110 amperes and for a 0 to 105 F ambient
Capacity:	75 ampere-hours
Cells:	20 cells with selector for 19 cells
Size:	19 by 11 by 8.5 inches
Weight:	85 pounds (maximum)

The capacity requirement is based on a flight duration of approximately 1 hour, a main battery flight usage of 48 ampere-hours, and an instrumentation battery flight usage of 50 ampere-hours. The 75 ampere-hour design requirement will provide a satisfactory margin for checkout and transfer test usage. An estimated main bus current profile is shown in Figure 10.2-93. The instrumentation bus current profile is estimated to be constant at approximately 50 amperes throughout flight.

The battery minimum voltage requirement is based on the minimum allowable of 24 volts at all using equipment and a maximum of 2 volts drop from the battery. The battery maximum voltage requirement is based on the maximum allowable of 31 volts for J-2S engine control and ignition buses.

A ground-operated heater and a temperature transducer will be incorporated inside each battery to maintain and verify proper operating temperature. The battery case will be sealed, and the battery case and individual cells will be provided with a relief valve and vents, respectively, which permit escape of gas without loss of electrolyte in a gravity free, vacuum environment.

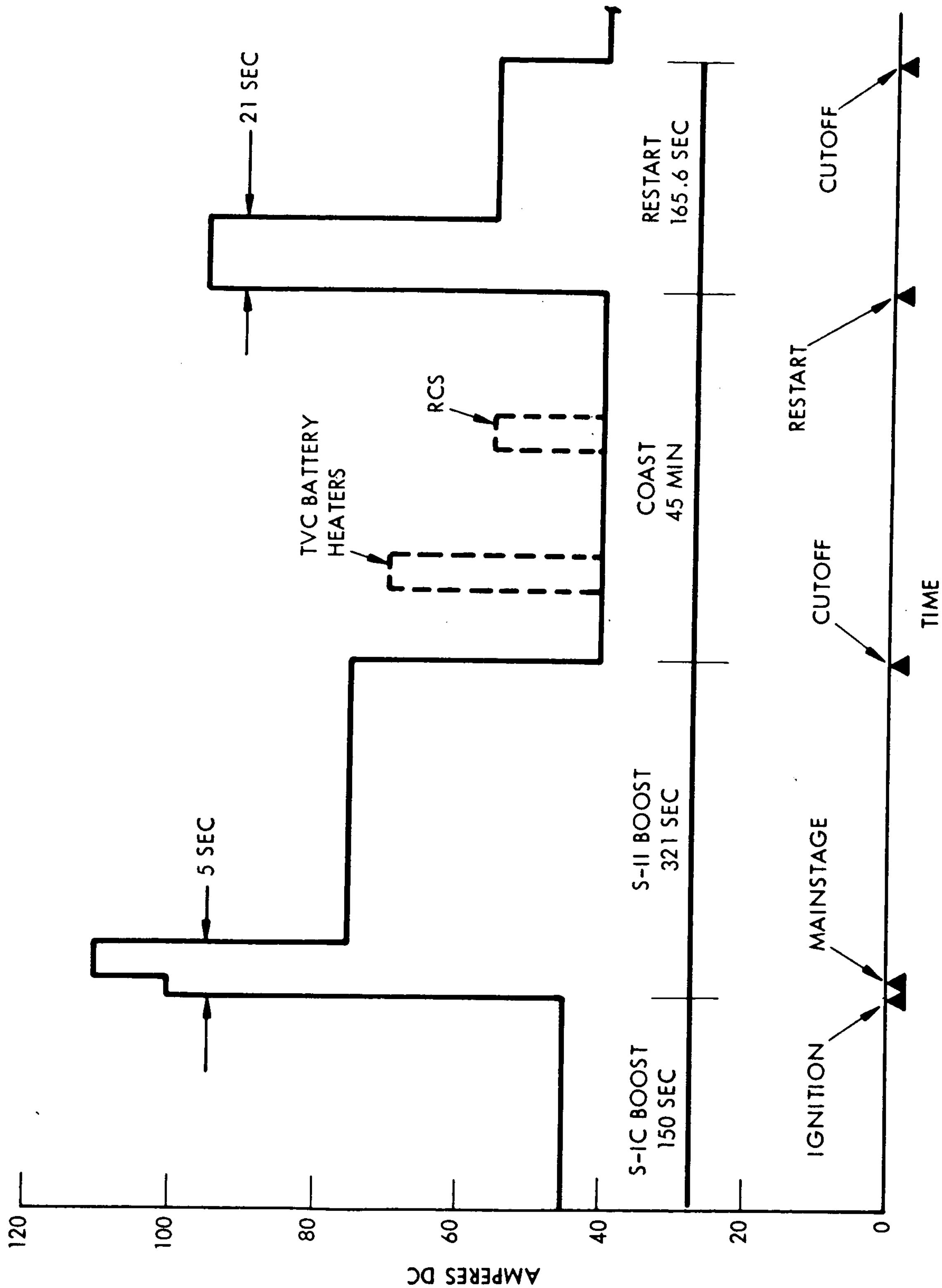


Figure 10.2-93. Main Bus Current Profile (LEO Mission)

The increase in size of the main and instrumentation batteries necessitates a redesign of the 206A31 and 207 containers. The main battery will be located within the redesigned 207 container and the instrumentation battery will be relocated in the 206A31 container since the 207 container cannot accommodate both batteries. Sketches of the revised 206A31 and 207 container layouts are shown in Figures 10.2-94 and 10.2-95.

If the coast period is lengthened to 105 minutes, the 75 ampere-hour battery described above could still support the main bus power requirements if the propellant utilization system is switched off during the coast period. This would result in a main battery flight usage of 57 ampere-hours. The increased coast period would result in an instrumentation battery flight usage of 100 ampere-hours. The 75 ampere-hour battery described above could be modified to a capacity of 110 ampere-hours with no increase in size or weight.

2. RCS. The reaction control system (RCS) provides vehicle attitude control during the coast period and J-2S engine restart. There are two RCS systems for each stage with mounting on opposite sides of the aft skirt area. The main battery will provide power to energize 24 solenoids in each RCS unit. The solenoids will be sequenced on and off intermittently to control the fuel and oxidizer valves for the RCS attitude control engines. Each RCS unit will use approximately 16 watt-hours of electrical power for maintaining stage attitude control. The instrumentation battery will provide 28 vdc power for the RCS flight measurements.

3. Flight Control System. The LEO mission requirements for J-2S engine restart and idle mode operation at the end of the coast period will require a thrust vector control system (TVC) for gimbaling the engines. The system must also be capable of providing gimbaling during ground checkout and of maintaining temperature control of the hydraulic fluid during the coast period. The main hydraulic pump is inoperative during idle mode operation, so an auxiliary motor pump (AMP) will be required to provide the hydraulic pressure necessary to gimbal the engines. It is proposed that the existing ground powered AMP be replaced by a modified S-IVB dc-powered AMP to provide these functions. A schematic of the electrical power distribution for the thrust vector control system is shown in Figure 10.2-96.

The 56-vdc power required by the TVC system will be provided by two battery packages. Each battery package will consist of two 28-vdc batteries which have the following characteristics:

Voltage:	26.5 to 31 volts for load current from 110 to 170 amperes and for a 0 to 105 degree F ambient. During motor start the voltage will not exceed the limits of 18 to 33.5 volts
Capacity:	25 ampere-hours
Cells:	21 cells with selector for 20 cells
Size:	16.5 by 10 by 8 inches
Weight:	60 pounds (maximum)

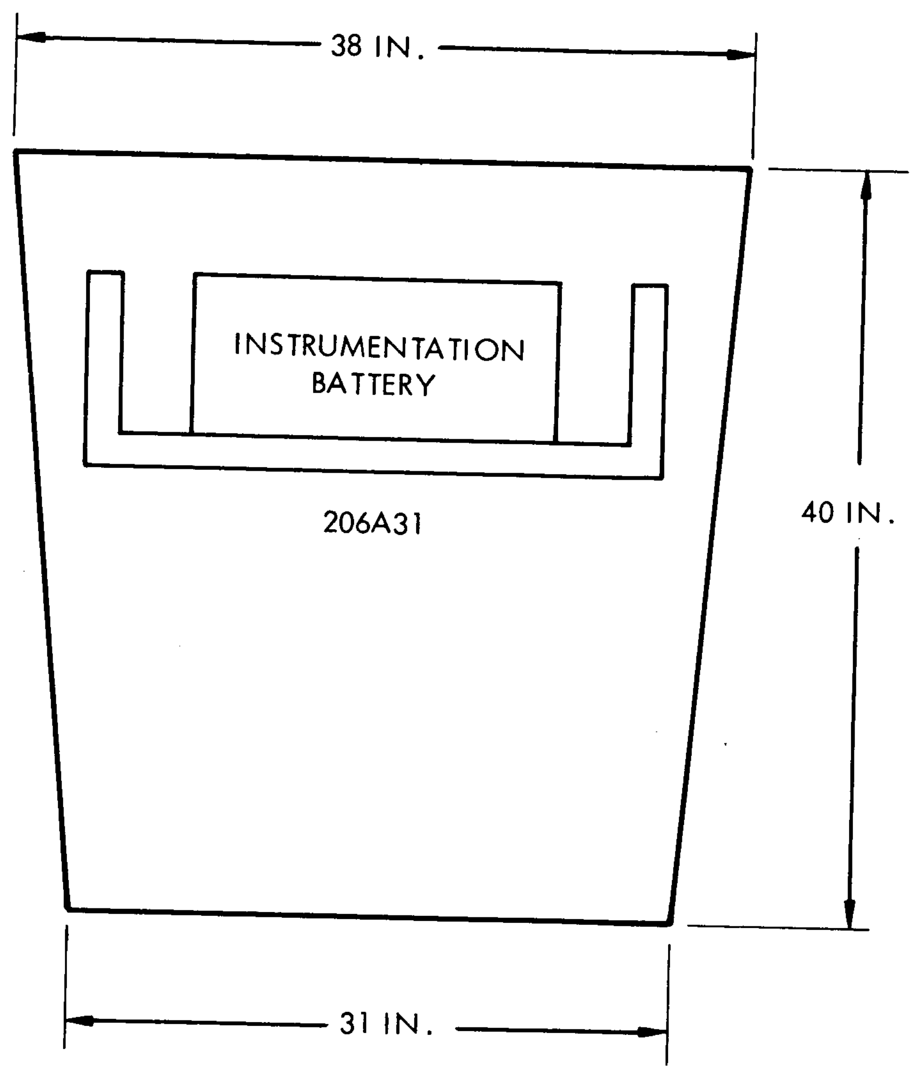


Figure 10. 2-94. Redesigned 206A31 Container

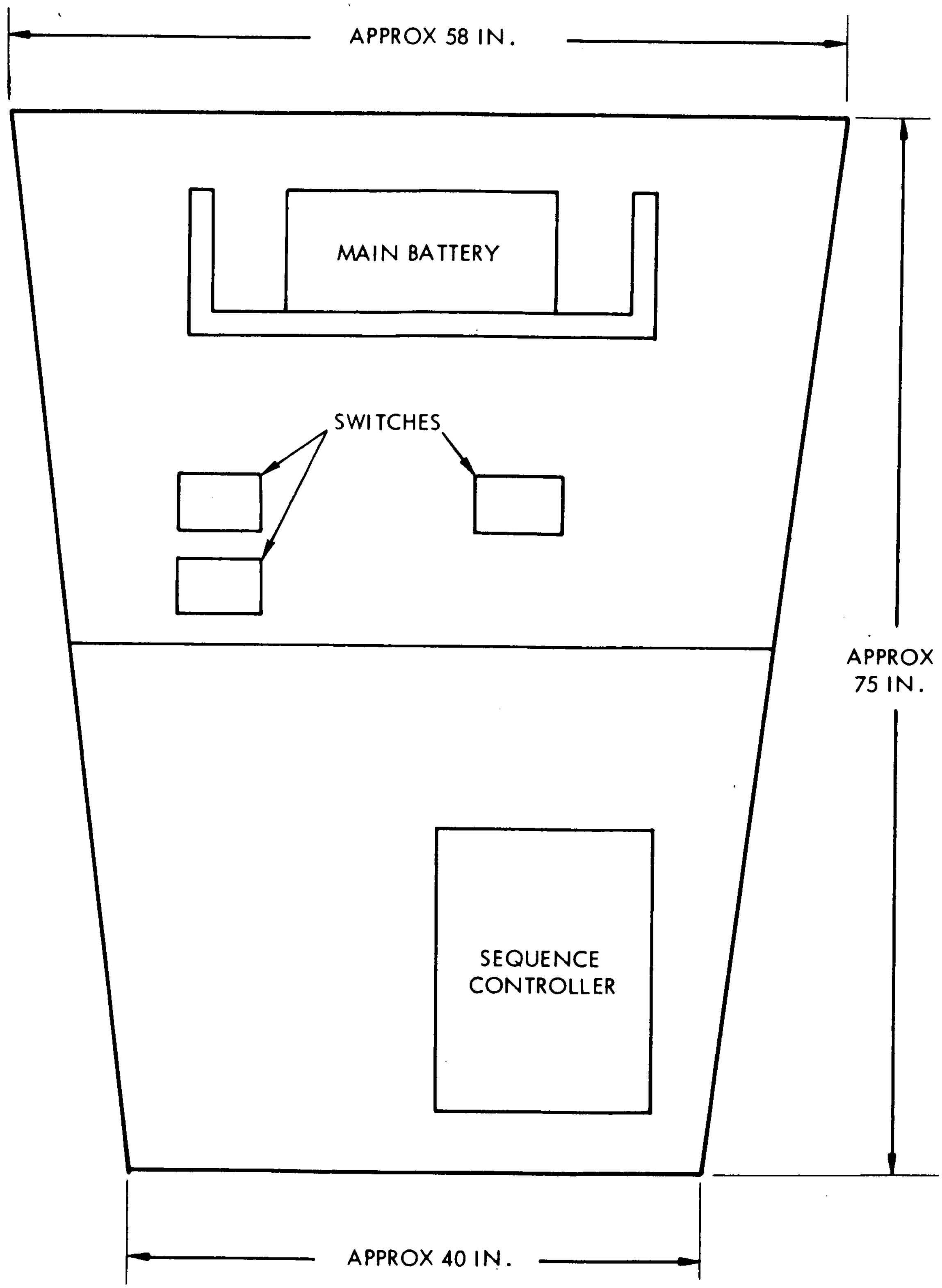


Figure 10. 2-95. Redesigned 207 Container

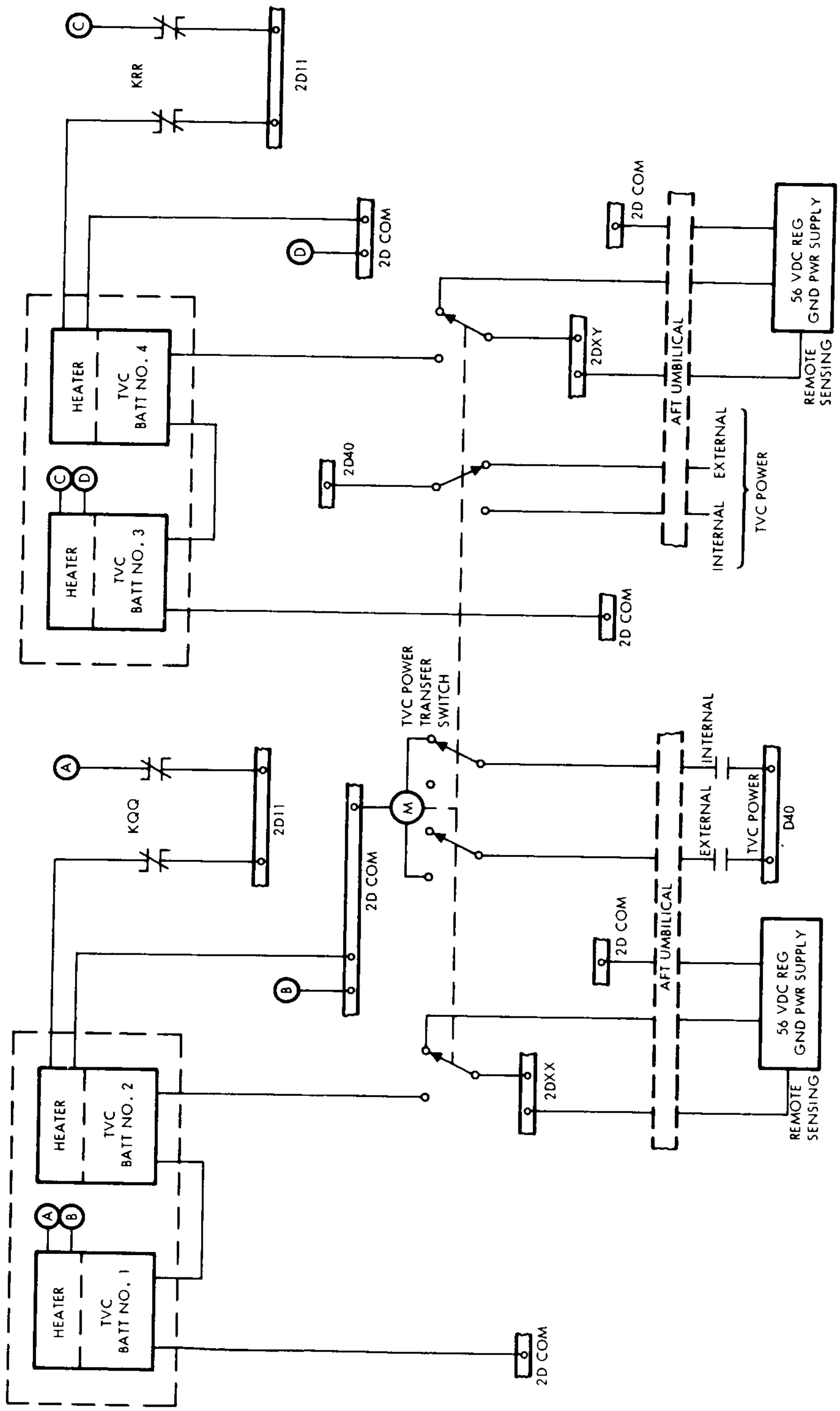


Figure 10.2-96. TVC System Electrical Power Distribution (Sheet 1 of 2)

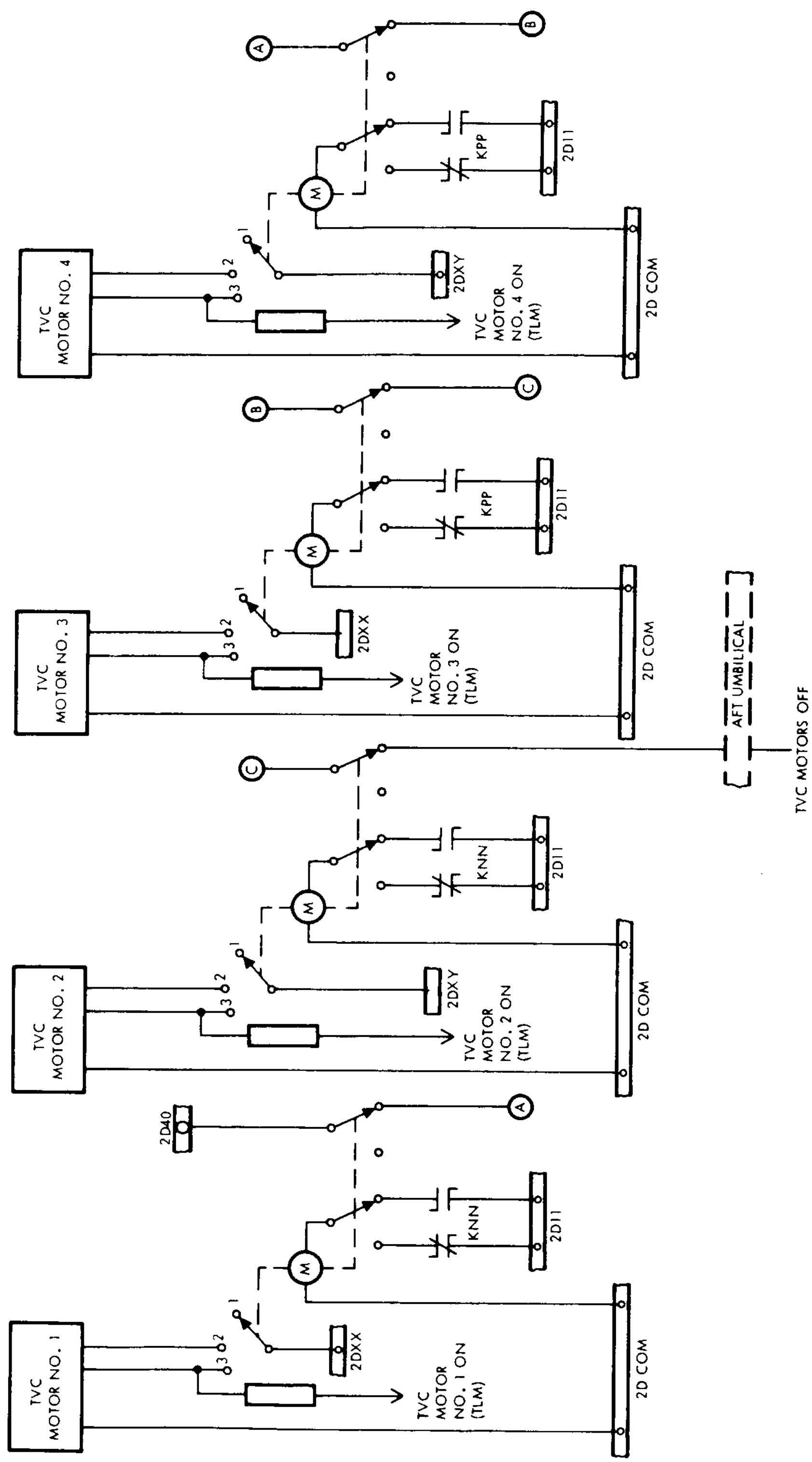


Figure 10.2-96. TVC System Electrical Power Distribution (Sheet 2 of 2)

The capacity requirement is based on a flight usage of 12 ampere-hours and a ground usage (checkout and transfer tests) of 6 ampere-hours. The 25 ampere-hour design requirement will provide a satisfactory margin for mission variations. An estimated bus current profile which is typical of either TVC bus 1 or 2 is shown in Figure 10.2-97.

The battery minimum steady state voltage requirement is based on a minimum allowable of 51 volts at the motors and a maximum of 2 volts drop from the battery. The battery minimum during motor start is based on S-IVB battery specifications. The battery maximum steady state voltage requirement is based on the maximum allowable voltage at the motors plus a minimum voltage drop from the batteries of one volt.

A ground and flight-operated heater and a temperature transducer will be incorporated inside each battery to maintain and verify proper operating temperature. Stage networks will be planned to inhibit heater operation during J-2S engine ignition. The battery case will be sealed, and the battery case and individual cells will be provided with a relief valve and vents, respectively, which permit escape of gas without loss of electrolyte in a gravity-free, vacuum environment.

Power and control switching will require one additional ME452-0026-0011 power transfer switch to transfer the two TVC buses from ground power to batteries and four motor start switches (MDD 1B32647) for motor start and stop.

The TVC batteries, power transfer switch, and motor control switches will be installed in a new container located on the thrust cone approximately 11 degrees from Position IV toward Position I (see Figure 10.2-98). The location chosen provides adequate space for the new container with a minimum of redesign to the surrounding area. A sketch of the TVC container layout is shown in Figure 10.2-99.

An alternate proposal to the S-IV dc AMP is the use of a flight ac power system to supply the existing ground-powered AMP. An ac system would eliminate problems with dc motor brush life and eliminate the requirement for a unique air supply to pressurize the motor throughout ground and flight operation. The ac system would require a special motor winding to match the output of the inverter, but similar windings are already in use on the recirculation pump motors. The inverter can be designed to incorporate a low frequency, low voltage output during motor start to limit current in-rush without reducing starting torque. However, implementation of the ac system would require extensive development and qualification programs since there is no qualified, off-the-shelf inverter of the above type available at present.

4. Fuel Tank Pressurization System. The LH_2 tank main vent valves will not require electrical lockup during the coast period. The LH_2 step pressurization switch selector Channel 7 command, used during mainstage operation, will not be required later in the mission. A balanced LH_2 tank vent system will be required during the coast period. With this system, the LH_2 tank will be

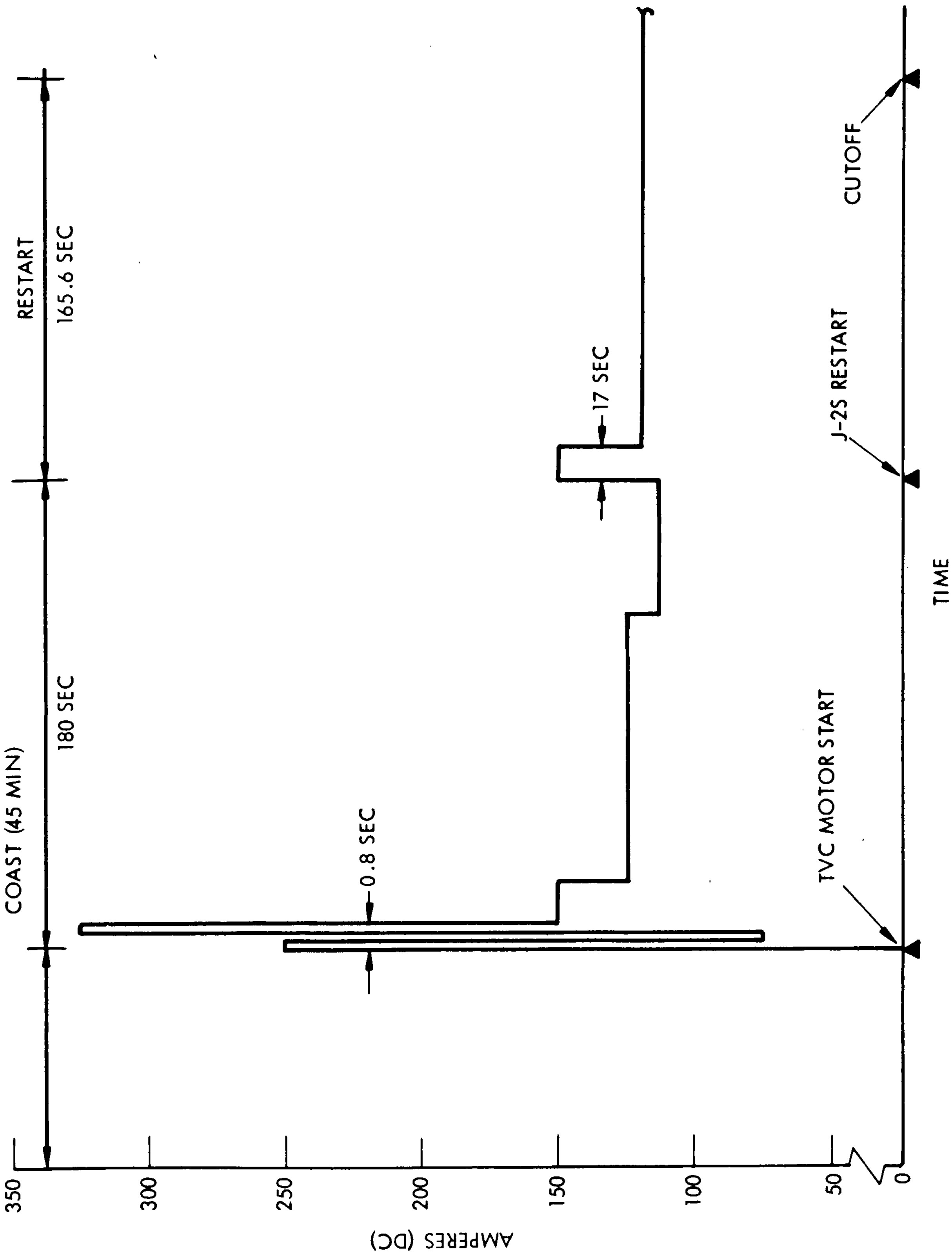


Figure 10.2-97. TVC Bus Current Profile

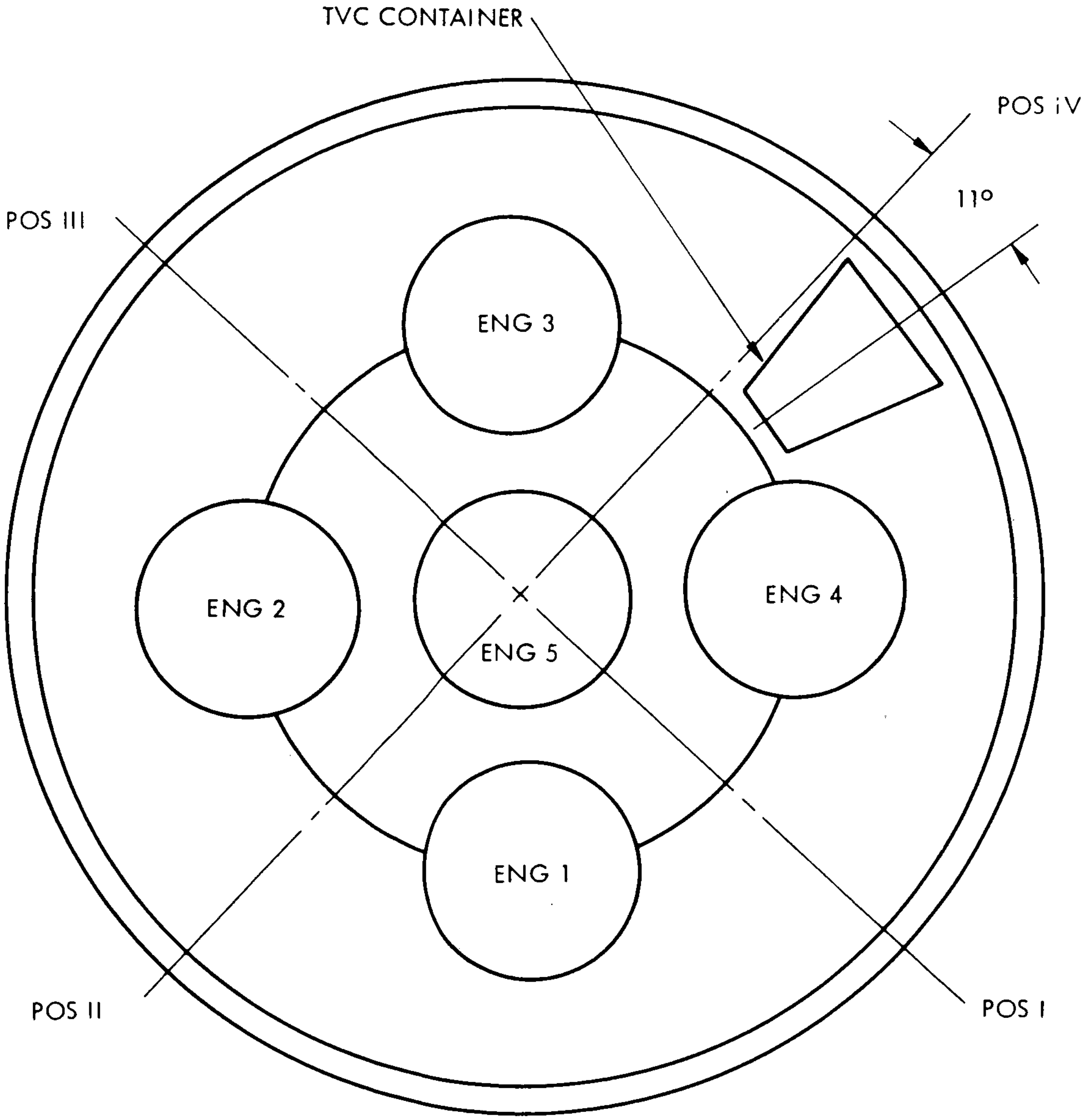


Figure 10.2-98. TVC Container Location

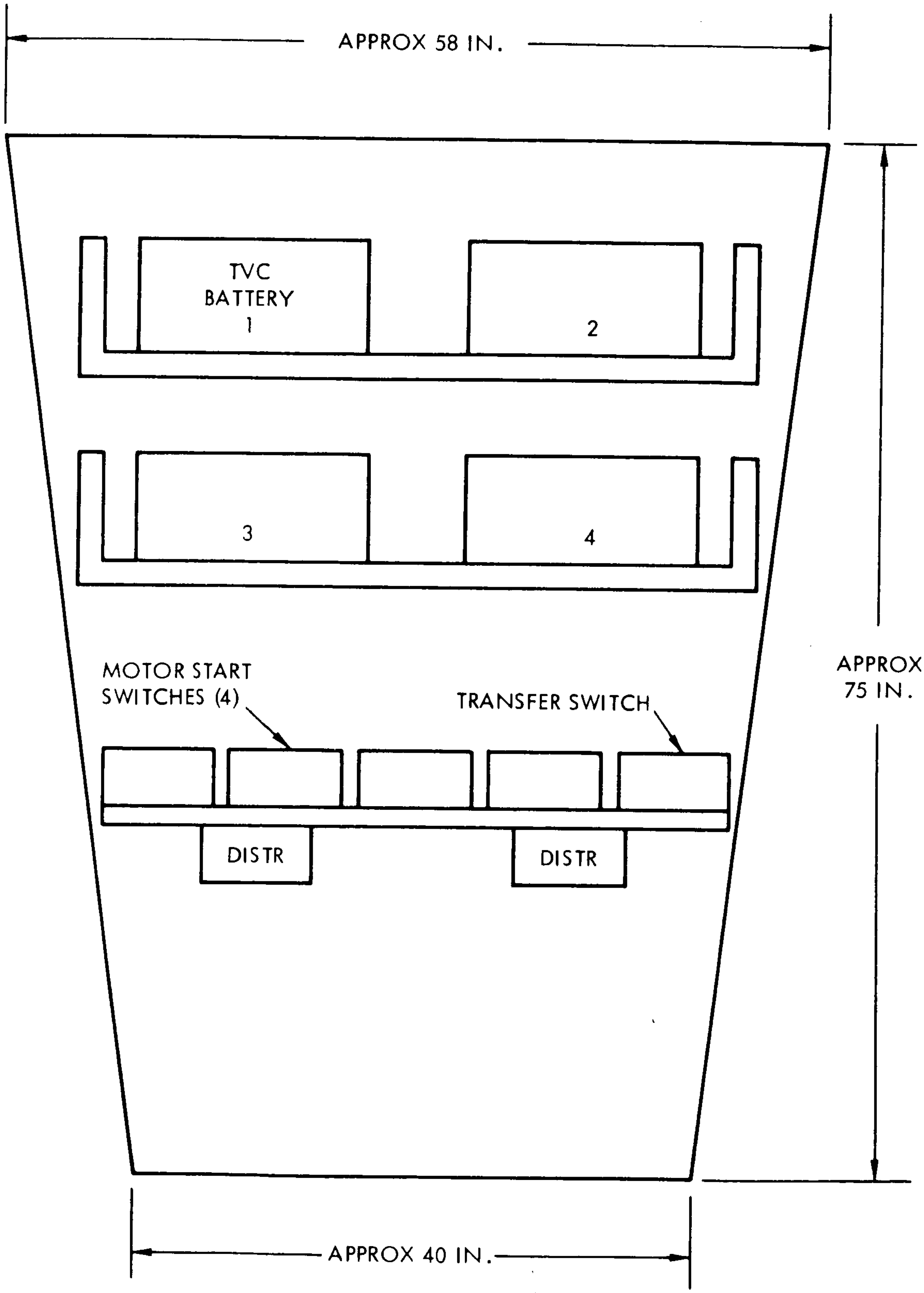


Figure 10. 2-99. TVC Battery Container

vented through propulsive nozzles to maintain settled propellants and to control tank pressure. The main bus will provide 28-vdc electrical power to operate the balanced venting system.

10.2.3.1.2 Operational Changes

This section summarizes changes to the electrical power system checkout requirements associated with J-2S hardware changes.

a. LOR Mission. The electrical power system checkout requirements to incorporate J-2S engine for the LOR mission will affect Specification MA0701-1034-111, "Manual Checkout, Electrical Power System." Deletion of the fuel recirculation system will result in approximately a 25-percent revision to the above specification and a 25-percent reduction in effort required to perform the checkout operation.

b. LEO Mission. The electrical power system checkout requirements will affect Specification MA0701-1034-111, "Manual Checkout, Electrical Power System." Addition of the TVC system results in approximately a 20-percent revision to the above specification and a 20-percent increase in effort required to perform the checkout operation.

10.2.3.2 Electrical Control System

10.2.3.2.1 Design Changes

This section contains the hardware changes to the electrical control system associated with the J-2S engine incorporation. A brief description of the change, the change rationale, and a list of the electrical control system components affected are included.

a. LOR Mission

1. Engine System. The incorporation of the J-2S engines on the S-II stage requires modification to the electrical control system, primarily to provide stage electrical control for the new J-2S engine solid propellant turbine starter (SPTS) system.

   Required changes to the switch selector commands for the LOR mission are as follows:

   <u>Deleted S-II Switch Selector Commands</u>

   LH_2 Recirculation Pumps Off

   Ullage Trigger

   Chilldown Valves Close

   LOX Depletion Sensors Cutoff Arm

New S-II Switch Selector Commands

Prevalves Close Arm Reset

All Engines Start No. 2

Mainstage Start No. 1

Mainstage Start No. 2

Revised Title

All Engine Start No. 1 was Engines Start

The J-2S engine will require an engine ready bypass signal from the stage prior to engine start. The stage will provide signals from engine ready bypass buses 1 and 2 when commanded through the switch selector. One additional relay will be required to incorporate this change. Figure 10.2-100 shows the circuitry required to mechanize these functions. These relays are reset by the engine control reset bus and the S-II engine start enable bus.

J-2S engine start will also require an "all engine start No. 1" command followed by an "all engine start No. 2" command from the switch selector. Each command is capable of initiating engine start and two relays will be required to implement these functions as depicted in schematic Figure 10.2-101. These engine start control relays are reset by the engine control relay reset bus and the S-II engine start enable bus. The S-II engine start enable bus is hot until S-IC separation during flight.

J-2S engine mainstage mode (following engine start) will require "mainstage start No. 1" command followed by "mainstage start No. 2" command from the switch selector. Each command is capable of initiating mainstage operation. Two relays are required to implement these functions. Figure 10.2-102 depicts the circuitry required to mechanize these functions. These relays are reset by the engine control reset bus and the S-II engine start enable bus.

The inflight LOX depletion engine cutoff arm command will be provided by a signal from two out of five sensors in the LOX tank without a switch selector arm command. Five relays will be required to implement this change as depicted by Figure 10.2-103. A hardwire LOX depletion arm indication will be provided through an umbilical connector for ground checkout.

The normal engine cutoff at termination of the S-II boost period will be accomplished by using the mainstage OK pressure switch dropout. The LOX tank depletion (two out of five engine dry sensor signal, as previously described) will be required to arm the engine cutoff circuitry (close relay contacts K_V and K_W in Figure 10.2-103). Then the first engine with both mainstage OK pressure switches to dropout will simultaneously cut off the remaining engines.

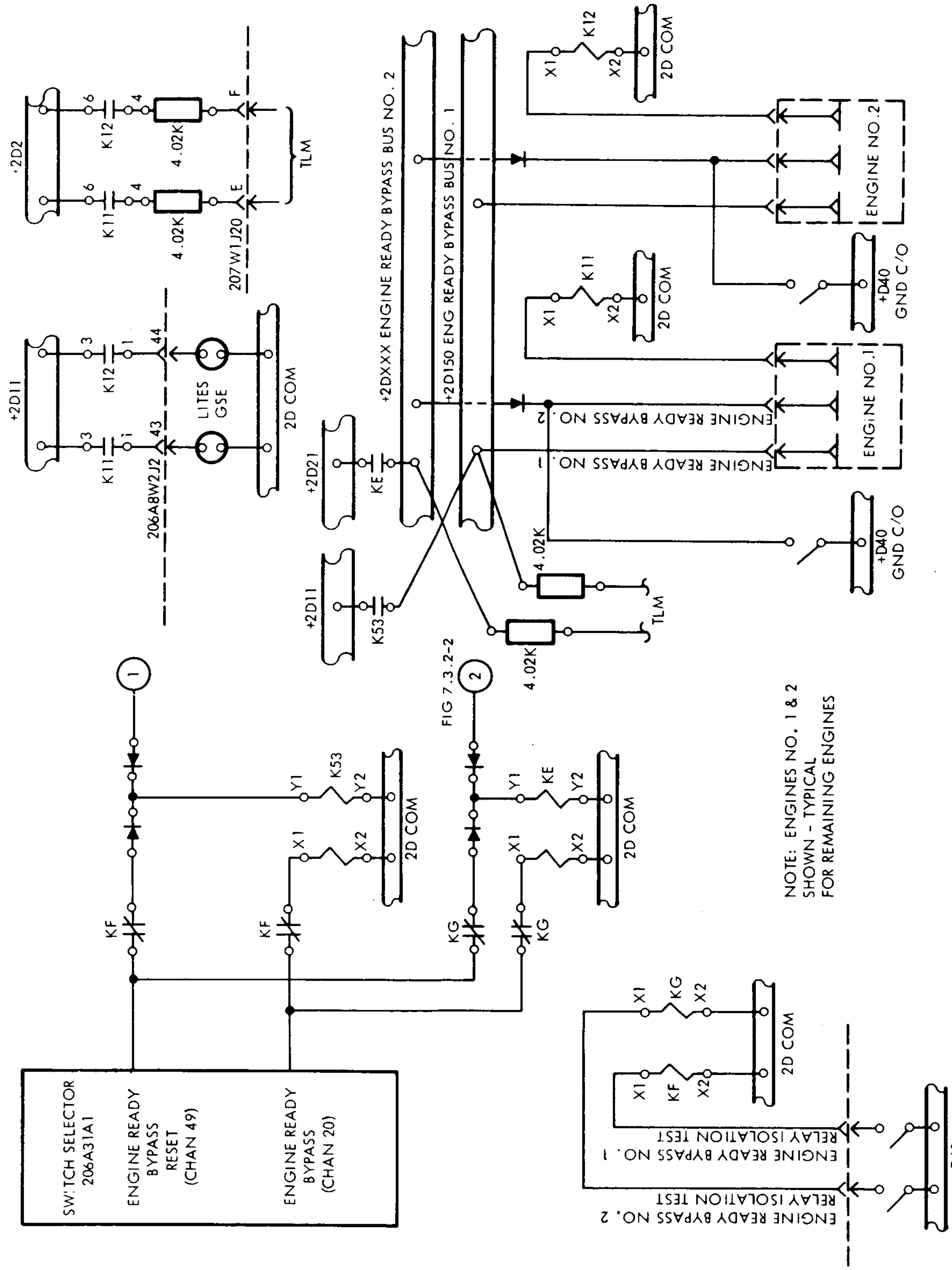


Figure 10.2-100. Redundant Engine Ready Bypass

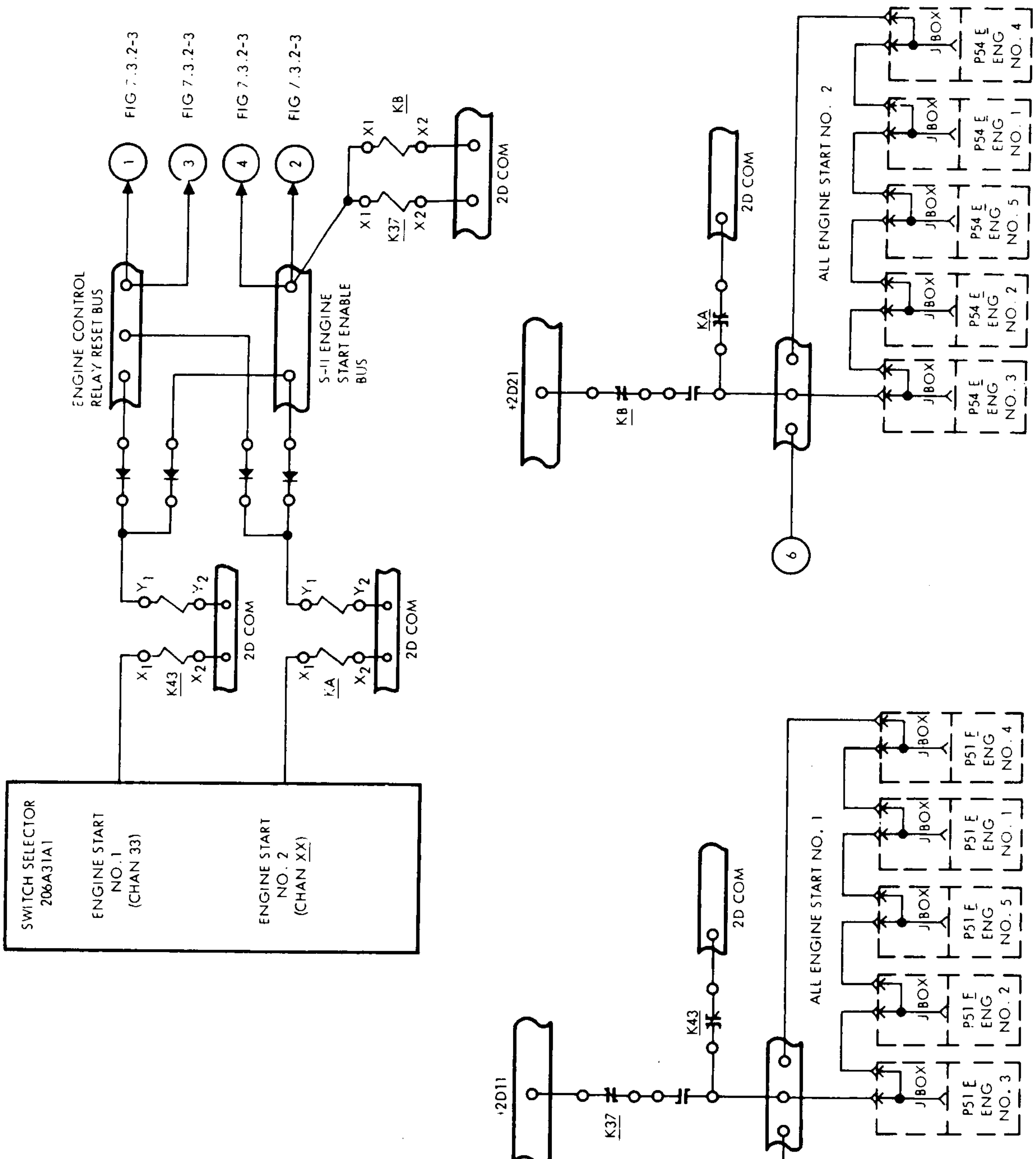


Figure 10. 2-101. Redundant Engine Start

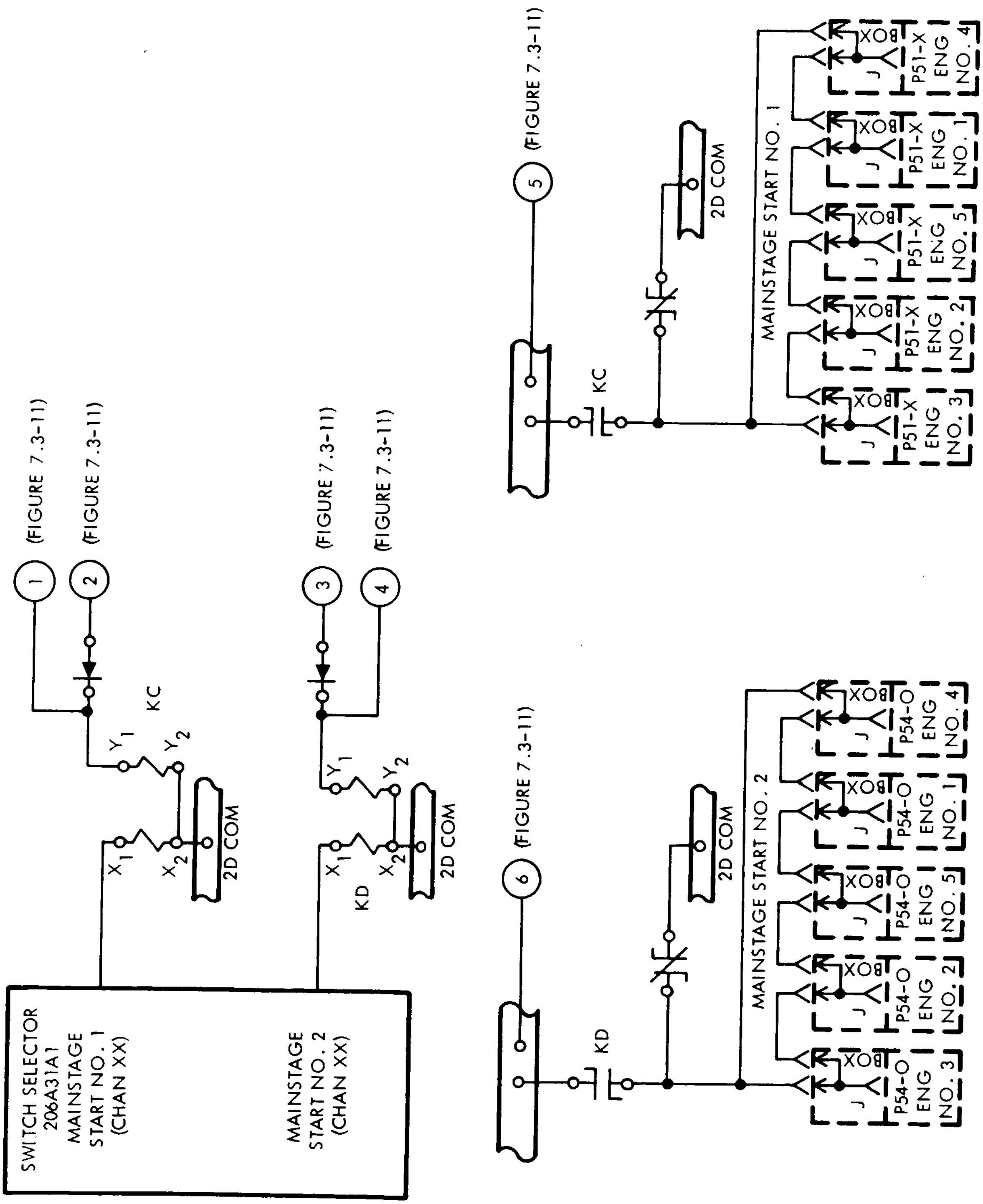


Figure 10.2-102. Redundant Mainstage Start

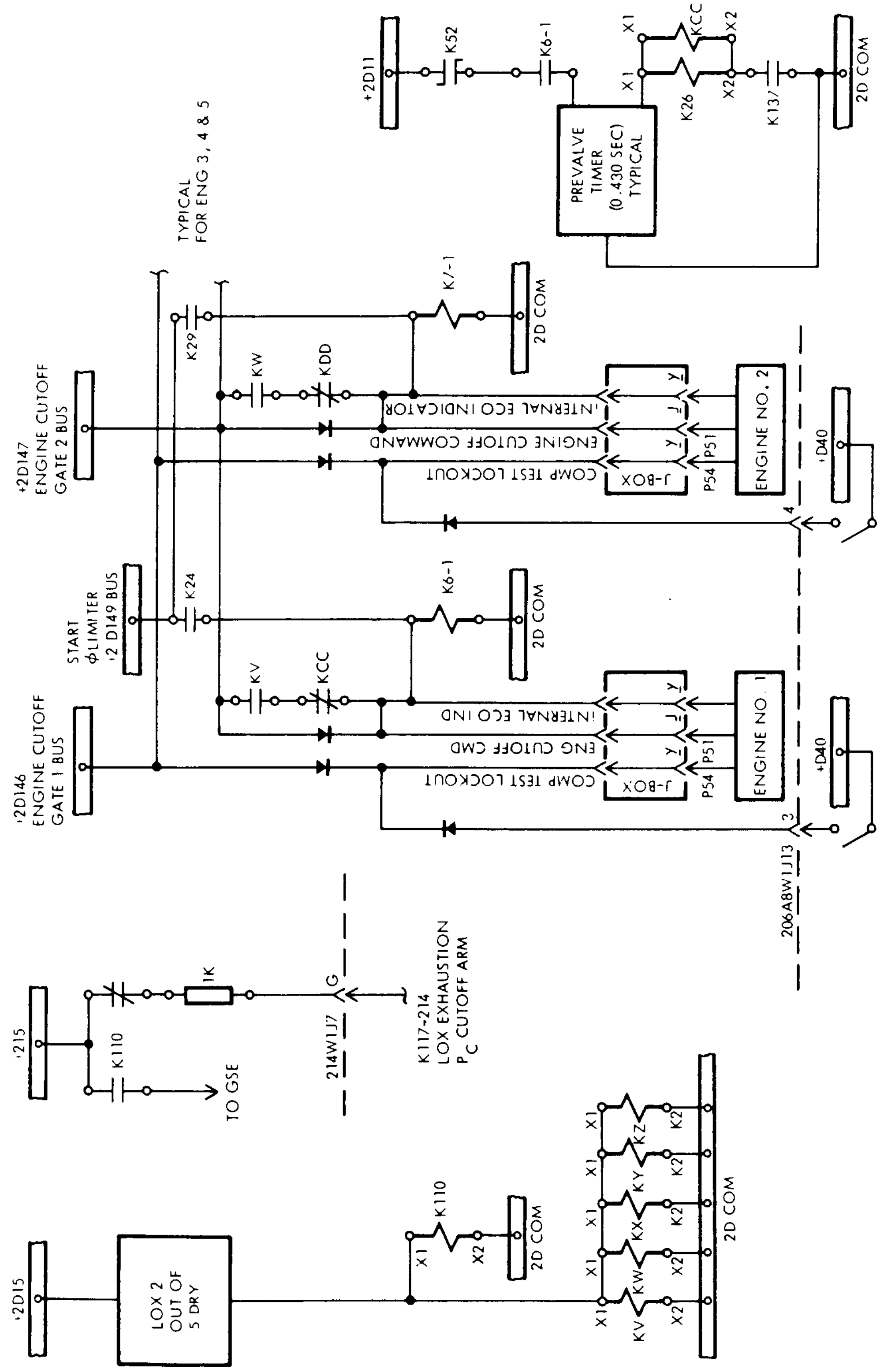


Figure 10.2-103. LOX Exhaustion P_C Cutoff

The one-engine-out capability during mainstage operation will be retained. An additional feature is required in the case of an early engine-out to prevent that engine from cutting off the remaining four engines at LOX exhaustion cutoff arm (two out of five dry sensors). Figure 10.2-103 shows the necessary circuitry to mechanize the new requirement. If Engine 1 terminates prematurely during mainstage operation, an internal engine cutoff signal will energize relay K6-1 and initiate the 430 millisecond prevalve timer for Engine 1. Relay K-137 contact will be in the closed position during this operation. Relay Kcc will be energized by the run-out of the prevalve timer.

Relay contact Kv will remain open until LOX exhaustion ECO arm, preventing Engine 1 from terminating the other engines early. Relay contact Kcc will be opened by the Engine 1 prevalve timer run-out, preventing the other four engines from being terminated by the 2-out-of-5 LOX dry sensor signal that closes relay contacts Kv, Kw, Kx, Ky, and Kz. The remaining four engines will continue to burn until LOX exhaustion produces an internal engine cutoff signal. The signal will be transmitted to cut off the remaining three engines before the prevalve timer, for that engine, can run out. The normal external switch selector command and emergency engine cutoff circuitry remain unchanged. Five relays will be required to implement this change.

The following stage harnesses in the 206 container area will be deleted: 206A7W12, 206W13, 206W210, 206W211, 206W212, 206W213, 206W214, 206W215, 206W419, 206W420, 206W421, and 206W423. In the 200 container area the following wire harnesses will be deleted: 200W9, 200W10, 200W12, 200W14, 200W17, 200W19 and 200W23.

The additional electrical control relays are required to incorporate the J-2S engine for the LOR mission because of the redundant idle mode start, redundant mainstage start, mainstage operations, LOX exhaustion engine cutoff, and redundant engine ready bypass functions.

2. Fuel Recirculation System. The fuel recirculation system is not required for J-2S engine start and the effect on the electrical control system is to delete functionally 10 relays as listed below. The relays which are no longer required will be utilized to implement control of new J-2S engine functions.

Control Relays Functionally Deleted

Relay Type	Part Number	Quantity
II	ME455-0005-	2
III	ME455-0005-	1
IV	ME455-0005-	7

3. LOX Recirculation System. The LOX recirculation system is not required for the J-2S engine start and the effect on the electrical control system is to delete functionally five relays as listed below. These spared relays will be used to implement new functions for the J-2S engine control.

Control Relays Functionally Deleted

Relay	Part Number	Quantity
Type III	ME455-005-	5

4. Separation System. Separation system EBW firing units 1A and 1B, which are used for the ullage motors, are not required because the ullage motors will be deleted from the stage. This permits six components to be deleted from the electrical control system as listed below.

Components Deleted

Part Name	Part Number	Number Deleted
Ullage EBW Firing Units (Government-furnished)	40M39515-119	2
Ullage Pulse Sensors (GFP)	40M02852	2
Ullage Trigger Control Relays	ME455-0005-0010	2

5. Electrical Interface Requirements. The list below defines connector, pin and functions as modified to incorporate the J-2S engine for the LOR mission.

Aft Umbilical Connector No. 1 (206A8W2J1)

Pin Letter	J-2S Functions
4	Spare
5	Spare
8	Command Engine Ready Bypass Relays Isolation Test Return
10	Spare
12	Command Engine Ready Bypass No. 1 Relays Isolation Test
22	Command Engine Ready Bypass No. 2 Relays Isolation Test
23	Command Engine 2 Mainstage Cutoff No. 1
24	Command Engine 3 Mainstage Cutoff No. 1
25	Command Engine 4 Mainstage Cutoff No. 1
26	Command Engine 5 Mainstage Cutoff No. 1
27	Measurement Engine Start Lockout Ind
44	Command LH_2 Prevalves Close
53	Command Engine 1 Mainstage Cutoff No. 1
56	Spare

Aft Umbilical Connector No. 2 (206A8W1J1)

Pin Letter	J-2S Function
6	Spare
7	Spare
11	Spare
19	Spare
21	Command Mainstage Cutoff No. 1 Relay Isolation Test
30	Spare
32	Spare
36	Spare
38	Spare
42	Command Mainstage Cutoff No. 2 Relay Isolation Test
47	Spare
51	Spare
57	Spare
59	Spare

Aft Umbilical Connector No. 3 (206A8W1J2)

Pin Letter	J-2S Function
8	Spare
23	Spare

Aft Umbilical Connector No. 4 (206A8W3J1)

Pin Letter	J-2S Function
3	Measurement Engine 4 Mainstage OK Bypass Throttle Enable
8	Measurement Engine 3 Mainstage OK Bypass Throttle Enable
17	Measurement Engine 5 Mainstage OK Bypass Throttle Enable
18	Measurement Engine 1 Mainstage OK Bypass Throttle Enable
22	Measurement Engine 2 Mainstage OK Bypass Throttle Enable
27	Measurement Engine 4 No. 1 SPTS; No. 2 EBW (0-5v)
42	Measurement Engine 2 No. 1 SPTS; No. 2 EBW (0-5v)
45	Measurement Engine 4 No. 1 SPTS; No. 1 EBW (0-5v)
46	Measurement Engine 3 No. 1 SPTS; No. 1 EBW (0-5v)
47	Measurement Engine 3 No. 1 SPTS; No. 2 EBW (0-5v)
48	Measurement Engine 5 No. 1 SPTS; No. 1 EBW (0-5v)
49	Measurement Engine 5 No. 1 SPTS; No. 2 EBW (0-5v)
51	Measurement Engine 1 No. 1 SPTS; No. 2 EBW (0-5v)
52	Measurement Engine 1 No. 1 SPTS; No. 1 EBW (0-5v)
53	Measurement Engine 2 No. 1 SPTS; No. 1 EBW (0-5v)

Aft Umbilical Connector No. 5 (206A8W2J2)

Pin Letter	J-2S Function
18	Spare
19	Spare
28	Spare
29	Spare
33	Spare
34	Spare
36	Spare
37	Spare
38	Spare
50	Spare

Aft Umbilical Connector No. 6 (206A8W8J1)

Pin Letter	J-2S Function
4	Command Engine 4 Component Test Idle Mode Solenoid
8	Command Engine 3 Component Test Idle Mode Solenoid
17	Command Engine 1 Component Test Idle Mode Solenoid
21	Command Engine 2 Component Test Idle Mode Solenoid
25	Command Engine 4 Component Test Mainstage Start Solenoid
29	Command Engine 3 Component Test Mainstage Start Solenoid
32	Command Engine 5 Component Test Mainstage Start Solenoid
34	Command Engine 5 Component Test Idle Mode Solenoid
36	Command Engine 1 Component Test Mainstage Start Solenoid
41	Command Engine 2 Component Test Mainstage Start Solenoid

Aft Umbilical Connector No. 7 (206A8W9J1)

Pin Letter	J-2S Function
3	Spare
4	Spare
5	Spare
20	Spare
21	Spare
22	Spare
23	Spare
26	Spare
27	Spare
28	Spare
29	Spare
39	Spare
40	Spare
41	Spare

Aft Umbilical Connector No. 7 (206A8W9J1) (Cont)

Pin Letter	J-2S Function
42	Spare
43	Spare
44	Spare
45	Spare
46	Spare
52	Spare
53	Spare
54	Spare
55	Spare
56	Spare
60	Spare

Aft Umbilical Connector No. 8 (206A8W4J1)

Pin Letter	J-2S Function
1	Spare
2	Spare
3	Spare
4	Spare
5	Spare
13	Measurement Engine 1 Ignition Detection No. 2 Simulate
14	Measurement Engine 2 Ignition Detection No. 2 Simulate
15	Measurement Engine 5 Ignition Detection No. 2 Simulate
34	Measurement Engine 3 Ignition Detection No. 2 Simulate
35	Measurement Engine 4 Ignition Detection No. 2 Simulate

Special Test Connector No. 7 (206A40J14)

Pin Letter	J-2S Function
D	Engine 1 Throttle Enable Mainstage OK Bypass
L	Engine 1 Ignition Detection No. 2 Simulate
M	Engine 1 No. 1 Heater On
J	Engine 1 No. 1 SPTS, No. 2 EBW (0-5v)
K	Engine 1 No. 1 SPTS, No. 1 EBW (0-5v)

Special Test Connector No. 8 (206A40J16)

Pin Letter	J-2S Function
D	Engine 2 Throttle Enable Mainstage OK Bypass
L	Engine 2 Ignition Detection No. 2 Simulate

Special Test Connector No. 8 (206A40J16) (Cont)

Pin Letter	J-2S Function
M	Engine 2 No. 1 Heater On
J	Engine 2 No. 1 SPTS No. 2 EBW (0-5v)
K	Engine 2 No. 1 SPTS No. 1 EBW (0-5v)

Special Test Connector No. 9 (206A40J18)

Pin Letter	J-2S Function
D	Engine 3 Throttle Enable Mainstage OK Bypass
L	Engine 3 Ignition Detection No. 2 Simulate
M	Engine 3 No. 1 Heater On
J	Engine 3 No. 1 SPTS No. 2 EBW (0-5v)
K	Engine 3 No. 1 SPTS No. 1 EBW (0-5v)

Special Test Connector No. 10 (206A40J20)

Pin Letter	J-2S Function
D	Engine 4 Throttle Enable Mainstage OK Bypass
L	Engine 4 Ignition Detection No. 2 Simulate
M	Engine 4 No. 1 Heater On
J	Engine 4 No. 1 SPTS No. 2 EBW (0-5v)
K	Engine 4 No. 1 SPTS No. 1 EBW (0-5v)

Special Test Connector No. 11 (206A40J22)

Pin Letter	J-2S Function
D	Engine 5 Throttle Enable Mainstage OK Bypass
L	Engine 5 Ignition Detection No. 2 Simulate
M	Engine 5 No. 1 Heater On
J	Engine No. 1 SPTS No. 2 EBW (0-5v)
K	Engine No. 1 SPTS No. 1 EBW (0-5v)

J-2S Engine Interface Connector P51

Pin Letter	J-2S Function
X	Mainstage Start Signal
g	Mainstage Cutoff Command
d	Mainstage Cutoff Monitor
h	Mainstage OK Bypass (Throttle Enable)

J-2S Engine Interface Connector P51 (Cont)

Pin Letter	J-2S Function
I	Pneumatic System Vent
K	Pneumatic System Vent
H	Reserved - Heater Power (+)
M	Reserved - Heater Power (+)
N	Reserved - Heater Power (+)
C	Reserved - Heater Power (-)
D	Reserved - Heater Power (-)
E	Reserved - Heater Power (-)

J-2S Engine Interface Connector P54

Pin Letter	J-2S Function
R	Component Test-Idle Mode Solenoid
g	Component Test-Mainstage Start Solenoid
S	Simulate Signal-Ignition Detect No. 2
J	Monitor - Idle Mode Control On
K	Monitor-Mainstage Start Control On
j	Monitor - SPTS Initiated
a	Monitor-Mainstage Cutoff Lock-in
b	Spare Wire
k	Monitor - No. 1 SPTS Ready
c	Monitor - SPTS Armed
d	Monitor - No. 1 Heater On (Reserved)
N	Monitor - No. 2 Heater On (Reserved)
q	Monitor - No. 1 SPTS; No. 1 EBW Monitor (0-5v)
r	Monitor - No. 1 SPTS; No. 2 EBW Monitor (0-5v)
B	Reserved - Heater 2 Simulate Signal
F	Spare Wire
G	Spare Wire
A	Reserved - Heater 1 Simulate Signal
e	Reserved - Redundant Engine Start Signal
C	Reserved - Redundant Mainstage Cut-Off Spare Signal
H	Reserved - Redundant Engine Ready Bypass Signal
O	Reserved - Redundant Mainstage Start Signal

J-2S Engine Interface Connector No. P108

Pin Letter	J-2S Function
G	Measure, Helium Tank Gas Temperature Resistance Thermometer, A-Sensor, Sensor Output
H	Spare
I	Measure, Helium Tank Gas Temperature Resistance Thermometer, A-Sensor, Input Common

J-2S Engine Interface Connector No. P108 (Cont)

Pin Letter	J-2S Function
J	Measure, Helium Tank Gas Temperature Resistance, Thermometer, A-Sensor, Output Common
K	Spare
L	Spare
M	Spare
N	Spare
O	Spare
P	Spare
R	Spare
S	Spare
T	Spare
U	Spare
V	Spare
n	Spare
p	Spare
s	Measure, Main Fuel Injection No. 2 Temperature Resistance Thermometer
t	Measure, Main Fuel Injection No. 1 Temperature Resistance Thermometer
u	Spare
v	Measure, Main Fuel Injection No. 2 Temperature Resistance Thermometer
w	Measure, Main Fuel Injection No. 1 Temperature Resistance Thermometer
x	Measure, Main Fuel Injection No. 2 Temperature Resistance Thermometer
y	Measure, Main Fuel Injection No. 1 Temperature Resistance Thermometer

J-2S Engine Interface Connector No. P105

Pin Letter	J-2S Function
C	Spare
J	Spare Shield Return

J-2S Engine Interface Connector No. P107

Pin Letter	J-2S Function
K	Measure Voltage On Propellant Utilization Valve For Position Potentiometer
L	Measure Voltage On Propellant Utilization Valve For Position Potentiometer

J-2S Engine Interface Connector No. P107 (Cont)

Pin Letter	J-2S Function
H	Supply, Positive 5 vdc Excitation for Hot-gas Tapoff Valve Position Potentiometer
M	Supply, Positive 5-vdc Excitation for Idle Mode Valve Position Potentiometer
N	Supply, Positive 28-vdc Duplicate Power for Instrumentation System Valve Position Switches
S	Supply, Positive 5-vdc Excitation for Fuel Bypass Valve Position
T	Measure, Idle Mode Valve Limit Switch Closed Signal
W	Spare
X	Measure, Idle Mode Valve Position Potentiometer Signal Output
Z	Measure, Idle Mode Valve Limit Switch Open Signal
c	Measure, Hot Gas Tapoff Valve Position Limit Switch Open Signal
d	Measure, Hot Gas Tapoff Valve Position Limit Switch Closed Signal
g	Measure, Fuel Bypass Valve Position Limit Switch Open Signal
i	Measure, Hot Gas Tapoff Valve Position Potentiometer Signal Output
n	Measure, Fuel Bypass Valve Position Limit Switch Closed Signal
p	Measure, Fuel Bypass Valve Position Potentiometer Signal Output

J-2S Engine Interface Connector No. P106

Pin Letter	J-2S Function
J	Measure, Dummy Pressure Transducer Signal Output
R	Spare

J-2S Engine Interface Connector No. P106 (Cont)

Pin Letter	J-2S Function
E	Command Dummy Pressure Transducer 20-Percent Calibration and Checkout Voltage Input
F	Command Dummy Pressure Transducer 80-Percent Calibration and Checkout Voltage Input
b	Measure, Idle Mode Chamber Pressure Transducer Signal Output
c	Command, Idle Mode Chamber Pressure Transducer 20-Percent Calibration and Checkout Voltage Input
d	Command, Idle Mode Chamber Pressure Transducer 80-Percent Calibration and Checkout Voltage Input
h	Measure, Helium Tank No. 1 Pressure Transducer Signal Output
i	Command, Helium Tank No. 1 Pressure Transducer 20-Percent Calibration and Checkout Voltage
r	Command, Helium Tank No. 1 Pressure Transducer 80-Percent Calibration and Checkout Voltage Input

b. LEO Mission. The LOR mission requirements described previously will have the following additional requirements for the LEO mission.

1. Engine System. An inflight instrument unit (IU) computer termination of the J-2S engine mainstage operation into the idle mode (not an engine cutoff) will be required for the LEO mission. A computer mainstage cutoff backup will be provided by LH_2 and LOX tank sensors (2 out of 5) dry indication signal. A 10-second idle mode operation will be required before the 45-minute coast period to allow the flight control system to arrest the vehicle attitude transient level to a maximum of $\pm$ 2.5-degree attitude error. The (RCS) will provide attitude control during the 45-minute coast period, and is limited to a small body rate and $\pm$ 2.5-degree attitude error. The IU computer will issue an "all engine cutoff" command that is timed 10 seconds from mainstage cutoff for insertion into the 100 nautical mile orbit. Figure 10.2-104 depicts a mainstage to idle mode thrust decay curve to indicate flight events during this period.

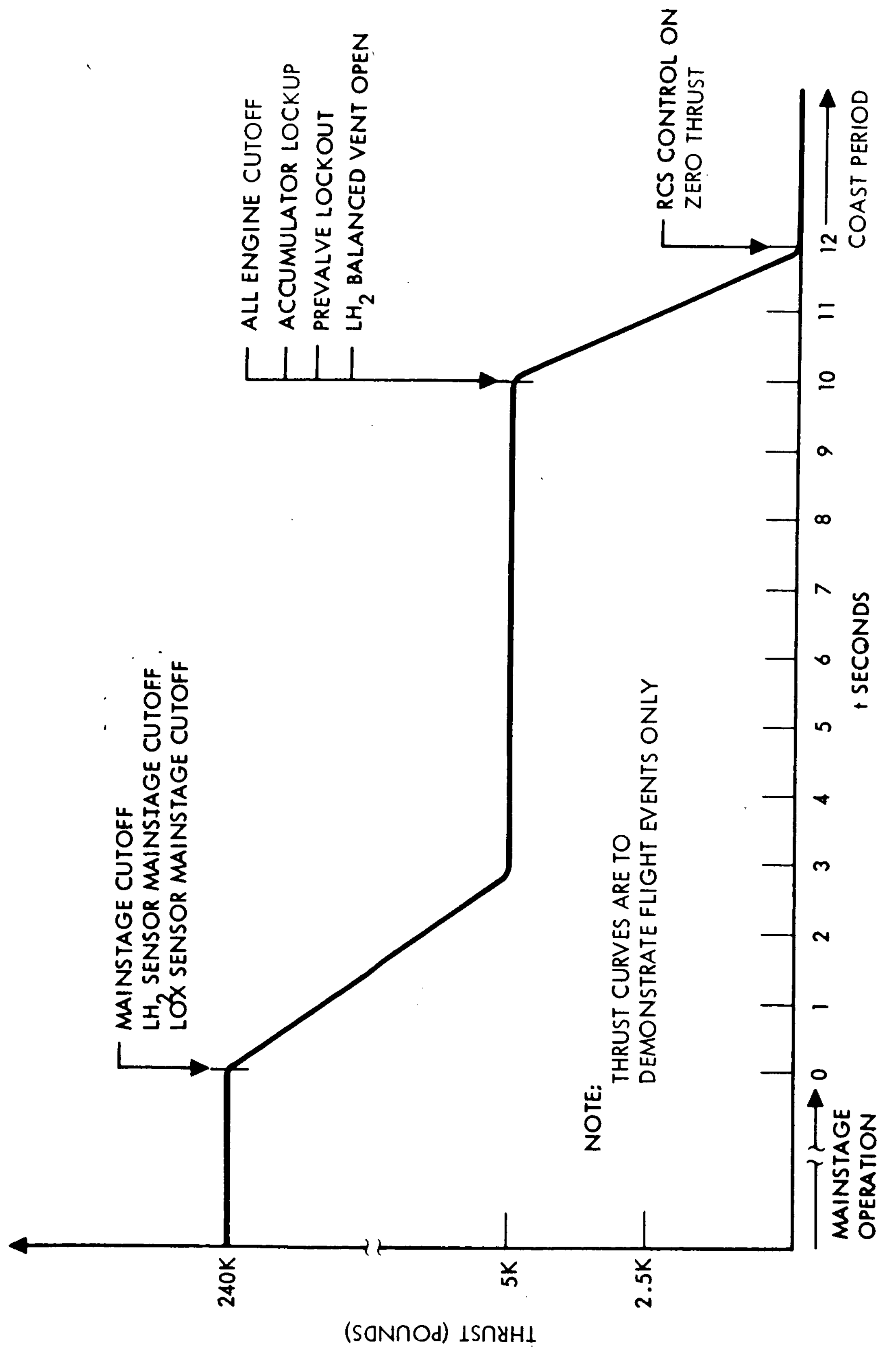


Figure 10.2-104. Flight Events--Mainstage to Idle Mode Cutoff

During the LEO mission, new switch selector commands are required for mainstage cutoff, thrust vector control, LH_2 balanced vent system, restart, and extended idle mode operation. The following switch selector changes are required:

New S-II Switch Selector Commands

"Mainstage Cutoff"
"Mainstage Cutoff Reset"
"Restart Arm"
"Restart Arm Reset"
"LH_2 Balanced Vent System Arm"
"TVC Auxiliary Hydraulic Pumps No. 1 and No. 2 On"
"TVC Auxiliary Hydraulic Pumps No. 3 and No. 4 On"
"TVC Auxiliary Hydraulic Pumps No. 1 and No. 2 Off"
"TVC Auxiliary Hydraulic Pumps No. 3 and No. 4 Off"

S-II Switch Selector Commands Deleted

"LH_2 Step Pressurization"

Two new switch selector commands will be required for the mainstage operation cutoff (mainstage operation cutoff and mainstage operation cutoff reset). Figure 10.2-105 is a schematic of the circuitry necessary to mechanize this new function. The switch selector mainstage cutoff or LH_2 or LOX dry sensor signal will terminate mainstage operation into idle mode. The LH_2 and LOX tank sensors will be moved up in the tanks to allow sufficient remaining propellants for restart and idle mode operation to complete the 300-nautical-mile orbit mission. The mainstage cutoff command will set the Km and Kn relays. The Km relay contact will connect the +2D11 main bus to the mainstage cutoff bus No. 1 which applies mainstage cutoff No. 1 to the engine. The engine mainstage cutoff-on signal energizes Kq relay which provides a hardwire indication of mainstage cutoff via Kq contacts to the S-II/S-IVB interface. The Kn relay contact provides a redundant mainstage cutoff No. 2 to the engine utilizing the +2D21 instrumentation bus.

Mainstage cutoff bus No. 1 or No. 2 will terminate J-2S engine mainstage operation and return the engine to idle mode operation. Telemetry measurements will be provided from each mainstage cutoff bus. Mainstage cutoff reset switch selector command or the +2D145 engine control relay reset bus will remove power from the two mainstage cutoff buses by resetting Km and Kn relays. Two latching and five non-latching type control relays are required to implement the new redundant mainstage cutoff function.

The prevalves will be locked out (open) beginning with "all engine cutoff" at the 100 nautical mile altitude and will remain locked open for the remainder of the LEO mission. Figure 10.2-106 depicts the all engine cutoff circuitry necessary to terminate idle mode operation prior to the 45-minute coast period and to provide the prevalve lock out signal. Switch selector channel 18 will set K22 relay which applies the main and instrumentation bus (diode isolated) via K22 contacts to engine cutoff relays K70-1, K70-2, K71-1

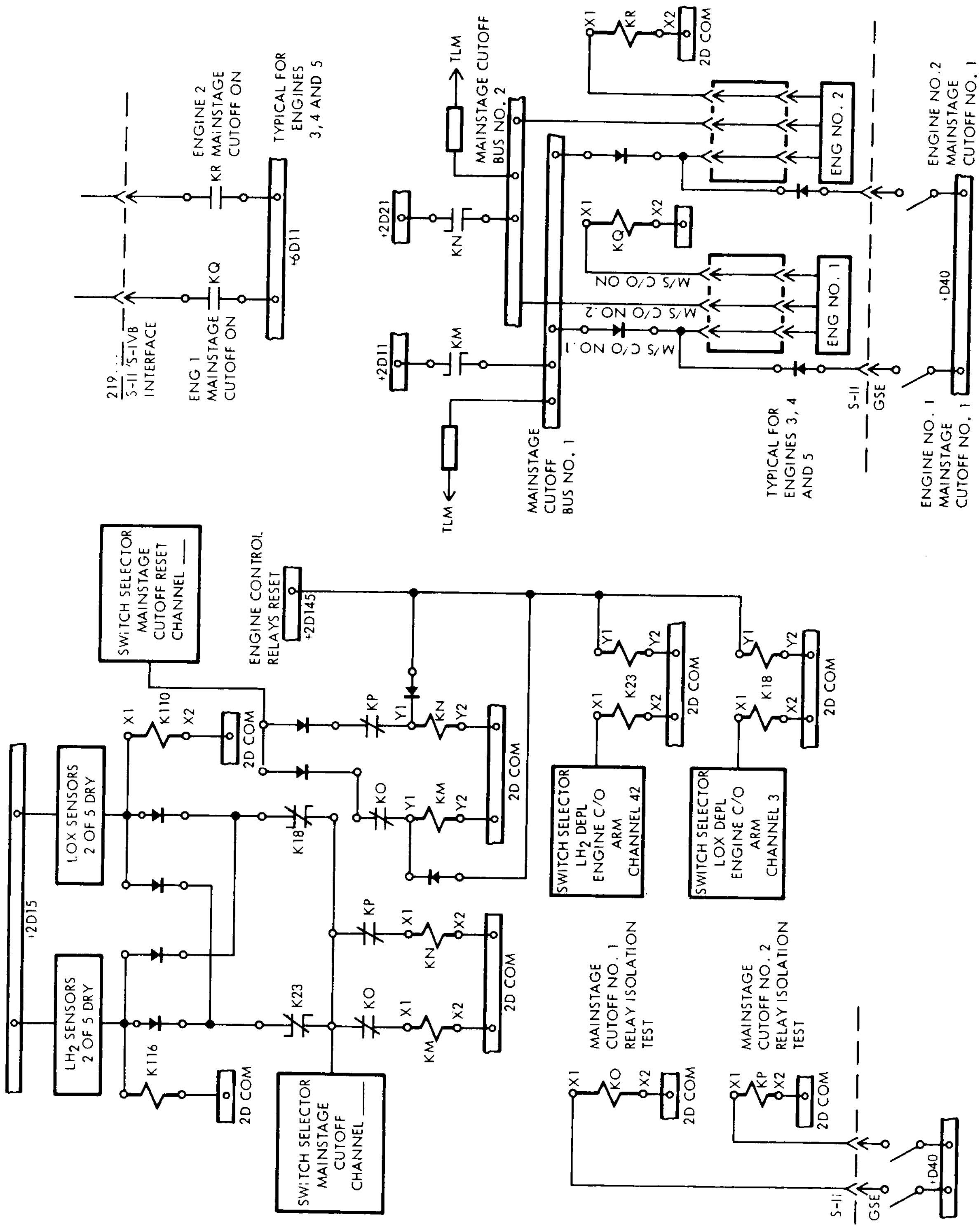


Figure 10.2-105. Mainstage Cutoff Circuitry

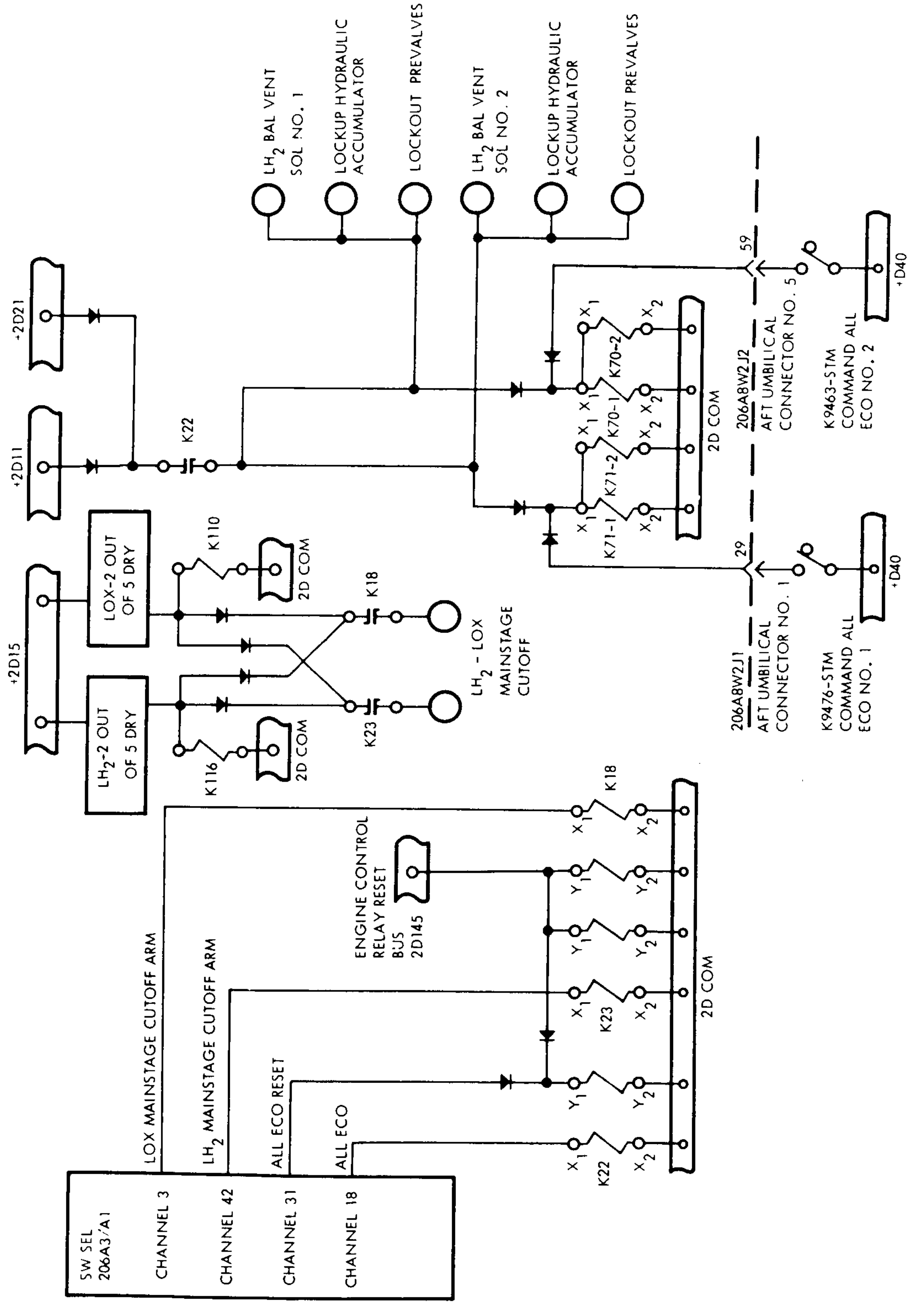


Figure 10.2-106. All-Engine Cutoff Circuitry

and K71-2. The same +28-vdc signal will be utilized to lock open the prevalves, lock up the hydraulic accumulators, and open the LH_2 balanced vent system. Switch selector commands will be required to arm the LH_2 and LOX sensor mainstage cutoff circuitry prior to switch selector command for mainstage cutoff. During mainstage operation the one engine out and emergency engine cutoff capability will be retained. In case of an early engine out during mainstage operation, the prevalve for that engine will close at the run out of the 430-millisecond prevalve timer. The remaining four engines will continue mainstage operation until mainstage cutoff. For the LEO mission an internal engine cutoff will not terminate the remaining engines under any circumstance. Figure 10.2-104 (shown earlier) shows a thrust decay curve and indicates events beginning with mainstage cutoff and continuing until start of the 45-minute coast period. The lock out of the prevalves is indicated on this curve.

The J-2S engines have been designed for inflight restart, and during the LEO mission this new function will require two commands (restart arm and restart arm reset) from the switch selector. Figure 10.2-107 shows the flight events prior to restart and Figure 10.2-108 shows the circuitry necessary to provide restart control. A switch selector S-II restart arm command will set Kaa relay which applies the +2D11 main bus to the +2D145 engine control relay reset bus. A switch selector restart arm reset command will be required after the S-II restart arm reset command to assure the prevalve lock out (open) is maintained during engine restart and extended idle mode operation. An all engine start command will now restart the engines into the extended idle mode operation. One latching type control relay will be required to implement the restart arm function.

The J2-S engines during the LEO mission will only be cutoff (zero thrust) by the switch selector channel 18 all engine cutoff command or the PD and EDS emergency termination. Figure 10.2-109 depicts the cutoff circuitry necessary to provide zero thrust cutoff. Engine cutoff bus No. 1 or No. 2 will provide each engine with redundant cutoff signals. The internal engine cutoff indication from an individual engine will energize a relay (K6-1 for Engine 1) resulting in Engine 1 prevalve closure. Blocking diodes prevent one engine from terminating another. Ten nonlatching type relays will be deleted from the LOR engine cutoff circuitry for the LEO mission.

2. RCS. The reaction control system (RCS) will be required during the coast period and J-2S engine restart for maintaining vehicle attitude control. There will be two S-IVB type RCS units mounted on opposite sides of the S-II aft skirt area. Figure 10.2-110 shows the RCS quad redundant series - parallel injector valves for one RCS engine. Two parallel injector valves must fail to prevent engine start and two series injector valves must fail to prevent cutoff. Figure 10.2-111 depicts the control circuitry necessary for one RCS engine. Relays K1, K2, K3 and K4 are DPDT with the four coils connected in parallel. The IU computer will provide the +28 VDC and return control signal necessary to sequence the RCS engines. One set of relay contacts will be used to control an oxidizer valve and the other a fuel valve. The +2D11 main bus shall provide the power to energize the solenoids. Telemetry measurements will be provided from one normally closed set of contacts

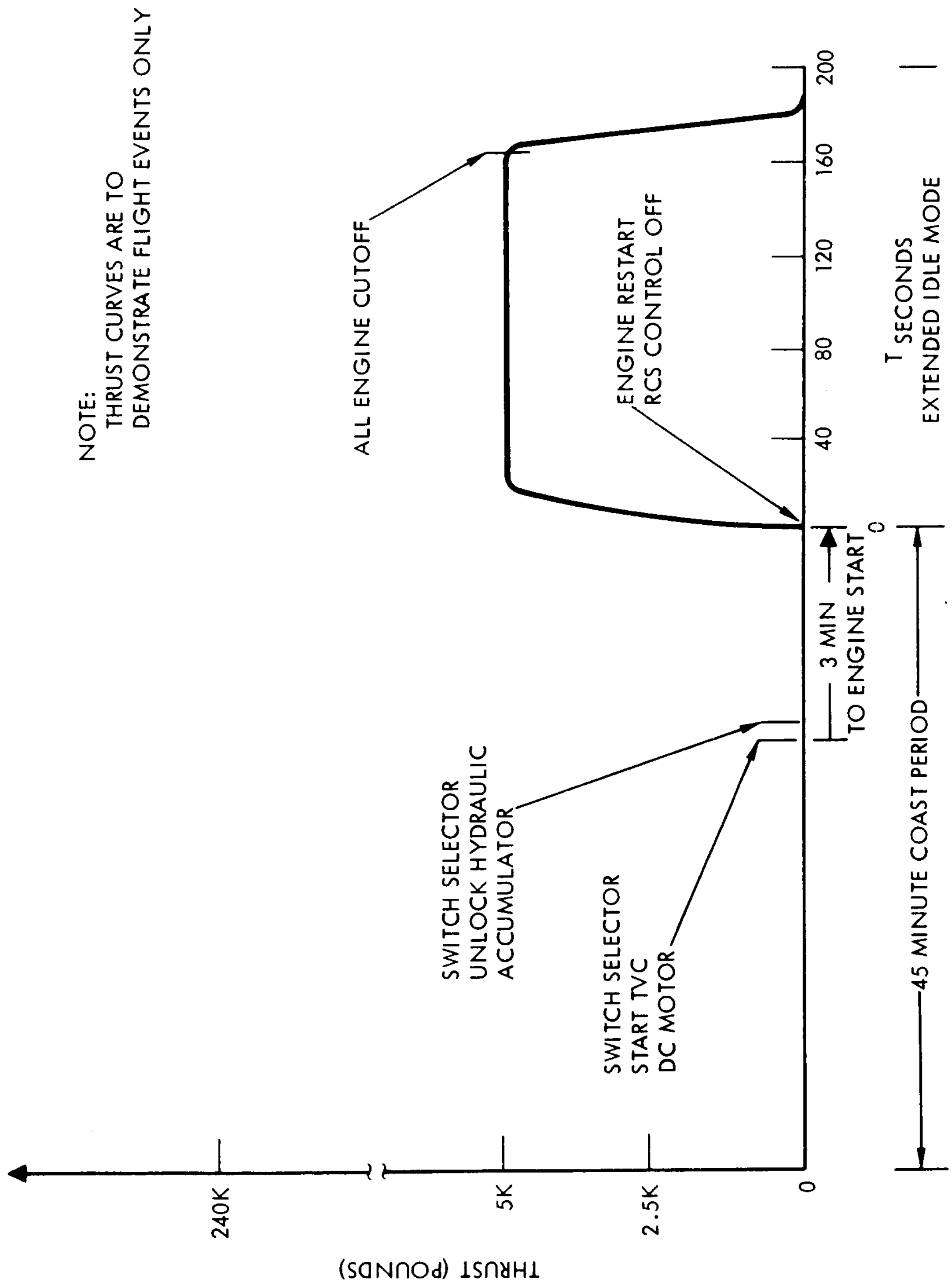


Figure 10. 2-107. Flight Events: Coast Period-Restart-Extended Idle Mode to Cutoff

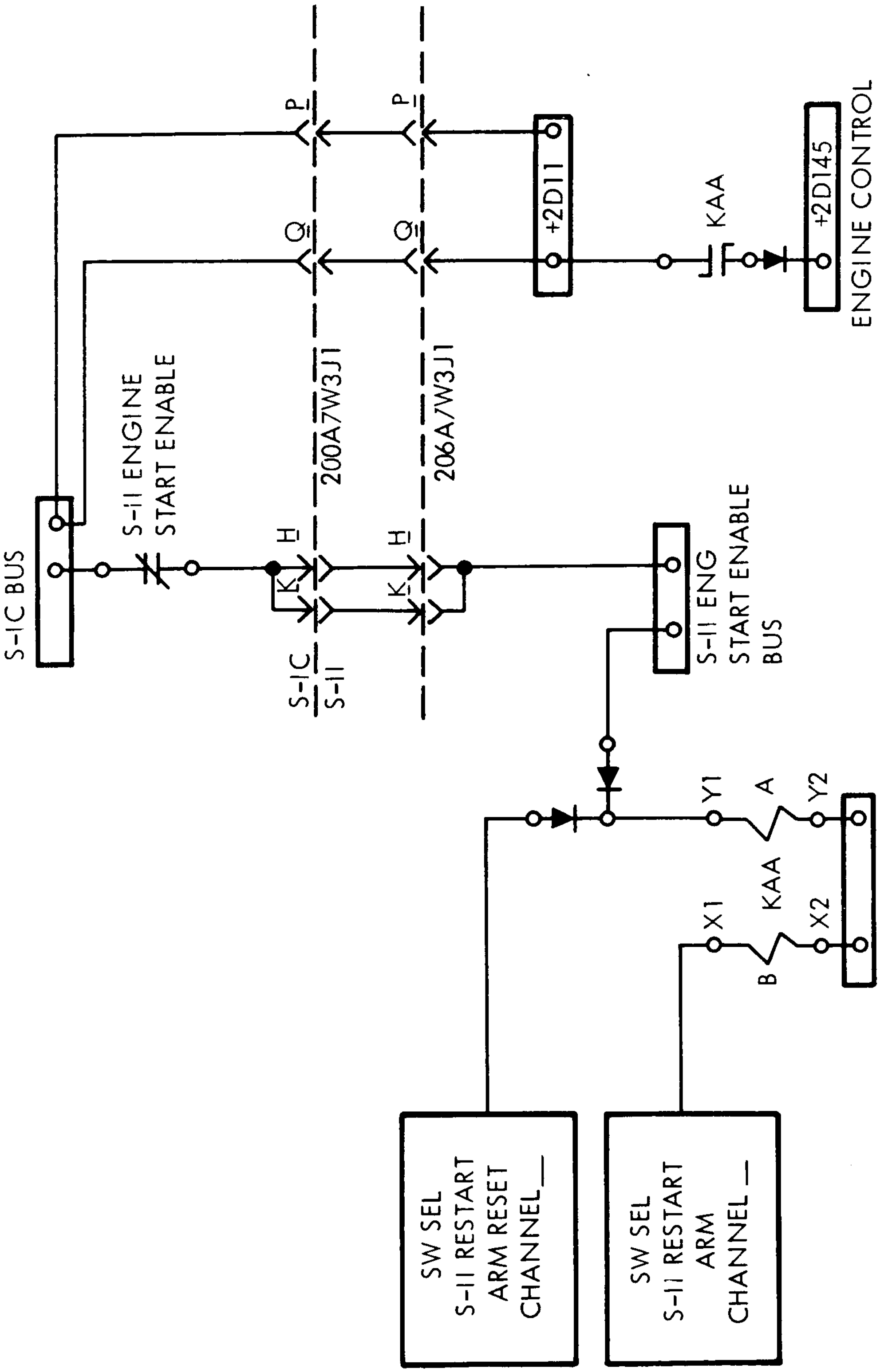


Figure 10.2-108. Restart Control Circuitry

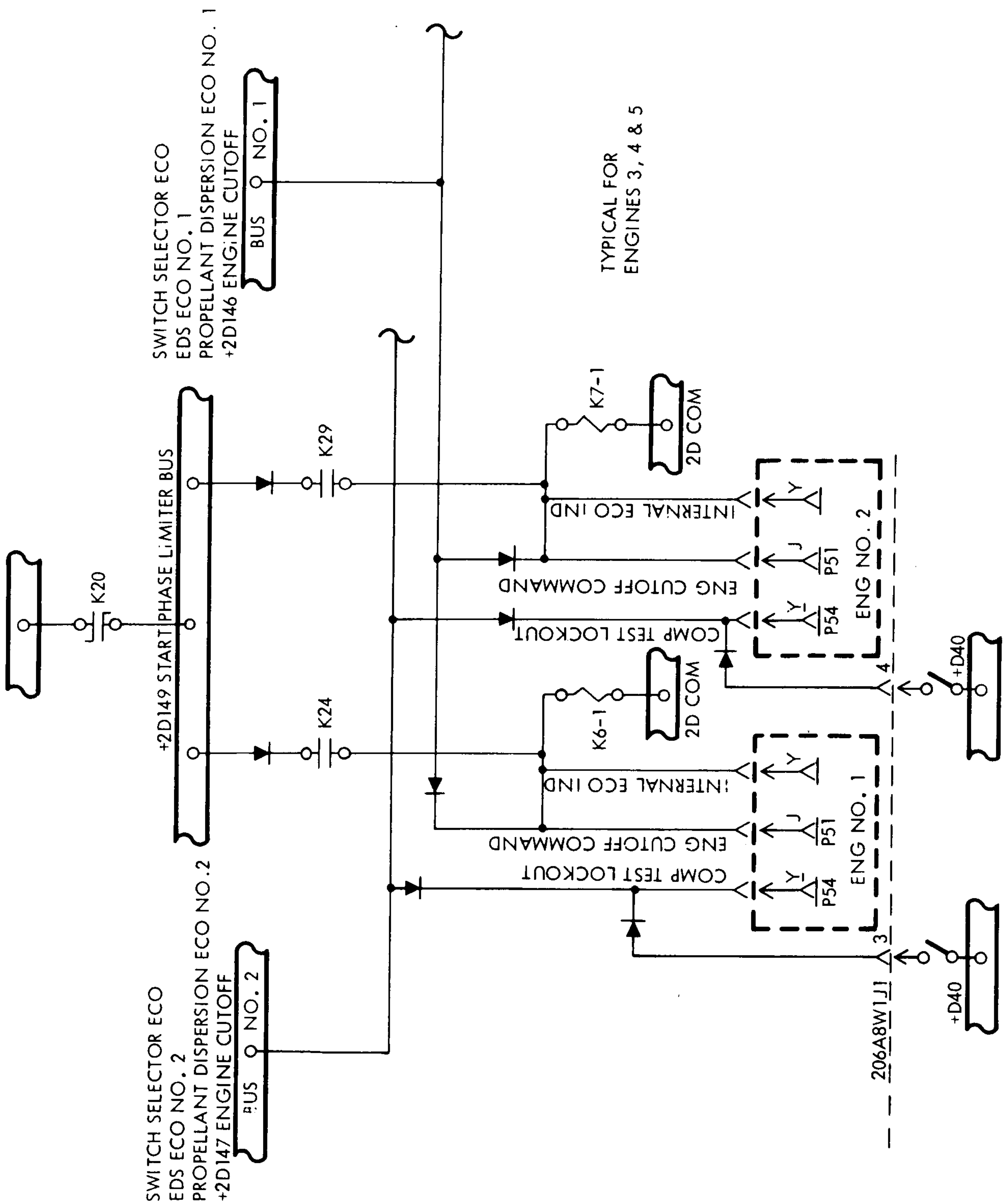


Figure 10.2-109. Zero-Thrust Cutoff Circuitry

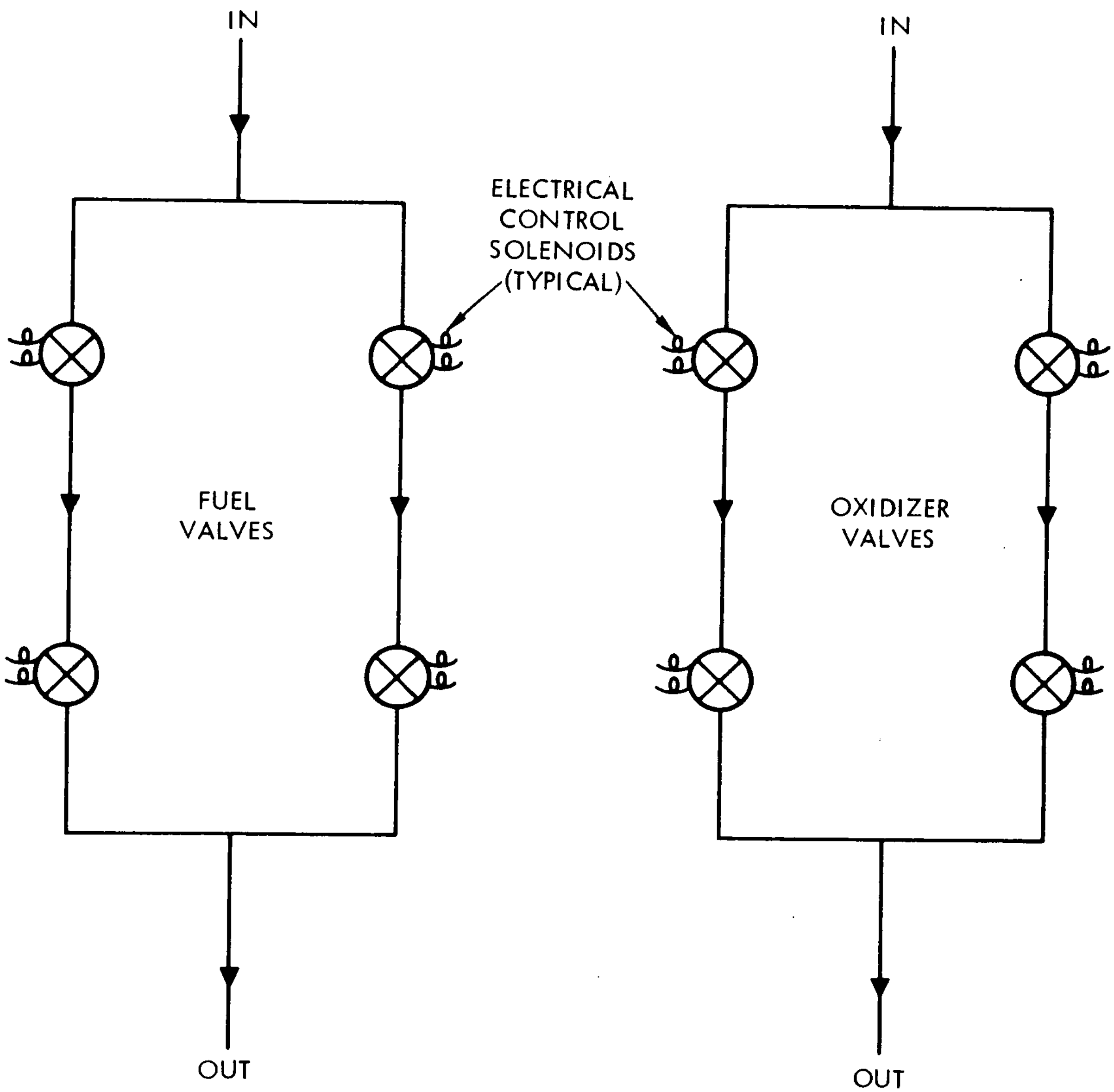


Figure 10. 2-110. RCS Quad Redundant Series--Parallel Injector Valves

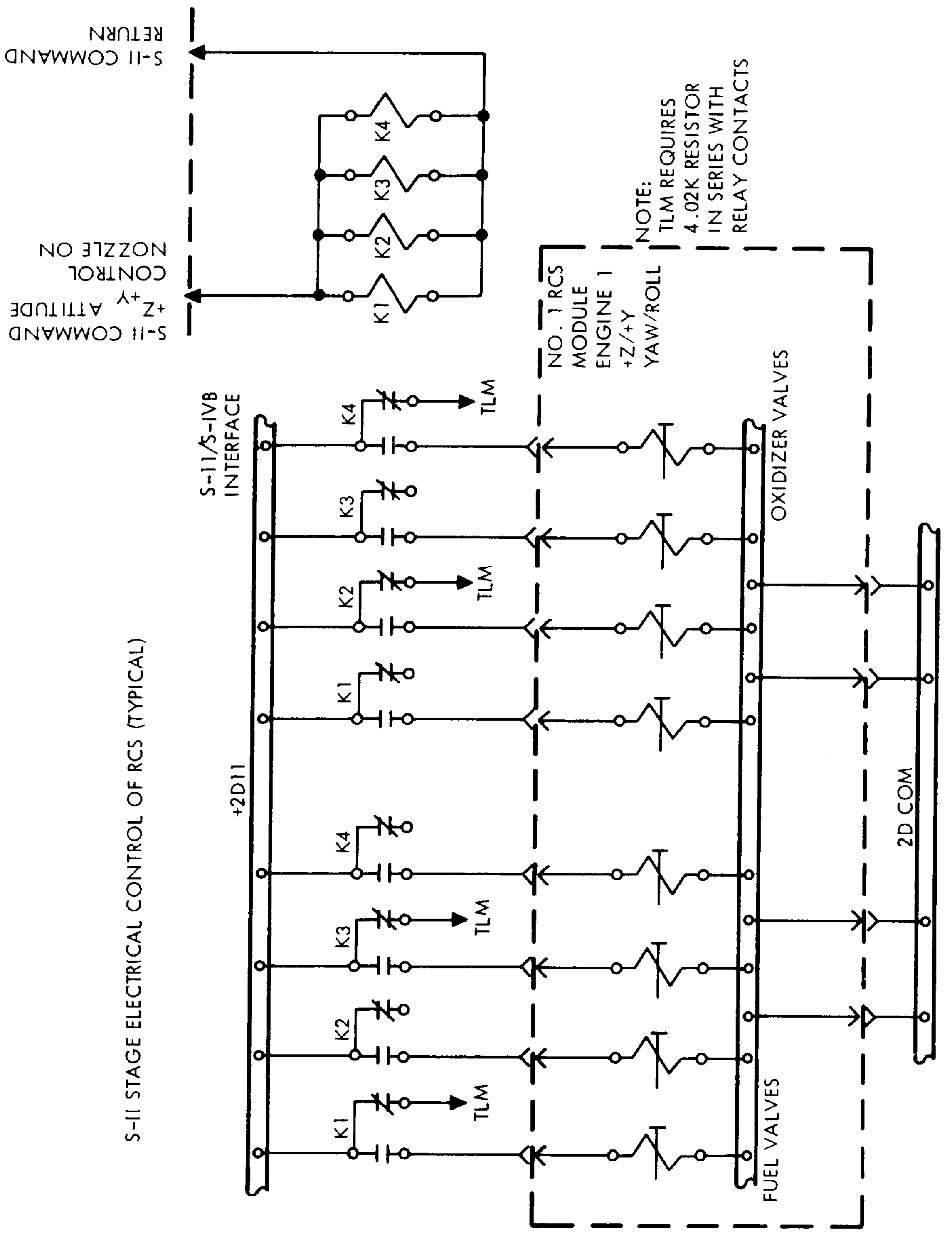


Figure 10.2-111. S-II Stage Electrical Control of RCS (Typical)

on each relay. There are three engines for each RCS unit, resulting in the addition of 24 nonlatching type control relays for both RCS units. The ground servicing and checkout will be provided at KSC for the RCS units. Figure 10.2-104 indicated when the IU computer will take control of the RCS unit and Figure 10.2-107 indicated when the IU computer will terminate control.

3. Flight Control System. During the LEO mission, the S-II flight control system will require a thrust vector control dc motor hydraulic pump system for engine gimbaling during the extended idle mode operation to complete the 300-nautical-mile orbit mission. The present S-IVB dc motor auxiliary hydraulic pump system is proposed for this function. This system will require a 56-vdc battery package, power transfer switch, and electrical control for start sequencing of the dc motors. Adjacent dc motor pumps will be powered from separate 56 vdc battery packages. Two new switch selector commands will be required to start the motors during the coast period at approximately three minutes before engine restart for temperature control of the hydraulic fluid and to build up the pressure in the accumulators. The first switch selector command will start dc motors for Engines 1 and 2 from separate 56 vdc battery packages. The second switch selector command will be delayed approximately 0.5 seconds to allow the starting surge to diminish sufficiently before applying power to start the dc motors for Engines 3 and 4. This process will reduce the high current drain on the batteries during motor start.

A new switch selector command will also be required to lock up the hydraulic accumulator reservoir at the time of "all engine cutoff" prior to the coast period during the LEO mission. Switch selector channel 12 command, "unlock the hydraulic accumulator", will be required during engine restart for the LEO mission.

Figure 10.2-105 showed a thrust decay curve to indicate events beginning with mainstage cutoff and continuing until start of the 45-minute coast period. Figure 10.2-107 depicted events beginning approximately three minutes prior to engine restart and continuing through the extended idle mode operation. The following steps provide the requirements for switch selector commands and electrical control relays to implement the flight events for engine gimbaling by the auxiliary dc motor pump system.

(a) There will be approximately a 10-second idle mode operation starting with termination of mainstage operation.

(b) The hydraulic energy in the accumulator at the termination of mainstage operation will provide for engine gimbaling during the 10-second idle mode period.

(c) The hydraulic accumulators will be locked up beginning with "all engine cutoff" at the 10-second idle mode termination.

(d) The hydraulic accumulators will be unlocked shortly after commanding the thrust vector control dc motor pump system on. The dc motor pump will be turned on three minutes prior to engine restart for hydraulic fluid warm up and to charge up the accumulator.

Figure 10.2-112 shows the circuitry necessary to mechanize control of the TVC battery heaters. The battery heater power is supplied from the +2D11 main bus and the heaters will automatically cycle on and off with temperature. The electrical control provided by the circuit in the figure will allow the heater circuit to be disabled to reduce the battery drain during S-II engine start. Relay K25 will be reset by the engine control relays reset bus +2D145 prior to liftoff. A set of closed contacts on K25 relay will reset relay KQQ and KRR. The contacts of KQQ and KRR relay will switch the +2D11 main bus to the TVC battery heater circuit. Switch selector command "all engine cutoff reset" Channel 31 will set relay K25 which will remove the TVC battery heater circuit from the +2D11 main bus to reduce the load during S-II engine start. Switch selector command "prevalves close arm" Channel 99 will reset K25 which will switch power to the heater circuits for the coast period. Switch selector command "all engine cutoff reset" Channel 31 will be required prior to S-II engine restart to remove the TVC heater load from the +2D11 main bus. The heater circuit will remain disabled because of the small time duration for the remainder of the LEO mission. Current limiting resistors are incorporated to provide GSE monitoring of the TVC battery heater power. Two new DPDT latching type relays are required to implement this change.

Figure 10.2-113 shows the control circuitry necessary to sequence the TVC auxiliary dc motor pump system on and off. A new switch selector command will set relay KNN. The KNN relay contacts will apply +28 vdc power from the +2D11 main bus to the motor start switches for TVC dc motors No. 1 and No. 2. The motor start switch contacts will apply +2DXX bus to start TVC motor No. 1 and +2DXY to start TVC motor No. 2. These two new TVC buses are powered from separate +56-vdc battery sources. A second switch selector command will set relay KPP approximately 0.5 seconds after relay KNN was set. Relay KPP contacts will connect TVC motor No. 3 to +2DXX bus and TVC motor No. 4 to +2DXY bus via the individual motor start switch contact. The +2D40 will provide summation monitoring of four motor start switches to indicate power off for all DC auxiliary hydraulic motor pumps. Switch selector commands are required to reset relays KNN and KPP for removing power from the four dc motors. Two DPDT latching type relays are required to implement this change. A telemetry measurement will be provided for each TVC motor to indicate that power is on.

4. Fuel Tank Pressurization System. A balanced LH_2 tank vent system will be required during the coast period of the LEO mission. A new switch selector command will be required to arm the LH_2 tank balanced vent system prior to the coast period. The "all engine cutoff bus" will provide the signal to open the balanced LH_2 tank vent system. The balanced vent system will be open for the remainder of the mission. Four latching type relays will be required to implement this new function.

Figure 10.2-114 depicts the circuitry necessary to implement the LH_2 tank balanced vent valve control system. A new switch selector command, "LH_2 balanced vent arm," will be required to arm the system by setting relays KJJ and KHH. The balanced vent switch selector arm command will be required

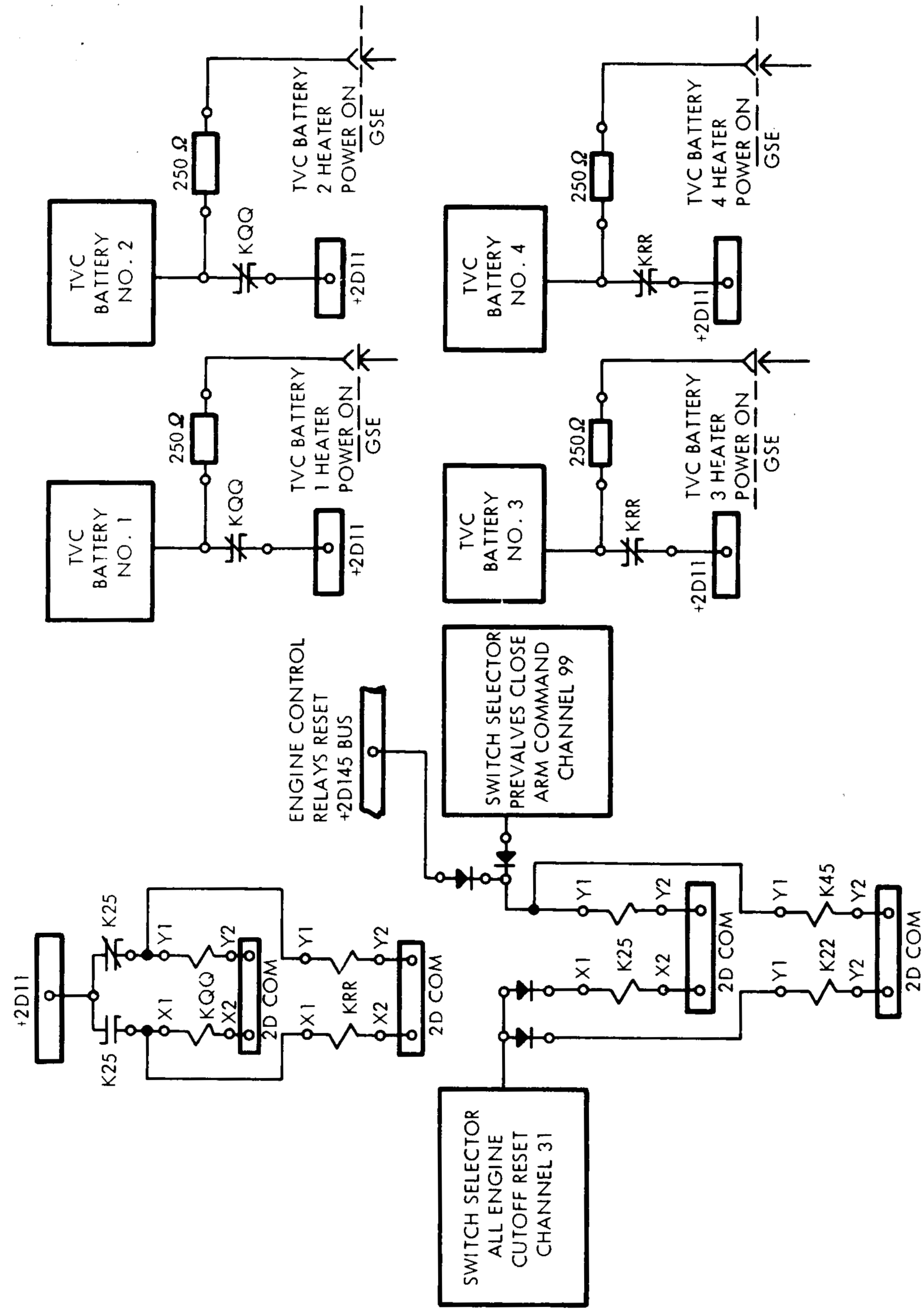


Figure 10.2-112. TVC Battery Heater Power

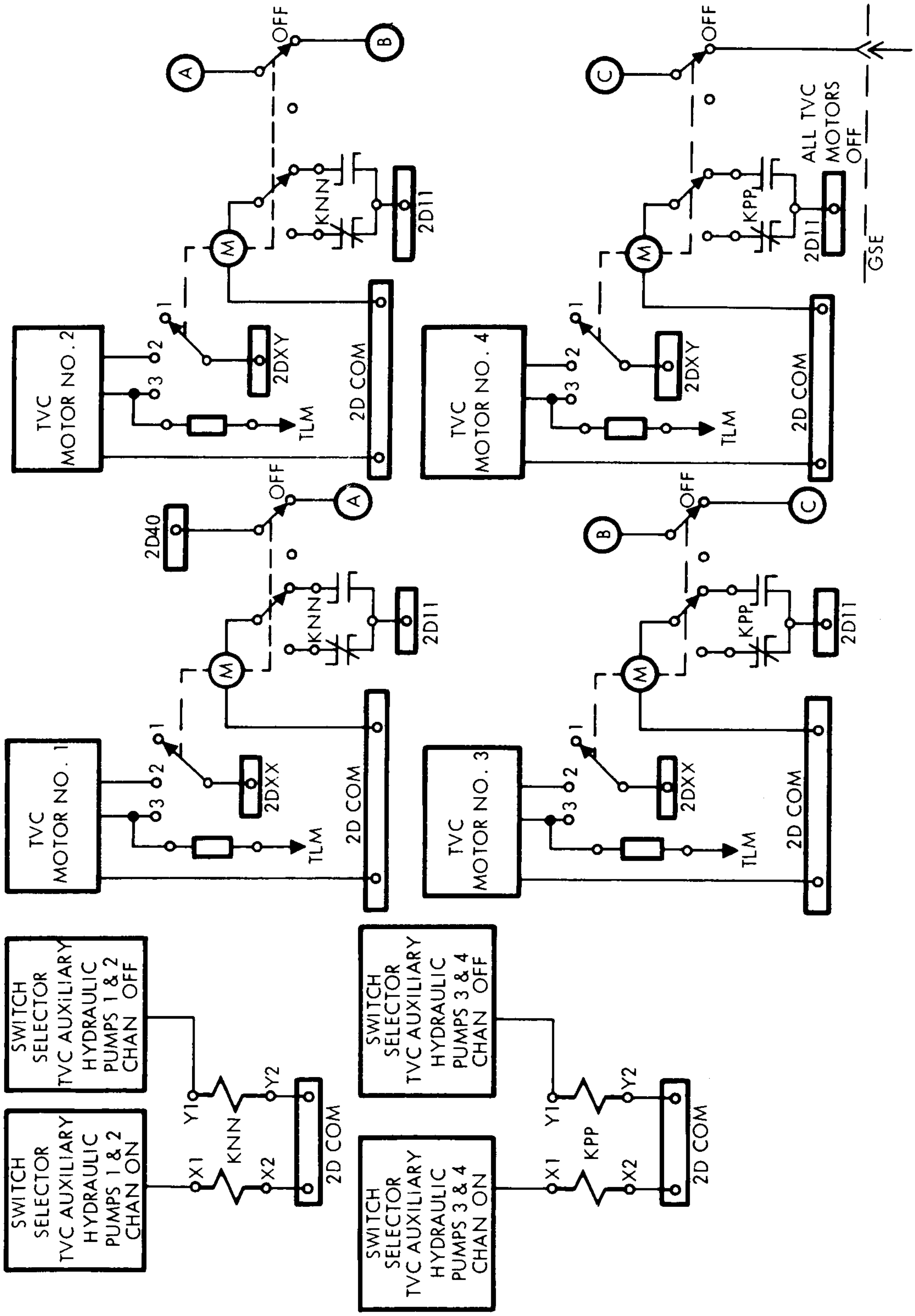


Figure 10.2-113. TVC Pump Motor Control

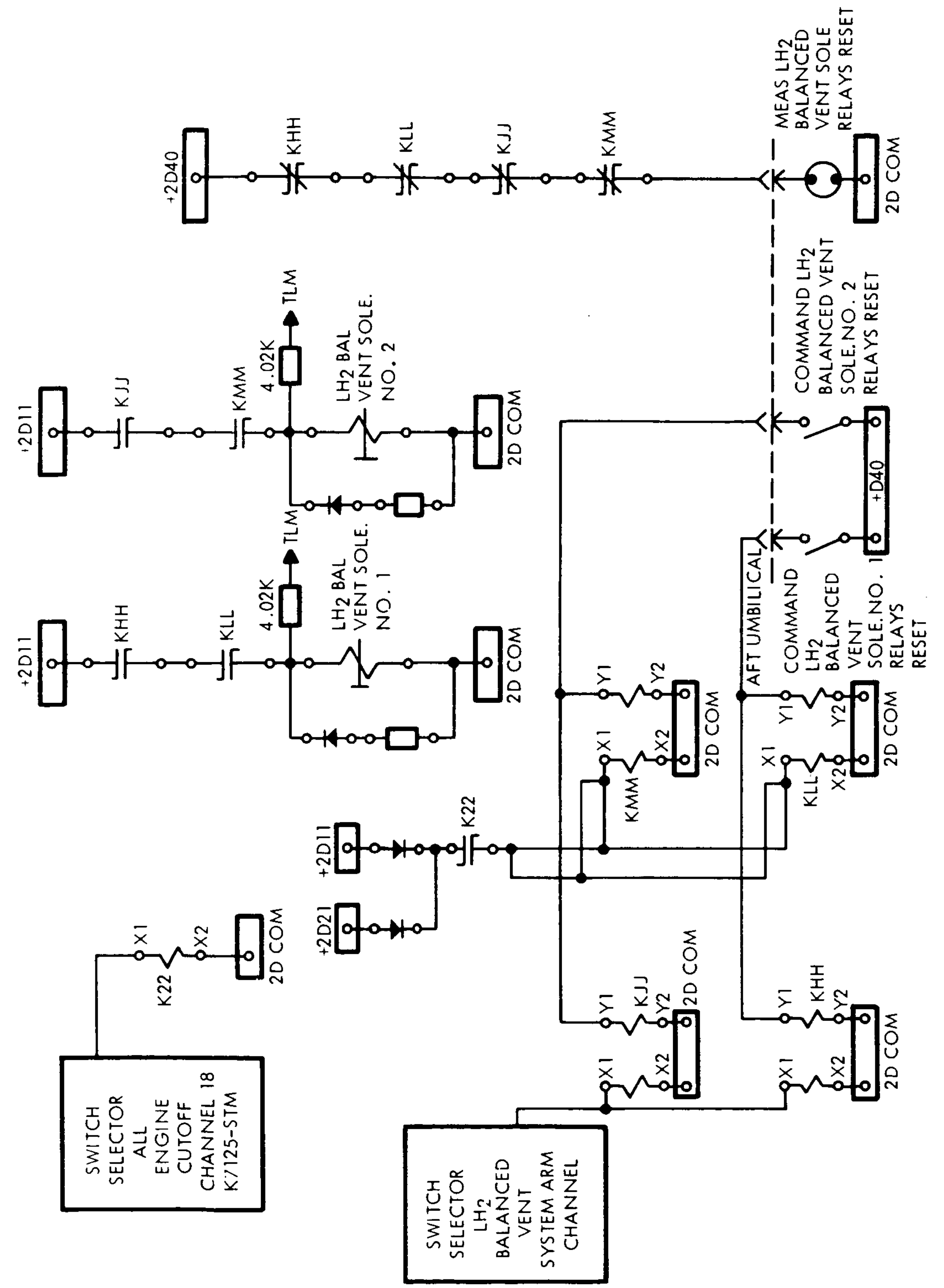


Figure 10.2-114. LH_2 Balanced Vent Valve Control

prior to "all engine cutoff" Channel 18 command. Relay contacts KHH and KJJ will apply 2D11 bus to both solenoids through KLL and KMM relay contacts. Switch selector Channel 18 "all engine cutoff" command will set relays KMM and KLL. The four relays will be reset prior to liftoff by a GSE command and verified by monitoring the +2D40 bus response through a series of four relay contacts.

5. Propellant Management Power Control. The new batteries for the LEO mission (45-minute coast period) will be capable of an alternate two-hour coast mission by removing propellant management power during the extended coast period. The power was calculated for an identical LEO mission (one mainstage operation and one three-minute idle mode period) except for a 105-minute coast period.

Figure 10.2-115 shows the circuitry required for controlling the propellant management power during an alternate two-hour coast LEO mission. Power will be removed from the propellant utilization and propellant level monitor dc power buses (2D14 and 2D15) during the 105-minute coast period to conserve power. Power will be switched on again prior to engine restart to obtain temperature measurements.

Figure 10.2-116 shows power off for the propellant management system. Before liftoff, the GSE will command power on for the propellant management system by applying +28 vdc through aft umbilical connector No. 1 to the power transfer switch motor. The power transfer switch will cycle to the opposite position and switch the main +2D11 bus to power the +2D14 and +2D15 buses.

During the S-IC boost phase the S-II engine start enable bus will hold relays KSS, KTT, and KUU in the reset state. Relay KUU will be required to arm the power control circuit. A new switch selector command, "PU power control arm" will be required to set relay KUU to arm the power transfer switch circuit. The switch selector "all engine cutoff" Channel 18 command just before the 105-minute coast period will set relay KSS causing the power transfer switch to cycle and remove power from the +2D14 and +2D15 buses. The contacts in series with KSS and KTT coils prevent power from being available simultaneously at each input to the motor of the power transfer switch.

Approximately three minutes before engine restart, relays KSS and KTT will be reset by the new switch selector command "PU power control reset". A new switch selector command, "PU power on" will be required to set relay KTT to cycle the transfer switch and apply power to the propellant management system again. Power will be on for the remainder of the LEO mission (approximately six minutes).

Three new switch selector commands and three DPDT latching type control relays are required to implement this change.

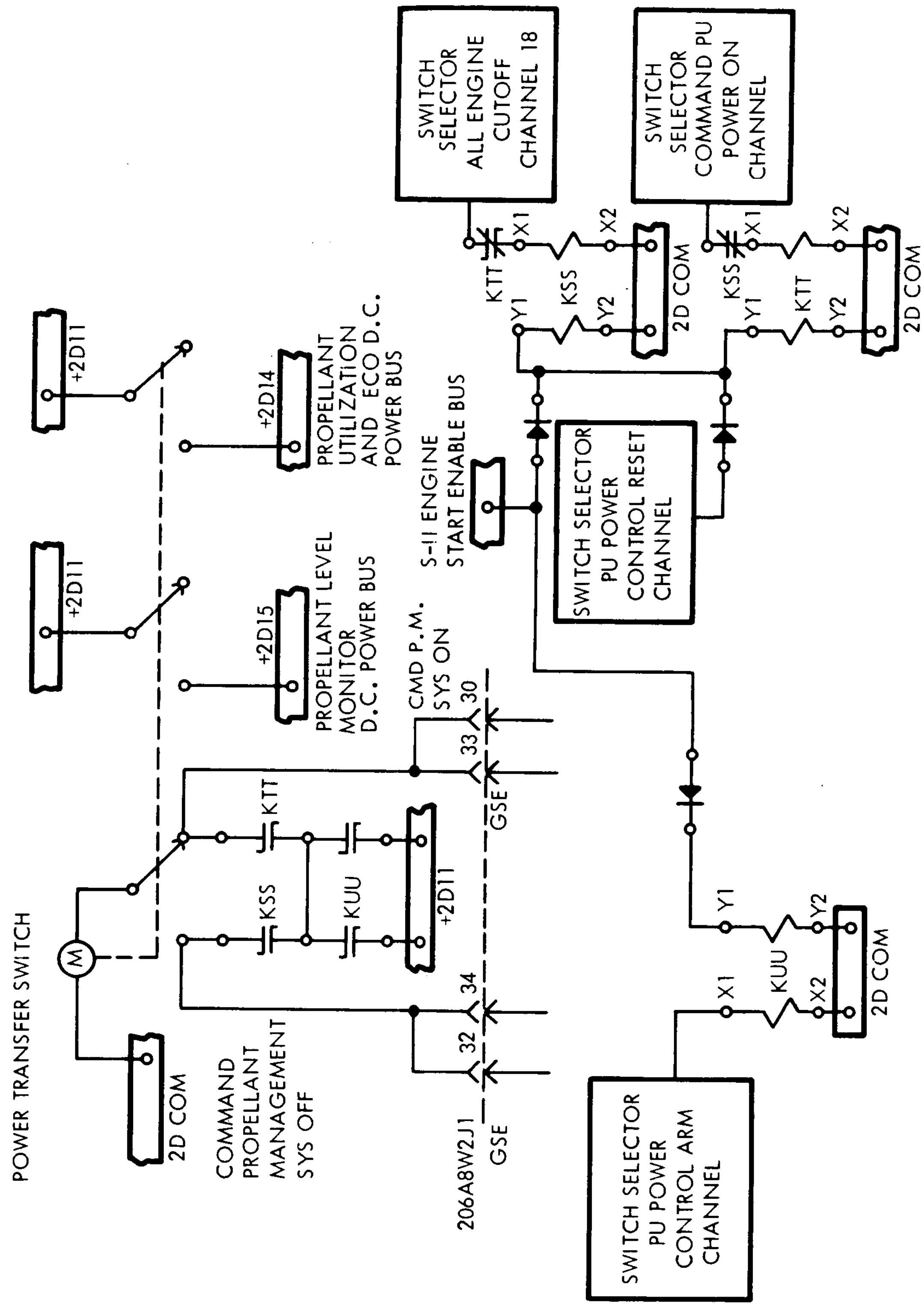
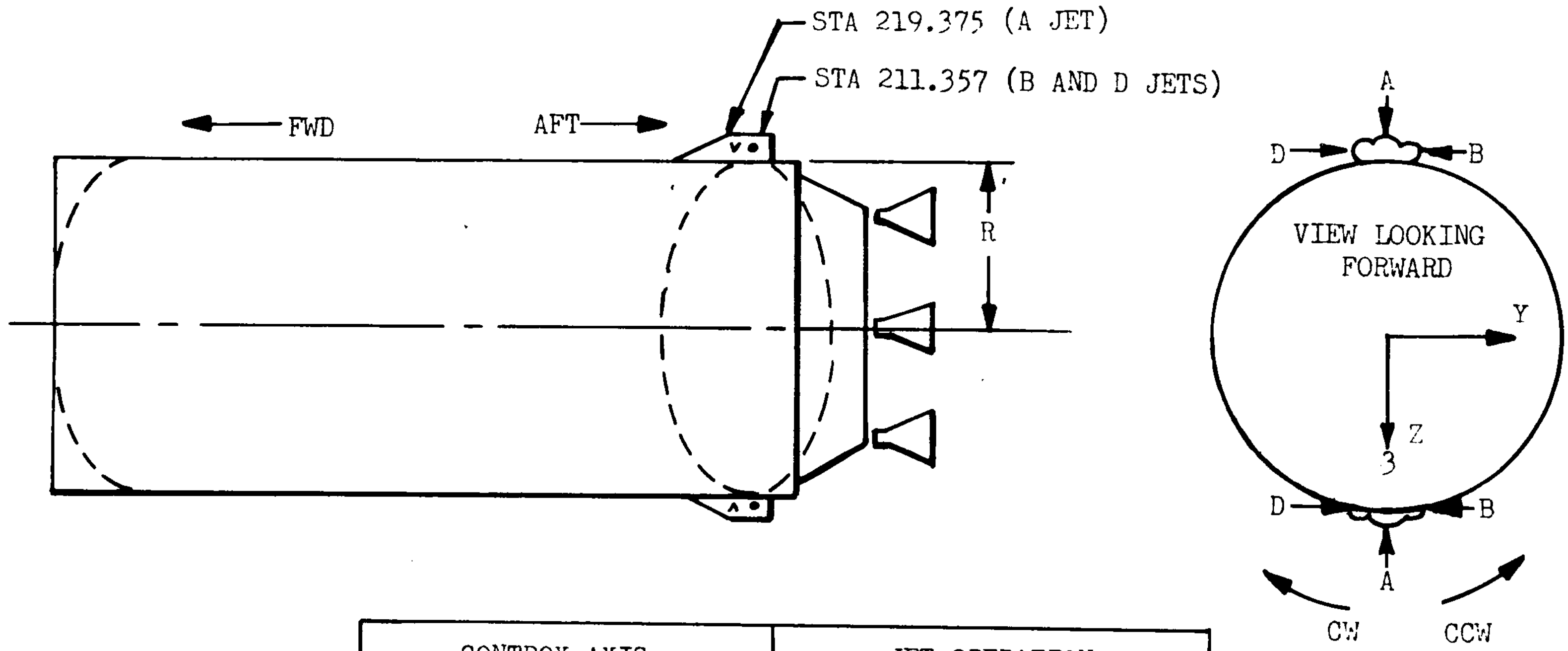


Figure 10.2-115. Propellant Utilization Power Control (Two-Hour Mission)



CONTROX AXIS		JET OPERATION
ROLL	CW(+X)	1D, 3D
	CCW(-X)	1B, 3B
PITCH	UP(+Y)	1A
	DOWN(-Y)	3A
YAW	RIGHT(+Z)	1B, 3B
	LEFT(-Z)	ID, 3B

Figure 10.2-116. RCS Jet Designation and Selection for S-II

10.2.3.2.2 Operational Change

This section summarizes changes to the electrical control system checkout requirements associated with the J-2S hardware changes.

a. LOR Mission. The electrical control system checkout requirements to incorporate the J-2S engine for the LOR mission will affect the following:

1. MA0203-1343 — 207 Electrical Container, Bench Checkout

2. MA0203-1342 — 207A2 Electrical Sequence Controller, Bench Checkout

   Deletion, addition, and reassignment of control relays for the LOR mission result in a 20 percent change to each bench checkout specification. The checkout time will remain approximately the same, because the additions and reassignment of control relays should approximately equal the deletions.

3. MA0701-1067-210 — Stage Electrical/Pneumatic Control Systems Checkout

   Deletion of the recirculation and ullage motor systems plus the additions to control the J-2S engines result in approximately a 15 percent change to this specification. The checkout time should be reduced slightly due to the deletions.

b. LEO Mission. The electrical control system checkout requirements for the LEO mission will affect the following:

1. MA0203-1343 — 207 Electrical Container, Bench Checkout

2. MA0203-1342 — 207A2 Electrical Sequence Controller, Bench Checkout

   Deletion, addition and reassignment of control relays for the LEO mission will result in a 20 percent change to each bench checkout specification. The checkout time will remain approximately the same.

3. MA0701-1067-210 — Stage Electrical/Pneumatic Control Systems, Checkout

   Addition of the LH_2 tank balanced vent system and prevalve lockout with an all engine cutoff will require approximately a 10 percent change to this specification. The checkout time will be increased slightly due to the additions.

10.2.3.3 Instrumentation System

10.2.3.3.1 Design Changes

This section contains the hardware changes to the instrumentation system associated with the J-2S engine incorporation. A brief description of the change and why it is required and a list of the instrumentation components is included.

The changes listed in this section are based on there being no change to the S-II-11 instrumentation program and components list, dated 5 January 1968. S-II design changes, approved after this date may alter the information contained in this section.

a. LOR Mission.

1. Engine System. The changes to the instrumentation system are a result of the engine configuration changes reflected at the engine interface.

   The measurements added and deleted from the flight measurement system are shown in Table 10.2-XI. The measurement changes, required in the ground measurement system, are listed in Table 10.2-XII. The only new hardware required for the flight system will be 15 new temperature bridge modules.

   Sufficient hardware and open channels are available to accommodate the added 35 measurements. Wiring and harness changes and additions will be required for all measurement changes.

2. Fuel Recirculation System. The measurement system changes, as a result of the deletion of the LH_2 recirculation pumps, are listed in Table 10.2-XIII. Except for the deletion of wire harnesses, only the following two hardware items are deleted:

Attenuator 0-70 vdc	V7-750300
Current Monitoring System	ME431-0019-0007

3. LOX Recirculation System. The changes to the flight measurement system, as a result of the deletion of the LOX recirculation and helium injection system are listed in Table 10.2-XIV. Wire harness changes are the only hardware changes affected by the deletion of the LOX recirculation system.

4. Separation System. The changes to the flight measurement system as a result of the ullage motors deletion are listed in Table 10.2-XV. Wire harnesses, associated with these measurements, as well as the following hardware will be deleted:

4 Dc/dc transducers	ME478-0030-0005

Table 10.2-XI. Engine Flight Measurement Deletions and Additions

Measurement	Function	
	Deletions	
VXC011-201/205	Start Tank Gas Temperature	
VXC329-201/205	Thrust Chamber Jacket Temperature	
D014-201/205	Gaseous Generator Chamber Pressure	
VXD016-201/205	Start Tank Pressure	
G001-201/205	Turbine Bypass Position	
G002-201/205	Start Tank Discharge Valve Position	
G005-201/205	Gaseous Generator Valve Position	
VK004-201/205	Gaseous Generator Spark System	
K008-201/205	O2 Bleed Valve Closed	
K009-201/205	H2 Bleed Valve Closed	
K196-201/205	H2 Start Valve Closed	
VK197-201/205	H2 Start Valve Open	
VK200-201/205	LOX Turbine Bypass Valve Open	
VK201-201/205	LOX Turbine Bypass Valve Closed	
VK202-201/205	Gaseous Generator Valve Open	
K203-201/205	Gaseous Generator Valve Closed	
K204-201/205	Start Tank Control Solenoid	
	Additions	
C326-201/205	LOX Turbine Outlet Temperature Temperature Low Level Probe (50) Asymmetrical Bridge	0-1800 F 2493 V7-750463-301
C005-201/205	Electric Control Box Temperature Temperature Low Level Probe (50) Asymmetrical Bridge	M300-200 F 2471 V7-750463-291

Table 10.2-XI. Engine Flight Measurement Deletions and Additions (Cont)

Measurement	Function	
	Additions (Cont)	
C006-201/205	Primary Instrumentation Pkg Temperature Temperature Low Level Probe (50) Asymmetrical Bridge	M300-200 F 2471 V7-750463-291
C014-201/205	Main Fuel Injection Temperature Temperature Low Level Probe (1256) Asymmetrical Bridge	M425-M100 F 134 EM V7-750463-371
C012-201/205	Helium Tank Gas Temperature Temperature Low Level Probe (1256) Asymmetrical Bridge	M350-100 F 134 EG V7-750463-461
D008-201/205	LOX Turbine Inlet Pressure Press dc/dc ABS (C-200)	0-200 psia PA 418-100-3
DXXX-201/205	Fuel Turbine Inlet Pressure Pressure dc/dc ABS	0-1500 psia
KXXX-201/205	Idle Mode Control (Solenoid) On	
KXXX-201/205	Mainstage Start (Solenoid) On	
KXXX-201/205	EBW Firing Units Armed	
KXXX-201/205	SPTS Initiated	
KXXX-201/205	SPTS Ready Signal No. 1	
KXXX-201/205	Mainstage Cutoff Lock-in On	
KXXX-201/205	Engine Cutoff Lock-in ON	
KXXX-201/205	Fuel Bypass Valve, Open Switch	
KXXX-201/205	Fuel Bypass Valve, Closed Switch	
KXXX-201/205	Idle Mode Valve, Open Switch	
KXXX-201/205	Idle Mode Valve, Closed Switch	

Table 10.2-XI. Engine Flight Measurement Deletions and Additions (Cont)

Measurement	Function
Additions (Cont)	
KXXX-201/205	Hot Gas Tapoff Valve, Open Switch
KXXX-201/205	Hot Gas Tapoff Valve, Closed Switch
GXXX-201/205	Fuel Bypass Valve, Position, Percent
GXXX-201/205	Idle Mode Valve, Position, Percent
GXXX-201/205	Hot Gas Tapoff Valve, Position, Percent
MXXX-201/205	Control Voltage Monitor

Table 10.2-XII. Engine Ground Measurement Deletions and Additions

Measurement	Function
	Deletions
BC009-201/205	Gaseous Generator LOX Valve Inlet Temperature
BC011-201/205	Start Tank Gas Temperature
WC015-201/205	Gaseous Generator Over Temp Cutoff Temperature
BC329-201/205	Thrust Chamber Jacket Temperature
C9112-GND	Thrust Chamber Purge Supply Temperature
C9115-GND	Thrust Chamber Chill Supply Temperature
C9282-GND	Thrust Chamber Chill Outlet Temperature
C9283-GND	Start Tank Chill A7-71 Out Temperature
BD002-201/205	Gaseous Generator Fuel Injection Pressure
BD007-201/205	Gaseous Generator LOX Injection Pressure
BD014-201/205	Gaseous Generator Chamber Pressure
BD016-201/205	Start Tank Pressure
WD127-206	He Inject Supply Pressure
WD128-206	He Inject Regulator Outlet Pressure
D9118-GND	Thrust Chamber PRG/Chill Supply Pressure
D9221-GND	Start Tank Chilldown Pressure
D9222-GND	Thrust Chamber Chilldown Pressure
D9223-GND	Thrust Chamber Chilldown Pressure
D9224-GND	Start Tank Chilldown Pressure
BG001-201/205	Turbine Bypass Valve Position 0-100 Percent
BG002-201/205	Start Tank Discharge Valve Position 0-100 Percent
BG005-201/205	Gaseous Generator Valve Position
WM107-201/205	Gaseous Generator Spark Monitor 1
WM108-201/205	Gaseous Generator Spark Monitor 2
	Additions
BBXXX-XXX	External Acoustic Station 220 Radius 198 Over Flame Bucket
BBXXX-XXX	External Acoustic Station 220 Radius 198 45 Degrees from Flame Bucket
WCXXX-201/205	Fuel Turbine Inlet Over Temperature Cutoff Temperature
BGXXX-201/205	Fuel Bypass Valve Position, 0-100 Percent
BGXXX-201/205	Hot Gas Tapoff Valve Position, 0-100 Percent

Table 10.2-XIII. LH_2 Recirculation System Measurement Deletions

Measurement	Function
	FLIGHT SYSTEM
C540-200	Recirculation Battery 1 Temperature
C541-200	Recirculation Battery 2 Temperature
VK251-207	Engine 1 Recirculation Pump Valve Open
VK252-207	Engine 2 Recirculation Pump Valve Open
VK253-207	Engine 3 Recirculation Pump Valve Open
VK254-207	Engine 4 Recirculation Pump Valve Open
VK255-207	Engine 5 Recirculation Pump Valve Open
VK256-207	Engine 1 Recirculation Pump Valve Closed
VK257-207	Engine 2 Recirculation Pump Valve Closed
VK258-207	Engine 3 Recirculation Pump Valve Closed
VK259-207	Engine 4 Recirculation Pump Valve Closed
VK260-207	Engine 5 Recirculation Pump Valve Closed
K313-207	Chilldown Valve Close Command
VK317-207	Recirculation Cutoff Indicator
K410-207	Recirculation Pump Valve Close Command
K412-207	Engine 1 Recirculation Pump On Command
K413-207	Engine 2 Recirculation Pump On Command
K414-207	Engine 3 Recirculation Pump On Command
K415-207	Engine 4 Recirculation Pump On Command
K416-207	Engine 5 Recirculation Pump On Command
XM111-207	Recirculation Dc Bus Voltage
XM114-207	Recirculation Battery Current
	GROUND SYSTEM
BM111-207	Recirculation Dc Bus Voltage
BM112-207	Recirculation Battery 1 and 2 Voltage
WM113-207	Recirculation Battery 2 Voltage
M9137-GND	Recirculation Bus Current
BT006-218	Engine 1 Recirculation Pump Speed
BT007-218	Engine 2 Recirculation Pump Speed
BT008-218	Engine 3 Recirculation Pump Speed
BT009-218	Engine 4 Recirculation Pump Speed
BT010-218	Engine 5 Recirculation Pump Speed

Table 10.2-XIV. LOX Recirculation System Flight Measurement Deletions

Measurement	Function
XD127-206	He Inject Supply Pressure
XD128-206	He Inject Regulator Outlet Pressure
VK271-207	Engine 1 LOX Return Line Valve Open
VK272-207	Engine 2 LOX Return Line Valve Open
VK273-207	Engine 3 LOX Return Line Valve Open
VK274-207	Engine 4 LOX Return Line Valve Open
VK275-207	Engine 5 LOX Return Line Valve Open
VK276-207	Engine 1 LOX Return Line Valve Closed
VK277-207	Engine 2 LOX Return Line Valve Closed
VK278-207	Engine 3 LOX Return Line Valve Closed
VK279-207	Engine 4 LOX Return Line Valve Closed
VK280-207	Engine 5 LOX Return Line Valve Closed
K390-207	LOX Return Line Valve Close Command

Table 10.2-XV. Separation System Flight Measurement Deletions

Measurement	Function
D054-200	Ullage Rocket 2 Chamber Pressure
D056-200	Ullage Rocket 4 Chamber Pressure
D058-200	Ullage Rocket 6 Chamber Pressure
D060-200	Ullage Rocket 8 Chamber Pressure
VK237-206	EBW Pulse Indicator Ullage 1A
VK238-206	EBW Pulse Indicator Ullage 1B

5. Thermal Analysis. The changes in thermal environment, as a result of the system changes on the stage, will require a change in the flight measurements (as indicated in Table 10.2-XVI on the following page.) The additional hardware required, as a result of these changes, is as follows:

6	Temperature Probes	ME449-0078-0001
2	Temperature Bridges	V7-750463-201
4	Temperature Bridges	V7-750463-191
4	Temperature Bridges	V7-750463-311
9	Flexible Surface Transducers (dual)	ME449-0026-0005
9	Dc amplifiers	ME473-0003-0001
9	H-SH Temperature Submodules	ME901-0588-0001

The hardware deletions are ME449-0011-0001, and two bridge modules, V7-750463-201.

Sufficient telemeter channels are available to accommodate the additional measurements. Changes to the wire harnesses, associated with all measurements, as well as the installation drawings for the new measurements will be required.

b. LEO Mission. The instrumentation system requirements established for the LEO mission are in addition to those identified in the LOR mission section.

1. Engine System. There are no additional instrumentation system requirements.

2. RCS. The reaction control systems will add a total of 48 new flight measurements: 24 identical measurements for each of the two systems. The new measurements for each system are shown in Table 10.2-XVII.

Table 10.2-XVII. RCS Measurement Requirements

Measurement	Function
CXXX-XXX	Attitude Control Oxidizer Module 1 Temperature
CXXX-XXX	Attitude Control Fuel Module 1 Temperature
CXXX-XXX	APS Helium Pressure Tank - Module 1 Temperature
DXXX-XXX	Attitude Control Chamber 1-1 Pressure
DXXX-XXX	Attitude Control Chamber 1-2 Pressure
DXXX-XXX	Attitude Control Chamber 1-3 Pressure
DXXX-XXX	Attitude Control Helium Pressure Tank 1 Pressure
DXXX-XXX	Helium Regulator Outlet Module 1 (APS) Pressure
DXXX-XXX	Fuel Supply Manifold Module 1 (APS) Pressure
DXXX-XXX	Oxidizer Supply Manifold Module 1 (APS) Pressure
DXXX-XXX	Fuel Tank Ullage Volume Module 1 Pressure
DXXX-XXX	Oxidizer Tank Ullage Volume Module 1 Pressure
KXXX-XXX	There are twelve (12) 28 vdc discrete measurements required for each RCS unit associated with the electrical control relays for the fuel and oxidizer valves on the attitude control engines.

Table 10.2-XVI. Thermal Analysis Flight Measurement Additions and Deletions

Measurement	Function
Additions	
C188-206	Thrust Cone Aft Ambient Temperature
C189-206	Thrust Cone Aft Ambient Temperature
C190-206	Thrust Cone Aft Ambient Temperature
C191-206	Thrust Cone Aft Ambient Temperature
C251-206	Thrust Cone Aft Stringer Temperature
C252-206	Thrust Cone Aft Stringer Temperature
C252-206	Thrust Cone Aft Stringer Temperature
C253-206	Thrust Cone Aft Stringer Temperature
C254-206	Thrust Cone Aft Stringer Temperature
C671-206	Engine Compartment Gas Temperature
C672-206	Engine Compartment Gas Temperature
C683-306	Heat Shield Aft Surface Temperature
C710-206	Heat Shield Aft Surface Temperature
C711-206	Heat Shield Aft Surface Temperature
C713-206	Heat Shield Aft Surface Temperature
C714-206	Heat Shield Aft Surface Temperature
C715-206	Heat Shield Aft Surface Temperature
CXXX-206	Heat Shield Aft Surface Temperature
CXXX-206	Heat Shield Aft Surface Temperature
Deletions	
C232-206	Thrust Cone Forward Surface Temperature
C241-206	Thrust Cone Forward Surface Temperature

The additional measurement hardware required to make these measurements are as follows:

4	Asymmetrical Bridges	V7-750463-471
2	Asymmetrical Bridges	V7-750463-291

In addition to this hardware, 5 vdc power will be required for the last six pressure measurements listed in Table 10.2-XVIII. RACS wiring will also be required for the remaining three pressure measurements in the table.

3. Flight Control System. The thrust vector control system required for gimbaling the engines during the extended idle mode operation of the LEO mission will be powered by a dc motor pump system. The dc motor will be housed in a container that will be pressurized. Each outboard engine will require one dc motor pump. The air pressure in each dc motor container will be monitored during flight. The hardware required to implement the flight measurements is listed below:

4	Dc/dc Pressure Transducers	ME478-0028-0007
1	1 Transducer Potentiometer	MC449-0007-0001

4. Fuel Tank Pressurization System. Two new measurements are required for the LH_2 balanced vent system for the LEO mission. The two measurements with the hardware required to make them are as follows:

CXXX-XXX	LH_2 Balanced Vent Inlet Temperature
DXXX-XXX	LH_2 Balanced Vent Inlet Pressure

1	Dc/dc Pressure Transducer	ME478-0028-0007
1	LL Temperature Probe	ME449-0072-0001
1	Dc Amplifier	ME473-0003-0001
1	Temperature Bridge Submodule	

5. Thermal Analysis. New measurements required for flight thermal analysis are as follows:

C343-207	Battery Shelf Temperature
C344-207	Electrical Container Equipment Mount Temperature
C349-209	Instrumentation Container Equipment Mount Temperature
C351-210	Instrumentation Container Equipment Mount Temperature
C356-221	Tracking Aid Equipment Mount Temperature
C357-214	PM Control Equipment Mount Temperature
C359-225	Telemetry Control Equipment Mount Temperature
C360-206	Switch Selector Control Equipment Mount Temperature
C511-223	Propellant Dispersion Container Equipment Mount Temperature

Addition of these measurements will require the following measurement hardware:

9	Low Level Flexible Surface Transducers	ME449-0026-0001
9	Asymmetrical Bridges	V7-750463-311

A change to the telemetry equipment will also be required to provide the required number of low level channels. In either of the RASM's, the following cards will be required to provide the capability for twenty additional low level temperature measurements:

2	Low Level Differential Amplifier Assembly	50M60204-050
4	Low Level Switch Assembly	50M10940-1
1	Decoding Matrix Assembly	50M10920-1

6. Electrical System. Addition of new batteries to satisfy power requirements for the LEO mission has established the requirement for the twelve new measurements listed below:

MXXX-XXX	TVCS 1 Bus Voltage
MXXX-XXX	TVCS 2 Bus Voltage
MXXX-XXX	TVCS 1 Bus Current
MXXX-XXX	TVCS 2 Bus Current
KXXX-XXX	Engine 1 M-P On Indication
KXXX-XXX	Engine 2 M-P On Indication
KXXX-XXX	Engine 3 M-P On Indication
KXXX-XXX	Engine 4 M-P On Indication
CXXX-XXX	TVCS 1 Battery Temperature
CXXX-XXX	TVCS 2 Battery Temperature
CXXX-XXX	TVCS 3 Battery Temperature
CXXX-XXX	TVCS 4 Battery Temperature

The additional hardware required to implement these measurements is as follows:

2	Current Monitoring Systems	ME431-0019-0007
2	0- to -7 v Attenuators	V7-750300-41
4	Low Level Flexible Surface Transducer	ME449-0026-0001
4	Asymmetrical Bridges	V7-750463-241

10.2.3.3.2 Operational Changes.

This section contains a description of the instrumentation system checkout requirements associated with J-2S hardware changes.

a. LOR Mission. Revisions to the following checkout procedures and measurement lists will be required:

J7-976411	Ground Instrumentation Measurements List
V7-976411	Saturn S-II Instrumentation Program and Components List, S-II-11 Stage

MA0201-1892	Measurement System Test and Checkout, S-II-6 stage and Subsequent Stages
MA0201-1921	Measurement System Test and Checkout, Automatic Mode, S-II-6 and Subsequent Stages
MA0201-1975	S-II-6 and Subsequent Stage Ground Measurement System Test and Checkout (Seal Beach)
MA0201-4363	S-II Ground Instrumentation System Test and Checkout (MTF)

The above checkout procedures and measurement lists would require from a 20 to 80 percent revision. The checkout procedures would not change appreciably in complexity and checkout time required.

b. LEO Mission. The operational changes for the LEO mission are the same as those listed for the LOR mission.

10.2.3.4 Ordnance System

10.2.3.4.1 Design Changes

This section contains the hardware changes to ordnance system associated with the J-2S engine incorporation. A brief description of the changes why they are required, and a list of the deleted ordnance system components are included.

a. LOR Mission. Listed below are the ordnance components which are associated with the ullage motor ignition system, that will be deleted because the ullage motors are deleted from the J-2S S-II stage.

Components Deleted

Part Name	Part Number	Number Deleted
EBW Detonator	7865742-1	2
CDF Manifold	ME288-0001-0005	2
CDF Assembly	ME901-0052-3525	1
CDF Assembly	ME901-0052-3575	1
CDF Assembly	ME901-0052-3577	1
CDF Assembly	ME901-0052-3663	1
CDF Assembly	ME901-0052-3665	1
CDF Assembly	ME901-0052-3715	1
CDF Assembly	ME901-0052-3717	1
CDF Assembly	ME901-0052-3765	1
CDF Assembly	ME901-0052-3767	1
Pyrogen Initiator	ME453-0004-0001	8

b. LEO Mission. No additional ordnance system design changes have been identified for the LEO mission.

10.2.3.4.2 Operational Changes

This section contains a description of the ordnance system checkout requirements associated with J-2S hardware changes.

a. LOR Mission. The ordnance system checkout requirements to incorporate the J-2S engine for the LOR mission will affect the following:

1. For the engine solid propellant turbine start (SPTS) system, a process specification will be prepared for electrical checkout of the exploding bridgewire (EBW) initiators (two required for each SPTS) before their installation for MTF static firing and KSC flight.

2. For the engine SPTS system, a process specification will be prepared for installing and arming the EBW initiators (disconnecting the dual pulse sensors) prior to MTF static firing.

3. For the engine SPTS EBW system live ordnance test at KSC, process specification MA0301-1004, "Preparation of Live Ordnance Test, S-II Vehicle, Kennedy Space Center High Bay Area, Procedures for" will be revised. This specification will also be revised to provide for the deletion of the ullage motor ignition system.

4. For the engine SPTS EBW system and deleted ullage motor ignition, process specification MA0301-1005, "Exploding Bridgewire Detonator Installation and Arming, Procedure for" will be revised. This specification will be revised to install and arm the SPTS EBW initiators at KSC prior to flight.

b. LEO Mission. No additional checkout requirements have been identified for the LEO mission.

10.2.4 THERMAL PROTECTION

This section discusses the aerodynamic and base heating rates applicable to the J-2S study and the resulting hardware temperatures.

10.2.4.1 Aerodynamic Heating

The S-II stage surface and protuberances are subjected to aerodynamic heating during first stage boost.

10.2.4.1.1 Heat Transfer Rates

Analyses of aerodynamic heating rates for the LOR, SO, PO, and LEO mission trajectories have been conducted at MSFC. The results indicate that the aerodynamic heating rates for all of the missions are less severe than the AS-501 - 503 mission trajectory heating rates. Therefore, the AS-501 - AS-503 nominal aero heating design environment is applicable to the LOR, SO, and LEO missions. However, proturberance factors used for the AS-504 - AS-510 design thermal environment should be used for these missions since the Mach number-time histories for these trajectories and for the J-2S trajectories resemble each other closely.

For the PO mission, the AS-504 - AS-510 thermal design environment should be used since the PO mission trajectory is even less severe than that of the AS-504 - AS-510 missions.

10.2.4.1.2 Design Temperatures

Aerodynamic heating rates for the J-2S study are the same as for the present S-II design; thus, existing S-II-11 structural design temperatures will not change.

10.2.4.2 Base Heating

The S-II stage base region is subjected to convective heating resulting from the reverse flow of the J-2S engine exhaust plume. This is due to the mutual impingement of the exhaust plumes at high altitude, which results in high temperature and pressure interaction regions. These impingements regions are also the primary source of the base region radiative heating.

10.2.4.2.1 Gas Recovery Temperature and Shear Loads

The same gas recovery temperatures used for the J-2 engine configuration were used for the base heating analyses of the J-2S configuration. These values were 3100 F for the rigid heat shield and flexible curtain, and 2325 F for the thrust structure and other base regions. It is anticipated that because of higher chamber pressure, the shear loads on the base heat shield will be higher than those experienced with the J-2 configuration.

10.2.4.2.2 Convective Heat Transfer Rates

A study employing an approximate analytical technique was made to quantitatively determine the difference in stagnation convective heating rates at the base heat shield between the J-2 and J-2S engine configurations.

Based on S-II model static and flight test data with the J-2 engine configuration, a curve of base pressure at the heat shield as a function of engine chamber pressure is available. Using this curve, a base pressure for the J-2S of 0.096 psia was obtained at the nominal chamber pressure, 1185 psia. This value was used to obtain the base heating estimates. The comparable value for the J-2 engine is 0.065 psia at the nominal chamber pressure of 715 psia.

Both J-2 and J-2S base heating analyses employed the same gas recovery temperatures of 3100 F for the rigid heat shield and flexible curtains and 2325 F for thrust structure and other base regions. No effects of engine gimbaling were included.

Results of the analysis indicate that convective heating rates for the J-2S will increase approximately 20 percent over the J-2. However, flight test data with the J-2 engine indicate that actual convective heating rates at the heat shields are significantly less than the design environment. Therefore, it should be satisfactory to retain this J-2 design convective heating environment for the J-2S configuration.

10.2.4.2.3 Radiant Heat Transfer Rates

Using the J-2S nominal plume properties, radiation levels have been calculated at various locations on the base heat shield and the thrust structure. These levels have been calculated at the same locations using an average J-2 plume; the radiation from the J-2S was found to be as much as 20 percent higher than that produced by the J-2. For this reason, NR is in concurrence with the NASA recommendation that the design radiation environment be increased 25 percent above the level used with the J-2 engines.

Because the S-II stage is used above 195,000 feet for all of the proposed missions, the radiation levels should not change significantly from mission to mission.

10.2.4.2.4 Structural Temperatures

a. Base Heat Shield. The predicted maximum temperature for the heat shield aft laminate is 1575 F. The maximum temperatures for the center and forward laminate were 790 F and 450 F, respectively. The aft laminate temperature of 1575 F is 275 F higher than the postflight evaluation one-engine-out maximum predicted temperature (1300 F) for the J-2 configuration, and 105 F higher than the 1470 F test temperature for the final 48-inch by 48-inch panel tests. It should be noted that the final panel test had a local rupture in the aft laminate. The local rupture itself was determined non-detrimental to the base heat shield, but loss of a large section of the aft laminate could expose the center laminate to direct base heating and cause the center laminate temperature to exceed 1000 F.

   The predicted shear loads resulting from the gas flow across the aft face of the heat shield are expected to double. It is possible that these higher shear loads could peel off a large section of the aft laminate and cause the heat shield to fail. It is therefore recommended that the heat shield be redesigned for the J-2S configuration to eliminate the possibility of a local rupture of the aft laminate.

b. Thrust Cone and Station 196 Web. Results of the thrust structure analysis predict a maximum thrust structure temperature of 345 F at the end of S-II boost. This is 18 F above the present maximum temperature. The temperature trends are similar.

   Results of the Station 196 web analysis predict a maximum temperature of 457 F for the web skin. This is an increase of 8 F over the present maximum web surface temperatures. The temperature trends are similar.

c. Interstage. The interstage analysis was conducted for the first 30 seconds of S-II boost (from S-IC separation to S-II interstage separation); it was based on the present external and internal insulation systems and a 235 F temperature at start of S-II boost. Results of the interstage analysis predict maximum temperatures at the time of interstage separation of 470 F. This is 10 F higher than the present maximum predicted temperatures, and 30 F below the present 500 F maximum design temperature.

10.2.5 REACTION CONTROL SYSTEM

For the LEO mission, the S-II stage performs an orbital circularization function with the payload following a coast period to apogee. During this coast period (approximately 45 minutes), a system for attitude control is required to remove vehicle perturbations due to initial shutoff of the J-2S engine cluster from idle, and to maintain vehicle attitude. Sizing of the reaction control system (RCS) included propellant to perform a 180 degree maneuver about the roll, pitch and yaw axes.

The propellant weight and time, for stabilization of the initial disturbance due to the J-2S engine shutdown from idle mode, were calculated from the following equations:

$$t = \dot{\gamma}_o \; I \; / \; F \; \frac{1'}{12} \quad (57.3)$$

$$\omega p = NF\, t \; / \; I_{sp}$$

The value of $\dot{\gamma}_o$ (initial disturbance rate) was assumed to be ± 0.6 degree per second.

During the analysis of this disturbance, it was assumed that $\dot{\gamma}_o$ could occur about any one, or all three, of the axes. Therefore, the yaw and roll jets could be firing simultaneously. For a clockwise roll and left yaw, it can be seen from Figure 10.2-116 that the opposing jets, 1d and 1b, are both firing. This also occurs for counter-clockwise roll and a right yaw with the opposing jets being 3d and 3b. To reduce this obvious waste in fuel, a change in the flight computer logic is necessary to shut down any two opposing jets that may fire during the same time interval. For the purpose of calculating the fuel requirements for this mission, the assumption was made that this logic did exist.

Reorientation of the vehicle is necessary because of its change in angular position caused by the initial disturbance rate. The equation used to calculate the fuel for this case is as follows:

$$\omega p = \frac{2NF}{I_{sp}} \sqrt{I \, \dot{\gamma}_o \, t / MF \, 1'}$$

The fuel for reorientation is also dependent upon the initial disturbance rate, $\dot{\gamma}_o$. In Figure 10.2-117, the total fuel per cluster for stabilization and reorientation is plotted for various values of $\dot{\gamma}_o$. This was done so that the fuel requirement for both stability and reorientation can be updated with any additional revision in the disturbance rate, $\dot{\gamma}_o$.

Fuel and time were calculated to rotate the vehicle at a rate equal to its orbital rate. In this way, it will maintain an orientation along the trajectory. The major and minor axis of the orbital ellipse varies from approximately 4100 to 4300 miles. The vehicle can therefore be considered as remaining along the local horizontal.

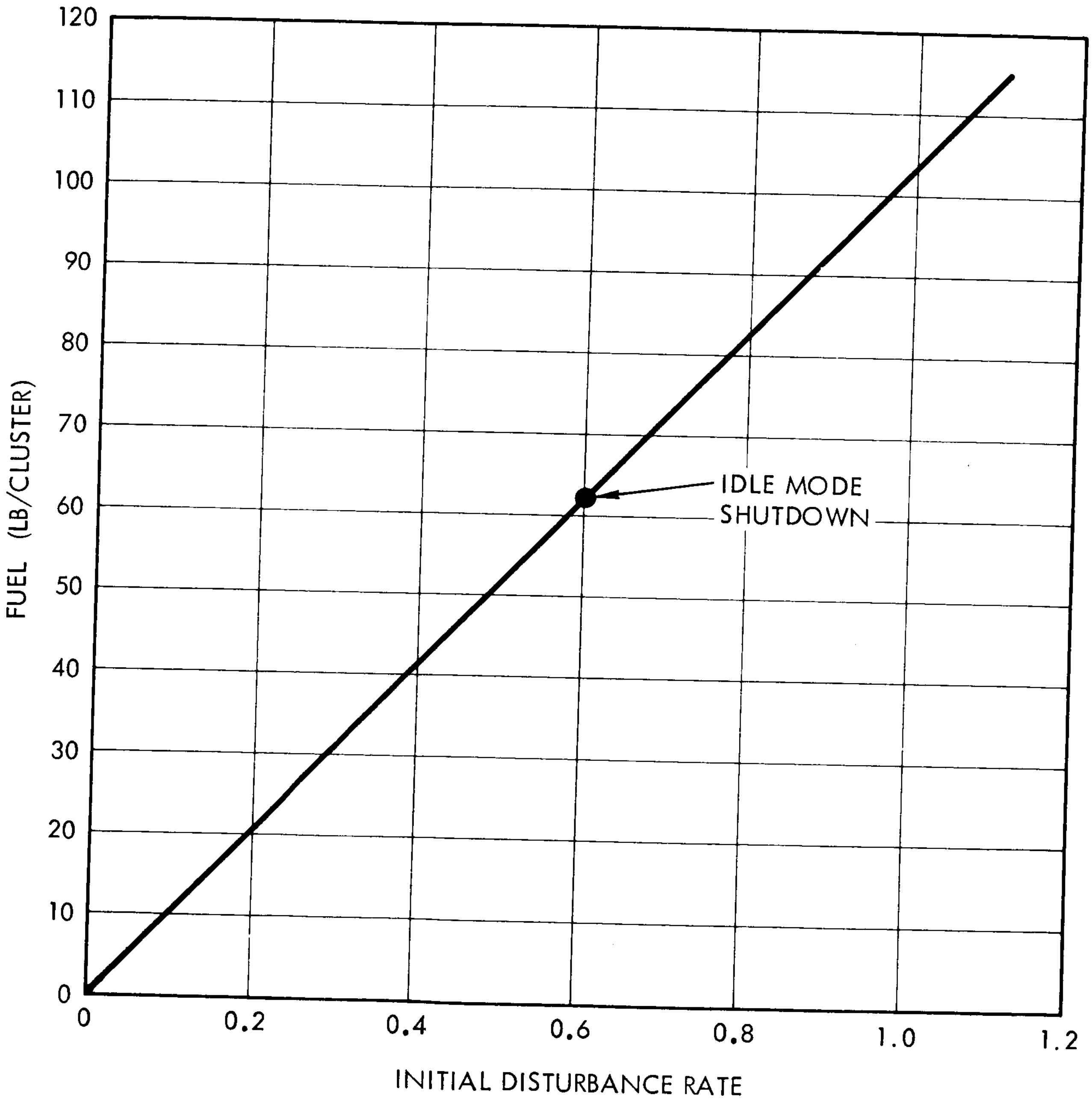


Figure 10. 2-117. Fuel for Stabilizing and Reorientating Due to the Initial Disturbance Rate

A 5.0-degree deviation from the local horizontal was used in the analysis due to the orientation along the trajectory and a deadband of 2.5 degrees. Included also were gravity gradient and aerodynamic disturbance torques.

A 180-degree maneuver about the roll, pitch, and yaw axes was included in the tabulation of the fuel requirements. Figure 10.2-118 presents the rate limits as a function of time for all three axes. Figure 10.2-119 is a plot of fuel expenditure per cluster for roll, ptich, and yaw; both individually and as a sum, for various rate limits.

The maximum duty cycle was calculated from the ratio of disturbance torque to control torque. This expression gives the thruster on-time as a percentage of the total mission time.

$$\text{Duty cycle} = 0.094 \text{ percent}$$

Tables 10.2-XVIII and 10.2-XIX present the basic data used in the analysis, with the results presented in Table 10.2-XX.

10.2.5.1 System Selection and Description

10.2.5.1.1 System Selection

The reaction control systems studied for S-II application were selected based on the consideration that they were available proven hardware. Another factor in choosing candidate systems was to select systems which have performance characterisitics compatible with the S-II mission requirements. Three systems were initially considered: the Apollo service module RCS, the uprated Saturn I-B/S-IVB attitude control system, and the Saturn V/S-IVB attitude control system. All three systems employ hypergolic bipropellant rocket engines and positive expulsion feed systems to provide propellant feed capabilities under zero and random gravity environment.

The Apollo RCS module employs four radiation-cooled, pulse-modulated rocket engines. The engine has a rated vacuum thrust of 100 pounds and a steady state specific impulse of 276 seconds. The engines use N_2O_4 oxidizer and MMH fuel at a mixture ratio of 2.1. There are two system configurations: Block I design, with two propellant tanks and a total propellant capacity of 120 pounds; and Block II with four propellant tanks and a capacity of 326 pounds. For the Apollo application, the RCS is mounted to a removable panel; when installed on the service module, this is an integral part of the service module structure.

The Saturn I-B/S-IVB attitude control system consists of three ablative pressure-fed rocket engines with a vacuum thrust of 150 pounds each and a steady state specific impulse of 292 seconds. The engine uses N_2O_4 and MMH propellants at a mixture ratio of 1.6:1. The system is installed in a self-contained detachable module with the thrusters positioned to provide pitch, yaw, and roll control. The system has a propellant capacity of 62 pounds.

The Saturn V/S-IVB attitude control system has an engine cluster consisting of three 150 pound thrust attitude control engines for pitch, yaw, and roll control (identical to the Saturn I-B/S-IVB engines) and a 70-pound engine to provide axial thrust. The system has a propellant capacity of 330 pounds.

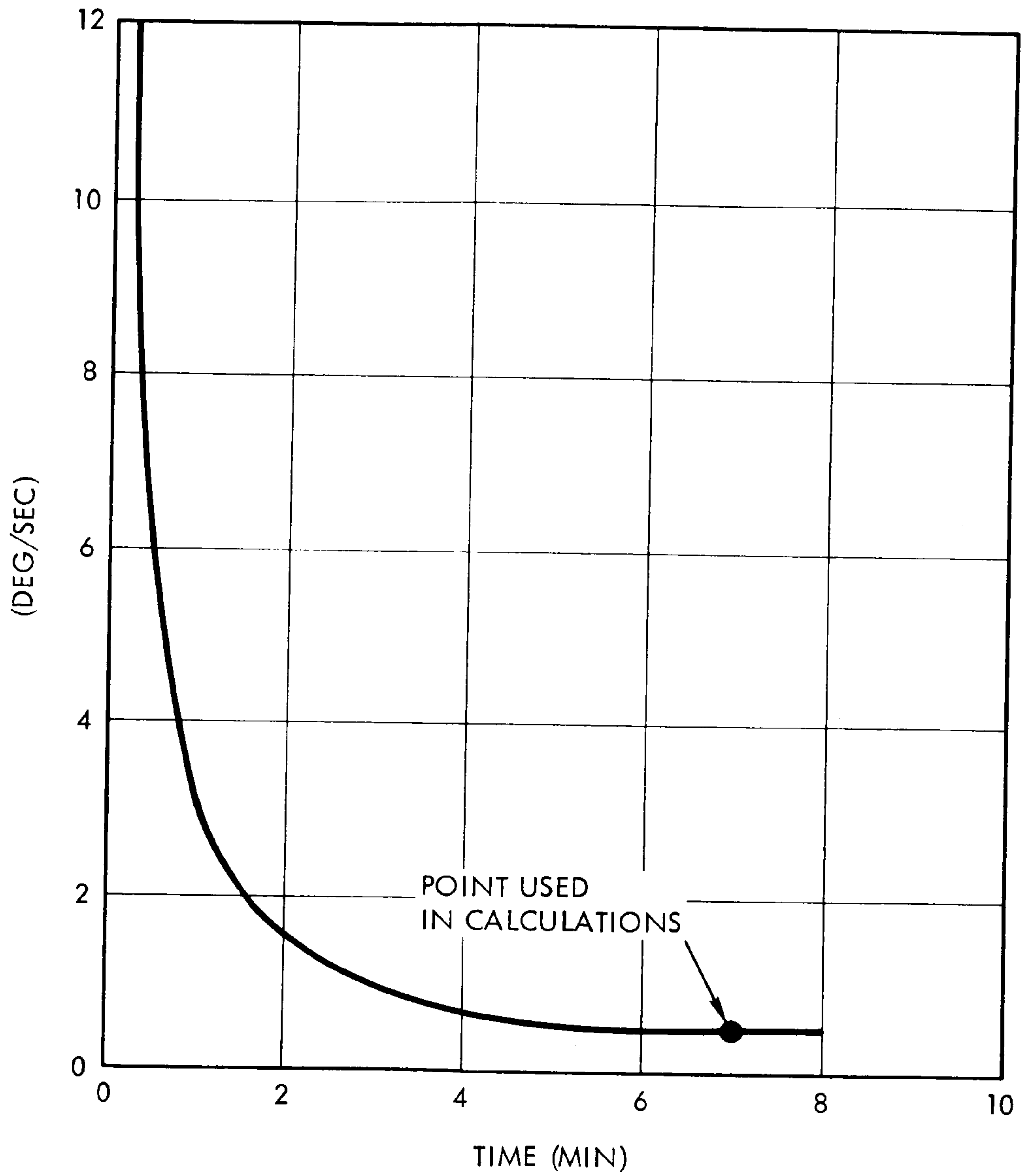


Figure 10.2-118. Angular Rate Versus Time for 180-Degree Maneuver

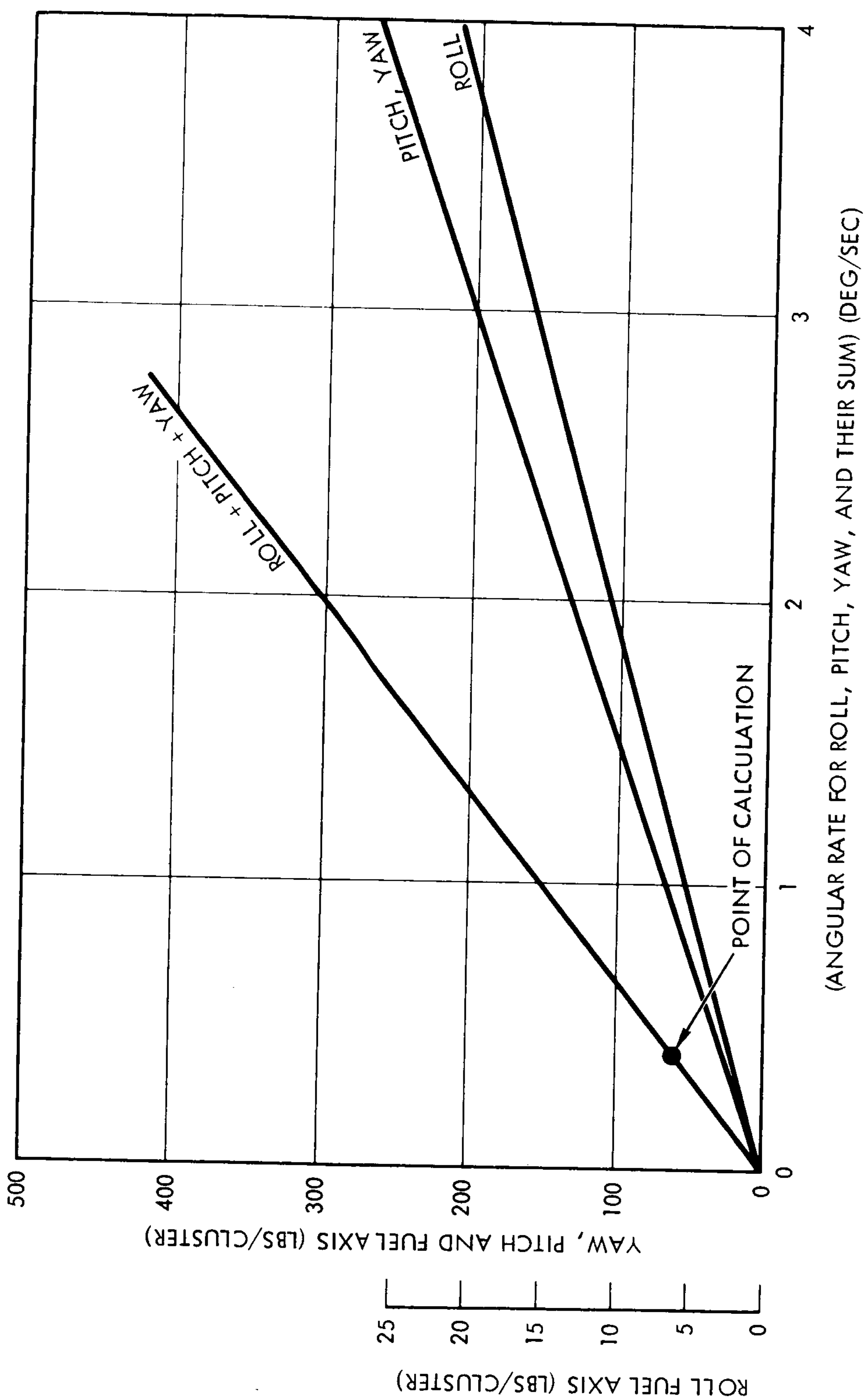


Figure 10.2-119. Fuel Expenditure Versus Rate Limit for 180-Degree Maneuver

Table 10.2-XVIII. RCS Thruster Characteristics

Symbol	Description	a, b, d Jets
I_{sp}	Specific impulse, seconds (steady state)	292
F	Thrust magnitude, pounds	147.2
R	Distance from vehicle centerline, inches	210.34
δ	Jet cant angle, degrees	10

Table 10.2-XIX. Vehicle Mass Properties Data

Symbol	Description	J-2S Engine Shutdown
w	Total vehicle weight, pounds	360,000
X_{cg}	Vehicle center of gravity from reaction control jets, inches	786
$I_{yy} = I_{ss}$	Vehicle pitch or yaw moment of inertia, slug-ft^2	35,500,000
I_{xx}	Vehicle roll momet of inertia, slug-ft^2	1,250,000
ℓ'	RCS effective moment arm	
	Roll, $\ell' = R \cos \delta$	200
	Pitch, $\ell' = X_{cg}$	790
	Yaw, $\ell' = X_{cg} \cos \delta$	771

Table 10.2-XX. Flight Control Sequence of Events and Associated Three-Axes RCS Propellant Usage Requirements for the LEO Mission

Event	RCS Attitude Control Maneuver Requirement	1/2 Orbit		1/2 Orbit	
		Wp* (lbs)	T* (sec)	Wp (lbs)	T (sec)
Injection into 100/300 nautical-orbit	None				
J-2S engine shutdown from idle mode	Stabilize a disturbance of ±0.6 degree/second	31.2	66.0	31.2	66.0
	Reorientate vehicle	31.5	88.3	31.5	88.3
Coast in elliptical orbit	Limit cycle at ±2.5 degrees about all three axes	0.2	0.0	0.6	0.0
	Maintain attitude control in the presence of disturbing torques	1.3	2.5	4.0	7.5
	Rotate vehicle at orbital rate to maintain attitude along trajectory	2.2	4.2	2.2	4.2
	Orientation maneuver of 180 degrees in 7 minutes about all three axes	58.0	51.0	58.0	51.0
Propellant requirements summary	Subtotal	124.4		127.5	
	Add 10% margin	12.5		13.0	
	Total RCS weight/cluster	136.9		140.5	
	Maximum RCS operational usage/jet		212.0		217.0

Notes: *W_p is the propellant weight requirement at rated I_{sp}/cluster. Total propellant weight for RCS is two times greater.
*T is the total jet on-time requirement for the most used jet.

For the purpose of this study, all of the attitude control systems considered were assumed to be of the type desired with respect to engine thrust levels and minimum impulse bit capabilities. The study then became a process of examining the advantages and disadvantages of the various system configurations. The system must have sufficient redundancy for man rating. A detachable modular RCS was considered desirable to maintain an independent propulsion system to facilitate manufacturing, checkout, and replacement, if required. The modular concept is appropriate for isolating the thermal control problems associated with space environment and cryogenic space vehicles. Compact system modules with shorter lines and buried engine installation simplify the thermal control problem during flight by permitting the use of passive protection.

A summary of the factors considered most significant in arriving at the selection of the S-II reaction control systems is presented in Table 10.2-XXI. As a result of assessing these factors, the Saturn V/S-IVB system was selected for the S-II. The primary factors which led to its selection were that it requires a minimum modification to the existing design for S-II application, requires only two modules to provide adequate control capability with required redundancy features, and can meet the propellant loading capacity for S-II mission requirements.

The propellant capacity of the Saturn IB/S-IVB system does not meet the minimum requirement currently estimated for the S-II mission. It is felt that modifying the systems to provide larger propellant tanks would result in design changes requiring system requalification. Although it is recommended that the axial thrust engine on the Saturn V/S-IVB attitude control system be deleted, it is felt that this will have little impact on system qualification status.

The Apollo service module RCS has two major disadvantages in regard to the S-II mission application. First, the engine employs single propellant injector valves with dual activation coils (automatic and manual). This configuration does not provide redundancy for a valve failure. In order to provide such redundancy, a four-module configuration is required. This is compared to the quad redundant series-parallel injector valves configuration used in the S-IVB design, which requires only two modules to provide the same redundancy capability. Another disadvantage of the Apollo RCS is that it would require considerable system modifications to repackage the unit into a detachable module design.

10.2.5.1.2 System Description

The system selected for the S-II RCS application is the McDonnell Douglas Company (MDC) Saturn V/S-IVB RCS module. The modules, two required per vehicle, are demountable and will be procured to the configuration presently being manufactured by the MDC. A module is shown in Figure 10.2-120. The module has a tri-lobe body section with external skin that mounts the propellant tanks, pressurization tank, and thrusters. An aerodynamic nose fairing with a 30 degree profile provides additional space for mounting the penumatic controls.

The module is shown schematically in Figure 10.2-121.

Table 10.2-XXI. S-II RCS System Criteria

Factors	Apollo Service Module RCS		S-IVB Attitude Control	
	Block I	Block II	Saturn I-B	Saturn V
Maximum thrust	100		150	
Minimum impulse Bit, Lb_f - sec	0.5		7.5	
Propellant loading (lb)	125	330	62	330
Dry weight (lb)	238	264*	350	413
Redundancy capabilities	Single propellant valve arrangement. Requires 4 modules to satisfy control capability in case of valve failure		Series-parallel propellant valve arrangement. Requires only two modules to satisfy failure mode requirements for manned mission.	
Engine life	1000		330 (meets mission requirements)	
Modifications	Extensive modification required to repackage RCS into a detachable modular concept.		Requires new propellant tanks to meet S-II requirements. Results in repacking existing module.	Delete axial thrust engines. Results in little impact on existing system design.

*Estimated weight of module fairing is 156 pounds.

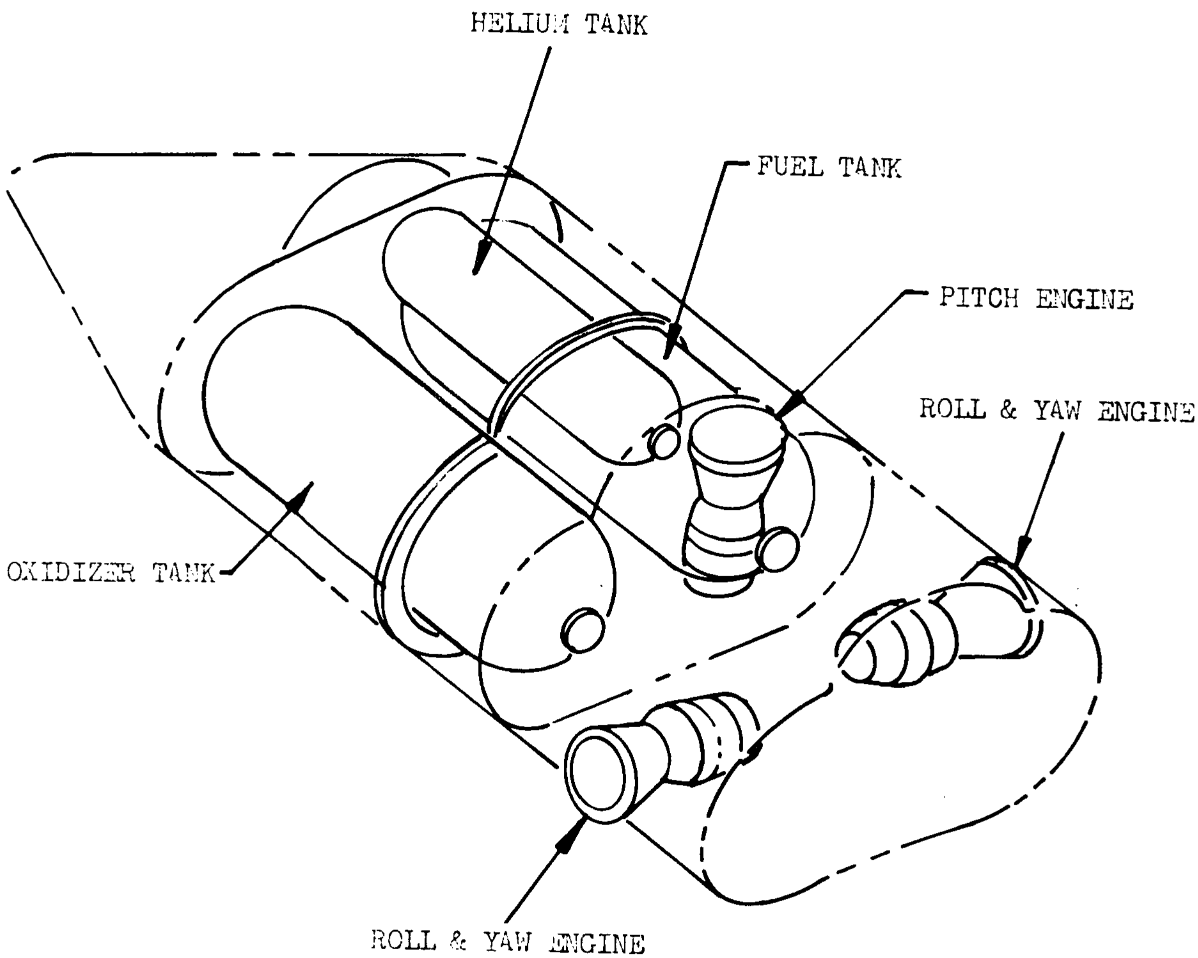


Figure 10. 2-120. Reaction Control System Module Perspective

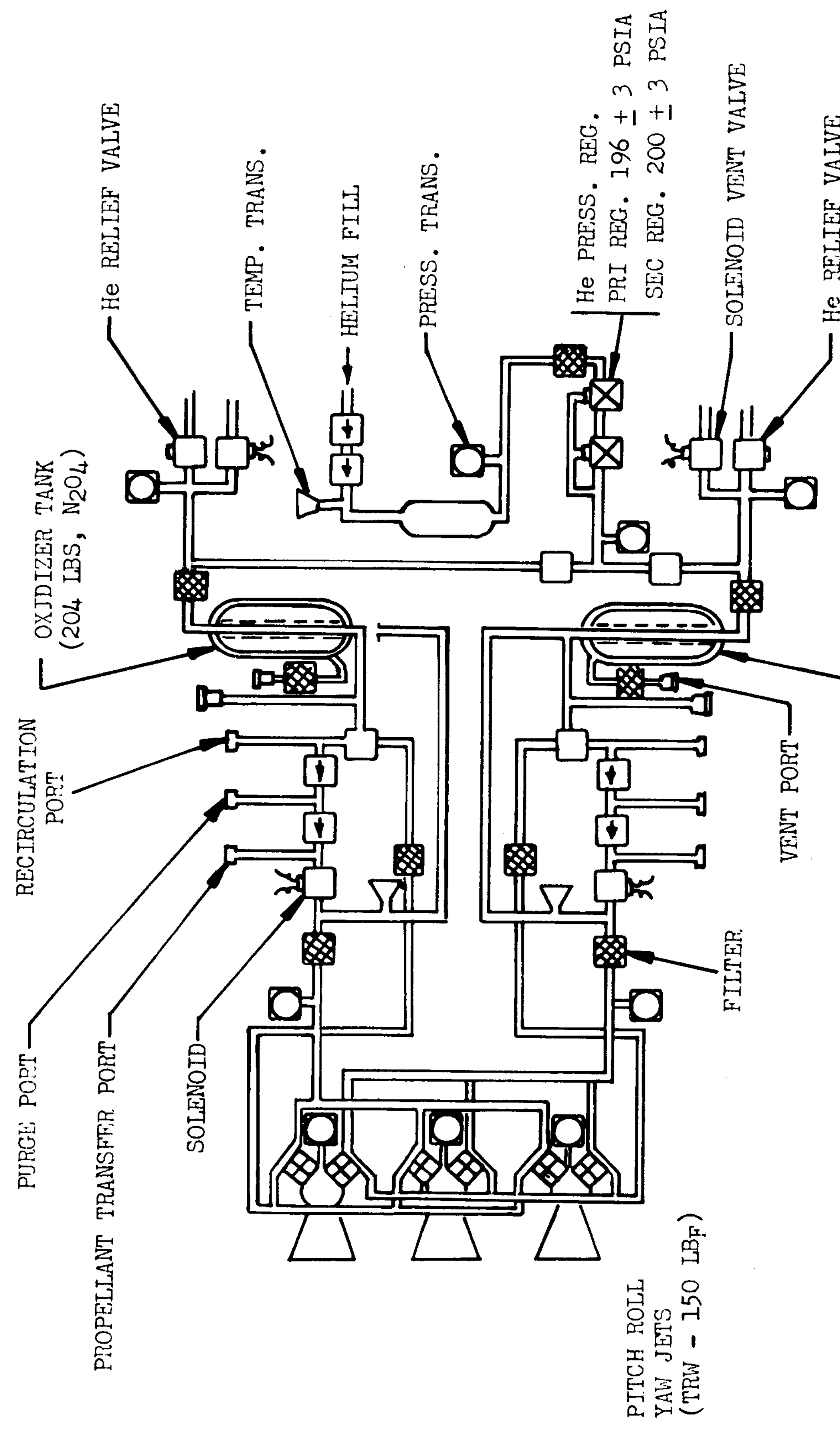


Figure 10.2-121. Saturn V/S-IVB Reaction Control System Schematic

a. Pressurization System. The RCS pressurization system stores helium at 3100 ± 100 psia. Two regulators in series reduce the storage pressure to 196 ± 3 psia for propellant tank ullage pressure. Because a common pressurization system is used for both propellant tanks, quadruple check valves are located between the regulator and the tanks. These prevent the hypergolic propellants from mixing as a result of positive expulsion bladder leakage or permeation. A solenoid operated vent valve is provided for the low-pressure regulated helium system to provide pneumatic control of the expulsion bladders during loading and checkout. A relief valve gives protection against over-pressurization of the regulated helium system during ground and flight operations. All helium entering the regulated pressure area is filtered in the helium control module upstream of the pressure regulator.

b. Propellant Feed System. The positive expulsion propellant feed system provides propellant transfer to the engine under zero and random gravity conditions. The propellants are contained in two titanium cylindrical tanks with hemispherical ends, a Teflon expulsion bladder, and an axially-mounted concentric tube arrangement for fill, drain, and recirculation. The propellant feed distribution system incorporates filters and auxiliary ports for servicing, venting, and purge operations.

c. Thrusters. Three 150-pound-thrust engines are employed in each module. Each engine combustion chamber is constructed of ablative material. The chamber is integrally fabricated and composed of the combustion chamber, nozzle section and nozzle expansion cone. The engine employs quadruple propellant injector valves for redundant valve action. The injector consists of twelve pairs of unlike doublets arranged to minimize hot spots in the combustion chamber. The valve side of the injector is filled with a silver braze heat-sink that reduces injector operating temperatures. During the qualified engine life (330 seconds) the external wall temperature does not exceed 1060 degrees R and the maximum valve body external temperature does not exceed 625 degrees R. Other engine characteristics are given below.

RCS Engine Characteristics

Steady-state thrust (vac), lb_f	150
Steady-state specific impulse (vac), seconds	292
Minimum impulse bit, lb_f-seconds	7.5
Chamber pressure, psia	100
Nozzle expansion ratio	33.9:1
Fuel	MMH
Oxidizer	N_2O_4
Mixture ratio	1.6:1
Rated life, seconds	330
Weight, lb_m	27

d. Mounting. Figure 10.2-116 shows the selected location of the two RCS modules and their relationship to existing faired protrustions. The modules are offset 0 degrees 50 minutes from the vehicle position planes for alignment with existing external hat section stringers. This allows symmetrical location of module attachment structure relative to existing S-II structure.

The aft skirt structure is reinforced as shown in Figures 10.2-20 and 10.2-21. This provides load paths to S-II structural frames at Stations 223, 240, and 283. and provides adaptation for the RCS module to S-II interface, using present S-IVB module attachment locations and fittings. Design of these fittings incorporates insulation to prevent thermal transfer, and allows relative motion for thermal expansion between the module and the stage.

e. Fairing and Insulation. A closeout fairing between the RCS module and the LH_2 tank sidewall insulation is required for aerothermodynamic smoothness. Figure 10.2-23 shows changes to the LH_2 tank sidewall insulation and also shows the area of application of external cork insulation required to protect the insulation or structure in the vicinity of the RCS module against aerodynamic heating or thruster impingement.

f. Electrical System. Electrical control of the RCS is discussed under Astrionics.

10.2.5.2 Environment

10.2.5.2.1 Vibration Levels

Figures 10.2-122 and 10.2-123 give the sinusoidal and random vibration criteria derived for the RCS module based on predicted flight environments only. It is proposed that these data be used for qualification requirements, since the RCS is required only for flight and is not installed during static firing.

10.2.5.2.2 Aerodynamic Heating

Analysis was conducted comparing fairing temperatures using heat rates of the NR design aeroheating trajectory and the NASA design heat rates for the present fairing configuration. Results of the analysis using a simplified thermal model show that the fairing temperature using the NR heat rates will be lower than using the NASA design heat rates. It is therefore concluded that temperature caused by aerodynamic heating will not result in either a fairing structure or RCS component problem.

Additional insulation will be required on the S-II structure around the fairing due to protuberance effects on the S-II stage, as shown in Figure 10.2-124. This insulation will be similar to the present S-II insulation system around the LH_2 feedline fairing.

10.2.5.2.3 Base Heating

An analysis has been conducted to determine the possible effects of J-2S plume radiation on the base of the RCS unit. The analysis was conducted for the most outboard point on the RCS since this point should receive the greatest amount of radiation from the plumes. The J-2S plumes used for the analysis have the following properties:

MR	= 5.87	T_c	= 3520 degrees K
P	= 1231 psia	Altitude	= 300,000 feet

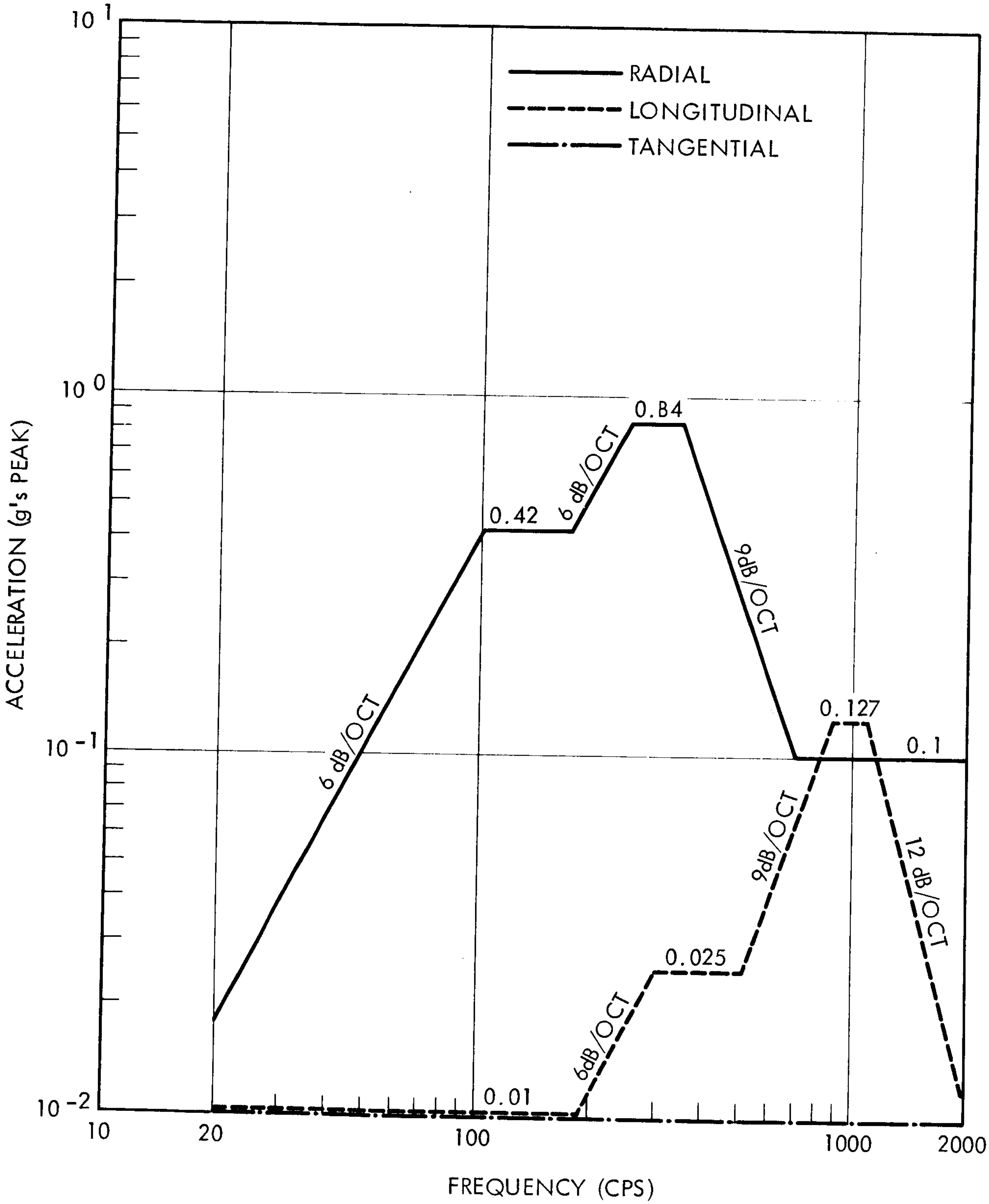


Figure 10. 2-122. Reaction Control System Random Vibration

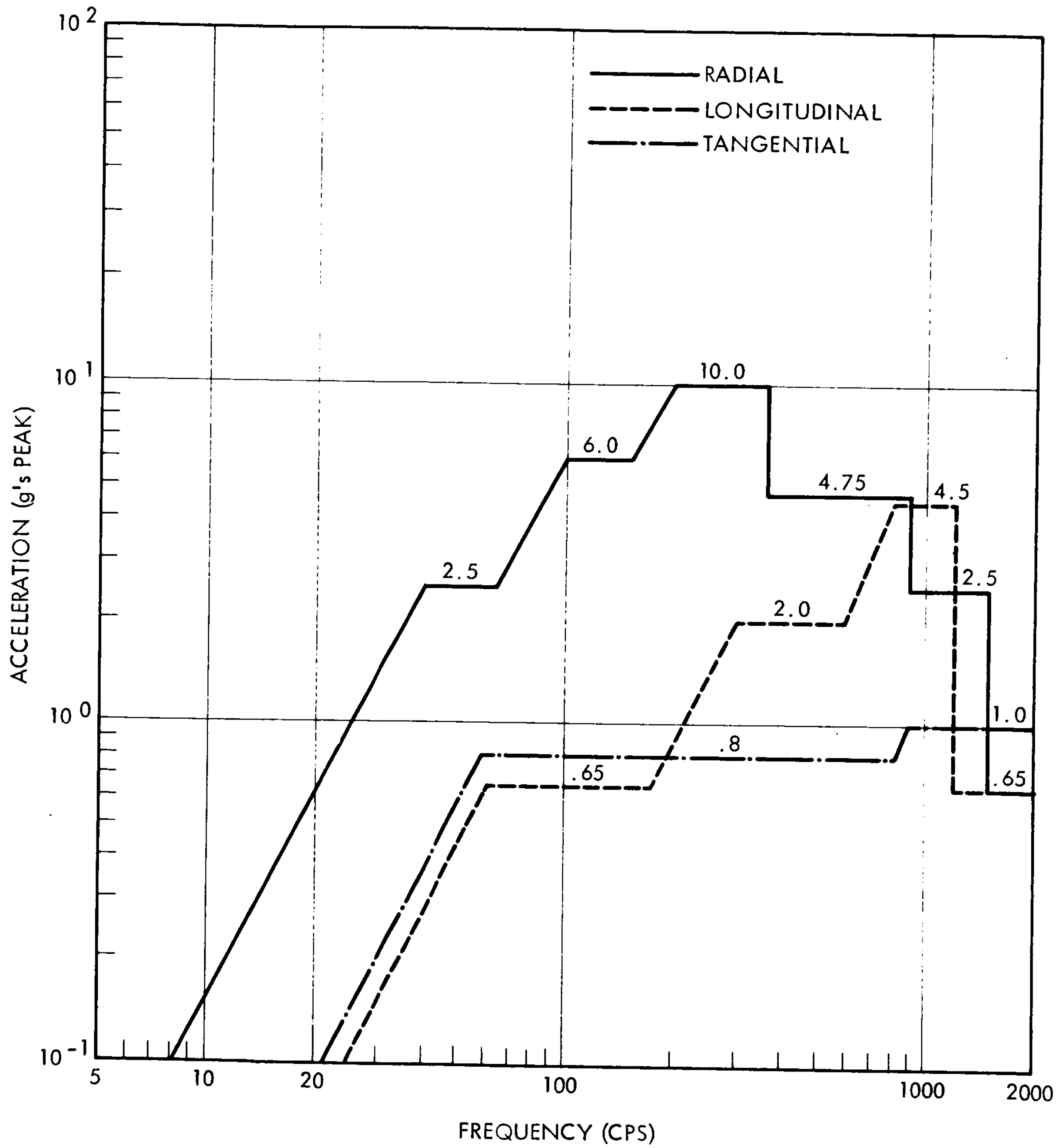


Figure 10. 2-123. Reaction Control System Sinusoidal Vibration

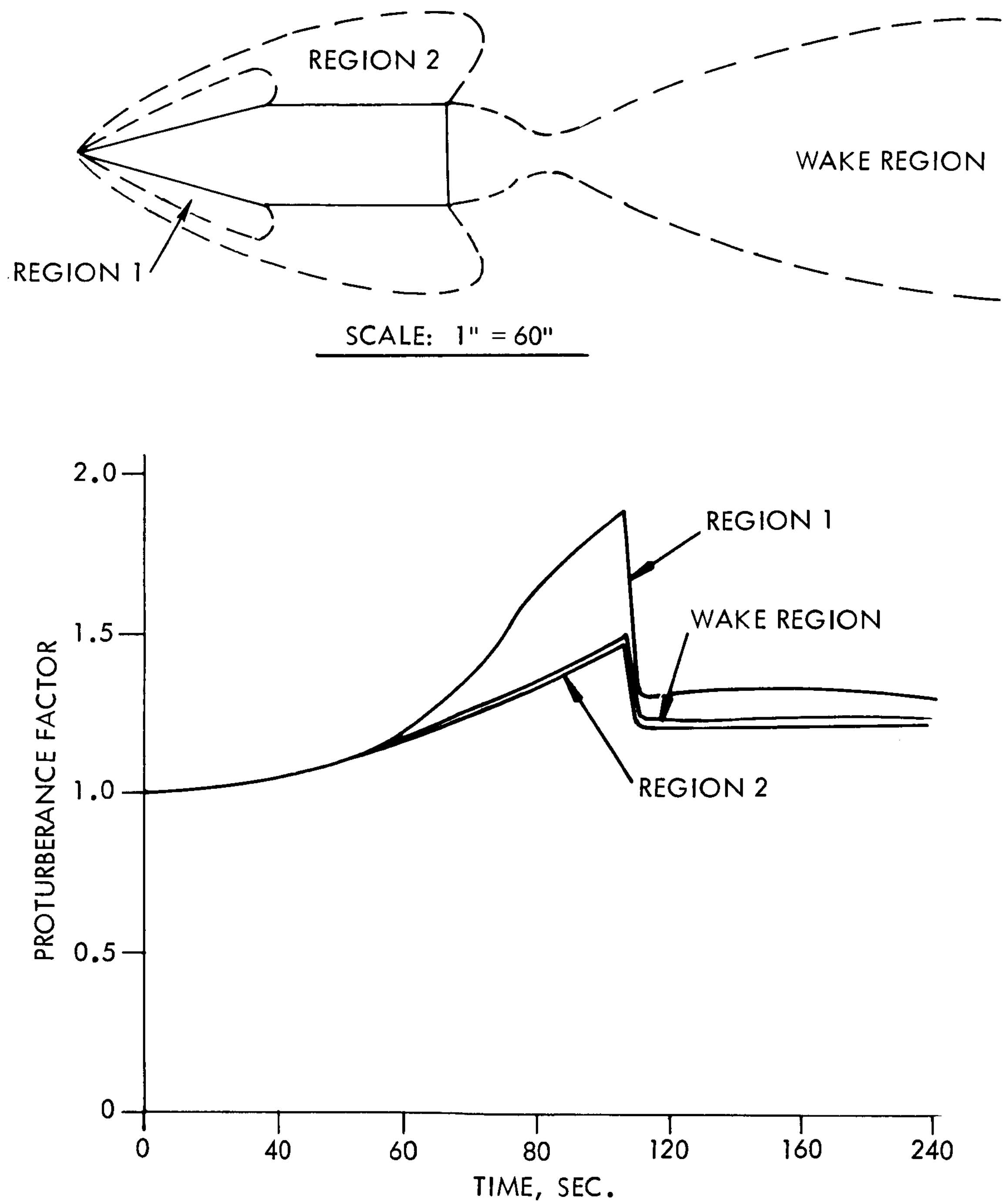


Figure 10. 2-124. Surface Regions Influenced by Interference Heating from Reaction Control System Fairing

The radiation resulting from this plume was computed for the condition with the interstage off. This will give the greatest amount of radiation since more of the plume is exposed to the RCS unit.

The analysis resulted in a radiation heat flux of 0.06 Btu per square foot per second.

No additional thermal protection for radiant heating will be required since the design will be based on higher aerodynamic heating rates.

10.2.5.2.4 Aerodynamic Loading

Analysis of a limited nature has indicated a possible problem associated with the 30-degree nose cone profile on the module as presently designed. A quick-look has shown a 15-degree nose cone profile to be more favorable on the module when mounted on the S-II stage from an aerodynamic loading and heating standpoint. Consideration was given to converting the 30-degree nose cone profile to the 15-degree nose profile, but design problems were encountered regarding the capability either to cantilever the longer 15-degree nose cone on the current RCS module and/or supporting structurally the 15-degree nose cone from the stage sidewall. For the purpose of this study, a decision was therefore made to retain the current 30-degree nose cone profile.

Further analytical work for a 30-degree nose cone profile to define further aerodynamic loading and heating environment, and a comparison of the results of these analyses with the current configuration capability, must be accomplished to make a final resolution of fairing configuration.

10.2.5.3 Ground Servicing Equipment

The GSE requirements for the RCS modules consist of installation and checkout equipment for the VAB, and module servicing equipment for the launch facility. Specific equipment requirements for the VAB have not been defined, but GSE is required to provide the following functions:

- RCS module transport and checkout
- Pneumatic and electrical checkout equipment for preinstallation checks
- Module installation equipment
- Postinstallation electrical continuity checkout equipment

At the launch facility, the RCS module servicing operations are performed manually with the exception of helium storage sphere pressurization. Propellants are loaded on board before the launch countdown begins. Fluid and gas distribution drag-in flex hoses for system purge, propellant transfer and pneumatic requirements for module servicing are connected manually to the RCS module. A description of the servicing equipment is given below.

10.2.5.3.1 Propellant Purge Servicing

The RCS module purge system provides gaseous nitrogen to the fuel and oxidizer subsystems for moisture removal prior to propellant loading. Individual purge equipment is used for purging the fuel and oxidizer systems of the RCS.

Nitrogen purge gas is supplied to the module servicing console (Douglas designation: Model 472 for fuel and Model 473 for oxidizer) and transferred to the RCS through the fill line (Figure 10.2-125). The gas circulates through the individual propellant systems and is vented to atmosphere. The system purge lasts for approximately five minutes. The engine valves are then opened, allowing the purge gases to flow through the engines. The engine valves are closed after approximately two minutes, and the system purge is complete.

The system supplies gaseous nitrogen to the module at a pressure of 30 ± 2 psig and at ambient temperature.

10.2.5.3.2 Propellant Circulation and Loading System

The system provides for circulation of propellants between the North American Rockwell mobile servicer and the RCS models 472 and 473 servicing consoles until the propellants are conditioned to the proper temperature and pressure for RCS loading (Figure 10.2-126). When the propellant conditioning is achieved, the propellant is diverted to the module through the fill line. As propellant accumulates in the module tank, displaced gases are forced through the recirculation lines back to the console and down to the NR mobile servicer. An increase in propellant pressure will indicate when the module tank is full. Observation of a sight glass on the console will show when the system is free of bubbles. After the tank is filled, a measured amount of propellant will be off-loaded to meet the module propellant loading requirements. The propellant is off-loaded through the fill line back to the console and down to the NR mobile servicing unit.

Fuel system operating parameters are as follows:

Pressure, propellant transfer	30 ± 2 psig
propellant return	23 ± 1 psig
Temperature	80 ± 5 F
Flowrate	0.5 to 1.5 gpm

The fuel gas bleed system is part of the console and is connected to the RCS module tank recirculation line by a flex hose (Figure 10.2-127). The system removes gas which has accumulated above the propellant inside the RCS tank bladder.

10.2.5.3.3 Temperature Control

It is assumed that the prelaunch thermal conditioning nitrogen purge requirements for the RCS modules will be the same as those on the S-IVB: 33 pounds per minute (each) at approximately 100 F. The three following methods of providing these requirements were investigated:

1. New independent system. This method has the obvious advantage of allowing optimization of flow rate and temperature without compromising other system requirements. However, it has the equally obvious disadvantage of requiring

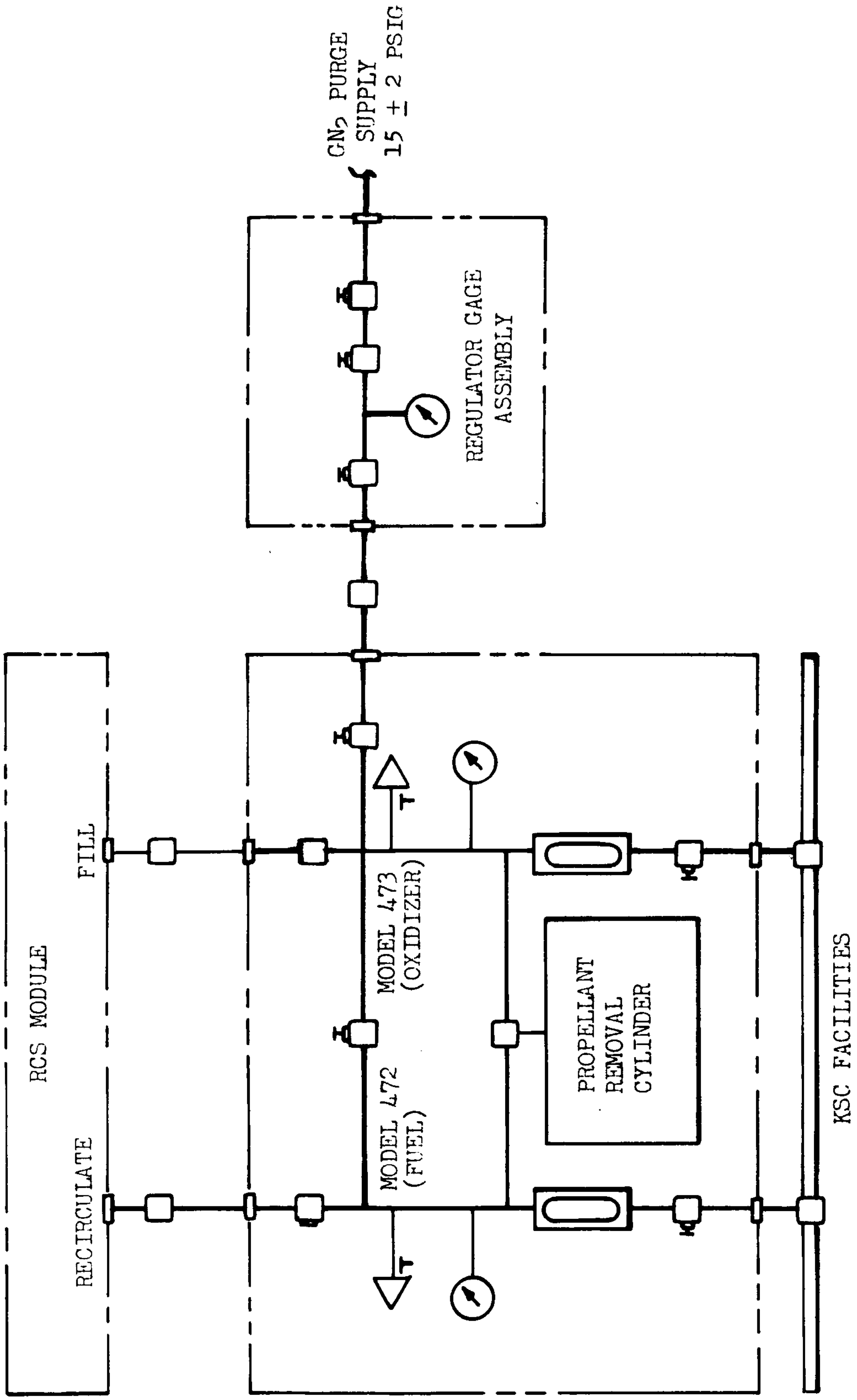


Figure 10.2-125. Reaction Control System Module Purge System

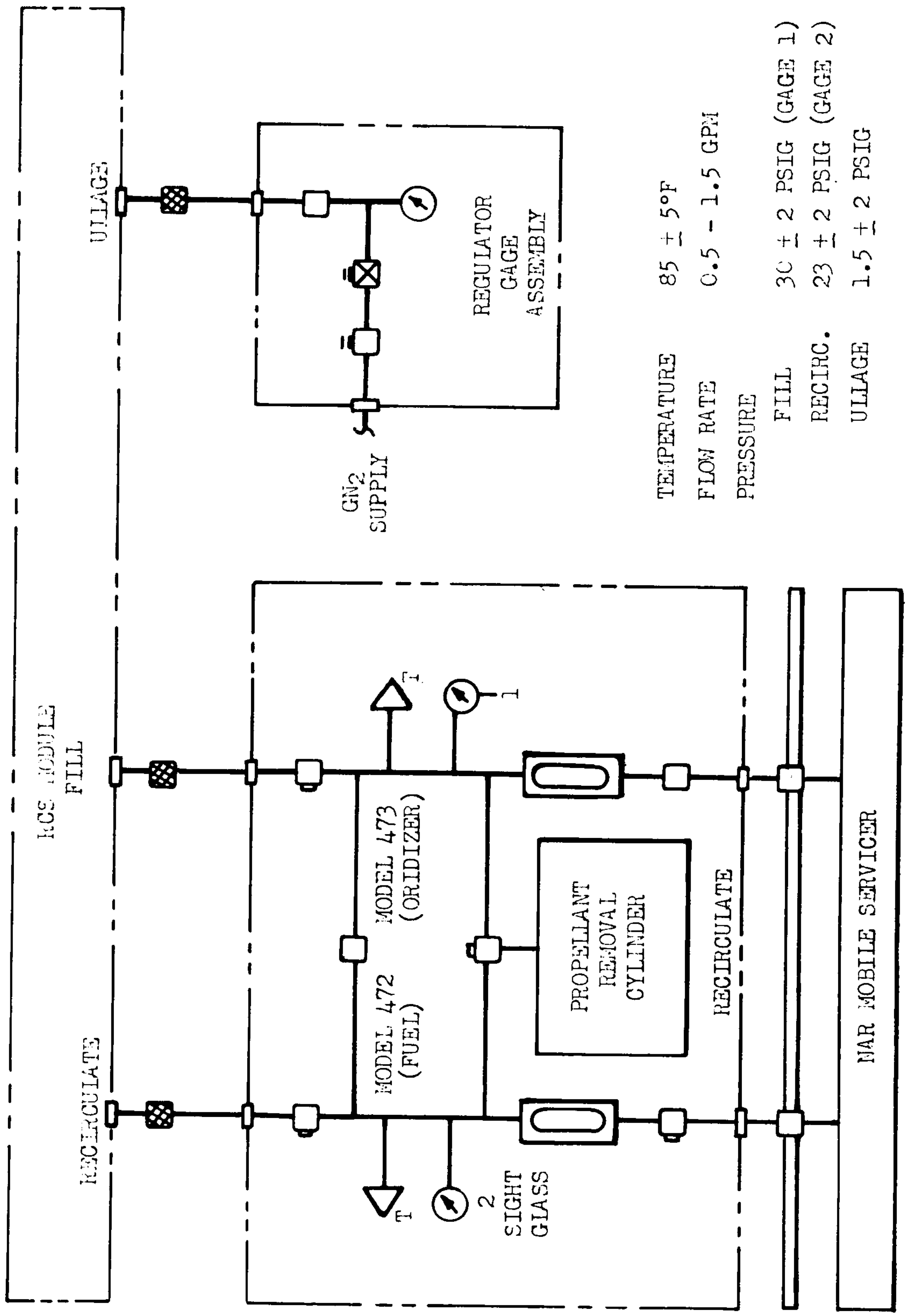


Figure 10.2-126. Reaction Control System Servicing System

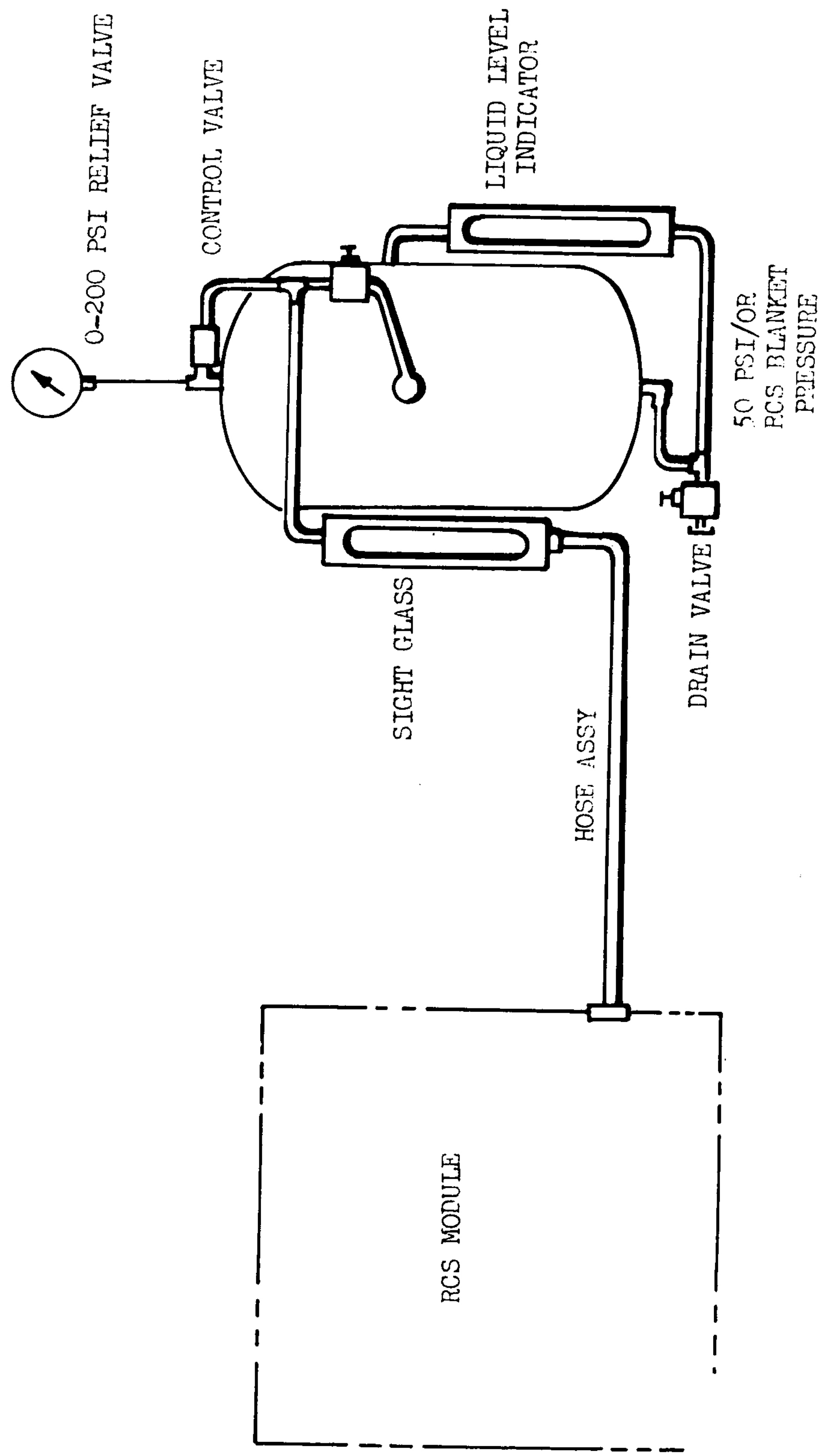


Figure 10. 2-127. Reaction Control System Gas Bleed System

a new umbilical connection and a revision to the facility ducting. (It is possible that the S-IVB engine compartment duct would be modified to suit this purpose.)

2. Tap-off from aft thermal control system. This method would require modification of the umbilical connection and the facility duct system, since the new total flow would be approximately three times that of the present system. The one attractive feature is that the temperature requirements of both systems are compatible.

3. Tap-off from engine compartment conditioning system (ECCS). Preliminary studies show that since there will be no thrust chamber chilldown on the J-2S concept, it appears feasible to reduce the ECCS purge temperature to a value which will satisfy both systems. However, the required purge temperature will call for operation of the facility system below its present calibration limits. Therefore, the existing control circuitry (thermistors in S-II engine compartment and associated controls) must be calibrated down to approximately 0 F (presently calibrated to 30 F minimum) or the concept revised to that employed in the thermal control systems (control by temperature sensor in duct).

Tapping off from the ECCS appears to require the least modifications and will be employed unless shown not to be feasible by more detailed analyses. Assuming this method is feasible, two 4-inch-diameter stubs will be added to the V7-417501 duct at Station 196, near position II. From these take-off locations, 4-inch-diameter, light gauge, aluminum ducts will be routed under the existing thermal control system ducts to each of the two RCS modules. Figure 10.2-128 shows the proposed routing.

The new ducting will be supported, where possible, by attaching to the existing ducts with clamps and brackets as shown in Figure 10.2-129. In locations where there is no existing thermal control duct, the new duct will be routed closer to the Station 196 frame and attached to the basic structure. All RCS ducting, excluding the stub-outs on the ECCS duct, will be field-installed at KSC to preclude interference with the static firing ECCS manifold (GSE) used only at MTF.

10.2.5.3.4 Pneumatic Pressurization System

The RCS pneumatic pressurization system pressurizes the helium storage tank for flight through a stage-mounted system (Figure 10.2-128). The system also maintains propellant ullage pressure after propellant loading, before pressurization of the helium tanks. The pneumatic system is pressurized in a three-step operation to maintain ullage pressure on the propellants during standby and to meet safety and vehicle system requirements. The operating parameters are listed below:

Operation	Pressure (psig)	Temperature (F)	Flowrate (lb/min maximum)
Propellant ullage control	50	Ambient	0.01
Attitude control system Helium tank prepressurization	1450	Ambient	0.35
Attitude control system Helium tank pressurization	3100	Ambient	0.75

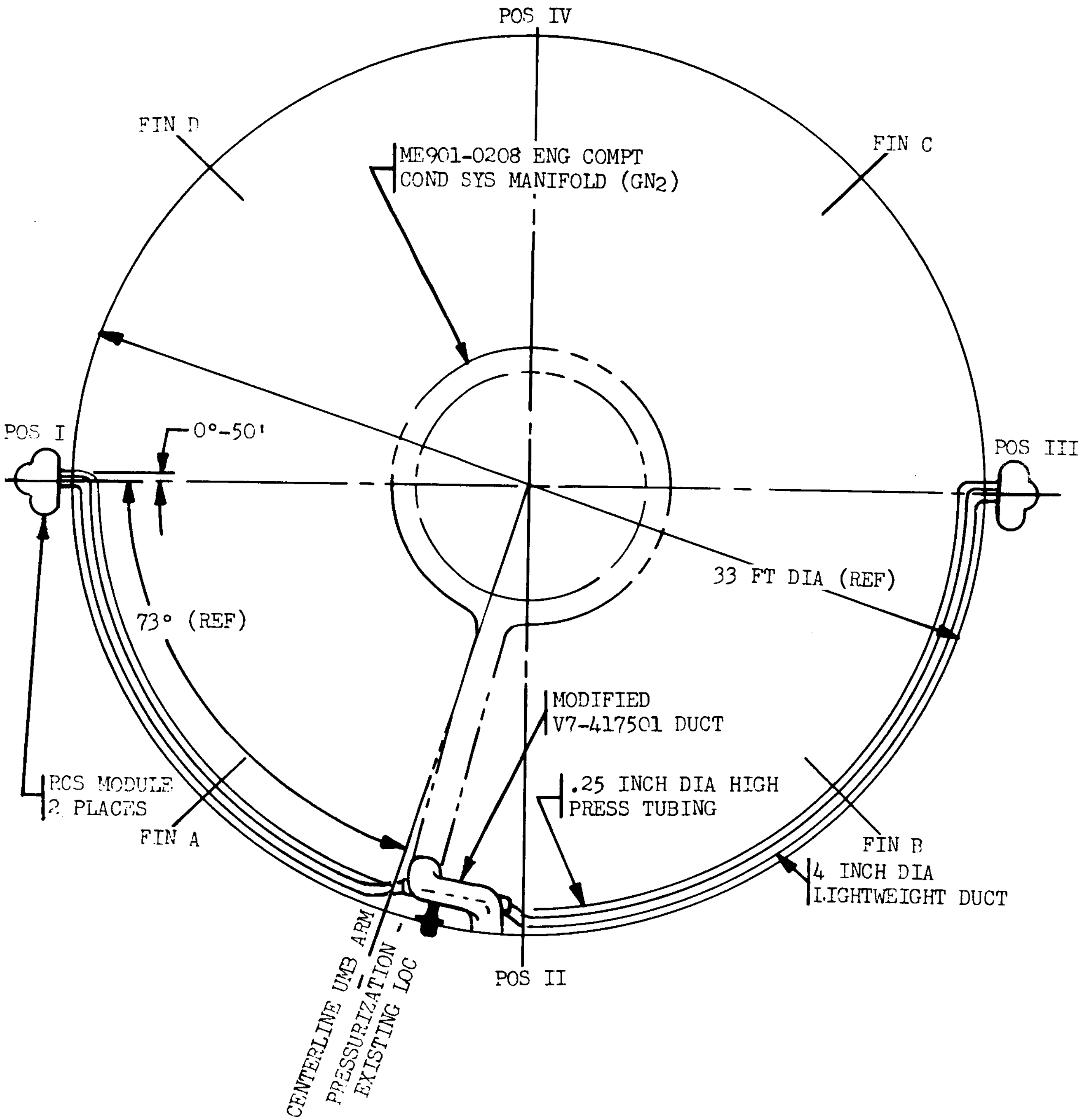


Figure 10.2-128. Reaction Control System Conditioning (GN_2) and Helium Pressurization System

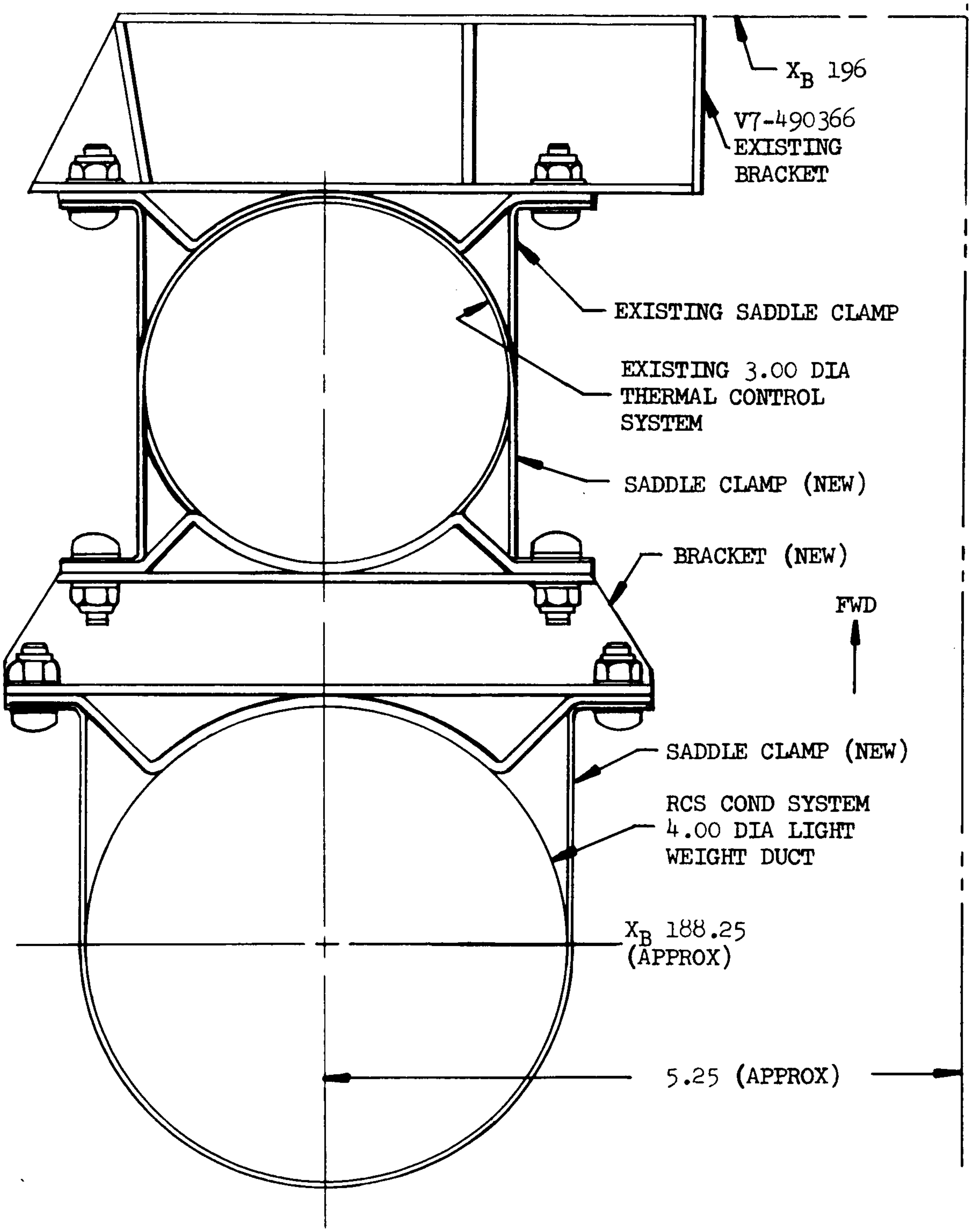


Figure 10.2-129. Reaction Control System Conditioning Ducts

10.2.6 WEIGHT AND MASS PROPERTIES

This section contains the weight and mass properties of the S-II stage with J-2S engines for the LOR and LEO missions. The data are presented in the form of tables and charts. Detailed computer printouts of the data are available, if desired. The baseline configuration for this data is the MSFC J-2S Improvement Study Baseline Launch Vehicle SA-511, dated September 1968.

10.2.6.1 LOR Mission

The installation of the J-2S engine will increase the S-II stage dry weight by 1862 pounds. A brief description of the affected components is as follows:

a. Engine and Accessories. The replacement of J-2 engines with J-2S engines will result in a net weight increase of 1218 pounds.

b. Thrust Structure. Structure will be redesigned to accommodate the increased thrust loads of the J-2S engines, resulting in a net weight increase of 2000 pounds.

c. Fairings. The propellant recirculation line fairings are deleted, because the propellant recirculation system is not required with the J-2S engine. The weight decrease is 81 pounds.

d. Base Heat Shield. The rigid base heat shield thickness is increased by 0.5-inch, and a steel wire mesh is added to the exposed surface, resulting in a weight increase of 30 pounds.

e. Purge and Leak Detection System. Modification to existing purge system to meet J-2S configuration will result in a 30-pound weight increase.

f. Fuel and Oxidizer Recirculation System. Propellant recirculation systems are not required for the J-2S configuration. A fuel system weight decrease of 910 pounds and oxidizer system weight decrease of 305 pounds are realized.

g. Electrical System. A 120-pound weight reduction results from the removal of the propellant recirculation system.

h. S-IC/S-II Interstage. A total weight reduction of 1099 pounds results from the following: deletion of the ullage system, 834 pounds; deletion of propellant recirculation system components, 155 pounds; decrease of measurement system commensurate with ullage system deletion, 70 pounds; decrease of structure by 40 pounds.

10.2.6.1.1 S-II Stage Dry Weight and Longitudinal Centers of Gravity

These data are presented in Table 10.2-XXII.

10.2.6.1.2 S-II Stage Propellant Weights and Longitudinal Centers of Gravity

These data are presented in Table 10.2-XXIII.

10.2.6.1.3 S-II Interstage Weights and Longitudinal Centers of Gravity

Data for the lower and upper aft interstage structures are presented in Table 10.2-XXIV

10.2.6.1.4 S-II Drop Weight Event History

Data are shown on Table 10.2-XXV.

10.2.6.1.5 S-II Stage Mass Characteristics

The vehicle weights, centroids, and moments of inertia are shown in Table 10.2-XXVI in English and International units.

10.2.6.1.6 S-II Stage Weight and Center of Gravity Versus Burn Time

These data are shown in Figure 10.2-130.

10.2.6.1.7 S-II Stage Moment of Inertia Versus Burn Time

Roll and pitch moments of inertia in International units versus burn time are shown in Figure 10.2-131.

10.2.6.1.8 S-II Weight Distribution and Cantilevered Items at Ground Ignition

Weight distribution is listed in Table 10.2-XXVII and is plotted in Figure 10.2-132. Cantilevered items are shown in Table 10.2-XXVIII.

10.2.6.1.9 S-II Weight Distribution and Cantilevered Items at Engine Cutoff

Weight distribution are listed in Table 10.2-XXIX and plotted on Figure 10.2-133. Cantilevered items are shown in Table 10.2-XXX.

10.2.6.2 LEO Mission

In addition to the weight changes described above for the LOR mission, the following weight changes will be in effect for the LEO mission:

a. Propellant Container. Addition of a lower antislosh baffle in the LOX tank will result in a 150-pound increase.

b. Fairings. Additional fairings for the LH_2 balanced vent system and the reaction control system will produce a 40-pound increase.

c. Fuel System. The added LH_2 balanced vent system causes a 25-pound increase.

d. Stage Control System. The existing four auxiliary hydraulic motors will be replaced with four 28 vdc motors, a four-battery power supply, and associated cabling, mounts, and plumbing. The net weight increase is 500 pounds.

e. Reaction Control System. A completely new system will be added, with a total dry weight of 946 pounds.

Table 10.2-XXII. S-II Stage Dry Weight and Longitudinal Centers of Gravity, LOR Mission

Item	Weight (lb)	S-II Body X-Station (in.)	S-V Body X-Station (in.)
Structure	(49,013)	(414.8)	(1978.8)
Propellant Container	29,179	465.5	2029.5
Forward Skirt	4,057	885.3	2449.3
Aft Skirt	4,130	241.0	1805.0
Thrust Structure	9,302	157.3	1721.3
Fairings and Associated Structures	1,351	340.4	1904.4
Base Heat Protection	660	60.7	1624.7
Paint and Sealer	334	592.3	2156.3
Propulsion System and Accessories	(26,472)	(121.9)	(1685.9)
Engine ans Accessories	19,248	715.4	2279.4
Purge System	416	266.8	1830.8
Fuel System	3,166	374.4	1938.4
Oxidizer System	2,519	172.1	1736.1
Stage Control System	1,123	113.0	1677.0
Equipment and Instrumentation	(6,645)	(381.2)	(1945.2)
Environmental Control System	1,121	370.2	1934.2
Guidance System	481	570.3	2134.3
Telemetry and Measurement Equipment	2,734	380.7	1944.7
Propellant Utilization System	634	374.4	1938.4
Electrical System	829	215.8	1779.8
Range Safety Equipment	319	757.4	2321.4
Pneumatic System	410	221.4	1785.4
Separation System	117	468.5	2032.5
Growth Allowance	418	321.3	1885.3
Total S-II Dry Stage Weight	82,548	317.7	1881.7

Table 10.2-XXIII. S-II Stage Propellant and Longitudinal Centers of Gravity, LOR Mission

Item	Weight (lb)	S-II Body X-Station (in.)	S-V Body X-Station (in.)
Residuals and Reserve Propellants	(13,390)	(295.5)	(1859.5)
Fuel Pressurizing Gas	1,594	650.2	2214.2
Fuel - Propellant Utilization Residual Allowance	530	369.2	1933.2
Fuel - Thrust Decay	200	374.3	1938.3
Fuel - Trapped	2,651	334.2	1898.2
Oxidizer Pressurizing Gas	3,992	283.8	1847.8
Oxidizer - Thrust Decay	455	159.0	1723.0
Oxidizer - Trapped	3,898	141.3	1705.3
Service Items	70	115.3	1679.3
Standard Propellant Consumption	(967,890)	(331.5)	(1895.5)
Fuel	154,289	638.5	2202.5
Oxidizer	813,601	273.1	1837.1
Other Weight Items	(2,550)	(455.6)	(2019.6)
Thrust Buildup			
Fuel	725	896.9	2460.9
Oxidizer	1,825	367.4	1931.4
Consumed prior to Ignition			
Service Items	(105)	(190.5)	(1754.5)
Vented Gases - Common Bulkhead	38	371.1	1935.1
Engine Start Cartridge Propellant	67	88.0	1652.0

Table 10.2-XXIV. S-II Stage Interstage Weights and Longitudinal Centers of Gravity, LOR Mission

Item	Weight (lb)	S-II Body X-Station (in.)	S-V Body X-Station (in.)
S-IC/S-II Interstage Documented to S-IC Stage Stations 0 to -23			
Structure	(1481)	(- 12.3)	(1551.7)
Interstage Structure	1471	- 12.3	1551.7
Paint and Sealer	10	- 12.0	1552.0
Equipment and Instrumentation	(67)	(- 2.4)	(1561.6)
Separation System	57	- 2.6	1561.4
Guidance System	10	- 1.6	1562.4
Total S-IC/S-II Interstage Documented to S-IC Stage Dry Weight	1548	- 11.9	1552.1
S-IC/S-II Interstage Documented to S-II Stage Stations 0 to 196.0			
Structure	(8127)	(93.9)	(1650.6)
Interstage Structure	7838	93.6	1657.6
Fairing and Associated Structures	149	107.5	1456.6
Paint and Sealer	140	98.2	1465.8
Propulsion System	(7)	(160.0)	(1404.0)
LOX Fill and Drain Disconnect	7	160.0	1404.0
Equipment and Instrumentation	(202)	(133.9)	(1430.1)
Telemetry and Measuring	60	107.3	1456.7
Separation System	85	169.2	1394.8
Guidance System	57	109.3	1454.7
S-IC/S-II Interstage Documented to S-II Stage Dry Weight	8336	94.9	1658.9

Table 10.2-XXV. S-II Drop Weight Event History
LOR Mission

Mission Event Time (sec)	Item Description	Weight (lb)
S-IC Mainstage		
	Bulkhead Purge Gas	38
161 (S-IC OECO)		
	S-IC/S-II Interstage (Small)	1,548
	J-2S Start Cartridge Propellant	67
165 (S-II Ignition) (Ignition to 90% P. C.)	LOX Thrust Buildup	1,825
	LH_2 Thrust Buildup	725
Mainstage Burn		
194 (S-II Prior to Interstage Jettison)		
	S-IC/S-II Interstage (Large)	8,336
Mainstage Burn		
	LOX Thrust Decay	455
	LH_2 Thrust Decay	200
499 (S-II Cutoff and Separation)		
	Dry Stage	82,548
	LOX Remaining in Tank (Including sump)	2,545
	LH_2 Remaining in Tank	2,940
	LOX Tank Pressurant	3,992
	LH_2 Tank Pressurant	1,594
	LOX Trapped (Outside of Tank)	1,353
	LH_2 Trapped (Outside of Tank)	241
	Service Items (Hydraulic oil, etc.)	70

Table 10.2-XXVI. S-II Stage Mass Characteristics LOR Mission (English Units)

Event	Weight (lb)	Station (in.)			I_{xx} Slug Ft^2	I_{yy} Slug Ft^2	I_{zz} Slug Ft^2
		X	Y	Z			
Vehicle Stage Dry Weight	82,548	317.7	5.9	-3.3	445,567	1,527,363	1,536,679
Total S-IC/S-II Interstage Documented to S-IC Stage	1,548	-11.9	1.6	-0.8	13,012	6,647	6,633
Total S-IC/S-II Interstage Documented to S-II Stage	8,336	94.9	3.3	-1.4	69,972	41,344	42,595
S-II Stage at S-II Engine Start Command	1,074,719	328.7	0.5	-0.3	522,025	5,840,517	5,851,329
S-II Stage at Second-Plane Separation - Before	987,342	319.2	0.5	-0.3	522,019	5,182,639	5,193,498
S-II Stage at Second-Plane Separation - After	979,006	321.1	0.5	-0.3	452,039	5,049,827	5,059,580
S-II Stage at LES Jettison	963,635	319.5	0.5	-0.3	452,039	4,943,415	4,953,179
S-II Cutoff Signal	95,943	314.6	5.1	-2.8	451,405	1,613,255	1,622,924

Note: S-II Body Station 0 is Saturn V Station 1564.0

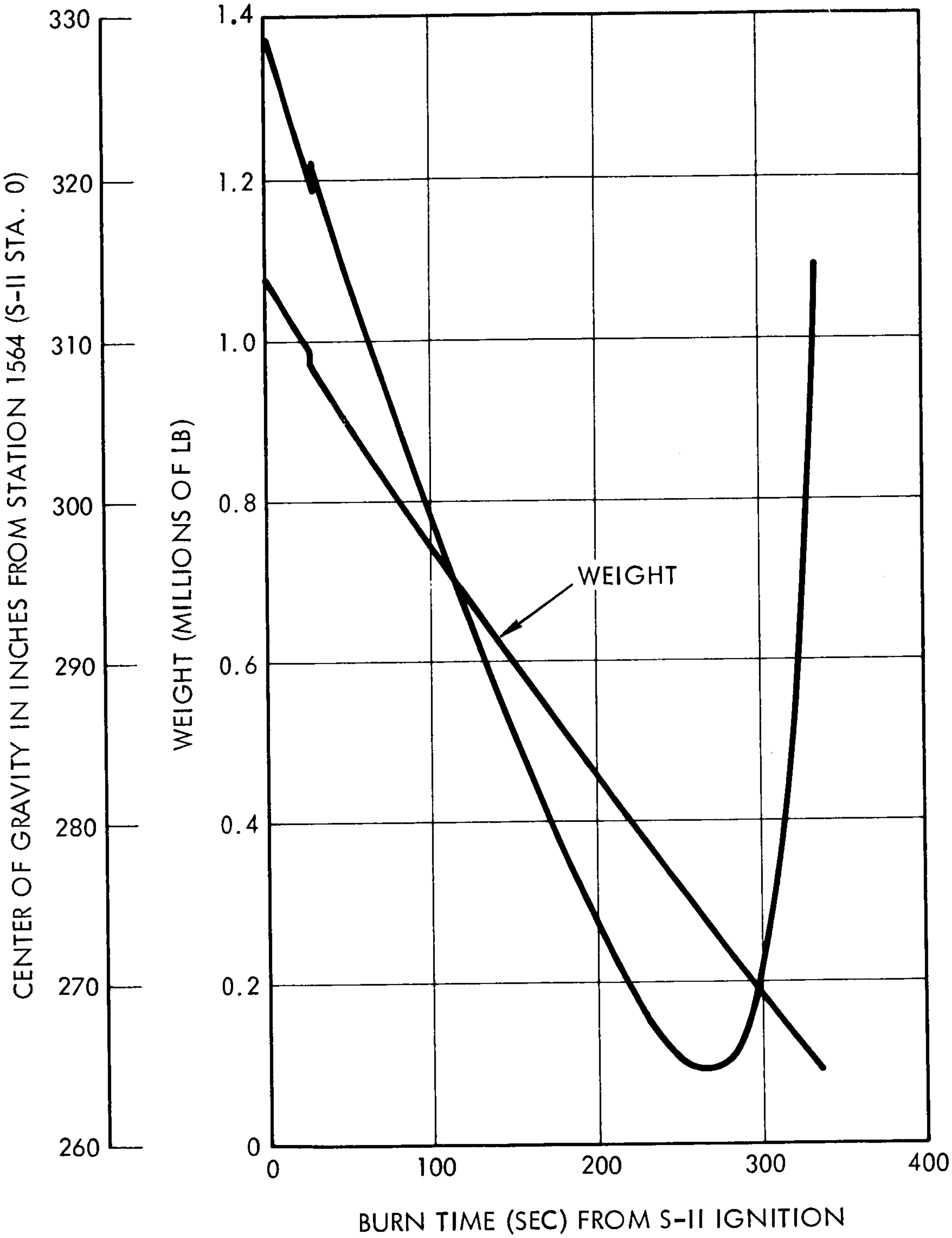


Figure 10.2-130. S-II Stage Weight and Center of Gravity Versus Burn Time (LOR Mission)

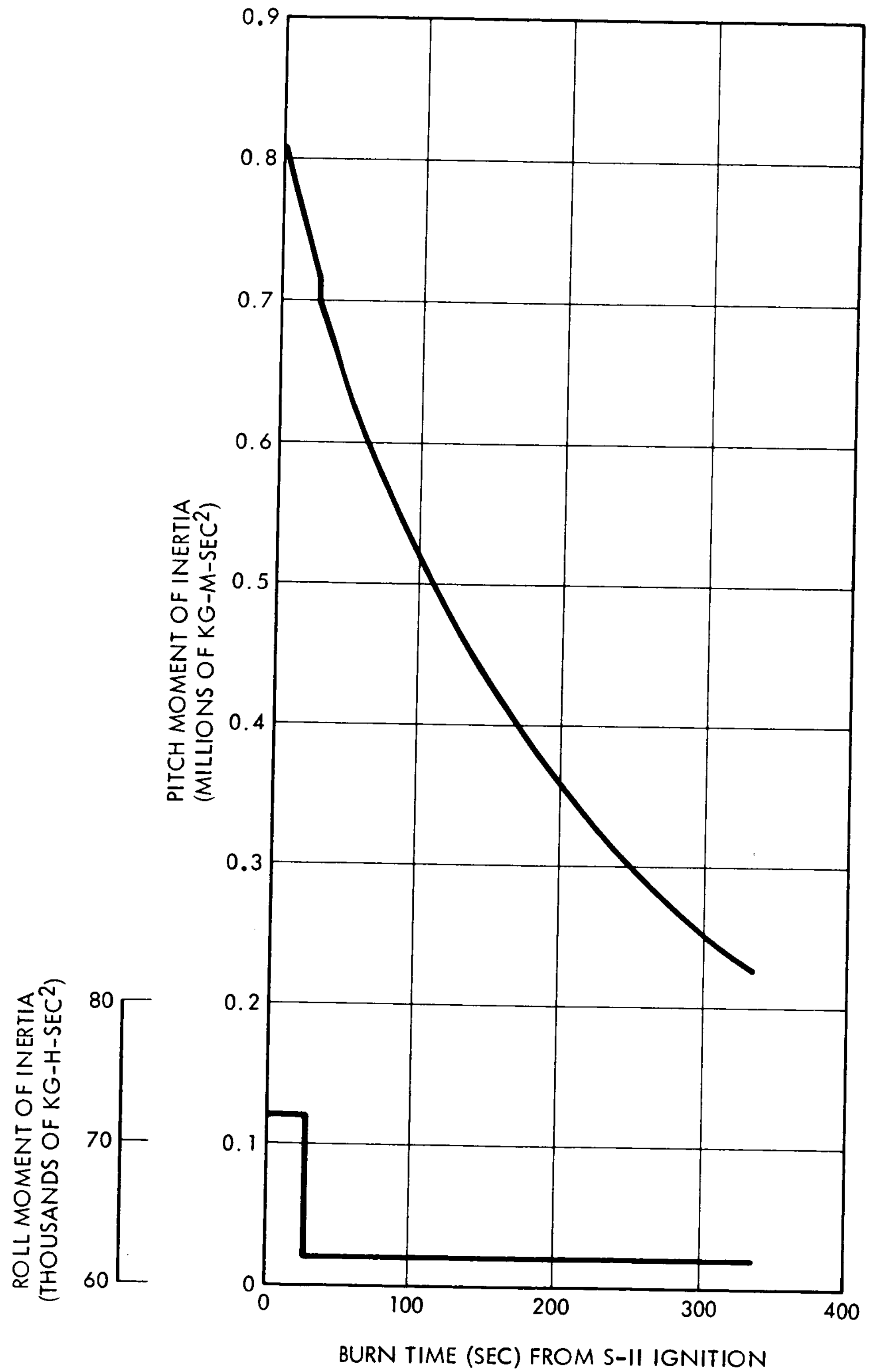


Figure 10.2-131. S-II Stage Roll and Pitch Inertia Versus Burn Time (LOR Mission)

Table 10.2-XXVI. S-II Stage Mass Characteristics, LOR Mission (International Units) (Cont)

Event	Weight (lb)	Station (in.)			I_{xx} (Kg-M-sec^2)	I_{yy} (Kg-M-sec^2)	I_{zz} (Kg-M-sec^2)
		X	Y	Z			
Vehicle Stage Dry Weight	82,548	317.7	5.9	- 3.3	61,602	211,166	212,454
Total S-IC/S-II Interstage Documented to S-IC Stage	1,548	- 11.9	1.6	- 0.8	1,799	919	917
Total S-IC/S-II Interstage Documented to S-II Stage	8,336	94.9	3.3	- 1.4	9,674	5,716	5,889
S-II Stage at S-II Engine Start Command	1,074,719	328.7	0.5	- 0.3	72,178	807,482	808,977
S-II Stage at Second-Plane Separation - Before	987,342	319.2	0.5	- 0.3	72,177	716,527	718,028
S-II Stage at Second-Plane Separation - After	979,006	321.1	0.5	- 0.3	62,501	698,165	699,513
S-II Stage at LES Jettison	963,635	319.5	0.5	- 0.3	62,501	683,453	684,802
S-II Cutoff Signal	95,943	314.6	5.1	- 2.8	62,413	223,041	224,377

S-II Body Station 0 is Saturn V Station 1564.0

Table 10.2-XXVII. S-II Weight Distribution at Ground Ignition LOR Mission

X-Axis	Weight (lb)	Weight/Inch (lb)	X-Axis	Weight (lb)	Weight/Inch (lb)	X-Axis	Weight (lb)	Weight/Inch (lb)
-23--12	799.1	72.6	312-324	1014.3	84.5	636-648	4064.9	338.7
-12- 0	810.9	67.6	324-336	1000.7	83.4	648-660	4118.8	343.2
0- 12	882.6	73.5	336-348	1154.3	96.2	660-672	4056.3	338.0
12- 24	399.1	33.3	348-360	1456.5	121.4	672-684	4051.9	337.7
24- 36	469.7	39.1	360-372	1478.7	123.2	684-696	4105.4	342.1
36- 48	564.1	47.0	372-384	2610.1	217.5	696-708	4042.9	336.9
48- 60	425.6	35.5	384-396	3001.2	250.1	708-720	4050.5	337.5
60- 72	426.7	35.6	396-408	3202.1	266.8	720-732	4044.2	337.0
72- 84	627.7	52.3	408-420	3248.5	270.7	732-744	4076.8	339.7
84- 96	437.4	36.4	420-432	3449.6	287.5	744-756	4033.3	336.1
96-108	422.8	35.2	432-444	3644.0	303.7	756-768	4045.1	337.1
108-120	597.9	49.8	444-456	3736.5	311.4	768-780	4114.8	342.9
120-132	376.1	31.3	456-468	3897.5	324.8	780-792	4067.9	339.0
132-144	387.3	32.3	468-480	4115.8	343.0	792-804	4057.4	338.1
144-156	398.8	33.2	480-492	4182.7	348.6	804-816	4154.4	346.2
156-168	642.3	53.5	492-504	4190.4	349.2	816-828	4344.4	362.0
168-180	426.3	35.5	504-516	4246.0	353.8	828-840	4253.5	354.5
180-192	529.6	44.1	516-528	4220.4	351.7	840-852	4401.4	366.8
192-204	1170.2	97.5	528-540	4270.1	355.8	852-864	4466.5	372.2
204-216	669.9	55.8	540-552	4127.8	344.0	864-876	4006.0	333.8
216-228	728.8	60.7	552-564	4070.1	339.2	876-888	2841.3	236.8
228-240	860.4	71.7	564-576	4070.0	339.2	888-900	2599.2	216.6
240-252	855.0	71.3	576-588	4127.7	344.0	900-912	676.4	56.4
252-264	708.3	59.0	588-600	4069.4	339.1	912-924	642.4	53.5
264-276	718.1	59.8	600-612	4068.7	339.1	924-936	679.9	56.7
276-288	1096.9	91.4	612-624	4148.4	345.7	936-948	677.3	56.4
288-300	942.3	78.5	624-636	4082.2	340.2	948-960	531.7	44.3
300-312	731.8	61.0						
						Total	200,193.8	

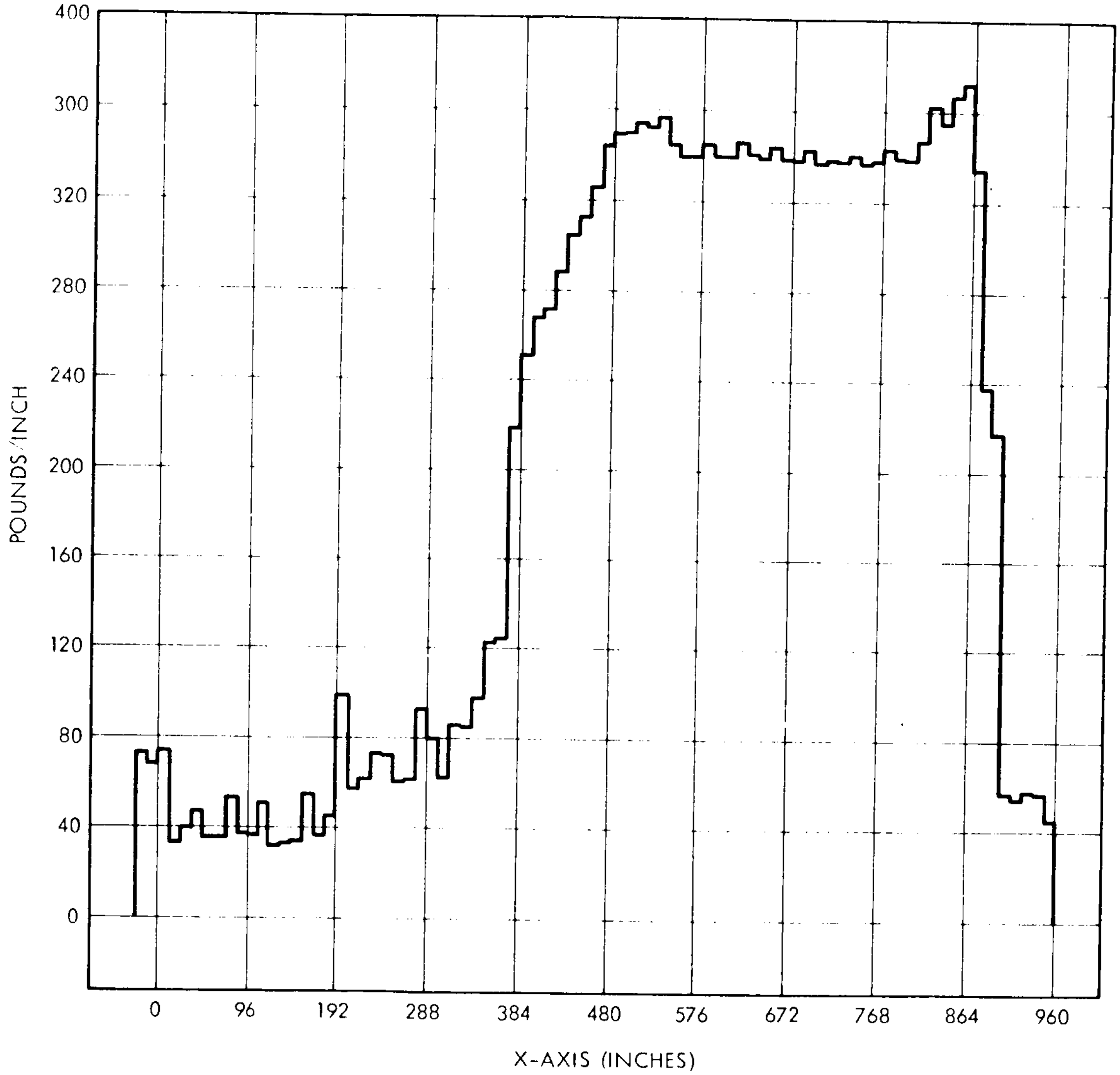


Figure 10. 2-132. S-II Weight Distribution at Ground Ignition (LOR Mission)

Table 10.2-XXVIII. S-II Stage at Ground Ignition
Cantilevered Items LOR Mission

Thrust Structure	
Point of Attachment	S-II Station 223 Saturn-V Station 1787
Weight	38,447 pounds
Center of Gravity	S-II Station 108.1 Saturn-V Station 1672.1
Pitch Moment of Inertial about Point of Attachment	26,366 Kg-M-sec^2 190,705 Slug-Feet^2
LOX Tank	
Point of Attachment	S-II Station 284 Saturn-V Station 1848
Weight	835,187 pounds
Center of Gravity	S-II Station 273.6 Saturn-V Station 1837.6
Pitch Moment of Inertia about Point of Attachment	266,289 Kg-M-sec^2 1,926,068 Slug-Feet^2
Forward Bulkhead	
Point of Attachment	S-II Station 823 Saturn-V Station 2387
Weight	2540 pounds
Center of Gravity	S-II Station 898.3 Saturn-V Station 2462.3
Pitch Moment of Inertia about Point of Attachment	1363 Kg-M-sec^2 9859 Slug-Feet^2

Table 10.2-XXIX. S-II Weight Distribution at Engine Cutoff, LOR Mission

X-Axis	Weight (lb)	Weight/Inch (lb)	X-Axis	Weight (lb)	Weight/Inch (lb)	X-Axis	Weight (lb)	Weight/Inch (lb)
-23--12	......	-0.0	312-324	1014.3	84.5	636-648	337.2	28.1
-12- 0	......	-0.0	324-336	1000.7	83.4	648-660	394.4	32.9
0- 12	......	-0.0	336-348	1154.2	96.2	660-672	335.7	28.0
12- 24	......	-0.0	348-360	1456.5	121.4	672-684	335.0	27.9
24- 36	......	-0.0	360-372	1478.7	123.2	684-696	392.3	32.7
36- 48	0.1	0.0	372-384	680.0	56.7	696-708	333.5	27.8
48- 60	0.7	0.1	384-396	440.4	36.7	708-720	344.9	28.7
60- 72	1.4	0.1	396-408	476.9	39.7	720-732	342.4	28.5
72- 84	1.4	0.1	408-420	358.8	29.9	732-744	378.7	31.6
84- 96	1.4	0.1	420-432	395.5	33.0	744-756	339.0	28.2
96-108	1.4	0.1	432-444	425.5	35.5	756-768	354.5	29.5
108-120	2.1	0.2	444-456	353.5	29.5	768-780	428.0	35.7
120-132	3.5	0.3	456-486	350.1	29.2	780-792	384.8	32.1
132-144	8.5	0.7	468-480	404.0	33.7	792-804	378.0	31.5
144-156	18.5	1.5	480-492	345.4	28.8	804-816	478.8	39.9
156-168	25.8	2.1	492-504	345.1	28.8	816-828	672.5	56.0
168-180	34.6	2.9	504-516	400.8	33.4	828-840	585.4	48.8
180-192	59.6	5.0	516-528	375.2	31.3	840-852	520.5	43.4
192-204	751.5	62.6	528-540	347.3	28.9	852-864	350.5	29.2
204-216	669.9	55.8	540-552	399.5	33.3	864-876	474.0	39.5
216-228	728.8	60.7	552-564	341.8	28.5	876-888	348.7	29.1
228-240	860.4	71.7	564-576	341.7	28.5	888-900	616.9	51.4
240-252	855.0	71.3	576-588	399.4	33.3	900-912	676.4	56.4
252-264	708.3	59.0	588-600	341.1	28.4	912-924	642.4	53.5
264-276	718.1	59.8	600-612	340.4	28.4	924-936	679.9	56.7
276-288	1096.9	91.4	612-624	420.1	35.0	936-948	677.3	56.4
288-300	942.3	78.5	624-636	353.9	29.5	948-960	531.7	44.3
300-312	731.8	61.0						
						Total	35,296.4	

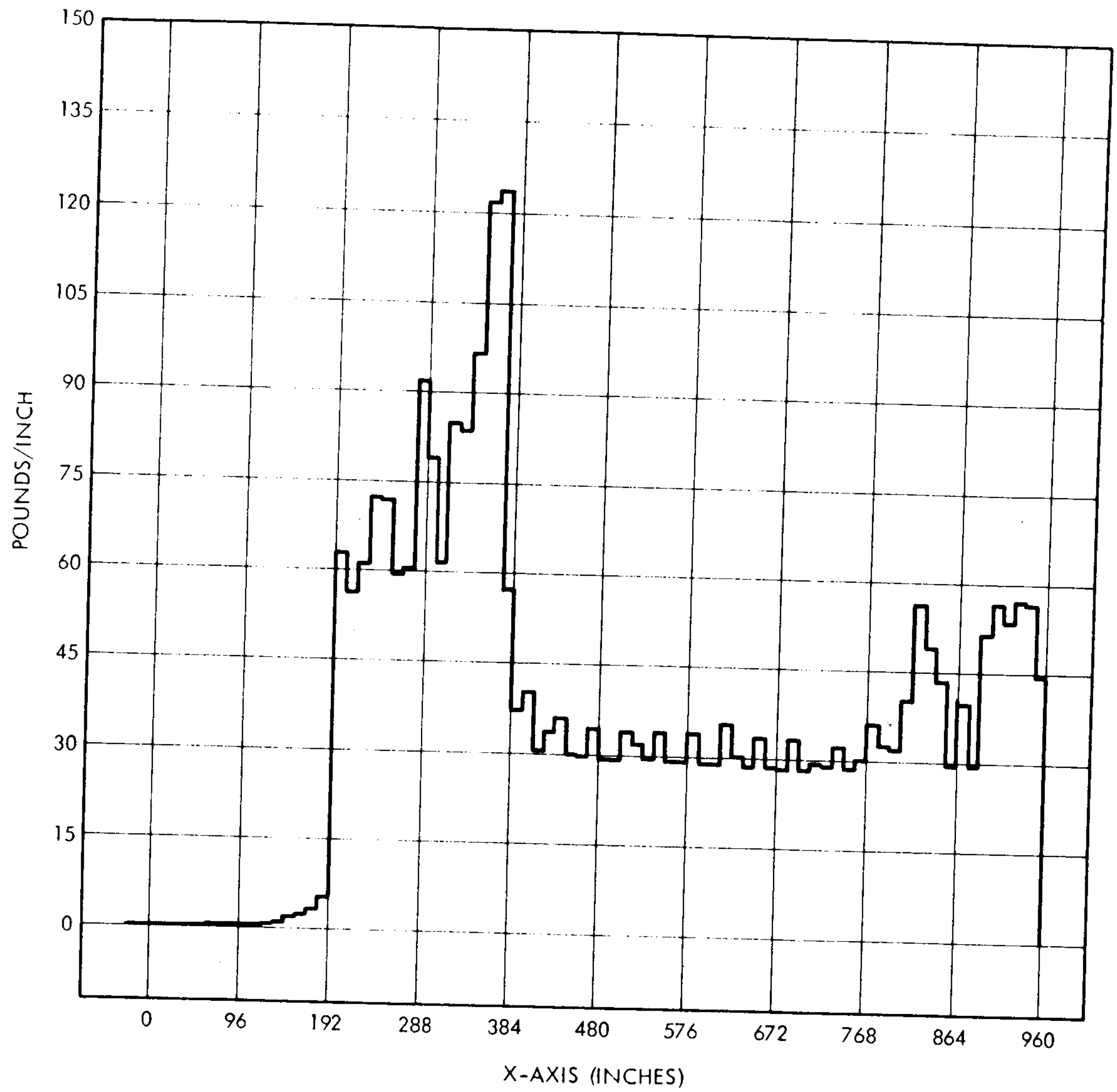


Figure 10. 2-133. S-II Weight Distribution at Engine Cutoff (LOR Mission)

Table 10.2-XXX. S-II Stage at Cutoff Signal
Cantilevered Items LOR Mission

Thrust Structure	
Point of Attachment	S-II Station 223 Saturn-V Station 1787
Weight	38,380 pounds
Center of Gravity	S-II Station 108.1 Saturn-V Station 1672.1
Pitch Moment of Inertia about Point of Attachment	26,366 Kg-M-sec^2 190,705 Slug-Feet^2
LOX Tank	
Point of Attachment	S-II Station 284 Saturn-V Station 1848
Weight	19,723 pounds
Center of Gravity	S-II Station 263.5 Saturn-V Station 1827.5
Pitch Moment of Inertia about Point of Attachment	8211 Kg-M-sec^2 59,391 Slug-Feet^2
Forward Bulkhead	
Point of Attachment	S-II Station 823 Saturn-V Station 2387
Weight	2540 pounds
Center of Gravity	S-II Station 898.3 Saturn-V Station 2462.3
Pitch Moment of Inertia about Point of Attachment	1363 Kg-M-sec^2 9859 Slug-Feet^2

f. Environmental Control System. One equipment container with shock mounts for stage control batteries will be added and one additional container will be shock-mounted. The total increase is 210 pounds.

g. Telemetry and Measuring. A larger 28-vdc battery, and 48 additional measurements will be installed, with a total weight of 200 pounds.

h. Electrical System. A larger 28-vdc battery will be installed, resulting in a 30-pound increase.

The total weight increase compared to the LOR mission to accomplish the LEO mission is 2001 pounds.

10.2.6.2.1 S-II Stage Dry Weight and Longitudinal Centers of Gravity

These data are presented in Table 10.2-XXXI.

10.2.6.2.2 S-II Stage Propellant Weights and Longitudinal Centers of Gravity

These data are presented in Table 10.2-XXXII.

10.2.6.2.3 S-II Interstage Weights and Longitudinal Centers of Gravity

Data for the lower and upper aft interstage structures are presented in Table 10.2-XXXIII.

10.2.6.2.4 S-II Drop Weight History

These data are shown in Table 10.2-XXXIV.

10.2.6.2.5 S-II Stage Mass Characteristics

The vehicle weights, centroids, and moments of inertia are shown in Table 10.2-XXXV in English and International units.

10.2.6.2.6 S-II Stage Weight and Mass Data Versus Flight Time

These data are shown in Figure 10.2-134.

10.2.6.2.7 S-II Weight Distribution and Cantilevered Items at Ground Ignition

Weight Distribution is shown in Table 10.2-XXXVI and Figure 10.2-135. Cantilevered items are shown in Table 10.2-XXXVII.

10.2.6.2.8 S-II Weight Distribution and Cantilevered Items at Engine Cutoff

Weight Distribution is shown in Table 10.2-XXXVIII and Figure 10.2-136. Cantilevered items are shown in Table 10.2-XXXIX.

Table 10.2-XXXI

S-II Stage Dry Weight and Longitudinal Center of Gravity

LEO Mission

Item	Weight (lb)	S-II Body X Station (In)	S-V Body X Station (In)
Structure	(49, 203)	(413.9)	(1977.9)
Propellant Container	29, 329	464.0	2028.0
Forward Skirt	4057	885.3	2449.3
Aft Skirt	4130	241.0	1805.0
Thrust Structure	9302	157.3	1721.3
Fairings and Associated Structure	1391	336.6	1900.6
Bast Heat Protection	660	60.7	1624.7
Paint and Sealer	334	592.3	2156.3
Propulsion System and Accessories	(27, 943)	(126.4)	(1690.4)
Engine and Accessories	19, 248	71.2	1635.2
Purge System	416	266.8	1830.8
Fuel System	3191	379.2	1943.2
Oxidizer System	2519	172.1	1736.1
Stage Control System	1623	121.3	1685.3
Reaction Control System	946	223.3	1787.3
Equipment and Instrumentation	(7085)	(367.3)	(1931.3)
Environmental Control System	1331	338.3	1902.3
Guidance System	481	570.3	2134.3
Telemetry and Measurement Equipment	2934	364.7	1728.2
Propellant Utilization System	634	374.3	1938.3
Electrical System	859	213.7	1777.7
Range Safety Equipment	319	755.2	2319.2
Pneumatic System	410	221.4	1785.4
Separation System	117	468.5	2032.5
Growth Allocation	418	321.3	1885.3
Total S-II Dry Stage Weight	84, 649	314.6	1878.6

Table 10.2-XXXII

S-II Stage Propellant Weight and Longitudinal Center of Gravity, LEO Mission

Item	Weight (lb)	S-II Body X Station (in)	S-V Body X Station (in)
Residuals and Reserve Propellants	(19,470)	(272.0)	(1836.0)
Fuel Pressurizing Gas	1704	650.2	2214.2
Fuel - Residual Allowance	530	356.3	1920.3
Fuel - Main Stage Reserve	1025	365.7	1929.7
Fuel - 2nd Idle Mode Mixture Ratio Shift Allowance	1690	377.9	1941.9
Fuel - Trapped	1441	313.6	1877.6
Oxidizer Pressurizing Gas	4092	283.8	1847.8
Oxidizer - Main Stage Reserve	4735	162.1	1726.1
Oxidizer - 2nd Idle Mode Mixture Ratio Shift Allowance	1830	157.2	1721.2
Oxidizer - Trapped	2353	131.0	1695.0
Service Items	70	115.3	1679.3
Standard Propellant Consumption			
Mainstage	(940,332)	(333.7)	(1897.7)
Fuel	148,113	652.6	2216.6
Oxidizer	792,219	274.0	1838.0
Thrust Decay - Mainstage to Idle	(319)	(245.3)	(1809.3)
Fuel	98	413.)	1977.0
Oxidizer	221	171.0	1735.0
1st Idle Mode	(1184)	(285.7)	(1849.7)
Fuel	565	411.9	1975.9
Oxidizer	619	170.5	1734.5
2nd Idle Mode	(12,650)	(261.7)	(1825.7)
Fuel	5160	398.1	1962.1
Oxidizer	7490	167.7	1731.7
Other Weight Items			
Thrust Buildup	(2550)	(454.3)	(2018.3)
Fuel	725	901.3	2465.3
Oxidizer	1825	365.0	1929.0
Consumed During Coast	(814)	(259.1)	(1823.1)
RCS Propellant	664	225.0	1789.0
Vented Gas - LH_2 Tank	150	410.0	1974.0
Consumed Prior to Ignition			
Service Items	(105)	(190.5)	(1754.4)
Vented Gas - Common Blkhd	38	371.1	1935.1
Engine Start Cartridge Propellant	67	88.0	1652.0

Table 10.2-XXXIII

S-II Interstage Weight and Longitudinal Centers of Gravity, LEO Mission

Item	Weight (lb)	S-II Body X Station (In)	S-V Body X Station (In)
S-IC/S-II Interstage Documented to S-IC Stage Stations 0 to -23			
Structure	(1481)	(-12.3)	(1551.7)
Interstage Structure	1471	-12.3	1551.7
Paint and Sealer	10	-11.5	1552.5
Equipment and Instrumentation	(67)	(- 1.9)	(1562.1)
Separation System	57	- 2.0	1562.0
Guidance System	10	- 1.6	1562.4
TOTAL S-IC/S-II INTERSTAGE DOCUMENTED TO S-IC STAGE DRY WEIGHT	1548	-11.9	1552.1
S-IC/S-II Interstage Documented to S-II Stage Stations 0 to 196.0			
Structure	(8127)	(93.9)	(1657.9)
Interstage Structure	7838	93.6	1657.6
Fairing and Associated Structure	149	107.5	1671.5
Paint and Sealer	140	98.2	1662.2
Propulsion System	(7)	(160.0)	(1724.0)
LOX Fill and Drain Disconnect	7	160.0	1724.0
Equipment and Instrumentation	(202)	(133.9)	(1697.9)
Telemetry and Measuring	60	107.4	1671.4
Separation System	85	169.2	1733.2
Guidance System	57	109.3	1673.3
S-IC/S-II INTERSTAGE DOCUMENTED TO S-II STAGE DRY WEIGHT	8336	94.9	1658.9

Table 10.2-XXXIV

S-II Drop Weight History, LEO Mission

Mission Event Time (sec)	Item Description	Weight (lb)
S-IC Mainstage		
	Bulkhead Purge Gas	38
161 (S-IC OECO)		
	S-IC/S-II Interstage (Small)	1548
	J-2S Start Cartridge Propellant	67
165 (S-II Ignition) (Ignition to 90% PC)	LOX Thrust Build-up	1825
	LH_2 Thrust Build-up	725
Mainstage Burn		
194 (S-II Prior to interstage jettison		
	S-IC/S-II Interstage (large)	8336
Mainstage Burn		
	LOX Thrust Decay	221
	LH_2 Thrust Decay	98
484 (Shift to 1st Idle Mode)		
	LOX Idle Mode	619
	LH_2 Idle Mode	565
497 (S-II at 1st Idle Mode Cutoff)		
Coast/Transfer Orbit to 300 Nautical Mile Altitude		
	Reaction Control Propellant	664
	Vented LH_2 Tank Pressurant	150
3203 (S-II at 2nd Idle Mode Ignition)		
	LOX Idle Mode	7490
	LH_2 Idle Mode	5160
3370 (S-II at 2nd Idle Mode Cutoff)		
	Dry Stage	84, 649
	LOX Remaining in Tank (including sump)	7585
	Unusable 1000	
	Mainstage flight reserve 4735	
	2nd idle mode MRS allow. 1830	

Table 10.2-XXXIV

S-II Drop Weight History, LEO Mission (Cont)

Mission Event Time (sec)	Item Description		Weight (lb)
3370 (S-II at 2nd Idle Mode Cutoff) (Cont)			
	LH_2 Remaining in Tank		4445
	Unusable	1200	
	Mainstage Bias	530	
	Mainstage Flight Reserve	1025	
	2nd Idle Mode MRS Allowance	1690	
	LOX Tank Pressurant		4092
	LH_2 Tank Pressurant		1704
	LOX Trapped (outside of tank)		1353
	LH_2 Trapped (outside of tank)		241
	Service Items		70

Table 10.2-XXXV
S-II Stage Mass Characteristics
(English Units)
LEO Mission

Event	Weight (lb)	Station (in)			I_{xx} (Slug Ft^2)	I_{yy} (Slug Ft^2)	I_{zz} (Slug Ft^2)
		X	Y	Z			
Vehicle Stage Dry	84,649	314.7	6.0	-4.0	458,283	1,549,374	1,549,425
Total S-IC/S-II Interstage Documented to S-IC Stage	1,548	-11.9	1.6	-0.8	13,012	6,647	6,633
Total S-IC/S-II Interstage Documented to S-II Stage	8,336	94.9	3.3	-1.4	69,972	41,344	42,595
S-II Stage at S-II Engine Start Command	1,070,891	328.7	0.5	-0.3	541,546	5,953,660	5,958,594
S-II Stage at Second-Plane Separation - Before	983,448	319.3	0.5	-0.4	541,503	5,280,939	5,285,871
S-II Stage at Second-Plane Separation - After	975,111	321.2	0.5	-0.3	471,524	5,148,035	5,151,854
S-II Stage at LES Jettison	959,741	319.7	0.5	-0.4	471,522	5,040,204	5,044,023
S-II Stage at Mainstage Cutoff	119,606	300.5	4.3	-2.8	470,932	1,730,848	1,734,435
S-II Stage at Begin Coast	118,512	301.2	4.3	-2.8	470,926	1,726,403	1,729,987
S-II Stage at Begin Second Idle Mode	117,289	301.4	4.4	-2.9	464,599	1,724,446	1,724,867
S-II Stage at Final Cutoff	104,639	306.1	4.9	-3.2	464,515	1,668,203	1,668,591

NOTE: S-II Body Station 0 is S-V Station 1564.0

Table 10.2-XXXV
S-II Stage Mass Characteristics (Cont)
(International Units)
LEO Mission

Event	Weight (lb)	Station (in)			I_{xx} Kg-M-sec^2	I_{yy} Kg-M-sec^2	I_{zz} Kg-M-sec^2
		X	Y	Z			
Vehicle Stage Dry Weight	84,649	314.7	6.0	-4.0	63,360	214,209	214,216
Total S-IC/S-II Interstage Documented to S-IC Stage	1,548	-11.9	1.6	-0.8	1,799	919	917
Total S-IC/S-II Interstage Documented to S-II Stage	8,336	94.9	3.3	-1.4	9,674	5,716	5,889
S-II Stage at S-II Engine Start Command	1,070,891	328.7	0.5	-0.3	74,877	823,184	823,859
S-II Stage at Second-Plane Separation - Before	983,448	319.3	0.5	-0.4	74,871	730,171	730,850
S-II Stage at Second-Plane Separation - After	957,111	321.2	0.5	-0.3	65,195	711,795	712,322
S-II Stage at LES Jettison	959,741	319.7	0.5	-0.4	65,195	696,885	697,408
S-II Stage at Mainstage Cutoff	119,606	300.5	4.3	-2.8	65,113	239,316	239,811
S-II Stage at Begin Coast	118,512	301.2	4.3	-2.8	65,113	238,702	239,274
S-II Stage at Begin Second Idle Mode	117,289	301.4	4.4	-2.9	64,238	238,431	239,003
S-II Stage at Final Cutoff	104,639	306.1	4.9	-3.2	64,226	230,654	231,208

NOTE: S-II Body Station 0 is S-V Station 1564.0

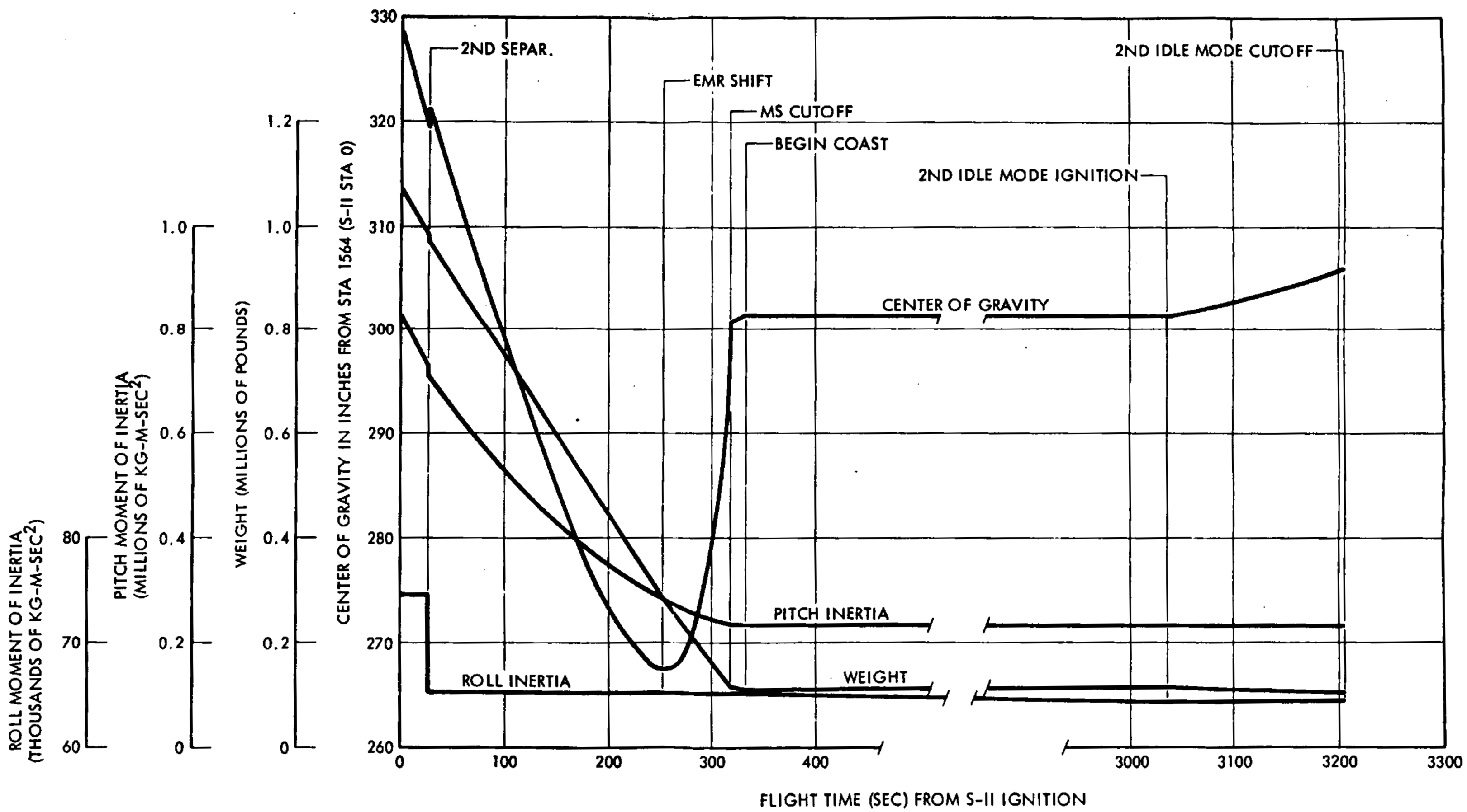


Figure 10.2-134. S-II Stage Weight and Mass Data Versus Flight Time (LEO Mission)

Table 10.2-XXXVI
S-II Weight Distribution at Ground Ignition
LEO Mission (Pounds)

X-Axis	Weight	Weight/Inch	X-Axis	Weight	Weight/Inch	X-Axis	Weight	Weight/Inch
-23--12	799.1	72.6	312-324	1000.4	83.4	636-648	4065.1	338.8
-12- 0	810.9	67.6	324-336	992.7	82.7	648-660	4118.9	343.2
0- 12	882.6	73.5	336-348	1153.6	96.1	660-672	4056.5	338.0
12- 24	399.1	33.3	348-360	1455.9	121.3	672-684	4052.0	337.7
24- 36	469.7	39.1	360-372	1478.2	123.2	684-696	4105.6	342.1
36- 48	564.1	47.0	372-384	2609.8	217.5	696-708	4043.1	336.9
48- 60	425.7	35.5	384-396	3001.0	250.1	708-720	4050.7	337.6
60- 72	426.9	35.6	396-408	3201.9	266.8	720-732	4044.4	337.0
72- 84	627.9	52.3	408-420	3248.4	270.7	732-744	4077.0	339.7
84- 96	437.5	36.5	420-432	3449.7	287.5	744-756	4033.5	336.1
96-108	422.9	35.2	432-444	3644.1	303.7	756-768	4045.3	337.1
108-120	597.4	49.8	444-456	3736.7	311.4	768-780	4115.0	342.9
120-132	374.3	31.2	456-468	3897.7	324.8	780-792	4068.1	339.0
132-144	380.5	31.7	468-480	4116.0	343.0	792-804	4057.5	338.1
144-156	382.9	31.9	480-492	4182.9	348.6	804-816	4154.6	346.2
156-168	620.2	51.7	492-504	4190.5	349.2	816-828	4344.5	362.0
168-180	397.1	33.1	504-516	4246.2	353.8	828-840	4253.7	354.5
180-192	486.5	40.5	516-528	4220.6	351.7	840-852	4401.5	366.8
192-204	1268.8	105.7	528-540	4270.3	355.9	852-864	4466.5	372.2
204-216	954.8	79.6	540-552	4127.9	344.0	864-876	4006.0	333.8
216-228	1249.9	104.2	552-564	4070.2	339.2	876-888	2841.3	236.8
228-240	1265.8	105.5	564-576	4070.2	339.2	888-900	2599.2	216.6
240-252	984.9	82.1	576-588	4127.8	344.0	900-912	676.4	56.4
252-264	708.8	59.1	588-600	4069.6	339.1	912-924	642.4	53.5
264-276	686.4	57.2	600-612	4068.9	339.1	924-936	679.9	56.7
276-288	1075.3	89.6	612-624	4148.6	345.7	936-948	677.3	56.4
288-300	926.2	77.2	624-636	4082.4	340.2	948-960	531.7	44.3
300-312	717.3	59.8						
						Total	201,413.3	

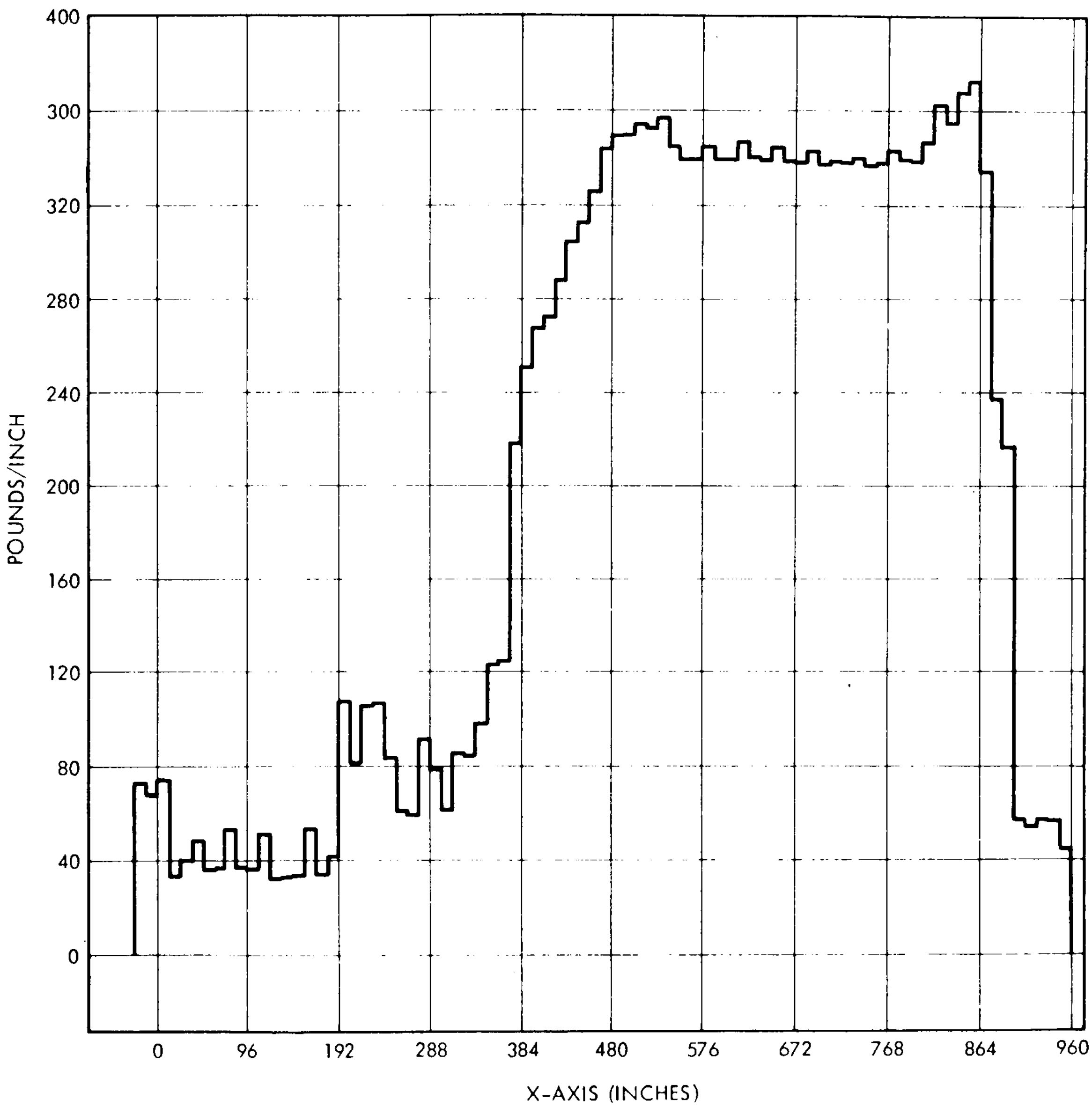


Figure 10. 2-135. S-II Weight Distribution at Ground Ignition (LEO Mission)

Table 10.2-XXXVII
S-II Stage at Liftoff, Cantilevered Items
LEO Mission

Thrust Structure	
Point of Attachment	S-II Station 223 Saturn-V Station 1787
Weight	38300 pounds
Center of Gravity	S-II Station 109.2 Saturn-V 1673.2
Pitch Moment of Inertia about Point of Attachment	26735 Kg-M-sec^2 193374 Slug-Ft^2
LOX Tank	
Point of Attachment	S-II Station 284 Saturn -V Station 1848
Weight	835456 pounds
Center of Gravity	S-II Station 273.1 Saturn-V 1837.1
Pitch Moment of Inertia about Point of Attachment	267049 Kg-M-sec^2 1931565 Slug-Ft^2
Forward Bulkhead	
Point of Attachment	S-II Station 823 Saturn-V Station 2387
Weight	2540 pounds
Center of Gravity	S-II Station 898.3 Saturn-V Station 2462.2
Pitch Moment of Inertia about Point of Attachment	1423 Kg-M-sec^2 10292 Slug-Ft^2

Table 10.2-XXXVIII
S-II Weight Distribution at Cutoff,
LEO Mission (pounds)

X-Axis	Weight	Weight/Inch	X-Axis	Weight	Weight/Inch	X-Axis	Weight	Weight/Inch
-23--12	......	-0.0	312-324	1000.4	83.4	636-648	337.4	28.1
-12- 0	......	-0.0	324-336	992.7	82.7	648-660	394.6	32.9
0- 12	......	-0.0	336-348	1153.6	96.1	660-672	335.9	28.0
12- 24	......	-0.0	348-360	1455.9	121.3	672-684	335.2	27.9
24- 36	......	-0.0	360-372	1478.2	123.2	684-696	392.5	32.7
36- 48	0.2	0.0	372-384	679.6	56.6	696-708	333.7	27.8
48- 60	0.8	0.1	384-396	440.2	36.7	708-720	345.0	28.8
60- 72	1.6	0.1	396-408	476.7	39.7	720-732	342.5	28.5
72- 84	1.6	0.1	408-420	358.8	29.9	732-744	378.9	31.6
84- 96	1.6	0.1	420-432	395.7	33.0	744-756	339.1	28.3
96-108	1.6	0.1	432-444	425.6	35.5	756-768	354.6	29.6
108-120	1.6	0.1	444-456	353.7	29.5	768-780	428.1	35.7
120-132	1.6	0.1	456-468	350.3	29.2	780-792	385.0	32.1
132-144	1.7	0.1	468-480	404.1	33.7	792-804	378.2	31.5
144-156	2.5	0.2	480-492	345.6	28.8	804-816	478.9	39.9
156-168	3.7	0.3	492-504	345.3	28.8	816-828	672.7	56.1
168-180	5.4	0.5	504-516	400.9	33.4	828-840	585.6	48.8
180-192	16.5	1.4	516-528	375.3	31.3	840-852	520.6	43.4
192-204	758.6	63.2	528-540	347.5	29.0	852-864	350.5	29.2
204-216	817.5	68.1	540-552	399.7	33.3	864-876	474.0	39.5
216-228	1112.5	92.7	552-564	342.0	28.5	876-888	348.7	29.1
228-240	1128.4	94.0	564-576	341.9	28.5	888-900	616.9	51.4
240-252	847.5	70.6	576-588	399.6	33.3	900-912	676.4	56.4
252-264	685.9	57.2	588-600	341.3	28.4	912-924	642.4	53.5
264-276	686.4	57.2	600-612	340.6	28.4	924-936	679.9	56.7
276-288	1075.3	89.6	612-624	420.3	35.0	936-948	677.3	56.4
288-300	926.2	77.2	624-636	35411	29.5	948-960	531.7	44.3
300-312	717.3	59.8						
						Total 35851.9		

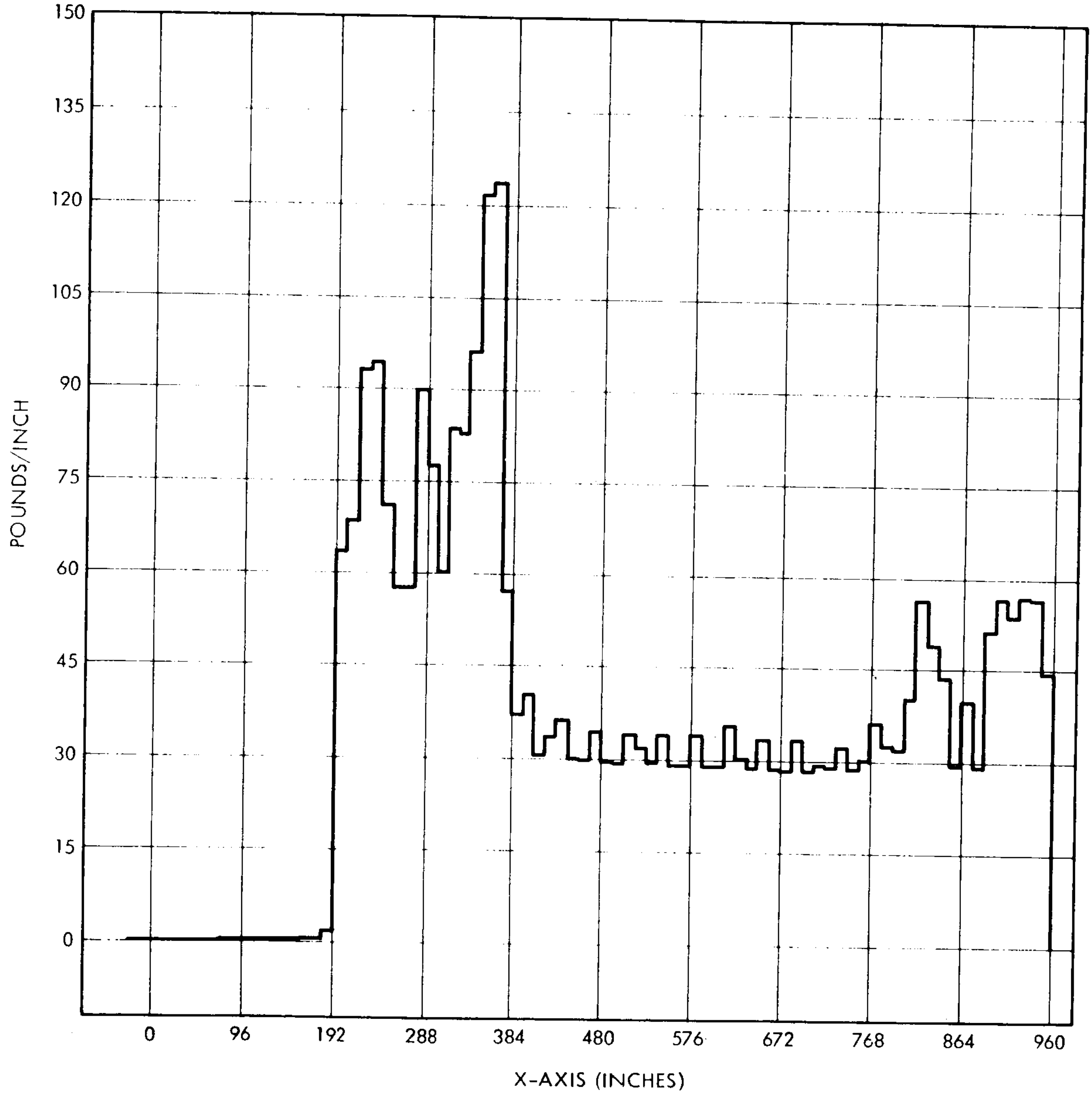


Figure 10.2-136. S-II Weight Distribution at Engine Cutoff (LEO Mission)

Table 10.2-XXXIX
S-II Stage at Cutoff Signal, Cantilevered Items
LEO Mission

Thrust Structure	
Point of Attachment	S-II Station 223 Saturn-V Station 1787
Weight	38233 pounds
Center of Gravity	S-II Station 109.3 Saturn-V Station 1673.3
Pitch Moment of Inertia about Point of Attachment	26209 Kg-M-sec^2 189566 Slug-Ft^2
LOX Tank	
Point of Attachment	S-II Station 284 Saturn-V Station 1848
Weight	21133 pounds
Center of Gravity	S-II Station 254.0 Saturn-V Station 1818.0
Pitch Moment of Inertia about Point of Attachment	9295 Kg-M-sec^2 67232 Slug-Ft^2
Forward Bulkhead	
Point of Attachment	S-II Station 823 Saturn-V Station 2387
Weight	2540 pounds
Center of Gravity	S-II Station 898.3 Saturn-V Station 2462.3
Pitch Moment of Inertia about Point of Attachment	1423 Kg-M-sec^2 10292 Slug-Ft^2

10.2.7 GROUND SUPPORT EQUIPMENT

This section describes the changes required to the mechanical ground support equipment (MGSE) and the electrical support equipment (ESE) for J-2S engine incorporation on the LOR and LEO missions. The changes for the LEO mission are described as additions to those required for the basic LOR mission.

The baseline configuration for GSE is described in the Configuration Control Master Record (CCMR), Parts 1 and 2, dated 13 November 1968.

The equipment and GSE for servicing and checkout of the J-2S engine as a component, as well as for the solid propellant turbine starter (SPTS), is assumed to be provided as GFE, or by the engine contractor as directed by NASA.

10.2.7.1 Mechanical Ground Support Equipment (MGSE) Changes

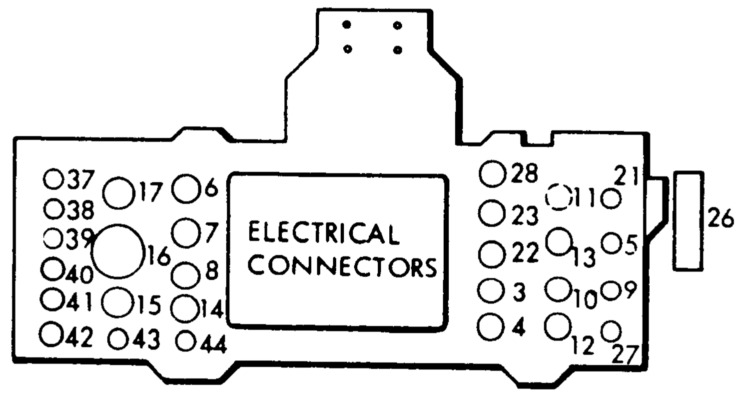
The basic philosophy for modifying the MGSE required to check out, service, and launch a J-2S-powered S-II will be to deactivate, modify existing systems, and add new hardware only when deactivated hardware cannot be used. No changes will be made solely to up-grade existing systems. Removal of deactivated systems will be accomplished in the detail design phase only if a degradation in system reliability can be demonstrated, or to provide space for added systems.

10.2.7.1.1 Auxiliary MGSE Changes for the LOR and PO Missions

a. A7-41, Umbilical Carrier Plate, S-II Intermediate Swing Arm. The carrier plate provides the structural termination for the individual fluid and electrical system connectors. It contains the latching and push-off systems that provide a near simultaneous separation of the individual disconnects shortly after vehicle liftoff.

1. Design Changes. The disconnects at the following locations are deleted and the opening capped with a metal plate to maintain the purge gas atmosphere between carrier plate and stage (Figure 10.2-137).

Disconnect No.	Find No.	J-2 Nomenclature
7	A7741	Turbine Start Bottle GH2 Vent Control Pressure
10	A7753	Turbine Start Bottle Pressurization
11	A7752	LH_2 Pump Seal Drain
17	A7755	Prevalves and Recirculation System
		Actuation System Pressure



LOR MISSION

DICONNECT NO.	FIND NO.	NOMENCLATURE	SIZE (IN.)	TYPE
6	A7742	LOX DOME PURGE	1/2	MC 144-0010
7	A7741	DELETED		
10	A7753	DELETED		
11	A7752	DELETED		
12	A7754	FUEL PUMP DRAIN	1	MC 144-0011
17	A7755	DELETED		
22	A7736	PRE VALVE ACTUATION SYSTEM PRESS	1/2	MC 144-0010
27	A7750	LH_2 FEED LINE PURGE	1	MC 144-0011
LEO MISSION				
10	A7753	RCS PRESSURIZATION	1/2	MC 144-0010
17	A7755	EAS AIR SUPPLY	1/2	MC 144-0010

Figure 10.2-137. A7-41 Umbilical Plate Engine Systems Changes

The following disconnects are redesignated as follows:

Disconnect No.	Find No.	J-2 Nomenclature
6	A7742	LOX Dome Purge was Turbopump Purge
12	A7754	Fuel Pump Drain was Turbine Start Bottle Vent and Relief
22	A7736	Prevalve Actuation system Pressure was Inflight Helium Injection
27	A7750	LH_2 Feedline Purge was Recirculation LH_2 Line Purge

2. Qualification and Test. Qualification of the carrier plate will not be affected by the change. No testing will be required to revalidate the carrier system above that currently performed at KSC for the J-2/S-II program.

3. Modification. Hardware modifications can be accomplished in the field or at the umbilical refurbishment facility (currently NR/LAD) after support of the last J-2/S-II commitment.

4. Contractual documents affected are the following:

   (a) Saturn Stage S-II Ground Support Equipment, Contract Specification CP 190 M0001A

   (b) 13M50097, Saturn Interface Control Document, Saturn V Vehicle S-II Stage Fluids Requirements

   (c) 65ICD9746, Umbilical and Service Connections, S-II (Intermediate)

b. A7-51, Side Load Arresting Mechanism (SLAM). The SLAM provides a structural tie, attaching to the engine bridles, to react side loads during ground engine start. The outboard engine attachments can be separated on command to permit engine gimbaling. The center engine attachment is fixed. The outboard engine bridle is provided as GFE with the engine, whereas the center engine bridle is provided by NR/SD. The outboard bridle attachment points to the SLAM have been designed by Rocketdyne to interface with existing equipment.

1. Design Change. The change in the J-2S side load attachment points will require modification of the center engine bridle collar struts. These changes, shown in Figure 10.2-138, consist of shortening the vertical struts, a redesign of the center strut attachment fitting, and a redesign of the turnbuckle barrel to develop a shorter length adjustment.

2. Retest. The redesigned and modified struts will be individually proof-tested to limit design load. No structural testing of the entire SLAM system is required.

3. Modification. Modified components of the center engine SLAM system can be accomplished in the field. No contractual documents are affected.

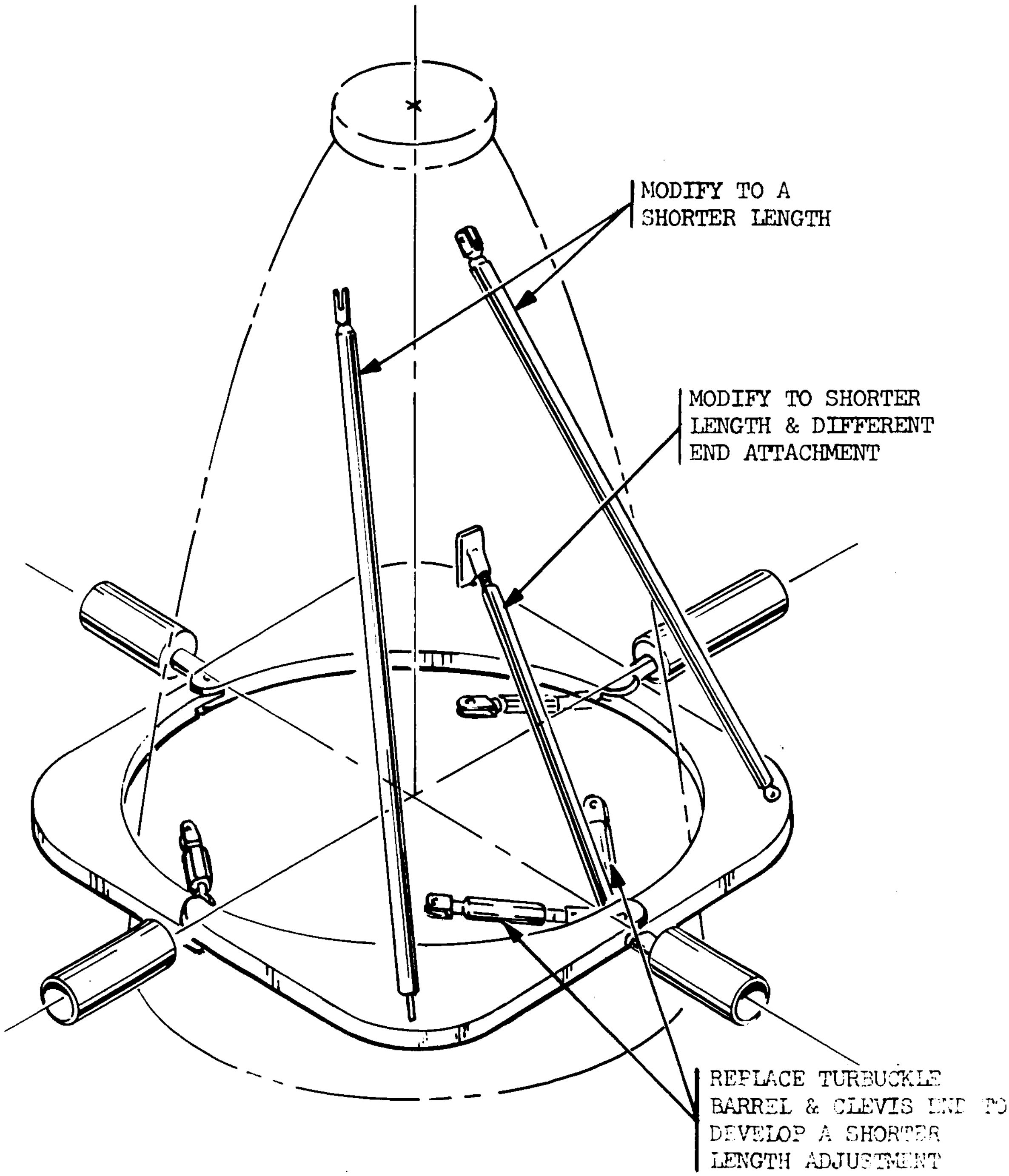


Figure 10.2-138. A7-51 SLAM Center Engine Bridle Changes for J-2S Engine

c. A7-61 Umbilical Carrier Plate, Aft - Static Firing. The A7-61 provides the termination support structure for the hard fluid lines (used as a substitute for the disconnects during static firing) and electrical systems servicing the stage. The carrier plate is fastened securely to the stage to withstand the static firing environment

1. Design Changes. The changes to the A7-61 will be similar to those described for the A7-41 except that physical change will consist of removal of support bracketry on deleted systems and nomenclature changes, as appropriate. No retesting is required nor are contractual documents affected.

d. A7-71, LH_2 Heat Exchanger. The A7-71 heat exchanger is essentially a double-walled vacuum-jacketed vessel encompassing three chill coils in a liquid and gaseous bath of hydrogen. The coils chill helium and hydrogen gas for use with the S-II stage.

1. Design Change. The start tank chill and thrust chamber chill circuits will be deactivated by capping openings at Find Numbers A9247, A9241, A9248, and A9242 (Figure 10.2-139).

2. Retest. Retesting will consist of verifying total system functional compatibility in conjunction with tanking tests at MTF and CDDT at KSC.

3. Modification. All modification is to be accomplished in the field.

4. Affected Contractual Documents:

Contract Specification CP362M0001A

65ICD9765, Physical ICD S-II Heat Exchanger

65ICD9766, Piping Criteria - S-II Pneumatic Servicing, Saturn V

e. A7-84 Engine Compartment Platform Set. The A7-84 consists of three distinct platform assemblies: two circumferential platform sets attaching to the aft interstage at S-II Stations 30 and 114, and a set of protective platform panels fitting over the heat shield. The platforms provide personnel access to the engine compartment.

1. Design Changes. Only the heat shield platform will be affected due to a change in the heat shield support configuration. The platform panels will be modified to clear the revised strut locations. A fit test at KSC low bay will be required. Modification can be accomplished in the field. No contractual documents are affected.

10.2.7.1.2 Auxiliary MGSE Changes for the LEO Mission.

These changes are delta to those required for the LOR mission.

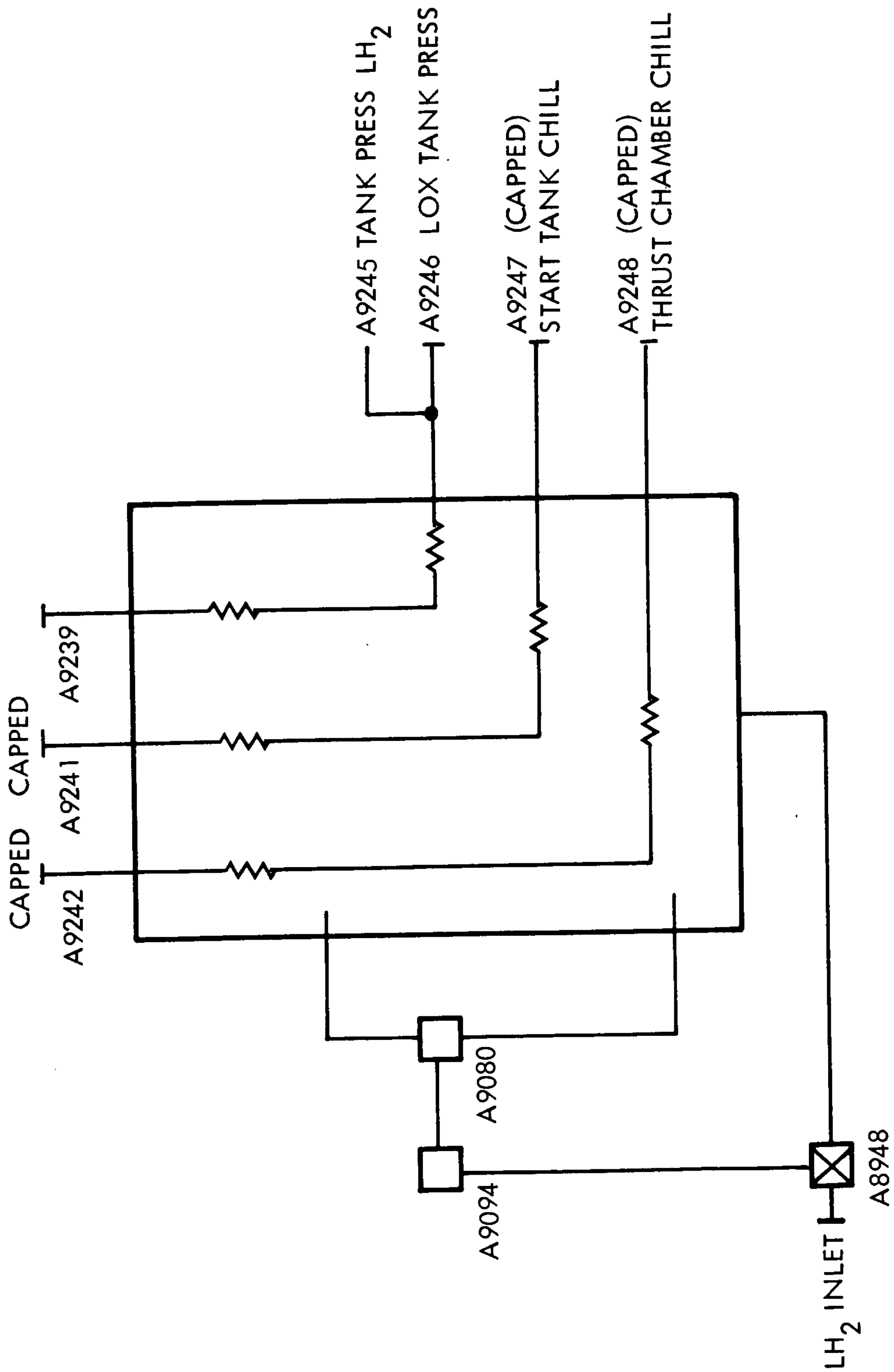


Figure 10.2-139. J-2S A7-71 Heat Exchanger Schematic Diagram

a. A7-41, Umbilical Carrier Plate, S-II Intermediate Swing Arm.

1. Design Changes. Add connects at the following locations (Figure 10.2-137):

Disconnect No.	Find No.	Nomenclature
10	A7753	RCS Pressurization
17	A7755	EAS air supply

2. Qualification and Test. Testing is not affected by the test.

3. Modification. Modification will be accomplished in the field.

4. Contractual Documents Affected.

Contract Specification CP190M001A

Saturn ICD 13M50097

Saturn ICD 65ICD9746

b. A7-61 Umbilical Carrier Plate, Aft - Static Firing

1. Design Changes. Hardline support bracketry and designation will be added for the EAS Air Supply system at Find No. A7755. Modification is to be accomplished in the field; no retest is required nor are contractual documents affected.

10.2.7.1.3 Checkout MSGE Changes for the LOR and PO Missions

a. C7-53, Pneumatic Checkout Blanking Plate Set. The C7-53 contains the plates and fittings required to block off subsystems for leak and functional tests (Figure 10.2-140).

1. Design Changes. Additional blanking plates are to be provided for the oxidizer dome purge system leak check at the disconnect (KSC only). The existing turbopump purge blanking plate will be utilized at the engine connect panel. The oxidizer dome purge plate will also satisfy prevalve actuation system requirements. There is to be no retest, modification can be accomplished in the field, and no contractual documents are affected.

b. C7-70 S-II Pneumatic Console Test Set. The C7-70 is a portable item of GSE containing the controls and monitoring devices to operate and checkout individual electrical and electro-mechanical components within the S7-41.

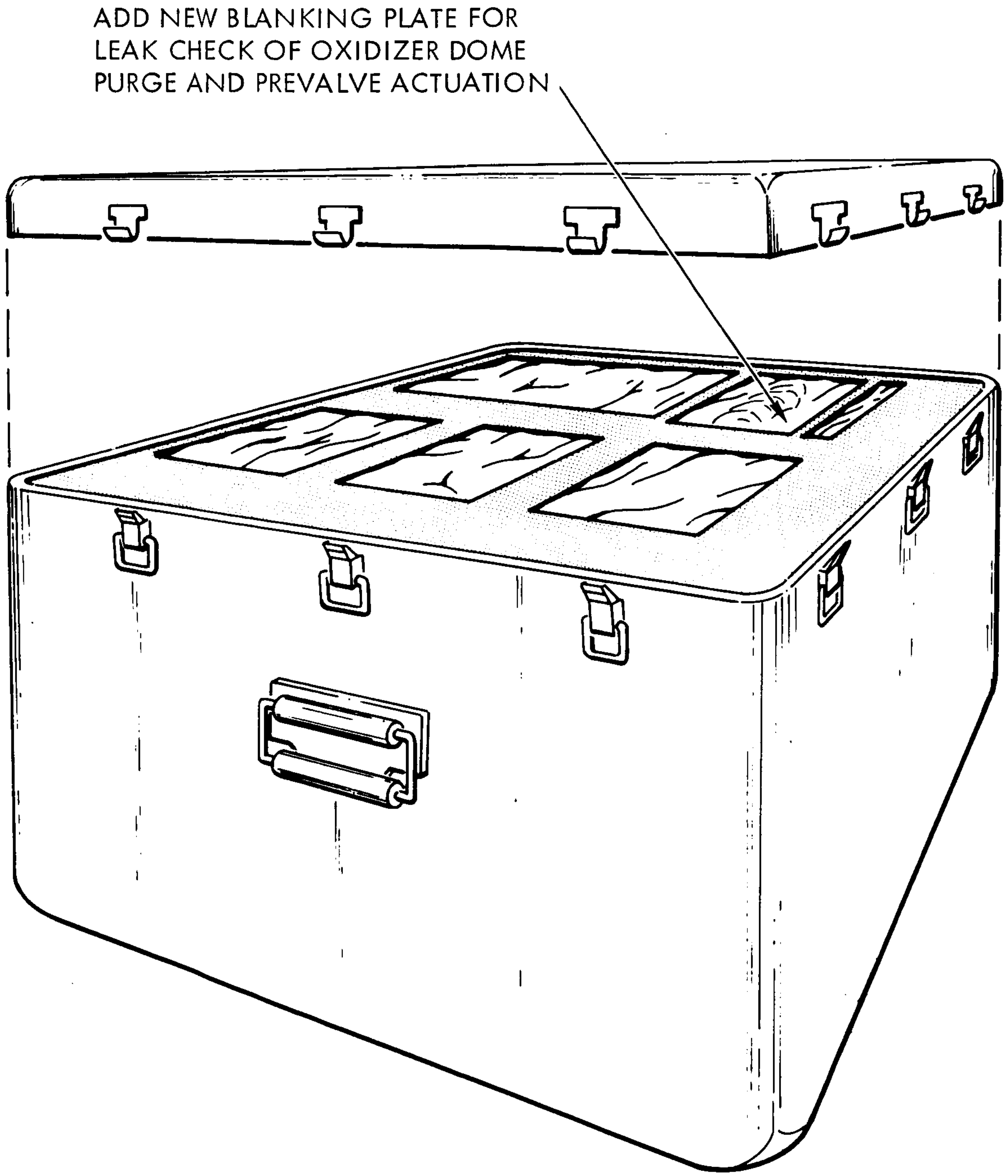


Figure 10.2-140. C7-53 Pneumatic Checkout Blanking Plate Set

1. Design Changes. Control and monitoring circuitry are to be added for the LOX dome purge pressure switch and solenoid valve. The nomenclature of recirculation system pressure will be changed to prevalve actuation pressure, and LH_2 recirculation system purge will be changed to LH_2 feedline purge. Added control system integrity will be verified, and modification can be accomplished in the field. The contractual document affected is Contract Specification CP490M0003A.

c. C7-603 Console, Pneumatic. The C7-603 provides the pneumatic stimuli to check out the various stage systems. It provides pressure levels to proof-pressure test and leak test both components and subsystems. The C7-603 is used at the manufacturing and static firing facilities only.

1. Design Changes. The 400-psig GH_2 start tank vent control supply system will be inactivated by capping fluid junction FJ109 and removing nomenclature from the rear panel and from the control switch on the test control panel (Figures 10.2-141 through 10.2-144).

The 750-psig recirculation System Helium Fill Disconnect Actuation Supply system will be inactivated by capping fluid junction FJ147 and removing nomenclature from rear panel and from the control switch on the test control panel (Figures 10.2-142 and 10.2-144).

The 1000 psig recirculation pressurization receiver requirements are deleted for the J2-S, and an actuation supply, and leak check pressurization of 725 ± 25 psig for the LOX and LH_2 prevalves is added. Therefore, the deleted system will be used to provide the prevalves 725 ± 25 psig actuation supply and leak check pressure. This is to be accomplished by changing pressure regulator G9PR7, pressure switch G9PS8, relief valve G9RV6, and revising nomenclature on the rear panel and control switch on the test control panel (Figures 10.2-143 and 10.2-144).

The 0-20 psig GH_2 start tank fill manifold disconnect relief check system will be inactivated by capping fluid junction FJ170, removing solenoid actuation control and removing nomenclature from front and rear leak check regulation panels (Figures 10.2-145 and 10.2-147).

The 30 psig LH_2 pump seal drain manifold supply system will be inactivated by capping fluid junction FJ135 and removing solenoid actuation control and nomenclature from front and rear leak check regulation panel (Figures 10.2-145 and 10.2-147).

The 0-20 psig LH_2 pump seal drain manifold disconnect relief check supply system will be inactivated by capping fluid junction FJ174, removing solenoid actuation control and nomenclature from front and rear leak check regulation panel (Figures 10.2-145 and 10.2-147).

The 30 psig LH_2 recirculation system purge supply system will be inactivated by capping fluid junction FJ194 and removing solenoid actuation control and nomenclature from front and rear leak check regulation panel (Figure 10.2-145 and 10.2-147).

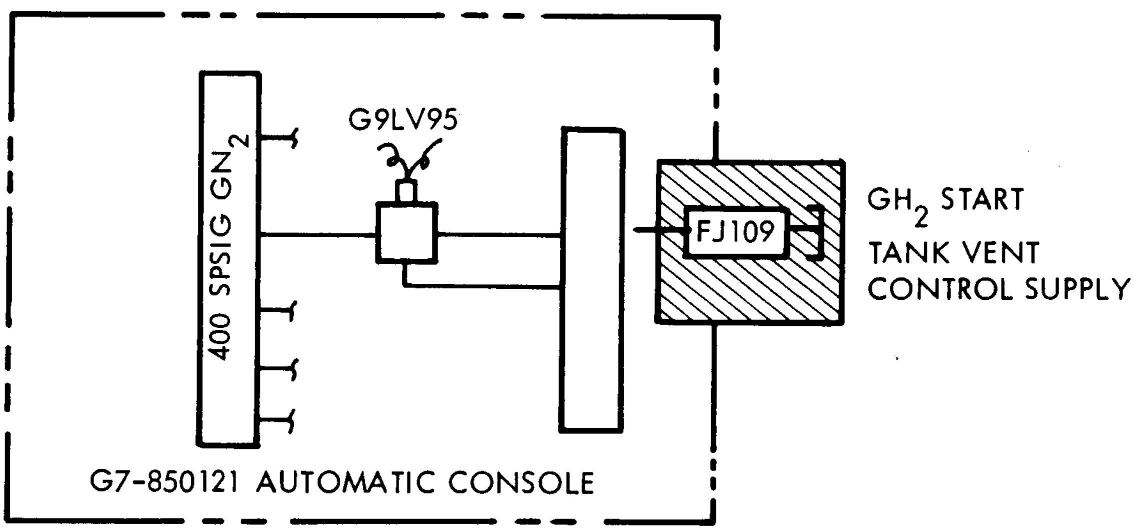


Figure 10.2-141. C7-603 System Changes (GH_2 Start Tank Vent Control Supply)

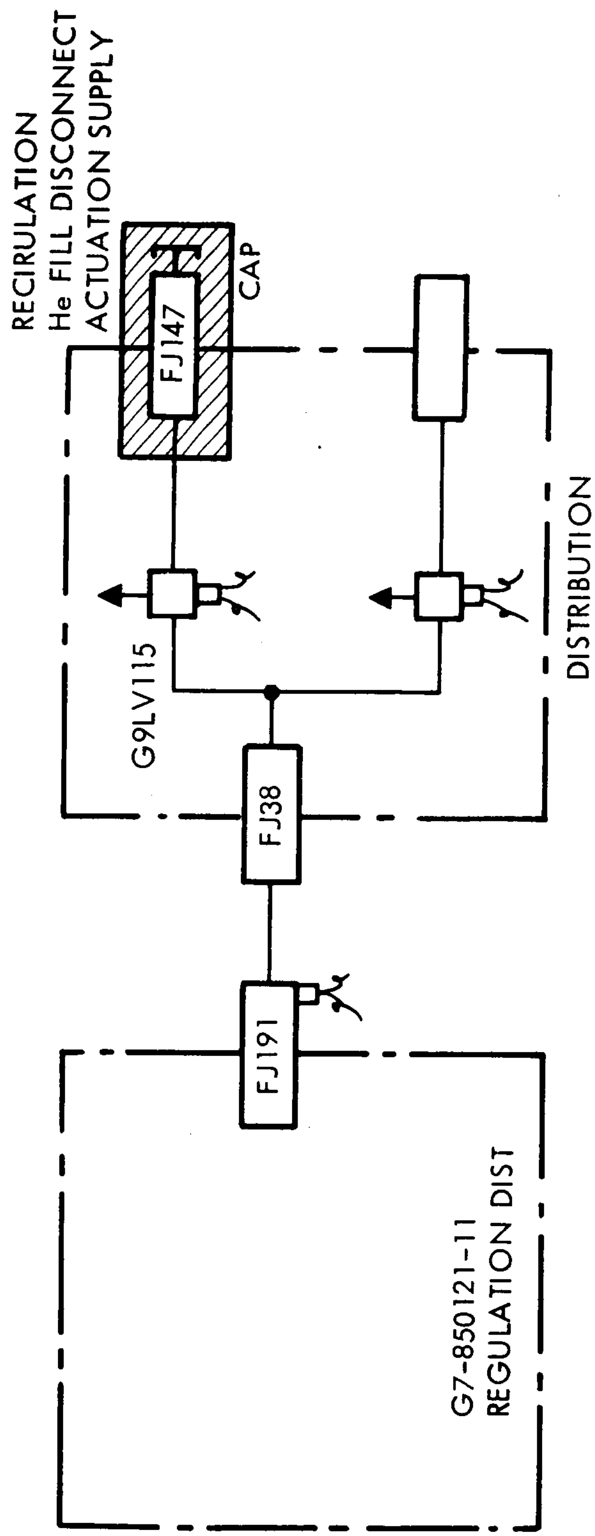


Figure 10.2-142. C7-603 System Changes (Recirculation System Helium Fill Disconnect)

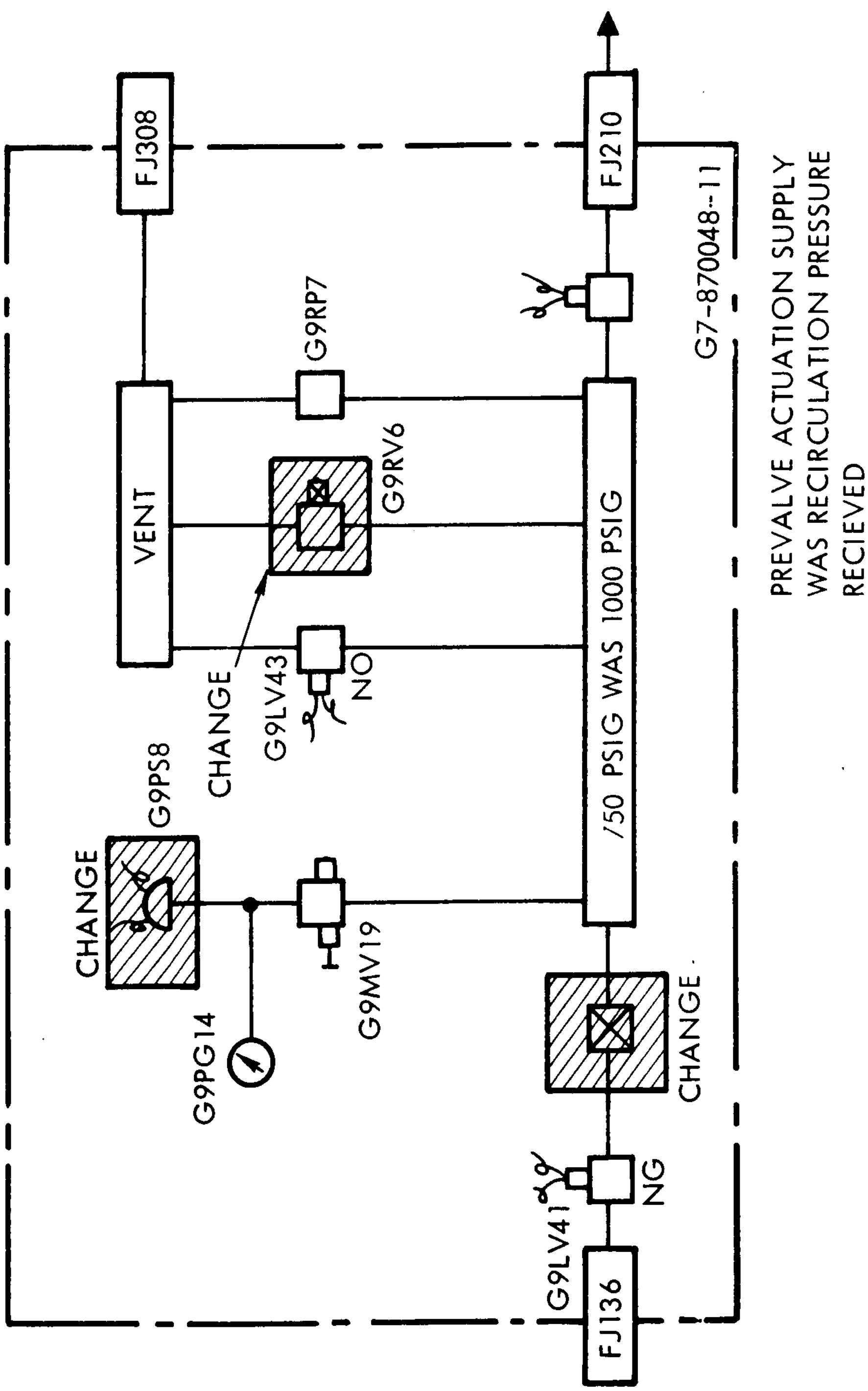


Figure 10.2-143. C7-603 System Changes (Recirculation Pressurization Receiver)

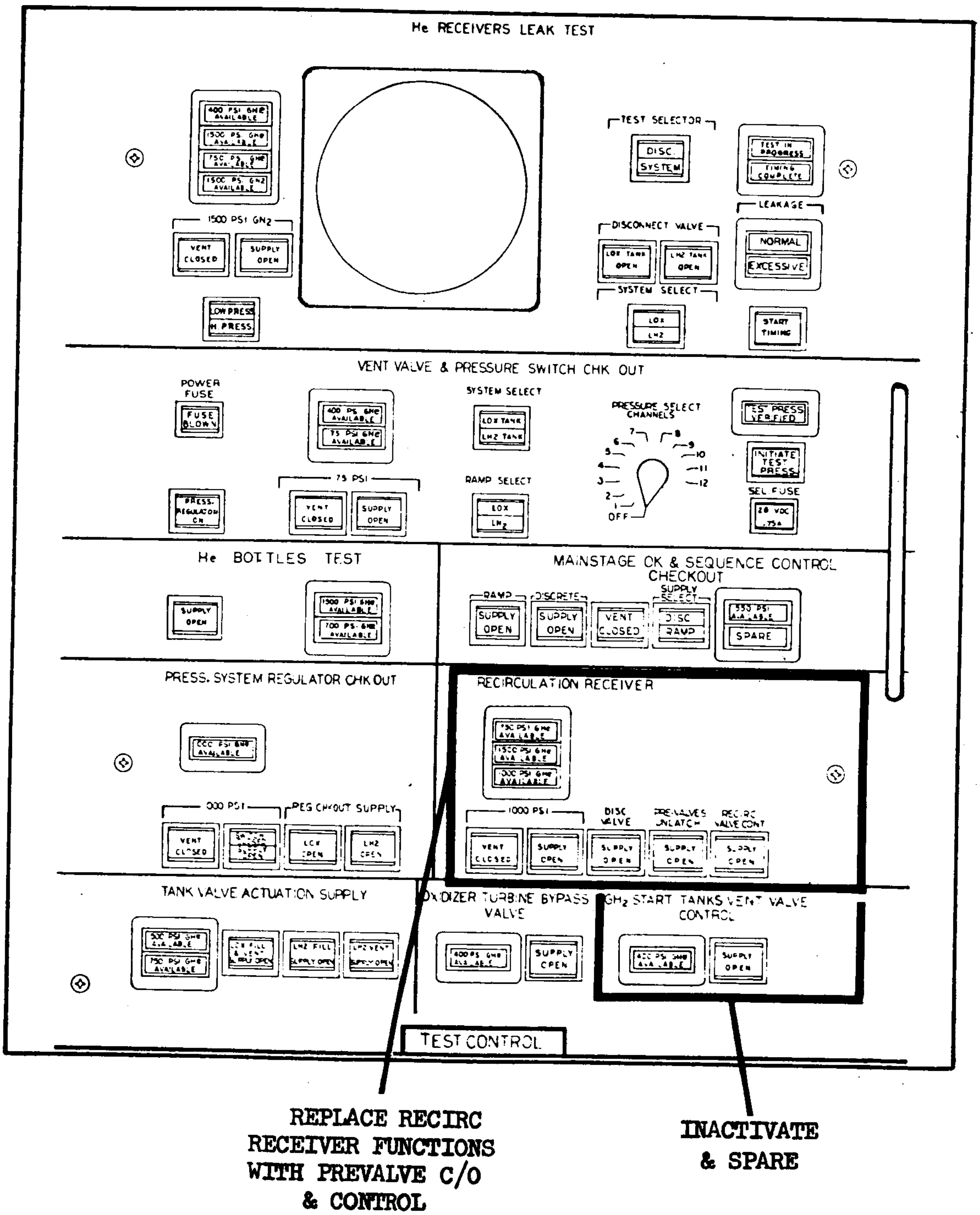


Figure 10.2-144. C7-603 Test Control Panel Changes

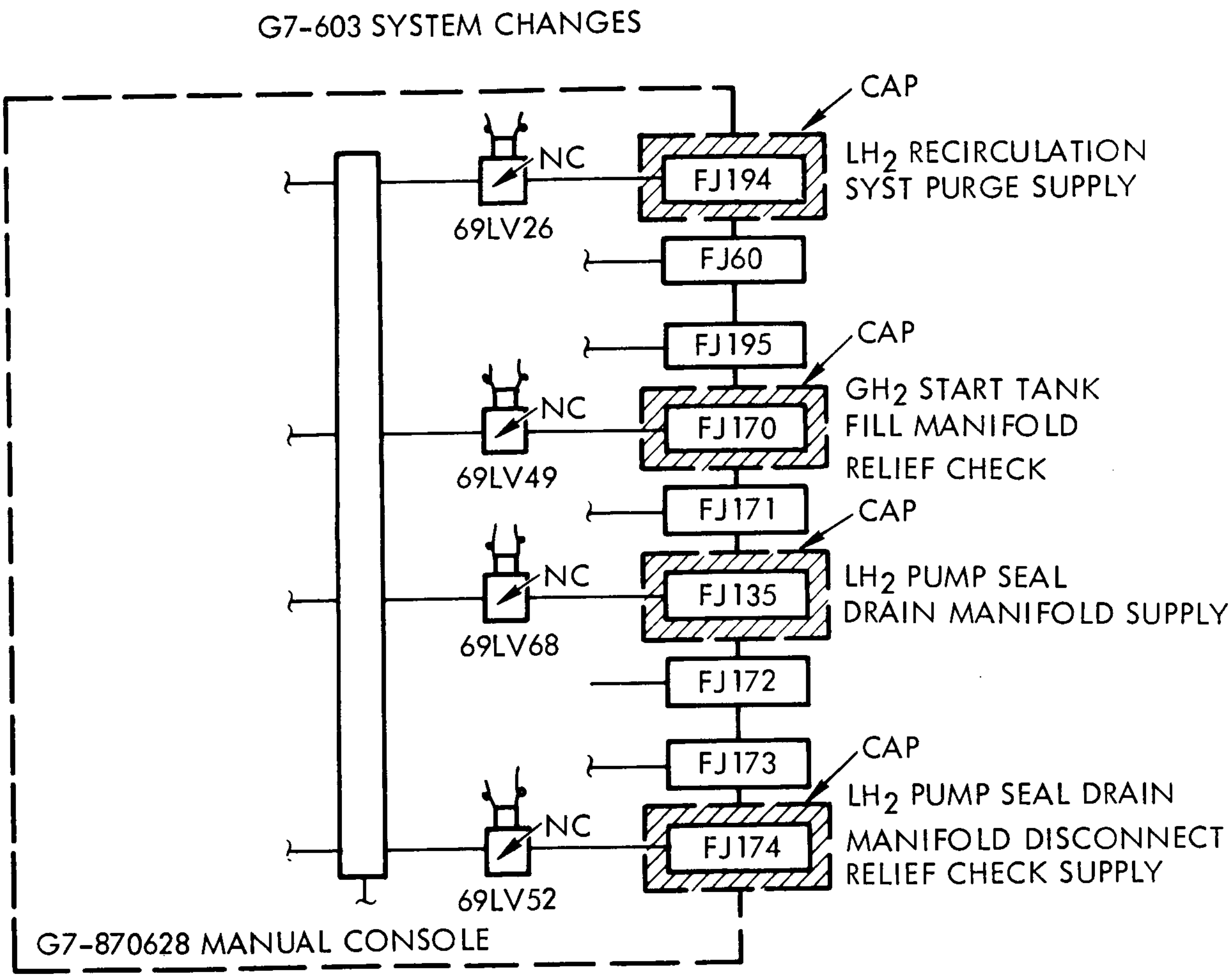


Figure 10.2-145. C7-603 System Changes (LH_2 Pump Seal Drain Manifold Disconnect Relief Check Supply)

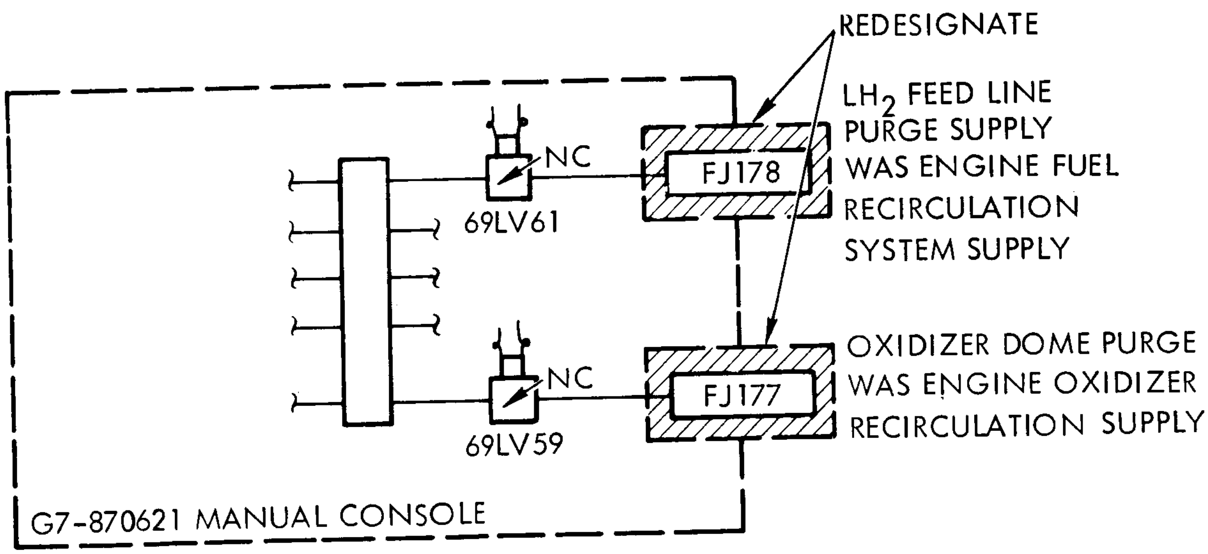


Figure 10.2-146. C7-603 System Changes (LH_2 Feedline Purge Supply)

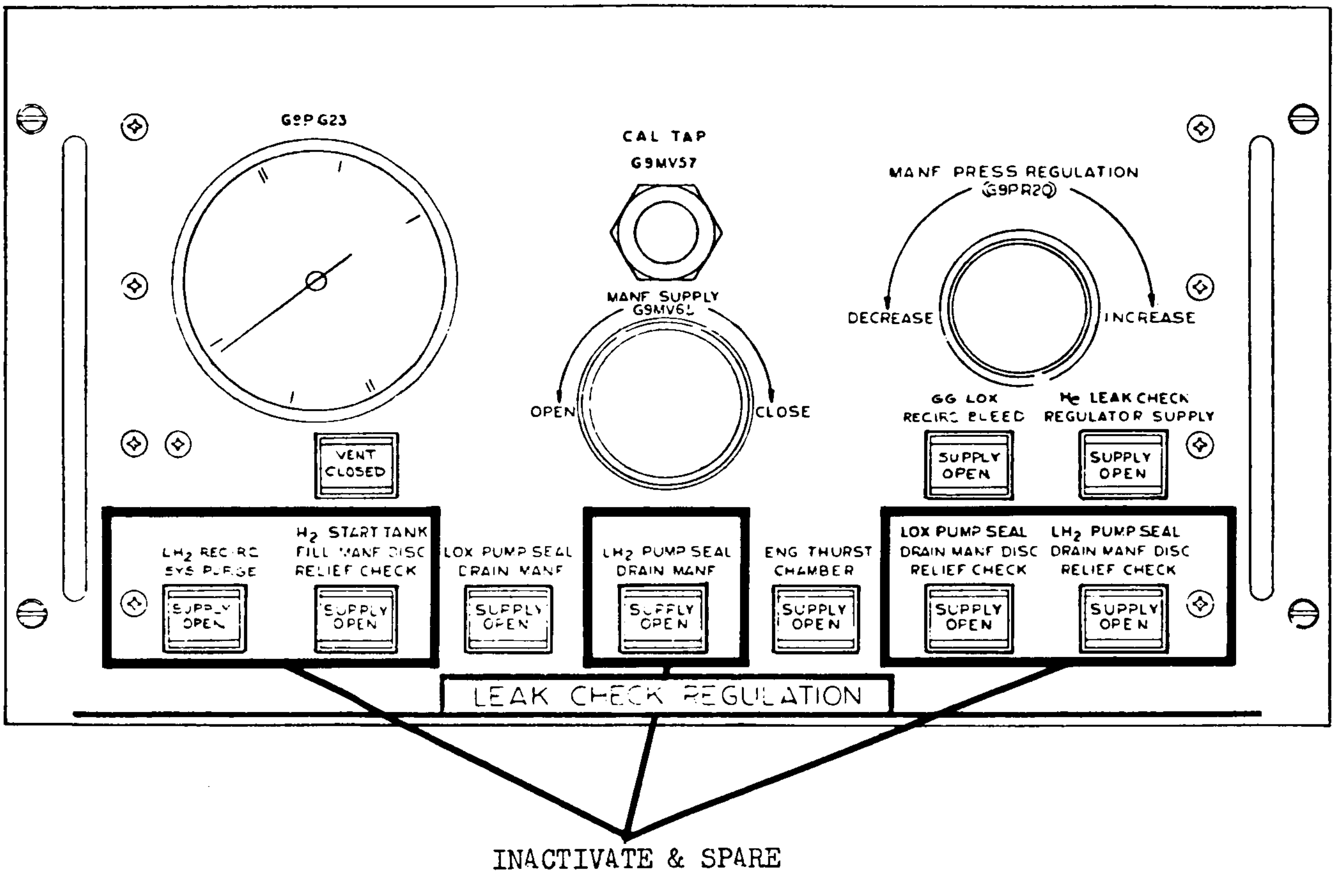


Figure 10. 2-147. Leak Check Regulation Test Control Panel Changes

The LH_2 feedline purge supply will be provided by redesignating the engine fuel recirculation system supply - 30-80 psig. Nomenclature will be changed on the high pressure regulation panel (Figures 10.2-146 and 10.2-148).

The oxidizer dome purge supply will be provided by redesignating the engine oxidizer recirculation system supply - 30-80 psig. Nomenclature will be revised on the front and rear high pressure regulation panel (Figures 10.2-146 and 10.2-147).

Calips mainstage OK pressure switches checkout requirements are 700 psig for the J2-S and 550 psig for the J-2. Pressure switches G9PS11 and G9PS15 will be reset to the higher pressure (Figure 10.2-149).

The 500 psig GH_2 start tank supply system will be inactivated by capping fluid junction FJ122 and removing nomenclature from front and rear of high pressure regulation and distribution panel (Figures 10.2-150 and 10.2-151).

Inactivate engine leak check panel 1 and spare the following circuits are (Figure 10.2-152):

GH_2 Start Tank Vent Control

Emergency Start Tank Vent Control

GH_2 Start Tank Pressure Meters

Revise nomenclature on the following circuits to reflect new functions:

GH_2 start tank control valves changed to component test-mainstage start solenoid

Oxidizer turbine bypass valves changed to mainstage cutoff lock-in

Gas generator valves changed to fuel bypass valve open

Engine leak check panel 2 will have engine gas generator bleed valve nomenclature changed to idle mode valve (Figure 10.2-153).

On the engine prevalve control and recirculation system panel the LH_2 and LOX return line valve circuits will be inactivated and spared (Figure 10.2-154).

Retest. All modified functional systems, excepting nomenclature change only, will be leak checked, and functionally tested as a subsystem. All modification can be accomplished in the field, and no contractual documents are affected.

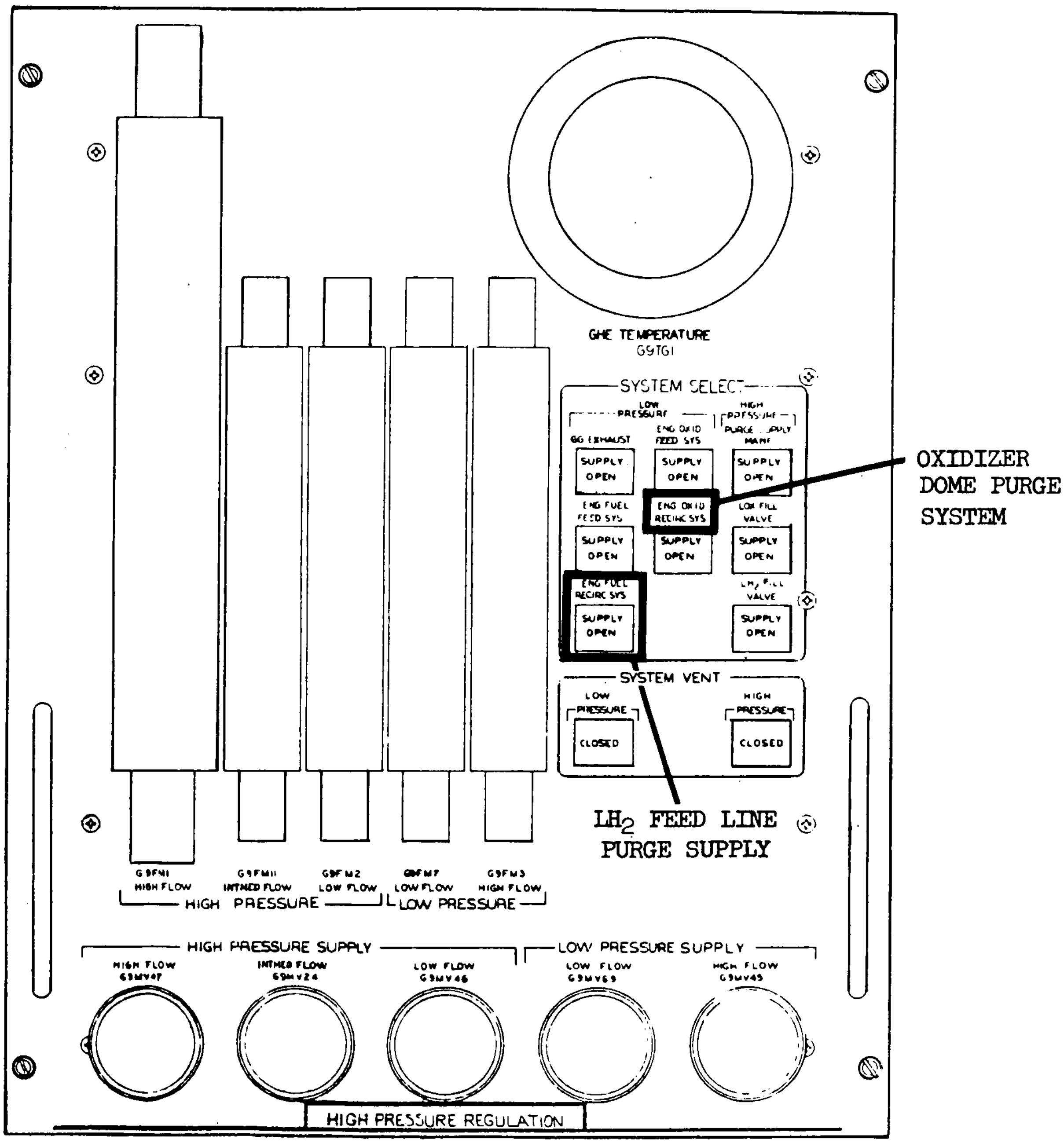


Figure 10.2-148. C7-603 System Changes (High Pressure Regulation Panel)

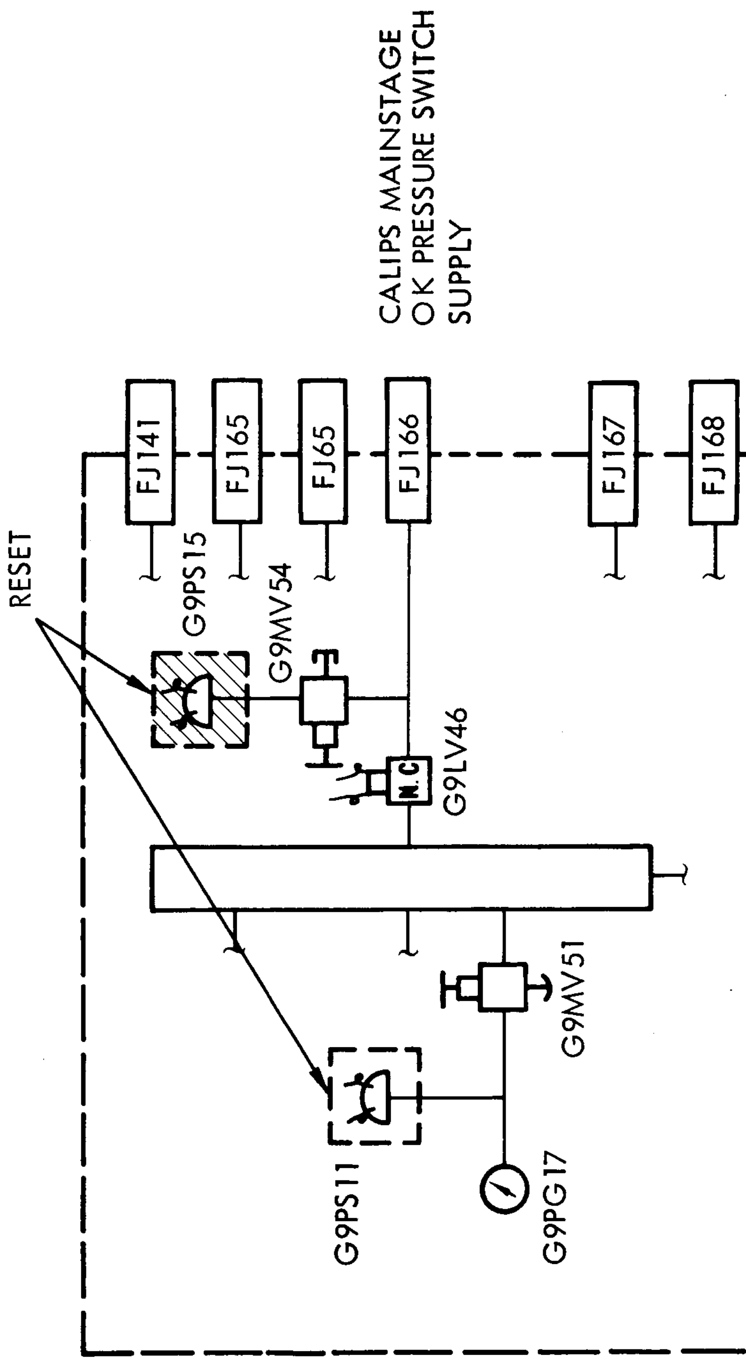


Figure 10.2-149. C7-603 System Changes (Calips Mainstage OK Pressure Switch Supply)

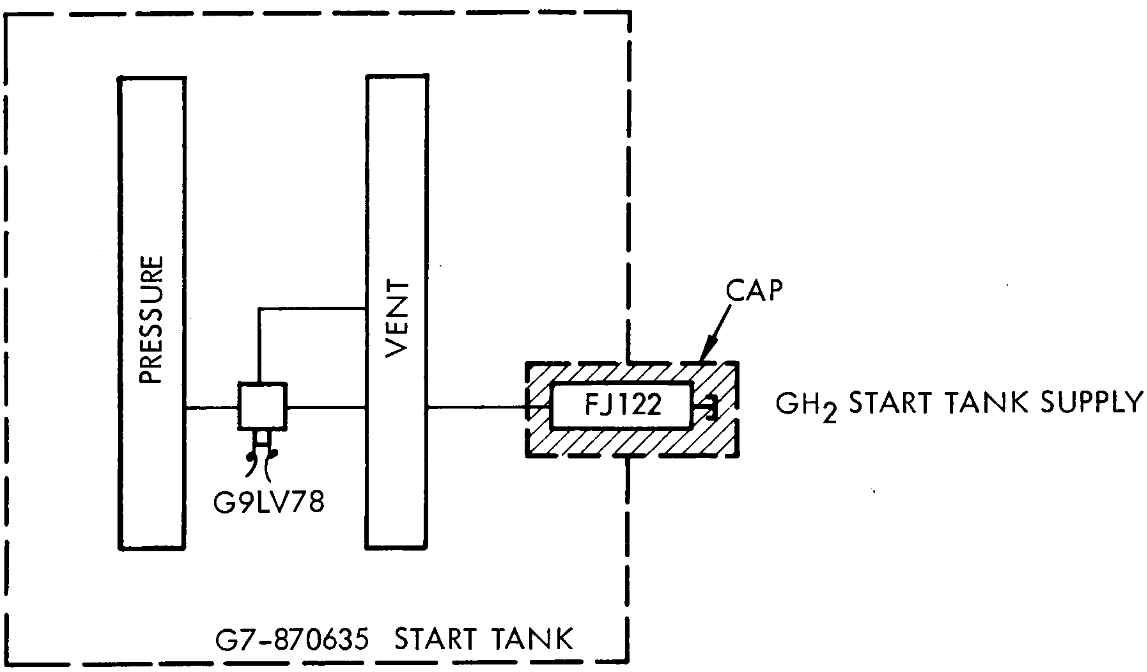


Figure 10.2-150. C7-603 System Changes (GH_2 Start Tank Supply)

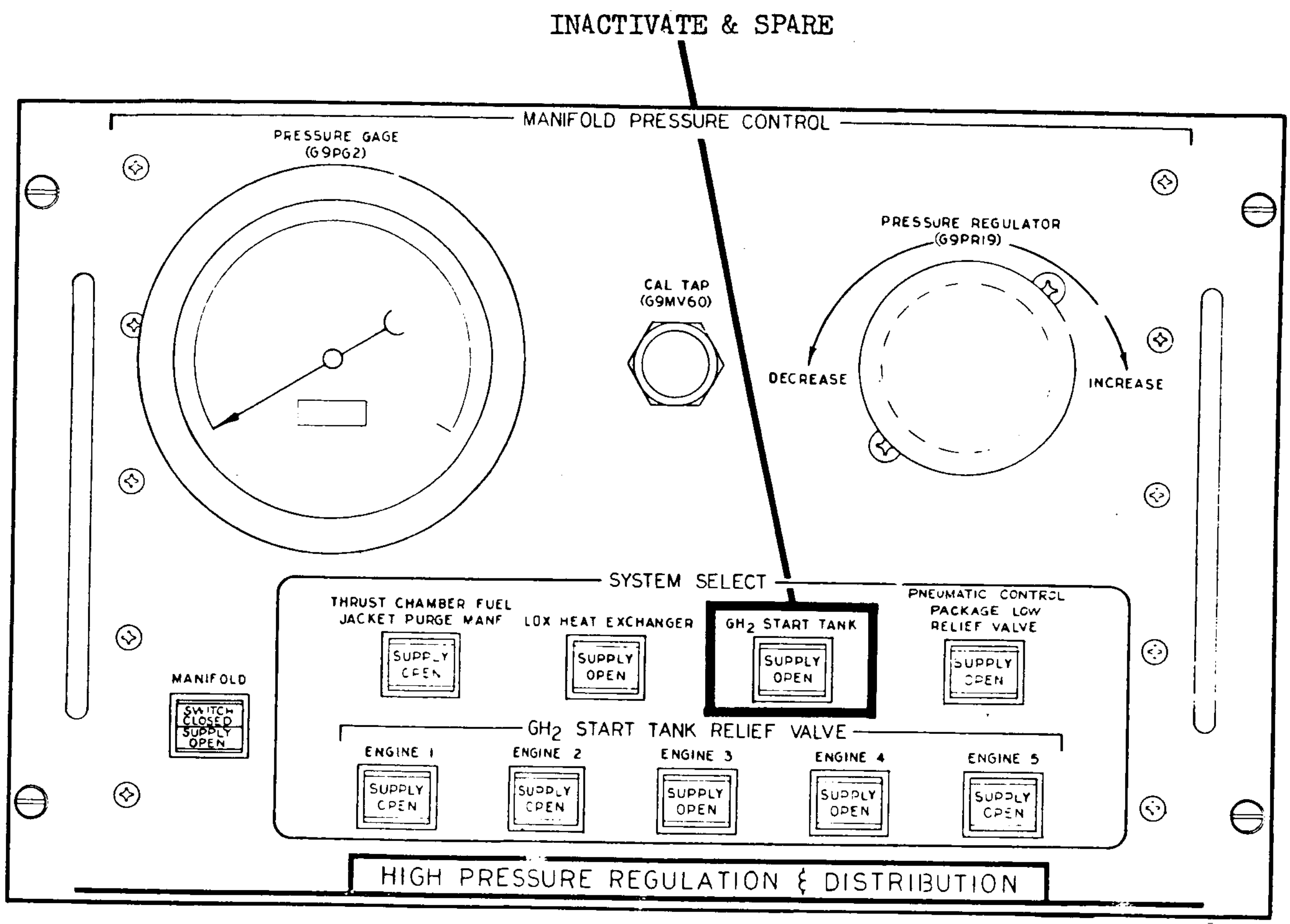


Figure 10.2-151. C7-603 System Changes (High Pressure Regulation and Distribution Panel)

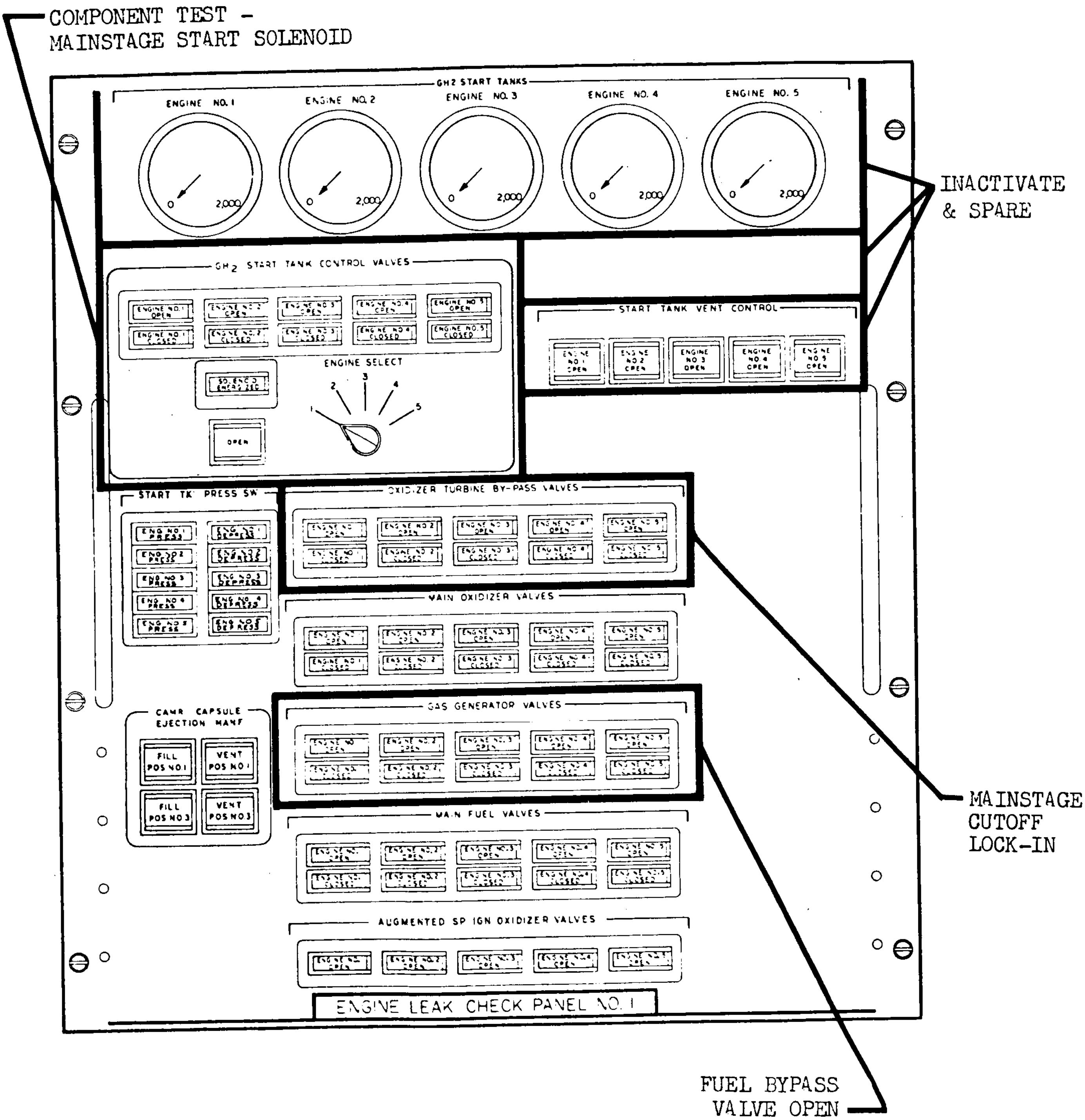


Figure 10. 2-152. C7-603 System Changes (Engine Leak Check Panel 1)

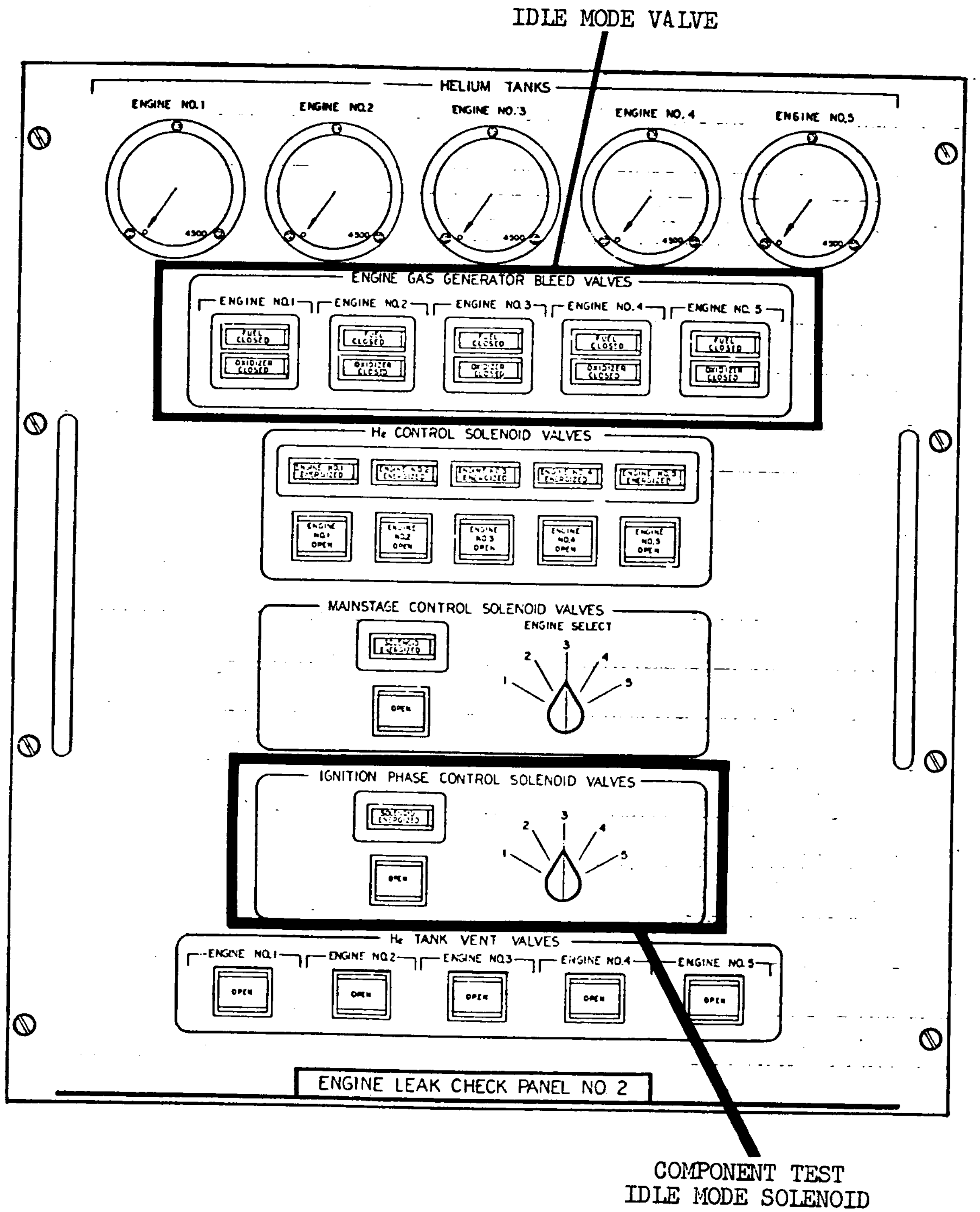


Figure 10.2-153. C7-603 System Changes (Engine Leak Check Panel 2)

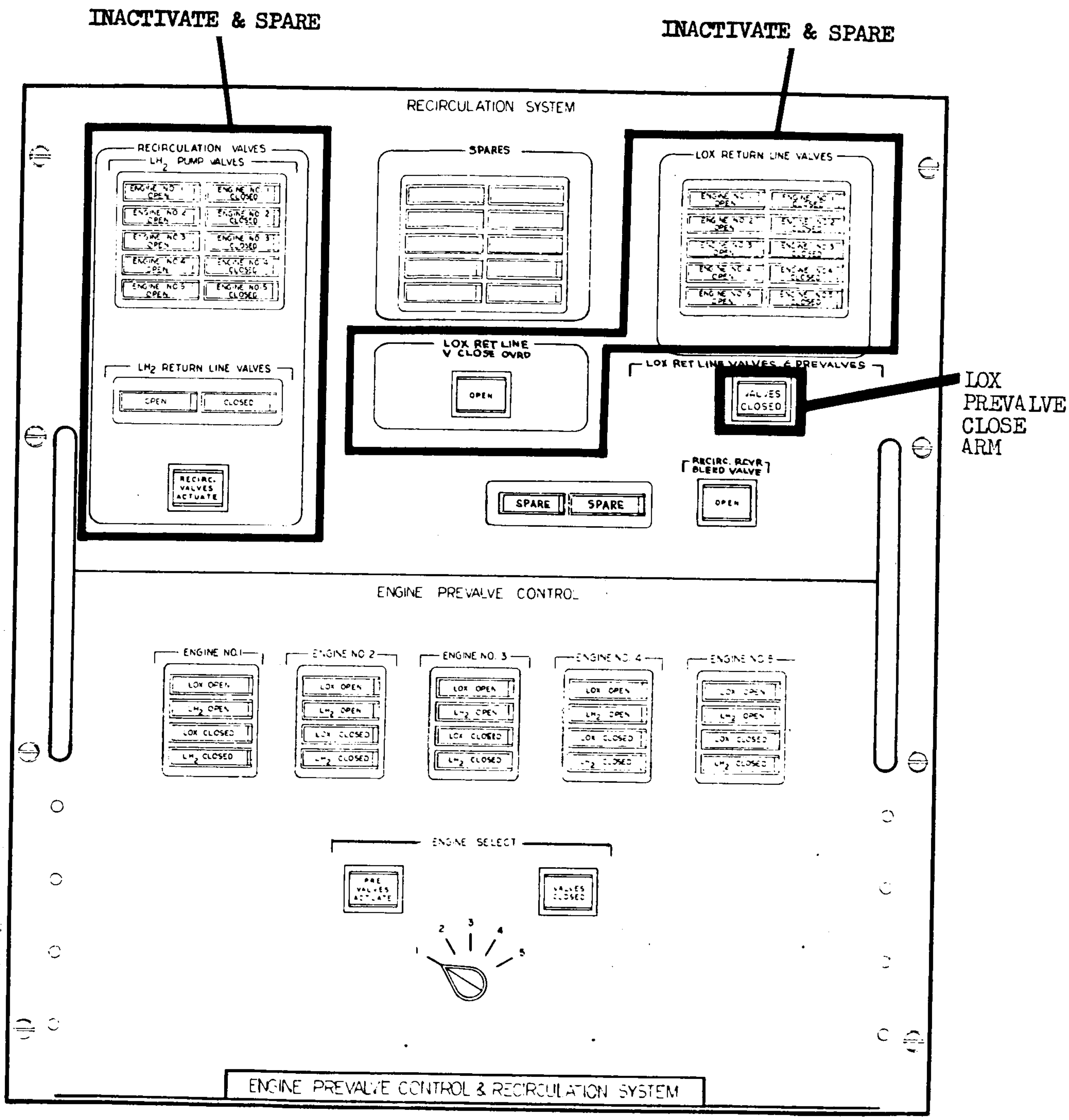


Figure 10. 2-154. C7-603 System Changes (Engine Prevalve Control and Recirculation System Panel)

10.2.7.1.4 Checkout MGSE Changes for the LEO Mission

a. C7-603, Pneumatic Checkout Console. The EAS air supply system function will be accomplished by deleting the G7-870648 module from the manual console and replacing it with a new module. With the removal of the G7-870648 module, the following fluid junctions will be capped and become nonfunctional: FJ56, FJ58, FJ321, FJ325 and FJ333 (Figure 10.2-155).

   1. Leak check and ramp pressure control module (G7-871294) will be revised to provide control for air supply solenoid valve, controls for stage's bleed valve for air reservoirs, and meters to monitor air pressure.

   2. Pressure selector module (G7-871297) will be revised to provide recording capabilities for C7-50 transducers and 20 percent and 80 percent calibration control for the C7-50 transducers.

   3. Modification will be accomplished in the field and leak and functional test will be performed after modification. Contractual documentation is not affected.

b. C7-50 Manual Pressure Checkout Transducer Set. Two 0-1000 psig pressure transducers and two calibration modules will be added, and storage will be provided. Modification is to be performed in the field; no retest is required. Contractual documentation is not affected.

c. C7-70 S-II Pneumatic Console - Test Set. New circuitry will be added to control and monitor the RCS and EAS air supply systems added to the S7-41. Modifications will be performed in the field, and retest will consist of functional test only. Contractual documents are not affected.

10.2.7.1.5 Servicing MGSE Changes for the LOR/PO Missions

The fluid distribution systems provide the plumbing interconnecting the servicing and checkout MGSE with the stage through the A7-61 carrier plates and H7-21 static firing skirt. The following are affected:

S7-27 Fluid distribution system, Test Stand A2, MTF

S7-33, Fluid distribution system, Test Stand A1, MTF

SDD-198, Fluid distribution system, Station VIII, Seal Beach

SDD-199, Fluid distribution system, Station IX, Seal Beach

1. Design Changes. Inactivated lines will be capped and active lines redesignated as appropriate. Inactivated flex lines interconnecting the fluid distribution system are to be removed to either the A7-61 or H7-21. There is no retest required, and all modification will be accomplished in the field. Contractual documents are not affected.

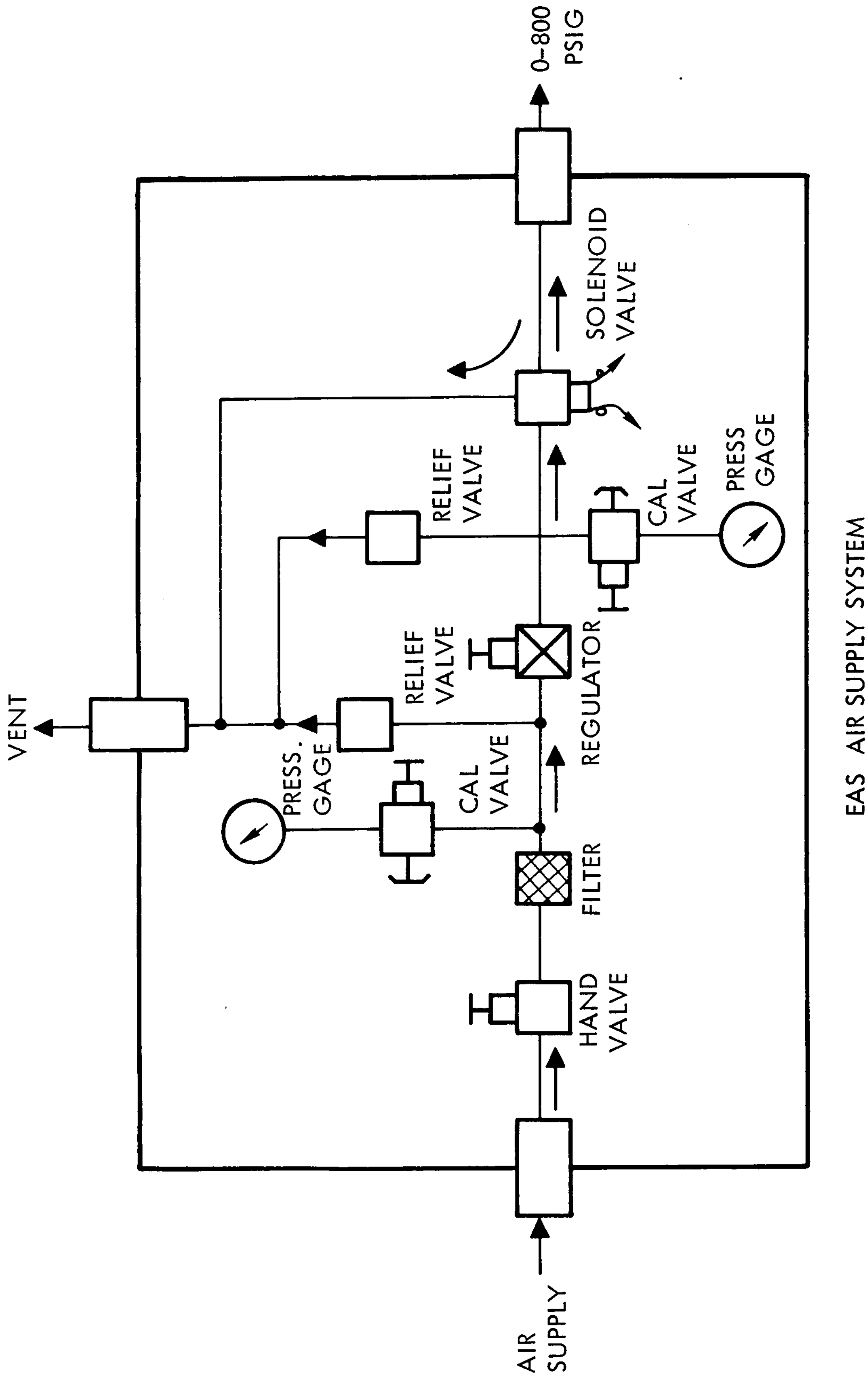


Figure 10.2-155. Engine Actuation Air Supply System

b. S7-41, S-II Pneumatic Console Set. The S7-41 provides the various pneumatic services required to check out and pressurize the S-II stage from prelaunch through launch.

1. Design Changes. The new LH_2 feedline purge requirement will be accomplished by using the existing LH_2 recirculation purge system. This will require changes in S7-41 panel nomenclature only. The new oxidizer dome purge requirement will be accomplished by the addition of a new system in the S7-41 A-section. This new system will include a pressure regulator, gauge, relief valve, calibration valves, manual valve, solenoid valve, pressure switch, filter, and pressure transducer. The fluid medium for this system will be gaseous helium. It is assumed that no heaters will be required to heat the gas at KSC (Figure 10.2-156).

The new prevalve actuation system servicing criteria will require a new regulation system in the S7-41 "A" section, and a new console interface and supply line to valve A9133 in the "B" section. The new regulation system will be provided by modification of the current thrust chamber chill system (which is no longer required). This will require replacing regulator A8991, relief valve A8995, transducer A15598, pressure switch A9154 and associated tubing with lower pressure and flow capacity ports. All other components in the existing system shall be used (Figure 10.2-157).

The revised engine thrust chamber and turbopump purge requirements will be provided by utilizing the existing thrust chamber purge system. Due to the new pressure and flow requirements, it will be necessary to replace transducer A15599 and relief valve A15557 in the "B" section and change relief valve A9001 and the setting of pressure regulator A8997 in the "A" section (Figure 10.2-158).

Deletion of the thrust chamber chill requirements will result in the inactivation of the portion of that system from console interface fitting A9199 to the outlet of check valve A9126 in the "B" section. Removal of these components may be required to provide space for the addition of other systems in that console (Figure 10.2-159).

Deletion of the turbine start bottle vent valve control system will result in the inactivation of that system by capping the inlet of regulator A9035 and interface fitting A9220 in the "C" section (Figure 10.2-160).

The existing turbopump purge system in the "C" section will be inactivated by capping the inlet of solenoid valve A9043 and interface fitting A9221 (Figure 10.2-161).

Deletion of the recirculation system will result in the inactivation of the recirculation bottle helium supply disconnect actuation system at MTF and KSC by capping at the inlet of solenoid valve A9066 and interface A9234. At MTF only, the recirculation shut-off valves actuation system and the recirculation shut-off actuation pressure sensing system will both be inactivated by capping at the inlet of solenoid A15513 and interfaces FJ79 and FJ80 (Figure 10.2-162).

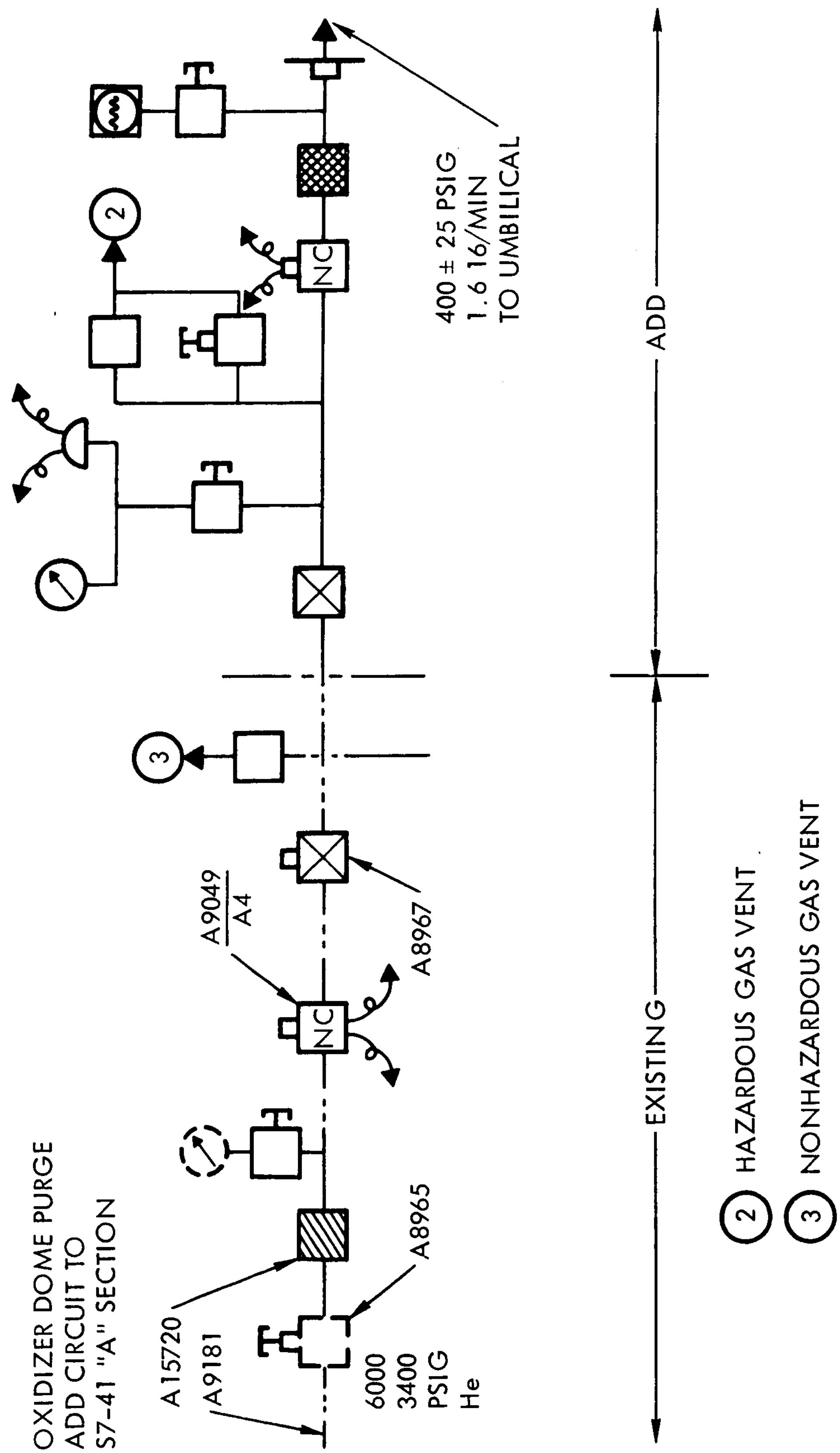


Figure 10.2-156. S7-41 Pneumatic Console Feedline Purge Changes

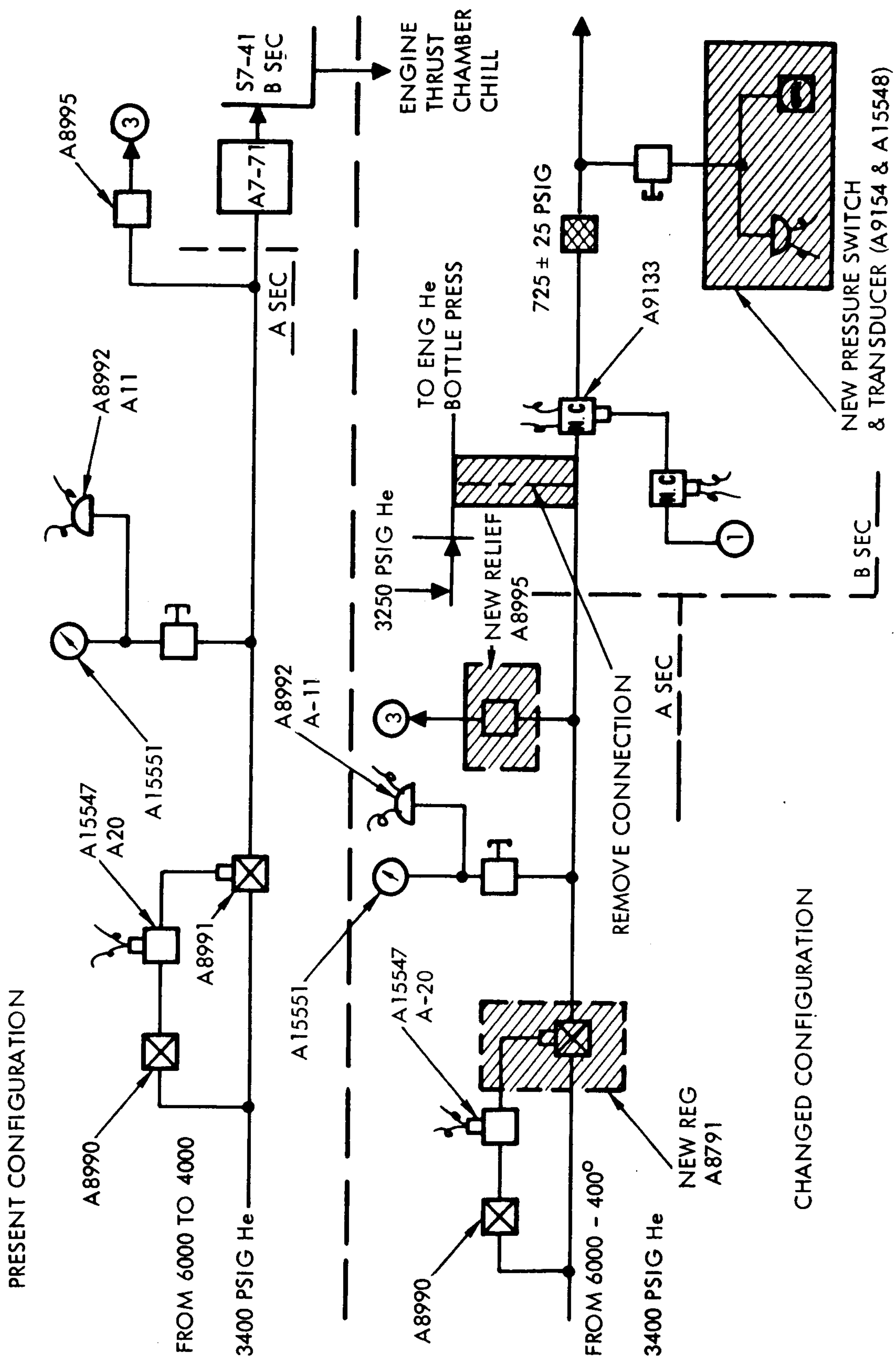


Figure 10.2-157. Prevalve Actuation System Thrust Chamber Chill Changes

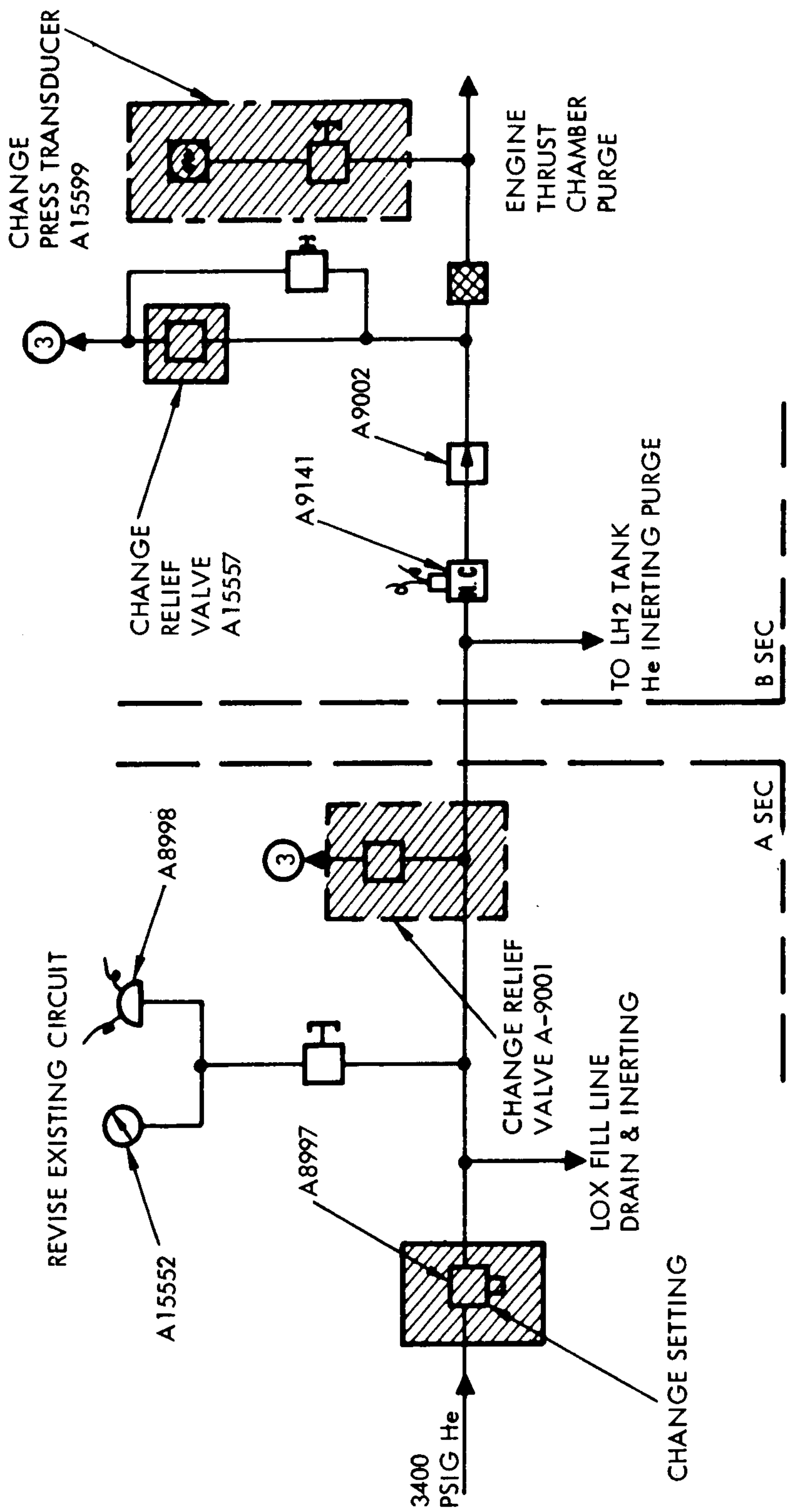


Figure 10.2-158. Engine Thrust Chamber and Turbopump Purge Changes

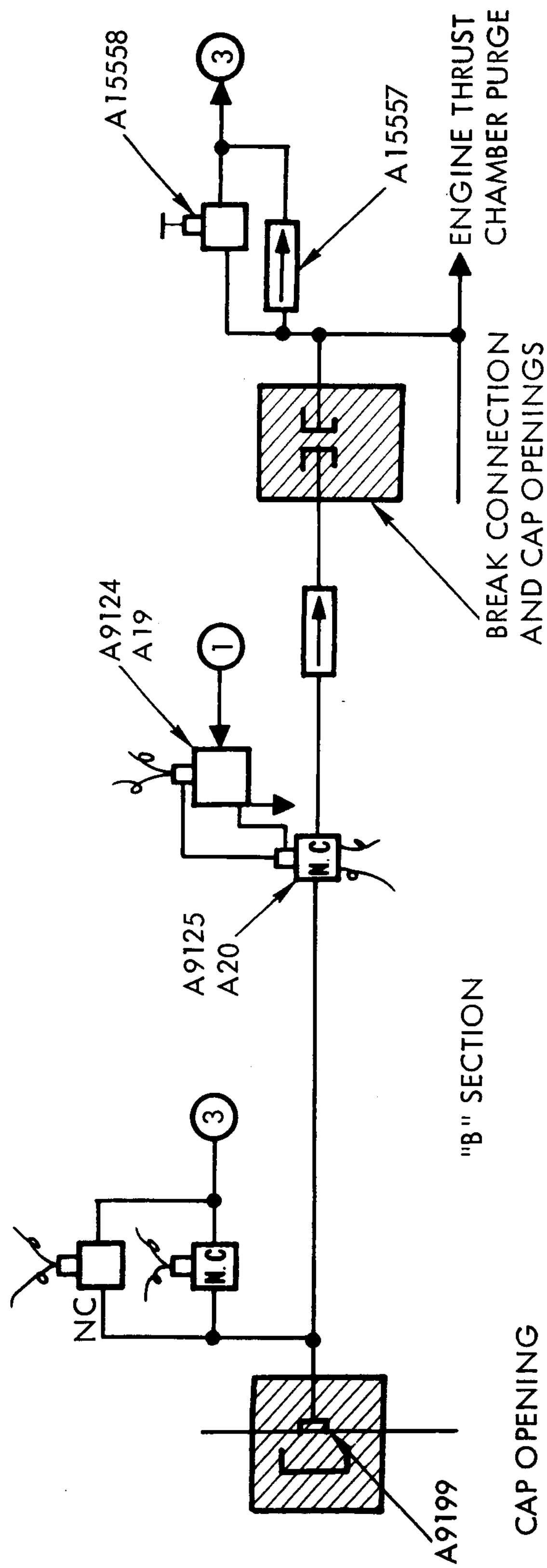


Figure 10.2-159. Thrust Chamber Chill System Deactivation

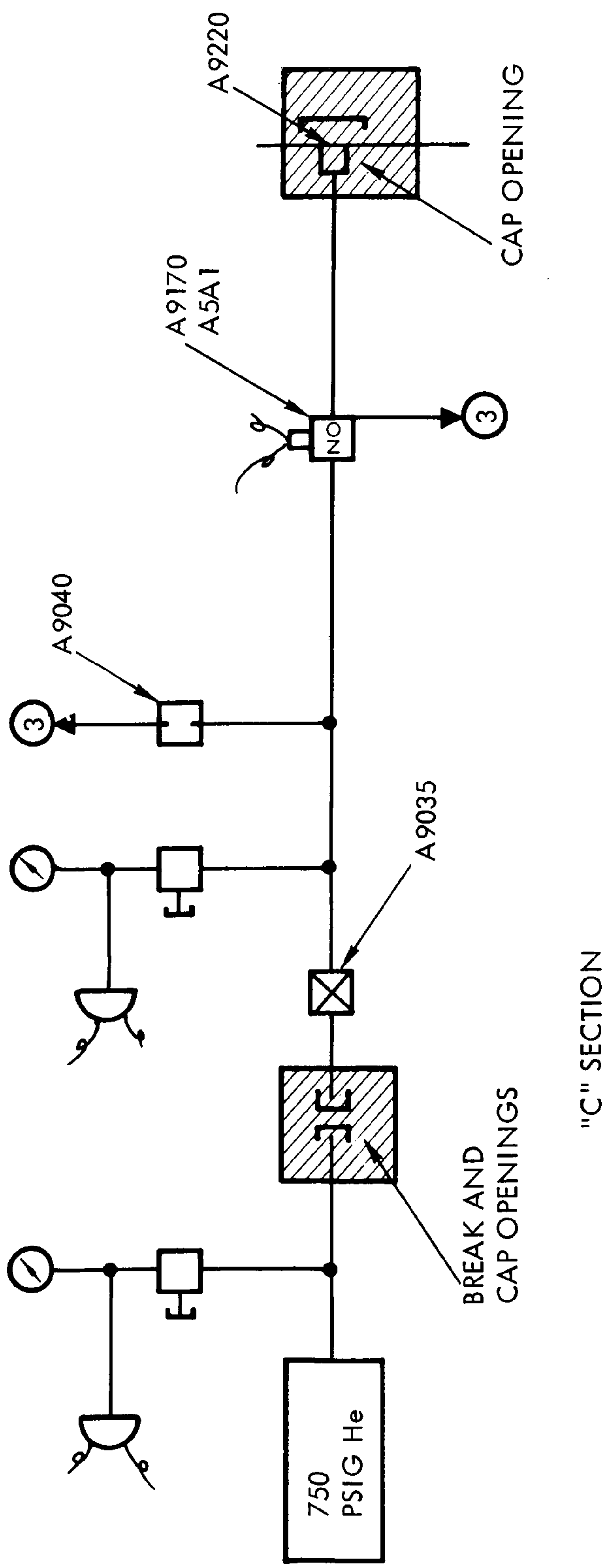


Figure 10.2-160. Turbine Start Bottle Vent Valve Control System Deactivation

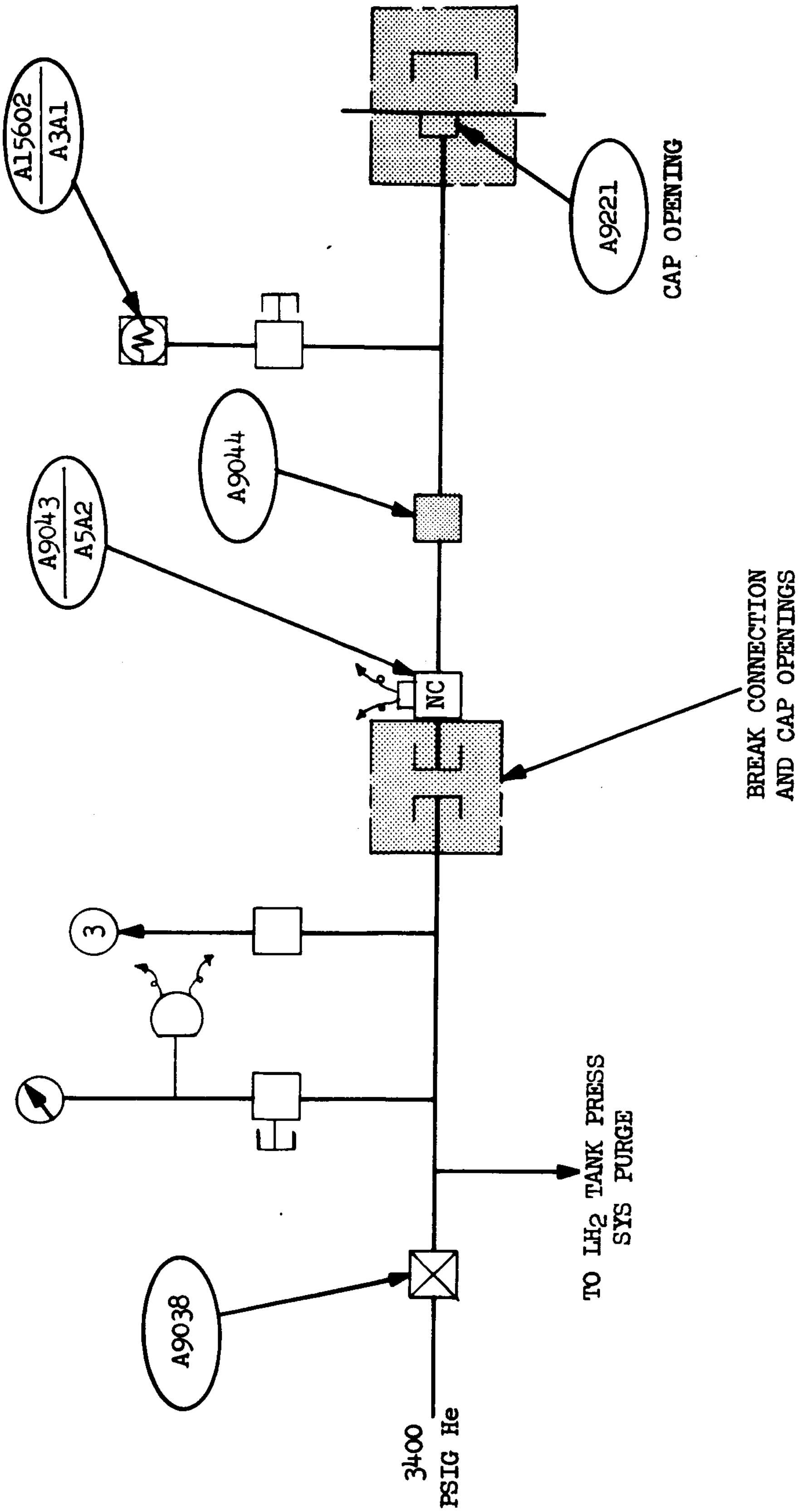


Figure 10.2-161. Turbopump Purge System Deactivation

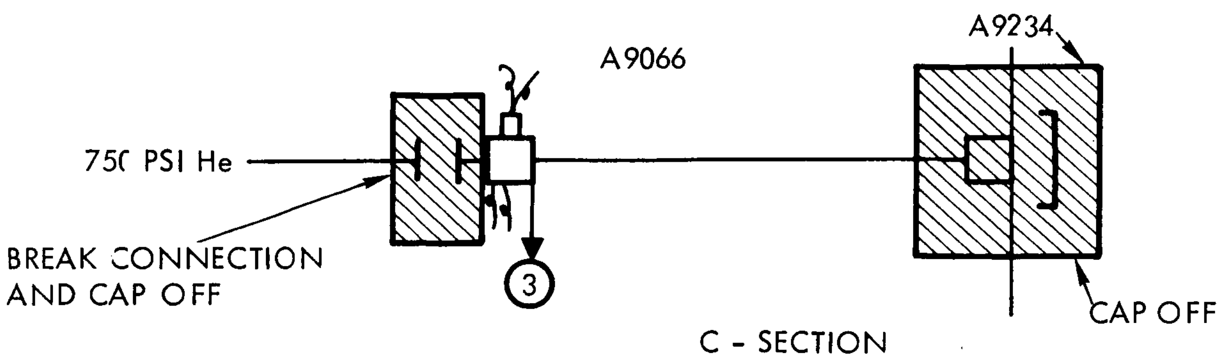


Figure 10. 2-162. Recirculation Bottle Helium Supply Disconnect Valve Actuation Pressure Deactivation (MTF and KSC)

The change in the checkout pressure for the mainstage OK pressure switches will result in readjusting regulator A9075, and pressure switches A9176 and A9177 in the "C" section (Figure 10.2-163).

Deletion of the stage turbine start bottles will result in eliminating the S7-41 "D" section in its entirety.

2. Retest requirements. Added and modified subsystems will be leak and functionally tested, and modification will be accomplished in the field.

3. Affected Contractual Documents:

Contract Specification CP181M0002A

Physical ICD, Pneumatic Console 65ICD9762

Pneumatic Console S7-41 C/B, 65ICD9763, Saturn V

Pneumatic Console S7-41 A/D, 65ICD9764, Saturn V

Piping Criteria S-II Pneumatic, 65ICD9766, Servicing

c. S7-42, Pneumatic Servicing - Electrical Console. The S7-42 contains the electrical control and monitoring circuitry to operate the A7-71 and S7-41 remotely when the A7-71 or S7-41 is isolated from the launch control center.

1. Design Changes. The LOX dome purge will have control and monitoring circuitry added for new pressure switch and solenoid valve added to the S7-41 console.

The control and feedback logic of the engine thrust chamber chill will be inactivated to inactivate system.

Control and feedback logic of the start tank pressure will be inactivated to inactivate system.

Prevalve actuation nomenclature will be changed to convert thrust chamber chill to new requirement.

Feedline purge nomenclature will be changed to convert LH_2 system purge to new requirement.

"Operating Pressures OK," "System in Standby," etc., logic will be revised to eliminate inactivated components.

2. Retest. The modified system will be tested for functional compatibility with the S7-41. All modifications will be accomplished in the field, and no contractual documents are affected.

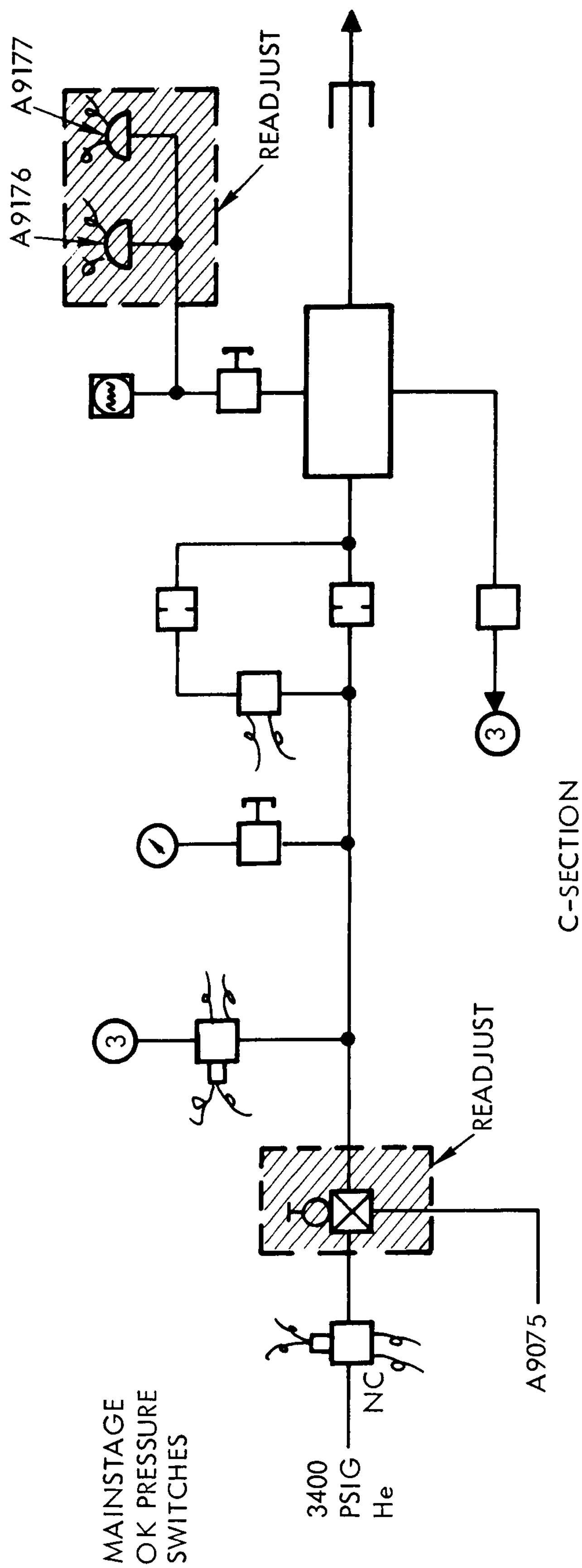


Figure 10.2-163. Mainstage OK Pressure Switches Readjustment

10.2.7.1.6 Servicing MGSE Changes for the LEO Mission

a. Design changes require the reactivation of lines on the following systems inactivated for LOR for the EAS air supply and checkout system:

S7-27, Fluid distribution system, Test Stand A2, MTF

S7-33, Fluid distribution system, Test Stand A2, MTF

SDD-198, Fluid distribution system, Station VIII, Seal Beach

SDD-199, Fluid distribution system, Station IX, Seal Beach

b. S7-41, S-II Pneumatic Console Set

1. Design Changes. A reaction control helium pressurization system will be added to the "B" section (Figure 10.2-164). The LOX return line helium injection subsystem will be modified to provide EAS air supply (Figure 10.2-165).

2. Retest. Added and modified subsystems will be leak and functionally tested. Modifications are to be accomplished in the field.

3. Affected Contractual Documents

Contract Specification CP181M002A

Physical ICD, S-II Pneumatic Console S7-41 B/Saturn V, 65ICD9762

Piping Criteria S-II Pneumatic, 65ICD9766, Servicing

c. S7-42 Pneumatic Servicing - Electrical Console

1. Design Changes. Circuitry will be added to control and monitor the RCS and EAS air supply subsystems added to the S7-41. Modifications are to be accomplished in the field. Retest will consist of functional verification only, and no contractual documents are affected.

10.2.7.1.7 Handling MGSE Changes for LOR/PO Mission

a. H7-21, Static Firing Skirt. The H7-21 provides the aft structural interface to the S-II for handling and transportation. The electrical and fluid systems required to check out the S-II, not serviced through the umbilicals, terminate at the H7-21 for redistribution to the stage systems.

1. Design Changes. Inactive termination will be capped and active connections redesignated as appropriate. No retest is required, and all modifications are to be accomplished in the field. Contractual documents are not affected.

10.2.7.1.8 Handling MGSE Changes for the LEO Mission

a. SDD-258, LOX Tank Internal Access Platform

1. Design Changes. Modify lower stabilization strut will be modified to clear added slosh baffle (Figure 10.2-166). Modification will be accomplished by reworking the existing strut, and retest will consist of proof loading the reworked strut. No contractual documents are affected.

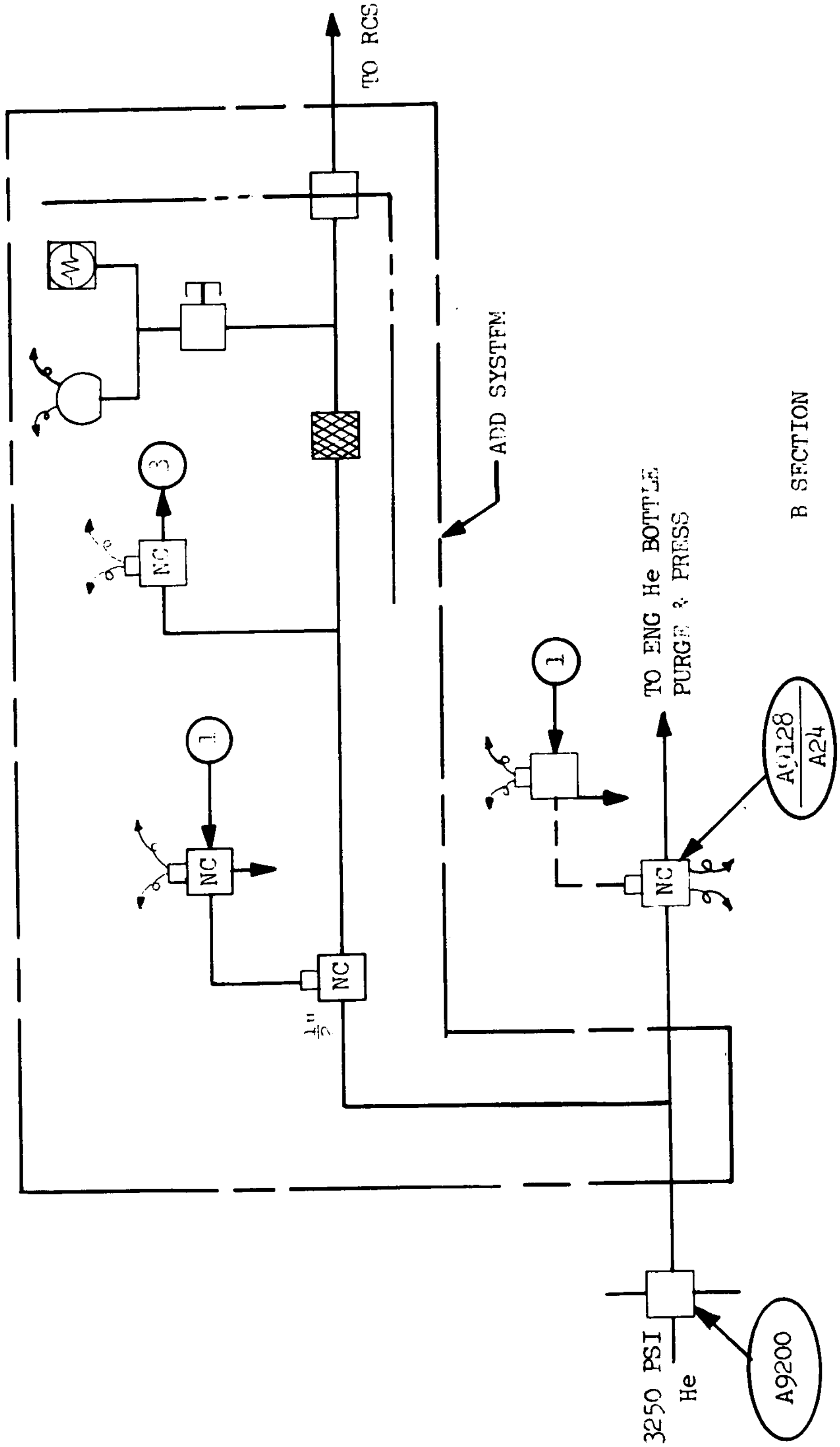


Figure 10. 2-164. Reaction Control Helium Pressurization System

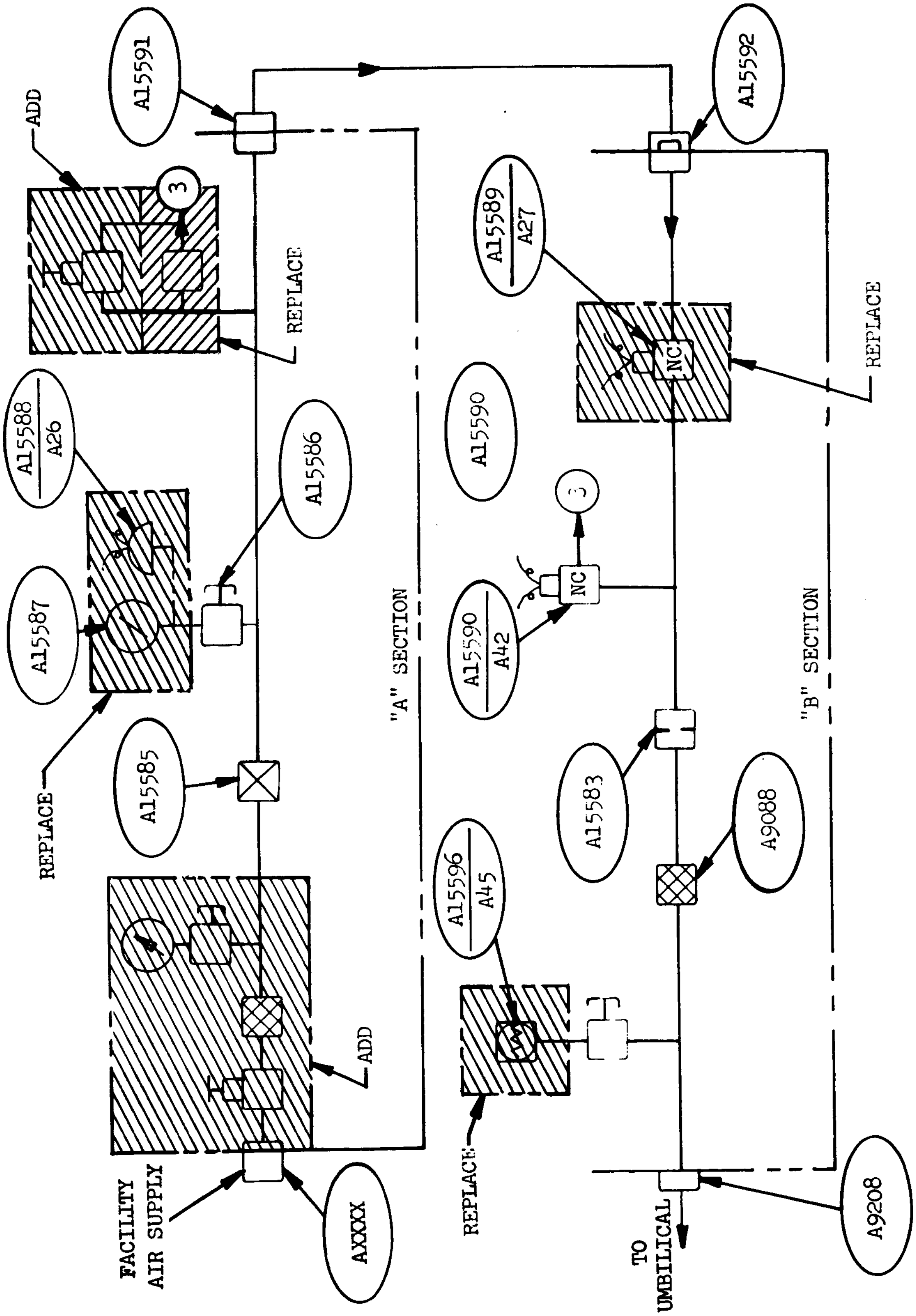


Figure 10.2-165. Air Atmosphere Supply System, S7-41 Pneumatic Servicing Console

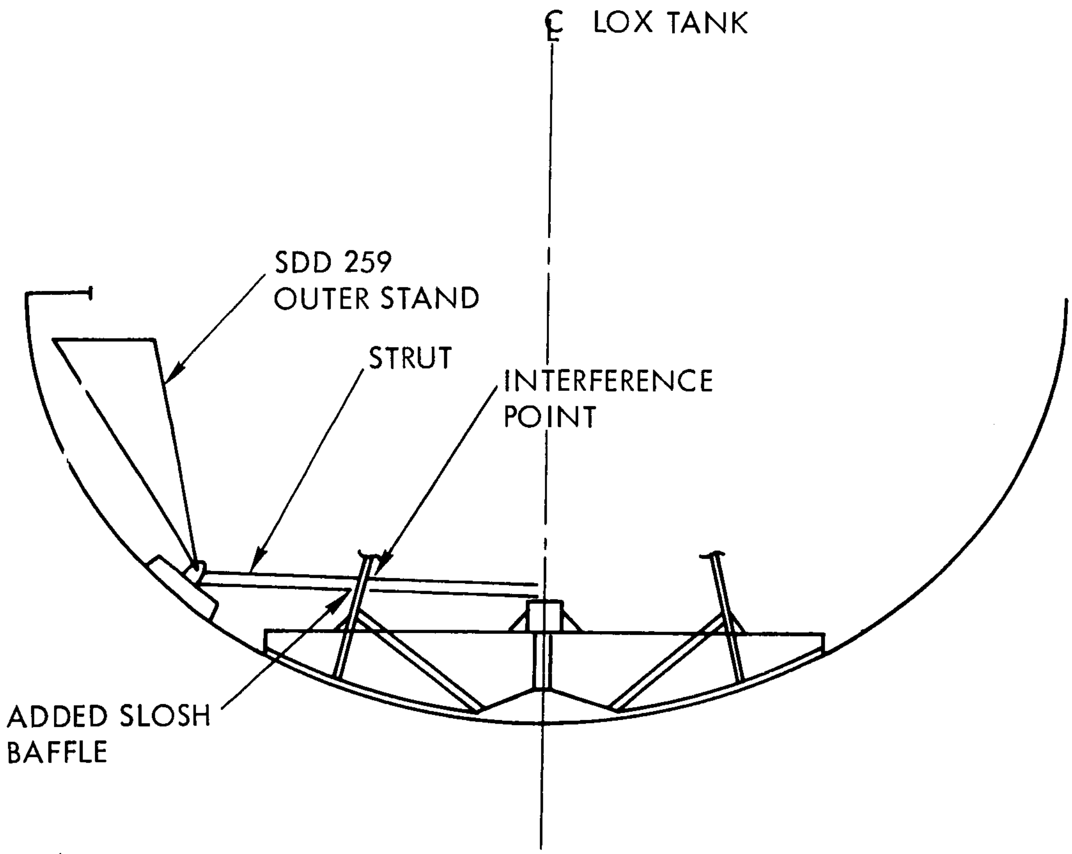


Figure 10.2-166. LOX Tank Internal Access Kit
Modification for LEO Mission

10.2.7.2 Electrical Support Equipment (ESE)

The changes to the S-II Stage ICD's and checkout requirements to incorporate the J-2S engines will provide the ESE contractor with the ESE hardware change requirements necessary for the LOR and LEO missions.

10.2.7.3 Automatic Checkout Equipment (ACE)

The primary purpose of the automatic checkout system is to perform a comprehensive checkout of the Saturn S-II stage (with J-2S engine implementation) before shipment to KSC. A series of GSE end items are used to test the Saturn S-II stage after manufacture under conditions which most nearly simulate actual flight of the stage. Seal Beach test sites provide a thorough check of the electrical, mechanical, fluid, and telemetry systems. MTF test sites perform these same functions and also provide structural testing through the static firing.

The GSE is designed to provide as much automatic testing as practicable, but manual functions are retained as backup to automatic functions and to provide certain functions not feasible to automatic control.

10.2.7.3.1 ACE System Requirements - LOR/PO Missions

The Saturn S-II stage automatic checkout system has been designed as a group of checkout stations controlled by a computer complex. These specialized checkout stations must be able to test the following systems on the stage:

1. Engine system
2. Propellant feed system
3. Propellant management system
4. Engine compartment conditioning system
5. Pressurization system
6. Electrical power system
7. Electrical control system
8. Separation system
9. Destruct system
10. Emergency detection system
11. Flight control electronic system
12. Thermal control system
13. RF systems
14. Measurement system

'he computer complex serves as the central automatic control for the checkout tations. Each checkout station has facilities for checkout of the mechanical systems, lectrical system, RF systems, digital data acquisition systems and telemetry sys- ɜms. Figure 10.2-167 is a block diagram of the automatic checkout system.

. Checkout Station Requirements. The automatic checkout system for the J-2S engine implementation consists of a controlling computer complex linked to a number of specialized checkout stations. These checkout stations, in turn, are linked to the functional systems on the stage. Associated with its particular function, the checkout station must provide stimuli to the stage; provide control signals to associated checkout stations and the stage; monitor GSE and stage system responses; store system responses and results of evaluation, as required: translate system responses into a suitable format for facilitating evaluation; evaluate system responses (further testing or corrective action is to be undertaken on the basis of evaluated results; provide visual readouts and complete records of test procedure and system responses; and provide a self-check capability for maintenance and machine confidence.

1. Computer Complex Checkout Station (C7-100) - The C7-100 checkout station consists of the C7-101 checkout computer and its peripheral equipment (C7-103 program input rack, C7-104 data printout rack, C7-105 auxiliary memory rack; and C7-110 high speed data printout rack), C7-102 test conductor console, C7-106 buffer equipment rack, C7-107 local digital driver link rack (MTF only), C7-108 remote digital driver link rack (MTF only), C7-109 isolation and drive rack, and C7-111 magnetic tape transport rack (MTF only). The purpose of the computer complex checkout station is to link the automatic operations of the GSE checkout stations and S-II stage systems. Via the checkout stations (discussed subsequently), the checkout computer will control the sending of stimuli to the S-II stage systems and the monitoring of responses from the systems.

(a) C7-100 Station Design Changes. There are no changes to the C7-100 computer complex end items because of J-2S engine implementation, except for the C7-102 test conductor console. The following five functions are deleted from the hazardous monitor panel because of deletion of the recirculation system:

Recirculation system dc greater than 64v

Recirculation system dc greater than 60v

Recirculation system dc low

Recirculation system helium pressure bottle greater than 1600 psig

Recirculation system helium bottle greater than 825 psig

These changes require removal of the nameplates and deactivation of the cabling in the J-boxes which connect the hazard functions to the C7-102.

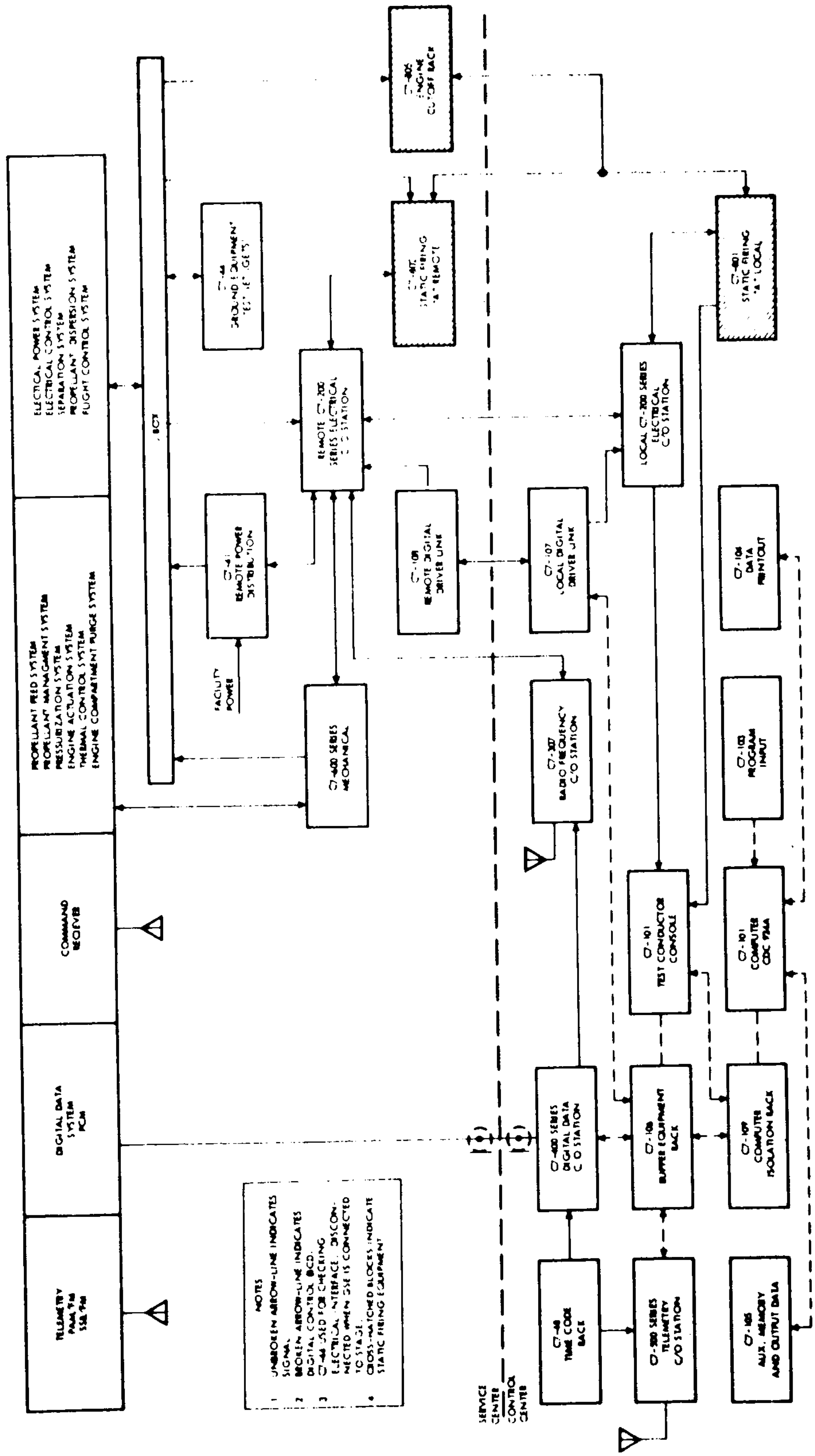


Figure 10.2-167. Automatic Checkout System Block Diagram

Affected documentation includes the following:

MA0201-1048	Hazardous Monitor Process Specification
MA0705-1015-111	Semiautomatic Checkout C7-100, (Stations VIII, IX)

The principal software items affected by electrical GSE changes are:

C7DXXX-127-XXX	GSE Integrated Test
C7DXXX-250-XXX	Electrical Power System Checkout
C7DXXX-251-XXX	Flight Measurements System Checkout
C7DXXX-252-XXX	Stage Networks Acceptance Program
C7DXXX-253-XXX	Flight Control System Checkout
C7DXXX-254-XXX	Pressurization System Checkout
C7DXXX-255-XXX	Simulated Flight System Checkout

2. Electrical Checkout Station (C7-200). The electrical checkout station consists of the C7-201 automatic control rack, C7-202 manual control and display rack, C7-204 signal distribution rack, C7-205 special data rack, C7-208 station control and display rack, C7-209 local control and display rack, C7-210 stage substitutes rack, C7-211 scanner rack, C7-212 discrete display rack, and C7-213 interlock relay rack.

The station provides hardware for the electrical checkout of many stage systems. Checkout can be done in either manual or automatic mode.

In the automatic mode, hardware is provided for electrical checkout under computer control, allowing the computer to select and direct station equipment operation and to transform data into a format for computer analysis and status recognition. Display of stimuli and responses provides rapid station status as well as easy troubleshooting and fault isolation. Interlocks of stage power, stimuli and responses and associated limit condition sensing of the power buses assure safe checkout operation. The electrical checkout station is used in testing the following stage systems:

Propellant Dispersion	Engine	Propellant Management
Separation	Pressurization	Flight Control
Electrical	Measurement	Thermal Control

(a) C7-200 Station Design Changes. The C7-202 manual control and display rack changes are in the switch selector control and display drawers, and are due to the addition and deletion of switch selector commands. These consist of deleting two commands, revising two commands, and adding seven new commands. The changes are accomplished by changing nomenclature on the display panels and by adding and rewiring diodes in the control and display drawers.

The following C7-202 rack documentation is affected due to switch selector command additions and deletions:

G7-852600	Rack Schematics
G7-872322	Switch Selector Control Drawer
G7-872323	Switch Selector Control Panel
G7-872326	Switch Selector Display Drawer
G7-872327	Switch Selector Display Panel
MA0201-1235	Switch Selector Display C7-202 Rack
MA0701-1012-111	Manual Checkout C7-200, Stations VIII, IX

Six new drawings are required for the switch selector control and display drawer harnesses, component boards, and drawer schematics.

The C7-204 signal distribution rack changes are in the engine system drawers. They consist of deleting 21 commands, revising 11 commands, and adding 25 commands. Twenty-one added commands will use the spared positions, and four new commands will require additional hardware (the addition of four flip-flops, four relays, and four relay drivers). This will be accomplished by repatching and adding approximately twenty new patchcords. These changes are caused by the deletion of the recirculation system, helium injection system, and start tank system. Functions to engine mainstage operation, engine ready and engine ignition detect signals were added.

Documentation affected by these changes is as follows:

G7-852300	Control Logic No. 1 (20 Percent)
G7-852640	Command Distributor Relay Logic No. 1 (20 Percent)
G7-857100	Rack Schematic (15 Percent)
MA0701-1012-111	Manual Checkout C7-200, Station (VIII or IX)
MA0701-1014-212	Static Firing Checkout

These modifications require three new rack harnesses and one new output patch panel assembly.

The C7-209 local control and display rack will be changed by revising the nomenclature on six switch positions on the engine stimuli control panels and by revising 26 legend light nomenclatures on the engine stimuli display panels.

The following documentation is affected:

G7-986072	Rack Schematic (20 Percent)
MA0701-1012-111	Manual Checkout C7-200 Station VIII or IX

C7-209 rack changes are due to deletion of recirculation system, helium injection system, and start tank system. Principal additions are functions for engine mainstage operation, engine cutoff, engine ready, and engine ignition detect.

Changes to the C7-211 scanner rack consist of adding approximately 60 patchcords. Documentation affected by the change is as follows:

G7-855810	Rack Top Drawing (15 Percent)
G7-981683	Rack Schematic (20 Percent)
G7-981684	Rack Schematic (20 Percent)
MA0201-1532-001	Rack Process Specification (20 Percent)
MA0201-1532-002	Rack Process Specification (20 Percent)

C7-211 rack changes are caused by the deletion of the recirculation system, start tank system and helium injection system commands and measurements, and the addition of the engine mainstage operation, engine ready, engine detect, and engine start system signals. In some cases the C7-211 rack provides a lamp driver for stage measurements with a 4.02K resistor in series.

The C7212 discrete display rack changes require the revision of 60 legend lights and the addition of 30 new legend light connections.

G7-981688	Rack Schematic (20 Percent)
G7-981689	Rack Schematic (20 Percent)
MA0201-1535	Rack Process Specification (20 Percent)

The deletions are caused by deletion of recirculation and helium injection systems. The additions are due to new engine cutoff commands.

The C7-213 interlock rack changes consist of revising the patch panels. Approximately 100 patchcords are deleted and 90 are added. Interlock requirements for J-2S engine implementation have not yet been fully defined.

Documentation affected by modifying the C7-213 rack are:

G7-855850	Rack Top Assembly (15 Percent)
G7-981679	Rack Schematic (15 Percent)
G7-981682	Rack Schematic (15 Percent)
MA0201-1786-001	Rack Process Specifications
MA0201-1786-002	Rack Process Specifications

C7-213 Rack changes are caused by deletion of recirculation pump start interlock and start tank vent open commands. The additions are not yet clearly identified, but the changes entail only patching change in the C7-213 rack.

Functional test documentation affected by the changes to the C7-211, C7-212, and C7-213 is as follows:

MA0701-1012-111	Manual Checkout C7-200 Station VIII or IX
MA0701-1014-212	Static Firing Checkout

No changes result to the C7-201 automatic control rack, C7-205 special data rack, C7-208 station local control and display rack and the C7-210 stage substitutes rack.

3. Range Safety Command Receiver (RSCR) Checkout Station (C7-307). The RSCR checkout station provides equipment and circuitry for checking the radio command receivers on the stage. Test signals from the rack are transmitted to the stage radio command receivers via hardwire or air link. Signals indicating response of stage receivers are returned to the rack for comparison with specific requirements. This signal comparison is performed by personnel operating the rack.

   The rack verifies proper operation of stage-installed receivers, power dividers, filters, and hybrid and interconnecting coaxial cable links. The command antenna subsystem test verifies the operation of each command antenna and its associated voltage standing wave ratio (VSWR) measurement. Partial automatic rack operation is provided by the electrical checkout station and the digital data acquisition station.

   (a) C7-307 Changes. There are no changes to the C7-307 rack caused by J-2S engine incorporation.

4. Digital Data Acquisition System (DDAS) Checkout Station C7-400. The DDAS checkout station is composed of four racks, C7-401 automatic control and display rack, C7-402 local control and display rack, C7-403 PCM (DRS-1) rack, and C7-406 computer adapter rack. This station provides the hardware and circuitry to retrieve data from the pulse code modulated (PCM) telemetry system of the stage. Code pulse trains from the PCM systems

are regenerated to remove transmission noise and are converted from serial to parallel format to produce information suitable for computer processing. Capability for recording and limited playback is made available through control and feedback lines to the tape recorder rack (C7-516) of the telemetry checkout station.

The station receives, demodulates, and decommutates PCM data from either the stage or from magnetic tapes, and routes the data to display at the station or the telemetry ground station tape recorder (C7-516).

(a) C7-400 Station Design Changes Due to LOR Mission. There are no changes to the C7-400 checkout station due to J-2S engine incorporation.

5. Telemetry System Checkout Station (C7-500). The telemetry checkout station is composed of 9 racks: C7-510 automatic control and display rack, C7-511 PCM format rack, C7-512 oscillograph rack, C7-513 decommutation rack, C7-514 discriminator rack, C7-515 receiver rack, C7-516 tape recorder rack, C7-518 single sideband rack, and C7-519 telemetry receiver station, Model TRS-1. The purpose of the C7-500 station is to check the airborne telemetry systems of the stage. The primary functions are as follows:

   1. Receive, monitor, and detect RF carrier frequencies
   2. Provide for the magnetic recording and playback of undecoded intelligence transmitted via the FM/FM, and PCM/FM telemetry systems
   3. Decode FM/FM data
   4. Convert analog data to the PCM format for computer access during automatic checkout
   5. Provide for local monitoring of single-channel PCM data
   6. Provide for local control of the stage RF equipment and telemetry calibrations
   7. Process and record a limited amount of telemetered data for post-test evaluation.

(a) C7-500 Station Design Changes. The C7-500 telemetry checkout station is unaffected by changes to incorporate the J-2S engine.

6. Static Firing Control Station (C7-800). The C7-800 station exists at MTF only, and is activated only for static firing tests. The station consists of the C7-801 (local static firing rack), the C7-802 (remote static firing rack), and the C7-805 (eingine cutoff rack).

The C7-801 and C7-802 racks permit manual control, display, and signal distribution of all 28-vdc parameters required for a static firing of the S-II stage (except engine gimbaling, which is computer controlled). These racks also provide for automatic sequencing (with manual override capability) during the interval from completion of propellant loading to initiation of propellant detanking. The racks are also integrated with the facility programming bay to provide a time-oriented countdown. The C7-801 rack consists of four bays and is located in the test control center. The C7-802, consisting of three bays, is located in the ground service center and is remotely controlled by the C7-801 rack.

The C7-805 rack permits the monitoring of certain engine parameters and provides automatic cutoff of all engines in the event of a dangerous condition during static firing. The rack indicates the cause of cutoff by means of auxiliary monitoring devices. The C7-805 rack, consisting of two bays, is located in the ground service center and is remotely operated and self-tested from the C7-801 rack.

(a) C7-800 Station Design Changes. Changes to the C7-801 local static firing "A" rack consist of adding 7 new switch selector channels, adding 10 new switches with indicator lights, deleting 13 switches and 9 associated indicator lights, adding one new switch panel, and adding 4 new diode boards. Because of the deletion of the start tanks, two start tank meter panel drawers are inactivated.

C7-801 rack changes require documentation change of the following:

G7-981676	Rack Schematic (20 Percent)
MA0201-1902	Rack Process Specification (20 Percent)
MA0201-1384	Rack Process Specification (20 Percent)

One new panel process specification is required.

C7-801 rack modifications are required because of deletion of the switch selector commands, recirculation system, helium injection system, and start tank system. The new additions are for switch selector commands, engine mainstage operation commands, and solid propellant turbine spinner (SPTS) system and engine ready and engine ignition detection signals.

Changes to the C7-802 remote static firing "A" rack include 17 relay function additions and 31 relay function deactivations. These are accomplished by approximately 100 patchboard deletions and approximately 75 patchcord additions.

G7-981677	Rack schematic (20 percent)
G7-871393	Program board assembly
G7-871394	Program board assembly
MA0201-1904-001	Rack process specification (20 percent)

These changes are required because of deletion of the recirculation system, helium injection system, and start tank system. The additions are caused by new engine mainstage operation commands and engine relay isolation test commands.

The C7-805 engine cutoff rack is modified by inactivating the five drawers for monitoring the gas generator temperature.

G7-981678	Rack Schematic

Functional test specification affected for C7-801, C7-802 and C7-805 is:

MA0701-1014-212	Static Firing Countdown (Test Stand A2)

7. Cable and J-Box Requirements - The cabling and J-box installations provide multiconductors for the interconnection and the transmission of electrical power and signals between the S-II stage and GSE, and between GSE end items. The installed cabling is capable of sustaining the maximum load requirements during any checkout phase, protecting where necessary circuits with fuses or resistors and isolating rack interconnections with diodes. MTF cabling, in addition to the above, has the capability of sustaining load conditions of a static firing.

Cabling change requirements are made for all four sites (two at Seal Beach and two for MTF). In practice, however, only one site will be modified at each location. The cabling information gives all necessary information for choosing the respective sites.

(a) Acceptance Stand No. 1 Cable Installation (C7-35), Acceptance Stand No. 2 Cable Installation (C7-40), MTF Firing Control Center Cable Installation (C7-38) Design Changes. Changes to these installations require revision of the terminal distribution rack wire lists in the control centers and test stands. Changes to cabling and terminal distribution racks consist of moving jumper wires, and equipment cable wires.

Changes to this equipment are summarized as follows:

C7-35 (Test Stand A2) Move 110 jumper wires, revise 1300 terminations

C7-40 (Test Stand A1) Move 200 jumper wires, revise 860 terminations

C7-38 (C2) Move 55 jumper wires, revise 560 terminations

C7-38 (C1) Move 160 jumper wires, revise 340 terminations

Documentation affected by the changes is as follows:

C7-35 (Test Stand A2)

F02-000208	Field Site Installation - Cable Assembly
G7-850035	Cable Set (C7-35)
G7-879081	Wire Harness, Terminal Distribution Rack
G7-980340	Interconnecting Diagram

C7-38 (C1/C2)

F02-000056 (C1)	Field Site Installation - Cable Assembly
F02-000256 (C2)	Field Site Installation - Cable Assembly
G7-850038 (C1/C2)	Cable Set (C7-38)
G7-856791 (C1)	Wire Harness, Terminal Distribution Rack (Jumpers)
G7-871216 (C1)	Wire Harness, Terminal Distribution Rack (Cables)
G7-879072 (C2)	Wire Harness, Terminal Distribution Rack
G7-985104 (C2)	Interconnecting Diagram
G7-985604 (C1)	Interconnecting Diagram

C7-40 (Test Stand A1)

F02-000108	Field Site Installation - Cable Assembly
G7-850040	Cable Set (C7-40)
G7-855256	Wire Harness, Terminal Distribution Rack (Cables)
G7-856792	Wire Harness, Terminal Distribution Rack (Jumpers)
G7-985105	Interconnecting Diagram

(b) Electrical Terminal Distributor Stations VIII and IX (SDD 154), Cable Installation, Station VIII (SDD 196), Cable Installation, Station IX (SDD 197) Changes. Changes to the SDD cabling consist of moving jumper wires, revising equipment cable wires, fuses, and diodes.

These SDD cabling changes are summarized as follows:

SDD 154 (Station VIII): Move 131 jumpers, revise 338 terminations, add 16 fuses

SDD 154 (Station IX): Move 131 jumpers, revise 338 terminations, add 16 fuses

SDD 196 (Station VIII): Move 160 jumpers, revise 100 terminations

SDD 197 (Station IX): Move 160 jumpers, revise 100 terminations

Affected documentation includes:

<u>SDD 154</u>

G7-875215	(Station VIII) Wire Harness, Terminal Distributor, JB1
G7-875225	(Station IX) Wire Harness, Terminal Distributor, JC1

<u>SDD 196 (Station VIII)</u>

G7-875218	Wire Harness, Terminal Distributor, JB4
G7-875244	Drag-on Junction Box Assembly Drawing

<u>SDD 197 (Station IX)</u>

G7-875228	Wire Harness, Terminal Distributor, JC4
G7-875244	Drag-on Junction Box Assembly Drawing

(c) Instrumentation Drag-On Cables A2 (SDD 345), Instrumentation Drag-On Cables A1 (SDD 346), Changes. Changes to the instrumentation drag-on cables at MTF consist of the deactivation of existing measurements and the addition of new measurements which affect cable routing and receptacle box harnesses. For each of SDD 345 and SDD 346 cable sets, there are five cable assembly revisions, eight harness assemblies changed, and 10 cables deleted. Changes to the SDD 345 and SDD 346 affect the following documentation:

F02-000123	Field Site Installation - Cable Sets (SDD 346)
S02-000223	Field Site Installation - Cable Assembly (SDD 345)

(d) Stage Station Test Electrical Harness (C7-43), Field Site Installation of Stage Mounted GSE Changes. Changes to the C7-43 consist of revising the cable sets. This entails the deletion of six cables from each of the eight cable sets, in support of deletion of the recirculation system.

Changes to the field site installations of stage-mounted GSE consist of revising drawings showing installation of carry-on instrumentation cables and drag-on instrumentation cables to show new mountings, brackets, cable clamp locations, and drilling details of Saturn J-2S instrumentation. These drawings affect MTF, KSC, and the C7-43.

Documentation affected by these changes includes:

Carry-On Cables

F02-000416 (Test Stand A2)	Field Site Installation - Cable Assembly
F02-000420 (Test Stand A1)	Field Site Installation - Cable Assembly
F02-000421 (KSC)	Field Site Installation - Cable Assembly
F02-000465 (Test Stand A2)	Field Site Installation - Cable Assembly
F02-000467 (Test Stand A1)	Field Site Installation - Cable Assembly

C7-43

G7-850043	Cable Set Installation Assembly
G7-870241	Cable Set Installation Assembly
G7-871004	Cable Set Installation Assembly
G7-987173	Cable Set Installation Harness Assembly
V7-760900	Cable Installation - Stage-Mounted

(e) Power Distribution System, Test Stand A2 (C7-80), Power Distribution System, Test Stand A1 (C7-81) Changes. The power distribution systems for test stands A2 and A1 will be revised to show deactivation of the 56-vdc recirculation system and the engine ignition battery simulator.

8. Other GSE Items and Related Changes.

(a) Digital Events Recorder (C7-77). The digital events recorder detects and records changes of state

(ON = $+8\,^{+26}_{-0}$, OFF = $0\,^{+7.5}_{-0}$) on the S-II discrete command and measurement lines. When a change is detected, the state to which it changes, the time of the change to the nearest millisecond, and an identifying number are recorded on a hard copy printout and punched paper tape or magnetic tape (MTF only). There are no hardware changes to this equipment, only revision to J-box/TDR input lines and a revision to the interface control document (ICD). The ICD is used to correlate the identifying number with the discrete on the input line. Revision of the ICD's is necessary to delete 48 functions, add 64 functions (using the spares) and revise 49 functions.

(b) Remote Distribution Rack (C7-41). The C7-41 remote distribution rack is used to monitor and transfer the power from facility power to GSE power, from GSE power to GSE power, and from GSE power to stage power. It is provided with interrupt circuits and interlocks to protect all stage and GSE buses. Provision is made for isolating the C7-200 station, the C7-800 station and the C7-603 pneumatic checkout console set during testing and servicing at the static firing sites (MTF) and at Seal Beach. The changes to the C7-41 Rack are deactivation of the entire 56 vdc recirculation power system and the electrical control, sensing and power transfer system for the 56 vdc rectifier; deactivation of the electrical control and power switching for the stage recirculation bus and battery simulator circuits; and deactivation of the engine ignition load bank and engine ignition battery simulator.

MTF, in addition to the above changes, requires deactivation of the isolation valve control and summary indicating circuits. Indicators on the isolation valve drawers are changed to show spare.

C7-41 documentation affected by LOR mission requirements includes:

G7-987011	Rack Schematic
G7-987012	Rack Schematic
MA0201-1595	Functional Testing Rack Process Specification
MA0201-1586	Functional Testing Rack Process Specification
MA0201-1581	Functional Testing Rack Process Specification
MA0201-1582	Functional Testing Rack Process Specification
MA0201-1583	Functional Testing Rack Process Specification
MA0201-1609	Functional Testing Rack Process Specification
MA0201-1611	Functional Testing Rack Process Specification
MA0201-1615	Functional Testing Rack Process Specification
MA0701-1014-212	Static Firing Countdown (Test Stand A2)
MA0701-1027-111	ACE Activation (Station VIII or IX)

The C7-41 changes are due to deletion of the recirculation system. The LOR mission does not use a 56 vdc battery since the recirculation system has been removed, but this battery system can be used for the thrust vector control system (TVCS) of the LEO mission. If a C7-41 hardware change is made for both the LOR mission and LEO mission simultaneously, the hardware changes can be minimized.

(c) Time Code Rack (C7-48). The C7-48 time code rack provides the time code signals for the various systems requiring a synchronous timing signal: strip chart recorders, tape recorders, local and remote time displays, C7-101 computer and C7-77 digital events recorder. This serves as the master time synchronization throughout the GSE computers checkout complex. There are no changes to this equipment because of the J-2S engine modification.

(d) Ground Equipment Test Set (C7-44). The ground equipment test set (GETS) is used to verify the functional readiness of the ground support equipment checkout stations. It receives the electrical stimuli at the stage umbilical connectors (like the stage) from the GSE, processes these stimuli, and sends back responses to the GSE. Signal acceptance and response generation is such that the electrical functional characteristics of the stage checkout equipment are verified for proper operation. The equipment has the capability of processing analog and digital signals and encoded discrete commands that simulate the instrument unit (IU). GETS performs hazardous condition simulation, hazardous monitor switching, flight control simulation, decoder switching, GETS control switching, stage systems simulation, time delay patching, electrical power operation, and audio communication network operations. Changes to the C7-44 will require the addition of two new integrated program board assemblies, the addition of two new static firing program board assemblies, the revision of the self-test program boards and a revision of the IU command decoder for the addition of seven new switch selector commands and deletion of two switch selector functions. Patchboard rework will require approximately 1000 new patchcords.

Documentation affected by the C7-44 changes includes the following:

MA0201-1776	GETS IU Decoder Process Specification (20 Percent)
MA0201-1953	GETS Functional Test Process Specification (20 Percent)
MA0701-1014-212	Static Firing Countdown (Test Stand A2)

(e) Engine Sequence Recorder (SDD 273) Power Supply 56 vdc (SDD 337) (Stations VIII and IX only), S-II Ordnance. The SDD 273 oscillograph recorder is used to monitor the engine valve positions during engine sequence testing. These recorders are used in lieu of the C7-205 special data rack for engine valve analog measurements.

The SDD 337-vdc power supply provides the electric power to operate the five LH_2 recirculation pump motors during checkout and simulated prelaunch operations. The SDD 273 recorder requires a change in documentation for the reidentification of new engine function assignments on each channel. The SDD 337-56-vdc power supply will be disconnected and documented as spare on the power distribution drawings and system schematics. The S-H ordnance requires only changes in documentation due to addition of the engine SPTS system. Two new specifications are required:

EBW Initiator Electrical Checkout Specification

EBW Initiator Installation and Arming Specification

There are two revised process specifications:

MA0301-1004	Live Ordnance Test Specifications
MA0301-1005	Installation and Arming Specification

b. GSE Operational Specification Changes

1. MA0701-1012-111, C7-200 Station Manual Checkout, (Station VIII or IX), affects C7-202, C7-204, C7-209, C7-211, C7-212, and C7-213. The amount of change is approximately 20 percent. The specification is easier to run, because of deletion of 56-vdc power supply switching test (recirculation system), addition and deletion of switch selector cards, and addition and deletion of C7-204 commands controlled by C7-209 and C7-202.

2. MA0701-1014-212, Static Firing Countdown, (Test Stands A1 and A2), affects C7-41, C7-44, C7-204, C7-211, C7-212, C7-123, C7-801, C7-802, and C7-805. Total change is approximately 30 percent. The specification has the same ease of performance as before, because of addition and deletion of engine system functions (including recirculation system), changes in switch selector functions, and changes in the automatic sequence test.

3. MA0701-1027-111 AGE/GSE Activation, Stations VIII, IX, affects C7-41, with a total change of approximately 5 percent. The specification is essentially easier to run than before, because of deactivation of 56 vdc power supply system of C7-41.

4. MA0705-1015-111, Semiautomatic Activation C7-100 Stations (VIII and IX), affects C7-102 hazardous monitor. The total change is approximately 5 percent, and testing is easier since there are fewer hazards to check (deletion of recirculation power and pressurization conditions).

10.2.7.3.2 ACE System Requirements - LEO Mission

The Saturn S-II stage automatic checkout system has been designed as a group of checkout stations controlled by a computer complex, and will be the same for the LEO mission.

a. Checkout Station Requirements. The automatic checkout system for the J-2S engine implementation consists of a controlling computer complex linked to a number of specialized checkout stations. This concept will remain unchanged for the LEO mission.

1. Computer Complex Checkout Station (C7-100). The only changes to the C7-100 computer complex end items resulting from J2-S engine implementation are to the C7-102 test conductor console. The following functions are added to the hazardous monitor panel:

   Thrust Vector Control System DC greater than 64v.

   Thrust Vector Control System DC greater than 60v.

   Thrust Vector Control System DC low.

   These changes require addition of legend plates and cabling in the J-boxes that connect the hazardous functions to the C7-102. Cabling changes can be minimized by combining LOR and LEO mission changes in this area. The following documentation is affected by this change:

   MA0201-1048 Hazardous Monitor Process Specification

   MA0705-1015-111 Semiautomatic Checkout C7-100, Stations VIII and IX

   The principal software items affected by electrical GSE changes are same as for the LOR mission.

2. Electrical Checkout Station (C7-200). The C7-202 manual control and display rack changes consist of adding a new switch selector control panel to provide for the nine new switch selector commands required, and adding 17 new commands using an existing spare panel assigned for static firing 2. The latter commands are implemented by changing nomenclature on the panel.

   The following C7-202 rack documentation is affected by the above changes:

G7-852600	Rack Assembly
MA0201-1235	Switch Selector Display, C7-202 Rack

New drawings are required for the new switch selector control panel, plus applicable rack harnesses.

The C7-204 signal distribution rack changes consist of adding 17 each of flip-flops, logic gates, relays and relay drivers, plus the necessary patching (34 patchcords) to implement these new commands. Documentation affected by these changes is as follows:

G7-852200	Rack Assembly
G7-852300	Control Logic No. 1
G7-852640	Command Distributor Relay Logic No. 1
G7-857100	Rack Schematic

These modifications also require 4 new rack and drawer harnesses and a new patch panel assembly. The C7-211 scanner rack changes are due to approximately 27 added functions to be monitored. This is accomplished by adding 49 patchcords to the A3 patch panel and 29 patchcords to the A17 patch panel. Documentation affected:

G7-855810	Rack Assembly
G7-981683	Rack Schematic
G7-981684	Rack Schematic
MA0201-1532-001	Process Specification
MA0201-1532-002	Process Specification

Also affected are the two new program board assemblies (A3 and A17) which were added for the LOR mission. The C7-212 discrete display rack changes require the addition of approximately 12 legend lights to display new functions being monitored. These are implemented by the addition of 24 patchcords. Documentation affected is as follows:

G7-855870	Rack Assembly
G7-857047	Program Board Assembly
G7-857048	Program Board Assembly
G7-981688	Rack Schematic
G7-981689	Rack Schematic
MA0201-1535-001	Process Specification
MA0201-1535-002	Process Specification

Also affected are the legend plate drawing and the display panel concerned.

The C7-213 interlock relay rack changes due to the LEO mission have not yet been determined. It is presently assumed that a maximum of 20 new functions may need to be interlocked. Based on this assumption, approximately 60 patchcords would be added to implement existing spare relays for these functions. Documentation affected by this change is as follows:

G7-855850	Rack Assembly
G7-981679	Rack Schematic
G7-981682	Rack Schematic
MA0201-1786-001	Process Specification
MA0201-1786-002	Process Specification

Also affected are the two new program board assemblies (A4 and A15) which were added for the LOR mission. Functional test documentation affected by LEO mission changes to the C7-200 Station is as follows:

MA0701-1012-111	Manual checkout, C7-200 Stations VIII and IX
MA0701-1014-212	Static Firing Checkout

There will be no changes to the following C7-200 Station racks for the LEO mission:

C7-201	Automatic Control Rack
C7-205	Special Data Rack
C7-208	Station Control and Display Rack
C7-209	Manual Control and Display Rack
C7-210	Stage Substitutes Rack

3. Range Safety Command Receiver (RSCR) Checkout Station (C7-307). There are no changes required for the C7-307 rack resulting from J2-S engine incorporation.

4. Digital Data Acquisition System (DDAS) Checkout Station (C7-400). No changes are required for the C7-400 station because of J2-S engine incorporation.

5. Telemetry System Checkout Station (C7-500). No changes to the C7-500 station are required for J2-S engine incorporation.

6. Static Firing Control Station (C7-800). Changes to the C7-801 local static firing rack consist of adding 9 new switch selector commands, 11 new switch

commands and 17 legend plates to spare indicator lights. The new switch selector commands will require the addition of 3 diode boards and wire harness changes to the switch selector encoder drawer, and 9 legend plates to existing spare switchlights on the switch selector panel. The 11 new switch commands will utilize spare switches on the new panel added for the LOR mission. Five of these will have associated indicator lights. The remaining 12 indicator lights will be located on various other panels of the C7-801, depending on the system involved and availability of spares. The documentation affected is as follow:

G7-852800	Rack Assembly
G7-872715	Switch Selector Encoder
G7-856835	Switch Selector
G7-981676	Rack Schematic
MA0201-1902-001	Process Specification

The C7-802 changes consist of adding patchcords to implement the nine switch selector commands of the C7-801, and to connect 17 existing spare relays into the circuits required. This will require approximately 60 additional patchcords. Documentation affected is as follows:

G7-871393	Program Board Assembly
G7-871394	Program Board Assembly
G7-981677	Rack Schematic
MA0201-1904-001	Process Specification

The C7-805 engine cutoff rack is not affected by LEO mission requirements. The functional test specification affected by C7-801 and C7-802 changes is:

MA0701-1014-210	Static Firing Countdown (Test Stand A2)

7. Cable and J-Box Requirements. The cabling and J-box installations provide multiconductors for the interconnection and the transmission of electrical power and signals between the S-II stage and GSE, and between GSE end items. The installed cabling is capable of sustaining the maximum load requirements during any checkout phase, protecting (as necessary) circuits with fuses or resistors and isolating rack interconnections with diodes. The same capability will be retained for the LEO mission.

(a) MTF J-Box and Cable Installation (C7-35, C7-38, and C7-40). Changes to the MTF cable sets require additional revisions to the terminal distribution rack wire lists for the control centers and test stands. Changes are summarized as follows.

C7-38 (Control Center)

Add 75 jumper wires

C7-35 and C7-40 (Service Center, Test Stand A2 or A1)

Add 250 jumper wires

Add 1 new cable assembly

Add 1 new umbilical cable and adapter

Add 6 drag-on cables

Add 3 stage mounted GSE harnesses

Add 7 receptacle/harness assemblies

Documentation affected by the above changes will be the same as for the LOR mission, plus new drawings required for approximately 19 new cables.

(b) Seal Beach J-Box and Cable Installation (SDD-154, SDD-196, SDD-197). Changes to the Seal Beach cable sets consist of the following:

Add 6 new drag-on cables

Add 1 new umbilical adapter and GETS cable

Add 2 new cables for battery simulation

Add 8 receptacles in junction box

Add 1 conduit hub to junction box

Add and/or change 50 fuses

Change 200 wire terminations

Documentation affected will be same as for the LOR mission, plus new drawings for the added requirements.

(c) Stage Station Test Electrical Harness (C7-43) Field Site Installation of Stage-Mounted GSE. These changes require revision to stage mounted GSE drawings at Seal Beach, MTF, and KSC to show new carry-on and drag-on cable clamp support locations, in addition to revision to the GSE static firing cable sets.

The following documentation is affected:

V7-760900	C7-43 (Seal Beach) Cable Installation
F02-000421	C7-92 (KSC) Cable Installation
G7-850043	C7-43 Assembly
G7-850692	C7-92 Assembly
G7-850693	Harness Assembly
G7-850694	Harness Assembly

New drawings are required for the added cable and harness assemblies.

(d) Power Distribution System, Test Stand A2 (C7-80), Test Stand A1 (C7-81). The 56 vdc circuit required for the thrust vector control system will require addition of a 56 vdc panel circuit with circuit breaker and a battery simulator circuit. With relatively minor modification, the 56 vdc recirculation system which was deactivated for the LOR mission can be used for this function.

8. Other GSE Items

(a) Digital Events Recorder (C7-77). There are no hardware changes required for this equipment. Revision to ICD's is needed to add 21 new channels at Seal Beach and 27 new functions at MTF. These are all existing spares.

(b) Remote Distribution Rack (C7-41). As stated above, the LOR mission deactivated the 56 vdc distribution system used for the recirculation bus. The LEO mission requires two 56-vdc systems for the thrust vector control hydraulic system. This will be accomplished by using the existing 56 vdc power system as TVC bus. This existing circuit fulfills one of the requirements. A complete new circuit will be added requiring 1 circuit breaker, 1 shunt, a relay, and interface connections.

The existing capabilities of the main and instrumentation power circuits are sufficient to accommodate the anticipated 60- to 80-ampere loads.

Since the spare capability of the C7-41 has been filled, an additional panel will be required to accommodate the above hardware, and an interface connection and new harness will be required.

Documentation affected includes the following:

Rack Schematics

G7-987011

G7-987012

G7-981938

G7-985749

G7-981939

G7-985748

Process Specifications

MA0201-1586

MA0201-1595

MA0201-1609

Circuit Status Panel

G7-850306

New drawings are required for new panel, interface connection and harness, revised interface plate, and assembly drawing.

(c) Time Code Rack (C7-48). There are no changes to this rack resulting from the LEO mission.

(d) Ground Equipment Test Set (C7-44). The changes to the C7-44 will consist of adding approximately 32 patchcords to each of two patch panels and revising one wire harness in the IU recorder drawer (approximately 12 wires).

Documentation affected is as follows:

MA0201-1776	GETS IU Decoder Process Specification
MA0201-1953	GETS Functional Test Process Specification

MA0701-1013-211 Static Firing Countdown (Test Stand A1)

MA0701-1014-210 Static Firing Countdown (Test Stand A2)

The two new patch panels added for the LOR mission will also be affected.

(e) Engine Sequence Recorder (SDD 273), Power Supply 56 vdc (SDD 337). There are no changes to this equipment resulting from the LEO mission.

b. GSE Operational Specifications

1. GSE Operational Specification Changes for LEO Mission.

(a) MA0701-1012-111, C7-200 station manual checkout (Stations VIII and IX,) affects C7-41, C7-202, C7-204, C7-211, C7-212, C7-213, with the total change approximately 10 percent. The specification reverts to essentially the same ease of performing as before LOR mission changes, because of reinstatement of 56-vdc power switching test for thrust vector control system (TVCS), and addition of reaction control system (RCS). Additional command monitoring, recording, and response monitoring are required due to nine new switch selector commands. The addition of LH_2 balanced vent valve system requires additional command and response monitoring by GSE.

(b) MA0701-1013-211, Static Firing Countdown (Test Stand A1), and MA0701-1014-210, Static Firing Countdown (Test Stand A2), affect C7-41, C7-44, C7-77, C7-102, C7-208, C7-211, C7-212, C7-213, C7-801, and C7-802. The total change is approximately 15 percent. Specification ease of performance will be essentially the same as before LOR mission changes, for reasons stated above.

(c) MA0701-1027-111, ACE/GSE Activation, C7-200, Stations VIII and IX, affects C7-41. The total change is approximately 5 percent. Specification ease of performance is essentially the same as before the LOR mission, due to reactivation of 56 vdc power supply for TVCS.

(d) MA0705-1015-111, Semiautomatic Activation, C7-100, Stations VIII and IX, affects C7-102 hazardous monitor. The total change is approximately 5 percent. Testing is essentially the same as before LOR mission, due to addition of TCVS monitoring functions.

10.2.8 INTEGRATED SYSTEM TESTS

The primary objective of this test program is to develop S-II Stage Systems as modified to incorporate the J-2S engine for a LOR or LEO mission to ensure a highly reliable man-rated stage that meets all operating and performance requirements of the design specification. This test program entails a minimal amount of design verification testing due to the extensive applicable experience gained in development of the present S-II stage.

Testing is to be accomplished on the S-II Battleship Stand (COCA 1) at the Santa Susana Field Laboratory (SSFL) and on the first two production vehicles during acceptance testing at the Mississippi Test Facility (MTF). A certain versatility is present should an "all systems" vehicle be made available, which would then replace the Battleship, however, test plans as described in this report assume that the Battleship facility will be employed. Structural testing and base heat testing (exhaust plume) will be accomplished on NASA facilities at MSFC.

An overall summary of the test program is shown in Figure 10.2-168.

10.2.8.1 LOR Mission

10.2.8.1.1 Structural Testing

Changes to the S-II structure to accommodate the increased thrust level and base heat flux of the J-2S engine will require the accomplishment of design verification tests to prove the integrity of the modified design. In particular, the revised thrust structure/aft skirt and heat shield will be tested due to the addition of the J-2S. A more extensive structure test would be required for the PO mission. A series of model tests will be conducted to determine thermal effects of the J-2S plume on the S-II.

a. Thrust Structure/Aft Skirt Tests. The thrust structure is the primary load carrying system for transmitting engine loads to the vehicle. The critical environments for the thrust structure include compression, shear and bending loads during J-2S/S-II operation, tensile loading during S-IC operation, and elevated temperature gradients during J-2S/S-II operation. Additional local loading is imposed by propellant lines and equipment mountings.

   The aft skirt is the load carrying system for transmitting axial loads and bending moments during S-IC boost flight and S-II thrust loads during J-2S/S-II operation. The design environment for the aft skirt consists of the applied structural loads, thermal gradients and local loading due to umbilical loads applied during pre-launch and launch.

   1. Test Objectives. The primary objectives of this test series are to verify the structural integrity of the thrust structure/aft skirt assembly when subjected to the static loads, thermal environment, local loads and edge conditions of the flight installation and to determine stress and deflection influences of individual engine and actuator loads. (Precant angle for engine installation is a secondary function of this objective.)

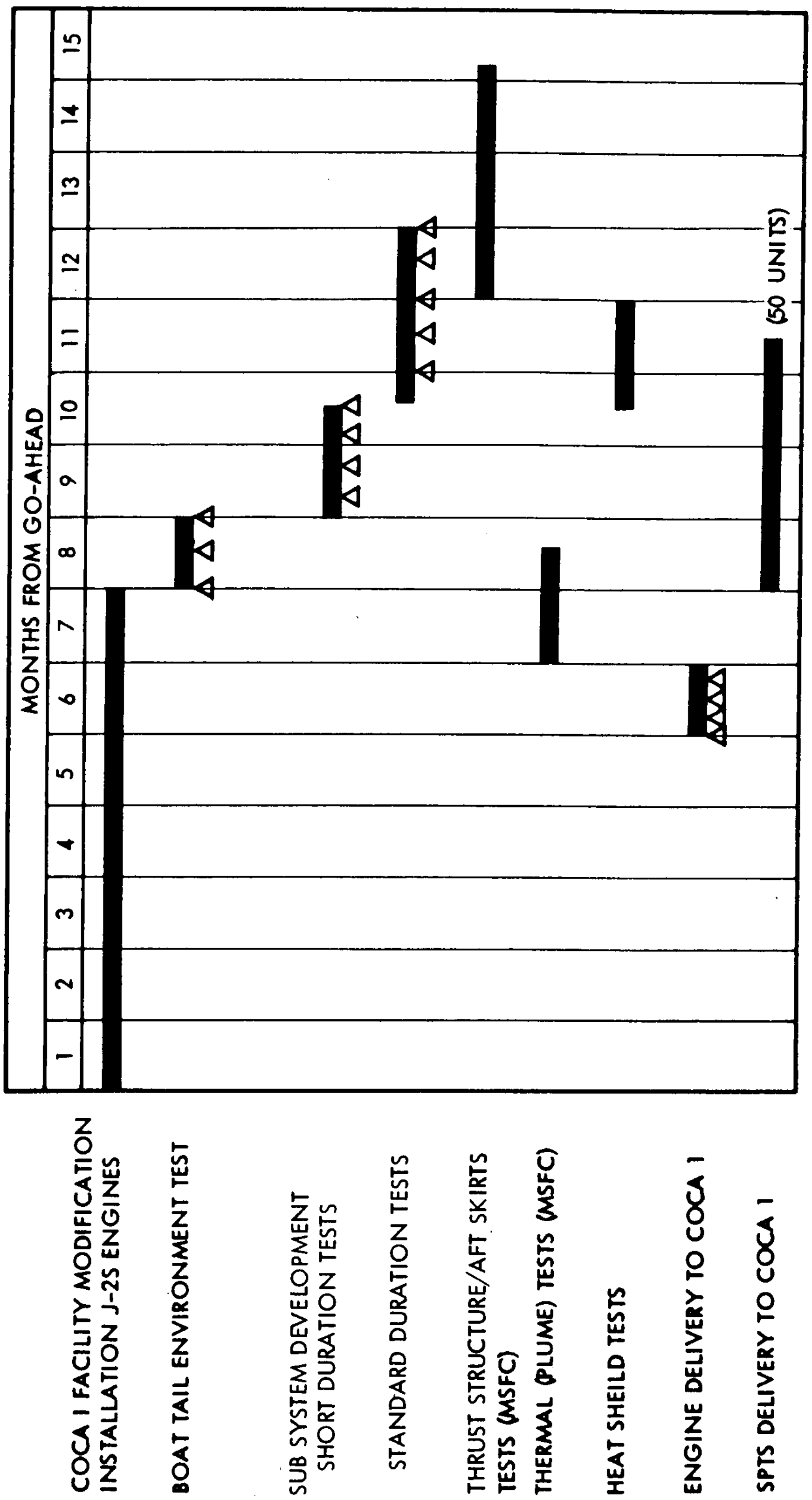


Figure 10.2-168. J-2S Implementation Integrated System Tests

2. Test Setup. The structural tests of the thrust structure/aft skirt will be conducted at the NASA R-P&VE Laboratory Test facilities at MSFC. The test specimen will be a structurally complete S-II stage thrust structure and aft skirt assembly. Additional equipment required to conduct this test series, assumed to be available at MSFC from previous testing includes the following:

   (a) Dummy S-II interstage

   (b) A load ring to be attached to the forward end of the aft skirt

   (c) Fixturing and loading devices to apply various point loads

   (d) Thermal control equipment to simulate heat inputs and outputs

3. Test Conditions. The thrust structure/aft skirt shall be subjected to the following load conditions while installed in the R-P&VE test stand:

   (a) Individual applied loads (thrust, umbilica, etc.) to determine influence coefficients for single loading parameters on the stress and deflection characteristics of the various structural elements.

   (b) Application of limit loads in conjunction with a thermal environment to verify analytical determination of structural stress and strain under the following load conditions:

      Limit thrust - all engines - 0 degree gimbal

      Limit thrust - all engines - 6 degree gimbal

      Limit thrust - one engine out - full gimbal

   (c) Application of ultimate loads to verify structural integrity of the thrust structure/aft skirt assembly under the following critical conditions:

      Maximum thrust - all engines - 0 degree gimbal (no heat)

      Maximum thrust - all engines - 0 degree gimbal (with heat)

      Maximum thrust - all engines - 6 degree gimbal (no heat)

      Maximum thrust - one engine out - full gimbal (no heat)

      Maximum thrust - one engine out - full gimbal (with heat)

4. Special Instrumentation. There is no special instrumentation requirement beyond those used during previous thrust structure/aft skirt testing to determine stress, deflection, and temperature values.

b. Heat Shield Tests. The heat shield has the function of protecting the J-2S engines and the S-II afterbody from hot exhaust gas recirculation and exhaust plume radiation.

1. Test Objectives. The objective of this test series is to verify the structural and thermal integrity of the rigid portion of the base heat shield when subjected to the design base environment. The most critical condition is that represented by one engine out.

2. Test Setup. The heat shield testing will be conducted at NR laboratory and test facilities.

   The test specimen for structural verification will be a flight design quarter panel heat shield. Other test equipment shall include fixtures, heating lamps, vibrators, and vacuum facility similar to those used during original S-II heat shield testing.

   A primary change from previous testing will be that the test panel orientation is to be horizontal rather than vertical.

   The test specimens for thermal testing will consist of small panels (12 by 12 inches) utilizing the flight design fabrication techniques.

3. Test Conditions. The heat shield quarter panel while mounted to a suitable vibration fixture and an enclosure which maintains the hot side of the panel in an inert atmosphere (2 percent maximum oxygen content) will be subjected to the sinusoidal vibration spectrum performed at room temperature and the random vibration spectrum performed at room and elevated temperatures. The detailed pressure, temperature, and vibration values will be those established for the J-2S engine. The small panel tests will be performed in a vacuum chamber simulating S-II operating altitudes and temperatures. The capability of the thermal side of the panel to exhaust the internal pressure will be monitored.

4. Special Instrumentation. There is no special instrumentation requirement over those utilized during previous heat shield testing.

c. Dynamic/Acoustic Tests. Based upon engine contractor information that the acoustic levels generated by the J-2S engine are no more severe than those generated by the J-2, it is not planned to perform any special tests for this parameter. Should a change in the acoustic environment be noted during engine testing, this test plan may be revised to add tests on the first production article to verify stage integrity with the new values.

10.2.8.1.2 Battleship Testing

To conduct the development phases of the J-2S implementation program, the Battleship stand (COCA 1) at the Santa Susana Field Laboratory will be utilized. All of the following described tests may be run on an All Systems Vehicle at MTF instead of the Battleship should it be made available.

The Battleship test stand affords a useful tool that permits installation and tests of the various subsystem changes required for J-2S implementation. A capability exists with the Battleship stand to withstand a malfunction with a lesser cost and schedule impact on the program than that provided by an All Systems Vehicle.

a. Boattail Environment Tests. To ascertain propellant conditions available to the engine at start command, a series of boattail environment tests will be conducted. This test series is planned early in the test program to permit use of the information in programming single J-2S engine tests at Rocketdyne simulating the S-II stage environment.

   To reproduce the environment within the engine compartment in the stacked configuration at KSC, a simulated aft interstage enclosure, flight heat shield, and flight engine compartment conditioning system (ECCS) will be installed on the Battleship stand (COCA 1).

   1. Test Objectives. The primary test objectives of this series are determination of propellant conditions within the feed ducts and pump inlets, determination of hydraulic fluid conditions within the hydraulic system, and determination of temperatures within the electrical and instrumentation containers.

   2. Test Procedure. This test series shall be conducted using the prelaunch sequence with a minimum amount of changes to fit facility needs. The anticipated prelaunch sequence is shown in Figure 10.2-169.

      Minimum propellant tanking for this series will be 60 percent LOX load and 40 percent LH_2 load. This quantity of propellant will satisfy the requirement for cryogenics in the engine compartment.

   3. Special Instrumentation. This test series does not require special instrumentation beyond that utilized in the previous boottail environment test series.

b. Subsystem Development Tests. To determine and verify various subsystem operating parameters, a series of tests will be performed on the Battleship stand. Although several test objectives may be obtained utilizing a single J-2S engine with four J-2 engines, other objectives require a complete five engine cluster, therefore the minimal program that satisfies all objectives uses the five engine cluster.

   This test series is programmed as four 60-second-duration tests and three full duration tests. A total data evaluation analysis and correlation to previous test results will be performed between tests.

   1. Test Objectives. Each test of the Battleship will have multiple objectives. Certain tests will have specialized objectives. The overall test objectives are listed below:

      (a) Determine the capability of the engine servicing system on the stage to properly service each engine in the cluster.

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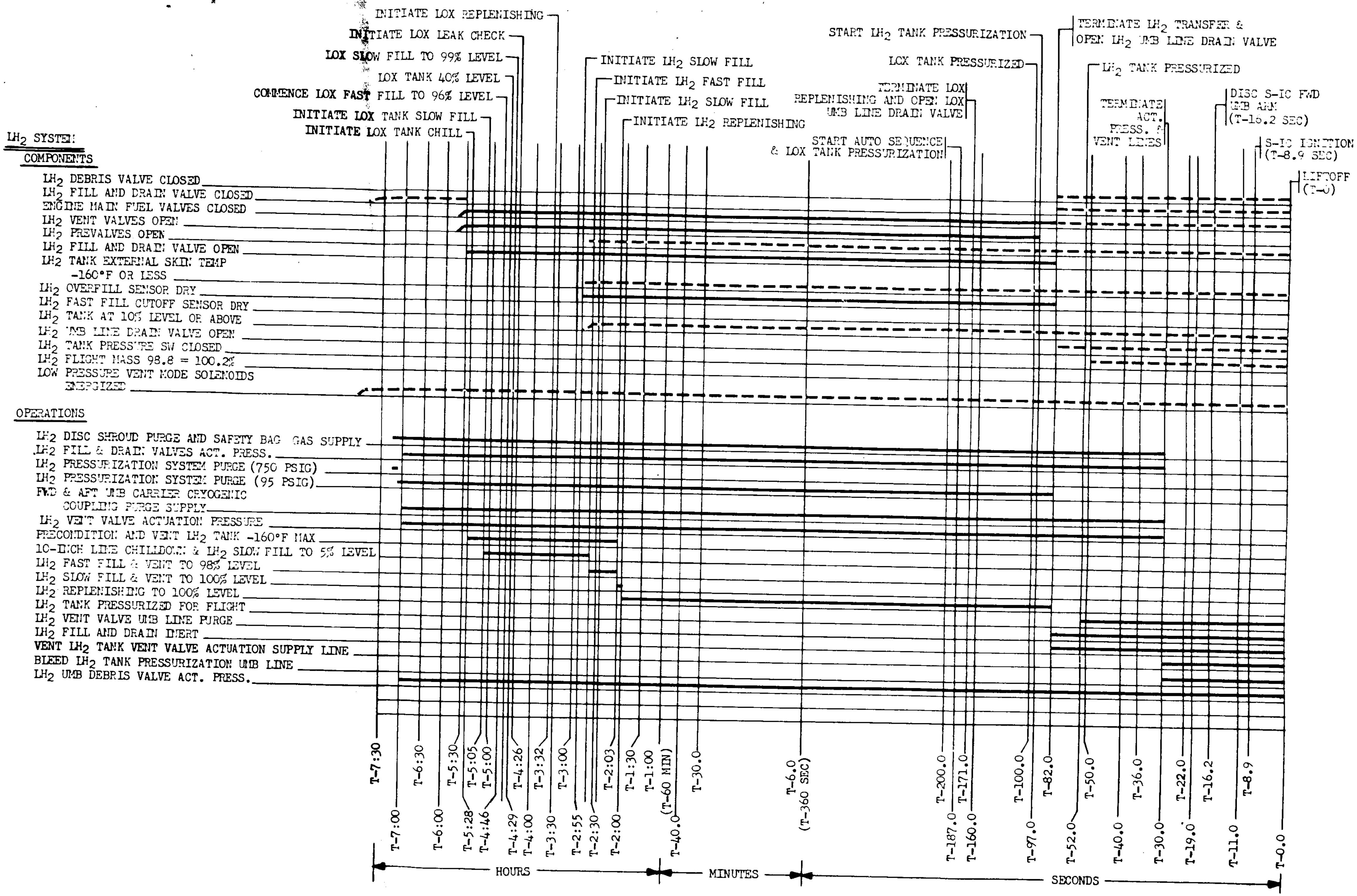

Figure 10.2-169. Prelaunch Sequence--S-II/J-2S (Sheet 1 of 3)

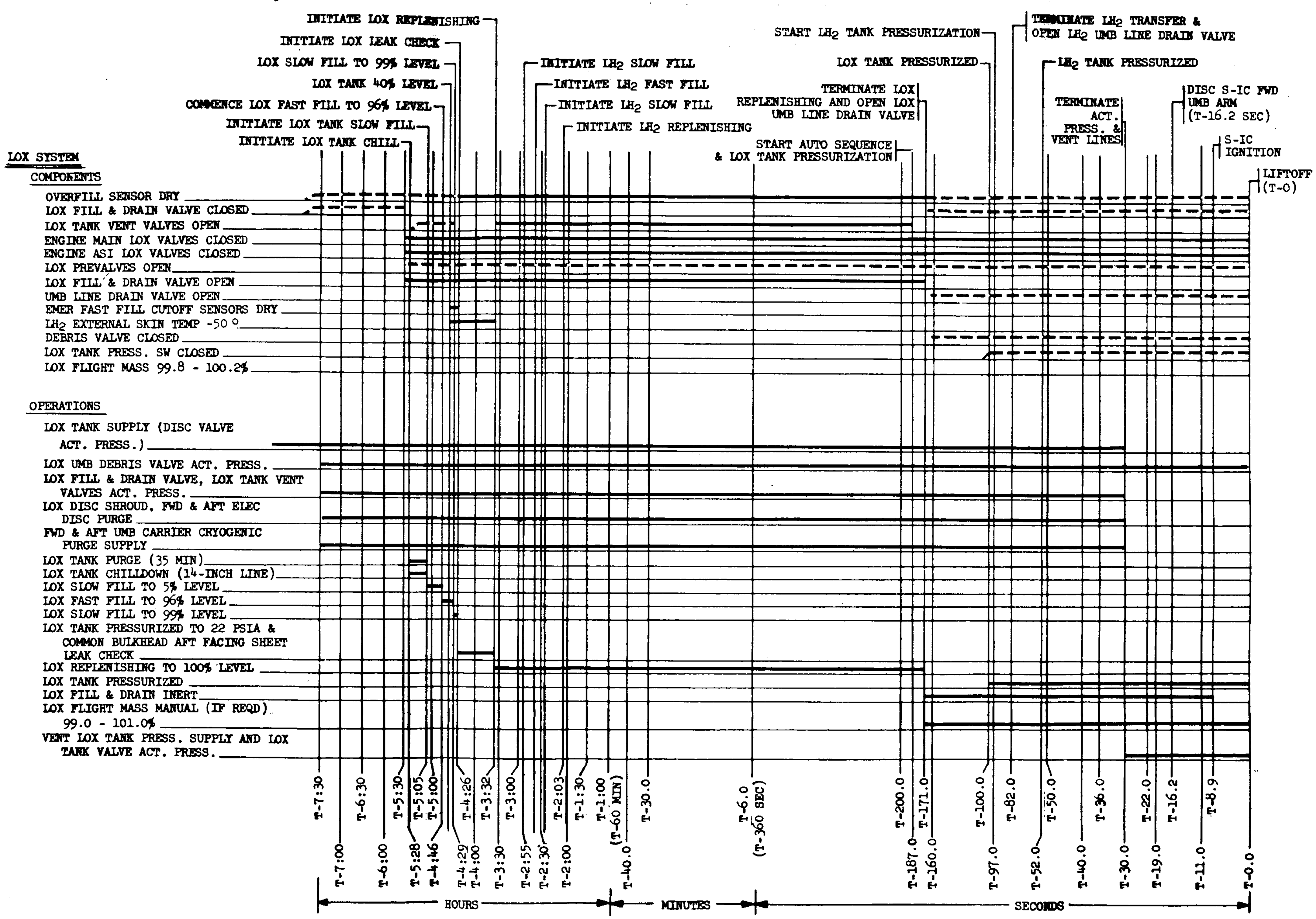

Figure 10.2-169. Prelaunch Sequence—S-II/J-2S (Sheet 2 of 3)

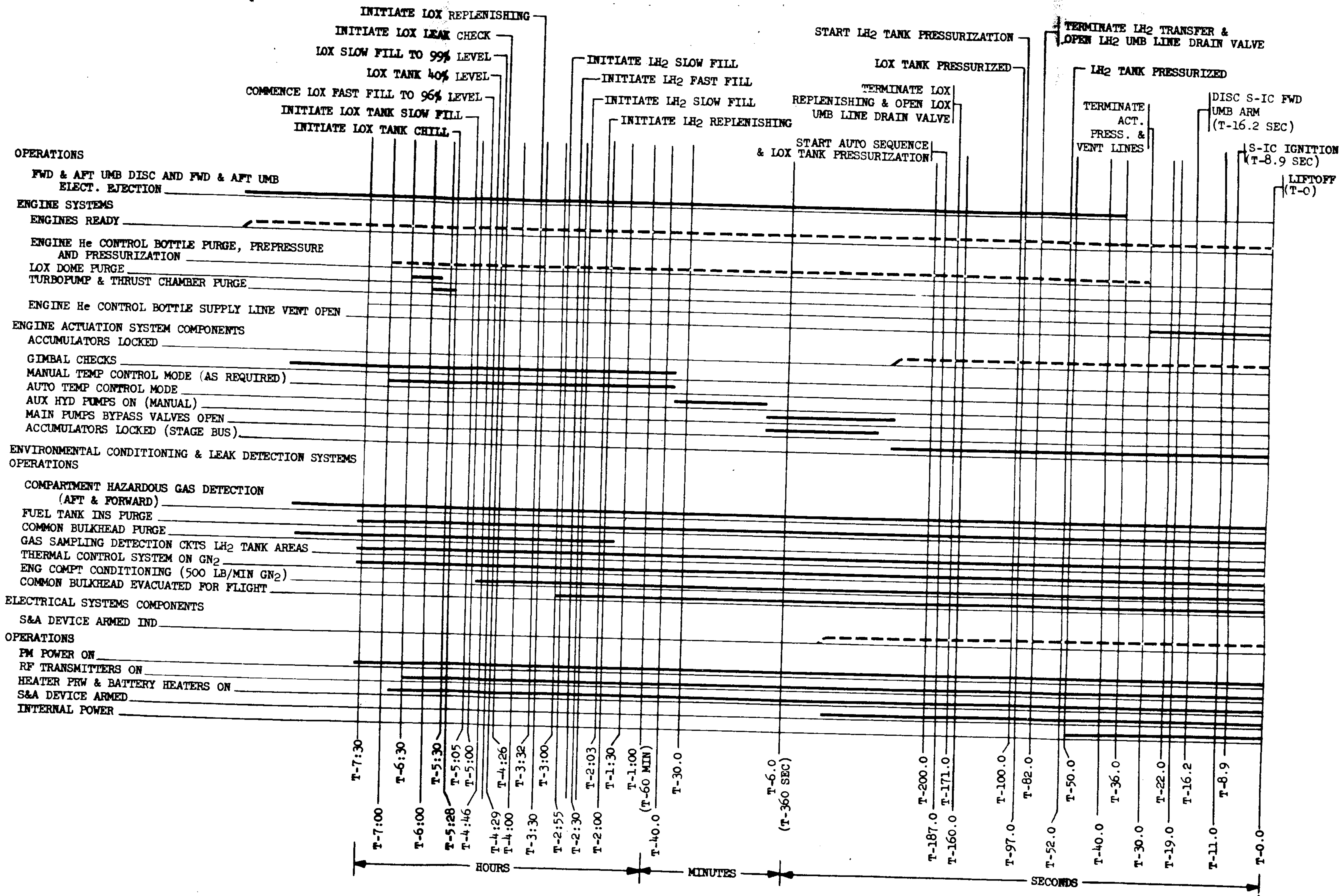

Figure 10.2-169. Prelaunch Sequence—S-II/J-2S (Sheet 3 of 3)

(b) Determine and evaluate propellant conditions at the engine during the starting sequence, and note engine-to-engine variations due to clustering.

(c) Determine and evaluate pressurization system gas conditions and regulator operation

(d) Determine and evaluate cluster environment data

(e) Determine and evaluate the effects of a LOX exhaustion cutoff. The unsymmetrical thrust tail-off associated with this type of cutoff will be determined.

(f) Determine adequacy of the redesigned hydraulic pump to satisfy the engine actuation system requirements of a minimum gimbal rate.

(g) Determine an integrated stage electrical requirement.

2. Test Procedure. This test series will be conducted utilizing the presently developed Battleship firing procedures with modifications to accommodate the requirements of the J-2S engines. The basic nature of the test program and the inherent ease of operation of the Battleship preclude the need for special procedures.

3. Special Instrumentation. To satisfy the test objectives, the following instrumentation will be required in addition to that for a normal Battleship test.

(a) Twenty high response pressure transducers installed at the various tank outlets and engine inlets

(b) Ten temperature transducers installed at the various tank outlets and engine inlets

(c) Approximately fifteen vibration transducers installed at selected points on the thrust structure and engines

4. Data Requirements. The accumulated data from these tests will be presented as detailed engineering reports from the various affected engineering groups. These reports will include performance analysis and a cause and effect analysis for all observed anomalies.

10.2.8.1.3 All Systems Verification Tests

The first two production stages incorporating the J-2S engine will be subjected to a full mission duration test in addition to the normal acceptance test. These tests will serve as the total system integration design verification and will provide the first stage subsystems and GSE integration verification.

a. LOR Mission. Due to the high confidence factor in successful operation based upon the large background of experience and available operating data on the S-II stage, knowledge of the effects each of the present subsystems has upon the others and knowing the nature of the changes to incorporate the J-2S, the integrated design verification tests will be conducted upon the first production stage prior to its acceptance test firings. Data from the acceptance tests will also be utilized in the overall development test evaluation.

1. Test Objectives. The primary objective of this portion of the overall test program is the verification that the stage meets all design parameters and is not adversely affected by incorporation of the J-2S engine.

   Detailed test objectives will be present for each of the stage subsystems that have been affected plus a final determination of the influence of each subsystem upon overall stage operation and performance. These objectives are listed below:

   (a) Verify the countdown sequence and associated activities for S-II static firing and launch with the J-2S engine installed

   (b) Verify compatibility of the revised stage hardware and requirements with the revised GSE

   (c) Verify the stage/GSE capability to service and checkout a J-2S engine cluster properly

   (d) Verify satisfactory engine starting characteristics and determine the thrust increase envelope for the flight stage

   (e) Verify satisfactory mainstage operation with emphasis upon the stage pressurization system performance and thrust structure compliance

   (f) Verify satisfactory hydraulic system operation under flight weight mounting conditions

   (g) Verify satisfactory operation of the revised valve actuation system and determine the operational margin remaining with the retained components

   (h) Verify satisfactory operation of the electrical power and control systems and determination of the power margin available with the redistributed loads

   (i) Verify satisfactory cutoff characteristics including adequacy of the tank screens and baffles, determination of residual propellant quantity, and determination of the stage thrust decay rate and thrust unbalance during cutoff

2. Test Procedure. The test procedures to be used in this test series will be those developed during Battleship testing and detailed in operational specifications for test stand installation and handling, pretest checkout, acceptance testing, and post-test checkout. As on the present S-II static firings, the sequence of events for a KSC launch will be adhered to as closely as possible.

3. Special Instrumentation. The proper verification and evaluation of stage operation will require use of the following special instrumentation on this stage:

   (a) Approximately twenty-four strain transducers installed on the thrust structure

   (b) Twenty high response pressure transducers installed at the various tank outlets and engine inlets

   (c) Ten temperature transducers installed at the various tank outlets

   (d) Approximately fifteen vibration transducers installed at selected points on the stage structure and engines

4. Data Requirements. The data presentation will be that illustrated in the S-II-4 Static Test Final Report.

10.2.8.2 LEO Mission

The S-II stage test program for the LEO mission will be the basic program called out for the LOR mission, with test program modifications as noted below.

10.2.8.2.1 Structural Testing

Testing of the thrust structural/aft skirt will include simulation of the additional loads caused by mounting the RCS. A vibration test series with an RCS mounted on an aft skirt section will be conducted to verify both the mount and the RCS integrity when subjected to the S-II vibration levels.

10.2.8.2.2 Battleship Testing

During subsystem development testing, the test program will be revised to include the following test objectives:

a. Determine and evaluate the stage effects of cluster cutoff to idle mode and operation of approximately 14 seconds of idle mode.

b. Determine and evaluate operating parameters of the balanced vent system in the LH_2 tank over a 45-minute coast period

c. Determine stage characteristics when the engines are started to and operate in idle mode only

d. Verify operating characteristics of the auxiliary hydraulic system

10.2.8.2.3 All Systems Verification Testing

The verification testing of the first two production articles at MTF will be as described previously for the LOR mission with the following revisions:

a. Electrically simulate operation of the RCS modules

b. Revise the cutoff procedures to provide mainstage cutoff through 14 seconds of idle mode

c. Verify tank pressure characteristics during a coast period with continuous venting of the LH_2 tank

d. Verify auxiliary hydraulic system operation when the J-2S cluster is in extended idle mode operation

10.2.9 SYSTEMS TEST AND CHECKOUT REQUIREMENTS

Employment of J-2S engines on S-II vehicles will require certain changes to stage test requirements, checkout procedures, and other contractual documents. This section describes the effort required to accommodate these changes for the two major S-II/J-2S missions.

10.2.9.1 Automatic Checkout

The automatic checkout effort involves the analysis of stage and GSE systems, establishing configuration and checkout requirements, the preparation, release, and control of checkout tapes, and the publication of program description documents (PDD's). Automatic checkout is performed at Seal Beach and at MTF using the CDC 924A computer, the C7 series of checkout stations, the S-II stage, and an ATOLL software system.

10.2.9.1.1 LOR Mission

The revisions described below apply to the listed checkout tapes and to the related PDD's.

a. Stage Networks Acceptance Program (SNAP). In the separation system section, the ullage motor firing system checkout will be deleted (10 percent reduction). In the propellant feed system section, checkout of the helium injection system, recirculation pumps, and recirculation valves will be deleted (90 percent reduction), and checkout procedures for the valve actuation system and the prevalves control system will be revised (5 percent revision). In the engine system section, checkout of new circuits will be added (20 percent increase) and the control and logic operations will be completely revised (100 percent revision).

b. Pressurization System. The operations related to regulator settings will be revised (5 percent revision).

c. Electrical Power System. The recirculation bus checkout will be deleted (10 percent reduction) and operations related to electrical control interlocks are to be revised (10 percent revision).

d. Flight Control System. Checkout to accommodate new pressure limits will be revised (20 percent revision).

e. Measurements System. Checkout of all recirculation measurements will be deleted (5 percent reduction) and other tests will be changed as necessary to accommodate new relocated, re-channelized, or otherwise changed flight measurements (30 percent revision).

f. GSE Integrated Checkout. Checkout of recirculation system control GSE will be deleted (5 percent reduction). Checkout of engine control GSE and monitor functions will be revised (30 percent revision).

g. Simulated Flight Test. New switch selector commands and alternate operational modes will be added (10 percent increase). The flight sequence will be revised (100 percent revision).

h. Interface Dictionary and Command Reference List. The Dictionary and list will be revised to reflect flight and hardware measurement changes (30 percent revision).

i. Common Subprocedures. Recirculation system operations will be deleted (5 percent reduction); engine system and related interface operations will be revised (30 percent revision).

10.2.9.1.2 Delta LEO Mission

a. Stage Networks Acceptance Program (SNAP). In the propellant feed system section, tests of the prevalve responses to engine cutoffs will be revised (5 percent increase). In the engine system section, checkout of the reaction control system (RCS) and engine restart logic will be added (5 percent increase).

b. Pressurization System. Functions for the balanced LH_2 tank vent system will be added (5 percent increase).

c. Electrical Power System. Checkout procedures for the airborne auxiliary hydraulic pump power supply and control will be added (20 percent increase).

d. Flight Control System. Add functions for the "Idle Mode (second engine start)" thrust vector control operations (10 percent increase).

e. Measurement System. Checkout of new flight measurements related to the RCS, the balanced LH_2 tank vent system, and the thrust vector control system will be added (15 percent increase).

f. GSE Integrated Checkout. Control and monitor functions will be revised (15 percent increase).

g. Simulated Flight Test. Sequencing will be revised and functions added for the RCS, balanced LH_2 tank vent system, thrust vector control system, prevalve control, and "Idle Mode" restart (10 percent increase).

h. Interface Dictionary and Command Reference List. The dictionary and list will be revised to reflect measurement changes (15 percent increase).

i. Common Subprocedures. Subprocedures will be revised to reflect the modified stage quiescent state verification routine (5 percent increase).

10.2.9.2 Acceptance Test Requirements (Non-static Firing)

Two contractual specifications establish the requirements for post-manufacturing and post-static firing non-cryogenic stage acceptance testing at Seal Beach and the Mississippi Test Facility. Supplier acceptance test specifications are reviewed to assure that subcontractor component procedures are acceptable.

10.2.9.2.1 LOR Mission

a. Seal Beach/MTF Acceptance Test Specification

1. Data Requirements. Data requirements for recirculation system and ullage motor firing unit monitors will be deleted (5 percent reduction).

2. Electrical Power. Recirculation power bus requirements will be deleted (10 percent reduction).

3. Measurements. Measurement accuracy requirements for turbine by-pass, start tank discharge, and gas generator valve position measurements will be deleted. Unique requirements will be added for accuracy checkout of new measurements (5 percent reduction).

4. Separation. Reference to ullage motor ignition functions will be deleted (10 percent reduction).

5. Engines. Engine cut-off and engine sequence test requirements will be revised (80 percent revision). Requirements will be added for checkout of new control and logic functions (20 percent increase).

6. Propellant Feed and Recirculation. Requirements for recirculation, and helium injection functions will be deleted (90 percent reduction). Prevalve control test requirements will be revised (10 percent revision).

7. Simulated Flight. Simulated flight pretest requirements, acceptance readiness configuration, and flight sequences to conform to revised systems requirements will be changed (70 percent revision).

8. Pressurization. Propellant tank pressure regulator test requirements will be revised (5 percent revision).

9. The following tables will be revised: engine sequence requirements, electrical power buses data evaluation requirements; flight analog measurement data and evaluation requirements, flight critical EMC monitoring points (20 percent revision).

b. Manual Functional and Leak Check Acceptance Specification. The specification is revised to reflect changes in the engine system leak and functional acceptance requirements resulting from the deletion of the gas generator, start tank, recirculation system, turbo pump seal purge and drain requirements, and related subsystem hardware and operational changes.

LH_2 and LOX recirculation systems are eliminated and the LH_2 and LOX feed system checkout testing criteria are modified accordingly.

Requirements must be added for the thrust chamber fuel bypass system and the LH_2 feed ducts purge system. Thrust chamber tapoff turbine drive checkout will be defined and LOX dome purge system checkout specified. Valve actuation system checkout details will be completely revised and the LOX and LH_2 tank pressurization system requirements modified to new pressure levels. The above changes require the development of detailed leak check procedures, from line section to terminations (closed valve or blanking plates), the establishment of pressure levels, and the resolution of functional testing definitions for controls (30 percent reduction).

c. Vendor Acceptance Test Specifications Review. There are no significant changes.

10.2.9.2.2 Delta LEO Mission

a. Seal Beach/MTF Acceptance Test Specification.

1. Electrical Power. Electrical power test requirements will be added to provide for idle mode thrust vector control system (20 percent increase).

2. Measurements. Measurement accuracy requirements will be added for new measurements (10 percent increase).

3. Engines. Test requirements will be added for engines restart capability in the idle mode (10 percent increase). Engine sequencing requirements will be revised (10 percent revision).

4. Prevalves. Prevalves control test requirements will be revised (5 percent revision).

5. Reaction Control System. Add test requirements to assure proper electrical control of the RCS (100 percent increase).

6. Thrust Vector Control. Add test requirements for electrical control of engine gimbaling during Idle Mode (10 percent increase).

7. Pressurization. Add electrical control test requirements for balanced LH_2 tank venting (five percent increase).

8. Simulated Flight. Revise flight sequence requirements (10 percent increase).

9. Data Requirements. Add and revise data and data evaluation requirements of new measurements (five percent increase).

b. Manual Functional and Leak Check Acceptance Specification. Functional and/or leak check requirements will be added for the LOX and LH_2 pump seal drain vent systems, the balanced LH_2 tank vent system, and the auxiliary hydraulic system (10 percent increase).

c. Vendor Acceptance Test Specifications Review. New subcontractor component procedures will be reviewed and approved for the reaction control system, auxiliary hydraulic system and the balanced LH_2 tank vent system. All new and revised vendor acceptance test procedures will be reviewed to assure performance, reliability, and quality control acceptability (five percent increase).

10.2.9.3 Static Firing Test and Acceptance Requirements

Two contractual documents are used to define the requirements for acceptable stage performance before, during, and immediately following each static firing test at the Mississippi Test Facility.

10.2.9.3.1 LOR Mission

a. MTF Static Firing Requirements Document. This document will be revised to reflect changes in pre-static, static, and post-static checkout requirements resulting from engine sequence, recirculation, measurement and related system and subsystem changes. Changes in test configuration, safety requirements, and shipping and handling requirements will also be incorporated (15 percent reduction).

b. MTF Static Firing Acceptance Test Specification. This document sets forth the Engineering requirements, tolerances and limits required for satisfactory stage performance during static firing tests.

This specification will be revised to reflect the new engine sequence, mechanical, and electrical operational limits, measurement system and engine cutoff changes, deletion of the recirculation system and changes in related subsystem purge and function requirements. Proper operation of new system parameters, such as the solid propellant turbine starter (SPTS), will be incorporated as stage acceptance requirements (15 percent reduction).

10.2.9.3.3 Delta LEO Mission

a. MTF Static Firing Requirements Document. This document will be revised to reflect changes in the pre-static, static, and post-static firing checkout requirements resulting from changes to the engine sequence, pump seal drain outlets, prevalve control system, and the addition of a balanced LH_2 tank vent system. Performance verification of the reaction control electrical system and the idle mode thrust vector control system will be incorporated. The flight control system will be verified for proper idle mode operation by use of the auxiliary motor pump. Changes in the measurement system, telemetry, and range safety command systems checkout requirements will be incorporated for proper pre-static and post-static firing checkout (5 percent increase).

b. MTF Static Firing Acceptance Test Specification. New flight sequence commands and operational modes will be added for the idle mode restart operations, coast operations, balanced LH_2 tank venting, and thrust vector control. Operation of the LOX low level sensors at the correct propellant level will be verified for second burn assurance (15 percent increase).

10.2.9.4 ICD Requirements

ICD's are contractual documents which establish interface design criteria. Preparation, and maintenance responsibility is retained by NASA. Preliminary Interface Revision Notices are submitted by the Contractor as required to achieve compatibility between the S-II stage and interfacing equipment and material.

10.2.9.4.1 LOR Mission

a. 13M07000 S-IC/S-II Functional. Electrical commands and loads will be revised.

b. 13M07001 S-II/S-IVB Functional. Electrical commands and loads will be revised.

c. 13M07005 J-2S/S-II Functional. New ICD defining functional interfaces will be issued.

d. 13M50097 S-II Fluid Requirements. Requirements will be deleted for start tank system, LOX and LH_2 recirculation systems, and gse generator and turbopump purge system. Requirements will be added for oxidizer dome purge system and LH_2 feed duct purge system. Requirements will be revised for thrust chamber purge system, fuel pump drain system, valve actuation system, and LOX and LH_2 pressurization systems.

e. 13M50405 J-2S/S-II Physical. A new ICD will be issued defining the physical interfaces.

f. 40M30593 S-II/S-IVB Electrical. The Separation System electrical control circuits will be revised.

g. 65ICD9000 S-II Protuberances. Ullage motors will be deleted.

h. 65ICD9746 Umbilical (A7-41) and Service Connections (aft). Fuel pressurization line purge, start tank actuation, fuel seal cavity bleed, valve actuation fill, and start tank fill will be deleted. Existing disconnects will be redesignated to new functions: fuel pump drain, oxidizer dome purge, LH_2 feedline purge, and helium injection.

i. 65ICD9762, 65ICD9763, 65ICD9764, S7-41 Physical Console (Pneumatic Servicing) "D" section will be deleted completely. In "A" section, LOX Dome Purge System and LH_2 Feedline Purge will be added. In "B" section, thrust chamber chill will be deactivated. In "C" section, turbine start bottle GH_2 vent control will be deactivated.

j. 65ICD9765 A7-71 Heat Exchanger. Thrust chamber chill and engine start bottle circuits will be deactivated.

k. 65ICD9766 Piping Criteria (Pneumatic). Functions will be deleted and revised to match S7-41 and A7-71 changes.

l. 40M3362X Flight Sequence. See Paragraph 10.2.9.7.1 below for description.

m. 40M11703 Electrical GSE Pneumatic Checkout Equipment. Electrical circuits will be revised, added, or deleted in the S7-41 and A7-71 units to correspond with system changes.

n. 40M35000 S-II/ESE Electrical. Control and measurement functions for the start tank system, LOX and LH_2 recirculation systems, and ullage motor ignition system will be deleted. New control and measurement functions will be added for the J-2S control/start/sequencing requirements. Functions for stage electrical control and measurement systems will be revised. Power requirements for the LH_2 recirculation system will be deleted.

10.2.9.4.2 Delta LEO Mission

a. 13M07005 J-2S/S-II Functional. Control functions for "Idle Mode" second burn will be added.

b. 13M50097 S-II Fluid Requirements. Requirements for RCS helium supply, RCS thermal conditioning, and TVC air pressurant supply will be added.

c. 40M11703 Electrical GSE Pneumatic checkout Equipment. Requirements for RCS helium supply and for TVC air pressurant supply will be added.

d. 40M30593 S-II/S-IVB Electrical. Functions from RCS control and thrust vector control will be added.

e. 40M3362X Flight Sequence. See paragraph 10.2.9.7.3 below for description.

f. 40M35000 S-II/ESE Electrical. Functions for RCS, thrust vector control, balanced LH_2 tank vents, idle mode second burn, LOX tank low level sensors, and electrical controls will be added.

g. 65ICD9000 S-II Protuberances. Balanced LH_2 tank vents and RCS modules will be added.

h. 65ICD9746 Umbilical (A7-41) and Service Connections (aft). Disconnect for RCS helium supply and TVC air supply will be added.

i. 65ICD9762, 65ICD9763, 65ICD9764; S7-41 Pneumatic Console. RCS helium supply and TVC air supply will be added.

j. 65ICD9766 Piping Criteria, S-II Pneumatics. Requirements for RCS helium supply and TVC air supply will be added.

10.2.9.5 KSC Test Specifications and Criteria (SID 66-1465)

The specifications and criteria document establishes all S-II test requirements at KSC.

10.2.9.5.1 LOR Mission (5 percent reduction)

a. J-2S Engine. Test requirements for the gas generator and start tank systems will be deleted. Provisions for verification of the solid propellant turbine start system, thrust chamber bypass system, and thrust chamber gas tapoff system will be added. The electrical sequence and timing requirements will be revised.

b. LOX Fill and Drain. Test requirements for the LOX depletion cutoff circuits will be revised.

c. Fuel Pressurization. Reference to the fuel recirculation system will be deleted. The regulator requirements and the feed duct purge check valve requirements will be revised.

d. LOX Pressurization. All reference to the LOX recirculation system will be deleted. The regulator requirements will be revised.

e. Ullage Motors. All requirements will be deleted.

f. Electrical Power. The power requirements for the fuel recirculation system will be deleted.

g. Control Pressure. Requirements will be revised to accommodate the simplified system.

h. Sequencing. The LOX and fuel recirculation functions will be deleted and the engine start sequencing test requirements will be revised. All ullage motor ignition test requirements will be deleted.

i. Separation. All tests of the ullage motor ignition system will be deleted.

j. Transducers and Signal Conditioning. Test requirements as required will be revised to account for deleted and new measurements.

10.2.9.5.2 Delta LEO Mission (10 percent increase)

a. Fuel Pressurization. Test requirements for the Balanced LH_2 tank vent system will be added.

b. LOX and LH_2 Fill and Drain. LOX and LH_2 low level sensor checkout will be revised.

c. Environmental Conditioning. RCS thermal conditioning and purge requirements for the balanced LH_2 vent nozzles will be added.

d. Reaction Control System. Requirements for this new system will be added.

e. Electrical Power. Requirements for power to the thrust vector control system will be added.

f. Sequencing. Test requirements will be added for "Idle Mode" restart, balanced LH_2 vent system and thrust vector control system. Reference to LH_2 step pressurization will be deleted.

g. Fluid Power. Requirements will be added for the thrust vector control system.

10.2.9.6 Schematic Drawings

System schematics depict the configuration of the S-II GSE and its functional interface with the stage. These schematic drawings are used to assist in the definition of design changes, development of test procedures, and conduct of trouble-shooting at the test sites. They are also used to assess the compatibility of stage and GSE.

10.2.9.6.1 LOR Mission

a. Electrical Checkout GSE (Seal Beach and MTF). These drawings will require extensive revisions because of deletion of the recirculation system, deletion of the ullage motors, deletion of the engine gas generator and start tank, addition of 10 EBW firing units and pulse sensors, revision of engine stimuli and measurements, simplification of the valve actuation system, and revision of stage stimuli and measurements.

b. Mechanical Checkout GSE (Seal Beach and MTF). The elimination of J-2 engine thrust chamber chill, start tank, LOX and LH_2 recirculation requirements and modification of the stage system purge requirements reduces the number of active components in the following GSE: A7-71 heat exchanger, S7-27, S7-33, SDD-198, and SDD-199 fluid distribution systems, and the A7-51 SLAM. The number of active circuits in the C7-603 pneumatic checkout console and the S7-41 pneumatic servicing consoles will also be reduced. The GSE schematic modifications will establish the revised line runs and interconnections to be compatible with the stage changes.

c. Static Firing Electrical GSE (MTF). The elimination of J-2 engine thrust chamber chill and start tank requirements, modification of the stage system purge requirements, total elimination of LOX and LH_2 recirculation, and the simplification of the static firing requirements reduces the electrical circuitry in the following GSE: C7-801, C7-802, C7-805, C7-211, C7-213, C7-41, S7-42, S7-41, A7-71, S7-27, S7-33, and C7-44. The changes plus the modifications to the engine control and measurements circuitry will result in extensive wire and function changes to both the A1 and A2 test stand system schematics.

d. S-II Stage Mechanical. The Stage System schematic drawings will require revision to incorporate changes in the recirculation, valve actuation, and engine systems. Sections requiring major modifications include the Engine Checkout, Recirculation, and Valve Actuation sections, while the remaining sections will require minor modifications.

10.2.9.6.2 Delta LEO Mission

a. Electrical Checkout GSE (Seal Beach and MTF). Revisions are required because of addition of idle mode restart, addition of RCS relays and bus, modification of prevalve control, addition of balanced LH_2 tank venting, addition of thrust vector control, modification of LOX and LH_2 low level sensor functions, and revision of stage stimuli and measurements.

b. Mechanical Checkout GSE (Seal Beach and MTF). Revisions are required because of addition of balanced LH_2 tank venting, addition of thrust vector control, addition of RCS helium supply, addition of RCS thermal conditioning, and addition of balanced LH_2 venting and purging.

c. Static Firing Electrical GSE (MTF). Revisions are required because of Idle Mode restart, thrust vector control, Balanced LH_2 venting and purging, LOX and LH_2 low level sensor function modification, modification of prevalve control, addition of RCS simulation, revision of flight sequencing, and revision of stage stimuli and measurements.

d. S-II Stage Mechanical. Revisions are required because of the addition of RCS, thrust vector control, balanced LH_2 tank venting system and related purging, RCS helium supply, and RCS thermal conditioning.

10.2.9.7 J-2/S-II Flight Sequence Program

The flight sequence program is a chronological list of the IU switch selector stimuli addressed to the S-II stage during flight. This sequence is incorporated in the automated simulated flight test which is conducted during acceptance testing at Seal Beach and at MTF, and is programmed into the static firing controller at MTF. Interface Control Document 40M3362X governs the timing, sequence, and channelization of flight sequence commands.

10.2.9.7.1 LOR Mission

The deletion of recirculation systems and the ullage motors, and the revised engine starting logic required some revisions to the flight sequence program which are used for J-2/S-II configurations. The following S-II switch selector commands have been deleted:

"LH_2 Recirculation Pumps Off"

"Ullage Trigger"

"Chilldown Valves Close"

"LOX Depletion Sensors Cutoff Arm"

The following S-II Switch Selector Commands have been added:

"Prevalves Close Arm Reset"

"All Engines Start No. 2"

"Mainstage Start No. 1"

"Mainstage Start No. 2"

The command "Engine Start" has been revised to "All Engines Start No. 1".

Using the MSFC J-2S Design Trajectory timing, a preliminary S-II Flight Sequence Program was evolved for the LOR mission. The program is shown in Table 10.2-XL. Several functions which are not addressed to the S-II stage are shown for reference purposes.

Additional switch selector commands were used on S-II-3 to modify the propellant utilization operational mode (ECP 5982) to an open loop operation; however, the timing and open loop mode have not been established yet as a permanent change. The commands for the "open loop" mode are as follows:

"Open Loop Arm"

"High EMR Select"

"High EMR Reset"

"Low EMR Select"

"Open Loop Arm Reset"

Although open-loop operation of the propellant utilization system may be employed in future missions, the flight sequence tables herein reflect the original closed loop mode. It is assumed that the propellant utilization sequencing and timing used on future S-II flights will also be used on missions which employ the J-2S/S-II configuration.

Table 10.2-XL. LOR Mission - Flight Sequence Program

Nominal Flight Time (Min:Sec)	Command	Switch Selector	
		Stage	Channel
	Prelaunch - Reference Information		
-24:00	Prevalves Lockout On	S-II	Ground
-24:00	All Engines Cutoff On	S-II	18
-04:00	Hydraulic Accumulators Locked On	S-II	Ground
00:00.0	Liftoff - Start of Time Base 1 (T_1)		
02:29.5	S-IC Inboard Engine Cutoff - Start of Time Base 2 (T_2)		
02:30.5	Start First FM/FM Calibration	S-II	30
02:35.5	Stop First FM/FM Calibration	S-II	9
02:36.5	Ordnance Arm	S-II	11
02:41.5	S-IC Outboard Engines Cutoff - Start of Time Base 3 (T_3)		
02:41.6	LH_2 Tank High Pressure Vent Mode	S-II	38
02:41.7	Prevalves Close Arm Reset	S-II	48
02:42.2	S-IC/S-II Separation No. 1	S-IC	15
02:42.3	S-IC/S-II Separation No. 2	S-IC	19
02:42.4	Switch Engine Control to S-II	IU	33
02:42.5	Engines Cutoff Reset	S-II	31
02:42.6	Engines Ready Bypass	S-II	20
02:42.7	Prevalves Lockout Reset	S-II	19
02:42.8	All Engines Start No. 1	S-II	33
02:42.9	All Engines Start No. 2	S-II	TBD
02:43.0	Mainstage Start No. 1	S-II	TBD
02:43.1	Mainstage Start No. 2	S-II	TBD
02:43.2	Engines Ready Bypass Reset	S-II	49
02:44.4	Hydraulic Accumulators Unlock	S-II	12
02:48.1	Start Phase Limiter Cutoff Arm	S-II	25
02:48.3	Activate Propellant Utilization System	S-II	32
02:49.1	Start Phase Limiter Checkout Arm Reset	S-II	6
02:49.2	Prevalve Close Arm	S-II	99
03:14.0	Second Plane Separation	S-II	23
04:43.0	Start Second FM/FM Calibration	S-II	30
04:48.0	Stop Second FM/FM Calibration	S-II	9
06:10.0	Start Third FM/FM Calibration	S-II	30
06:15.0	Stop Third FM/FM Calibration	S-II	9
TBD	LH_2 Step Pressurization	S-II	7
07:47.0	S-II/S-IVB Ordnance Arm	S-II	8
07:48.0	LH_2 Depletion Sensors Cutoff Arm	S-II	42
08:18.0	Cutoff S-II J-2S Engines - Start of Time Base 4 (T_4)		
08:18.8	S-II/S-IVB Separation	S-II	5

10.2.9.7.2 Delta LEO Mission

New switch selector commands are required because of the addition of the thrust vector control system, the "Idle Mode" restart operations, and the balanced LH_2 vent system. Further analysis is required to establish the possible use of ground-issued commands during the coast period and "Idle Mode" second burn period. The ground commands would be used as backup measures and also to control alternate sequence modes for alternate time bases and mission modes. The analysis would be performed in conjunction with the MSFC Launch Vehicle Ground Support Plan that is issued for every launch. The flight sequence program for the LEO mission is shown in Table 10.2-XLI. A summary listing is given below. The following S-II Switch Selector Command has been deleted:

"LH_2 Step Pressurization"

The following S-II Switch Selector Commands are new:

"Start Auxiliary Hydraulic Pumps No. 1"

"Start Auxiliary Hydraulic Pumps No. 2"

"Restart Arm"

"Restart Arm Reset"

"Balanced LH_2 Vent System Arm"

"Mainstage Cutoff"

"Start Auxiliary Hydraulic Pumps No. 1 Reset"

"Start Auxiliary Hydraulic Pumps No. 2 Reset"

"Mainstage Cutoff Reset"

"LOX Depletion Sensors Cutoff Arm"

The "S-II/S-IVB Ordnance Arm" and "S-II/S-IVB Separation" commands are not shown in the table since separation operations after the idle mode second burn were not considered to be a part of this study.

Table 10.2-XLI. LEO Mission - Flight Sequence Program

Nominal Flight Time Min:Sec	Command	Switch Selector	
		Stage	Channel
	Prelaunch - Reference Information		
-24:00	Prevalves Lockout On	S-II	Ground
-24:00	All Engines Cutoff On	S-II	18
-04:00	Hydraulic Accumulators Locked On	S-II	Ground
00:00.0	Liftoff - Start of Time Base 1 (T_1)		
02:29.5	S-IC Inboard Engine Cutoff - Start of Time Base 2 (T_2)		
02:30.5	Start First FM/FM Calibration	S-II	30
02:35.5	Stop First FM/FM Calibration	S-II	9
02:36.5	Ordnance Arm	S-II	11
02:41.5	S-IC Outboard Engines Cutoff - Start of Time Base 3 (T_3)		
02:41.6	LH_2 Tank High Pressure Vent Mode	S-II	38
02:41.7	Prevalves Close Arm Reset	S-II	48
02:42.2	S-IC/S-II Separation No. 1	S-IC	15
02:42.3	S-IC/S-II Separation No. 2	S-IC	19
02:42.4	Switch Engine Control to S-II	IU	33
02:42.5	Engines Cutoff Reset	S-II	31
02:42.6	Engines Ready Bypass	S-II	20
02:42.7	Prevalves Lockout Reset	S-II	19
02:42.8	All Engines Start No. 1	S-II	33
02:42.9	All Engines Start No. 2	S-II	TBD
02:43.0	Mainstage Start No. 1	S-II	TBD
02:43.1	Mainstage Start No. 2	S-II	TBD
02:43.2	Engines Ready Bypass Reset	S-II	49
02:44.4	Hydraulic Accumulators Unlock	S-II	12
02:48.1	Start Phase Limiter Cutoff Arm	S-II	25
02:48.3	Activate P.U. System	S-II	32
02:49.1	Start Phase Limiter C/O Arm Reset	S-II	6
02:49.2	Prevalve Close Arm	S-II	99
03:14.0	Second Plane Separation	S-II	23
04:43.0	Start Second FM/FM Calibration	S-II	30
04:48.0	Stop Second FM/FM Calibration	S-II	9
06:10.0	Start Third FM/FM Calibration	S-II	30
06:15.0	Stop Third FM/FM Calibration	S-II	9
07:34.0	LOX Depletion Sensors Cutoff Arm	S-II	3
07:34.2	LH_2 Depletion Sensors Cutoff Arm	S-II	42
07:34.4	Balanced LH_2 Vent System Arm	S-II	TBD
08:04.0 (approx)	Mainstage Cutoff (Initiated by IU upon satisfaction of orbital parameters. After this event has occurred the remaining commands in time base 3 are issued in the indicated sequence)	S-II	TBD

Table 10.2-XLI. LEO Mission - Flight Sequence Program (Cont)

Nominal Flight Time Min:Sec	Command	Switch Selector	
		Stage	Channel
08:05.0 08:14.0	Mainstage Cutoff Reset All Engines Cutoff	S-II S-II	TBD 18
	(Start of Time Base 5 (T_5))		
X(TBD)	Restart Arm	S-II	TBD
X+00:00.1	Restart Arm Reset	S-II	TBD
X+00:00.2	Prevalves Close Arm Reset	S-II	48
	NOTE: The following sequences will be repeated as required during the "coast" mode.		
TBD TBD	Start Tlm Calibrate Stop Tlm Calibrate	S-II S-II	30 9
	NOTE: Preliminary studies indicate that thermal control of the auxiliary hydraulic fluid will not be required during the "coast" mode. If future studies indicate the need of thermal control, additional commands can be added to turn the auxiliary hydraulic pumps on and off.		
50:00.0 (approx)	Begin Restart Preparations - Start of Time Base 6 (T_6)		
50:09.9	Start Tlm Calibrate	S-II	30
50:14.9	Stop Tlm Calibrate	S-II	9
50:15.0	Start Auxiliary Hydraulic Pumps No. 1	S-II	TBD
50:15.5	Start Auxiliary Hydraulic Pumps No. 2	S-II	TBD
50:16.0	Hydraulic Accumulators Unlock	S-II	12
53:09.4	Start Tlm Calibrate	S-II	30
53:14.4	Stop Tlm Calibrate	S-II	9
53:14.6	Engines Cutoff Reset	S-II	31
53:14.8	Engines Ready Bypass	S-II	20
53:15.0	All Engines Start No. 1	S-II	33
53:15.1	All Engines Start No. 2	S-II	TBD
53:15.5	Engines Ready Bypass Reset	S-II	49
56:00.6	Cutoff S-II Engines - Start of Time Base 7 (T_7) (Assumes 165.6 seconds of "Idle Mode" burn time)		
56:00.8	Start Auxiliary Hydraulic Pumps 1 Reset	S-II	TBD
56:00.9	Start Auxiliary Hydraulic Pumps 2 Reset	S-II	TBD
	NOTE: S-II/S-IVB ORDNANCE ARM, S-II/S-IVB SEPARATION, and any S-II deorbit or retro commands are not shown because flight operations after S-II "Idle Mode" cutoff are not part of this study.		

10.2.10 LAUNCH OPERATIONS REQUIREMENTS

The effect on the launch operations of the S-II stage resulting from incorporation of the J-2S engine for the LOR mission (and for those additional effects resulting from stage changes necessary to accommodate LEO mission requirements) are being assessed in a separate study by NASA/KSC. The information presented in the following paragraphs is a preliminary description of the launch operations impact as determined by NR/SD Seal Beach Operations. It was earlier submitted to NASA/KSC as an initial overview of the launch operations impact.

10.2.10.1 Facility and Launch Support Equipment Requirements

Changes required for the LOR mission and the additional changes required for the LEO mission are described in the following paragraphs.

10.2.10.1.1 LOR Mission

The ullage motor handling, storage, X-rays, inspection requirements, and the S-II GH_2 supply requirements are to be deleted. The umbilical fluid lines for engine start tank purge and chill, LOX recirculation helium injection, start tank vent control, LH_2 pump seal drain, and thrust chamber chill will be deleted or capped off, but the purge function will be retained. The Section "D" console of S7-41 ground pneumatic console and interconnecting lines will be deleted. The support requirements for the recirculation batteries will be deleted. Over All Test (OAT) equipment for recirculation electrical load simulation and control will be deleted. Electrical Support Equipment (ESE) will be revised as required for the modified electrical controls and modified measurement signals. The requirement for the engine solid propellant turbine start (SPTS) ordnance for handling, storage, and inspection will be added. Electrical cabling will be modified to new requirements.

10.2.10.1.2 Delta LEO Mission

a. Reaction Control System (RCS): Requirements for a bench checkout facility in the VAB for the RCS module leak and functional tests will be added. It may be possible to use the existing S-IVB facility for these tests. An RCS valve control unit will be added on Platform No. 1 of the mobile service structure (MSS). The hypergolic propellant loading, pressurization, and purging lines and connections on the MSS will be modified to provide for servicing the S-II RCS modules from Platform No. 1. Safety clothing and safety devices required for the toxic hypergolic propellant handling and servicing will be added. Handling and hoisting equipment for installation of the RCS modules will be provided. An umbilical line for the RCS helium supply (S7-41 connections) will be added. Storage and preservation requirements for the RCS modules are to be provided.

b. Thrust Vector Control (dc pump motors in pressurized containers). Dc ground power supply and controls will be added. Support requirements for the airborne batteries, umbilical line for auxiliary hydraulic motor air tank supply (S7-41 connections), and facility air supply on LUT's to feed the S7-41 TVC air circuit will be added.

c. Miscellaneous. Checkout equipment will be added to measure solar absorptivity and infared emissivity for optical property tests on airborne components. The ESE will be modified for required electrical control and measurement signals, and the electrical cabling will be changed to meet new requirements.

10.2.10.2 Low Bay Operations

Low Bay operations will be affected as stated below.

10.2.10.2.1 LOR Mission

The elimination of the engine start tanks and the stage recirculation systems will simplify the stage systems leakage and functional tests that are performed in Low Bay.

10.2.10.2.2 LEO Mission

The addition of the RCS bench checkout involving thorough leak and functional tests will increase the propulsion checkout tasks. The additional measurements will increase the measurements checkout. The increased electrical power and control systems will increase the electrical checkout.

10.2.10.3 High Bay Operations

High Bay operations will be affected as described below.

10.2.10.3.1 LOR Mission

a. Individual System Tests and Operations. The elimination of the stage recirculation systems and engine start tanks will reduce the propulsion system testing. The deletion of the ullage motors and the addition of the engine SPTS will balance each other in regard to EBW testing and checkout. The simplified fluid systems will reduce the S7-41 ground pneumatics operations and checkout. The A7-41 aft umbilical installation task and leak checks will be slightly reduced. The elimination of the ullage motor fairings will simplify the installation of fairings. The new switch selector functions will increase the switch selector and sequence control checkout tasks.

b. Launch Vehicle and Space Vehicle Integrated Tests. The integrated testing will not be significantly affected by the fluid system changes, since the tests are generally electrical in nature and fluid operations are at a minimum. The additional switch selector commands and flight sequencing will slightly increase the complexity. The elimination of the stage recirculation systems will have a very minor effect in reducing electrical checkouts. The danger of inadvertent dry spinning and damaging of the LH_2 recirculation pumps is eliminated. The elimination of the recirculation batteries and inverters reduces the electrical preparation tasks slightly.

c. Ordnance Operations. The deletion of the ullage motors will eliminate the large ordnance handling, hoisting, and installation tasks. The small ordnance tasks will be increased by the addition of the engine SPTS system. It is presumed that the SPTS cartridges will be installed in High Bay as is the other ordnance; only EBW detonators and the safe and arm device are installed at the Launch Pad during the countdown.

10.2.10.3.2 Delta - LEO Mission

a. Individual System Tests and Operations. The addition of the RCS system electrical provisions will require additional electrical checkout. The RCS modules will normally not be installed on the stage during High Bay testing except for an IU guidance and flight control test; therefore individual RCS module tests will not be performed in High Bay. The dc motor auxiliary hydraulic system with its dc ground power and airborne battery requirements will increase the electrical checkout tasks. The balanced LH_2 tank venting system will increase the checkout task. The idle mode restart logic will increase the electrical and engine systems checkout task. The revised flight sequencing will increase the switch selector and logic checkout task. The additional measurements will increase the measurements verification task. The revised prevalve control system will increase the electrical checkout task. The A7-41 aft umbilical installation task and leak checks will also be slightly increased.

b. Launch Vehicle and Space Vehicle Integrated Tests. The RCS modules will be installed on the stage for the IU end-to-end guidance and flight control test. For other integrated tests, an RCS simulator will be electrically connected to the stage with drag-on cables. The revised prevalve control system, the balanced LH_2 vent system, the thrust vector control system, the idle mode restart logic, the additional switch selector commands, the flight sequencing logic and simulated coasting flight will significantly increase the integrated testing requirements and lengthen the S-II stage active period during integrated testing. This will require engineers, technicians, and support personnel to spend a significantly greater amount of time on integrated testing.

c. Ordnance Operations. There is no change from the LOR mission.

10.2.10.4 Launch Pad Operations Prior to CDDT-Countdown

Operations will be affected as described in the following paragraphs.

10.2.10.4.1 LOR Mission

The miscellaneous tasks required prior to the CDDT-countdown are not significantly affected. The launch vehicle and space vehicle integrated tests performed at the launch pad are affected only slightly as described under High Bay Operations above.

10.2.10.4.2 LEO Mission

The RCS modules will be permanently installed at the launch pad just prior to the flight readiness test (FRT). During the FRT the modules will be pressurized and actual valve movements will occur during the simulated flight sequence. The other integrated test tasks performed at the pad will be significantly increased for the same reasons described in the High Bay Integrated Test Section.

10.2.10.5 CDDT and Launch Countdown

The effect on operations will be as described below.

10.2.10.5.1 LOR Mission

Figure 10.2-170 shows the NR/SD tasks that are required during the CDDT-countdown combination. The original tasks (J-2 engines) that are no longer required because of the J-2S engine implementation are shown for comparison and are lined out. The vacuum check on the jacketed fluid lines, performed just prior to the start of the CDDT, will be considerably simplified by the deletion of the stage recirculation systems. The net effect of the J-2S implementation on the CDDT countdown operations is a minor reduction in the tasks. The elimination of start tank purge, chill and pressurization, thrust chamber chill, LH_2 recirculation pump operations, and LOX helium injection purge and operation are also slight task reductions. The elimination of redlines associated with these operations, however, provides a significant increase in the probability of a successful countdown. The redlines are discussed in the Launch Mission Rules section below.

10.2.10.5.2 Delta - LEO Mission

Figure 10.2-170 shows the additional tasks required by the RCS, the thrust vector control system, and the balanced LH_2 tank venting system. A large additional task, in preparation for the CDDT, is the purging, flushing, propellant loading, and pneumatic servicing of the RCS modules. This task is performed using firing room control and RCS valve unit control on MSS Platform No. 1. Since the hypergolic propellants are extremely hazardous, this task is safety-oriented and performed cautiously. In addition, the RCS modules must be thermally controlled until launch time. The thermal control requires the periodic monitoring and recording of module temperatures while the engine compartment conditioning system continuously supplies conditioned air to the modules. In the event of an extensive delay, the hypergolic propellants must be drained with subsequent flushing, decontamination, and drying of the RCS modules. The addition of the RCS modules also adds additional redline requirements. The increased complexity of the electrical power system, the electrical control system, the flight sequencing commands, the coast mode, and the idle burn restart mode will significantly increase amount of checkout used in the various individual tasks. In addition, this increased complexity will increase the integrated checkouts performed during the CDDT-countdown. The over-all effect of the LEO mission requirements will significantly increase the CDDT-countdown tasks and will slightly reduce the probability of a successful countdown.

10.2.10.6 Launch Mission Rules

The effect on the Launch Mission Rules will be as described below.

10.2.10.6.1 LOR Mission

The J-2S implementation significantly reduces the number of redlines that could cause a mission cancellation. Of a total of 78 redline measurements, the following 34 measurements are deleted: LOX pump discharge temperatures (Engines 1-5), start tank gas temperatures (Engines 1-5), thrust chamber jacket temperatures (Engines 1-5), engine inlet LH_2 temperatures and pressures (Engines 1-5), start tank pressures

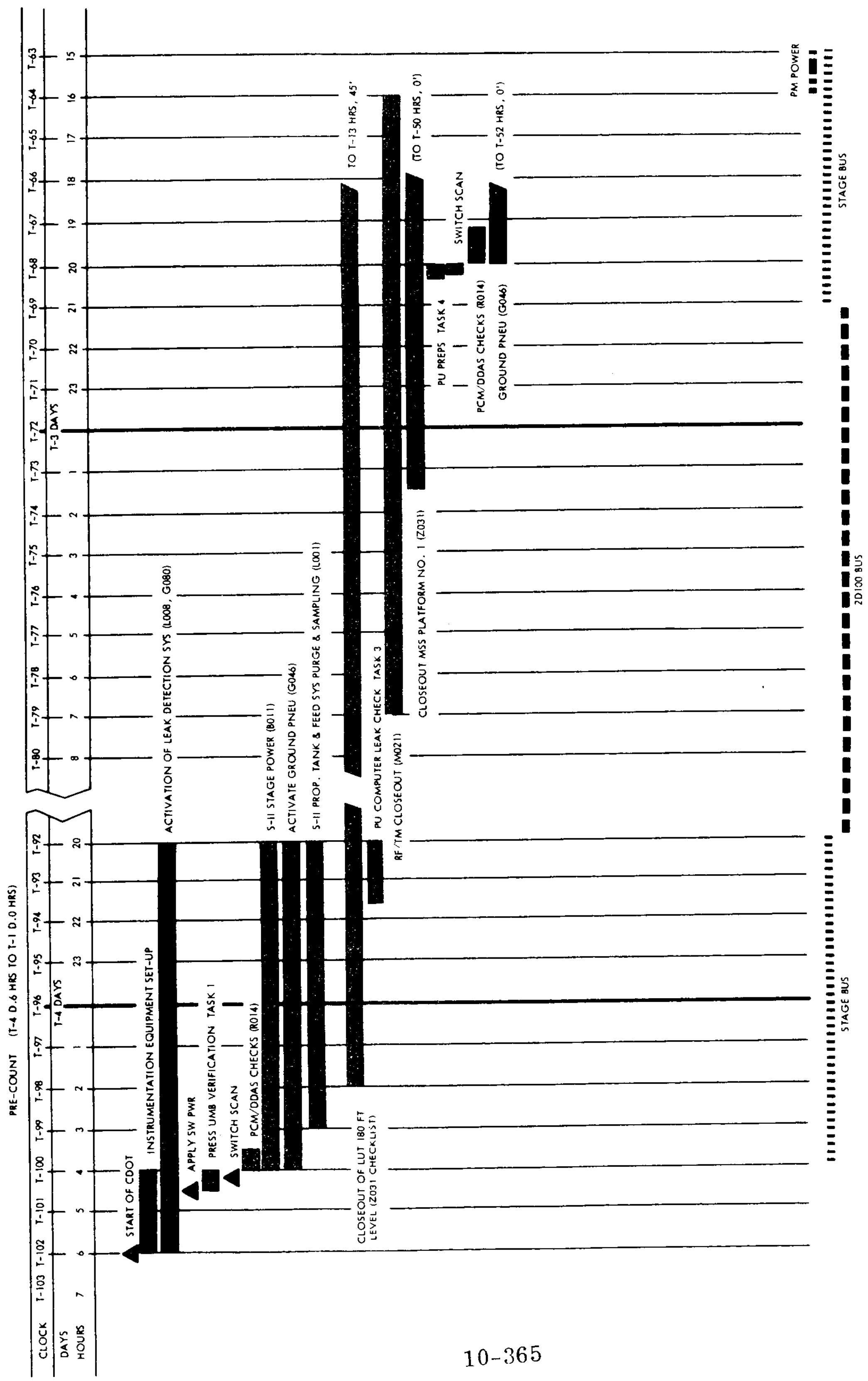

Figure 10.2-170. NR Tasks for CDDT and Countdown (Sheet 1 of 8)

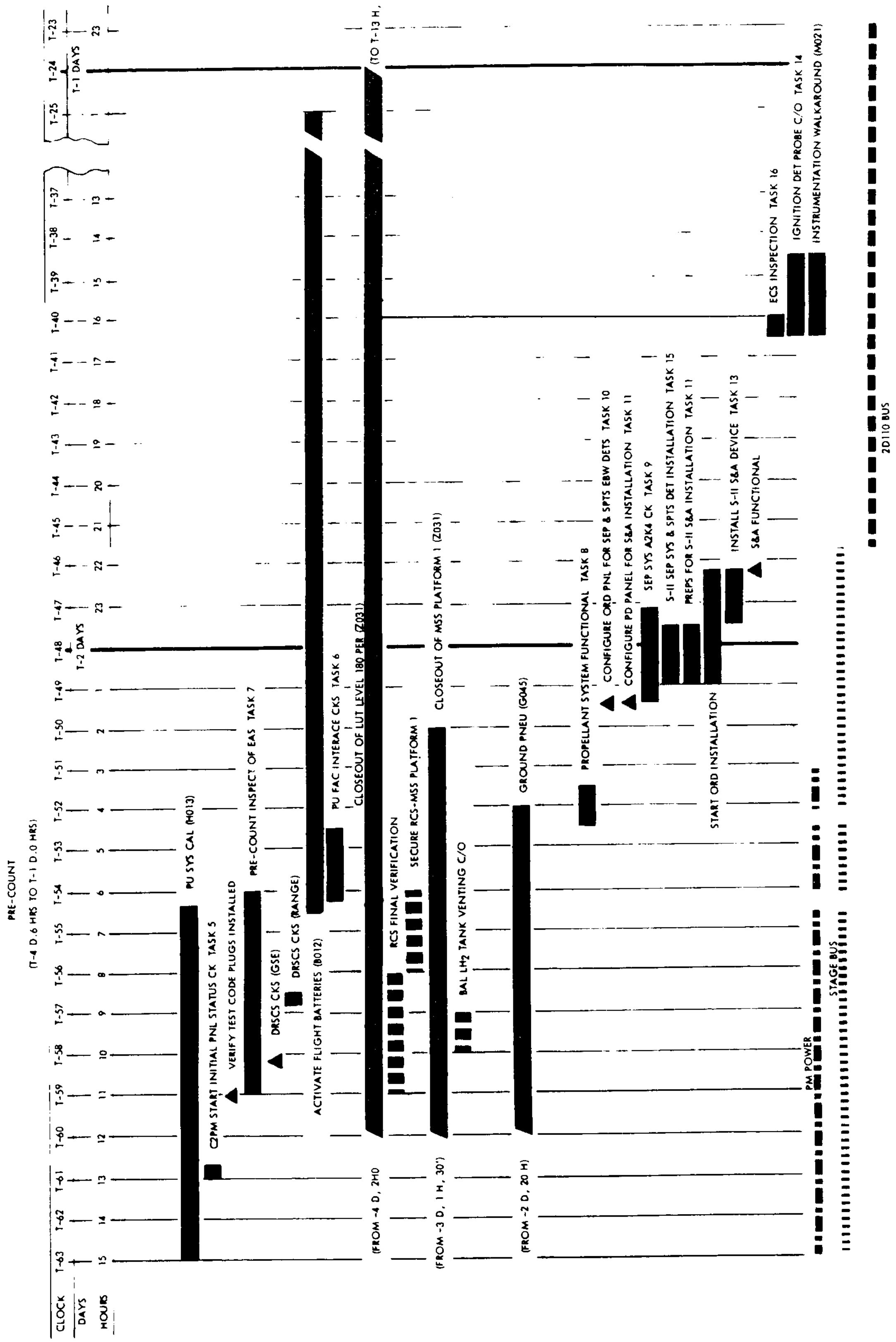

Figure 10.2-170. NR Tasks for CDDT and Countdown (Sheet 2 of 8)

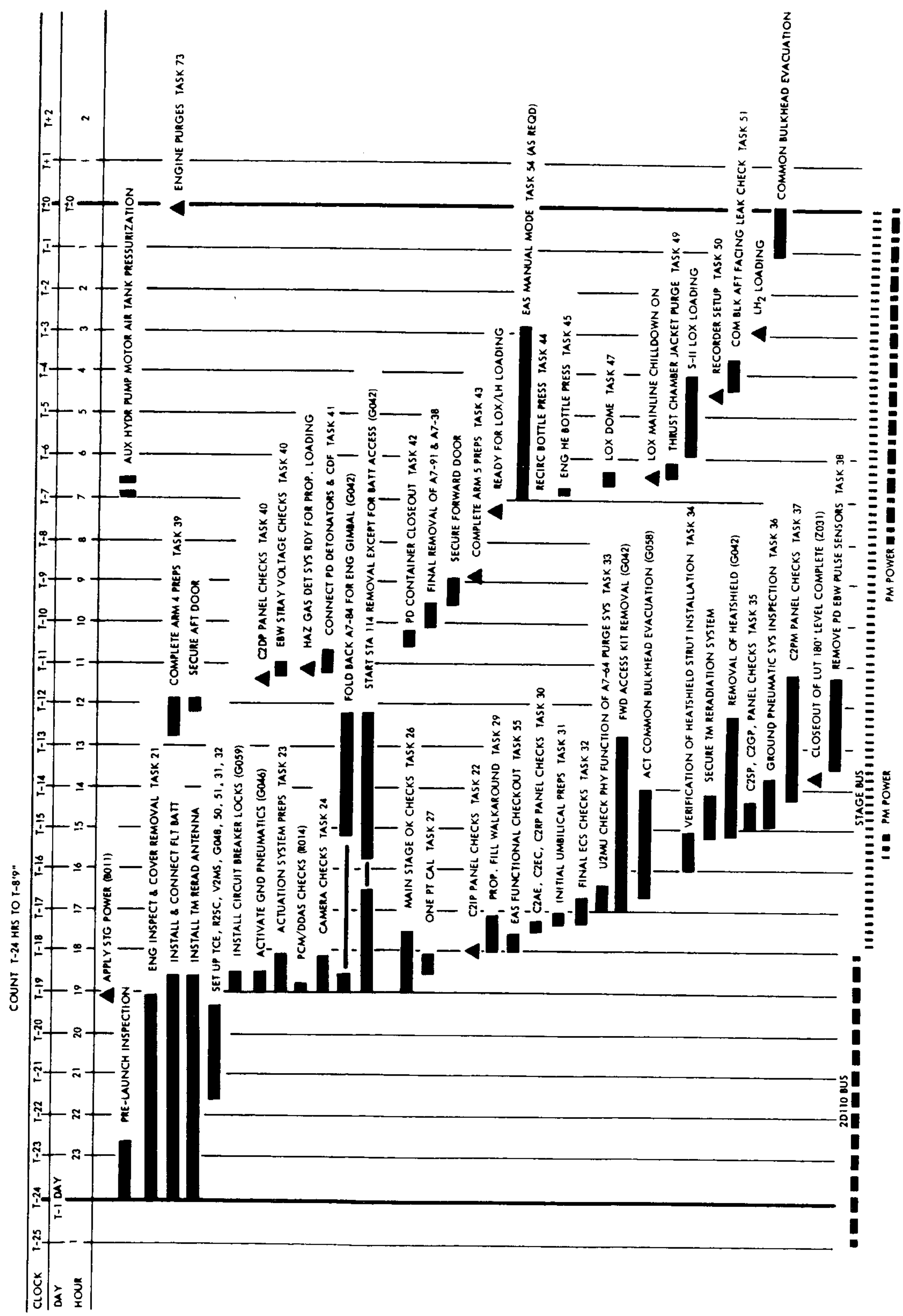

Figure 10.2-170. NR Tasks for CDDT and Countdown (Sheet 3 of 8)

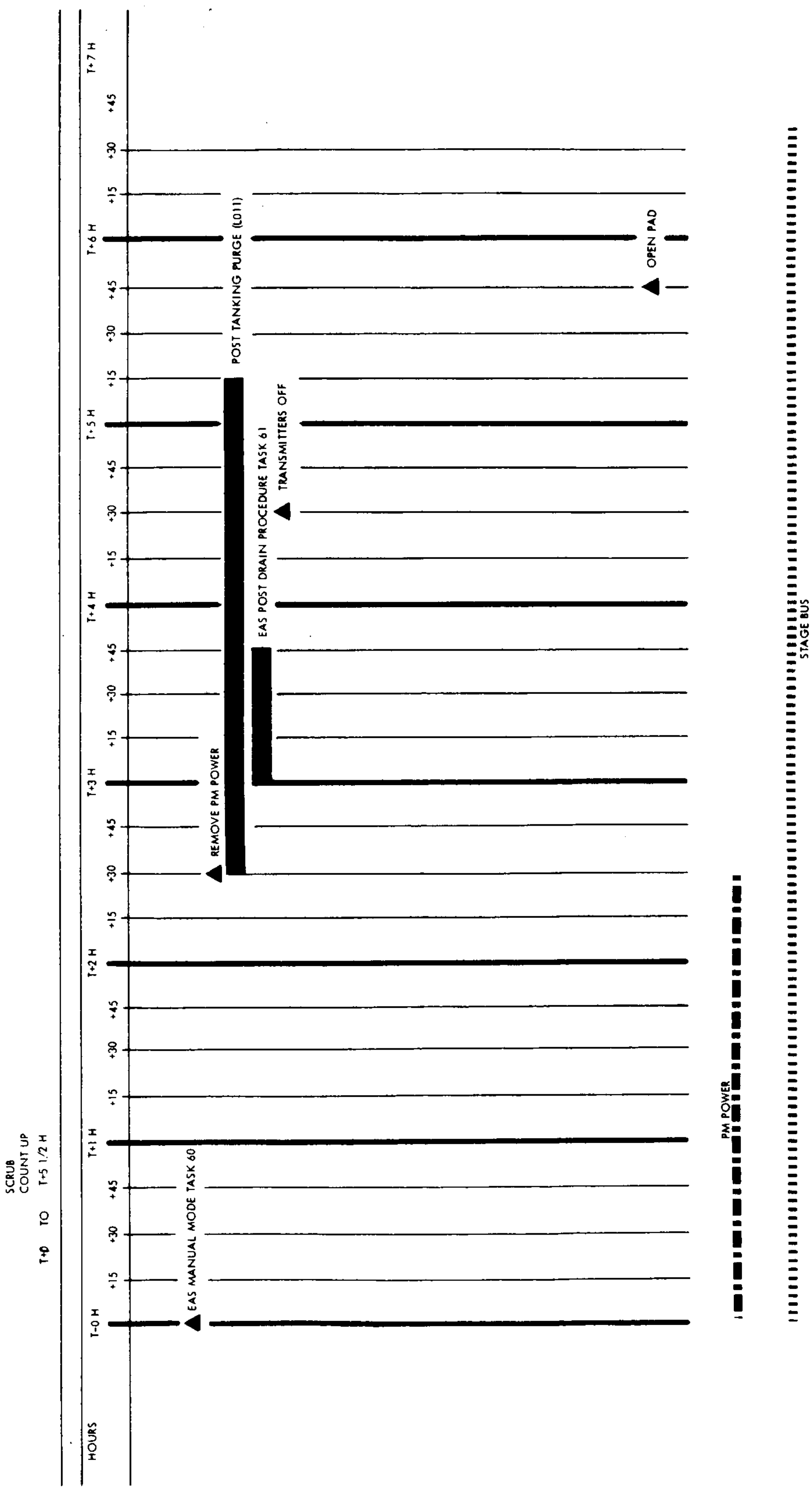

Figure 10.2-170. NR Tasks for CDDT and Countdown (Sheet 4 of 8)

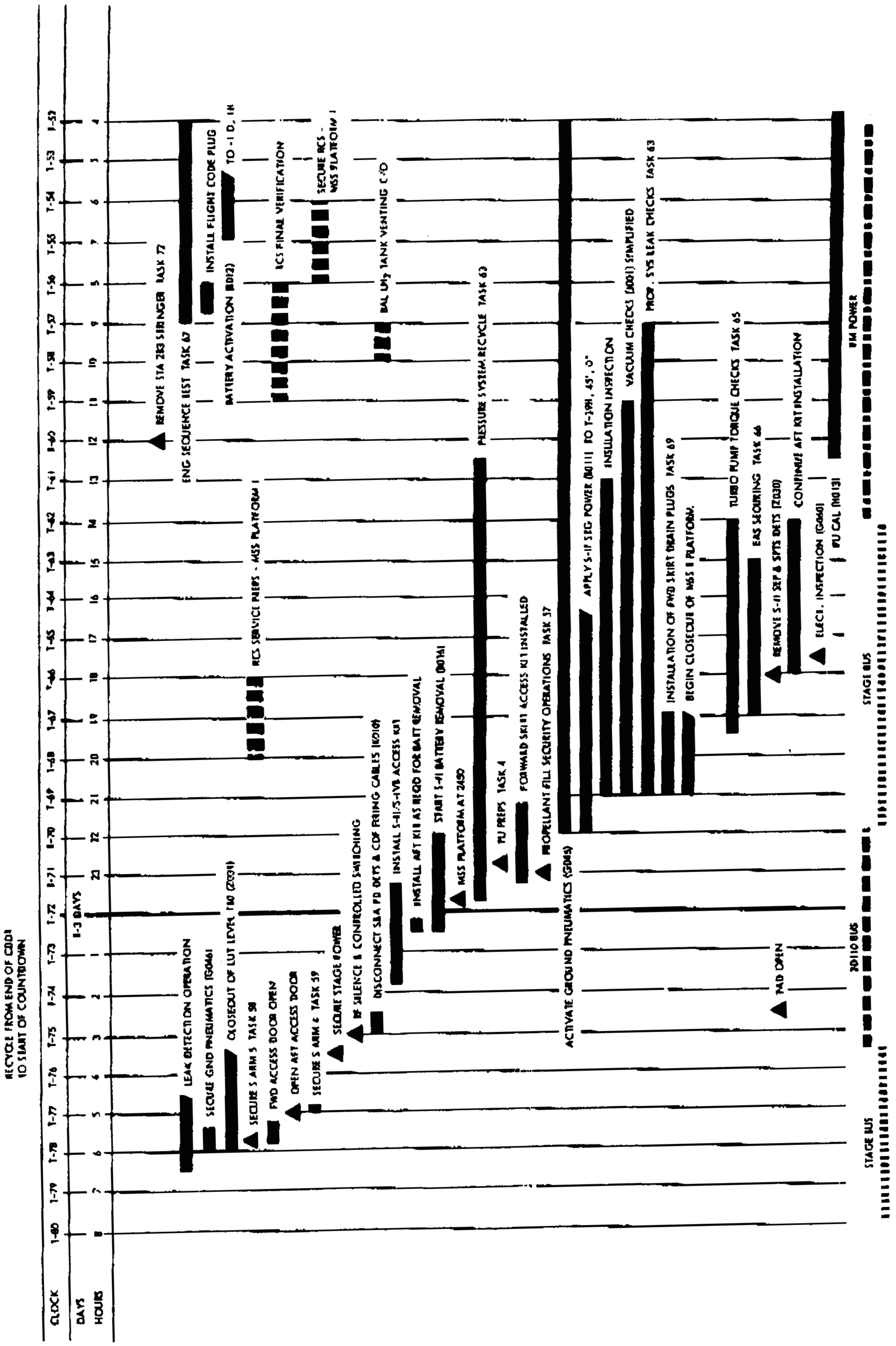

Figure 10.2-170. NR Tasks for CDDT and Countdown (Sheet 5 of 8)

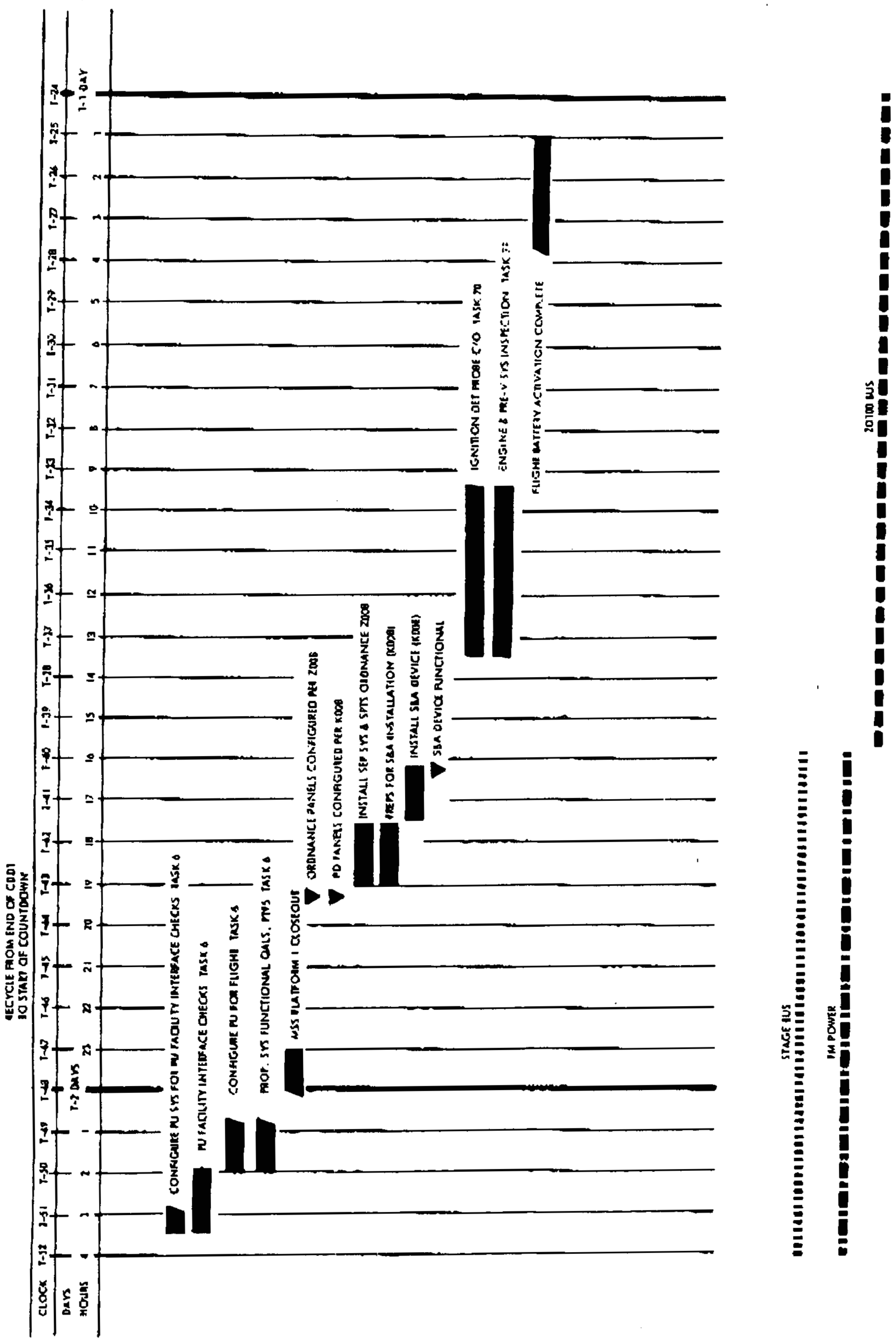

Figure 10.2-170. NR Tasks for CDDT and Countdown (Sheet 6 of 8)

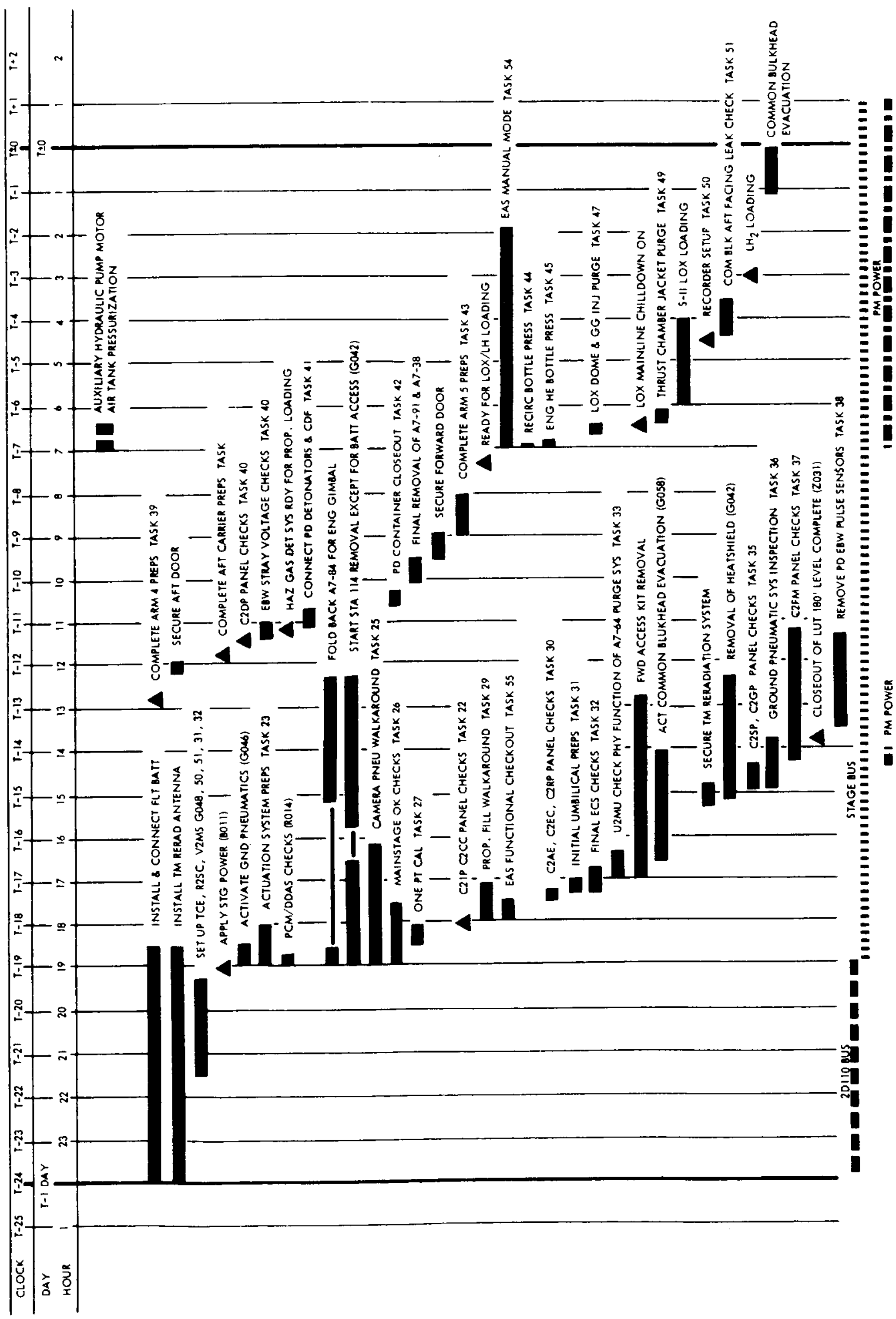

Figure 10.2-170. NR Tasks for CDDT and Countdown (Sheet 7 of 8)

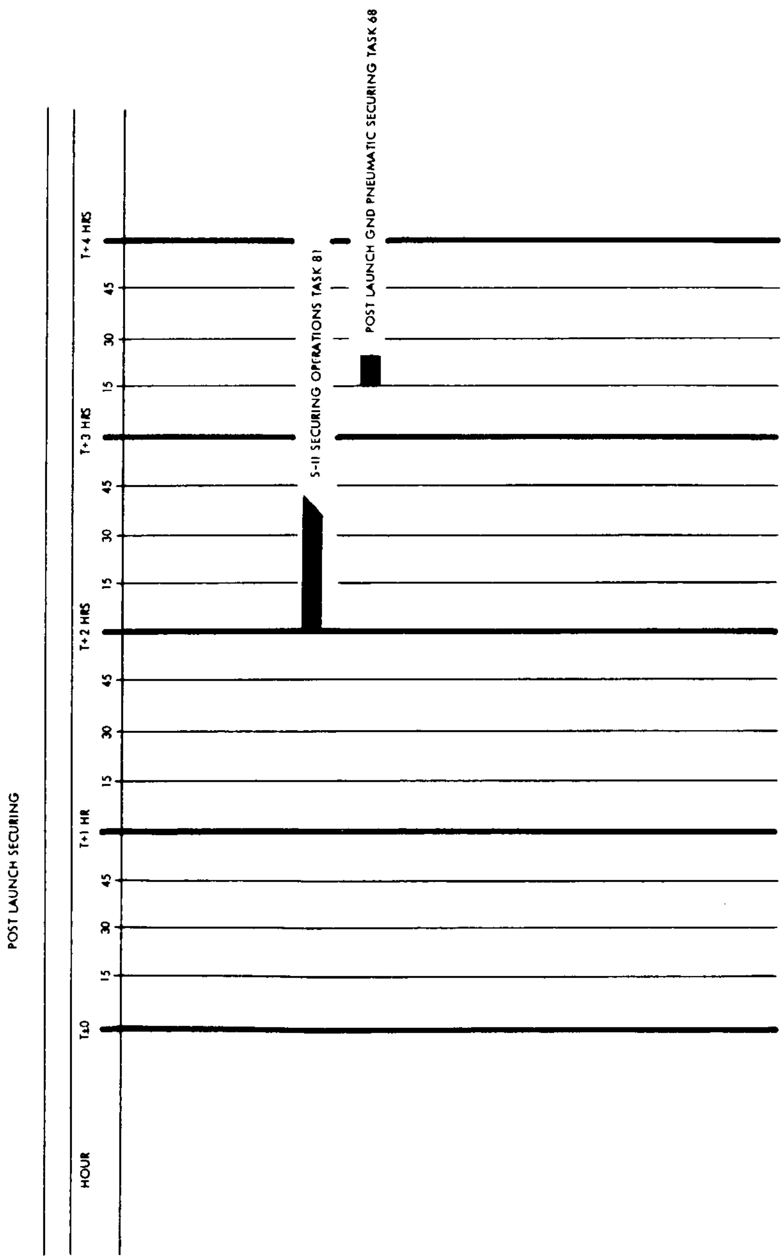

Figure 10. 2-170. NR Tasks for CDDT and Countdown (Sheet 8 of 8)

(Engines 1-5), valve actuation regulator out pressure, helium injection supply pressure, helium injection outlet pressure, and recirculation dc bus voltage. The deletion of these redlines also reduces the number of personnel required to monitor redlines in the Firing Room at the Mission Support Central Instrumentation Facility (CIF) and at the MSFC-Launch Information Evaluation Facility (LIEF).

10.2.10.6.2 Delta - LEO Mission

The addition of the RCS modules requires the following 10 additional redlines: RCS module temperatures (2), helium tank temperatures (2), helium tank pressures (2), fuel supply manifold pressures (2), and oxidizer supply manifold pressures (2). For the thrust vector control system, two additional redlines are required: auxiliary hydraulic pump motor pressurization air tank pressure, and auxiliary hydraulic dc bus voltage. These additional redlines will require an increase in the personnel monitoring tasks.

10.2.11 RELIABILITY ASSESSMENT

In the 1966 preliminary analysis J-2S study, Reference 1.1-2, it was determined that approximately 345, or 30 percent of the S-II stage single point failure modes that could cause mission loss or abort would be eliminated by the stage system's simplifications resulting from incorporation of the J-2S engine. In this present study, this analysis was reviewed and updated as necessary and the benefits of such simplification on the stage quantitative reliability values was then determined. The presently contracted quantitative reliability of the S-II stage (with J-2 engines) is 0.95 - meaning that 95 of every 100 attempted missions will be successful.

10.2.11.1 LOR Mission

Two methods of expressing the S-II stage quantitative reliability increase resulting from incorporation of the J-2S engine were considered. The first method directly evaluated the effect of the reduced number of components and their failure modes on the present reliability value of 0.95. It was determined that the new quantitative reliability value would be 0.954. An overall program cost savings is implied in this increase in stage reliability, but such an evaluation was beyond the scope of the present study.

The second method retained the current reliability value of 0.95, but reapportioned (downward) the required reliability values of each remaining stage system. Such an approach would also imply an overall program cost reduction through the reduced scope of the reliability testing program.

Table 10.2-XLII includes the overall stage reliability values for these two methods for the LOR mission. Typical values of the reapportioned reliability of the stage systems are shown in the "boost period" column of Table 10.2-XLII. The boost period includes the following phases: countdown, S-IC boost, S-IC/S-II separation, and S-II boost.

10.2.11.2 LEO Mission

The same two methods of expressing the S-II stage quantitative reliability changes for the LOR mission were employed for the LEO mission. The reliability values for the boost period are multiplied by the corresponding values for the orbital period to give the overall LEO mission reliability values. In addition to the above noted boost period phases, the orbital period includes the following phases: idle mode shutdown, coasting, and restart/burn in idle mode.

The first method of directly evaluating the effect of the reduced number of components and their failure modes on the current reliability value of 0.95 for the boost period gave a value of 0.954. The corresponding value determined for the orbital period was 0.945. The overall LEO mission reliability would then be 0.902.

With the second method, the present value of 0.95 was retained for the boost period and the remaining stage system apportioned reliabilities could then be reduced. The corresponding relative value for the orbital period was 0.940. The overall LEO mission reliability on this basis would then be 0.893.

Table 10.2-XLII shows the overall stage reliability values for these two methods for the LEO mission as well as for the LOR mission. Table 10.2-XLIII shows the reapportioned system reliability indices for the boost period, orbital period, and for the overall LEO mission.

10.2.12 SUMMARY OF S-II DESIGN MODIFICATIONS

The design changes required to incorporate the J-2S engine into the S-II stage for the LOR mission and those additional changes required to perform the two-stage LEO mission are summarized according to stage systems.

For additional design change definition, refer to the appropriate Engineering discipline section.

10.2.12.1 J-2S/S-II Stage Design Modifications (LOR Mission)

10.2.12.1.1 Engine Servicing System

a. Stage Mechanical Systems. Start tank chill and fill system and start tank vent valve control system will be deleted. The start tank V & R system will be reidentified as fuel pump seal drain system. The current fuel pump seal drain system will be deleted. Thrust chamber chill function will be deleted, and the line insulation removed. The turbopump purge system will be reidentified as LOX dome purge system. On engine/stage fluid interface panels, the center engine customer connect panels are to be reduced from three to one, and interface lines reduced from twelve to six. Interface lines for the outboard engine are to be reduced from thirteen to eight.

b. Stage Electrical Systems. Electrical circuitry for start tank vent control solenoid (stage pneumatics) is to be deleted, as is the electrical circuitry for start tank vent emergency solenoid (engine electrical).

c. Stage Structural Systems. Brackets will be eliminated for deleted stage systems.

Table 10.2-XLII. S-II Stage Quantitative Reliability Values, LOR and LEO Missions

Mission	Boost Reliability 0.95 in All Cases			Boost Reliability 0.95 for Original S-II/J-2 case only		
	Boost	Orbital	Total	Boost	Orbital	Total
J-2/S-II LOR	0.950	-	0.950	0.950	-	0.950
J-2S/S-II LOR	0.950	-	0.950	0.954	-	0.954
J-2S/S-II LEO	0.950	0.940	0.893	0.954	0.945	0.902

Table 10.2-XLIII. Reliability Apportionment Indices

System	Boost Period	Orbital Period	Composite LEO Mission
Electrical Control	0.991805	0.988598	0.980496
Electrical Power	0.997030	0.989638	0.986699
Emergency Detection	0.999947	0.999962	0.999909
Engine Compartment Conditioning	0.999942	0.999999	0.999941
Engine	0.992792	0.991220	0.984075
Engine Servicing	0.999474	0.999994	0.999468
Engine Actuation and Auxiliary Air Atmosphere	0.990640	0.997136	0.987803
Flight Termination	0.998443	0.995513	0.993963
Measurement	0.999970	0.999975	0.999945
Pressurization	0.996653	0.991275	0.987897
Propellant Valve Actuation	0.999247	0.999119	0.998367
Propellant Feed	0.996462	0.994068	0.990551
Propellant Management	0.987838	0.993321	0.981240
Radio Frequency	0.999902	0.999488	0.999390
Reaction Control	0.999741	0.999698	0.999439
Separation	0.999512	0.999999	0.999511
Structure	0.999553	0.999801	0.999354
Thermal Control	0.999955	0.999963	0.999918
LH_2 Balanced Vent	0.999999	0.999970	0.999969
S-II Stage	0.950000	0.940338	0.893321

10.2.12.1.2 Recirculation Systems

a. Stage Mechanical Systems

1. LOX. The five return lines and five valves are to be deleted from engine interface to tank. The helium injection system (tank, regulator, valves, and lines) will be deleted.

2. LH_2. Five pumps, six valves, five bypass lines, return lines, purge line and check valve are to be deleted. LH_2 feed line purge manifold and check valves will be added.

b. Stage Electrical Systems

1. LOX. Circuitry for control of helium injection solenoids will be deleted.

2. LH_2. Circuitry for control of recirculation pumps and monitoring circuitry for pump speed and valve position will be deleted. Recirculation batteries, inverters, filters, transfer switches and bus will also be deleted.

c. Stage Structural Systems

1. LOX. The necessity is eliminated for two special LOX tank bulkhead gore segments, since holes and bosses are not required. The support bracketry for return lines and helium injection systems will be deleted.

2. LH_2. LH_2 tank cylinder 2 will be simplified, eliminating recirculation holes and bosses. The return line fairing will be deleted. The insulation of LH_2 tank cylinder 2 will be simplified, and support bracketry will be eliminated.

10.2.12.1.3 Valve Actuation System

a. Stage Mechanical Systems. Tank pressure will be changed from 3000 to 750 psi for prevalve actuation. The regulator will be deleted, as will the solenoid valves and lines for return and discharge valve control. The relief valve will be replaced. The emergency LH_2 recirculation actuation lines and disconnect will be deleted. By utilizing the helium injection disconnect, the actuation fill disconnect can be deleted.

b. Stage Electrical System. Circuitry for control of recirculation actuation solenoid valves will be deleted.

c. Stage Structural System. Installation of bracketry for regulator, solenoid valves, lines, and disconnect will be eliminated.

10.2.12.1.4 Ullage Motors

a. Stage Mechanical Systems. Four ullage motors will be deleted.

b. Stage Electrical Systems. Power and control circuits will be deleted for ullage motor ignition.

c. Stage Ordnance System. Twenty-one ordnance devices will be deleted.

d. Stage Structural Systems. Ullage motor attach fittings and structural reinforcement will be deleted; cork insulation on interstage will be added.

10.2.12.1.5 Propellant Tank Pressurization

a. Stage Mechanical Systems. Regulators will be modified for higher pressure and flow; pressurization line gimbal joints will be modified for higher pressures. The pressurization (LOX) line will be deleted from center engine.

10.2.12.1.6 Engine Start Control System

a. Stage Electrical System. Ignition power will be added to main bus. Control for duplicate engine start commands will be added, as will control for duplicate mainstage start commands.

10.2.12.1.7 Engine Cut-Off Control System

a. Stage Electrical Systems. Control circuits will be added for LOX low level arm of all engine cut-off from Pc dropout; control circuits will be added to prevent low level arm from initiating all engine cutoff under premature engine-out conditions.

10.2.12.1.8 Thrust Structure

a. Stage Structural Systems. Thickness of stringers, frame longerons, and skin will be increased. Web and cap thickness on center engine support beam will be increased, and local redesign of engine thrust block will be accomplished.

10.2.12.1.9 Flight Control System

The wobbler plate on primary flight control pump will be redesigned.

10.2.12.1.10 Heat Shield

a. Stage Structural Systems. The rigid panels will be redesigned, and stainless wire mesh screen will be added to hot side of rigid panel. Support brackets will be relocated, and flex curtains and attachments will be redesigned to match engine and rigid panel change. Minor modification will be made to basic support tubes and fittings.

10.2.12.1.11 Instrumentation

The following changes are to be made in measurements.

a. Additions

Flight	Ground
34 Temperature	5 Temperature
10 Pressure	10 Position
15 Position	2 Acoustic
65 Discrete	
5 Voltage	
Total 129	Total 17

b. Deletions

Flight	Ground
14 Temperature	24 Temperature
16 Pressure	27 Pressure
15 Position	15 Position
81 Discrete	13 Voltage
1 Voltage	1 Current
1 Current	5 Tachometer
Total 128	Total 85

10.2.12.2 J-2S/S-II Stage Design Modifications (LEO Mission)

10.2.12.2.1 Balanced Vent System

a. Stage Mechanical Systems. Vent system from LH_2 tank will be added (two parallel solenoid valves, vent lines, and two nozzles). A vent system purge, single-line with orifice tapped off vent valve actuation supply, will be added.

b. Stage Electrical Systems. Power and control for vent solenoid valves will be added.

c. Stage Structural Systems. Vent line holes will be added, in forward skirt, and doublers will be added around holes to support nozzles. Support brackets will be added for vent lines. The tank forward bulkhead gore panel will be changed to provide holes and mounting for two solenoid valves, and insulation will be modified for compatibility.

10.2.12.2.2 Thrust Vector Control System

a. Stage Mechanical Systems. Ac motor/pumps will be replaced with dc motor/pumps of the S-IVB type. An air system (tanks, lines, and disconnect) will be added for dc motor compartment atmosphere. A case drain line from auxiliary pump to ARMA will be added, and the hydraulic package panel will be enlarged.

b. Stage Electrical Systems. Electrical power (batteries) and control for four dc motors will be added.

c. Stage Structural Systems. The forward mounting provision on thrust cone will be relocated for modified hydraulic package, and mounting provisions on thrust cone for new battery container will be added.

10.2.12.2.3 Reaction Control System

a. Stage Mechanical Systems. Two reaction control modules (S-IVB/S-V type) will be added. A system to pressurize RCS helium tanks will be added. RCS temperature conditioning system will be added, using ducts from ECCS.

b. Stage Electrical Systems. Electrical power and control for RCS will be added.

c. Stage Structural Systems. Aft skirt structure will be reinforced and mounting provisions will be added for the RCS modules. Holes will be added in the aft skirt for electrical, pressurization, and thermal control lines. Support brackets for electrical, pressurization, and thermal control lines will be added. Insulation shields will be added in areas of thruster impingement, and external cork insulation added fore and aft of RCS modules. A fairing close-out of RCS modules will be added.

10.2.12.2.4 LOX Tank Low Level Sensors

a. Stage Mechanical Systems. LOX sensors will be relocated from sump pedestal to cruciform baffle.

b. Stage Structural Systems. Mounting holes will be provided for sensors on cruciform baffle.

10.2.12.2.5 LOX Pump Seal Drain Outlet

a. Stage Mechanical Systems. Three of five LOX pump seal drain lines will be rerouted.

b. Stage Structural Systems. Holes for three LPSD outlets will be relocated on aft skirt.

10.2.12.2.6 Prevalve Control System.

a. Stage Electrical Systems. Electrical control will be added to disable prevalve closure at all engine cut-off command.

10.2.12.2.7 Idle Mode Shutdown

a. Stage Electrical Systems. Electrical control will be added to provide mainstage shutdown to idle mode from: switch selector command, or LOX or LH_2 (two of five) low level sensor signal. Electrical control will be added to lock up hydraulic accumulators at first burn all-engine cutoff.

10.2.12.2.8 Coast and Restart Controls

a. Stage Electrical Systems. Electrical control will be added to reset engine control relays for restart to idle mode. Electrical control will be added to start TVC auxiliary hydraulic system (three minutes) before restart, and electrical control added to unlock hydraulic accumulators immediately after TVC auxiliary units are on.

10.2.12.2.9 Slosh Control

a. Stage Structural Systems. Conical slosh control baffle will be added in the LOX tank.

10.2.12.2.10 Fairing Venting

a. Stage Structural Systems. Increased venting area will be added for the nose cone portion of the LH_2 feedline fairings.

10.2.12.2.11 Instrumentation

a. Stage Electrical Systems. The following flight measurements are to be added:

16	Temperature
21	Pressure
28	Discrete
1	Position
2	Voltage
2	Current
Total 70	

10.2.12.2.12 Miscellaneous

New main and instrumentation batteries for extended operational period will be added. 207 electrical container for mounting larger battery and 206A31 container to accept new battery will be redesigned. Density and thickness of cork insulation will be increased on all electrical containers.

10.2.12.3 J-2S/S-II GSE Design Modifications (LOR Mission)

10.2.12.3.1 Engine Servicing System

a. Umbilical Carrier Plate, Aft. Start tank vent valve actuation disconnect will be deleted. The fuel pump seal drain (F P S D) disconnect and the start tank chill and fill disconnect will be deleted. Reidentify start tank V and R disconnect to F P S D. Reidentify turbopump purge disconnect to LOX dome purge.

b. LH_2 Heat Exchanger. Thrust chamber chill circuit and start tank chill and fill circuit will be deactivated.

c. Pneumatic Checkout Console. Deactivate the start tank vent control supply, start tank fill supply, start tank fill disconnect relief check supply, fuel pump seal drain supply, and fuel pump seal drain disconnect relief check supply will be deactivated. The LOX recirculation system supply will be redesignated to LOX dome purge supply. Mainstage OK pressure switch, reset level, will be redesignated.

d. Pneumatic Servicing Console. LOX dome purge will be added. The thrust chamber chill, start tank vent valve control, and turbopump purge will be deactivated. The mainstage OK pressure switch level will be modified. Start tank chill, start tank fill, and start tank purge will be deleted.

e. Pneumatic Servicing Control Console. Control and signal conditioning will be changed to be compatible with the changes listed above.

f. Fluid Distribution System. The static firing skirt will be changed by capping and redesignating fluid lines as required for compatibility.

g. Electrical GSE. Electrical control circuitry will be deactivated for the start tank pneumatic vent and emergency vent.

10.2.12.3.2 Recirculation Systems

a. Aft Umbilical Carrier Plate. Recirculation LH_2 line purge disconnect will be redesignated to LH_2 feed line purge disconnect.

b. Pneumatic Checkout Console. LH_2 recirculation system purge manifold and the engine fuel recirculation system supply will be deactivated.

c. Pneumatic Servicing Console. LH_2 recirculation purge system will be redesignated to LH_2 feed line purge.

d. Electrical GSE. Electrical power and control for the LH_2 recirculation pumps, monitoring of pump speed and valve positions, and electrical power and control of helium injection solenoid valves will be deleted.

10.2.12.3.3 Valve Actuation System

a. Umbilical Carrier Plates. The fill disconnect will be deleted. The helium injection disconnect will be redesignated to prevalve actuation disconnect. The emergency LH_2 recirculation actuation disconnect will be deleted.

b. Pneumatic Checkout Console. The valve actuation system fill disconnect actuation supply will be deactivated.

c. Pneumatic Servicing Console. The valve actuation system helium fill disconnect actuation supply will be deactivated. The thrust chamber chill system will be redesignated to prevalve actuation system. The emergency LH_2 recirculation actuation supply system will be deactivated.

d. Electrical GSE. Power and control for recirculation valve actuation solenoid valves and emergency LH_2 recirculation actuation system valves will be deleted.

10.2.12.3.4 Ullage Motors

a. Handling and storage equipment and monitoring of ignition system will

10.2.12.3.5 Propellant Tank Pressurization

a. There is no change to this system.

10.2.12.3.6 Engine Start Control System

a. Electrical power, control, and monitoring of engine start system will be added.

10.2.12.3.7 Engine Cutoff Control System

a. Electrical. Power, control, and monitoring of engine cut-off control will be added.

10.2.12.3.8 Thrust Structure

a. There is no change to this system.

10.2.12.3.9 Heat Shield

a. Interstage Platform Set. The center engine access set will be modified to the revised heat shield configuration.

10.2.12.4 J-2S/S-II GSE Design Modifications (LEO Mission)

10.2.12.4.1 Balanced Vent System

Electrical ground power and control of vent system will be added.

10.2.12.4.2 Thrust Vector Control System

A disconnect for dc motor air supply will be added, as will a system to leak-check air supply system in the pneumatic checkout console. A system to check out and service the air supply system in the pneumatic servicing and control console will be added, and ground power, control, and monitoring of the dc motor/pump.

10.2.12.4.3 Reaction Control System

A disconnect for RCS helium tank pressurization system will be added on umbilical carrier plate. Pressurization system and control for the RCS helium system will be added on the pneumatic servicing and control units. Electrical power, control, and monitoring of RCS stage control relays will be added.

10.2.12.4.4 LOX Tank Low Level Sensors

A system to check out LOX low level sensor mainstage shutdown signal will be added.

10.2.12.4.5 LOX Pump Seal Drain Outlet

There is no change to this system.

10.2.12.4.6 Prevalve Control System

There is no change to this system.

10.2.12.4.7 Idle Mode Shutdown

Electrical power, control, and monitoring will be added for engine idle mode shutdown.

10.2.12.4.8 Coast and Restart Controls

Electrical power and control will be added for engine restart.

10.3 S-IVB STAGE

The distinguishing features of the J-2S engine as based on the Rocketdyne Interface Criteria Document, R-7211 are as follows:

- Increased thrust - 265,000 pounds maximum
- Increased expansion ratio - 40:1
- Idle Mode Operation
- Solid Propellant Turbine Starter
- Thrust Chamber Fuel Jacket Bypass
- Thrust Chamber Tapoff Drive
- Rigid High Pressure Propellant Ducts
- Purge and Drain Changes
- Engine Instrumentation Changes
- LOX Depletion Cutoff Capability
- Rapid Propellant Dump Capability

Each of these features has been closely analyzed to determine its potential application and impact on the S-IVB stage. Details of these analyses are presented in MDAC report DAC-56749, March 1969, a companion publication to this report. As a result of these analyses, feasible applications have been defined to the point where in-depth design changes have been determined.

This section presents a detailed description of the changes to the S-IVB stage and ground support equipment (GSE) resulting from incorporation of the J-2S. A description of the ground test program necessary to verify engine/stage operations and to demonstrate their operational compatibility is included. Weight and reliability data is also furnished. It should be noted that the information presented is quite detailed and is of such a nature that implementation (detail design) could proceed immediately upon authorization. However, the modifications presented are not based on the results of stage development testing but rather on predictions of engine performance; therefore, until an adequate battleship test program has been completed the changes noted in this section cannot be considered final.

Replacement of the J-2 by the J-2S engine results in a simpler S-IVB stage physically and functionally. However, the J-2S results in a more complex stage in that idle mode chilldown represents a new operational mode. For prediction and past flight analyses, idle mode operation requires an effort similar to that for mainstage.

10.3 (Continued)

Physically the J-2S features affect the S-IVB as follows. The increased thrust necessitates strengthening of the thrust structure. As a result of corresponding increased chamber pressure and turbopump speeds, redesign and requalification testing is required for the main hydraulic pump and fuel tank pressurization control systems. Increased expansion ratio has little physical affect on the stage. With similar sea level test side loads as experienced on the J-2, no changes to GSE are required.

Idle mode operation permits deletion of the recirculation chilldown system. Since the engine starts in an idle mode where the conditions of propellants can be of mixed phase, it is no longer necessary to ensure that liquid propellants are at the engine interface during engine start. Therefore, the solid ullage motors and aft firing APS engines are not required for propellant settling acceleration. The engine also has the capability of being shut down from mainstage to idle mode, thereby permitting slosh wave attenuation. Coupled with reduced start requirements, both solid ullage motors and APS ullage engines can be removed.

The Solid Propellant Turbine Starter (SPTS) eliminates several stage/engine and stage/umbilical tower interfaces associated with the J-2 start tank. Additionally, inflight restraints on times between restarts and engine mixture ratio (EMR) have been essentially eliminated.

The thrust chamber fuel jacket bypass permits ambient thrust chamber start conditions, thereby eliminating chilled helium for preconditioning. This along with start tank replacement by the SPTS has resulted in elimination of cold helium for the engine. Successful efforts were made to eliminate the stage cold helium requirements for prepressurization and cold helium bottle fill. As a result, the GSE requirement for cold helium has been eliminated.

Thrust chamber tapoff drive has little affect on the stage other than relaxing the 4.5:1 EMR restraint during engine restart. Rigid high pressure ducts are an engine improvement not directly affecting the stage.

Purge and drain changes have eliminated several stage/engine interfaces and all inflight purges. Flight data may ultimately permit reduction of the stage 4.5 cubic foot pneumatic helium sphere as a result of purge and actuator reductions.

Engine instrumentation changes result in proposed elimination of significant amounts of flight instrumentation. MDAC has expressed concern in the subsequent ability to establish S-IVB operational performance, verify idle mode chilldown models, and identify causes of malfunction should they occur. Discussions have been held with NASA and Rocketdyne on this subject. A recommended instrumentation list is given in Paragraph 10.3.9.

The LOX depletion cutoff capability permits elimination of the LOX low level sensor system. While LOX depletion will not generally be used, and since the S-IVB is controlled by velocity cutoff, it does permit a reduction in unusable propellants at engine cutoff. This is reflected as a pound-for-pound payload

10.3 (Continued)

increase. The rapid propellant dump capability will find application in stage safing and passivation required for orbital rendezvous and Workshop applications. There are no resulting stage hardware changes.

In the following subsections, the stage system changes required for accomplishment of a synchronous mission are also defined. Much of the analyses for these changes has been presented in MDAC report DAC-56584 of July 1967. Stage changes are required for increased coast duration, three engine burns, and higher mission altitude. The major changes are addition of heater blankets, propellant slosh baffles, and increased telemetry equipment.

The J-2S engine, because of its simpler requirements on stage supporting systems, provides an additional degree of flexibility to the S-IVB stage when considering the synchronous and other future missions. For example, in the synchronous mission, three burns result in only two simple engine-required changes. The first is addition of a third Solid Propellant Turbine Starter (SPTS) to the engine and addition of cold helium in the stage. There are other changes required for this mission but they are not associated with the engine operations. If the J-2 engine were used on the S-IVB rather than the J-2S, extending the capability from two to three burns would require not only additional cold helium but more electrical power, plumbing of additional pneumatic pressurant from the stage to the engine, and utilization of significant APS propellant to provide ullaging accelerations during the additional chill period prior to third burn.

The simplification of the stage as evidenced by the significant list of stage deletions noted in this section is accompanied by a significant increase in stage reliability and performance. Also the checkout, launch, and flight operations are simplified. Therefore, the net results of application of the J-2S, results in significant benefits to the S-IVB stage and its program.

10.3.1 Stage Design

A primary purpose of the S-IVB stage (Figure 10.3.1-1) structure is to withstand and transfer the loads created by the payload and the instrument unit to the S-II stage and to maintain structural integrity for the additional loads generated by the S-IVB stage. Local structure must also have the ability to withstand loads caused by the mounting of internal equipment, protuberances, and externally located systems.

The primary load carrying structural subsystem consists of five major subassemblies shown in Figure 10.3.1-2. These assemblies are the forward skirt, propellant tanks, aft skirt, thrust structure, and aft interstage. Structural assemblies which do not carry primary airframe loads are the longitudinal tunnels.

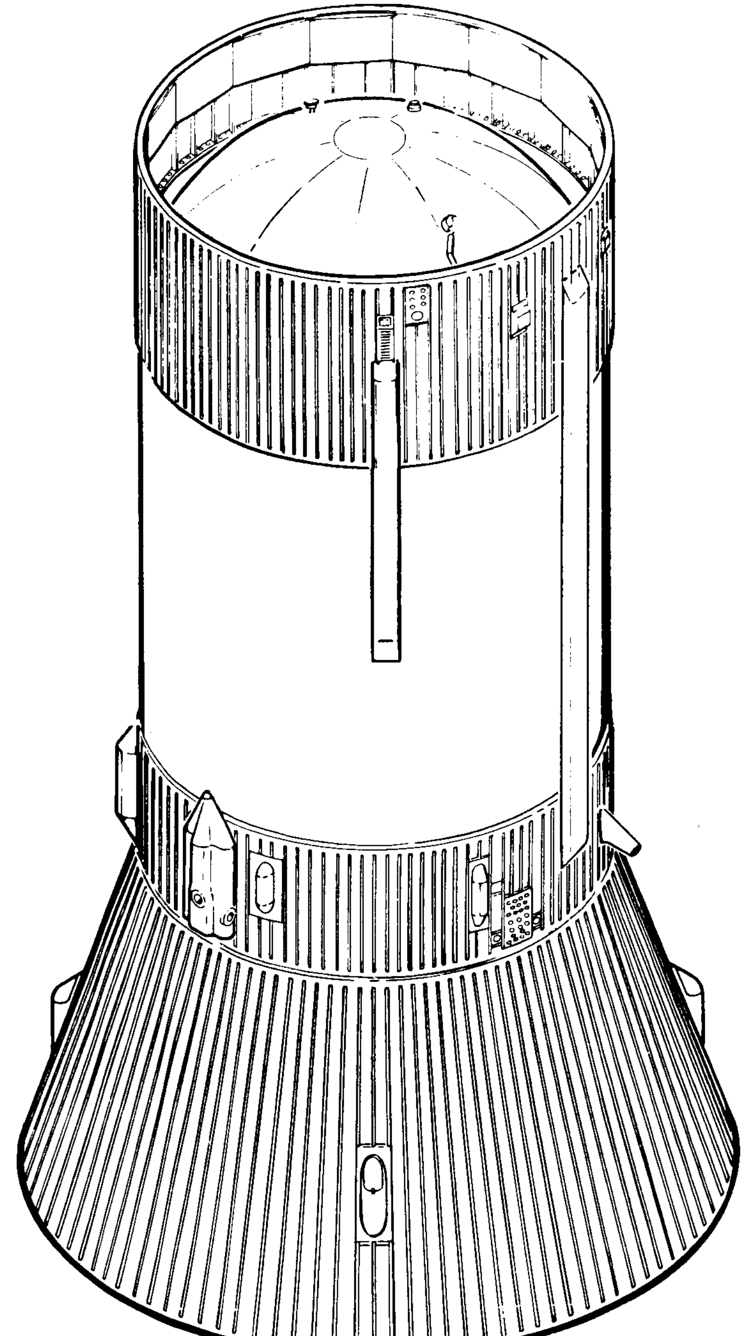

FIGURE 10.3.1-1. S-IVB/SATURN V STAGE

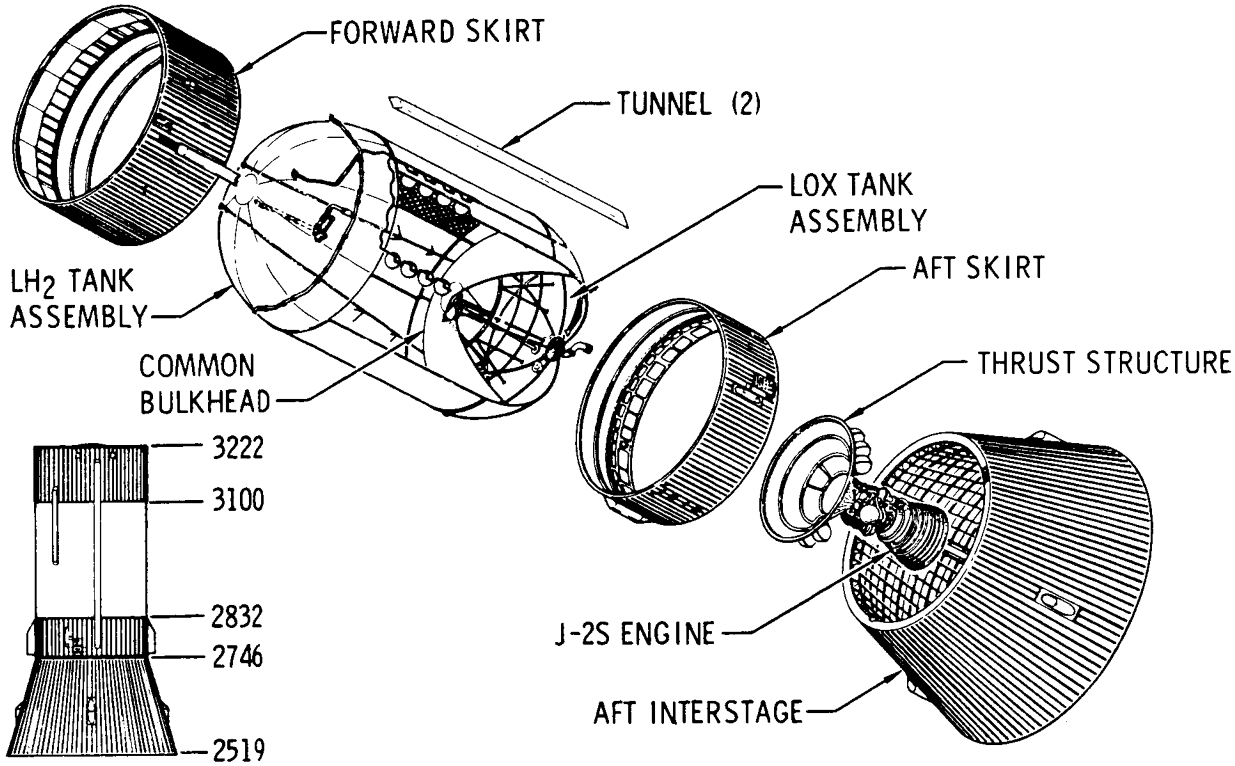

FIGURE 10. 3. 1-2. MAJOR STRUCTURAL SUB-ASSEMBLIES

10. 3. 1. 1 Structures

Structural modifications to the S-IVB stage in conjunction with the implementation of the J-2S engine for the LOR and 3-Burn Synchronous basic missions are discussed in the following paragraphs.

LOR and 3-Burn Synchronous Mission

Due to the elimination of the chilldown recirculation system, four feedthru fittings (LOX and LH_2 chilldown pump line and the LOX and LH_2 chilldown return line fittings) will be deleted from the aft dome assembly. The chemical milled pads in the dome segments (thick pads for welding fittings in place) will be left intact. The LH_2 tank aft dome internal insulation will be modified to be compatible with the LH_2 tank fitting deletions. The LH_2 chilldown pump line and the LH_2 chilldown return line fairings will be deleted from the aft skirt. The retrorocket plume impingement curtains will be modified to delete the feedthru boots required for the chilldown system piping lines. Wiring and piping supports for the chilldown system will also be deleted.

Eliminating the ullage rockets allows the deletion of the ullage rocket fairings support fittings, and replacing heavy duty machined intercostals with lighter weight standard sheet metal intercostals (Figure 10. 3. 1-3). The present location of stringers 44A, 45A, 116A, and 117A and the intercostals attached thereto will not be changed. Wiring supports will be deleted.

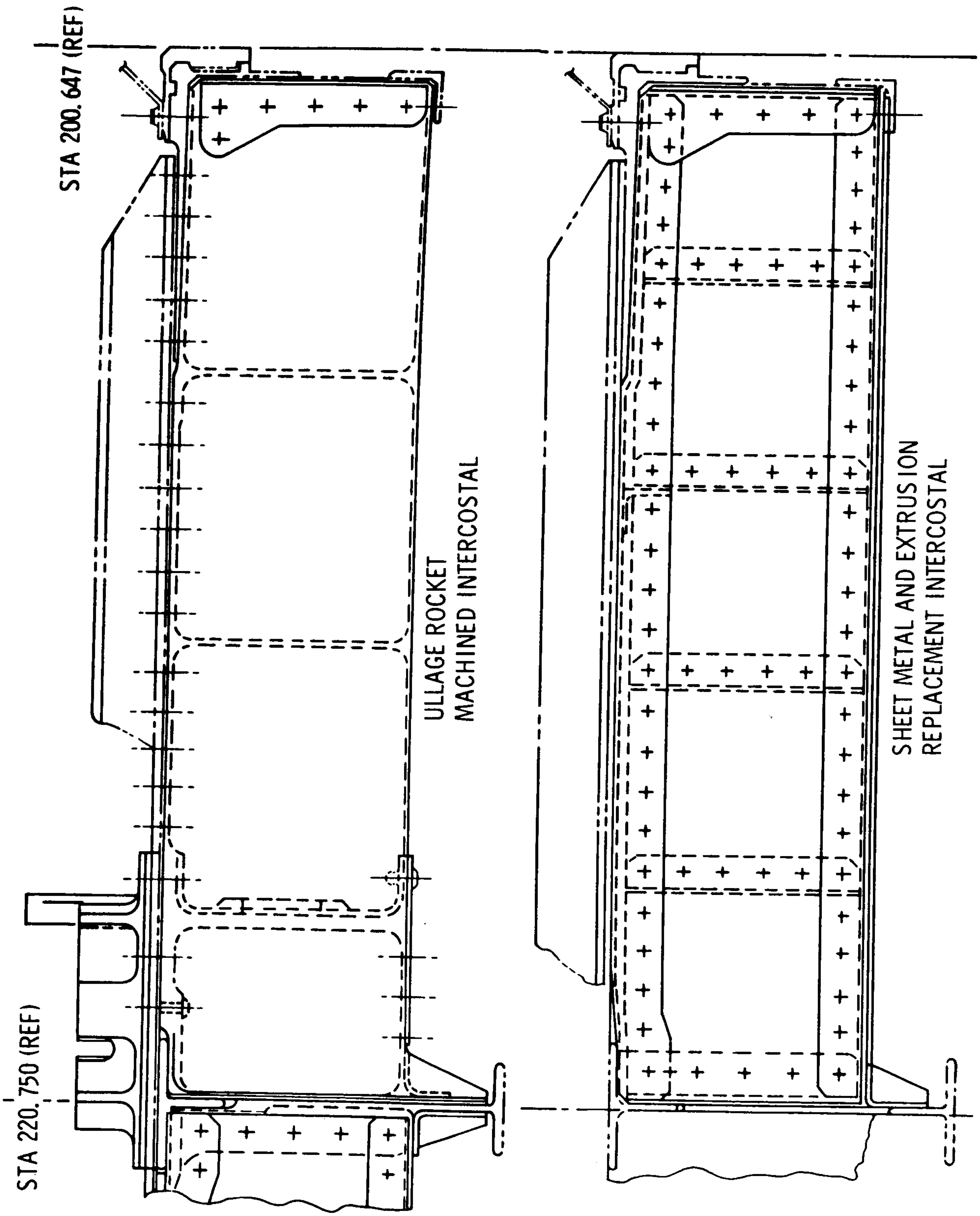

FIGURE 10.3.1-3. ULLAGE ROCKET INTERCOSTALS

10. 3. 1. 1 (Continued)

The thrust structure will be strengthened to sustain the 265, 000 pounds thrust of the J-2S engine by increasing the cross sectional area of the stringers and increasing the thickness of two skin areas shown in Figure 10. 3. 1-4. The existing extrusion will be utilized to fabricate the new stringers as shown in Figure 10. 3. 1-5.

In addition to the above structural changes, a battery mounting adapter will permit the use of a smaller aft number one battery in the aft skirt between stringers number 10 and 14 for LOR missions (Figure 10. 3. 1-6). For the 3-Burn Synchronous mission, the original capability will be retained by removing the adapter.

3-Burn Synchronous Mission

Revisions to the tank repressurization system permits the deletion of seven ambient helium storage spheres and retaining straps (Figure 10. 3. 1-7). The seven ambient helium sphere support pans provided for the LOR mission capability will be left in place. Three cold helium storage spheres will be added in the LH_2 tank. Three positions are presently available in the LH_2 tank for additional spheres; one in the main tunnel area and two in the auxiliary tunnel area. Piping supports will be added as required.

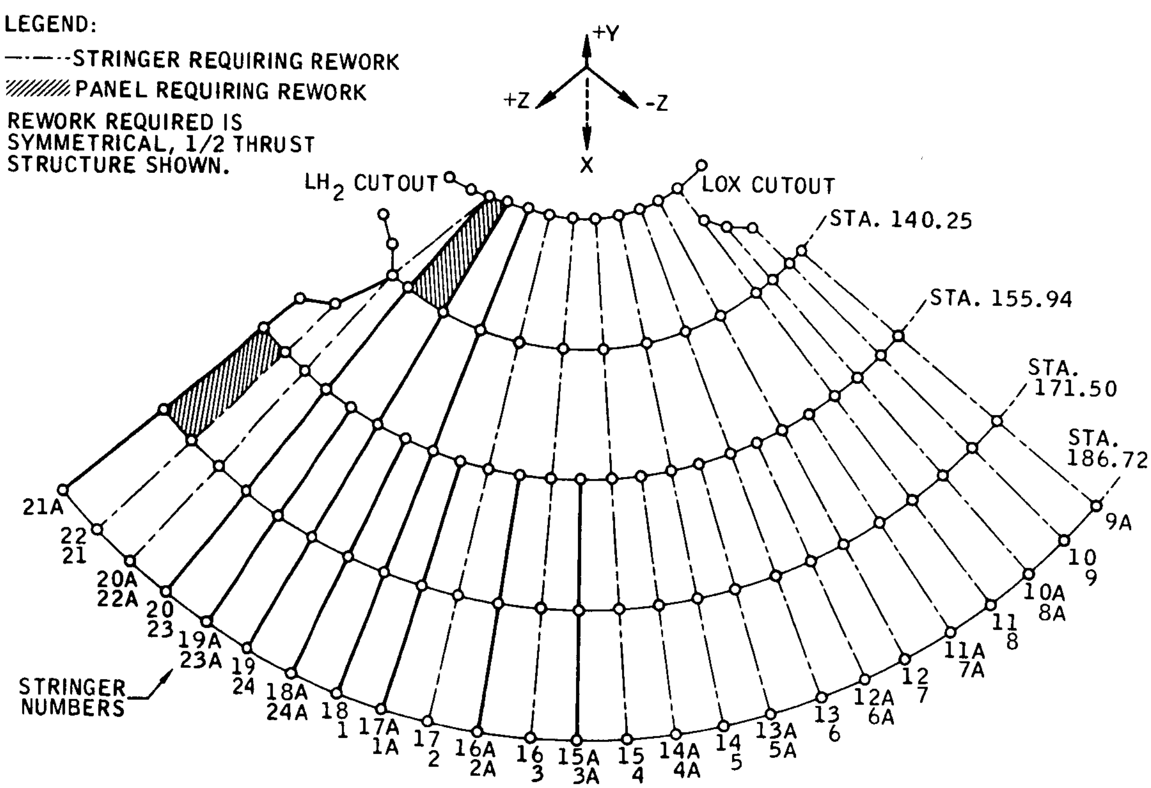

FIGURE 10. 3. 1-4. THRUST STRUCTURE MODEL

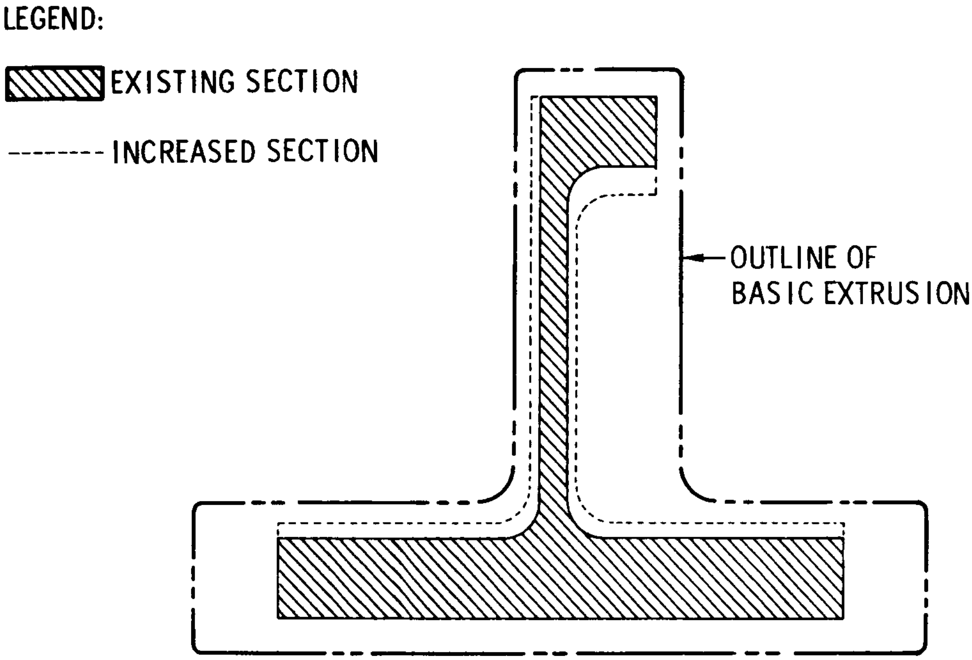

FIGURE 10. 3. 1-5. TYPICAL THRUST STRINGER SECTION

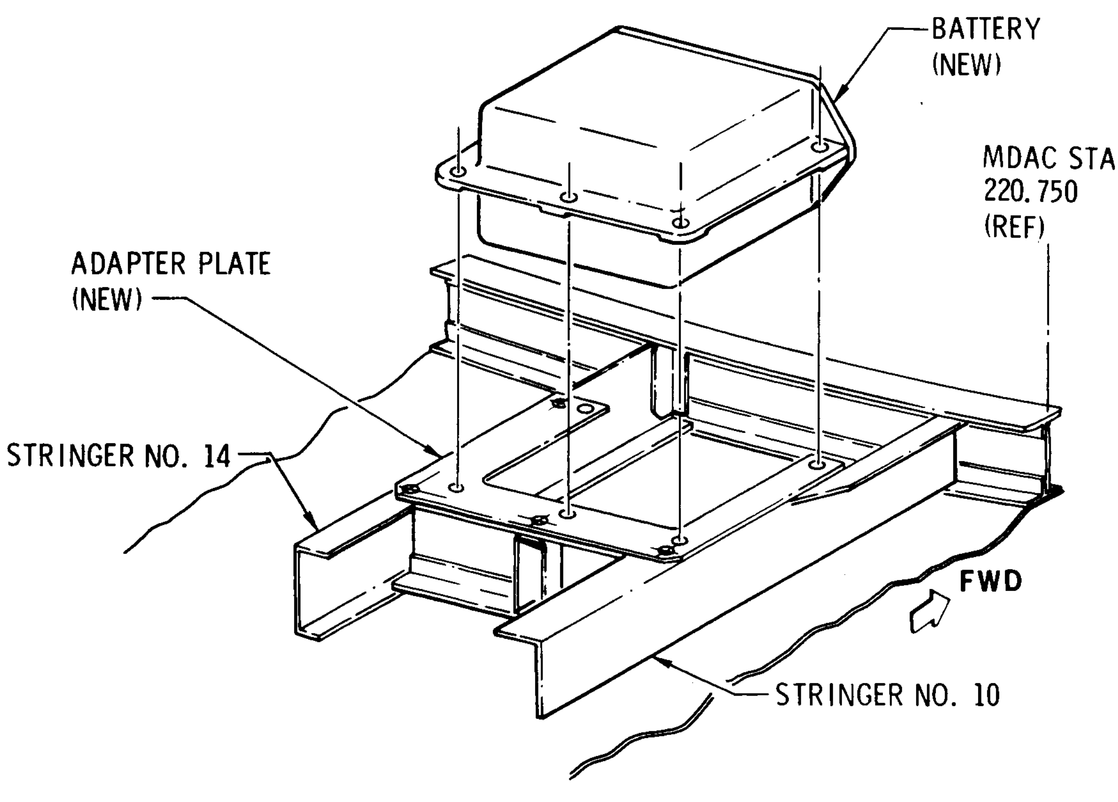

FIGURE 10. 3. 1-6. LOR MISSION-AFT NO. 1 BATTERY ADAPTER

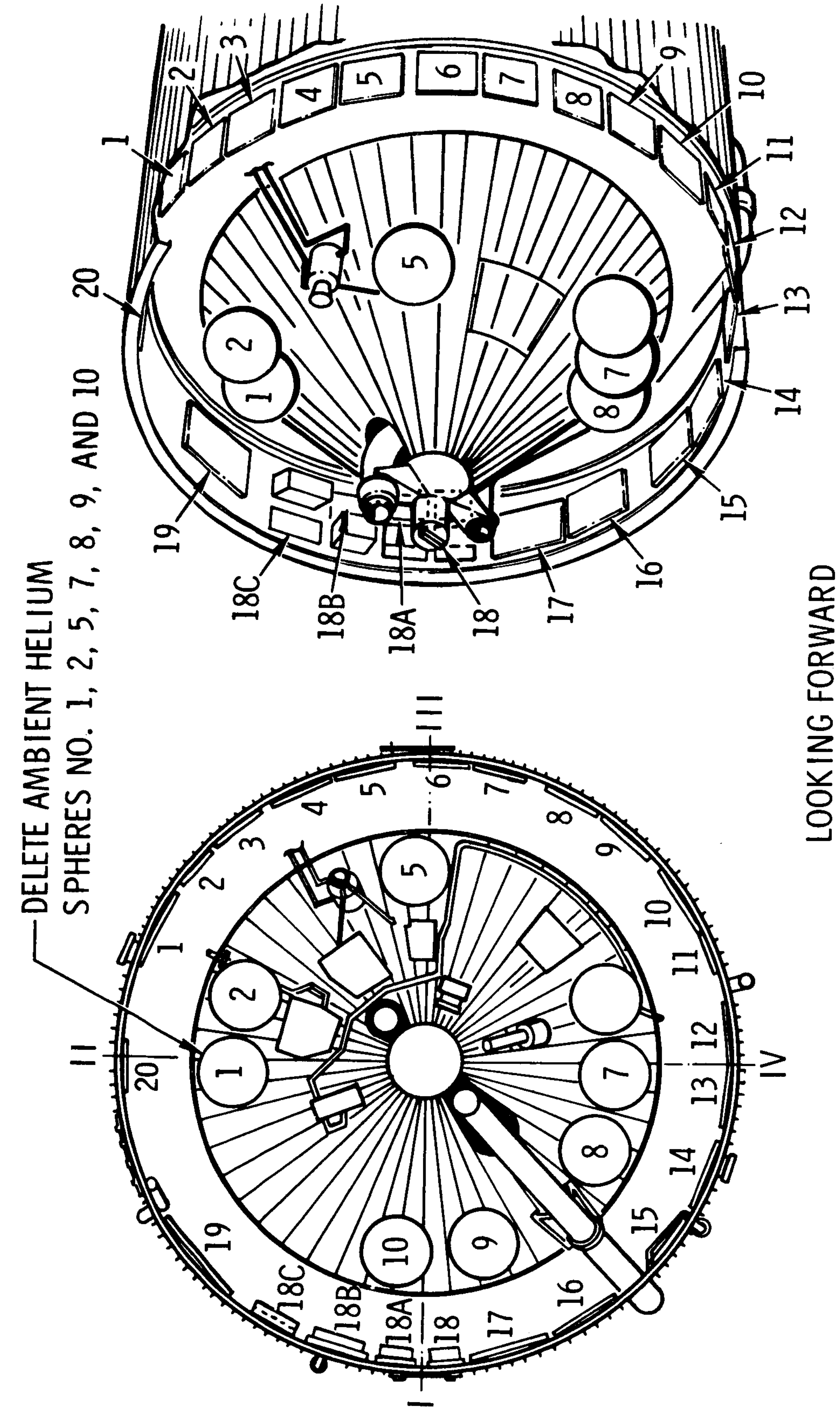

FIGURE 10. 3. 1-7. AFT SKIRT PANELS LOCATION

10. 3. 1. 1 (Continued)

Slosh baffles provided for the LOR mission S-IVB stages will not be compatible with the 3-Burn mission requirements (Figure 10. 3. 1-8), therefore additional baffles and deflectors will be added in the LOX tank and LH_2 tank as shown in Figures 10. 3. 1-9 thru 10. 3. 1-11. Attach points for the new baffles and deflectors have been added by previous authority. The external black and white paint pattern will be revised and low emissivity coatings will be applied to certain internal areas of the skirt structures for thermal control of electrical modules and components. Additional brackets and attach fittings required for piping, wiring, and electrical components unique to the synchronous mission, will be installed on all stages even though they may be used for LOR missions only.

Drawing Changes

Table 10. 3. 1-I lists the structural components affected by the J-2S engine implementation. Parts to be added, deleted, or modified are listed by subsystem and drawing number. The total number and size of the engineering drawings affected are listed below:

Structures Section

24 "J" size revisions
9 "J" size new
11 "D" size new
2 "C" size new
5 "D" size revisions

Installations Section

12 "J" size revisions
2 "J" size new

TABLE 10. 3. 1-I. S-IVB STRUCTURAL DRAWING AND HARDWARE CHANGES

LOR MISSION			
1. Delete chilldown recirculation system (Ref. 1A59098.)			
Qty.	Part No.	Title	Task
-	1B63286	Dome Assy. Aft, LOX Tank	Revise
2	1B63286-3	Flange	Delete
1	1B67065-1	Elbow Assy.	Delete
1	1B67066-1	Elbow Assy.	Delete
1	1B67067-1	Flange	Delete
1	1B67068-1	Flange	Delete
-	1A89613	Insulation Instl. Aft Dome	Revise
1	1B66688-1	Ring	Delete
1	TBD	Pad (Foam, same dia. as 1B66688-1)	Add
1	1B67216-1	Ring	Delete
1	TBD	Pad (Foam, same dia. as 1B67216-1)	Add

TABLE 10. 3. 1-I (Continued)

Qty	Part No.	Title	Task
1	1B65109	Curtain Instl, Retrorocket Plume Impingement	Revise
-	1A39315	Painting and Markings Instl.	Revise
-	1A39303	Tank Assy LOX and LH_2	Revise
-	1A39301	Structures Assy	Revise
-	1A39300	Vehicle Assy	Revise
-	1A39295	Skirt Assy Aft	Revise
2	1A39295-27	Stringer	Delete
1	1A39295-299	Stringer	Delete
1	1A39295-300	Stringer	Delete
1	1A39295-331	Angle	Delete
1	1A39295-332	Angle	Delete
1	1A39295-397	Angle	Delete
2	1A39295-647	Channel	Delete
1	1A68145-1	Fitting	Delete
1	1A79480-503	Fairing Instl	Delete
1	1A79481-1	Fairing Instl	Delete
1	1A84438-1	Fitting	Delete
1	1A39295-217	Skin (Replace with full skin)	Delete
1	1A39295-671	Skin (Replace with full skin)	Delete
1	1A39295-675	Skin (Replace with full skin)	Delete
2	1A39295-3	Stringer	Add
1	1A39295-297	Filler	Delete
1	1A39295-311	Filler	Delete
2	1A39295-711	Filler	Delete
-	1A93255	Angle, Cap	Revise
1	1B44031-1	Skin (Replace with full skin)	Delete
1	1B50906-1	Doubler, Forward	Delete
1	1B54542-1	Fitting	Delete
1	1B63115-1	Doubler, Aft	Delete
2. Delete ullage rockets (Ref 1A82765). See section 10. 3. 2. 6 for propulsion hardware.			
-	1A39295	Skirt Assy Aft	Revise
2	1A93015-1	Bracket	Delete
2	1A93610-1	Bracket	Delete
2	1B27471-1	Fitting	Delete
*2	1B37128-1	Machined Intercostal	Delete
*2	1B37128-2	Machined Intercostal	Delete
2	1B37182-1	Fitting	Delete
2	1B37182-2	Fitting	Delete

*Replace 1B37128-1 and 1B37128-2 machined intercostals with conventional type intercostal between MDAC Station 200 and 220.

TABLE 10.3.1-I (Continued)

Qty	Part No.	Title	Task
2	1B37183-1	Fitting	Delete
2	1B37183-2	Fitting	Delete
2	1B37184-1	Fitting	Delete
2	1B43826-1	Intercostal	Delete
-	1B34789	Structure Instl Ullage Rocket Support	Revise
4	1B34789-5	Shim	Delete
4	1B34789-7	Shim	Delete
2	1B34789-17	Clip	Delete
2	1B34789-18	Clip	Delete
2	1B34789-19	Intercostal	Delete
2	1B34789-23	Clip	Delete
2	1B34789-24	Clip	Delete
4	1B34789-25	Shim	Delete
2	1B34789-27	Spacer	Delete
4	1B34789-29	Shim	Delete
2	1B34789-31	Stiffener	Delete
2	1B34789-33	Doubler	Delete
2	1B34789-43	Clip	Delete
-	1A39301	Structures Assy	Revise
-	1A39300	Vehicle Assy	Revise
-	1A39315	Painting and Markings Instl	Revise
2	1B56624-1	Fairing Instl	Delete
3. Increase load carrying capability of thrust structure for 265,000 pounds thrust.			
1	TBD	Thrust Structure Assy (Similar to 1A39316)	Revise
-	1A39312	Thrust Structure Instl	Revise
-	1A39301	Structure Assy	Revise
-	1A39300	Vehicle Assy	Revise
SYNCHRONOUS MISSION			
1. Delete childown recirculation system (Ref 1A59098), same as for LOR mission. 2. Delete ullage rockets (Ref 1A82765), same as for LOR mission. 3. Increase slosh control capabilities.			
1	1B68278-1	Baffle Instl LOX Tank	Add
1	TBD	Baffle Instl LH_2 Tank, Upper	Add

TABLE 10.3.1-I (Continued)

Qty.	Part No.	Title	Task
1	TBD	Baffle Instl LH_2 Tank, Lower	Add
1	TBD	Deflector Instl LH_2 Tank, Upper	Add
-	1A39315	Painting and Markings Instl	Revise
-	1A39301	Structures Assy	Revise
-	1A39300	Vehicle Assy	Revise
1	1B56518	Baffle and Deflector Instl	Delete or Revise
4. Increase load carrying capability of thrust structure, same as for LOR mission.			
5. Revise tank repressurization system (Ref. 1B58009)			
-	1A57523	Sphere Instl Cold Helium	Revise
3	1A39297-1	Strap Assy	Add
3	1A48858-1	Sphere, Storage, Helium	Add
12	106265-16	Bolt	Add
3	55666-400A1	Gasket	Add
12	MS24665-152	Pin	Add
3	MVC62172-400A	Coupling	Add
12	NS102892-054	Nut	Add
3	S0046T239	O-Ring	Add
24	S0148A056-033P	Washer	Add
24	S0148A056-033R	Washer	Add
-	1B39870	Instl. Ambient Helium Bottle	Revise
7	1B27629-501	Strap Assy	Delete
14	1B27632-1	Fitting	Delete
14	1B27632-2	Fitting	Delete
7	1B66868-501	Sphere Assy	Delete
28	AN310-3	Nut	Delete
28	AN960-10L	Washer	Delete
28	H20-4	Nut	Delete
28	MS24665-151	Pin	Delete
28	NAS1303-7D	Bolt	Delete
28	NAS1351-4-24P	Screw	Delete
28	S0111L17H0458	Washer	Delete
6. Revise stage external surface emissivity value for component thermal control.			
-	1A39315	Painting and Markings Instl	Revise
-	1A39301	Structures Assy	Revise
-	1A39300	Vehicle Assy	Revise
1	TBD	Sphere, Ambient Helium Storage, Coated	Revise

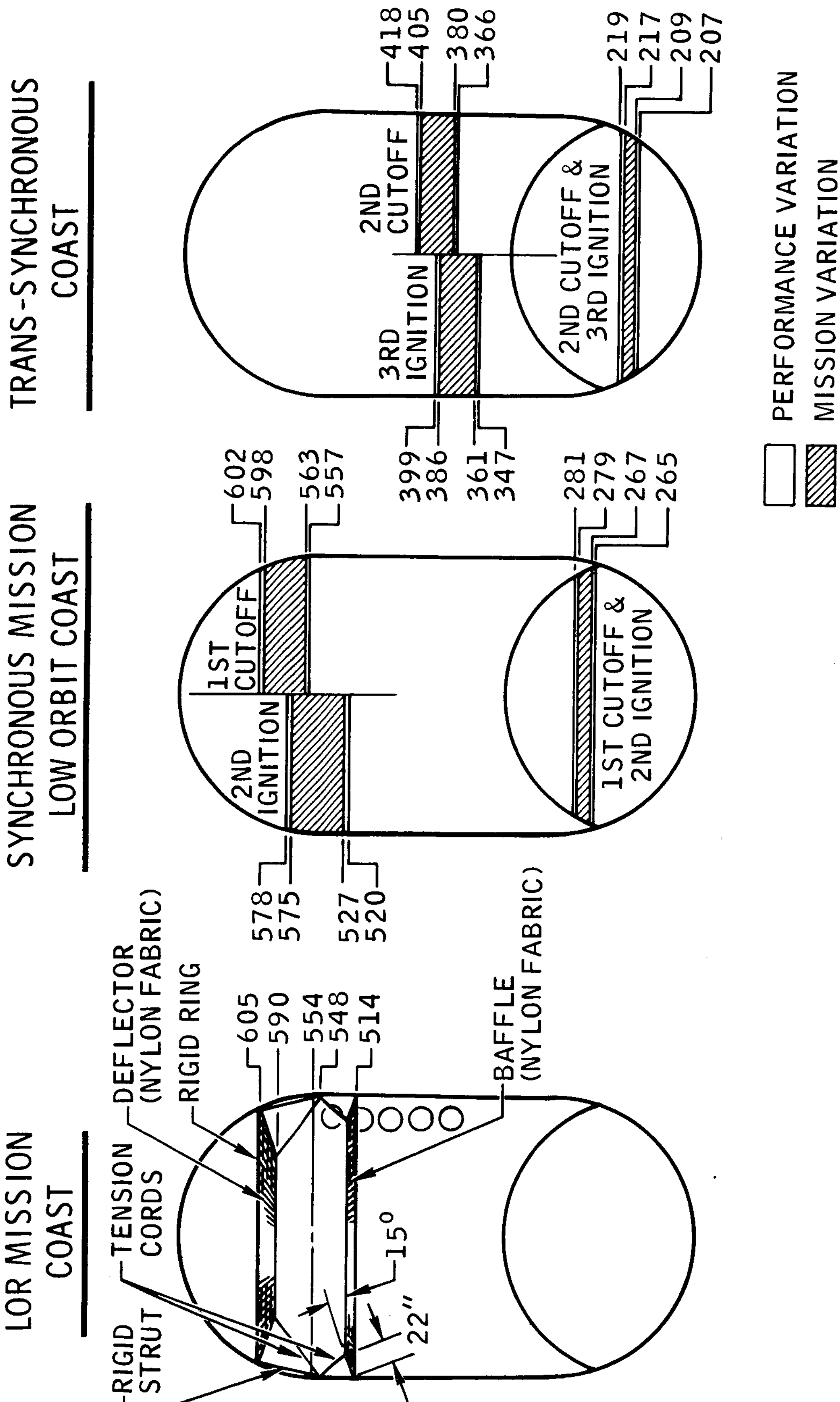

FIGURE 10. 3. 1-8 PROPELLANT TANK BAFFLE, DEFLECTOR AND PROPELLANT LEVEL COMPARISON

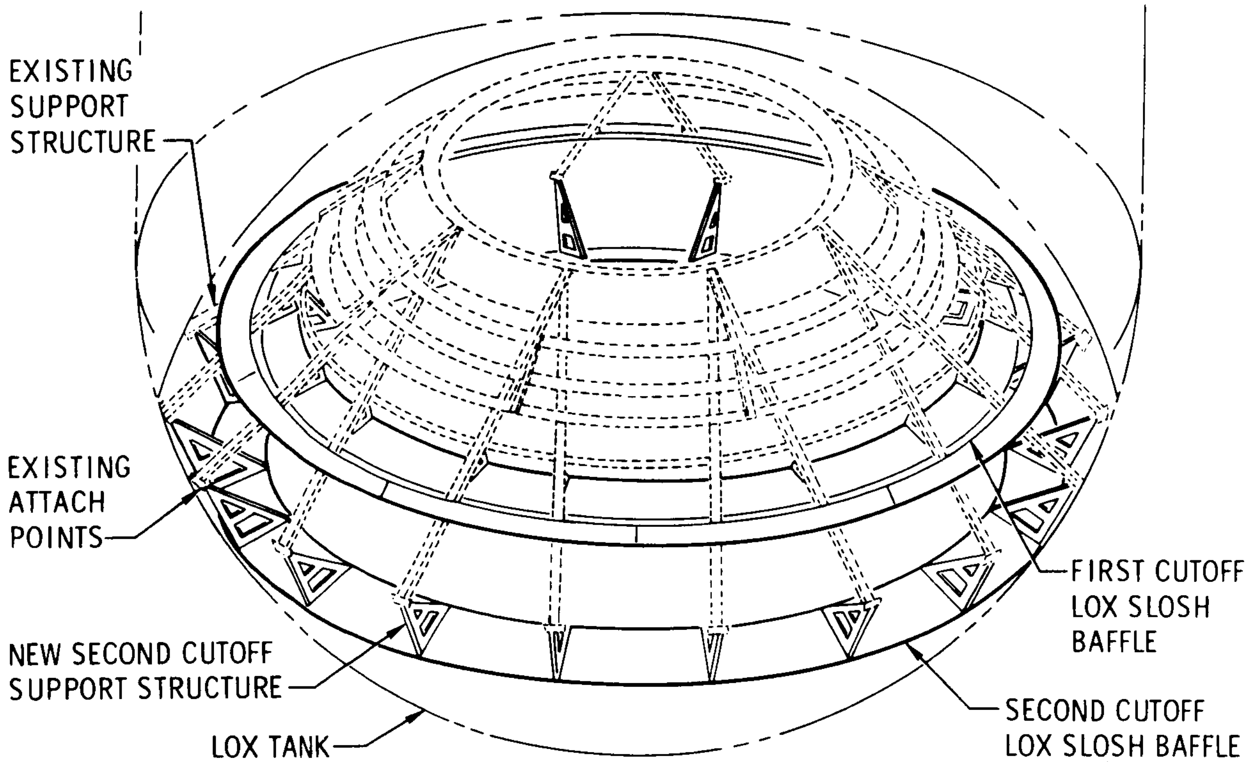

FIGURE 10. 3. 1-9. BAFFLE INSTALLATION-LOX TANK

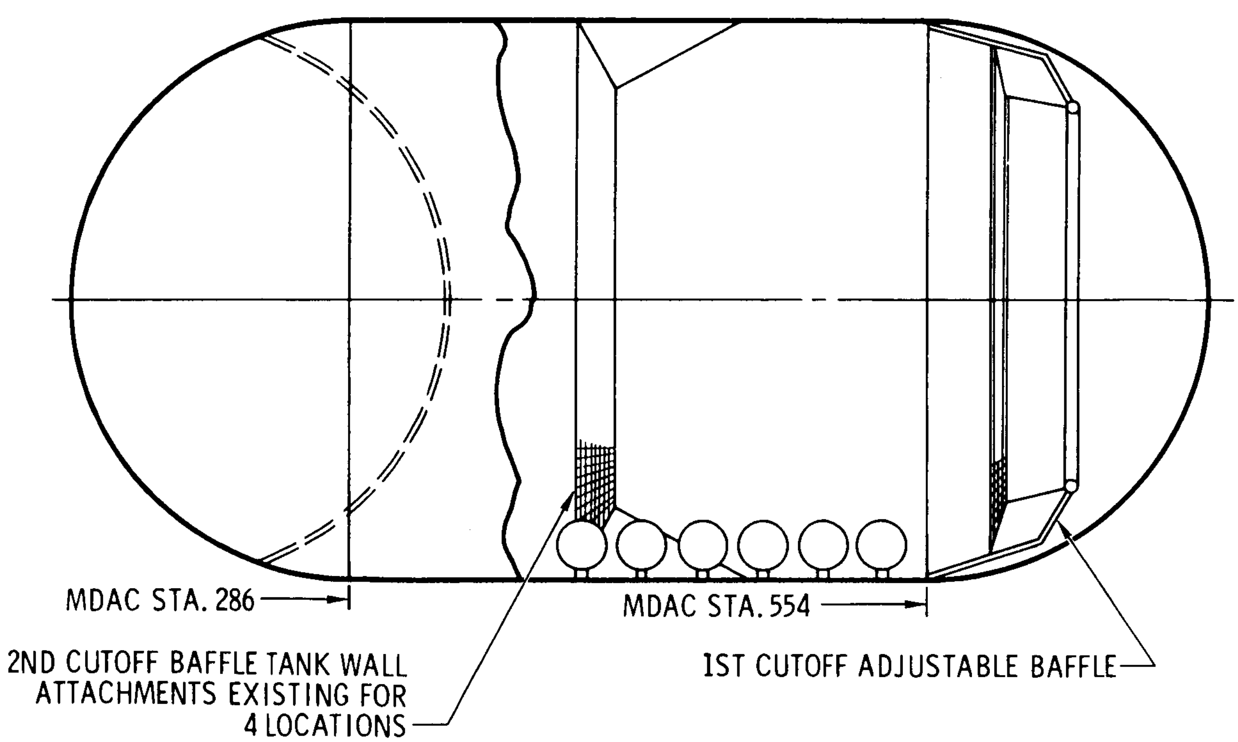

FIGURE 10. 3. 1-10. LH_2 TANK BAFFLE INSTALLATION-3 BURN SYNC MISSION

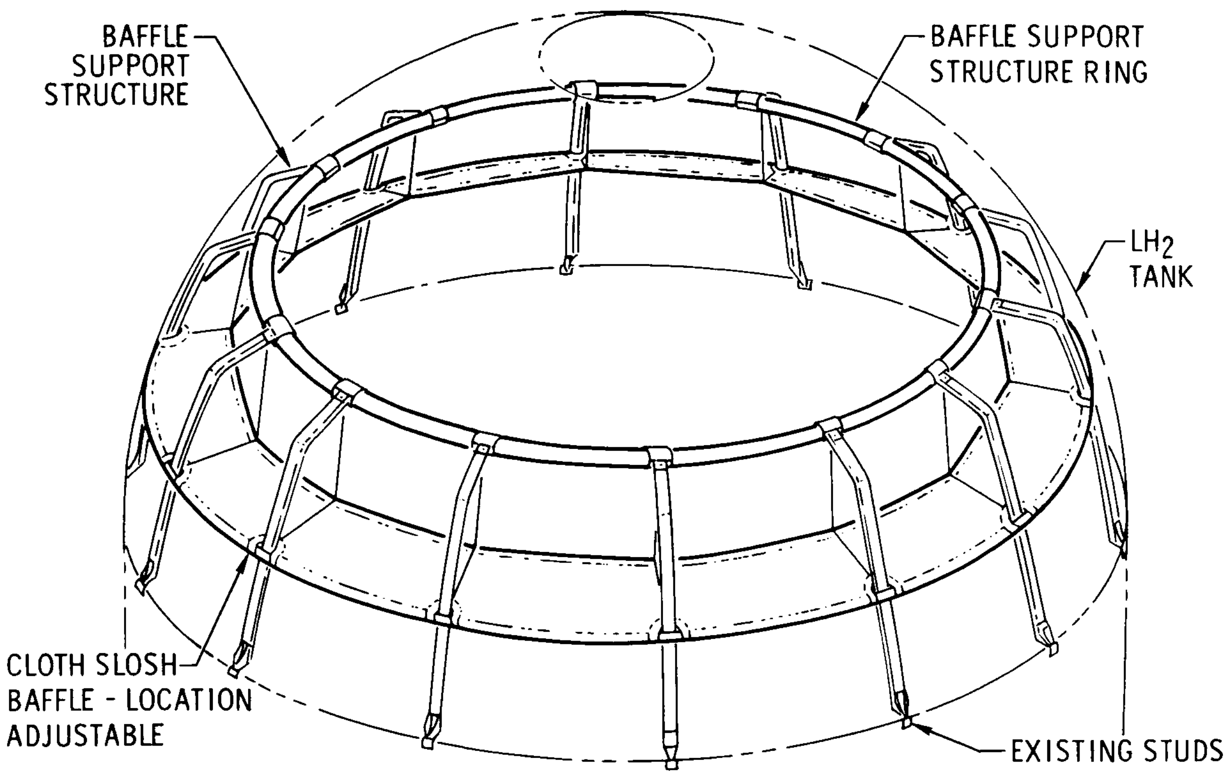

FIGURE 10. 3. 1-11. FIRST CUTOFF LH_2 SLOSH BAFFLE AND SUPPORT STRUCTURE SKETCH

10. 3. 1. 2 Stage Interfaces

This subsection describes the modifications to the customer connect panel and electrical interface (between the J-2S engine and the stage) and the disconnects at the umbilical panel (between stage and ground support equipment (GSE)).

The umbilical panel on the aft skirt (Figure 10. 3. 1-12) will be revised to eliminate the fuel pump drain disconnect, the pre-valve closing backup disconnect, and the start tank vent and relief drain disconnect. This is possible since the start bottle on the J-2S has been replaced with the solid propellant turbine starters. The disconnect left open by the deletion of the hydrogen start tank initial fill system will be utilized for LOX dome purge without the need for requalification.

Table 10. 3. 1-II gives a comparison of design requirements between the J-2 and J-2S engine.

TABLE 10. 3. 1-II. UMBILICAL PANEL LINE COMPARISON

Function	Media	Operating Pressure		Operating Temperature	
		J-2	J-2S	J-2	J-2S
LOX Dome Purge (Was Start Tank Fill	Helium (Was Hydrogen)	1200 to 1400 psia	475 ±25 psig	-300° to -140°F	+50° to +150°F
Start Tank Vent and Relief	Hydrogen Gas	———	Deleted	———	Deleted
Thrust Chamber Purge	Helium	1000 psia Prechill and 55 to 200 psia Purge	150 ±25 psig	-350°F (Prechill) and +50° to +200°F (Purge)	+50°F to +150°F
Prevalve Closing Backup	Helium	———	Deleted	———	Deleted
Fuel Pump Drain	Hydrogen Gas	———	Deleted	———	Deleted

10. 3. 1. 2 (Continued)

The customer connect panel (CCP) for the J-2S engine will not contain ports to accommodate the turbine start bottle vent valve actuation, hydrogen and LOX chilldown systems, or the engine pump purge system. Added to the J-2S panel is the oxidizer dome purge inlet port which was not a requirement on the J-2 engine. Comparison of the J-2S CCP with the J-2 CCP is shown in Figure 10. 3. 1-13. Design requirements for the CCP interface are noted in Table 10. 3. 1-III.

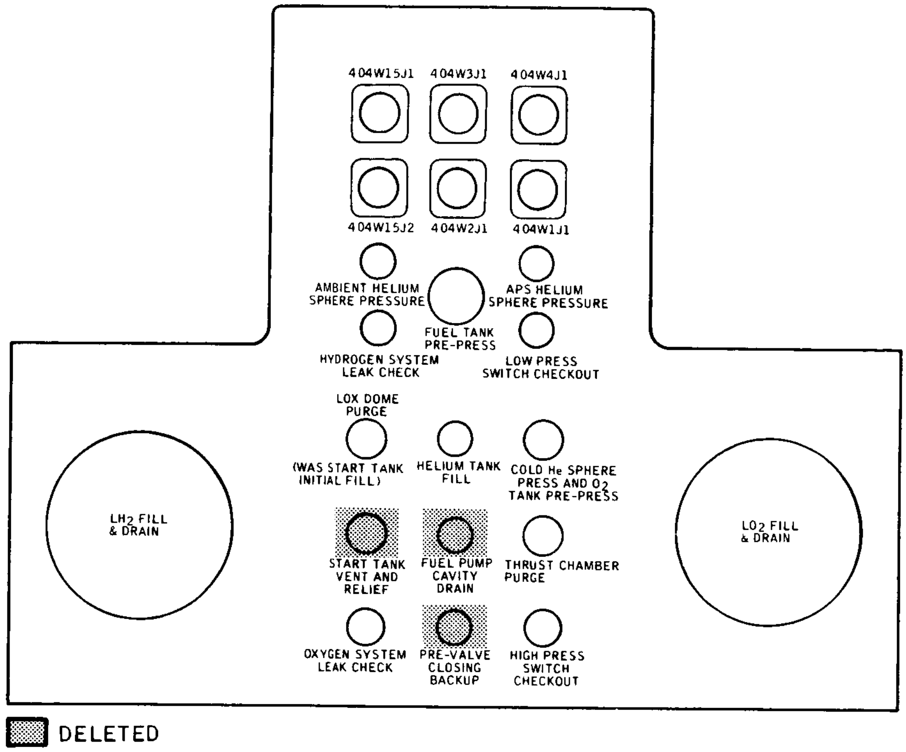

FIGURE 10. 3. 1-12. UMBILICAL PANEL MODIFICATION

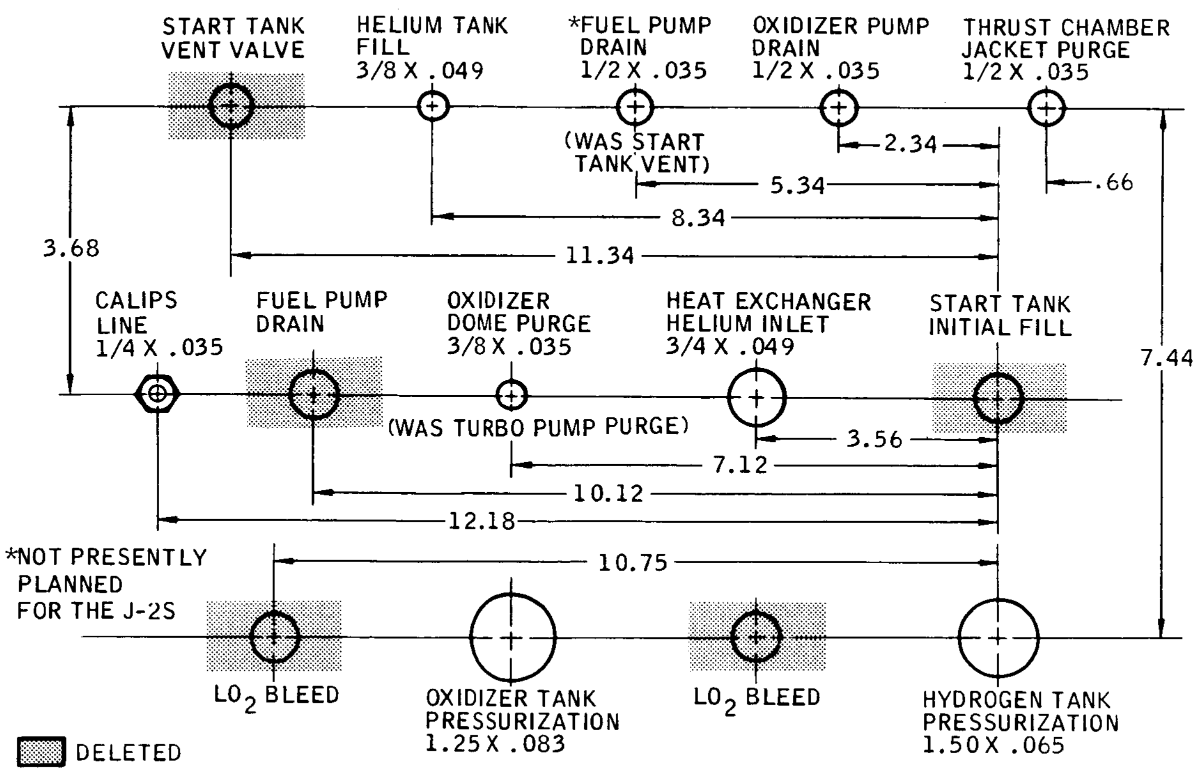

FIGURE 10. 3. 1-13. CUSTOMER CONNECT PANEL REVISIONS

TABLE 10.3.1-III CUSTOMER CONNECT PANEL COMPARISON CHART

Function	Media	Operating Pressure		Operating Temperature	
		J-2	J-2S	J-2	J-2S
Helium Tank Fill System	Helium	3,100 ± 100 psig	3,100 ± 100 psig	-170° to -300°F	Ambient
Fuel Pump Drain *	Gaseous Hydrogen	140 psia	140 psia	-425°F to Ambient	-425°F to Ambient
Oxidizer Pump Drain	Gaseous Oxygen	50 psia	50 psia	-298°F to +140°F	-298°F to +140°F
Thrust Chamber Jacket Purge	Helium	1,000 psia Prechill and 55 to 200 psia Purge	150 ± 25 psig	-350°F (Prechill) +50° to +200°F (Purge)	+50° to +150°F
Calips Line	Helium	3,100 ± 100 psig	3,100 ± 100 psig	Ambient	Ambient
Oxidizer Dome Purge (Not Used On J-2 Engine)	Helium		475 ± 25 psig		50° to 150°F
Heat Exchanger Helium Inlet	Helium	0 to 700 psia	0 to 700 psia	-423°F	-423°F
Oxidizer Tank Pressurization	Helium	150 to 430 psia	150 to 430 psia	-423°F to +715°F	-423°F to +715°F
Hydrogen Tank Pressurization	Gaseous Hydrogen	950 psia Max	1,350 psia Max	-270°F to +165°F	-270°F to +165°F

* Not presently planned for the J-2S.

10.3.1.2 (Continued)

The connectors which comprise the electrical interface between the stage and the J-2S engine are listed in Table 10.3.1-IV with a brief description of the requirements and disposition of these connectors. A detailed breakdown of the differences may be found in Tables 10.3.1-V through 10.3.1-X.

The connectors J13, J15, and J17 on wire harness 403W4 can be deleted because there is no auxiliary instrumentation package on the J-2S engine. The wiring modifications for J14, J16, and J18 of 403W4 and J1 of 403W5 are primarily the result of function switching and do not represent a large problem. The wiring modifications represented by J1 of 403W200 are the result of instrumentation changes. A new connector must be added to wire harness 403W5 to provide new restart control. To implement the additional engine instrumentation not presently on the engine, it will be necessary to add two new connectors to wire harness 403W200.

TABLE 10.3.1-IV. J-2S ENGINE S-IVB STAGE ELECTRICAL INTERFACE

Connector	Disposition
1. Wire Harness 403W4	
J12	No Change
J13	Delete
J14	Modify wiring to add 18 functions and delete 12 functions.
J15	Delete
J16	Modify wiring to add 9 functions and delete 10 functions.
J17	Delete
J18	Modify wiring to add 14 functions and delete 12 functions.
2. Wire Harness 403W5	
J1	Modify wiring to add 12 functions
J2	Delete
J (new)	Add new connector
3. Wire Harness 404W29	
J1	No Change

TABLE 10.3.1-IV (Continued)

4. Wire Harness 403W200	
Connector	Disposition
J1	Modify wiring to add 6 functions and delete 21 functions.
J5	No Change
J (new)	Add new connector.
J (new)	Add new connector.

TABLE 10. 3. 1-V. INDEX OF ABBREVIATIONS FOR TABLES 10. 3. 1-VI THROUGH 10. 3. 1-X

ADD + W	Add a new function and wire.
EXC + W	Delete J-2 function, replace with J-2S function and add a wire.
EXC	Delete J-2 function and replace with J-2S function.
NOM	Nomenclature change.
ADD	Add a new function.
N. C.	No change required.
ADD W	Add wire.
MON	Monitor.
RDNT	Redundant.
SIM	Simulate.
HTR	Heater.
RES	Reserved.
CT	Component Test.
DRD	Dropped.
CMD	Command.
CONT	Control.

TABLE 10.3.1-VI.
J-2S ENGINE/S-IVB STAGE ELECTRICAL INTERFACE DETAILED CHANGES - P54

J-2S Engine Interface Connector	Stage Connector/ Pin	403W4 Wire Harness		Change	Panel Conn Pin	Panel Conn Pin	Black Box Conn Pin
		J-2S Function	J-2 Function				
P54	J14 A	Res, No. 1 Htr Sim Sig	Spare	ADD +W	-		
	B	Res, No. 2 Htr Sim Sig	Meas, Control Assy Temp No. 2	EXC +W	-		
	C	Res, Rdnt Mnst Cutoff Sig	Spare	ADD +W	-		
	D	Mon, Ignition Volt (30V)	Meas, GSE Ign Bus Mon	NOM	J10F	P15F	SP
	E	Mon, He Cont On	Meas, He Cont On	N.C.	J10R	P15R	P3V
	F	Spare Wire	Meas, Cont Assy Temp No. 1	ADD W	-		
	G	Spare Wire	Meas, Cont Assy Temp No. 2	ADD W	-		
	H	Res, Rdnt Eng Rdy Byp Sig	Spare	ADD +W	-		
	J	Mon, Idle Mode Cont On	Meas, Ign Phase Cont On	EXC	J9b		
	K	Mon, Mnstg Start Cont On	Meas, St Tk Disc Vlv Cont On	EXC	J9c		
	L	Sim Sign Mnstg Ok	Cmd. Seq Test Sim Mnstg Ok Press Sw	NOM	J10V	P15V	PIR

TABLE 10.3.1-VI (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W4 Wire Harness		Change	Panel Conn Pin	Panel Conn Pin	Black Box Conn Pin
		J-2S Function	J-2 Function				
P54	M	Sim, Sign-Ign Det 1	Cmd, Seq Test Ign Det Sim	NOM	J10U	P15U	PIS
	N	Mon, No. 2 Htr On (Res)	Meas, Spare Mon	ADD	J9S		
	O	Res, Rdnt Mnstg St Sig	Spare	ADD +W	-		
	P	Mon, Mnstg Cont On	Meas, Mnstg Cont On	N.C.	J9T		
	Q	C.T., He Cont Sol	Cmd, Comp Test He Cont.	NOM	J10a	P1J	
	R	C.T., Idle Mode Sol	Cmd, Comp Test Ign	EXC	J10Z	P15Z	PIK
	S	Sim Sign-Ign Det 2	Spare Wire	ADD	J10T	P15T	PIP
	T	Mon, Mnstg Ok 1 Drd Out	Meas, Mnstg Ok 1 Depres-surized	NOM	J9N		
	U	Mon, Mnstg Ok 1 Drd Out	Meas, Mnstg Ok 1 Depres-surized	NOM	J9U		
	V	Mon, Cutoff Lock-In On	Meas, Eng Cutoff Lock-In	NOM	J9M		
	W	Mon, Ign Det	Meas, Ign Complete	NOM	J9V		
	X	CT No. 2 ASI Spk Sys	Cmd, C.T. Spk Sys 2	N.C.	J10C	P15C	PIC
	Y	CTPwr & Cont Buss Mon (30V)	Supply, C.T. Pwr	NOM	J10d	P15d	P3r

TABLE 10.3.1-VI. (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W4 Wire Harness		Change	Panel Conn Pin	Panel Conn Pin	Black Box Conn Pin
		J-2S Function	J-2 Function				
P54	J14 Z	CT No. 1 ASI Spk Sys	Cmd, Comp Spk Sys 1	NOM	J10X	P15X	PIM
	a	Mon. Mnstg Cutoff Lock-In	Meas, Oxid Turb Byp Vlv Cld	EXC	J9W		
	b	Spare Wire	Meas, Oxid Turb Byp Vlv Op	DEL	J9X		
	c	Mon, Spts Armed	Meas, Spare Mon	ADD	J10N	P15N	P3p
	d	Mon, No. 1 Htr On (Res)	Meas, Fuel Inj Temp Ok No. 1	EXC	J9Y		
	e	Res, Rdnt Eng St Sig	Spare Wire	ADD F	J9Z		
	f	CT Mnstg Cont Sol	Cmd, CT Mnstg Cont	NOM	J10W	P15W	PIL
	g	CT Mnstg St Sol	Cmd, CT St Tk Disch Control	NOM	J10Y	P15Y	PIN
	h	Mon, ASI Spk On	Meas, ASI Spk On	N.C.	J9a		
	j	Mon, SPTS Initiated	Meas, GG Spk On	EXC	J9A		
	k	Mon, No. 1 SPTS Ready	Spare Wire	ADD	J10P	P15P	P3a
	m	GSE Gnd Ref	Supply GSE Gnd Ref	N.C.	J10S	P15S	P3U
	n	Mon, Ign Volt (5V)	Meas, Ign Volt (5V)	N.C.	J5h		

TABLE 10.3.1-VI (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W4 Wire Harness		Change	Panel Conn Pin	Panel Conn Pin	Black Box Conn Pin
		J-2S Function	J-2 Function				
P54	J14 p	Mon, ECA Temp	Meas, Cont Assy Temp No. 1	N.C.	J8V		
	q	Mon, No. 1 SPTS; No. 1 EBW Mon (0-5V)	Meas, No. 1 GG Spk Mon	EXC	J10J	P15J	P3i
	r	Mon, No. 1 SPTS; No. 2 EBW Mon (0-5V)	Meas, No. 2 GG Spk Mon	EXC	J10K	P15K	P3h
	s	Mon, Cont Volt (5V)	Meas, Cont Volt	N.C.	J5g		
	t	Instr Gnd Ref	Meas, Instr Rtn	NOM	J5f		
	u	Mon, ECA Temp	Meas, Cont Assy Temp No. 1	NOM	J8R		
	v	Mon, ECA Temp	Meas, Cont Assy Temp No. 1	NOM	J8P		
	w	Mon, No. 2 ASI Spk	Meas, No. 2 ASI Spk Mon	N.C.	J10H	P15H	P3j
	x	Mon, No. 1 ASI Spk	Meas, No. 1 ASI Spk Mon	N.C.	J10G	P15G	P3k
	y	CT Lockout (Cutoff Sig)	Cmd, CT Lockout	N.C.	-		
	z	Shield Return	Shield	N.C.	-		

TABLE 10. 3. 1-VII. J-2S ENGINE/S-IVB STAGE ELECTRICAL INTERFACE DETAILED CHANGES - P106

J-2S Engine Interface Connector	Stage Connector/ Pin	403W4 Wire Harness		Change	403A61 Panel
		J-2S Function	J-2 Function		
P106	J16 A	Spare	Spare	NC	-
	B	Spare	Spare	NC	-
	C	Spare	Spare	NC	-
	D	Spare	Spare	NC	-
	E	Cmd Dummy Press Xdrc 20 Pct Cal Co Volt In	Cmd, Fuel Pump Interstg 20 Pct Cal	EXC	J1-N
	F	Cmd, Dummy Press Xdcer 80 Pct Cal Co Volt In	Cmd, Fuel Pump Intsg 80 Pct Cal	EXC	J1-P
	G	Spare	Spare	NC	-
	H	Spare	Spare	NC	-
	J	Meas, Dummy Press Xdcr Sig Out	Meas, Fuel Pump Intsg Press	EXC	J5R
	K	Meas, Instr Pkg Temp Res Therm, Sens Out	Meas, Pri Instr Pkg Temp	NOM	J8U
	L	Spare	Spare	NC	-
	M	Supply, Pos 28 Vdc Dup Pwr For Instr Sys	Supply, Pri Inst Pkg 28 Vdc Pwr (Redun)	NOM	J3P

TABLE 10.3.1-VII. (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W4 Wire Harness		Change	403A61 Panel
		J-2S Function	J-2 Function		
P106	J16 N	Supply, 28 Vdc Dup Pwr Instr Sys	Supply, Pri Instr Pkg 28 Vdc Rtn (Redun)	NOM	J3R
	P	Spare	Spare	NC	-
	R	Spare	Supply, Pri Instr Pkg 5 Vdc Pwr (Rdnt)	DEL	J2P
	S	Meas, Pri Instr Pkg Temp Res Therm Out C	Meas, Pri Instr Pkg Temp	NOM	J8T
	T	Meas, Helium Tk 2 Press Xdcr Sig Out	Meas, He Tk 2 Press Xdcr Sig Out	NOM	J5j
	U	Cmd, He Tk No. 1 Press Xdcr 20 Pct Cal-Co In	Cmd, He Tk Press, 20 Pct Calib	NOM	J1r
	V	Cmd, HE Tk 2 Press Xdcr 80 Pct Cal-Co Volt In	Cmd, He Tk Press 80 Pct Calib	NOM	J1q
	W	Cmd, Fuel Pump Disc Press 20 Pct Cal-Co Volt In	Cmd, Fuel Pump Disch Press, 20 Pct Calib	NOM	J1p

TABLE 10.3.1-VII (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W4 Wire Harness		Change	403A61 Panel
		J-2S Function	J-2 Function		
P106	J16 X	Cmd, Fuel Pump Disch Press 80 Pct Cal Co Volt In	Cmd, Fuel Pump Disch Press 80 Pct Calib	NOM	J1n
	Z	Supply, Dup Press Xdcr Sig Rtn Instr Sys	Supply, Dup Press Instr Pkg	NOM	J2N
	a	Meas, Instr Pkg Temp Res Therm In Com	Meas, Instr Pkg Temp	NOM	J8S
	b	Meas, Idle Mode Cmbr Press Xdcr Sig Out	Meas, Eng Start Tk Press	EXC	J5b
	c	Cmd, Idle Mode Chmber Press 20 Pct Calib	Cmd, Eng St Tk Press 20 Pct Calib	EXC	J1m
	d	Cmd, Idle Mode Chmbr Press 80 Pct Calib & Co V Iu	Cmd, Eng St Tk Press 80 Pct Calib	EXC	J1z
	e	Meas, Oxid Pump Disch Press Xdcr Sig Out	Meas, Oxid Pump Disch Press	NOM	J5c
	f	Meas, Fuel Pump Disch Press Xdcr Sig Out	Meas, Fuel Pump Disch Press	NOM	J5a

TABLE 10.3.1-VII (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W4 Wire Harness		Change	403A61 Panel
		J-2S Function	J-2 Function		
P106	J16 g	Cmd, Thrust Chmbr Press Xdcr 20 Pct Calib	Cmd, Thrust Chmber Press 20 Pct Calib	NOM	J1y
	h	Meas, He Tk No. 1 Press Xdcr Sig Out	Meas, Gas Gen Chmbr Press	EXC	J7K
	j	Cmd, He Tk 1 Press Xdcr 20 Pct Cal & Co Volt	Cmd, Gas Gen Chmbr Press 20 Pct Calib	EXC	J1w
	k	Cmd, Oxid Pump Disch Press Xdcr 20 Pct Cal	Cmd, Oxid Pump Disch Press 20 Pct Calib	NOM	J1d
	m	Cmd, Oxid Pump Disch Press Xdcr 80 Pct Calib	Cmd, Oxid Pump Disch Press 80 Pct Calib	NC	J1c
	n	Cmd, Thr Chmbr Press Xdcr 80 Pct Calib	Cmd, Thrust Chmbr Press 80 Pct Calib	NC	J1x
	p	Meas, Thr Chmbr Press Sig Output	Meas, Thrust Chmbr Press	NC	J6K
	r	Cmd, He Tk 1 Press Xdcr 80 Pct Calib & C/O	Cmd, Fuel Turbine 80 Pct Calib	EXC	J1f
	s	Shield Rtn	Shield Rtn	NC	Shld 2

TABLE 10. 3. 1-VIII
J-2S ENGINE/S-IVB STAGE ELECTRICAL INTERFACE DETAILED CHANGES - P107

J-2S Engine Interface Connector	Stage Connector/ Pin	403W4 Wire Harness		Change	Inter- Conn Panel	Inter- Conn Panel	Black Box Conn
		J-2S Function	J-2 Function				
P107	J18 A	Spare	Spare	NC			
	B	Spare	Spare	NC			
	C	Spare	Spare	NC			
	D	Spare	Spare	NC			
	E	Supply, Pos 5 Vdc Exc PU Vlv Pos Pot	Supply, PU Vlv Pos Pot 5 Vdc	NOM			
	F	Spare	Spare	NC			
	G	Spare	Spare	NC			
	H	Supply, 5 Vdc Exc Hot Gas Tap-Off Vlv Pos Pot	Supply, Start Tk Disch Vlv Pos Pot 5 Vdc	EXC	J6-E		
	J	Spare	Spare	NC			
	K	Meas, Volt On PU Vlv Pos Pot	Spare	ADD			
	L	Meas, Volt On PU Vlv For Pos Pot	Spare	ADD			
	M	Supply, Pos 5 Vdc Exc Idle Mode Vlv Pos Pot	Supply, Oxid Turb Bypass Vlv Pos Pot 5 Vdc	EXC	J6-F		
	N	Supply, Pos 28 Vdc Dup Pwr Instr Sys Vlv Pos Sw	Spare	ADD			

TABLE 10.3.1-VIII (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W4 Wire Harness		Change	Inter-Conn Panel	Inter-Conn Panel	Black Box Conn
		J-2S Function	J-2 Function				
P107	J18 P	Spare	Spare	NC			
	R	Spare	Spare	NC			
	S	Supply, Pos 5 Vdc Exc Fuel Byp Vlv Pos	Supply, GG Vlv Pos Pot 5 Vdc	EXC	J6-A		
	T	Meas, Idle Mode Vlv Lmt Sw Closed	Meas, Fuel Bld Vlv Pos, Closed	EXC	J11-E	P16E	P3c
	U	Supply, Pos 5 Vdc Exc Main Oxid Vlv Pos Pot	Supply, Main Oxid Vlv Pos Pot 5 Vdc	NOM	J-6C		
	V	Supply, Pos 5 Vdc Exc Main Fuel Vlv Pos Pot	Supply, Main Fuel Vlv Pos Pot 5 Vdc	NOM	J6-B		
	W	Spare	Meas, ASI Oxid Vlv Pos, Open	DEL	J11-C	P16-C	P3-q
	X	Meas, Idle Mode Vlv Pos Pot Sig Out	Meas, Oxid Turb Byp Vlv Pos Pot	EXC	J5d		
	Z	Meas, Idle Mode Vlv Lmt Sw Open Sig	Meas, Oxid Bld Vlv Pos, Closed	EXC	J11D	P16D	P3d
	a	Meas, Main Oxid Vlv Pos Lmt Sw Open Sig	Meas, Main Oxid Vlv Pos Open	NOM	J9E		

TABLE 10.3.1-VIII (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W4 Wire Harness		Change	Inter-Conn Panel	Inter-Conn Panel	Black Box Conn
		J-2S Function	J-2 Function				
P107	J18 b	Meas, Main Oxid Vlv Pos Lmt Sw Closed Sig	Meas, Main Oxid Vlv Pos Closed	NOM	J10A	P15A	P3a
	c	Meas, Hot Gas Tapoff Vlv Pos Lmt Sw Open Sig	Meas, Start Tank Disc Vlv Pos Open	EXC	J9G		
	d	Meas, Hot Gas Tapoff Vlv Pos Lmt Sw Closed Sig	Meas, Start Tank Disc Vlv Pos Closed	EXC	J11B	P16B	P3f
	e	Meas, PU Vlv Pos Pot Sig Output	Meas, PU Vlv Pos Pot	NC	J5e		
	f	Spare	Spare	NC	Spare	(No Wire)	
	g	Meas, Fuel Byp Vlv Pos Lmt Sw Open Sig	Meas, Gas Gen Vlv Pos Open	EXC	J9J		
	h	Meas, Main Oxid Vlv Pos Pot Sig Out	Meas, Main Oxid Vlv Pos Pot	NC	J5U		
	j	Meas, Hot Gas Tapoff Vlv Pos Pot Sig Out	Meas, Start Tk Disc Vlv Pos Pot	EXC	J5V		

TABLE 10.3.1-VIII (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W4 Wire Harness		Change	Inter-Conn Panel	Inter-Conn Panel	Black Box Conn
		J-2S Function	J-2 Function				
P107	J18 k	Meas, Main Fuel Vlv Pos Lmt Sw Open Sig	Meas, Main Fuel Vlv Pos Open	NOM	J9K		
	m	Meas, Main Fuel Vlv Pos Lmt Sw Closed Sig	Meas, Main Fuel Vlv Pos Closed	NOM	J11K	P16K	P3b
	n	Meas, Fuel Byp Vlv Pos Lmt Sw Closed Sig	Meas, Gas Gen Vlv Pos Closed	EXC	J11A	P16A	P3g
	p	Meas, Fuel Byp Vlv Pos Pot Sig Out	Meas, Gas Gen Vlv Pos Pot	EXC	J5W		
	r	Meas, Main Fuel Vlv Pos Pot Sig Out	Meas, Main Fuel Vlv Pos Pot	NC	J5X		
	s	Shield Return	Spare Wire	ADD W	J10E (Ref)	P15E	SP

TABLE 10. 3. 1-IX. J-2S ENGINE/S-IVB STAGE ELECTRICAL INTERFACE DETAILED CHANGES - P51

J-2S Engine Interface Connector	Stage Connector/ Pin	403W5 Wire Harness J-2S Function	403W5 Wire Harness J-2 Function	Change	BB Conn Pin
P51	J1 A	Eng Cont Pwr (K101)	Supply, Eng Cont	NC	P11H
	B	Eng Cont Pwr (K101)	Supply, Eng Cont	NC	P11K
	C	Res Htr Pwr (-)	Spare	ADD +W	ND
	D	Res, Htr Pwr (-)	Spare	ADD +W	ND
	E	Res, Htr Pwr (-)	Spare	ADD +W	ND
	F	Eng Cont Pwr (K101)	Supply, Cont Pwr	NOM	P11M
	G	Eng Cont Pwr (K101)	Supply, Cont Pwr	NOM	P11R
	H	Res, Htr Pwr (+)	Spare	ADD +W	ND
	I	PNEU Sys Vent	Spare	ADD +W	ND
	J	Spare Wire	Spare	ADD +W	ND
	K	PNEU Sys Vent	Spare	ADD +W	ND
	L	He Vent Cont Sol	CMD Emergency He Vent	NOM	PIT
	M	Res Htr Pwr (±)	Spare	ADD +W	ND
	N	Res, Htr Pwr (+)	Spare	ADD +W	ND

TABLE 10.3.1-IX (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W5 Wire Harness		Change	BB Conn Pin
		J-2S Function	J-2 Function		
P51	O	Ign Pwr (K103)	Supply, Ign Pwr	NC	P4A
	P	DC Gnd Rtn (K105N)	Ign & Cont Pwr Gnd Rtn	NC	403 E13
	R	DC Gnd Rtn (K105N)	Ign & Cont Pwr Gnd Rtn	NC	403 E2
	S	DC Gnd Rtn (K105N)	Ign/Cont Pwr Gnd Rtn	NC	403 E3
	T	DC Gnd Rtn (K105N)	Ign/Cont Pwr Gnd Rtn	NC	403 E4
	U	Ign Pwr (K103)	Supply, Ign Pwr	NC	P43
	V	Ign Pwr (K103)	Supply, Ign Pwr	NC	P4F
	W	Spare Wire	Spare	AW	ND
	X	MNSTG St Sig	CMD, MSTG ENAB	ADD +W	P5g
	Y	Eng Cutoff Mon	Meas, Eng Cutoff ON	NC	P3Y
	Z	Eng Rdy Byp (Cutoff Reset) Sig	CMD, Eng Ready Byp	NC	P5E
	a	Eng Rdy Mon	Meas, Eng Rdy	NC	P3X
	b	MNSTG Ok 2 Mon	Meas, MNSTG Ok 2 Press	NC	P2j
	c	Spare Wire	Spare	AW	ND
	d	MNSTG Cut-off Mon	Spare	ADD +W	ND

TABLE 10.3.1-IX (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W5 Wire Harness J-2S Function	403W5 Wire Harness J-2 Function	Change	BB Conn Pin
P51	e	MNSTG Ok 1 Mon	Meas, MNSTG Ok 1 Press	NC	P2i
	f	Eng Start Sig	CMD, Eng Start	NC	P5V
	g	MNSTG Cutoff CMD	Spare	ADD +W	ND
	h	MNSTG Ok Byp (Throttle ENABLE)	Spare	ADD +W	ND
	i	Eng Cut-off Sig	CMD, Eng Cutoff	NC	P5W
	k	Shield Rtn	Shield Rtn	-	-
	nc		Shield Floating		

TABLE 10.3.1-X. J-2S ENGINE/S-IVB STAGE ELECTRICAL INTERFACE DETAIL CHANGES - P108

J-2S Engine Interface Connector	Stage Connector/ Pin	403W200 Wire Harness J-2S Function	403W200 Wire Harness J-2 Function	Change	Dest Conn Pin
P108	J1 A	Spare	Spare	DEL W	
	B	Spare	Spare	DEL W	
	C	Spare	Spare	NC	
	D	Spare	Spare	NC	

TABLE 10.3.1-X (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W200 Wire Harness J-2S Function	J-2 Function	Change	Dest Conn Pin
P108	E	Spare	Spare	NC	
	F	Spare	Spare	NC	
	G	Meas, HE TK Gas Temp Res. Therm "A" Sens	Meas, Start TK Gas Temp "A" Sensor	EXC	P4-C
	H	Spare	Meas, ST TK Gas Temp "B" Sensor	DEL	
	I	Meas, HE TK Gas Temp Res Therm "A" Sensor	Meas, ST TK Gas Temp "A" Sensor	EXC	P4-E
	J	Meas, HE TK Gas Temp Res.Therm "A" Sensor	Meas, ST TK Gas Temp "A" Sensor	EXC	P4-B
	K	Spare	Meas, ST TK Gas Temp "B" Sensor	DEL	
	L	Spare	Meas, ST TK Gas Temp "B" Sensor	DEL	
	M	Spare	Meas, HE TK Gas Temp	DEL	P5-C
	N	Spare	Meas, Oxid Dis Temp "B" Sensor	DEL	
	O	Spare	Meas, Oxid Pump Dis Temp "B" Sensor	DEL	

TABLE 10.3.I-X (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W200 Wire Harness J-2S Function	J-2 Function	Change	Dest Conn Pin
P108	P	Spare	Meas, Fuel Pump Disc Temp "B" Sensor	DEL	
	R	Spare	Meas, Fuel Pump Disc Temp "B" Sensor	DEL	
	S	Spare	Meas, Oxid Pump Disc Temp "B" Sensor	DEL	
	T	Spare	Meas, HE TK Gas Temp	DEL	P5-E
	U	Spare	Meas, HE TK Gas Temp	DEL	P5-B
	V	Spare	Meas, Fuel Pump Disch Temp "B" Sensor	DEL	
	W	Meas, Oxid Pump Disc Temp "B" Sens Input Common	Meas, Oxid Pump Disch Temp "A" Sensor	NC	
	X	Meas, Oxid Pump Disc Temp "A" Sens Output Common	Meas, Oxid Pump Disch Temp "A" Sensor	NC	P8-E
	Z	Meas, Fuel Pump Disc Temp "A" Sensor	Meas, Fuel Pump Disc Temp "A" Sensor	NC	P12-E

TABLE 10.3.I-X (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W200 Wire Harness J-2S Function	J2 Function	Change	Dest Conn Pin
P108	a	Meas, Fuel Pump Disc Temp "A" Sensor	Meas, Fuel Pump Disc Temp "A" Sensor	NC	P12-B
	b	Meas, Oxid Pump Disc Temp "A" Sensor	Meas, Oxid Pump Disc Temp "A" Sensor	NC	P8-C
	c	Meas, Fuel Turb Inl Temp Common	Meas, Fuel Turb Inl Temp	NC	P1-E
	d	Meas, Fuel Turb Inl Temp	Meas, Fuel Turb Inl Temp	NC	P1-B
	e	Meas, Fuel Pump Disch Temp "A" Sensor	Meas, Fuel Pump Disc Temp "A" Sensor	NC	P12-C
	f	Meas, Oxid Turb Out Temp	Meas, Oxid Turb Out	NC	P10-E
	g	Meas, Oxid Turb Out Temp	Meas, Oxid Turb Out Temp	NC	P10-B
	h	Meas, Oxid Turb In Temp	Meas, Oxid Turb In Temp	NC	P2-E
	j	Meas, Oxid Turb In Temp	Meas, Oxid Turb In Temp	NC	P2-B

TABLE 10.3.I-X (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W200 Wire Harness J-2S Function	J2 Function	Change	Dest Conn Pin
P108	k	Meas, Fuel Turb In Temp	Meas, Fuel Turb In Temp	NC	P1-C
	m	Meas, Oxid Turb Out Temp	Meas, Oxid Turb Out Temp	NC	P10-C
	n	Spare	Meas, Thrust Chmbr Jkt Temp No. 1	DEL	
	p	Spare	Meas, Thrust Chmbr Jkt Temp No. 1	DEL	
	r	Meas, Oxid Turb Inlet Temp	Meas, Oxid Turb Inl Temp	NC	P2-C
	s	Meas, Main Fuel Inj No. 2 Temp	Meas, Thrust Chmbr Jkt Temp No. 2	EXC	P6-C
	t	Meas, Main Fuel Inj Temp No. 1	Meas, Fuel Inj Temp	NOM	P7-C
	u	Spare	Meas, Thrust Chmbr Jkt Temp No. 1	DEL	
	v	Meas, Main Fuel Inj Temp No. 2	Meas, Thrust Chmbr Jkt Temp No. 2	EKC	P6-E
	w	Meas, Main Fuel Inj Temp No. 1	Meas, Fuel Inj Temp	NOM	P7-E

TABLE 10.3.I-X (Continued)

J-2S Engine Interface Connector	Stage Connector/ Pin	403W200 Wire Harness J-2S Function	J2 Function	Change	Dest Conn Pin
P108	x	Meas, Main Fuel Inj Temp No. 2	Meas, Thrust Chmbr Jkt Temp No. 2	EKC	P6-B
	y	Meas, Main Temp Inj Temp No. 1	Meas, Fuel Inj Temp	NOM	P7-B
	z	Shield Rtn	Shield Rtn	NC	

10.3.2 Propulsion/Mechanical Systems

The propulsion systems of the S-IVB stage provide thrust for major vehicle velocity increases, attitude control, ullaging of propellants, and S-II/S-IVB separation. Figure 10.3.2-1 presents the overall stage main propulsion schematic as modified for incorporation of the J-2S engine. Succeeding schematics show detailed subsystem comparisons between the J-2 and J-2S versions of the S-IVB. Only those systems affected by the J-2S or synchronous mission are discussed in the following sections.

S-IVB mechanical systems are comprised of flight control, environmental control, and ordnance systems. Flight control hydraulic gimbaling is affected by changes in engine mass, thrust, and turbine speed. As a result of J-2S idle mode operation, the auxiliary hydraulic system must assume increased functions as the sole means of engine gimbaling during idle mode. Environmental control systems are chiefly affected by the extended stage lifetime needed for the synchronous orbit mission. The ordnance system comprised of solid ullage motor ignition, jettison hardware and J-2S solid propellant turbine starter ignition. Deletion of the ullage motor system hardware is partially offset by addition of the J-2S Solid Propellant Turbine Spinner (SPTS).

10.3.2.1 Engine Fluid Service Systems

The primary features of the J-2S engine that directly affect the stage engine fluid service system are deletion of the hydrogen start tank, fuel pump drain, and the addition of a LOX dome purge requirement.

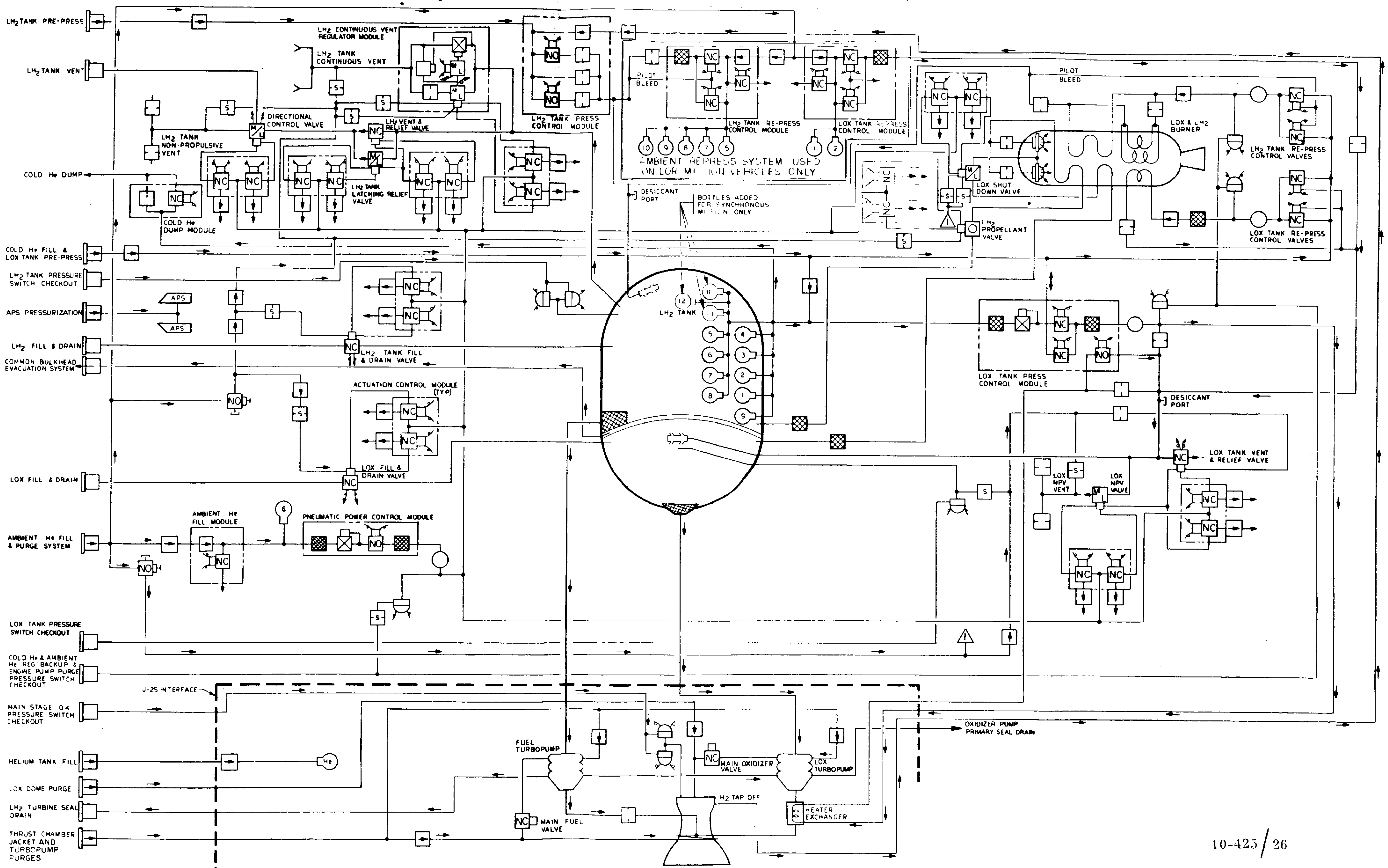

FIGURE 10. 3. 2-1 SCHEMATIC PROPULSION/MECHANICAL

10.3.2.1 (Continued)

Deletion of the hydrogen start tank allows the removal of the vent and relief line and disconnect between the umbilical panel and the Customer Connect Panel (CCP) (Figure 10.3.2-2). The vent and relief line and disconnect will also be removed between the carrier and CCP. The start tank emergency vent valve on the J-2 engine has been removed from the J-2S but no hardware change is necessary to the stage.

To purge the LOX dome, the disconnect that was previously used for start tank initial fill will be utilized. Tube assemblies making up the system from the umbilical panel to CCP will contain flexible hose assemblies. These disconnect and hose assemblies operating pressures are compatible with the requirement of the LOX dome purge and will not require requalification testing. The LOX dome purge will utilize the manifold purge connection left vacant on the CCP.

Sequencing will be revised from the present AS-511 specification as shown in Table 10.3.2-I and hardware changes are noted in Table 10.3.2-II. Stage configuration for the Synchronous Mission is the same as the LOR and data in the above mentioned tables are applicable to both missions.

The engine system H&CO drawing, 1B59461, J-2 Engine Leak Check, currently in use, would be deleted for Huntington Beach Facility VCL checkout. A new H&CO procedure for J-2S checkout would be required. A major portion of 1B66572, Propulsion System Test - SV, VCL, pertaining to engine testing, would need to be rewritten and reprogrammed.

10.3.2.2 Propellant Fill and Feed Systems

The J-2S engine is capable of operating on propellant tank ullage pressure alone (idle mode), without the highly pressurized propellant from the turbo-pumps which the standard J-2 engine requires. Idle mode operation can be achieved on hot gases, mixed phase, liquid, or any combination of these oxidizer and fuel conditions.

The thrust level in steady-state idle mode operation, with nominal tank operating pressures and liquid propellants is about 5000 pounds. However, for restart, the engine will be required to operate initially on gases and mixed phase propellant due to warm propellant feed ducts. Such operation will produce a thrust of approximately 1000 pounds, increasing gradually until liquid propellants have entered the engine and steady state idle mode operation is achieved. Propellant flow during idle mode operation chills down the engine and propellant feed system, thereby performing the current recirculation chilldown function. With the idle mode operation capable of performing the chilldown of each start, the existing LOX and LH_2 chilldown systems can be removed from the stage entirely as shown in Figures 10.3.2-3 and 10.3.2-4.

Deletion of the chilldown system eliminates the need for the LOX and hydrogen prevalves. These valves displace five inches of the feed duct system and will be replaced with short duct lengths (spacers), Figure 10.3.2-5. To avoid requalification testing involving system vibration, the spacers will be designed to the same weight as the prevalves, thereby not significantly affecting the system dynamic response characteristics.

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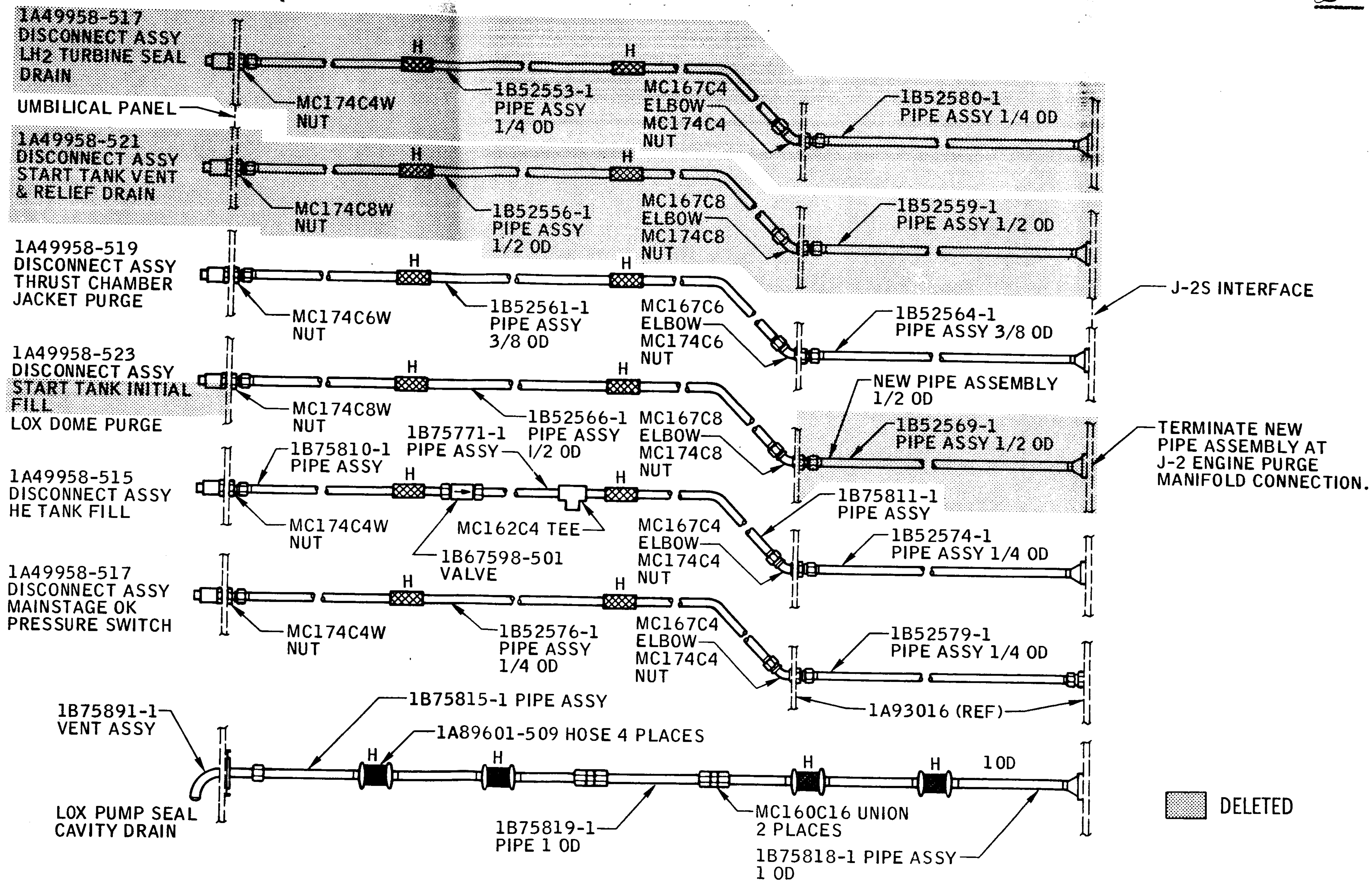

FIGURE 10.3.2-2. SCHEMATIC, ENGINE FLUID SERVICE SYSTEMS

TABLE 10.3.2-I. ENGINE SYSTEM LAUNCH SEQUENCE REVISIONS LOR AND SYNCHRONOUS MISSIONS

Event	Time (Hr:Min:Sec)	Change
Open Start Tank Vent	T - 00:20:00	Delete
Open Start Tank Purge	T - 00:20:00	Delete
Close Start Tank Supply	T - 00:14:30	Delete
Open Start Tank Supply	T - 00:14:30	Delete
Close Start Tank Vent	T - 00:05:30	Delete
Close Start Tank Supply	T - 00:05:00	Delete
Open Start Tank Supply Vent	T - 00:05:00	Delete
Start LOX Dome Purge	T - 07:30:00	ADD
Stop LOX Dome Purge	T - 07:00:00	ADD

TABLE 10.3.2-II. ENGINE SYSTEM HARDWARE CHANGES-LOR AND SYNCHRONOUS MISSIONS (Ref Dwg 1A66894)

Item	Pipe Length in.	Part Number	Qty	Change
LOX Dome Purge		To Be Determined		New Design
Pipe Assembly	132*	1B52556-1	1	Delete
Pipe Assembly	35**	1B52559-1	1	Delete
Nut		MC174C8W	2	Delete
Elbow		MC167C8	2	Delete
Pipe Assembly	132*	1B52553-1	1	Delete
Pipe Assembly	35**	1B52580-1	1	Delete
Nut		MC174C6	1	Delete
Hose Assembly		1A48852-507	2	Delete
Flange		1A89733-503	2	Delete
Pipe Assembly	35	1B52569-1	1	Delete

NOTE: Pipe Assemblies contain 1/2 x 0.028 Dia tubing with standard MC end fittings unless otherwise specified.

*Pipe assembly contains one 1A48852-507 hose assembly.
**Pipe assembly contains 1A89733-503 flange on one end.

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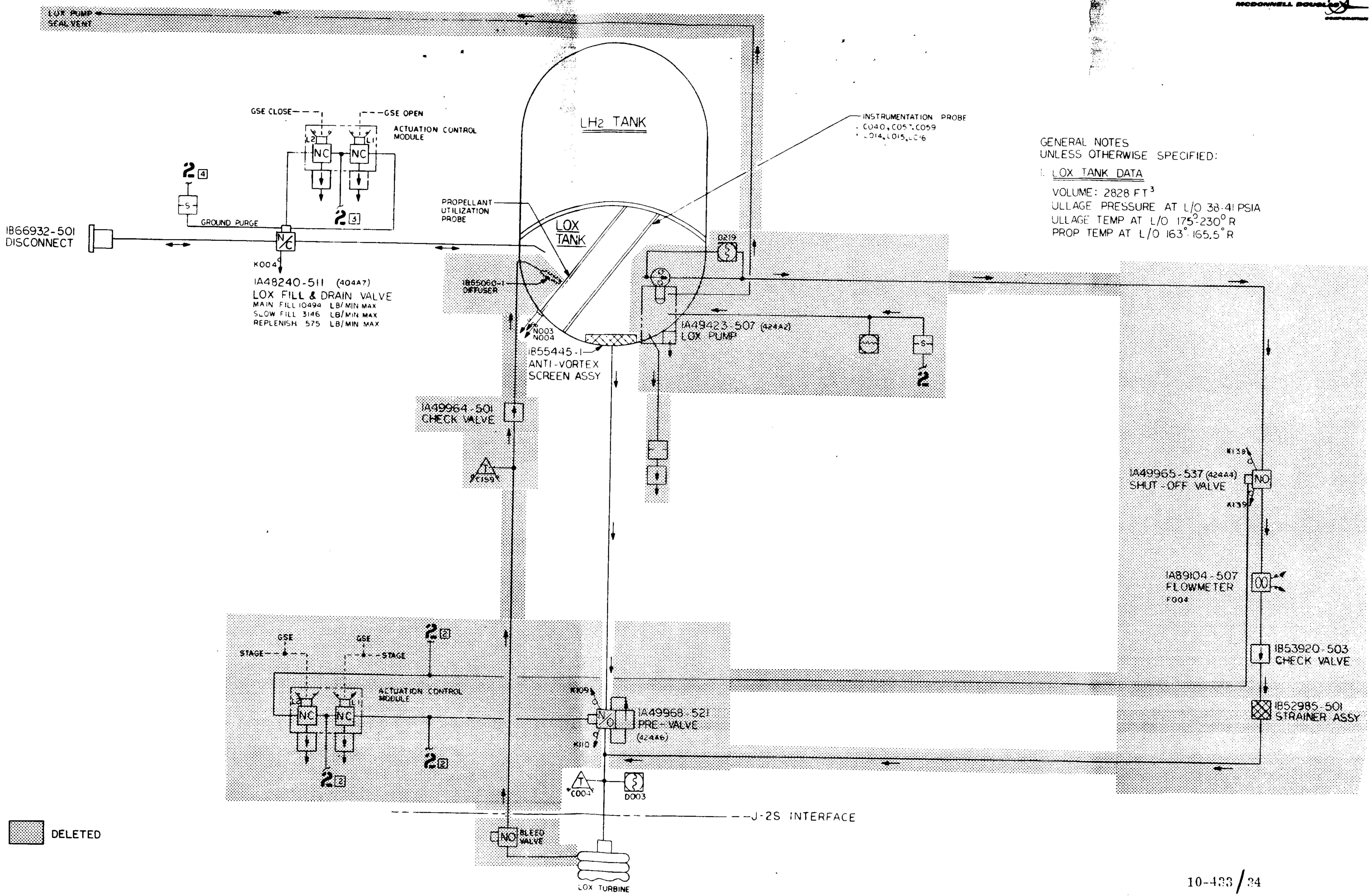

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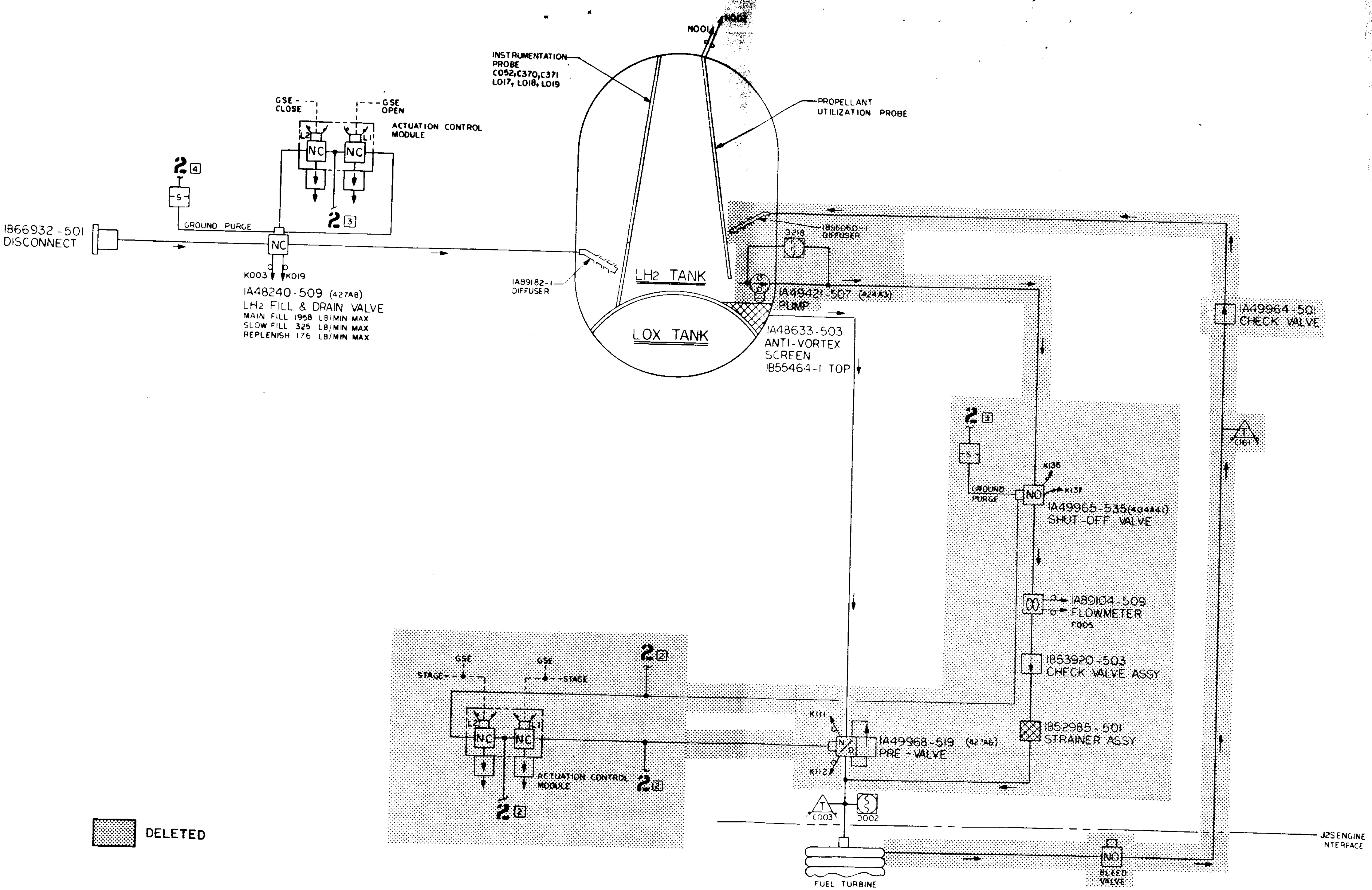
FIGURE 10. 3. 2-3 SCHEMATIC, LOX FEED AND CHILLDOWN (SEQUENCE 6)



GENERAL NOTES
UNLESS OTHERWISE SPECIFIED:

1. LH2 TANK DATA
   - VOLUME 10,409.5 FT^3
   - ULLAGE PRESSURE AT L/O 31-34 PSIA
   - ULLAGE TEMP AT L/O 150°-200° R
   - PROPELLANT TEMP AT L/O 37°-38° R

FIGURE 10.3.2-4 SCHEMATIC, LH_2 FEED AND CHILLDOWN (SEQUENCE 10)

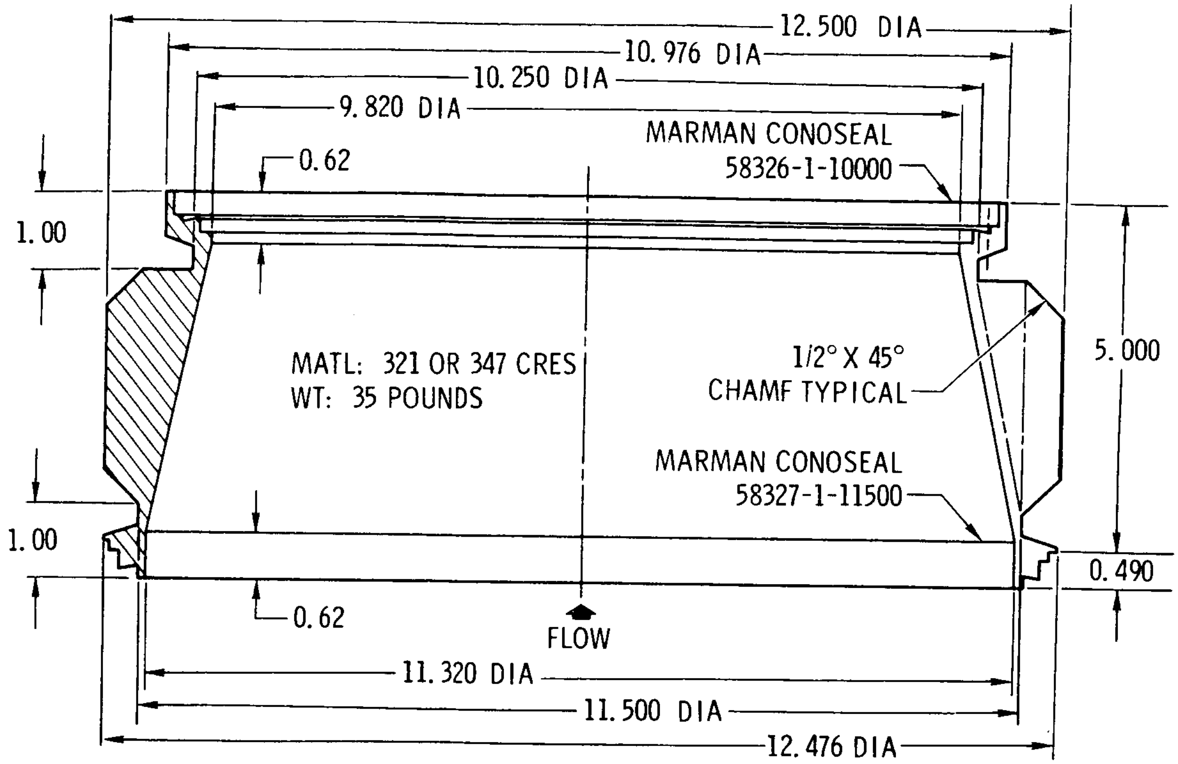


FIGURE 10.3.2-5. LO_2/LH_2 FEEDLINE SPACER

10.3.2.2 (Continued)

Omission of the chilldown system eliminates the need for the chilldown line elbow and the chilldown return line fittings which are welded into the hydrogen and LOX tanks. These fittings will be omitted and the holes will not be drilled in the tanks. Deleted hardware is listed in Table 10.3.2-III.

Deletion of the chilldown requirement eliminates several sequences from the prelaunch and flight events. Five prelaunch and ten flight sequences are no longer required for the LOR mission, Table 10.3.2-IV. Five prelaunch and sixteen flight sequences are eliminated for the Synchronous Mission. Continuous purging of the LOX chilldown pump is eliminated thus reducing total helium consumption by 200 standard cubic inches per minute for the entire prelaunch and flight duration. Total reduction of helium consumption for the LOR mission is 50 cubic feet, and 105 cubic feet for the Synchronous Mission.

The removal of the chilldown systems and prevalves results in moderate changes to 1B59459, Propellant Tanks Subsystem Leak Check Procedure. All changes are deletions in nature and would require minimal time expenditure to accomplish.

TABLE 10.3.2-III. CHILLDOWN SYSTEM HARDWARE DELETIONS-LOR AND SYNCHRONOUS MISSIONS (Ref Dwg 1A59098, 1B74477, 1B68375)

Item	Part Number	Qty	Change
LH_2 Chilldown Shutoff Valve	1A49965-527	1	Delete
Seal (Naflex)	VD261-0037-0504	2	Delete
Duct Assembly (LH_2)	1A49966-505	1	Delete
Resilient Mount (LH_2)	1A49962-527	2	Delete
Resilient Mount (LH_2)	1A49962-503	2	Delete
Prevalve (LH_2)	1A49968-507	1	Delete
Prevalve (LO_2)	1A49968-509	1	Delete
Duct Assembly	1B59760-1	1	Delete
Resilient Mount	1A49962-531	2	Delete
Resilient Mount	1A49962-513	2	Delete
Duct Assembly	1A87741-503	1	Delete
Resilient Mount	1A49157-505	1	Delete
Support	1B38989-1	1	Delete
Insulation	1B55325-1	1	Delete
Insulation	1B55326-1	1	Delete
Diffuser	1B55060-1	1	Delete
Gasket	1B55061	1	Delete
Duct Assembly (LO_2)	1A87736-501	1	Delete
Resilient Mount	1A49962-529	1	Delete
Resilient Mount	1A49962-515	1	Delete
Resilient Mount	1A49962-511	1	Delete
Swing Check Valve	1A49964-501	1	Delete
Misc Hardware			
LO_2 Chilldown Pump	1A49423-507	1	Delete
Seal (Naflex)	VD261-0037-1753	2	Delete
Jumper	MS25083-1EE3	1	Delete
Pipe Assembly	1B59313-1	1	Delete
Adapter	MC237C4W	4	Delete
Seal	MC252C4TA	4	Delete
Pipe Assembly	1B59314-1	1	Delete
LO_2 Chilldown Shutoff Valve	1A49965-533	1	Delete
Coupling - Marman	MVC61772-200A	18	Delete
Gasket - Marman	55666-200-A1	18	Delete
Coupling - Marman	MVC61772-225A	2	Delete
Gasket - Marman	55666-225-A1	2	Delete
Nut	1A94729-1	1	Delete
Strainer	1B52985-501	3	Delete
Nut	1A94729-501	3	Delete
Check Valve	1B53920-503	3	Delete
Duct	1A97740-1	1	Delete
Tube Assembly	1B66501-1	1	Delete
Union	MC160C4	2	Delete
Tube Assembly	1B52585-1	1	Delete

TABLE 10.3.2-III. (Continued)

Item	Part Number	Qty	Change
Tube Assembly	1B64162-1	1	Delete
Union	MC164C4	1	Delete
Nut	S0109C7	1	Delete
LH_2 Chilldown Pump	1A49421-505	1	Delete
Pipe Assembly	1A59315-1	1	Delete
Pipe Assembly	1B59312-1	1	Delete
Flange - Marman	58951-1-025	1	Delete
Hose Assembly	1A48850-507	1	Delete
Flow Meter	1A89104-507	2	Delete

TABLE 10.3.2-IV. SEQUENCE DELETIONS FOR LOR MISSION

Event	Time (Hr:Min:Sec)	Remarks
Chilldown Pumps on for 2 Min	T-00:30:00	
Open Thrust Chamber Chilldown	T-00:10:00	
LH_2 Chilldown Pump ON	T-00:05:00	
LO_2 Chilldown Pump ON	T-00:04:50	
Close LOX and LH_2 Prevalves	T-00:04:45	
Feed Duct Prevalves OPEN	T+00:08:39	
LOX Chilldown Pump OFF	T+00:08:30.4	
LH_2 Chilldown Pump OFF	T+00:08:41	
LOX Chilldown Pump ON	T+02:59:31.5	
LH_2 Chilldown Pump ON	T+02:59:46.5	
Feed Duct Prevalves OPEN	T+03:09:47.9	
LH_2 Chilldown Pump OFF	T+03:09:57.9	
LOX Chilldown Pump OFF	T+03:09:58.1	
LOX Chilldown Pump Purge Control Valve OFF	T+03:15:35.5	

T- = Prior to Launch
T+ = Flight

10.3.2.3 Fuel Tank Pressurization and Repressurization System

As a result of the J-2S increased chamber pressure, a higher pressure is experienced in the hydrogen tap off which is used to pressurize the hydrogen tank.

The present hydrogen tank pressurization system, designed for the J-2 engine is basically adequate for the higher pressure except for two pipe assemblies upstream of the pressurization module which will require increased wall thickness (no change to hose assemblies) and changing of the three orifices in the module. The three orifices will be rough calibrated in the Battleship program in order to eliminate the need for a development test on the module. At the present time the orifices are calibrated on each flight vehicle and it is planned to continue the practice on vehicles using the J-2S engine. Pressurization control module requalification to the higher operating pressure will be necessary.

10.3.2.3 (Continued)

Fuel tank pressurization helium requirements for the LOR Mission are identical to the AS-511 using a J-2 engine, but are different for the synchronous mission, Figure 10.3.2-6. The LOR vehicle requires nine cold helium and seven ambient bottles. Of these, one of the cold helium bottles and, if required, five of the ambient bottles are used for hydrogen tank repressurization. The synchronous mission requires twelve cold bottles and no ambient bottles. With the restartable burner the additional cold bottles will provide sufficient pressurization gas for three burns for both the hydrogen and LOX tanks.

Two of the additional cold helium bottles will be added in the secondary tunnel area, and one will be added in the main tunnel.

This can be accomplished by removing the caps from the tank fittings and installing the 3 bottles. In the secondary tunnel the present 1B67299-1 manifold will be retained. The MC plug and seal will be removed from the end of the manifold and a 1B67094-1 manifold and 1B67299-501 manifold will be installed to connect the two additional bottles.

The MC plug and seal removed from the end of 1B67299-1 manifold will be used to plug the 1B67299-501 manifold. Table 10.3.2-V lists hardware changes for the synchronous mission.

To add a bottle in the main tunnel area will require redevelopment of two sections of the hydrogen pressurization line and relocation of the supporting structure. The 1B67299-505 manifold will be removed and replaced with 1B67299-503 which is a three bottle manifold.

10.3.2.4 LOX Tank Pressurization and Repressurization System

The LOX Tank Pressurization System, Figure 10.3.2-7, will not change because of the use of the J-2S engine in place of the J-2, and will be identical to AS-511 for the LOR mission. Due to the requirement of two repressurizations during the synchronous mission, it will be necessary to add one cold helium bottle and delete the two ambient bottles used for LOX tank repressurization. The total cold helium bottle requirements are described in paragraph 10.3.2.3 for both LOX and LH_2 tanks. Table 10.3.2-VI lists hardware changes for the synchronous mission.

10.3.2.5 Pneumatic Control System

The pneumatic control system, Figure 10.3.2-8 is greatly simplified by the J-2S engine and the ensuing deletion of LH_2 chilldown, LO_2 chilldown, start tank, and the engine pump purge components.

Removal of the LH_2 and LO_2 chilldown systems allows deletion of the actuation control module and associated hardware and all purge lines to the LH_2 chilldown shutoff valve and LOX chilldown pump. Removal of the start tank from the J-2S eliminates the need for the actuation control module (normally closed side of pneumatic control module) that actuated the start tank vent and relief valve. The J-2S design requires that the engine pumps be purged only prior to liftoff. This change makes possible the deletion of the engine pump purge control module and

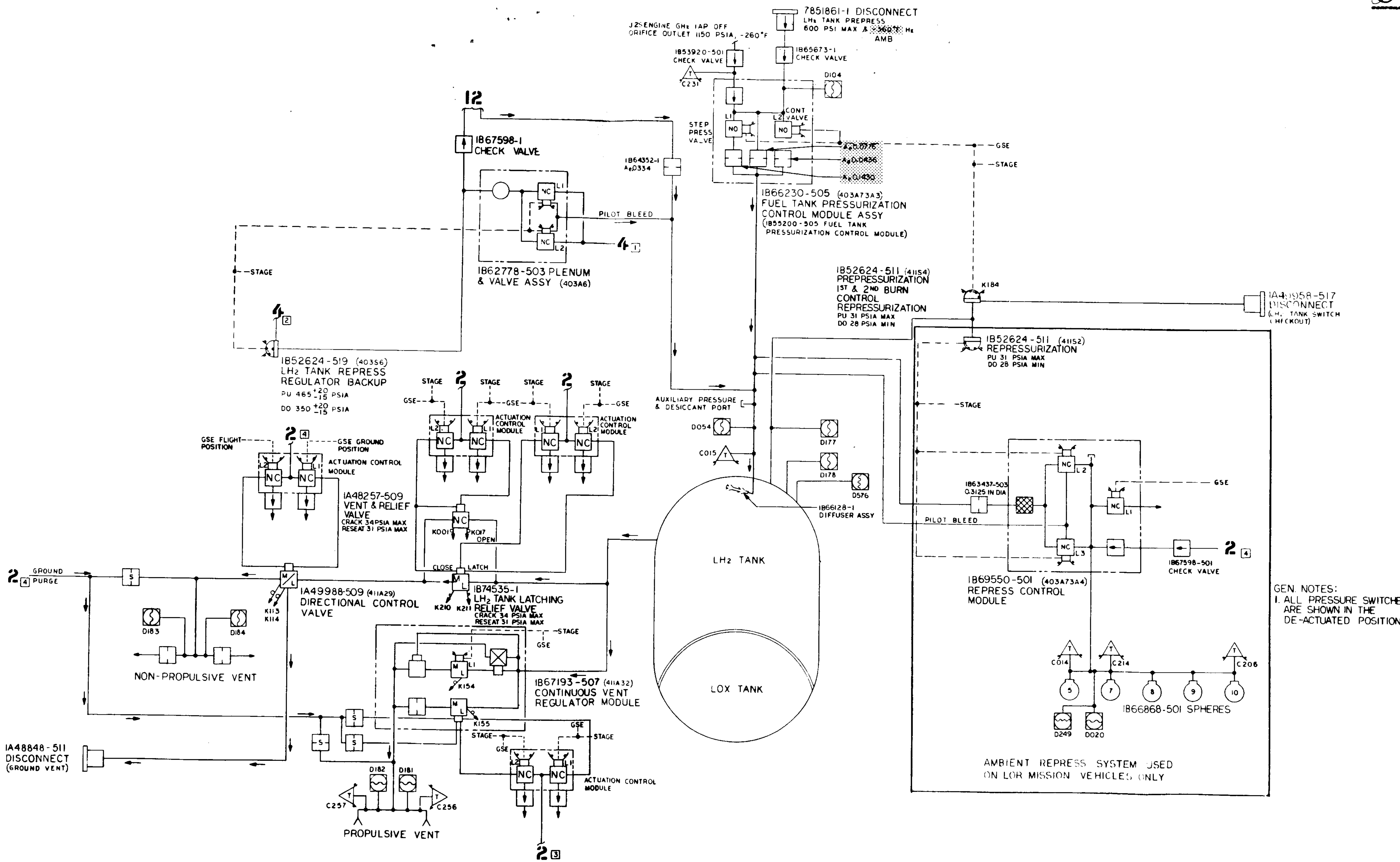


FIGURE 10.3.2-6 FUEL TANK PRESSURIZATION SYSTEM (SEQUENCE 8)

TABLE 10.3.2-V. FUEL TANK PRESSURIZATION SYSTEM HARDWARE CHANGES FOR SYNCHRONOUS MISSION ONLY (Ref Dwg 1B58009 and 1B58008)

Item	Part Number	Qty	Change
Ambient Helium Bottle	1B39870	5	Delete
Cold Helium Bottle	1A48858-1	3	Add
Pipe Assembly	1B64380-1	1	Redesign
Pipe Assembly	1B64384-1	1	Redesign
Associated Hardware			Added
Manifold	1B67094-1	1	Add
Adapter	MC237C8W	3	Add
Seal	MC252C8TA	3	Add
Plug	MC238C8W	1	Delete
Seal	MC252C8TA	1	Delete
Isolator	5045-002 Robinson	2	Add
Screw	MS35215-55	12	Add
Washer	AN960D10L	12	Add
Nut	NAS679C3	12	Add
Manifold	1B67094-503	1	Add
Manifold	1B67094-505	1	Delete
Pipe Assembly	1A98355-1	1	Redevelop
Pipe Assembly	1B43397-1	1	Redevelop
Module Assembly	1B69550-1		Delete
Cap Assembly	MC177C12W		Delete
Bolt	NAS1004-4H	4	Delete
Washer	AN960D416L	4	Delete
Pipe Assembly	1B58859-1		Delete
Pipe Assembly	1B68898-1		Delete
Clamp	NAS1715C4K		Delete
Clamp	NAS1715C8K		Delete
Bolt	NAS1003-1A		Delete
Washer	AN960D10L		Delete
Nut	NAS679C3		Delete
Check Valve	1B51361-1		Delete
Pipe Assembly	1B66254-1		Delete
Clamp	NAS1715C4K	2	Delete
Bolt	NAS1003-1A		Delete
Bolt	NAS1003-1A		Delete
Washer	AN960D10L	2	Delete
Nut	NAS679C3	2	Delete
Pipe Assembly	1B66254-1		Redesign
Pipe Assembly	1B66816-1		Delete
Union	MC164C12W		Delete
Nut	MC174C12W		Delete
Washer	AN906D1716		Delete
Pipe Assembly	1B66815-1		Delete
Clamp	TA11C55D24	5	Delete
Collar	1B64177-507	5	Delete

TABLE 10.3.2-V (Continued)

Item	Part Number	Qty	Change
Collar	1B64178-507	5	Delete
Bolt	NAS1003-1A	10	Delete
Washer	AN960C10L	10	Delete
Nut	NAS679C3	10	Delete
Nut	MC174C12W		Delete
Washer	AN960D1716		Delete
Pipe Assembly	1B66824-1		Redevelop Pipe Assy by Removing 1B63908 Tee and 1B44164-523 Tee.
Pipe Assembly	1B66814-1		Delete
Pipe Assembly	1B66803-1		Delete
Clamp	NAS1715C22K	2	Delete
Clamp	NAS1715G6K	2	Delete
Bolt	NAS1003-1A	2	Delete
Washer	AN960D10L	2	Delete
Nut	NAS679C3	2	Delete
Pipe Assembly	1B64614-1 (Ref Transducer 020)		Delete
Cap Assembly	MC177C4W		Delete
Clamp	TA11C55D24	3	Delete
Collar	1B64177-507	3	Delete
Collar	1B64178-507	3	Delete
Bolt	NAS1003-1A	6	Delete
Washer	AN960D10L	6	Delete
Nut	NAS679C3	6	Delete
Clamp	NAS1716D12K		Delete
Bolt	NSA1003-1A		Delete
Washer	AN960D10L		Delete
Nut	NAS679C3		Delete
Union	MC160C12W		Delete
Pipe Assembly	1B66808-1		Delete
Clamp	NAS1716D12K	12	Delete
Bolt	NAS1003-1A	24	Delete
Washer	AN960D10L	24	Delete
Nut	NAS679C3	24	Delete
Union	MC160C12W		Delete
Pipe Assembly	1B67697-1		Delete
Tee	MC122D12		Delete
Nut	MC124C12W		Delete
Reducer	MC247C12-4		Delete
Pipe Assembly	1B67699-1		Delete
Pipe Assembly	1B67696-1 (Ref Transducer D249)		Delete
Vent Seal	1B58239-1	4	Delete

TABLE 10.3.2-V (Continued)

Item	Part Number	Qty	Change
Bolt	NAS1004-6H	12	Delete
Washer	AN960C416L	12	Delete
Gasket - Marman	54973-12A1	4	Delete
O-Ring	S0046T26	4	Delete
Vent Seal	1B58239-1	6	Delete
Bolt	NAS1004-6H	9	Delete
Washer	AN960C416L	9	Delete
Gasket - Marman	54973-12A1	3	Delete
O-Ring	S0046T26	3	Delete

10.3.2.5 (Continued)

eliminates the requirement to thermally condition the pneumatic bottle prior to liftoff. In flight, automatic sequencing of the module is no longer required. Tables 10.3.2-VII thru 10.3.2-XI list sequence and hardware changes to the pneumatic control system.

The removal of actuation control modules, purge control modules, and various other components in this system result in a major change to 1B59457, Pneumatic Control Subsystem Leak Check. The majority of changes are deletions in nature; therefore, the time required to change the procedure would be held to a minimum.

Major portions of drawing 1B66572, Propulsion System Test - SV, VCL, pertaining to valve timing and other facets of the pneumatic control system would have to be rewritten and reprogrammed.

10.3.2.6 Ullage Control Solid Motor System

On stages using the J-2 engine, the ullage rockets are fired at the time of separation just prior to first burn. This imparts adequate stage acceleration to insure settled propellants during the separation transient and main engine start. Settled propellants are required to prevent chilldown pump cavitation and to insure liquid propellants at the engine/stage interface. The required acceleration is provided on the Saturn V vehicle by two solid propellant rockets mounted on the aft skirt 180° apart and angled with the thrust vector passing through the vehicle center of gravity. The spent ullage motors and their fairings are jettisoned after they have burned out (Figure 10.3.2-9).

With the idle mode capability of the J-2S engine, the ullage rockets are no longer required. The J-2S engine is capable of idle mode operation without the subcooled propellants which the standard J-2 engine requires. First start can be achieved on gaseous, mixed phase, liquid or any combination of these oxidizer and fuel conditions existing in the feed ducts. This is provided that run NPSH is available in propellants at the main tank outlets at engine start command.

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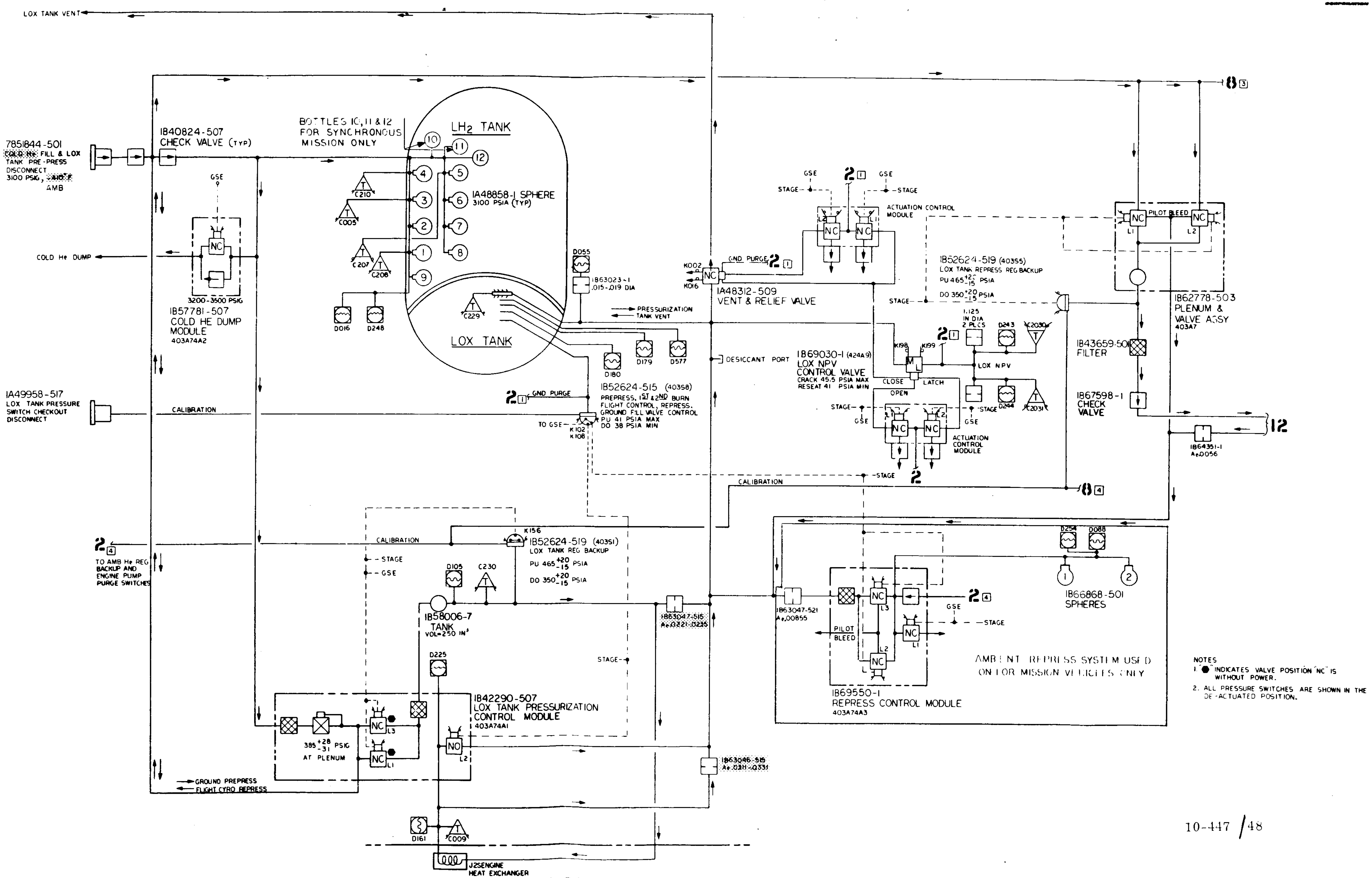

FIGURE 10.3.2-7 LOX TANK PRESSURIZATION SYSTEM (SEQUENCE 4)

TABLE 10.3.2-VI. LOX TANK PRESSURIZATION SYSTEM HARDWARE CHANGES FOR SYNCHRONOUS MISSION ONLY (Ref Dwg 1B58009 and 1B58008)

Item	Part Number	Qty	Change
Module Assembly	1B69550-1	1	Delete
Cap Assembly	MC177C12W		Delete
Bolt	NAS1004-4H	4	Delete
Washer	AN960D416L	4	Delete
Pipe Assembly	1B66842-1	1	Delete
Orifice Assembly	1B58007-535	1	Delete
Pipe Assembly	1B67108-1	1	Delete
Clamp	NAS1715C4K	3	Delete
Clamp	NAS1715C8K		Delete
Bolt	NAS1003-1A		Delete
Washer	AN960D10L		Delete
Nut	NAS679C3		Delete
Support	1B66863-1	1	Delete
Collar	1B64178-507	1	Delete
Collar	1B64177-507	1	Delete
Bolt	NAS1004-34A	2	Delete
Washer	AN960D416L	2	Delete
Nut	NAS679C4	2	Delete
Clamp	NAS1715C4K	2	Delete
Elbow	MC165C4 (Add in Place of MC162C4 Tee)		
Amb Helium Bottle	1B39870	2	Delete
Tee	MC162C4		Delete
Pipe Assembly	1B58853-1	1	Delete
Union	MC160C12W		Delete
Pipe Assembly	1B67386-1	1	Delete
Clamp	NAS1716D12K		Delete
Bolt	NAD1003-2A	2	Delete
Washer	AN960D10L	2	Delete
Clamp	NAS1716D12K	2	Delete
Bolt	NAS1003-1A	4	Delete
Washer	AN960D10L	4	Delete
Nut	NAS679C3	2	Delete
Pipe Assembly	1B66251-1	1	Delete
Pipe Assembly	1B58801-1 (Ref Transducer D088, & D254)	1	Delete
Pipe Assembly	1B69840-1	1	Redevelop Pipe Assembly by Removing 1B63809 Tee

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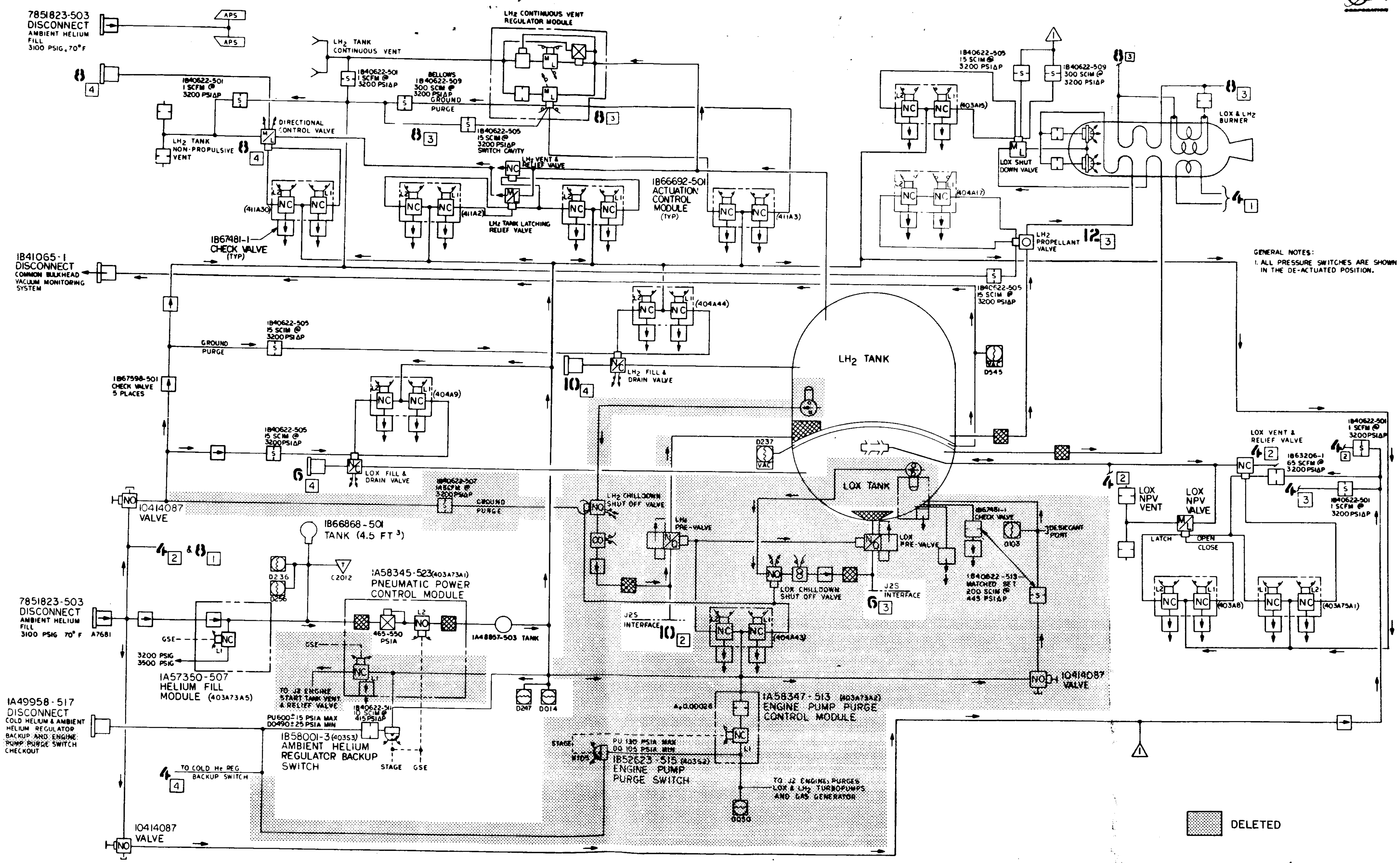

FIGURE 10.3.2-8 SCHEMATIC, PNEUMATIC CONTROL

TABLE 10.3.2-VII. ENGINE PUMP PURGE SEQUENCE CHANGES

Event	Time (Sec)	Remarks
Engine Pump Purge Control Valve Enable on	T + 639.5	Delete on LOR and SYNC Missions
Engine Pump Purge Control Valve Engine on (Back-up Signal)	T + 648	Delete on LOR and SYNC Missions

T = Time of J-2S Ignition

10.3.2.6 (Continued)

Deletion of the ullage motor system represents a significant simplification of the stage hardware and operational sequence. These deletions are common to the LOR and synchronous missions and are a direct result of the increased capabilities of the J-2S engine.

Deletion of telemetry talkback components include a pressure transducer, temperature transducer, temperature bridges, and electrical harnesses.

Ullage control and solid motor deletions for both the LOR and Synchronous missions are listed on Table 10.3.2-XII.

Ullage motors, support structure, and aft skirt penetrations will be deleted. See Figure 10.3.1-3.

Elimination of the solid propellant ullage motor system also deletes the following operational sequences.

- T* - 20 sec: Arm ullage rocket ignition system, charge firing units.
- T - 0.1 sec: Fire exploding bridgewire and Pyrogen igniters.
- T - 0.1 to T+ 4 sec: Ullage motor burn.
- T+ 12 sec: Fire exploding bridge wire and ignite confined detonation fuse, thereby exploding the frangible nuts and allowing jettison of the ullage rocket assembly, fairing, and mounting.

* Time of Separation

Checkout of initiators and circuitry as well as motor installation are significant operations that can now be omitted.

TABLE 10.3.2-VIII. LH_2 CHILLDOWN SYSTEM PNEUMATIC CHANGES LOR AND SYNCHRONOUS MISSIONS (Ref Dwg 1B58004)

Item	Pipe Length	Part Number	Qty	Change
Pipe Assembly	80	1B59292-1	1	Delete
Union		MC160C4W	11	Delete
Pipe Assembly	90	1B59294-1	1	Delete
Pipe Assembly	120	1B59291-1	1	Delete
Pipe Assembly	120	1B59298-1	1	Delete
Pipe Assembly	40	1B52502-1	1	Delete
TEE		MC162C4	2	Delete
Pipe Assembly	72	1B52610-1	1	Delete
Module		1B66692-501	1	Delete
Check Valve		1B67481-1	2	Delete
Strap		1B66290-1	2	Delete
Seal		MC252C4TA	4	Delete
Adapter		MC237C4W	2	Delete
Pipe Assembly	120	1B74809-1	1	Delete
TEE		1B74811-1	1	Delete
Pipe Assembly	120	1B52464-1	1	Delete
Pipe Assembly	70	1B59297-1	1	Delete
Pipe Assembly	120	1B59293-1	1	Delete
Pipe Assembly	50	1B59287-1	1	Delete
Reducer		MC235C7W	1	Delete
Pipe Assembly	120	1B64600-501	1	Delete
Bracket		1B64133-1	2	Delete
Pipe Assembly	72	1B64829-1	1	Delete
Pipe Assembly	60	1B64828-1	1	Delete
Pipe Assembly	96	1B64827-1	1	Delete
Pipe Assembly	60	1B64826-1	1	Delete
Restrictor		1B40622-501	1	Delete
Pipe Assembly	96	1B64825-1	1	Delete
Pipe Assembly	42	1B64824-1	1	Delete
Pipe Assembly (1B58004 zone 41)		1B64144-1 R1	1	Redevelop
Pipe Assembly (1B58004 zone 42)		1B52543-1	1	Redevelop
MISC Hardware				

TABLE 10.3.2-IX. LO_2 CHILLDOWN SYSTEM PNEUMATIC CHANGES LOR AND SYNCHRONOUS MISSIONS (Ref Dwg 1B58004)

Item	Pipe Length	Part Number	Qty	Change
Pipe Assembly	130*	1B52455-1	1	Delete
Pipe Assembly	42	1B52494-1	1	Delete
Pipe Assembly	34	1B74204-1	1	Delete
Pipe Assembly	27	1B74205-1	1	Delete
Pipe Assembly	86	1B69966-1	1	Delete
Pipe Assembly	43	1B74208-1	1	Delete
Pipe Assembly	89	1B66503-1	1	Delete
Pipe Assembly	17	1B75036-1	1	Delete
Pipe Assembly	11	1B75037-1	1	Delete
Pipe Assembly	140*	1B64141-1	1	Delete
Pipe Assembly	94	1B59285-1	1	Delete
Pipe Assembly	75**	1B67712-1	1	Delete
Pipe Assembly	25	1B75035-1	1	Delete
Pipe Assembly	4	1B75034-1	1	Delete
Pipe Assembly	14	1B75033-1	1	Delete
Pipe Assembly	3	1B75189-1	1	Delete
Pipe Assembly	12	1B75032-1	1	Delete
Block		1B43242-1	1	Delete
Union		MC160C4W	7	Delete
Clamp		NAS1715C4K	32	Delete
Adapter		MC287C4W	2	Delete
Seal		MC252C4TA	3	Delete
Orifice		1B40622-5	1	Delete
Check Valve		1B67481-1	1	Delete
Cross		MC163C4	1	Delete
CAP		MC177C4W	1	Delete
Orifice		1B40622-3	1	Delete
Pipe Assembly		New		Add
Hose Assembly		1A48850-507	2	Delete
Flange		Marman 58951-1025	1	Delete

* Pipe assembly contains hose assembly 1A48850-507.
** Pipe assembly contains 58951-1-025 flange.

TABLE 10.3.2-X. PNEUMATIC POWER CONTROL SYSTEM CHANGES
LOR AND SYNCHRONOUS MISSION
(Ref Dwg 1B58004)

Item	Pipe Length	Part Number	Qty	Change
Control Module		1A58345-523	1	Modify*
Pipe Assembly	85	1B67231-1	1	Delete
Flange		Marman 58951-1-025	1	Delete
Misc Hardware				Delete
Pipe Assembly			1	New

NOTE: Pipe is 1/4 diameter x .028 Cres with standard MC fitting on one end and Marman flange 58951-1-025 on the other.

* Module to be modified by removing solenoid from normally closed side and capping all open ports.

TABLE 10.3.2-XI. ENGINE PUMP PURGE SYSTEM CHANGE
LOR AND SYNCHRONOUS MISSIONS
(Ref Dwg 1B58004)

Item	Pipe Length	Part Number	Qty	Change
Module		1A58347-513	1	Delete
Pipe Assembly		1B67224-1	1	Redesign
Pipe Assembly	92*	1B67233-1	1	Delete
Clamp		NAS1715C416	3	Delete
Union		MC164C4	1	Delete
Nut		MC174C4W	1	Delete
Pipe Assembly	20**	1B52516-1	1	Delete
Vent Seal		1B58239-1	1	Delete
Misc. Hardware				
Tee		1A95010-1	1	Delete
Adapter		1A98722-1	1	Delete
Flange		MARMAN	1	Delete
		59097-1-12	1	Delete
Flange		1A89733-1	1	Delete
Pipe Assembly	52	1B52538-1	1	Delete
Adapter		MC237C4W		Delete
Seal		MC252C4TA	1	Delete
Pipe Assembly	15	1B63285-1	1	Delete

* Pipe assembly contains 1A95010-1 tee, 1A98722-1 adapter, 59097-1-12 flange, and one MC fitting.

** Pipe assembly contains 1A89733-1 flange and one MC fitting.

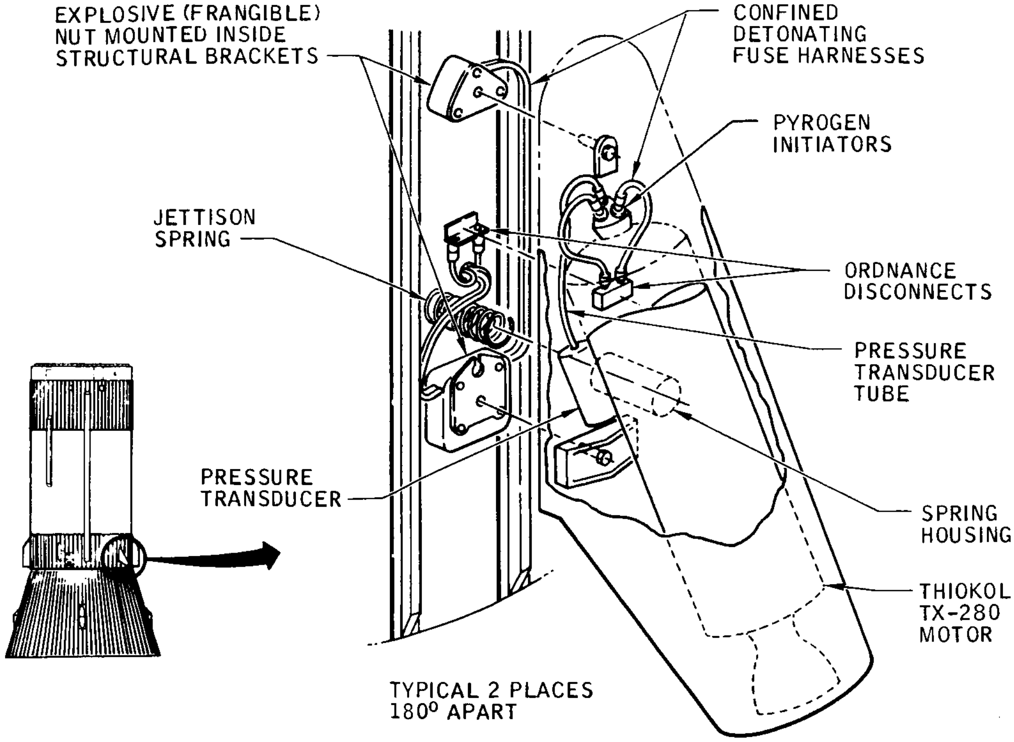

FIGURE 10.3.2-9 ORDNANCE SYSTEM

10.3.2.7 Auxiliary Propulsion System

The Auxiliary Propulsion System (APS) as it is designed for the Saturn V vehicle includes one Gemini 72 pound thrust ullage engine in each of the two modules. The J-2S engine with its idle mode capability will not require APS ullaging for either the LOR or synchronous mission. Slosh control and propellant settling will be accomplished with idle mode.

The Saturn V modules will be flown on both the LOR and synchronous missions. Fuel consumption for the synchronous mission, as shown in Table 10.3.2-XIII, is greater than for the LOR mission and is well within the APS capability.

Redesign of the Saturn V module requires the removal of the 72 pound thrust ullage engine, Figure 10.3.2-10 and engine support frame. The outer fairing will be fabricated without the ullage engine opening and the propellant manifolds will be redesigned to eliminate the ullage engine ports. Deleted electrical hardware includes the pressure transducer, associated connectors and wiring.

Table 10.3.2-XIV lists the components deleted from the APS module. A cutaway of the module is shown in Figure 10.3.2-11. The figure depicts the present configuration and the major components to be deleted when employed with the J-2S engine.

The deletion of the APS ullage engines would result in moderate changes to drawing 1B66902, Auxiliary Propulsion System Acceptance Test, S-IVB/SV, and drawing 1B69403, APS Subsystem Checkout, SV.

TABLE 10.3.2-XII. ULLAGE CONTROL SOLID MOTOR HARDWARE DELETIONS LOR AND SYNCHRONOUS MISSIONS (Ref Dwg 1A82765-501)

Item	Part Number	Qty	Change
Rocket Motor	1A81960-1	2	Delete
Cover	1A02612-501	2	Delete
Fairing Assembly	1A78552-1	2	Delete
Cable Assembly	1A02590-501	2	Delete
Ring Assembly	1B43420-1	2	Delete
Fairing Assembly	1A74671-1	2	Delete
Frame	1A82592-1	2	Delete
Bracket	1A02598-1	2	Delete
Bracket	1A02602-1	2	Delete
Stiffiner	1A0260-1	2	Delete
Bracket	1A02663-1	2	Delete
Column	1A02665-501	2	Delete
Column	1A02665-502	2	Delete
Beam	1A02666-501	2	Delete
Beam	1A02666-502	2	Delete
Frame, Thrust	1A02667-1	2	Delete
Ring	1A02669-1	2	Delete
Misc Hardware			Delete

NOTE: Hardware deletions listed above are sub parts to top drawing 1A82765-501 which will be deleted in its entirity. See paragraph 10.3.1.1 for structural hardware changes and paragraph 10.3.2.10 for ordnance changes.

10.3.2.7 (Continued)

Deletions of portions of drawing 1B69571, Auxiliary Propulsion System Module - SV, VCL pertaining to the ullage engines, would be moderate and would result in rewriting and reprogramming.

Due to extended periods of exposure to the sun and deep space during the synchronous mission, the temperature in the propellant tanks and the self sealing disconnects will go beyond acceptable limits as noted below.

	Allowable Temp (°F)		Actual (°F)	
Area	High	Low	High	Low
Propellant Tanks	105	40	102	35*
Self-Sealing Disconnects	160	-20	270*	-25*

* Beyond temperature limits.

TABLE 10.3.2-XIII. SYNCHRONOUS ORBIT APS PROPELLANT MANAGEMENT

	Module I		Module III	
	Fuel (Lbs)	Oxidizer (Lbs)	Fuel (Lbs)	Oxidizer (Lbs)
Module Capacity	125.6	204.0	125.6	204.0
Propellant Consumption Attitude Control	52.3	83.7	55.0	88.0
Propellant Consumption Ullaging	0.0	0.0	0.0	0.0
Residual (10%)	12.6	20.4	12.6	20.4
Propellant Required	64.9	104.1	67.6	108.4
Excess or Off-Loaded	60.7	99.9	58.0	95.6

TABLE 10.3.2-XIV. AUXILIARY PROPULSION SYSTEM HARDWARE CHANGES LOR AND SYNCHRONOUS MISSION

(Ref Dwg 1A65685 and 1A82258)

Item	Part Number	Qty	Change
Engine No. 4 - Ullage	Rocketdyne 21005	2	Delete
Retainer	1B63088-1	2	Delete
Cover	1B52658-1	2	Delete
Tube Assembly	1B57441-1	2	Delete
Expander	MC239C42	2	Delete
Bracket	1B57259-1	2	Delete
Jumper	MS25083-1BB7	2	Delete
Pipe Assembly	1B59580-1	2	Delete
Pipe Assembly	1B59670-1	2	Redevelope
Pipe Assembly	1B59679-1	2	Redevelope
Pipe Assembly	1B59572-1	2	Redevelope
Frame	1B51331-1	2	Delete
Frame	1B33333-269	2	Redesign*

*Redesign to omit opening for ullage motor.

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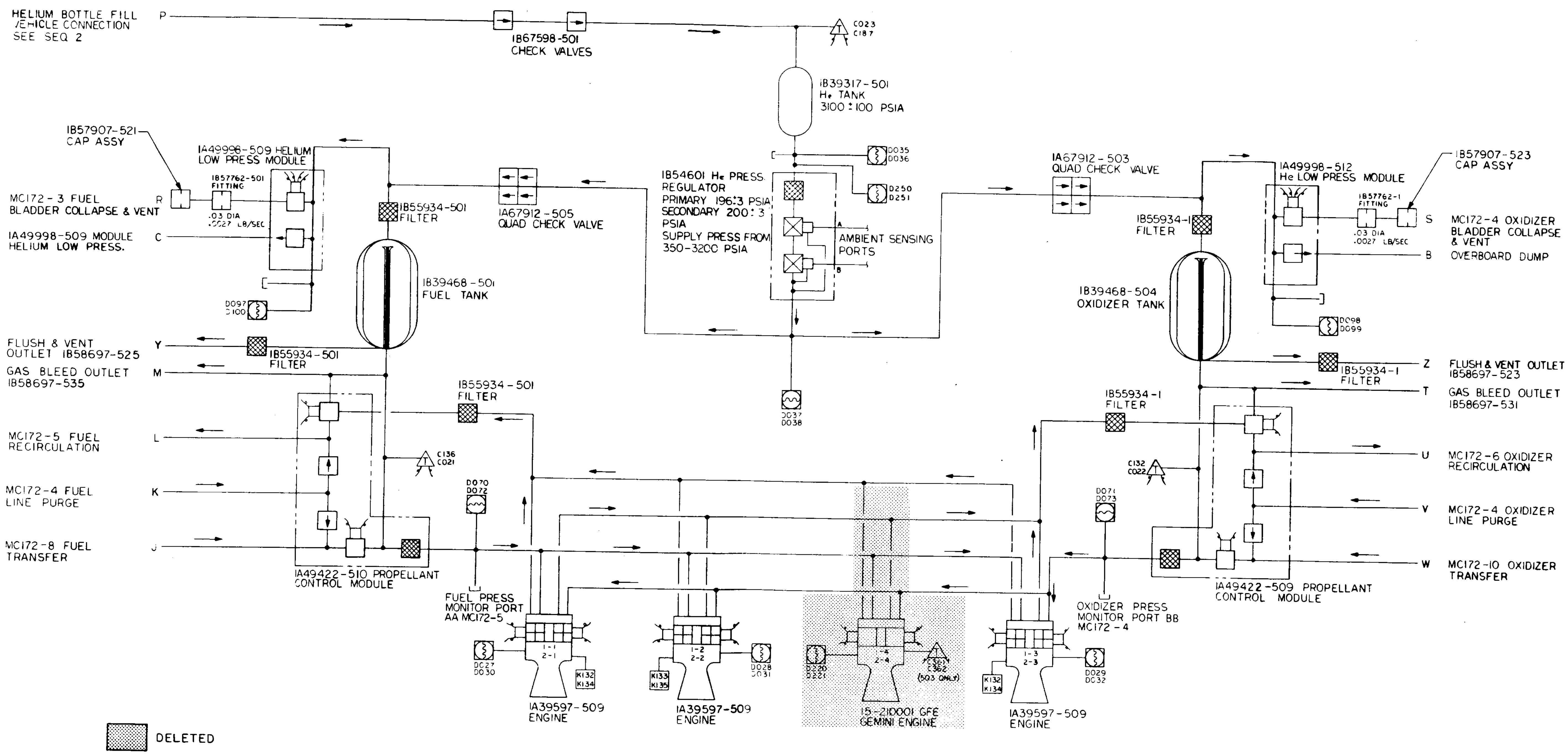


FIGURE 10.3.2-10 SCHEMATIC - APS (SEQUENCE 16)

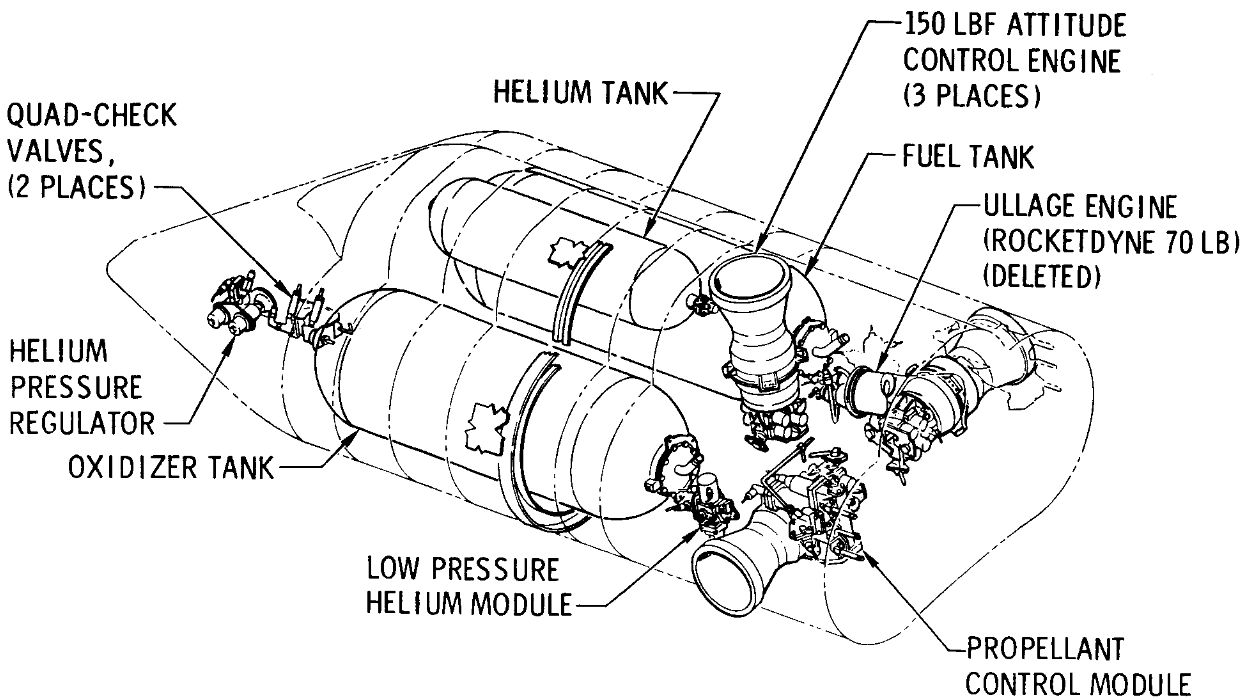


FIGURE 10.3.2-11 APS COMPONENT INSTALLATION

10.3.2.7 (Continued)

The minimum temperature attained by the propellant tank is 35°F which is 5°F below the minimum allowable. This temperature occurs in the vicinity of the forward tank support area. This problem is eliminated by fabricating the entire forward tank support of fiberglass. There will be no appreciable weight change.

The maximum temperature attained by the self-sealing disconnects is +270°F which is 110°F above the maximum allowable. Recent redesigns dictated by AS-501 flight results incorporate a shield over the exposed area. These shields have been designed and fabricated for future vehicles and are expected to eliminate the disconnect high temperature problems.

10.3.2.8 Flight Control System

The hydraulic system design, performance, operation and handling have been reviewed for compatibility with the J-2S engine and with the missions specified in the study. The hydraulic system performance capability was studied because of the increases in gimbal loads and changes in the engine start conditions. The significant increases in gimbal loads were caused by increased thrust. System operation was found to be satisfactory with a somewhat reduced margin.

The most significant change in operation of the hydraulic system is the requirement that the system provide stage control (pitch and yaw) during idle mode before restart. The idle mode duration may be as long as 100 seconds with the engine thrust varying from 1,350 lbs at start to 5,000 pounds maximum. The engine driven hydraulic pump will not be operating because the LOX turbine which drives the pump is not powered during this period. The auxiliary hydraulic pump and the accumulator must therefore supply all the flow required. The capability of the system was found to be satisfactory.

Engine Driven Pump and Isolator Assembly Modification

The most significant hydraulic system hardware change recommended as a result of this study is the modification and partial requalification of the engine driven hydraulic pump and isolator assembly. The changes are required because the speed of the LOX turbine, which directly drives the pump, has increased significantly and the clearance between the pump and the engine nozzle is reduced. A comparison of the J-2 and J-2S pump speed requirements is shown below:

RPM	J-2	J-2S
Maximum Speed	8,700	10,870
Nominal Speed	8,000	10,500
Overspeed (Pump Specification)	10,000	12,000

Analyses

The two basic problems to be resolved were the increased speed of the pump and the reduction in clearance between the pump and engine nozzle.

Possible solutions to the increased speed problem included (1) modification and requalification of the present pump, or (2) modification of another "off the shelf" pump which is designed for higher speed to suit this application but requires a complete requalification program.

The increased speed problem was reviewed with the pump manufacturer. It was mutually agreed that the present pump can be modified and requalified to meet the new requirements. The only hardware modification recommended by the manufacturer is to change the thickness of a spacer in the displacement control. This modification will reduce the stroke from 25 to 19 degrees and limit maximum flow to the same level as the present pump.

10.3.2.8 (Continued)

The endurance test on the present pump is 100 hours duration. It is recommended that this be reduced to 60 hours to compensate for operation at the higher speeds. This is considered conservative since the pump is operated during normal testing for less than 10 hours and during engine burn for only a few minutes.

Based upon these considerations a decision can be made solely on the basis of the analysis related to the pump-to-nozzle clearance problem.

The dimensions of the present hydraulic pump, Part No. 1A66240, which is part of the pump isolator assembly, Part No. 1A86847, have been reviewed and are not considered compatible with the envelope provided in Figure 32 of J-2S Interface Criteria Document, R-7211, Revision dated 7 May 1968. Further investigation into this area was considered necessary to (1) determine if additional clearance is available to allow installation of the pump isolator assembly, and (2) determine if sufficient clearance was allowed to assure that the pump clears the nozzle during engine operation considering thermal expansion, vibration, and dimensional tolerances.

The latest information from Rocketdyne indicates that under worst tolerance conditions the nozzle just contacts the present pump envelope, see Figure 10.3.2-12. Since the nozzle nominal dimension has increased 0.5 inch in that area, the installed pump envelope must be reduced by 0.5 inch to restore the original J-2 nozzle clearance condition. This of course assumes that tolerance buildups are the same for J-2 and J-2S. The installed pump length must be reduced approximately one inch to increase the nozzle clearance by 0.5 inch.

Several possible methods of obtaining additional clearance were considered:

a. Modify the thermal isolator and the seal and bearing group to reduce the overall length and use the present spline shaft. This change was found to be feasible only if the required reduction in the installed length of the pump isolator assembly did not exceed 3/8 inch.

b. Modify the thermal isolator to extend the present seal and bearing group up into the LOX turbine dome. This would require that Rocketdyne provide a shorter spline shaft. This change was found to be feasible to reduce the installed length of the pump isolator assembly by approximately one inch.

c. Adapt the S-II pump, which is a smaller unit, to the J-2S and the S-IVB hydraulic system. This is not considered feasible because substantial redesign is required.

d. Adapt another smaller pump to the J-2S and the S-IVB hydraulic system. This solution is considered most feasible if a very large additional clearance is required.

These methods were based on the analyses discussed in subsequent paragraphs.

10.3.2.8 (Continued)

The present pump and thermal isolator assembly installed length can be reduced approximately 3/8 inch and still use the present spline shaft. This would be accomplished by reducing several clearances in the seal and bearing group and by reducing the isolator length by an equivalent amount. The margin in the thermal isolation capability of the present design is such that additional requalification of this feature would not be required. The clearance obtained with this change was found to be insufficient.

The present pump and redesigned thermal isolator assembly installed length can be reduced approximately one inch if a new shorter quill shaft is used. This would be accomplished by redesign of the isolator assembly to make the present seal and bearing group extend up into the LOX turbine dome. Figures 10.3.2-12 and 10.3.2-13 show a comparison of the present and proposed design. No redesign of the seal and bearing group would be required. The thermal isolator would be changed to move the flange location and to reduce the diameter of the portion of the isolator that extends up into the dome to allow insertion at a more favorable angle. Sufficient clearance must remain between the two flanges to allow the isolator to be bolted to the pump and to the LOX dome. The margin in the thermal isolation capability of the present design is such that MDAC is confident that this change can be made but requalification of this feature is considered necessary.

The flexible lines terminating at the pump will not require change since the location of the pump will only move about one inch in the direction reducing the point to point length.

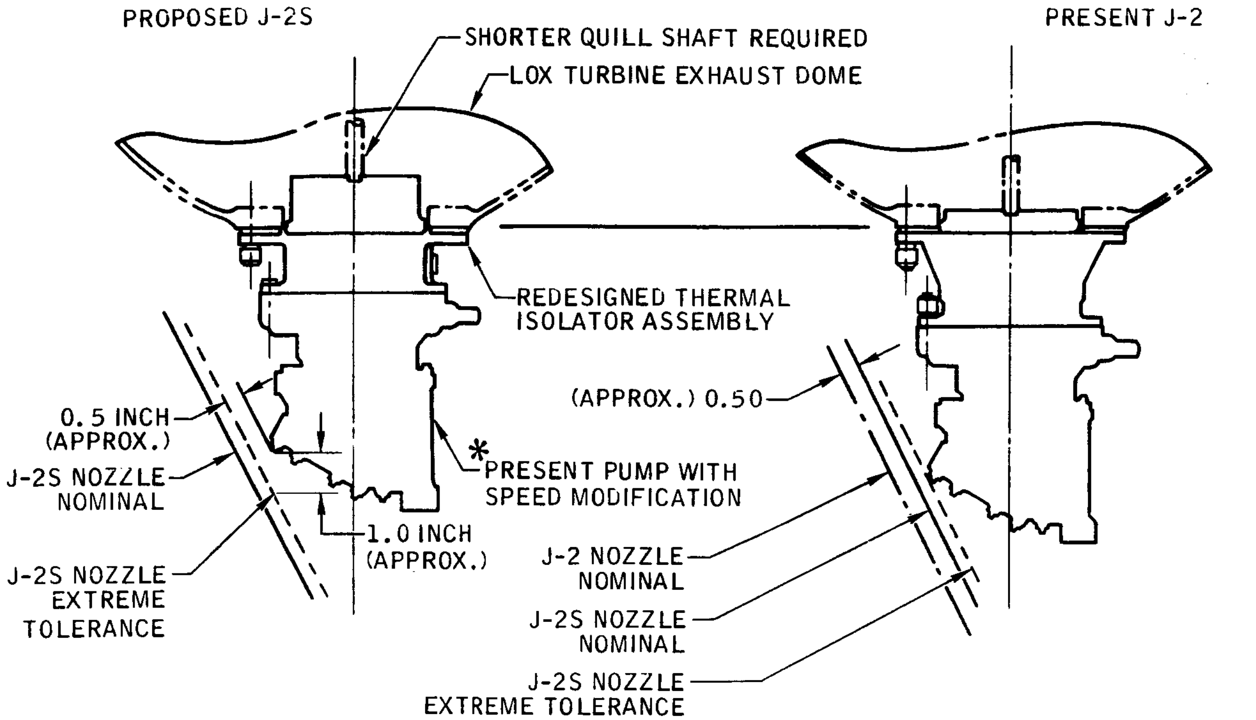
Not all details of the isolator redesign have been resolved, but some considerations are included below:

a. The present nut and bolt arrangement will not fit between the new isolator flanges. The Rocketdyne provided studs are presently too long and could be shortened. (Four .063 washers are presently used.) Bolts could also be used if the isolator flange holes are slotted to allow bolt installation. The pump-to-isolator bolt installation could be inverted to give more clearance, but slots in the isolator flange would also be required on this end because of the bolt length required.

b. Consideration must be given to moving the location of the seal vent port. Sufficient clearance will still be available between the flanges.

c. Redesign of the thermal isolator is expected to reduce the insulation capability. This will increase the maximum pump temperatures experienced during orbit and increase the heat loss during ground hold. The margin in thermal isolation capability of the present design is such that MDAC is confident that the isolator can be shortened but requalification of this feature is necessary.

The J-2S Interface Criteria Document, R-7211 specifies that the temperature range of the mounting pad face and drive spline with exhaust system gases is -250°F to +900°F. This is the same range specified for the J-2 engine. The



*PUMP - ISOLATOR - LOX TURBINE DOME IN WORST TOLERANCE POSITION IN RELATION TO NOZZLE

FIGURE 10.3.2-12 MAIN HYDRAULIC PUMP/ENGINE BELL INTERFERENCE MAIN HYDRAULIC PUMP

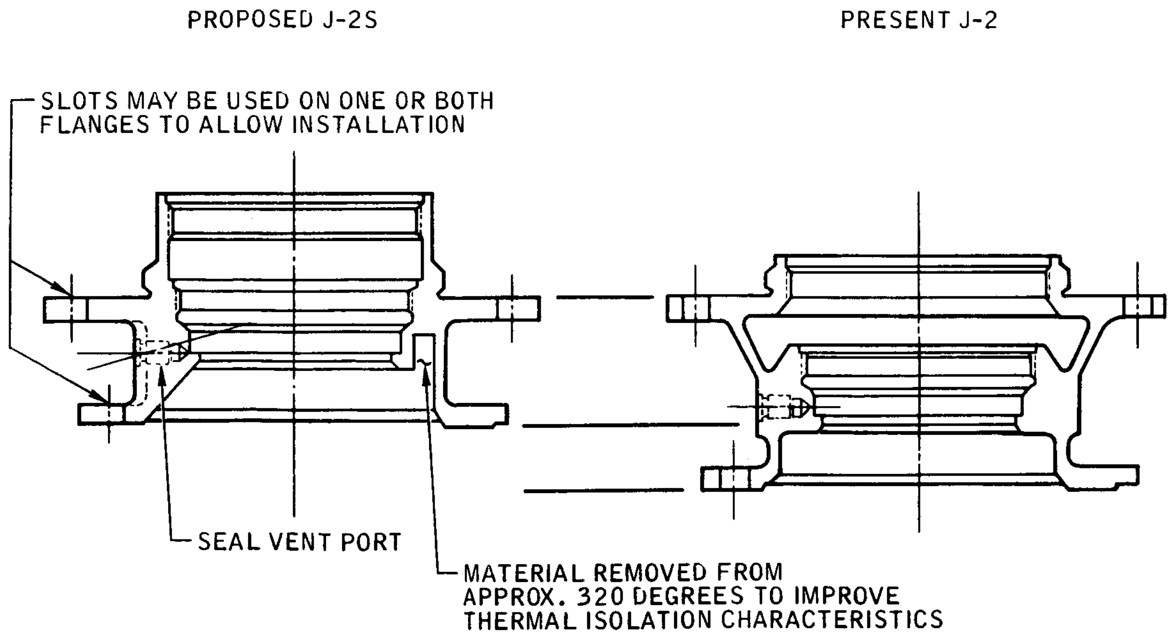


FIGURE 10.3.2-13 THERMAL ISOLATOR REDESIGN

10.3.2.8 (Continued)

high temperature is related to engine burn periods. The maximum acceptable oil temperature established for any point in the hydraulic system is 275°F. Flight test data on several missions shows that the oil temperature in the pump levels off at approximately 180°F after 35 minutes of soak after each engine burn period. The low temperature requirement is related to ground hold periods. During ground hold the hydraulic system is maintained at acceptable levels by periodic operation of the auxiliary pump with ground electrical power. A temperature sensor in the engine driven pump is monitored during ground hold and the auxiliary pump is operated as required. Reduction of the isolator capability is expected to increase the heat loss rate through the pump mounting pad and require additional auxiliary pump operation. The present system is considered capable of handling this additional heat requirement.

Adaptation of the present S-II pump manufactured by the American Brake Shoe Company is not considered feasible because the pump would have to be almost wholly redesigned and a hot gas shaft seal added. Specifically, the program would have to include the following:

a. The pump manifold assembly and case would have to be redesigned to accommodate the S-IVB hydraulic system method of recirculating fluid through the pump in order to keep the fluid at an operating temperature throughout the coast mode portion of the flight.

b. The pump would require a hot gas shaft seal, and since no such seal is available, the seal would have to be designed, developed and qualified. MDAC experience with the design, development and qualification of the hot gas seal on the present S-IVB pump indicates that this would be a major effort.

c. The modifications required to adapt the pump to the S-IVB hydraulic system configuration would invalidate the pump qualification.

A survey of other available pumps was made and it was concluded that another feasible unit would be a standard Vickers fixed angle variable (FAV) displacement pump, Model PV012, modified to adapt to the S-IVB hydraulic system plus the addition of a hot gas shaft seal. The factors supporting this conclusion are:

a. The FAV pump provides an envelope with substantial clearance mounted on the present isolator assembly design.

b. The hot gas shaft seal requirement could be met by using the hot gas seal developed and qualified on the current S-IVB pump. The qualified seal group can be fitted to the new pump as a unit, requiring only modification to the seal group housing to adapt to the new pump.

c. The pump manifold assembly and case would have to be redesigned to adapt to the S-IVB hydraulic system method of recirculating fluid.

10.3.2.8 (Continued)

d. The isolator assembly requires no change. Since the same seal group would be used, the thermal isolation feature need not be requalified. Therefore only a partial requalification of the assembly is required.

Two of the three flex lines terminating at the pump would be lengthened slightly because the port location will be different on the FAV pump.

Selected Design

Considering the latest information received from Rocketdyne on the clearance available, only two of the solutions discussed are considered feasible and provide adequate clearance. They are (1) adapt a new smaller pump to this application and use the present thermal isolator, and (2) redesign the thermal isolator, modify the present pump for the higher speeds, and use a new shorter quill shaft.

Change (2) which includes redesign of the thermal isolator has been selected based upon minimum change and minimum cost.

Component Test Plan

A limited requalification test program is required to assure satisfactory operation of the modified pump and isolator assembly. The test requirements will be similar to those established for the currently utilized assembly. Two specimens will be used in the program. The program will be conducted in accordance with the test plan in Table 10.3.2-XV.

Satisfactory operation must also be verified on Battleship type tests for both high and low temperature limits.

TABLE 10.3.2-XV. REQUALIFICATION TEST PLAN FOR THE ENGINE DRIVEN HYDRAULIC PUMP

	Spec. 1	Spec. 2	Structural	Operational	Repeat Cycle
Room Temp	X	X	X	X	X
High Temp	X	X	X	X	X
Proof	X	X	X	X	X
Leakage	X	X		X	
Vibration	X	X		X	
Flow ΔP	X	X		X	
Duty Cycle (Hot Gas)	X			X	X

10.3.2.8 (Continued)

Hydraulic System Installation

A hydraulic line support on the J-2S engine has been moved approximately 2.0 inches from its location on the J-2 engine. Three hard lines must be redesigned as a result of this change. Three flexible lines which terminate at the same location will not require change. The new hard lines may be requalified by similarity and no additional tests are required. The assumption here as well as with the remainder of the hydraulic system is that the vibration and acoustic environments are the same for the J-2/S-IVB stages and the J-2S/S-IVB stages. The three flexible lines terminating at the engine driven pump will not be modified since the pump location is only changing approximately one inch in the direction reducing the point to point length. The hydraulic system hardware drawings which must be modified are listed in Table 10.3.2-XVI.

The hardware changes include (1) change a spacer thickness in the pump displacement control to limit pump flow and power at the higher speeds, and (2) redesign the thermal isolator to reduce the installed pump envelope to allow more clearance with the J-2S nozzle.

TABLE 10.3.2-XVI. FLIGHT CONTROL SYSTEM HARDWARE CHANGES

Item or Drawing Title	Part Number	Quantity	Change
Tube Assembly - Low Press, Engine	1B32326	1	Redevelop
Tube Assembly - High Press, Engine	1B32327	1	Redevelop
Tube Assembly - Therm Cond Engine	1B32328	1	Redevelop
Engine Driven Hydraulic Pump (SCD)	1A66240	1	Modify and Requalify
Thermal Isolator	1A86457	1	Redesign
Baffle	1A86372	1	Redesign
Pump Isolator Assembly	1A86847	1	Revise
Hydraulic System Installation	1B62563	1	Revise
Actuator, Hydraulic (SCD)	1A66248	1	Revise

10.3.2.8 (Continued)

Auxiliary Hydraulic Pump Operating Schedule and Energy Requirements

The auxiliary hydraulic pump operation schedule and the on-board energy consumed from aft battery number 2 on the S-IVB/J-2S stages will be different than on previous S-IVB/J-2 stages. There are several significant contributors to this change.

a. The chilldown pumps which also consumed energy from aft battery number 2 have been eliminated. The auxiliary pump on the S-IVB/J-2 is started 310 seconds before the engine restart sequence to prevent the transient power surges associated with auxiliary pump start up from being sensed by the chilldown inverters. This lead time is no longer required.

b. The J-2S restart sequence requires that the auxiliary pump be operated during idle modes for stage attitude control.

c. Operation of the auxiliary pump is required for additional heating cycles because of increased coast duration for the synchronous mission.

The energy requirements for the two missions considered are shown in Table 10.3.2-XVII.

The following assumptions were made in determining the energy requirements.

a. The auxiliary hydraulic pump must be operated during all periods listed. The current shown is the average expected for a worst case flight condition.

    1. Prelaunch - 45 amp.

    2. Boost - 45 amp.

    3. 50 seconds before an idle mode or S-IVB burn period for filling accumulator and centering engine - 45 amp.

    4. All idle modes - 45 amp.

    5. All S-IVB burn periods - 30 amp.

    6. Auxiliary pump mixing heating cycles as required below - 45 amp.

b. Mixing and heating cycles are required at the following approximate times during all coast periods.

    1. Mixing cycle 48 seconds duration at every hour unless a heating cycle is scheduled at that time.

    2. Heating cycle 480 seconds duration at the end of every three hour period.

The auxiliary motor air supply mass has been evaluated in previous studies for missions as long as the synchronous mission and found to be adequate.

TABLE 10.3.2-XVII. AUXILIARY HYDRAULIC PUMP BATTERY ENERGY REQUIREMENTS J-2S ENGINE

Operating Mode	LOR Mission			Synchronous Mission		
	Time Power On Seconds	Current Amps	Energy Consumed Amp-Hrs	Time Power On Seconds	Current Amps	Energy Consumed Amp-Hr
Prelaunch (Internal Power)	30	45	0.37	30	45	0.37
Boost	498.9	45	6.24	489.9	45	6.24
First S-IVB Burn						
Idle Mode						
Full Thrust	120.8	45	1.51	58.4	45	0.74
First Coast						
Thermal Cycles						
No. 1 Mix	48	45	0.60	48	45	0.60
2 Mix	48	45	0.60	48	45	0.60
3 Heat	480	45	6.00	480	45	6.00
4 Mix	48	45	0.60	48	45	0.60
5 Mix				48	45	0.60
6 Heat				480	45	6.00
7 Mix				48	45	0.60
Prefire Operation	50	45	0.63	50	45	0.63
Second Burn						
Idle Mode	100	45	1.25	100	45	1.62
Full Thrust	291	45	3.63	208.7	45	2.61
Second Coast						
Thermal Cycles						
No. 1 Mix				48	45	0.60
2 Mix				48	45	0.60
3 Heat				480	45	6.00
4 Mix				48	45	0.60
Prefire Operation				50	45	0.63
Second Burn						
Idle Mode				100	45	1.62
Full Thrust				92.5	45	1.15
TOTAL ENERGY (AMP-HRS)			21.43			38.41

10.3.2.8 (Continued)

Handling and Checkout

Handling and checkout procedures associated with the hydraulic system have been reviewed to determine what changes are required to make them compatible with the J-2S engine. No significant changes are required in the fill, flush and bleed procedure, the hydraulic subsystem test requirements drawing, and the engine alignment procedure since the basic hardware remains unchanged. The nozzle exit dimensions to which the engine alignment fixture must mate are the same for the J-2S as the J-2 engine. Three drawings will be changed to reflect new hydraulic system part numbers.

Procedure	Dwg No.	Hours Saved or Added To Operations/Stage
Test Requirements Hydraulic Subsystem S-IVB Stage HB	1B40171	NONE
Fill, Flush, Bleed and Fluid Samples, Hydraulic System - HB	1B40973	NONE
Engine Alignment Procedure - S-IVB/HB	1B39095	NONE

Instrumentation

The flight instrumentation on the hydraulic system for J-2S/S-IVB stage will be the same as for the J-2 S-IVB stage with one exception, i.e., differential pressure transducers will be used on the actuators for early production stages as was done for early J-2/S-IVB stages. Data will be evaluated to determine actual gimbal loads.

10.3.2.9 Environmental Control System

Environmental control changes are necessary due to the increased coast periods of the synchronous mission. Thermal control is accomplished by active (heater blankets) and passive (radiation shield) insulation.

Heater Blankets - Twelve components on aft skirt panels Nos. 2, 3, 4, 8, 17 and 19 (Figure 10.3.1-7) will need active heating in addition to radiation shields. Table 10.3.2-XVIII indicates the units which require active heating, their location, temperature limit and heating capacity required. The active heating will be accomplished by form-fitting heater blankets made of synthetic rubber with heater elements in the walls, Figure 10.3.2-14. The heater blanket will have two thermostats in parallel, each one located adjacent to the area on the electronic component with the highest mass density. Either thermostat can activate the blanket, and in this way the operation of the heater blanket will be more stable. The outside of the heater blanket will be covered with a low emissivity aluminum surface. Inside in some areas, soft foam material (e.g., polyurethane) will be used. The purpose of the foam is to provide a mechanical bond and act as a shock absorber between the electronic component and the blanket walls. The foam material will be strategically located to assure maximum heat

TABLE 10.3.2-XVIII. ELECTRICAL COMPONENTS REQUIRING ACTIVE THERMAL CONTROL

Name	Location	Problem	Fix
Switch Selector	Aft Panel 2	Exceeds Low Temperature Limit of -13°F	15 Watt Heater Blanket On Each Unit
Remote Analog Submultiplexer (2)	Aft Panels 3, 8	Exceeds Low Temperature Limit of -4°F	
Remote Digital Submultiplexer	Aft Panel 3		
Multiplexer Assembly (4)	Aft Panel 4, 8		
50 Amp Motor Driven Switch (3)	Aft Panel 19	Exceeds Low Temperature Limit of -30°F	5 Watt Heater Blanket On Each Unit
50 Amp Motor Driven Switch	Aft Panel 17		
Cold Helium Bottle Pressure Transducer	Auxiliary Tunnel	Exceeds Low Temperature of -300°F	Recalibrate Unit At Lower Temperature Or Install Heater Blanket On Unit

10.3.2.9 (Continued)

transfer between the heater element and the component. The blanket will contain an electric receptacle and a stud fastener arrangement. The heater blankets will be in kit form, suitable for installation at KSC, with no need to disconnect cables or move the electronic components.

The electrical equipment heaters "on" and "off" command will permit thermal conditioning of the 12 components as required during the mission. The sequencer "on" command applies 28 volts dc to a heater power bus at a predetermined time. This enables all heater blankets simultaneously. The termostat control devices on each heater blanket will cycle power on and off to control the temperature within the required band. The weight of each heater will depend on the size of the component it heats; however, the weight of each heater should be less than eight ounces. The heater blankets will require development testing.

Passive Thermal Control

A number of stage components require passive protection from their respective ambients. This protection can be accomplished with low emissivity radiation shielding materials, coatings, and thermal isolators.

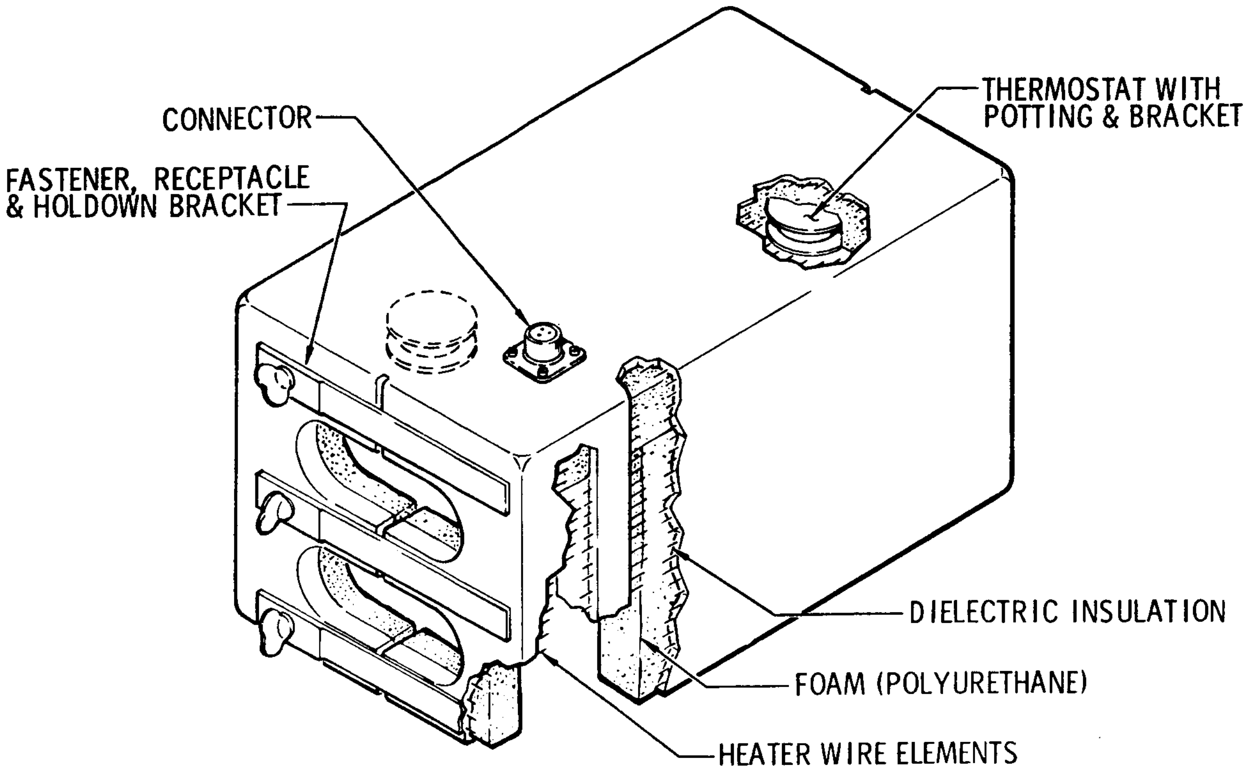


FIGURE 10.3.2-14 TYPICAL HEATER BLANKET - SYNC. MISSION

TABLE 10.3.2-XIX. ELECTRICAL COMPONENTS REQUIRING RADIATION SHIELDING

Name	Location	Problem	Fix
Sequencer	Aft Panel 1	Exceeds Low Temperature Limit Qualifying Component	Provide Low Emissivity Covers on Both Sides of Each Panel
56 V Power Distribution Panel	Aft Panel 17		
28 V Power Distribution Panel	Aft Panel 19		
Attitude Control Relay (2)	Aft Panels 6 and 18D		
Signal Conditioning Panels (6)	Aft Panels 7, 9, 12, 13, 14, and 16		
Liquid Level Control Units	Aft Panel 10		
Voltage Excitation Module Assembly	Aft Panel 18B	High Temperature Problem	Mount on Large Heat Sink; Locally Remove. Low Emissivity Coating From LOX Tank Dome.

10.3.2.9 (Continued)

Thermal Radiation Shielding

Radiation shielding consists of aluminized "Kapton" sheets covering the aft skirt electronic equipment. In accordance with appropriate thermal analysis, selected electronic equipment and panels must be completely covered with low emissivity ($\epsilon = 0.5$) material. The low temperature limit, at which the equipment was qualified, must not be exceeded to achieve mission accomplishment. The equipment and panels to be covered are listed in Table 10.3.2-XIX. Figure 10.3.2-15 shows a typical radiation shield application to a component support panel.

Coating and Mounting Devices

Certain components have thermal problems which require either isolation from the stage structure or a change in optical properties by the application of suitable coatings. These methods will permit them to stay within prescribed temperature limits:

Component	Location	Problem	Fix
Pneumatic Power Control Module	Thrust Structure Tray 1	Exceeds Low Temperature Limit of -125°F	Low Emissivity and Low Absorbtivity of Emissivity Ratio
Ambient Helium Fill Module	Thrust Structure Tray 1	Exceeds Low -125°F and High +165°F Temp Limits	Paint Thermal Isolators and Low ϵ and α/ϵ Paint
Ambient Helium Bottle (4-1/2 c.f.)	Thrust Structure	High Temperature	Low ϵ point
APS Tanks	Aft Skirt	Propellant Tank Temp exceeds lower limit	Make forward tank support of fiberglass
APS Self-Sealing Quick Disconnect	APS aft Bulkhead	Exceeds low Temp limit	Use fiberglass mounting bracket
Forward Battery No. 1 Forward Battery No. 2	Forward Skirt	Exceeds High Temp limit of +160°F	Paint section of forward skirt adjacent to batteries white

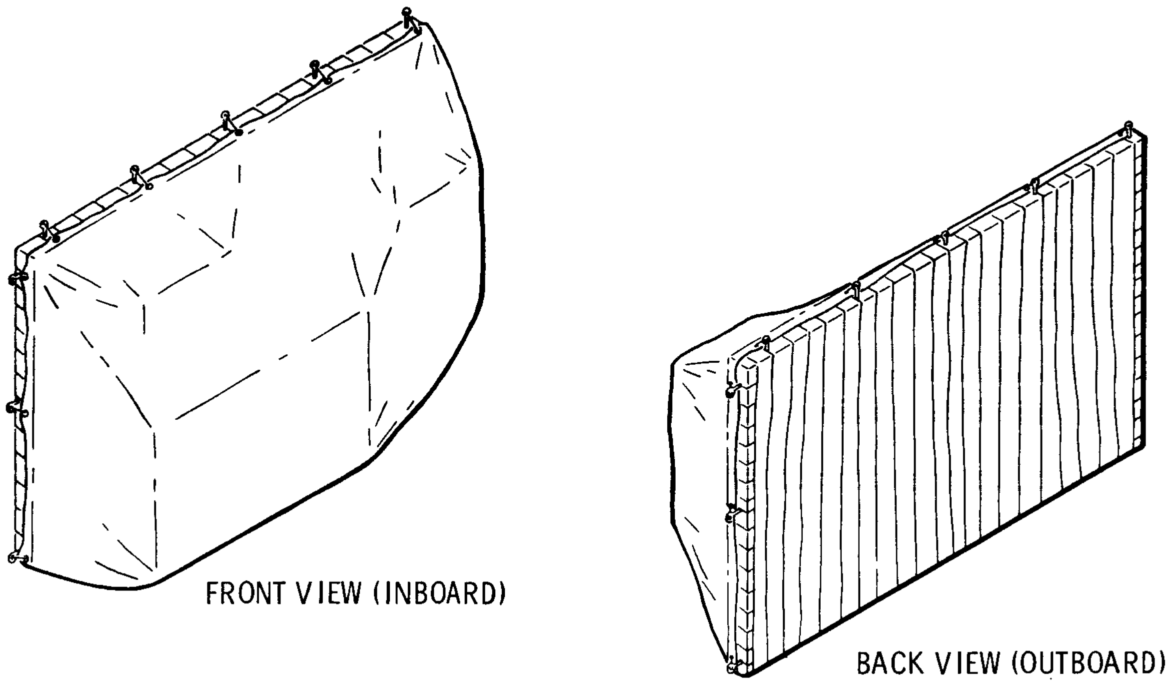


FIGURE 10. 3. 2-15 TYPICAL RADIATION SHIELD SYNCHRONOUS MISSION

10. 3. 2. 9 (Continued)

Existing Thermal Conditioning to be Removed

The insulation unit 1B39506 for the chilldown line which passes through the aft skirt and the aft skirt environmental control manifold will be removed. The ambient helium bottle shroud and thermal conditioning duct (1B57366 and 1A69978) will be deleted since the helium temperature conditioning required for inflight engine purges is not required with the J-2S engine. This duct is a branch of a larger duct which supplies purge gas to the thrust structure. The side outlet duct for the helium bottle will be capped, i.e., no design change will be made to the thrust structure purge gas duct.

Table 10. 3. 2-XX summarizes the hardware changes to the environmental control system.

10. 3. 2. 10 Ordnance Systems

Modifications to the existing stage ordnance systems as a result of the J-2S implementation result primarily in equipment deletions. The two solid propellant ullage motors and their fairings which are attached on the S-IVB aft skirt will be deleted, as will the two ordnance systems used for rocket ignition and jettison. These systems are exploding bridgewire (EBW) systems, requiring deletion of the EBW firing units, EBW detonators and detonator blocks, the fuse trains, and attaching hardware.

TABLE 10.3.2-XX. ENVIRONMENTAL CONTROL SYSTEM HARDWARE CHANGES FOR SYNCHRONOUS MISSION

Item	Part Number	Quantity	Change
Bottle Shroud	1B57366	1	Delete
Branch Duct	1A67978	1	Delete
Chill Return Line Insulation	1B39506	1	Delete
Heater Blankets	TBD	4	Add
Radiation Shields	TBD	3	Add

10.3.2.10 (Continued)

Hardware additions include providing doublers on the skirt over the area where the ullage rockets were mounted, as well as providing for new GFE ordnance associated with the J-2S engine solid propellant turbine starters. Stage hardware changes are tabulated in Table 10.3.2-XXI.

Deletion of the ullage rockets and their ignition and jettison systems will result in considerable time saving at KSC by eliminating certain test procedures and reducing the effort of others. However, this saving will be partially offset by added installations of the J-2S solid propellant turbine starters (GFE) and the associated EBW initiators (GFE).

Each of five ordnance systems on Saturn V are initiated by one redundant pair of EBW detonators. During a launch vehicle overall test with "plugs-out", a launch sequence is run with detonators mated to vehicle firing networks and fired into closed pressure chambers (test chambers). These same ten test chambers are refurbished and used for the plugs-out test for subsequent launches.

The plugs-out test for a J-2S launch will require a reduction in the quantity of test chambers for use with EBW detonators, and the new requirement for test chambers for use with the EBW initiator for SPTS. This latter test chamber may possibly require a new design, although other test chamber designs exist from previous Saturn programs.

TABLE 10.3.2-XXI. ORDNANCE SYSTEM HARDWARE CHANGES

Name	Number	Quantity	Change
Rocket Motor Instl. (1A82765)			
Rocket Motor (Ref)	1A81960	2	Delete
Rocket Motor Fairing Assy (Ref)	1A74671	2	Delete
Transducer Instl	1B58507	2	Delete
Live Components Instl	1A82766	2	Delete
Rocket Ignition System (1B57448)			
EBW Firing Unit	40M39515 (GFE)	2	Delete
EBW Detonator	7865742	2	Delete
CDF Manifold	1B58045 (GFE)	2	Delete
CDF Assembly	1B53581 (GFE)	9	Delete
CDF Pyrogen Initiator	1B53584 (GFE)	4	Delete
CDF Disconnect	1B74495	2	Delete
CDF Disconnect	1B56218	2	Delete
Frangible Nut	1A72620	4	Delete
Rocket Jettison System (1B57009)			
EBW Firing Unit	40M39515 (GFE)	2	Delete
EBW Detonator	7865742	2	Delete
Detonator Block	1A84223	1	Delete
CDF Assembly	1B56400	2	Delete
Explosives Kit, EBW Detonator	1B34834	1	Revise
Test Chamber Kit, EBW Detonator	1B58705	1	Revise*
Explosives Kit, Solid Propellant Turbine Starter	TBD	1	Add
Explosives Kit, EBW Initiator, SPTS	TBD	2	Add

* Assumes no new configuration chamber required for the SPTS EBW initiator

10.3.3 Electrical Systems

Three electrical subsystems are affected by the installation of the J-2S engine. These systems are: (1) the power and electrical distribution system which provides power to all of the components (valves, transmitters, range safety, etc.) at the proper time. This system is made up of batteries, power distributors, cables and connectors, etc., (2) the data acquisition system which provides data for automatic preflight checkout of the vehicle, monitoring of vehicles performance during powered flight, post flight evaluation of vehicle performance and verification of commands received in the vehicle from the ground station, and (3) the Propellant Utilization and loading system which controls engine mixture ratio to achieve simultaneous depletion of the propellants and provides signals which indicate mass of propellant on board for control of propellant loading.

10.3.3.1 Electrical Power Systems

The current profiles used to determine ampere-hour requirements for the batteries listed in Table 10.3.3-I are shown on Figure 10.3.3-1A through D, Figures 10.3.3-2A through D, and Figures 10.3.3-3A through D.

The impact of the J-2S engine on the power system of the LOR-J-2 vehicle is minimal as shown on Table 10.3.3-I. Considering present battery complement would be the replacement of the present aft No. 1 battery, a 1A59741-507, with a 1A83468-505 for the LOR mission with the J-2S. For this mission the options for aft No. 1 battery were 1A83470 (63 amp-hr), 1A83468-505 (87 amp-hr), and 1A59741-501 (256 amp-hr). The 1A83470 would fulfill mission requirements providing the battery heaters do not operate for an excessively long time. Another choice might be the 1A83468-505 (recommended) which would leave adequate safety margins. A comparison of the physical dimensions of batteries 1A83468-505 and 1A59741-507 indicates that the two batteries are nearly the same size but not interchangeable without physical rework. Even though the containers are comparable in size, twenty cells of the type used in battery 1A83468 do not appear to fit nicely into the container of battery 1A59741 because this container is divided into three smaller compartments causing space to be wasted. Therefore, a revision to the battery mounting structure would be necessary. The 1A83468-505 battery could be used without additional qualification testing since the qualification tests for the 1A83468-505 are virtually the same as those for the 1A59741-507. If change to the 1A83468-505 is considered undesirable, the 1A59741-507 presently installed could be utilized; however, the weight penalty over the 1A83468-505 would be 85 pounds with the battery loaded to only 18 percent of its capacity.

For the synchronous mission it will be necessary to change the present Forward No. 1 Battery from a 1A59741-507 to two parallel 1A59741-503 batteries rated at 366 amp-hr each. This results primarily from the additional power required for long range telemetry transmission. The Forward Bus No. 1 and the 366 amp-hr battery were originally sized for an R&D Data Acquisition System, so it is capable of handling the increased load. The additional 366 amp-hr battery is primarily for the increased stage operational time. This concept has been tested and used on the

TABLE 10.3.3-I. POWER SYSTEM EVALUATION

Battery	Part Number	Specification Rating (Amp-Hr)	Amp-Hr Required	Percent Utilization	Amperes Required	Weight Lbs
LOR MISSION WITH PRESENT J-2						
Fwd No. 1 (28 Volts)	1A59741-507	256	88	34	12.5 to 12.8	170
Fwd No. 2 (28 Volts)	1A68316-501	28	14.3	51	.05 to 5.5	20
Aft No. 1 (28 Volts)	1A59741-507	256	49	19	1.3 to 45.3	170
Aft No. 2 (56 Volts)	1A68317-503	84 @ 56V	41 @ 56V	49	0.4 to 88	83
Total Weight						443
LOR MISSION WITH J-2S						
Fwd No. 1 (28 Volts)	1A59741-507	256	82	32	12.9 to 13.2	170
Fwd No. 2 (28 Volts)	1A68316-501	28	10.3	37	.05 to 5.5	20
Aft No. 1 (28 Volts)	3 Options 1A83470 1A83468-505* 1A59741-507	 63 87* 256	 45.7	 73 53* 18	 0.9 to 45.4	 50 85* 170
Aft No. 2 (56 Volts)	1A68317-503	84 @ 56V	27.4 @ 56V	33	0 to 45	83
Weight Range						331 to 443
Weight of Best Option						*356
Weight Change from present J-2						-87

*Probable best option

TABLE 10.3.3-I. (Continued)

Battery	Part Number	Specification Rating (Amp-Hr)	Amp-Hr Required	Percent Utilization	Amperes Required	Weight Lbs
SYNCHRONOUS ORBIT WITH J-2S						
Fwd No. 1 (28 Volts)	Two 1A59741-503	366 (each)	617	84	32.0 to 49.7	400
Fwd No. 2 (28 Volts)	1A68316-501	28	7.8	28	.05 to 5.5	20
Aft No. 1 (28 Volts)	2 Options 1A59741-507* 1A59741-503	256* 366	91.6	36* 25	0.9 to 45.4	170* 200
Aft No. 2 (56 Volts)	1A68317-503	84 @ 56V	45 @ 56V	54	0 to 45	83
Weight Range						643 to 703
Weight of Best Option						*643
Weight Change from present J-2						+200

*Probable best option

10.3.3.1 (Continued)

S-IV with this type of battery. Additional testing with this particular battery will be done for increased confidence in the paralleling technique. The percent of utilization is greater than is normally used (84 percent) with the battery heaters operating with a 100 percent duty cycle. This is a smaller margin of safety than is normally used. However, with a constant current drain of approximately 25 amperes, the battery heaters will be activated infrequently, if at all. Assuming no heater activation, the percent of utilization is 76 percent which is acceptable. An E/I switch for each battery provides separation on external power. This prevents the batteries from bucking each other under no load and dissipating energy. On GSE command to internal power, the batteries are simultaneously paralleled and share the load. A complete evaluation (including circuit diagrams) has been performed in the Synchronous Orbit Capability Study under supplemental agreement 1145 of Contract NAS7-101.

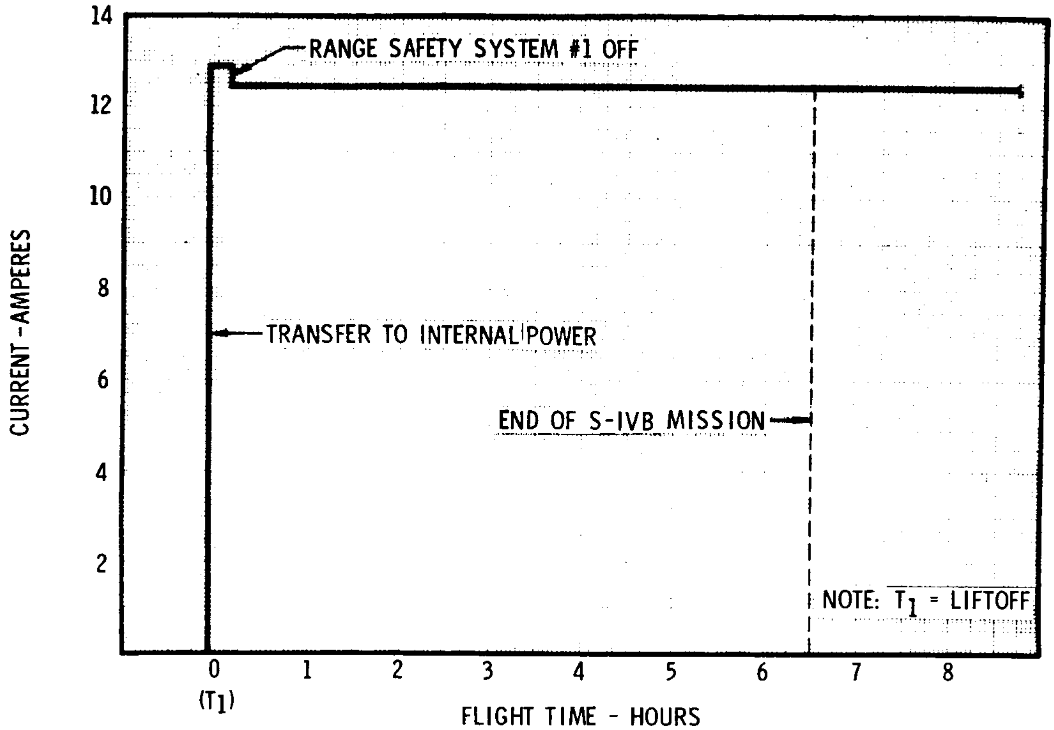


FIGURE 10.3.3-1A. 504N FWD. BATTERY NO. 1 CURRENT PROFILE (LOR MISSION USING J-2 ENGINE)

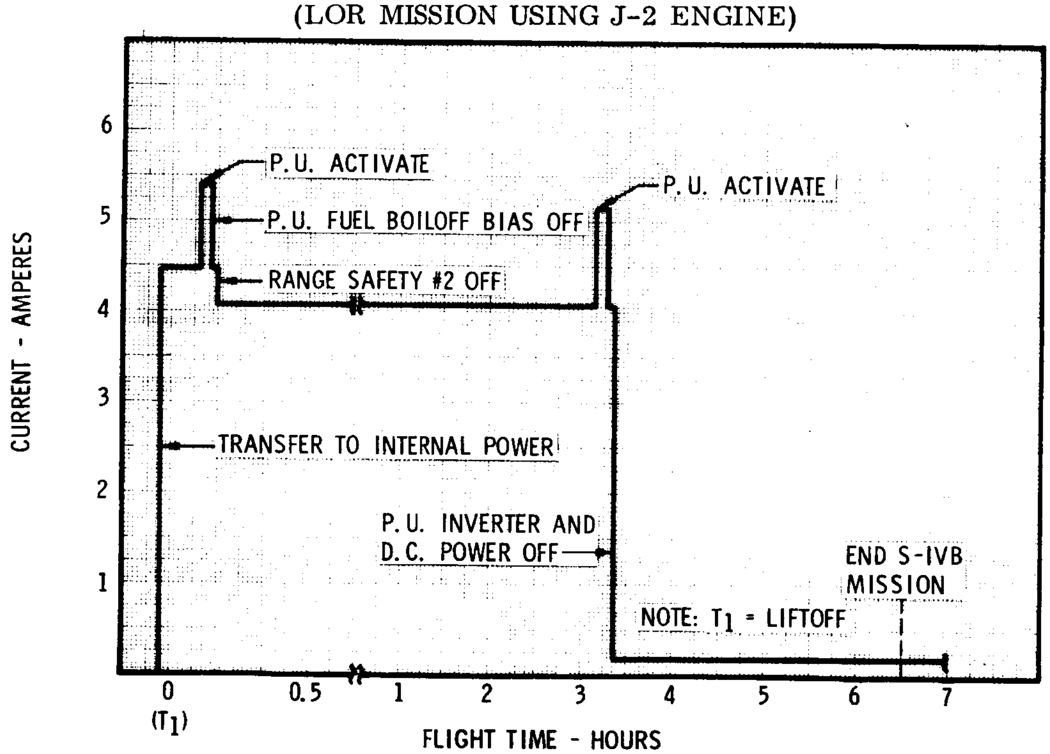


FIGURE 10.3.3-1B. 504N FWD BATTERY NO. 2 CURRENT PROFILE (LOR MISSION USING J-2 ENGINE)

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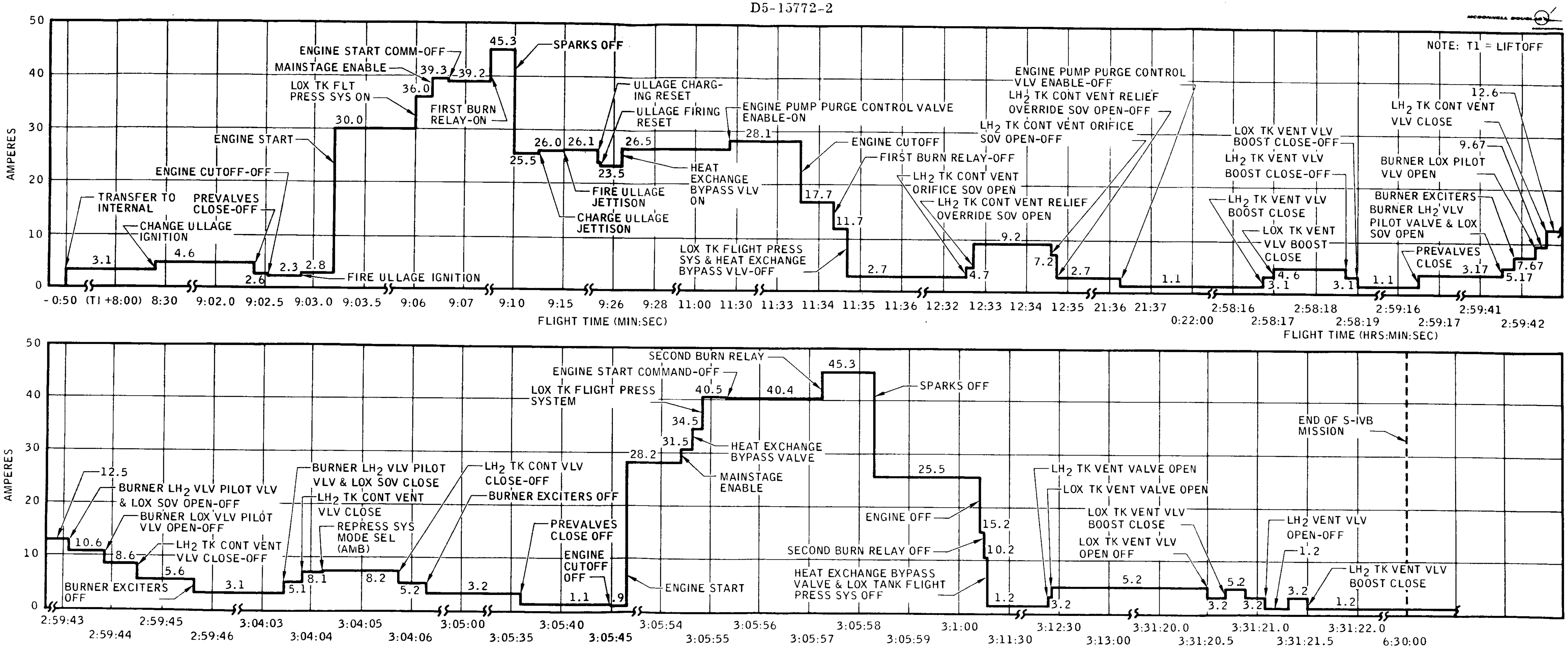

FIGURE 10.3.3-1C. 501N AFT BATTERY NO. 1 CURRENT PROFILE
(LOR MISSION USING J-2 ENGINE)

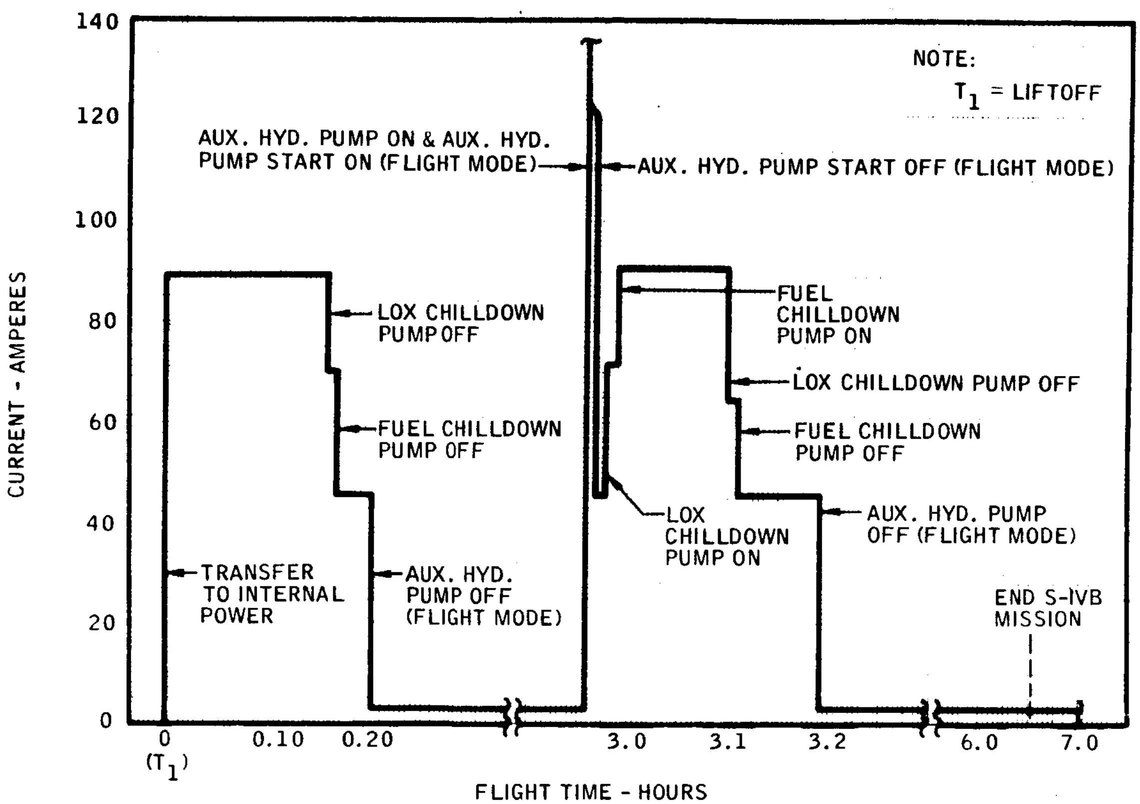

FIGURE 10. 3. 3-1D. 504N AFT BATTERY NO. 2 CURRENT PROFILE (LOR MISSION USING J-2S ENGINE)

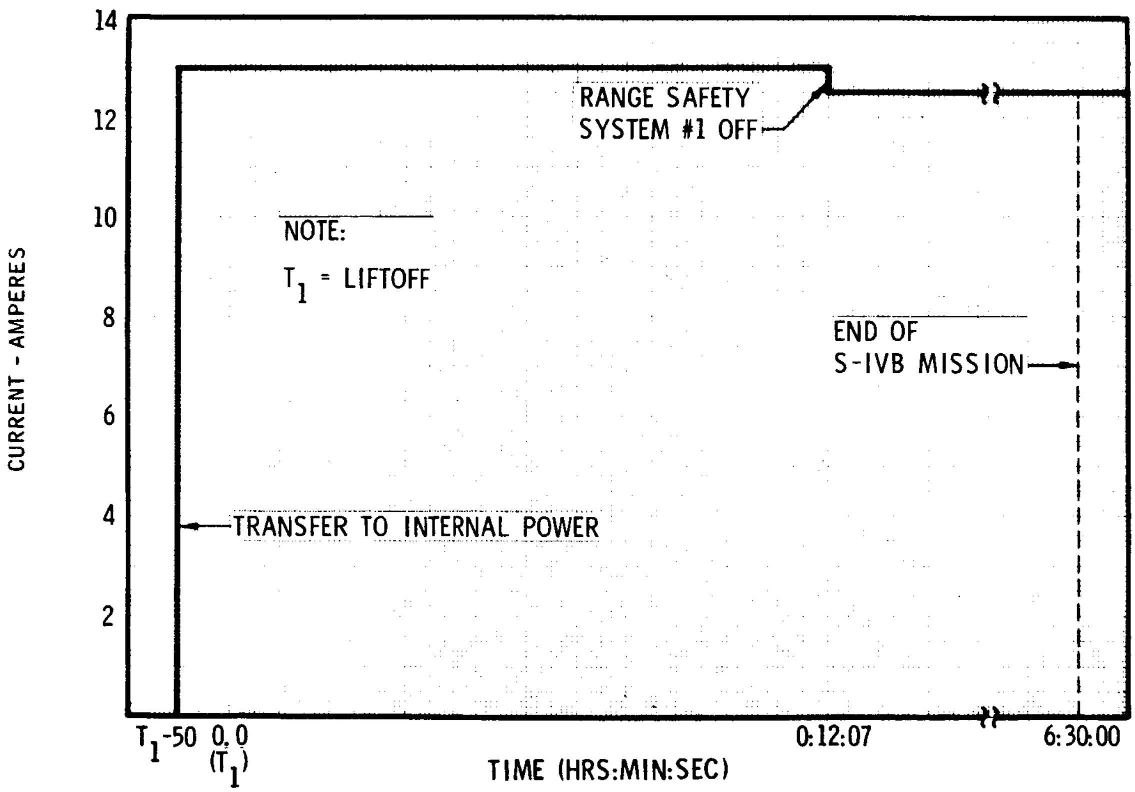

FIGURE 10. 3. 3-2A. FWD BATTERY NO. 1 CURRENT PROFILE (LOR MISSION USING J-2S ENGINE)

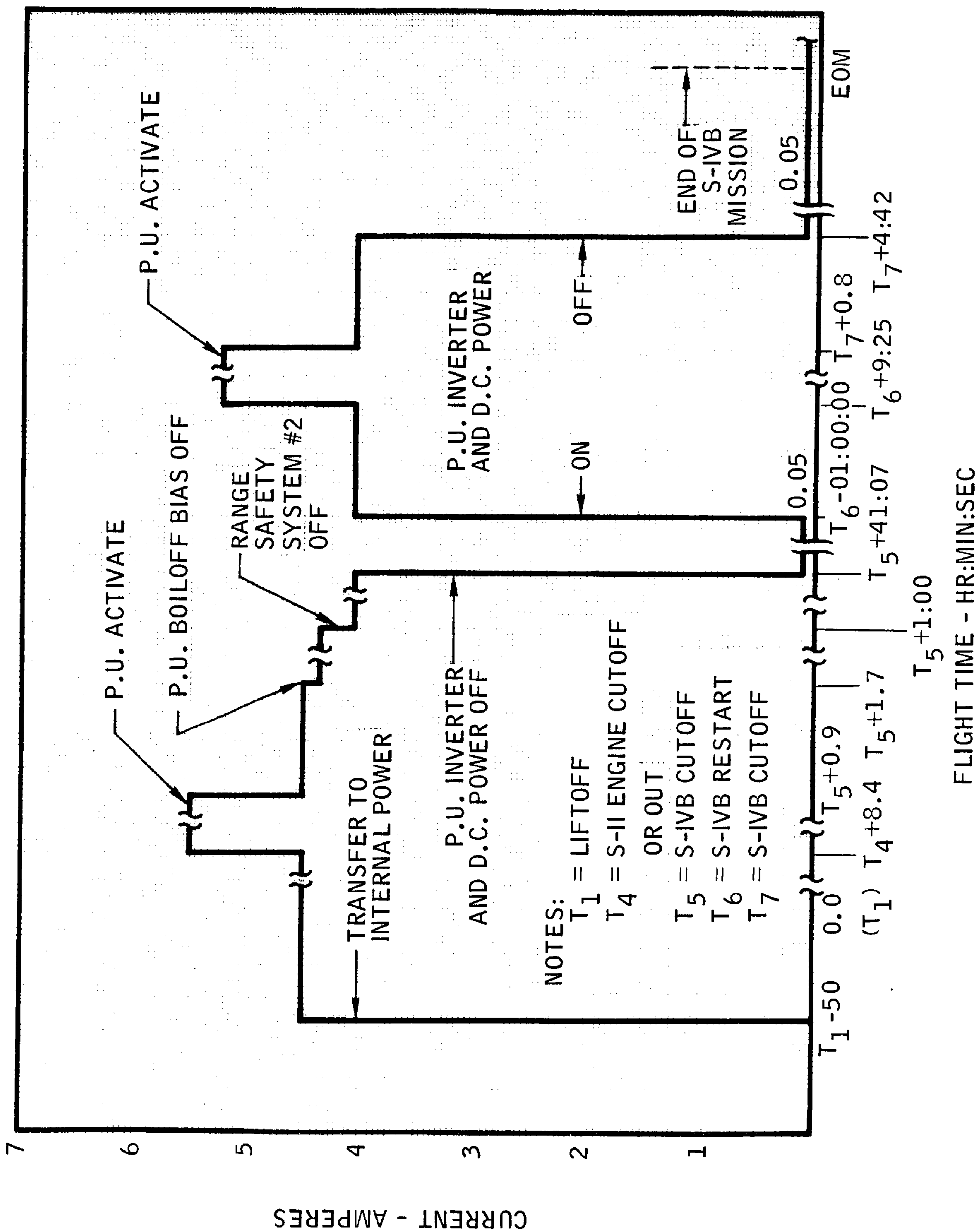

FIGURE 10.3.3-2B. FWD BATTERY NO. 2 CURRENT PROFILE (LOR MISSION USING J-2S ENGINE)

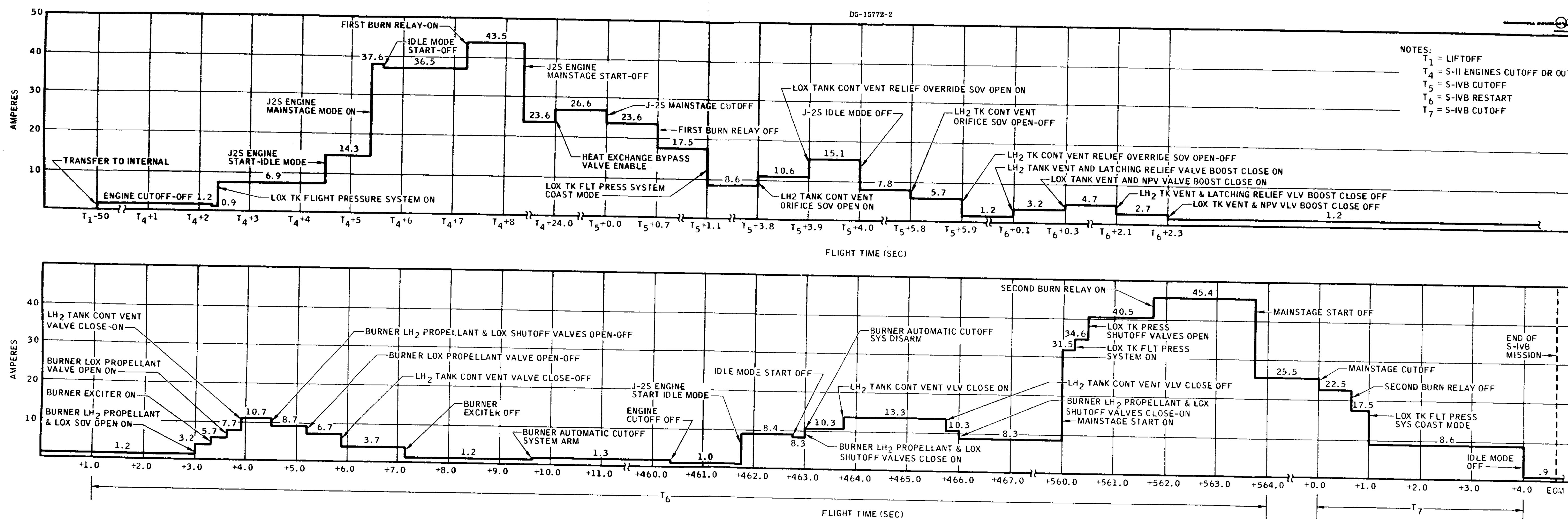

FIGURE 10.3.3-2C. AFT BATTERY NO. 1 CURRENT PROFILE (LOR MISSION USING J-2S ENGINE)

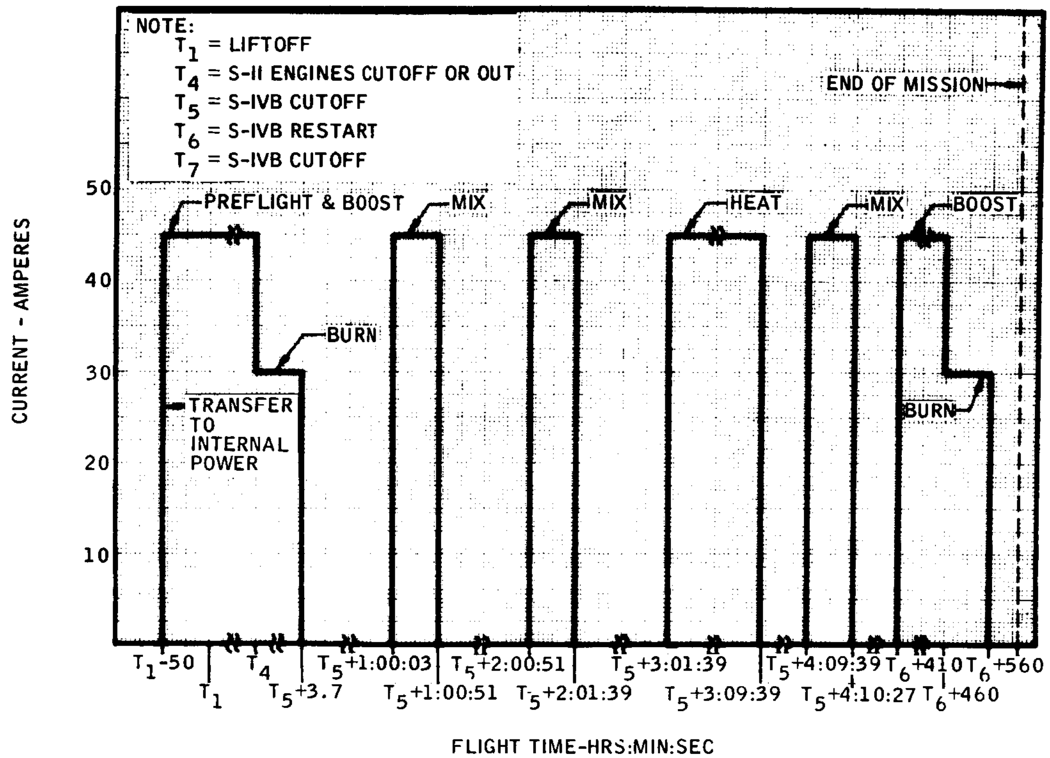

FIGURE 10.3.3-2D. AFT BATTERY NO. 2 CURRENT PROFILE (LOR MISSION USING J-2S ENGINE)

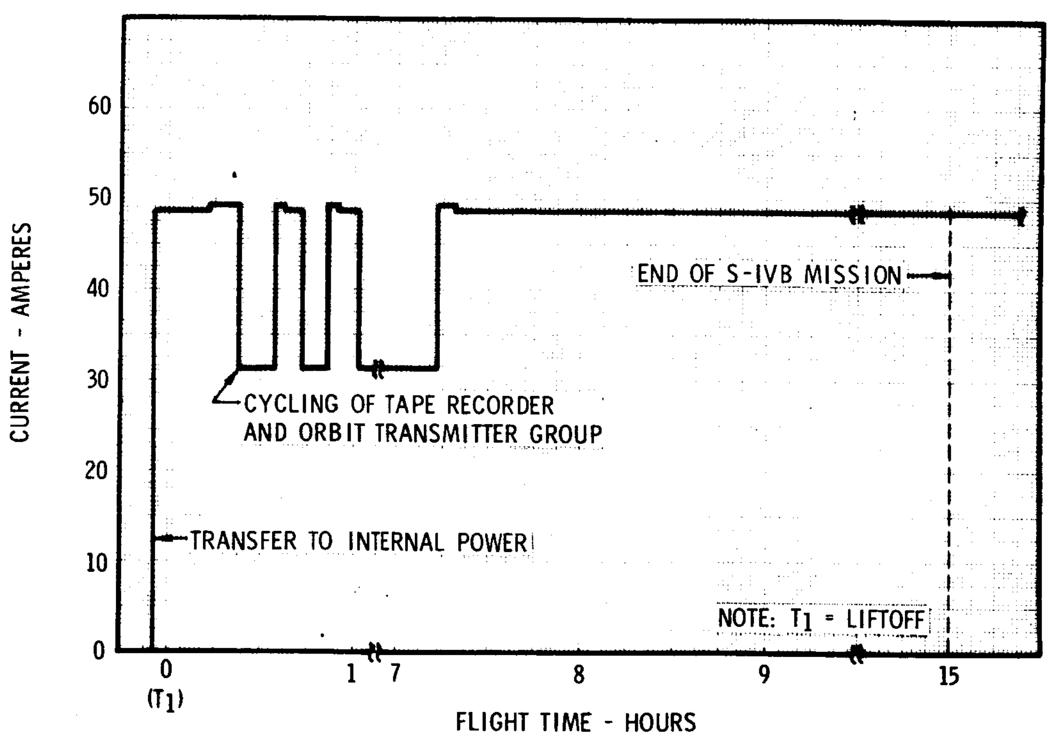

FIGURE 10.3.3-3A. FWD BATTERY NO. 1 CURRENT PROFILE (SYNC MISSION/J-2S ENGINE)

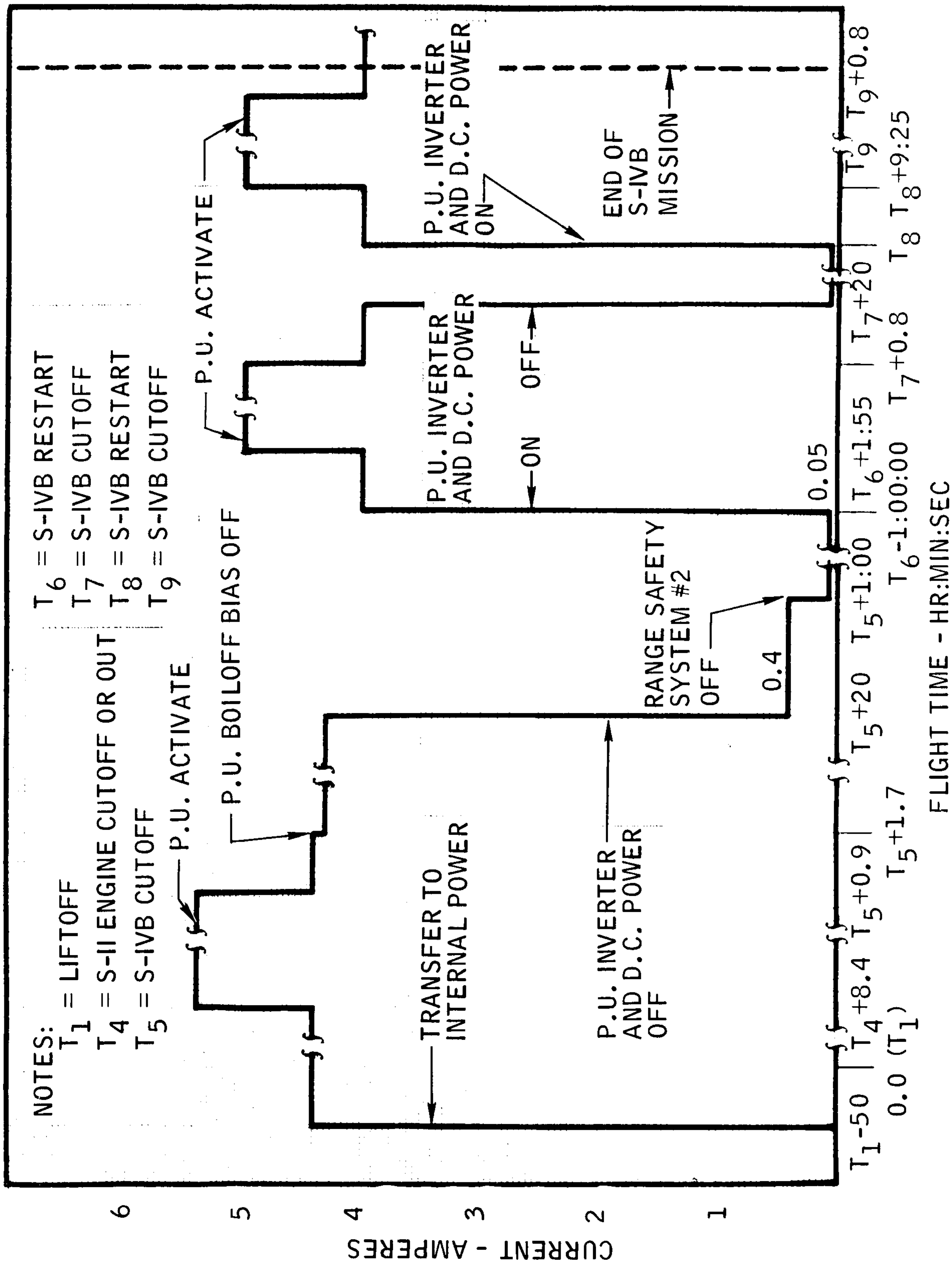

FIGURE 10.3.3-3B. FWD BATTERY NO. 2 CURRENT PROFILE (SYNC MISSION/J-2S ENGINE)

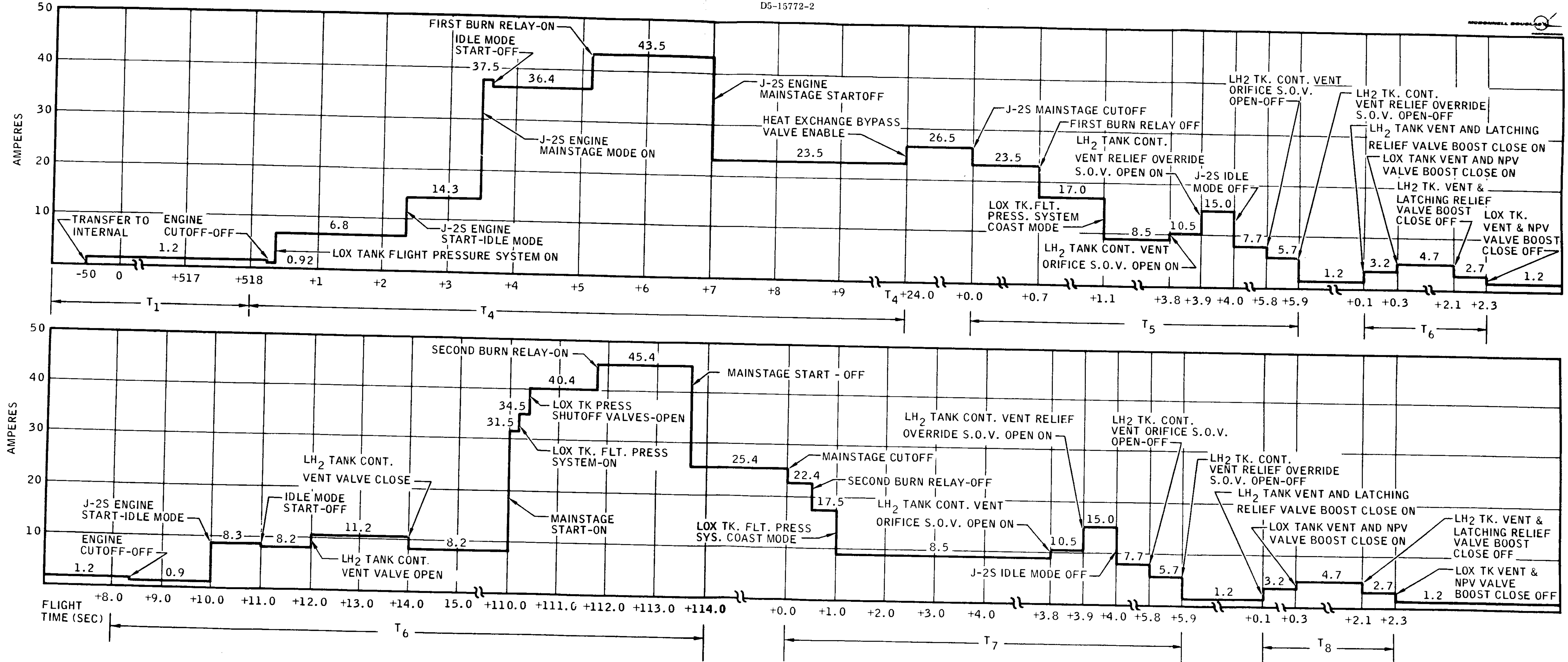

FIGURE 10.3.3-3C. AFT BATTERY NO. 1 CURRENT PROFILE (SYNC MISSION J-2S ENGINE) (Sheet 1 of 2)

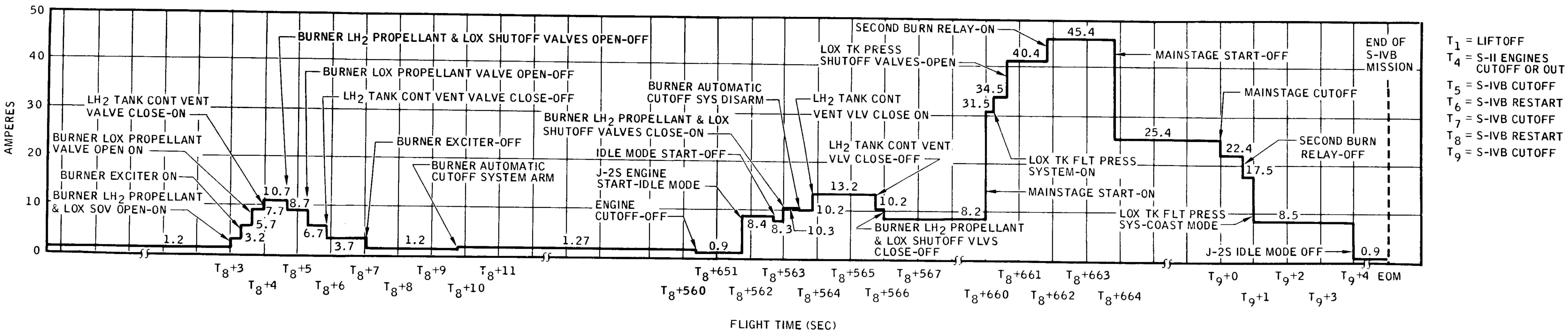

FIGURE 10.3.3-3C. AFT BATTERY NO. 1 CURRENT PROFILE (SYNC MISSION/J-2S ENGINE) (Sheet 2 of 2)

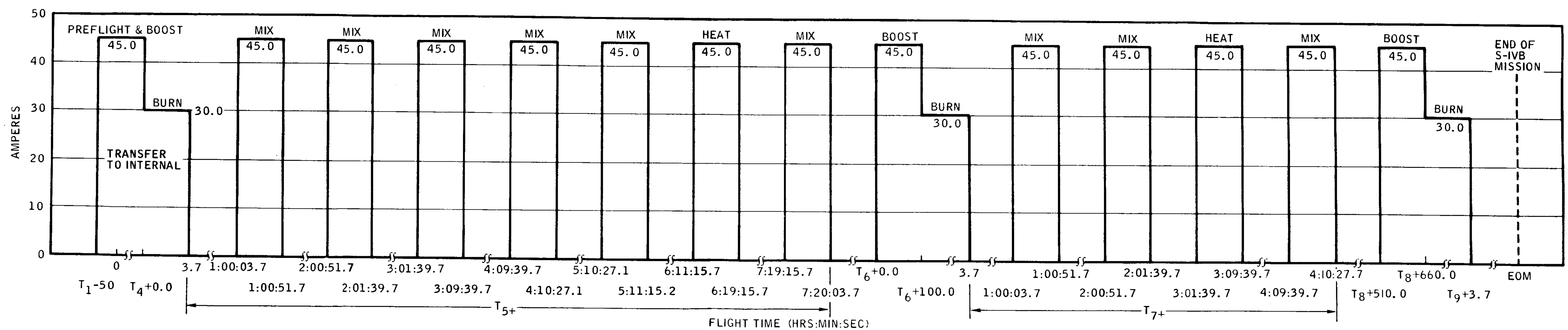

FIGURE 10.3.3-3D. AFT BATTERY NO. 2 CURRENT PROFILE (SYNC MISSION/J-2S ENGINE)

10.3.3.1 (Continued)

Pending a thermal analysis for the synchronous mission, the 1A57941-507 battery is recommended for aft No. 1 battery. If thermal analysis indicates the need for more battery heater power, it may be necessary to recommend a 1A59741-503 for the aft No. 1 battery. No structural support changes are required since the 1A59741-503 is the same physical size as the 1A59741-507.

10.3.3.2 Electrical Control Systems

The impact of the J-2S engine on the electrical control systems of the Saturn V Stage for both LOR and synchronous mission profiles, consists of (1) the deletion of the two chilldown inverters and associated controls, (2) deletion of the four EBW firing units, four EBW pulse sensors, and controls associated with the solid ullage rocket motors, (3) deletion of the LOX depletion sensors, control units, (4) depletion of the control circuits for the LOX and LH2 chilldown valves and prevalves, including the prevalve delay timer module, (5) deletion of the 72 pound OAMS ullage engine and associated controls from each APS, and (6) addition of a solid propellant turbine starter unit for each engine start. In addition, the engine pump purge has been deleted and the LOX dome purge has been delegated to the GSE. The start bottle and gas generator commands have been deleted through the use of SPTS to start the engine. The schematics for these circuits are shown in Figures 10.3.3-4 through 10.3.3-9.

Twenty-three switch selector channels will be deleted with six added to the switch selector for engine functions and twelve added for passivation. These channel assignments are required to operate the S-IVB/J-2S vehicle in LOR and synchronous modes. The list of deleted switch selector channel assignments are found in Table 10.3.3-II with the list of added channel assignments in Table 10.3.3-III.

Ten additional switch selector channel assignments are required on the synchronous mission over that of the LOR mission profile. These controls are related to the electrical equipment heater blankets and the tape recorder. These additional switch selector channel assignments are listed in Table 10.3.3-IV.

Due to the extreme cold environment encountered in the synchronous mission, heater blankets will be added to certain electrical equipment to maintain an optimum operating temperature. The tape recorder has been added to the synchronous mission vehicle to provide continuous coverage of the vehicle measurements throughout the insertion phase of the mission profile.

Stage modifications due to deletion of measurements or commands will be made in the wiring harnesses from either the function/engine interface and/or the first relay controlling the measurement or command. No modification is required internal to any of the major black boxes (the 56 volt aft power distribution assembly, the 28 volt aft power distribution assembly, aft control distribution assembly, sequencer and the switch selector). Any relay modules or switching assemblies not used as a result of deletions will remain in these units as spares.

The hardware removed as a result of the modifications discussed is listed in Table 10.3.3-V.

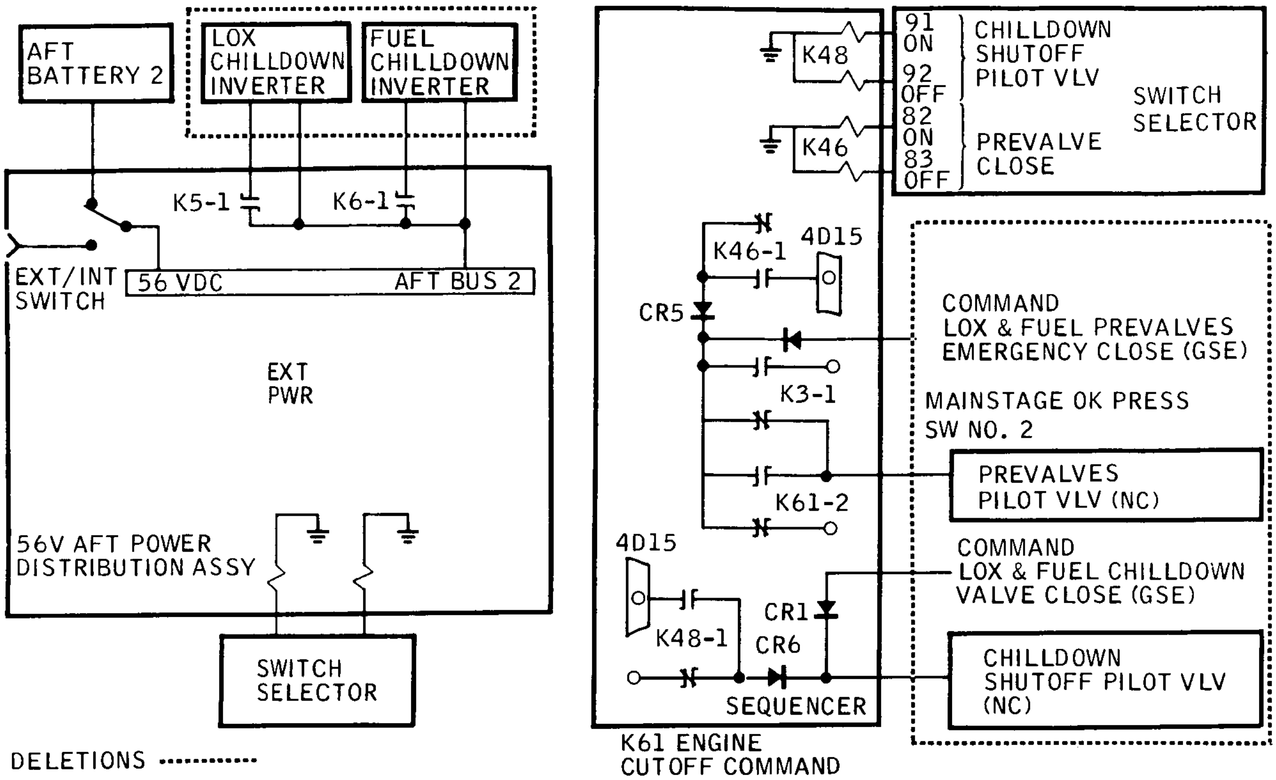

FIGURE 10.3.3-4. CHILLDOWN CIRCUIT DELETIONS

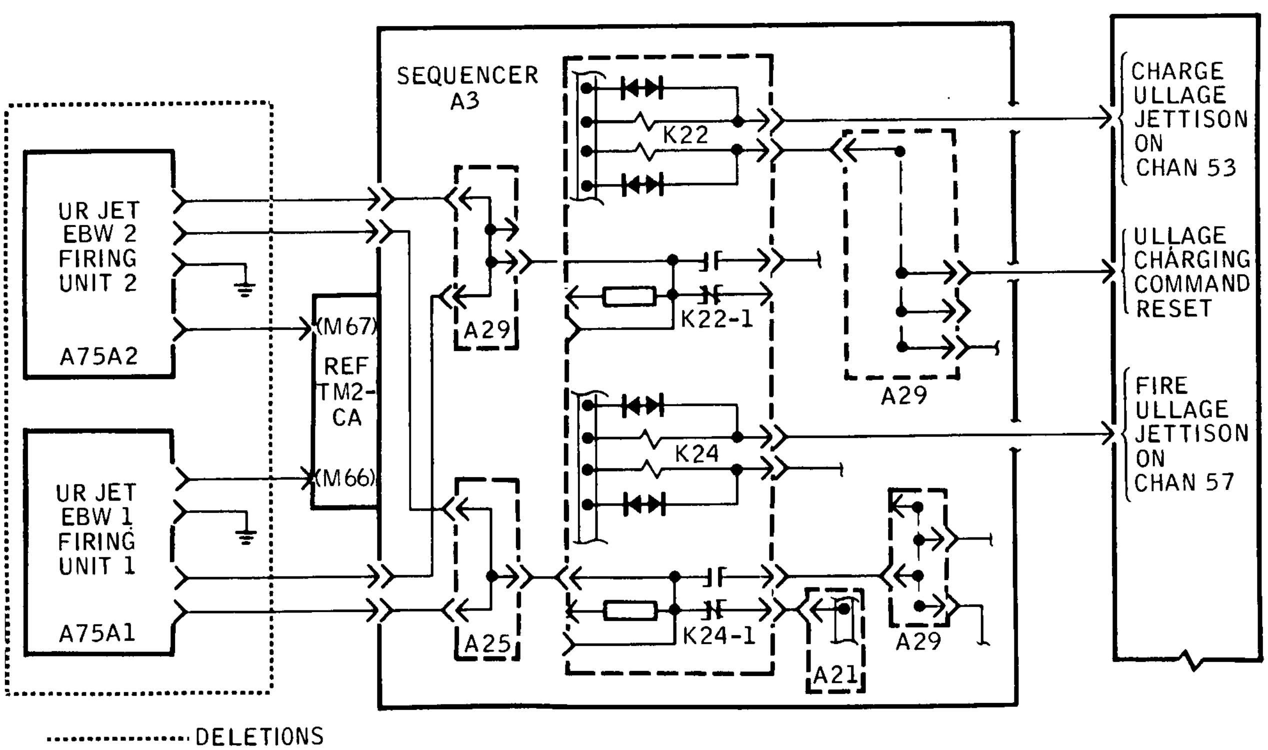

FIGURE 10.3.3-5. ULLAGE ROCKET JETTISON

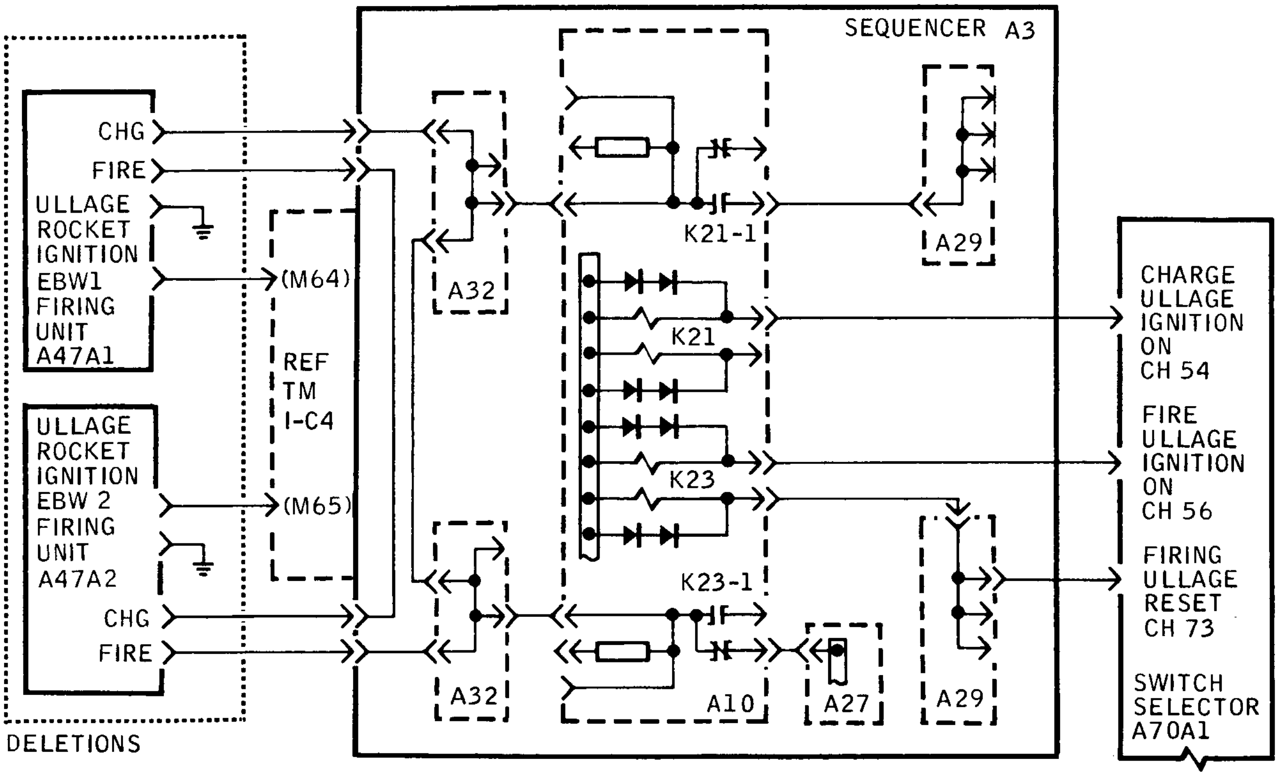

FIGURE 10.3.3-6. ULLAGE ROCKET IGNITION

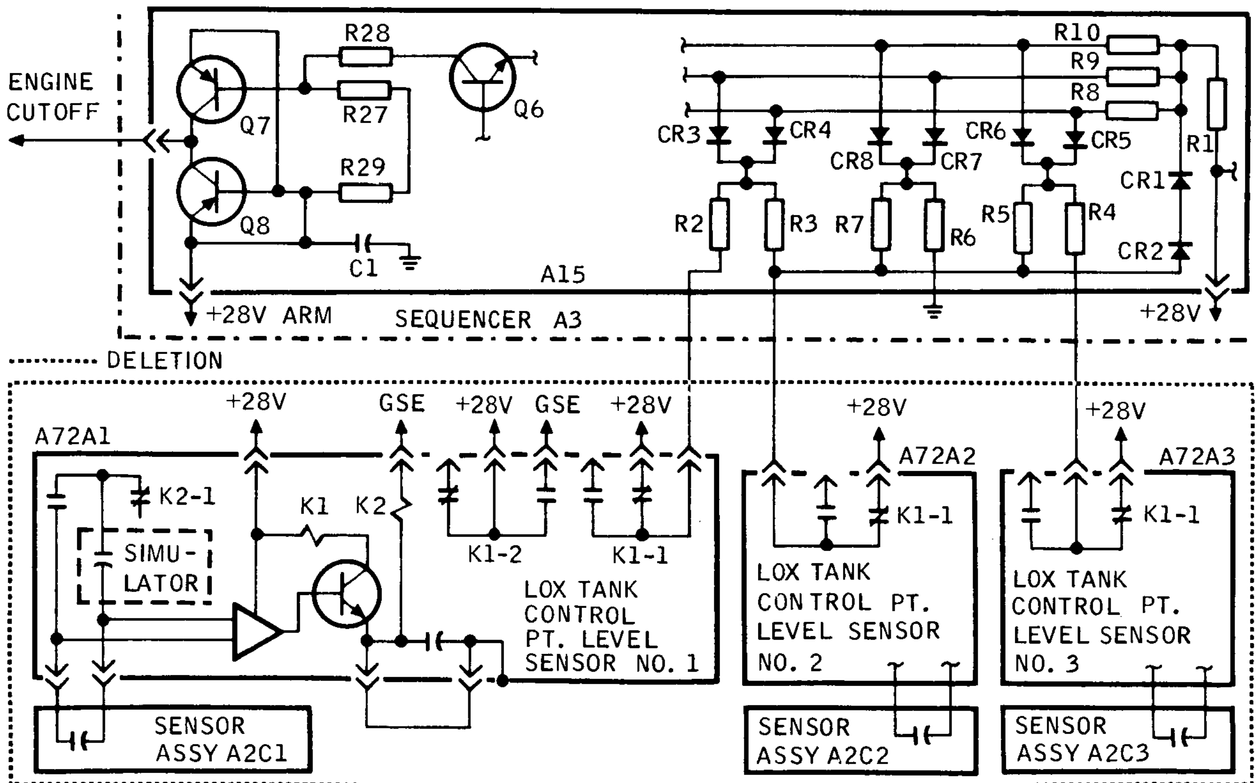

FIGURE 10.3.3-7. LOX POINT-LEVEL SENSOR DEPLETION

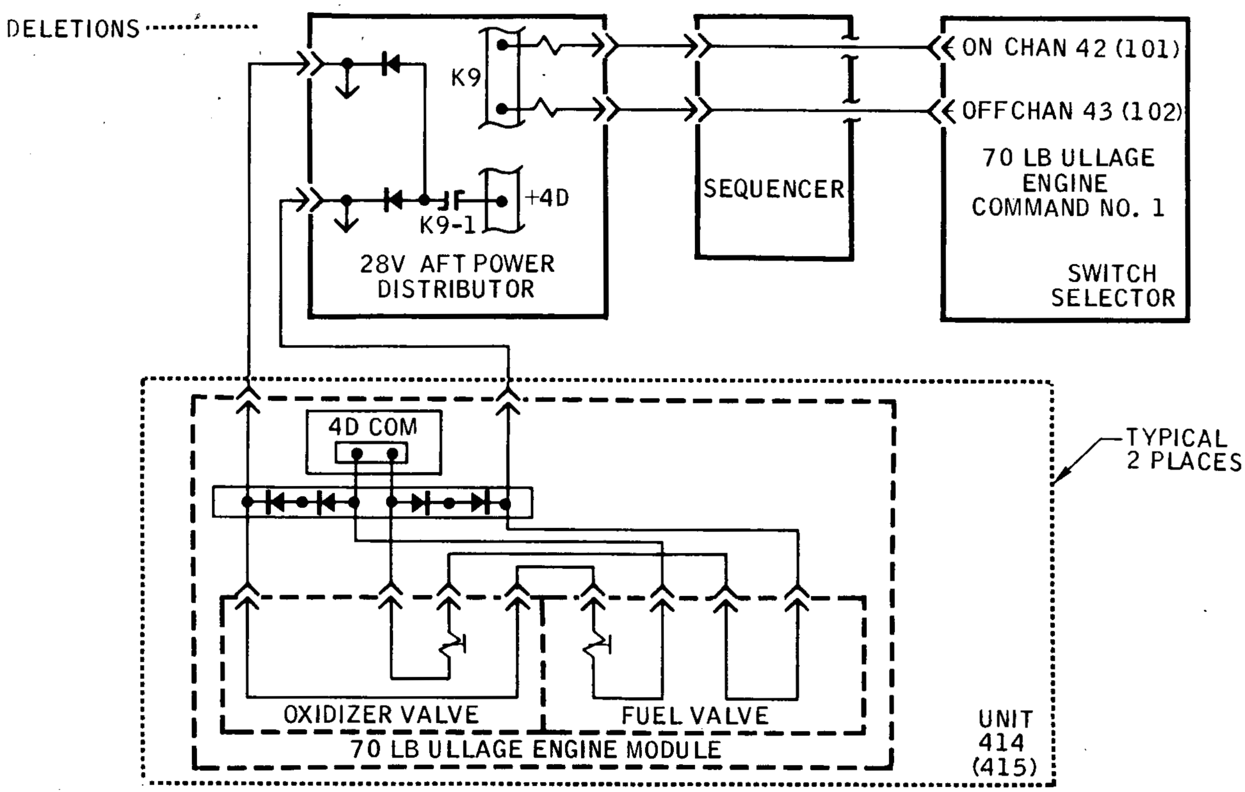

FIGURE 10.3.3-8. 70 LB ULLAGE ROCKET IGNITION

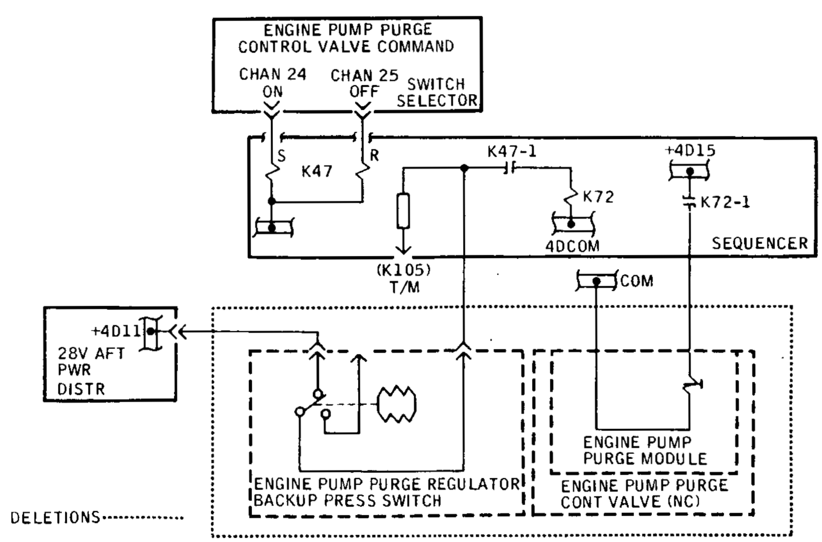

FIGURE 10.3.3-9. ENGINE PUMP PURGE

TABLE 10.3.3-II. SWITCH SELECTOR CHANNEL ASSIGNMENTS DELETED

Channel		Channel	
58	Fuel Chilldown Pump ON	2	Start Tank Vent Valve Open OFF
59	Fuel Chilldown Pump OFF	42	70 Lbs Ullage Engine Command No. 1 ON
22	LOX Chilldown Pump ON	43	70 Lbs Ullage Engine Command No. 1 OFF
23	LOX Chilldown Pump OFF	101	70 Lbs Ullage Engine Command No. 2 ON
91	Chilldown Shutoff Pilot Valve ON	102	70 Lbs Ullage Engine Command No. 2 OFF
92	Chilldown Shutoff Pilot Valve OFF	55	Charge Ullage Jettison ON
82	Prevalve Close Command ON	88	Ullage Charging Command Reset
83	Prevalve Close Command OFF	57	Fire Ullage Jettison ON
24	Engine Pump Purge Control Valve Command ON	54	Charge Ullage Ignition ON
25	Engine Pump Purge Control Valve Command OFF	56	Fire Ullage Ignition ON
1	Start Tank Vent Valve Open ON	73	Firing Ullage Reset
		11	Fuel Injection Temp OK Bypass (Main Stage Enable)

10.3.3.2 (Continued)

Deletion of the chilldown inverters, prevalves, chilldown shutoff valves, EBW firing units, LOX depletion sensors, and logic modules will afford a sizable reduction in component test effort. These deletions will also simplify installations, electrical test requirements, and H&CO drawings. However, the engine sequence for J-2S is sufficiently different from the J-2 engine sequence that the engine test requirements and H&CO drawings must be completely revised for dry area and static test firing checkout. The all-systems test is also affected by the changes in engine sequence.

TABLE 10.3.3-III. SWITCH SELECTOR CHANNEL ASSIGNMENT ADDED

1. Emergency He Dump ON*	11. Mainstage Start ON
2. Emergency He Dump OFF*	12. Mainstage Start OFF
3. Mainstage Cutoff ON	13. Idle Mode Control Valve (Solenoid) Open*
4. Mainstage Cutoff OFF	14. Idle Mode Control Valve (Solenoid) Closed*
5. Mainstage OK Bypass ON	15. Mainstage Start Control Valve Open*
6. Mainstage OK Bypass OFF	16. Mainstage Start Control Valve Closed*
7. Pneumatic System Vent ON*	17. Mainstage Control Valve Open*
8. Pneumatic System Vent OFF*	18. Mainstage Control Valve Closed*
9. Propellant Dump Solenoid Valve Closed*	
10. Propellant Dump Solenoid Valve Open*	
*Required for In-Orbit Passivation	

TABLE 10.3.3-IV. ADDITIONAL SWITCH SELECTOR CHANNEL ASSIGNMENT FOR THE SYNCHRONOUS MISSION

Thermal Conditioning	
1. Electrical Equipment Heaters Enable	2. Electrical Equipment Heaters Disable
Telemetry	
3. Slow Record ON	8. Fast Record OFF
4. Slow Record OFF	9. Emergency Playback Reverse ON
5. Recorder Playback ON	10. Emergency Playback Reverse OFF
6. Recorder Playback OFF	
7. Fast Record ON	

TABLE 10.3.3-V. ELECTRICAL/ELECTRONIC COMPONENT DELETIONS FOR LOR AND SYNC MISSIONS

Item	Part Number
Purge and Drain	
Engine Pump Purge Module	1A58347
Engine Pump Purge Reg Backup Press Sw	1B52623
Conn P33 (403W8)	PTIH-8-3P
Conn P32 (403W8)	PTIH-8-4P
Start Tank	
Start Tank Vent Pilot Valve	1A58345
Conn P30 (403W8)	PTIH-8-3P
Chilldown	
Prevalve Actuation Cont Module	1B66692
Chilldown Shutoff Valve Pilot Valve	1B66692
Prevalve Pilot Valve	1B66692
Fuel Tank Prevalve	1A49968
Fuel Tank Chilldown Shutoff Valve	1A49965
LOX Chilldown Inverter	1A74039
Fuel Chilldown Inverter	1A74039
LOX Chilldown Pump	1A49423
Fuel Chilldown Pump	1A49421
LOX Chilldown Shutoff Valve	1A49965
LOX Prevalve Pos Ind Sw	1A49968
LH2 Tank Prevalve Ind Sw	1A49968
Fuel Tank Chilldown Shutoff Valve Ind Sw	1A49965
Conn P12 (403W2)	PTIH-8-3P
Conn P11 (403W2)	PTIH-8-3P
Conn P11 (404W7)	1A82714
Conn P12 (404W7)	1A82714
Conn P10 (EMC Testing)	10-435452-15P
Conn P1 (404W26)	10-435452-15S
Conn P9 (404W7) EMC Testing	10-435452-15P

TABLE 10.3.3-V. (Continued)

Item	Part Number
Chilldown (Continued)	
Conn P1 (404W27)	10-435452-15S
Conn P2 (404W26)	1A82714
Conn P2 (404W27)	1A82714
Conn P24 (404W7)	1A82714
Conn P26 (404W7)	1A82714
Conn P11 (404W7)	1A82714
Conn P12 (404W7)	1A82714
LOX Depletion Sensors	
(4) LOX Tank Sensor Assy	1A68710
(4) LOX Sensor Control	1A68710
Wire Harness (406W1) (Delete)	1B58968
Conn P1 (406W1)	1B37872-503
Conn P2 (406W1)	1B37872-503
Conn P7 (404W30)	1B37872
Conn P3 (404W30)	DS07-19-2S-9
Conn P4 (404W30)	DS07-19-2S-9
Conn P5 (404W30)	DS07-19-2S-9
Conn P11 (404W30)	DS07-19-2S-9
(3) Conn J1 (L.S.C.)	DM5600-19-2P
(3) Conn J2 (L.S.C.) (Used in EMC Testing)	PTIH-12-10P
(2) Feed Thru Disconnect (424A5J8 & J9)	1B37873
Conn P18 (404W7) (Used in EMC Testing)	S0286E-12-10S
Conn P19 (404W7)	S0286E-12-10S
Conn P20 (404W7)	S0286E-12-10S
Ullage Rocket Ignition	
(2) Panel 18C (404A47)	1B51299
(2) U.R. Ignition EBW FU	40M39515
(2) Conn P54 (Dummy) (404W7)	10-150921-123

TABLE 10.3.3-V. (Continued)

Item	Part Number
Ullage Rocket Ignition (Continued)	
(2) Conn P55 (Dummy) (404W7)	10-150921-123
(2) Conn P51 (404W7)	PT07CE-12-8P
(2) Conn P53 (404W7)	RB10-42612-3S
U.R. Ign Manifold CDF Block	1A84223
Conn J1 (404A47-2)	10-42212-3P
Conn J2 (404A47-2)	10-42212-3P
Pulse Sensor Mtg Brkt Assy (Flt) (404A47A4)	1B52640
Pulse Sensor Mtg Brkt Assy (NF) (404A47A4)	DSV-4B-780
(2) UR Ign EBW Pulse Sensor	40M02852
(2) Conn J1	10-42212-3P
(2) Conn J2	PT07E-12-8P
(2) Test Chamber (404A47A4A3)	1A58816
(2) Test Chamber (404A47A4A4)	1A58816
(2) Conn J1	PTIH-8-3P
(2) Conn J2	PT07E-12-8P
Wire Harness (Int to 404A47) (404A47A4W1)	1B50054
Wire Harness (Int to 404A47) (404A47A4W2)	1B50242
Wire Harness (Int to 404A47) (404A47A4W3)	1B58950
Wire Harness (Int to 404A47) (404A47A4W4)	1B58950
(2) Conn	PT06CE-8-3S
(2) Conn	PT00CE-12-8P
(2) Conn	PT07CE-12-8P
(2) Conn	PT06CE-12-8S
(2) Diode Suppression Mod (404A47A4W3 and W4)	1B58950
Ullage Rocket Jettison	
Panel Pos 18A	1B51297
(2) Dummy Recep	10-150921-123
(2) U.R. Jet EBW FU	40M39515
(2) Conn (P51) (404W7)	PT07H-12-8P

TABLE 10.3.3-V. (Continued)

Item	Part Number
Ullage Rocket Jettison (Continued)	
(2) Conn (P52)	10-42612-3S
U.R. Jet CDF Block	1A84223
(2) Conn	10-42212-3P
EBW C/O Bracket (Flight)	1A97791
EBW C/O Bracket (NF)	DSV-4B-778
(2) U.R. Jet EBW Pulse Sensors	40M02852B
(2) Conn J1	10-42212-3P
(2) Conn J2	PT07E-12-8P
(2) Test Chamber (1 & 2)	1A58816
(4) Conn J1	PTIH-8-3P
Wire Harness (404A75A10W1	1B40164
Wire Harness (404A75A10W2)	1B40165
Wire Harness (404A75A10W3)	1B58950
Wire Harness (404A75A10W4)	1B58950
(2) Conn J1	PT02CE-12-8P
(2) Conn J2	PT06CE-8-3S
(2) Conn J1	PT00CE-12-8P
(2) Conn J2	PT06CE-12-8S
(2) Diode Supp Mod	1B41560
Ullage Engine Module (APS)	
70 Pound Ullage Engine Module (414A7)	NAA 15-210001
(4) Conn	PTIH-8-4P
(2) Conn (414W1) P21 MS3116E8-4S	MS3116E8-4S
(2) Conn (414W1) P22 MS3116E8-4S	MS3116E8-4S
(2) Circuit Supp. Mod	1B54522
Wire Harness (Mod) (414W1)	1B67267
70 Pound Ullage Engine Module (415A7)	NAA 15-210001
Wire Harness (Modify) (415W1) (Same as 414W1 APS W1)	1B67267

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10.3.3.3 Propellant Utilization System

Since the response of the PU System depends on the mainstage performance of the engine, a comparison was made between the J-2 and J-2S engine performance characteristics. A comparison between the J-2 engine PU valve characteristics presently being used for closed loop PU System operation and the estimated J-2S engine PU valve characteristics in R-7211 indicates that the nonlinear shape of PU valve characteristics will not change significantly. The PU valve LOX corrective flow gain for the J-2S engine is near 83.5 pounds LOX per second per unit EMR change at a bridge gain ratio of 5.0:1.0. This gain is based on the J-2S engine weight flow versus engine mixture ratio characteristics presented in the J-2S Interface Criteria Document of May 7, 1968. The gain is also shown to be independent of the calibrated thrust level of the J-2S engine. The corresponding LOX corrective flow gains for the J-2 engines presently being used for closed loop PU System operation are within one db of the J-2S value. This small gain difference indicates that although the J-2S engine thrust and flow characteristics have been increased over the characteristics of the J-2 engine, the LOX corrective flow gain does not display any significant change. Therefore, a ±4.5 db adjustment presently available in the PU System motor loop feedback network provides adequate PU System gain adjustment capability without PU System modifications.

The transfer of the LOX cutoff responsibility to the J-2S engine results in a change in the amount of trapped and unavailable LOX which must be used in revising the end point of the LOX calibration. No other changes are required.

Mission Requirements/PU System

Closed loop PU System operation is recommended for the S-IVB/J-2S Stage during the LOR and synchronous missions. No PU system modifications are required to utilize the J-2S engine. However, the open loop low EMR command for engine restart will not be required. Recommended PU system operation is based on Boeing design trajectories and status reports.

The recommended modes of PU system operation for the LOR mission are presented below.

Burn Period	Mode of Operation	Commanded EMR
First	Open Loop	5.0/1.0
Second	Closed Loop	5.0/1.0

The open loop mode of operation during first burn eliminates the need of an LH_2 boiloff bias and, therefore, the sequencing associated with the bias.

10.3.3.3 (Continued)

The recommended modes of PU system operation for the synchronous mission are presented below.

Burn Period	Mode of Operation	Commanded EMR
First	Open Loop	5.5/1.0
Second	Closed Loop	5.35/1.0
Third	Open Loop	5.0/1.0

The open loop mode of operation during first burn eliminates the need of an LH_2 first coast. The closed loop EMR for second burn reflects the results of past synchronous mission studies. Open loop operation during third burn eliminates the need of an LH_2 bias for second coast and eliminates thrust variations due to PU valve motion toward the end of flight. The elimination of LH_2 boiloff biases also eliminates the sequencing associated with the biases.

The capability of operating in the modes previously discussed were incorporated into the propellant utilization electronics at 504 and subs. The basic approach is shown in Figure 10.3.3-10.

10.3.3.4 Instrumentation System

Saturn V LOR Stage

The measurement changes for the operational J-2S LOR missions are itemized in Table 10.3.3-VI. There are a total of 54 deletions and 46 additions to the S-IVB. The deletions occur primarily in three areas; (1) ullage rocket and jettison, (2) J-2 engine peculiar measurements, and (3) chilldown system. The additions are primarily on the J-2S engine. A comparison of the S-IVB/J-2 and S-IVB/J-2S indicates that 24 measurements at the engine interface remain the same. Functionally, 9 other engine measurements remain the same; however, the measurement range is different from the J-2. Of the 54 measurement deletions, 17 are at the engine interface with the remainder in the vehicle. Nine additional measurements will be monitored from the stage. One of these measurements, C8, was flown on earlier vehicles but is not presently telemetered. The remainder are engine associated measurements such as commands and talkbacks which are picked up in distributors. The additional telemetry requirements at the engine interface fall into two categories as follows: (1) those that are presently wired to the interface and (2) those which are not presently wired to the interface, most of which have been requested by MDAC. It is assumed that transducers and wiring to implement these requested measurements can be installed on the engine by Rocketdyne.

Since the instrumentation net change is only 8 deleted measurements, it will not be necessary to change the present multiplexing and RF transmission system provided the sampling rates remain approximately the same as they are presently. However, as a result of these changes, it will be necessary to change the signal

P.U. OPERATION	MR	SWITCH POSITION	
		OPEN	CLOSED
CLOSED LOOP	5.35	2	
OPEN LOOP	5.0		2
OPEN LOOP	5.5	1, 52	51, 53
OPEN LOOP	4.5	1, 42	41, 43

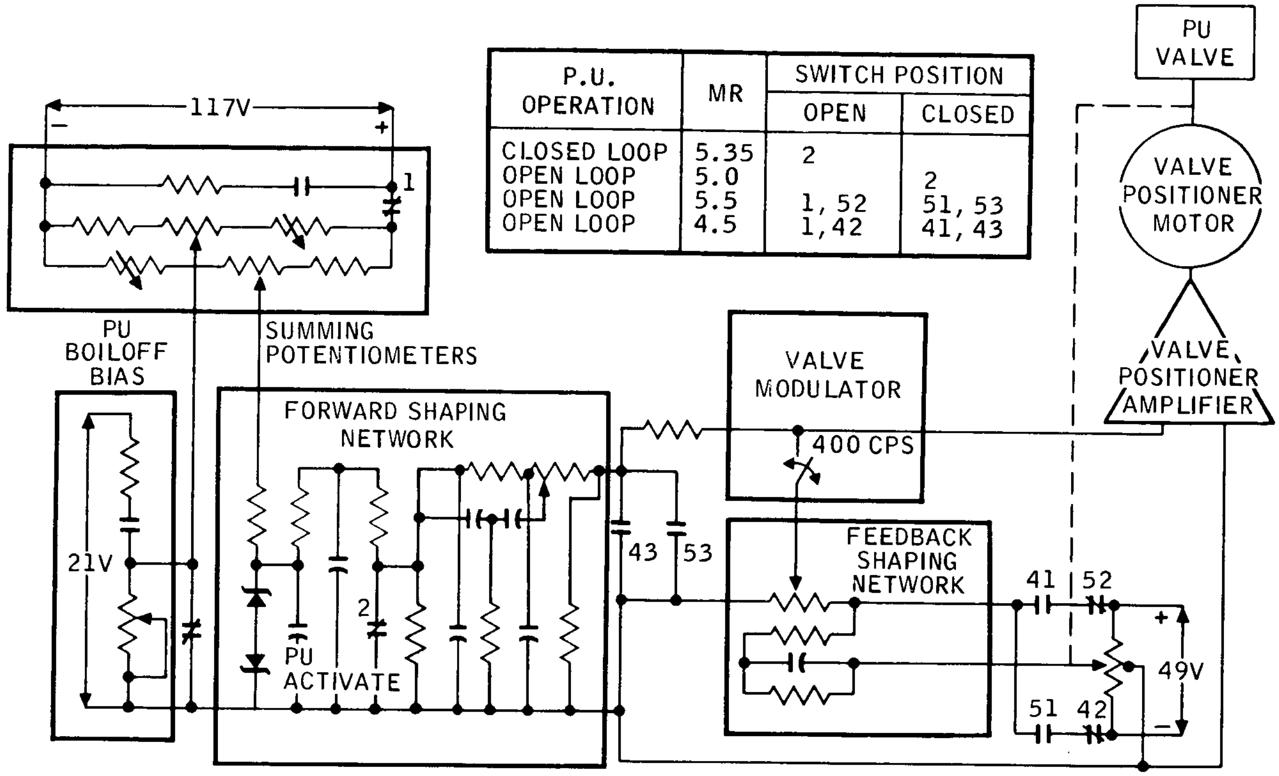


FIGURE 10.3.3-10. PROPELLANT UTILIZATION SYSTEM SCHEMATIC

10.3.3.4 (Continued)

conditioning rack module complement, revise cable drawings, cable network drawings, schematics, Instrumentation Program and Components List, and installation drawings.

Engine instrumentation power requirements during the S-IVB flight and coast period are reduced from 240 watts for the J-2 to 18 watts for the J-2S.

Virtually all the additional measurement points are on the engine, therefore, transducer component testing will be small. However, new amplifiers, temperature bridges, and other networks for the additional measurements must be designed and tested if present components are not acceptable.

Instrumentation test requirements and H&CO drawings must be revised for dry checkout areas and for the Beta Test Stands at Sacramento.

The transducer and signal conditioning rack module additons and deletions as well as wire harness changes for the LOR mission are listed in Table 10.3.3-VII.

Saturn V Synchronous Stage

The measurement changes for an operational J-2S synchronous mission are itemized in Table 10.3.3-VIII.

TABLE 10.3.3-VI. INSTRUMENTATION CHANGES FOR THE J-2S STUDY

Measurement No. and Title		LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
C1	Temp, Fuel Turbine Inlet	N. C.	N. C.			P108-c, d, k,	P108-c, d, k	
C2	Temp, Oxidizer Turbine Inlet	N. C.	N. C.			P108-h-j, r	P108-h, j, r	
C6	Temp, Start Tank Gas	Delete	Delete			P108-G, I, J		
C7	Temp, Helium Tank Gas	Delete	Delete	-350°F/+100°F		P108-M, T, U		
C7	Temp, Helium Tank Gas	Add	Add		-300°F/+100°F		P108-G, I, J	
C11	Temp, Electronic Control Assy	N. C.	N. C.			P54-p, u, v	P54-p, u, v	
C133	Temp, Oxidizer Pump Discharge	N. C.	N. C.			P108-W, X, b	P108-W, X, b	
C134	Temp, Fuel Pump Discharge	N. C.	N. C.			P108-Z, a, e	P108-Z, a, e	
C159	Temp, LOX Circ Return Line Tank Inlet	Delete	Delete					
C161	Temp, LH2 Circ Return Line Tank Inlet	Delete	Delete					
C197	Temp, Primary Instr Package	N. C.	N. C.			P106-K, S, a	P106-K, S, a	

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
C199 Temp, Thrust Chamber Jacket	Delete	Delete			P108-s, v, x		
C200 Temp, Main Fuel Injector No. 1	N. C.	N. C.			P108-t, w, y	P108-t, w, y	
C215 Temp, Oxidizer Turbine Outlet	N. C.	N. C.			P108-f, g, m	P108-f, g, m	
TBD Temp, Fuel Bypass Manifold	Add	Add		-425°F to 240°F			Signal not at Rkd Interface.
TBD Temp, SPTS Case No. 2	Add	Add		-160°F to 1540°F			Signal not at Rkd Interface.
TBD Temp, Thrust Chamber Skin	Add	Add		-425°F to 240°F			Signal not at Rkd Interface.

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
TBD Temp, Idle Mode Injector LOX Line	Add	Add		-300° F to 240° F			Signal not at Rkd Interface.
TBD Temp, ASI Idle Mode LOX	Add	Add		-300°F to 240°F			Signal not at Rkd Interface.
TBD Temp, ASI Idle Mode LH2	Add	Add		-425°F to 240°F			Signal not at Rkd Interface.
TBD Temp, LH2 Injector	Add	Add		-425°F to 240°F			Signal not at Rkd Interface.
TBD Temp, LH2 F/C Injector	Add	Add		-425°F to 240°F			Signal not at Rkd Interface.
TBD Temp, LOX Pump Bearing Coolant	Add	Add		-300°F to 240°F			Signal not at Rkd Interface.
TBD Temp, LH2 Pump Bearing Sump	Add	Add		-425°F to 240°F			Signal not at Rkd Interface.
TBD Temp, LOX Engine Interface (Wide Range)	Add	Add		-300°F to 240°F			Signal not at Rkd Interface.
TBD Temp, LH2 Engine Interface (Wide Range)	Add	Add		-425°F to 240°F			Signal not at Rkd Interface.

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
C3 Temp, Fuel Pump Inlet	N. C.	N. C.					Vehicle Measurement
C4 Temp, Oxidizer Pump Inlet	N. C.	N. C.					Vehicle Measurement
C8 Temp, Heat Exchanger Helium Inlet	Add	Add					Vehicle Measurement not telemetered on 509.
C9 Temp, Heat Exchanger Helium Outlet	N. C.	N. C.					Vehicle Measurement
C10 Temp, Engine Area Ambient	N. C.	N. C.					Vehicle Measurement
D1 Press, Thrust Chamber			0-1000 psi	0-1500 psi	P106-p	P106-p	
D8 Press Fuel			0-1500 psi	0-2500 psi	P106-f	P106-f	
D9 Press, Oxidizer Pump Discharge	Retain Range Change	Retain Range Change	0-2000 psi	0-2500 psi	P106-e	P106-e	
D10 Press, Gas Generator Pressure	Delete	Delete	0-1000 psi		P106-h		
D242 Press, Helium Tank No. 1	Changed Connector & Range	Changed Connector & Range	0-5000 psi	0-3500 psi	P154-g	P106-h	

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
D17 Press, Engine Start Tank	Delete	Delete			P106-b		
TBD Press, Idle Mode Chamber Press	Add	Add		0-50 psi		P106-b	
D19 Press, Helium Tank No. 2	N. C.	N. C.	0-3500 psi	0-3500 psi	P106-T	P106-T	
D50 Press, Engine Pump Purge Reg	Delete	Delete					
D103 Press, Helium Press to LOX Motor Ctr	Delete	Delete					
D218 Press, Diff LH2 Chilldown Pump	Delete	Delete					
D219 Press, Diff LOX Chilldown Pump	Delete	Delete					
D220 Press, Ullage Control Chamber 1-4	Delete	Delete					
D221 Press, Ullage Control Chamber 2-4	Delete	Delete					
D241 Press, GH2 Start Bottle Backup Measurement	Delete	Delete					

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
TBD Press, ASI Combustion Chamber	Add	Add		0-50 psia			Signal not at Rkd Interface.
D2 Press, Fuel Pump Inlet	N. C.	N. C.					Vehicle Measurement
D3 Press, Oxidizer Pump Inlet	N. C.	N. C.					Vehicle Measurement
D4 Press, Main Fuel Injection	Add	Add		0-2500 psia			Signal not at Rkd Interface.
D5 Press, Main Oxidizer Injection	Add	Add		0-2500 psia			Signal not at Rkd Interface.
D7 Press, Oxidizer Turbine Inlet	Add	Add		0-200 psia			Hardwire only at Rkd Interface.
D18 Press, Engine Regulator Outlet	Add	Add		0-750 psia			Hardwire only at Rkd Interface.
TBD Press, Fuel Turbine Inlet	Add	Add		0-1500 psia			Hardwire only at Rkd Interface.

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
TBD Press, ASI Idle Mode LOX	Add	Add		0-50 psia			Signal not at Rkd Interface.
TBD Press, ASI Idle Mode LH2	Add	Add		0-50 psia			Signal not at Rkd Interface.
TBD Press, LH2 Injection	Add	Add		0-50 psia			Signal not at Rkd Interface.
TBD Press, LH2 F/C Injector	Add	Add		0-50 psia			Signal not at Rkd Interface.
TBD Press, Idle Injector LOX Line	Add	Add		0-50 psia			Signal not at Rkd Interface.
D104 Press, LH2 Pressure Module Inlet	Retain Range Change	Retain Range Change		TBD			Vehicle Measurement
F1 Flowmeter, Oxidizer	Retain Range Change	Retain Range Change	0-3000 gpm	$\frac{1500/3600}{30/1500}$	P109-V, U, T, S, R	P109-V, U, T, S, R	
F2 Flowmeter, Fuel	Range Change	Range Change	0-9000 gpm	$\frac{7000/11000}{400/7000}$	P109-P, N, K, J	P109-P, N, K, J	
F4 Flow Rate, Oxidizer Circulation Pump	Delete	Delete					
F5 Flow Rate, Fuel Circulation Pump	Delete	Delete					
G3 Position, Main Oxidizer Valve Position	N. C.	N. C.			P107-$\underline{h}$	P107-$\underline{h}$	

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title		LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
G4	Position, Main Fuel Valve Position	N. C.	N. C.			P107-r	P107-r	
G5	Position, Gas Generator Valve Position	Delete	Delete			P107-p		
TBD	Fuel, Bypass Valve Position	Add	Add				P107-p	
G8	Position, Oxidizer Turbine Bypass Valve	Delete	Delete			P107-X		
TBD	Position, Idle Mode Valve Position Pot	Add	Add				P107-X	
G9	Position, Start Tank Discharge Valve	Delete	Delete			P107-j		
TBD	Position, Hot Gas Tapoff Valve	Add	Add				P107-j	
G10	Position, PU Valve	N. C.	N. C.			P107-e	P107-e	
G1	Position, Actuator Piston Pitch	N. C.	N. C.					Vehicle Measurement
G2	Position, Acutator Piston Yaw	N. C.	N. C.					Vehicle Measurement

TABLE 10.3.3-VI. (Continued)

Measurement and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
K5 Event, Mainstage Control ON (Sol)	N. C.	N. C.			P54-P	P54-P	
K6 Event, Ign Phase Cont ON	Delete	Delete			P54-J		
TBD Event, Idle Mode Cont ON (Sol)	Add	Add				P54-J	
K7 Event, Helium Control ON (Sol)	N. C.	N. C.			P54-E	P54-E	
K10 Event, ASI Spark ON	N. C.	N. C.			P54-h	P54-h	
K11 Event, GG Spark ON	Delete	Delete			P54-j		
TBD Event, SPTS Initiated	Add	Add				P54-j	
K12 Event, Engine Ready Signal	N. C.	N. C.			P51-a	P51-a	
K13 Event, Engine Cutoff Lockin ON	N. C.	N. C.			P54-V	P54-V	
K14 Event, Mainstage OK No. 1	N. C.	N. C.			P51-e	P51-e	
K20 Event, ASI Oxidizer Valve Position Open	Delete	Delete			P107-W		

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
K96 Event, Start Tank Discharge Valve Cont ON	Delete	Delete			P54-K		
TBD Event, Mainstage Start Cont ON (Sol)	Add	Add				P54-K	
K105 Event, Pump Purge Reg Backup De-en	Delete	Delete					
K109 Event, Oxidizer Prevalve Open	Delete	Delete					
K110 Event, Oxidizer Prevalve Closed	Delete	Delete					
K111 Event, Fuel Prevalve Open	Delete	Delete					
K112 Event, Fuel Prevalve Closed	Delete	Delete					
K117 Event, GG Valve Position Open	Delete	Delete			P107-g		
TBD Event, Fuel Bypass Valve Open	Add	Add				P107-g	
K118 Event, Main Fuel Valve Open	N. C.	N. C.			P107-k	P107-k	
K120 Event, Main Oxidizer Valve Open	N. C.	N. C.			P107-a	P107-a	

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
K122 Event, Start Tank Discharge Valve Open	Delete	Delete			P107-c		
TBD Event, Hot Gas Tapoff Valve Open Switch	Add	Add				P107-c	
K124 Event, Oxidizer Turb Bypass Valve Open	Delete	Delete			P54-b		
K125 Event, Oxidizer Turb Bypass Valve Closed	Delete	Delete			P54-a		
TBD Event, Mainstage Cutoff Lockin	Add	Add				P54-a	
K126 Event, Oxidizer Bleed Valve Closed	Delete	Delete			P107-Z		
TBD Event, Idle Mode Valve Open Switch	Add	Add				P107-Z	
K127 Event, Fuel Bleed Valve Closed	Delete	Delete			P107-T		
TBD Event, Idle Mode Valve Closed Switch	Add	Add				P107-T	
K131 Event, LOX Chill-down Purge Switch De-en	Delete	Delete					

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
K136 Event, Fuel SOV Chill System Closed	Delete	Delete					
K137 Event, Fuel SOV Chill System Open	Delete	Delete					
K138 Event, Oxidizer SOV Chill System Open	Delete	Delete					
K139 Event, Oxidizer SOV Chill System Closed	Delete	Delete					
K149 Event, Ullage Jettison 1 P/S	Delete	Delete					
K150 Event, Ullage Jettison 2 P/S	Delete	Delete					
K157 Event, Mainstage OK No. 2	N. C.	N. C.			P51-b	P51-b	Signal picked up in the 28V aft power distributor.
K158 Event, Mainstage OK Press Switch No. 1 Depress	N. C.	N. C.			P54-U	P54-U	
K159 Event, Mainstage OK Press Switch No. 2 Depress	N. C.	N. C.			P54-T	P54-T	
K176 Event, Ullage Rkt Ign P/S 1 Ind	Delete	Delete					

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
K177 Event, Ullage Rkt Ign P/S 2 Ind	Delete	Delete					
K123 Event, Start Tank Discharge Valve Closed	Delete	Delete			P107-d		Not telemetered on 509.
TBD Event, Hot Gas Tapoff Valve Closed Switch	Add	Add				P107-d	Signal picked up at the 56V aft power distributor.
K119 Event, Main Fuel Valve Closed	N. C.	N. C.			P107-m	P107-m	Not telemetered on 509. Signal picked up at the 56V aft power distributor.
K116 Event, Gas Gen Valve Position Closed	Delete	Delete			P107-n		Not telemetered on 509.
TBD Event, Fuel Bypass Valve Closed Switch	Add	Add				P107-n	Signal picked up at the 56V aft power distributor.
TBD Event, SPTS Armed	Add	Add				P54-c	
TBD Event, SPTS No. 1 Ready	Add	Add				P54-k	
K121 Event, Main Oxidizer Valve Closed	N. C.	N. C.			P107-b	P107-b	Not telemetered on 509. Signal picked up at the 56V aft power distributor.

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
TBD Event, SPTS No. 2 Ready	Add	Add				P53-B	
TBD Event, SPTS No. 3 Ready		Add				P53-C	
TBD Event, Mainstage Start Command	Add	Add				P51-X	Signal picked up from stage.
TBD Event, Mainstage cutoff ON cmd	Add	Add				P51-g	Signal picked up from stage.
K21 Event, Engine Start Command	N. C.	N. C.				P51-f	Signal picked up from stage.
K140 Event, Engine Cutoff ON	N. C.	N. C.				P51-j	Signal picked up from stage.
TBD Event, Helium Vent Control Cmd	Add	Add					Signal not at Rkd Interface.
TBD Event, Eng Pneu Sys Vent Cmd	Add	Add					Signal not at Rkd Interface.
TBD Event, Propellant Dump Solenoid Energ.	Add	Add					Signal not at Rkd Interface.
K8 Event, Ignition Detected	Delete	Delete					

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
M6 Voltage, Control Bus	N. C.	N. C.			P54-s	P54-s	
M7 Voltage, Ignition Bus	N. C.	N. C.			P54-n	P54-n	
M26 Voltage, Phase A-B Fuel Chilldown Inverter	Delete	Delete					
M27 Voltage, Phase A-B LOX Chilldown Inverter	Delete	Delete					
M28 Freq, Fuel Chill-down Inverter	Delete	Delete					
M29 Freq, LOX Chill-down Inverter	Delete	Delete					
M40 Voltage, Phase A-C LOX Chilldown Inverter	Delete	Delete					
M41 Voltage, Phase A-C Fuel Chilldown Inverter	Delete	Delete					
M60 Voltage, PU Valve Control	N. C.	N. C.					Monitored from PU Electronics Assembly.
M64 Voltage, F/U 1 EBW Ullage Rocket Ignition	Delete	Delete					

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
M65 Voltage, F/U 2 EBW Ullage Rocket Ignition	Delete	Delete					
M66 Voltage, F/U 1 EBW Ullage Rocket Jettison	Delete	Delete					
M67 Voltage, F/U 2 EBW Ullage Rocket Jettison	Delete	Delete					
T1 Speed, Oxidizer Pump	N. C.	N. C.	0-12000 rpm	0-12000 rpm	P109-H, G, F, E	P109-H, G, F, E	

TABLE 10.3.3-VI. (Continued)

Measurement No. and Title	LOR	SYNC	J-2 Measurement Range	J-2S Measurement Range	J-2 Interface Connector	J-2S Interface Connector	Remarks
T2 Speed, Fuel Pump	Retain Range Change	Retain Range Change	0-30000 rpm	0-33000 rpm	P109-D, C, B, A	P109-D, C, B, A	

TABLE 10.3.3-VII. INSTRUMENTATION CHANGES FOR THE J-2S OPERATIONAL LOR

Component	Part Number	Number Added	Number Deleted	Modify
Temp Bridge	1A98088		2	
Temp Bridge	1A82274		2	
Temp Bridge	TBD	17		
DC Amplifier	1A82395		2	
DC Amplifier	TBD	3		
Temp Sensor	NA5-27215		1	
Temp Sensor	1A67863		2	
Temp Sensor	NA5-27321		1	
Temp Sensor	TBD	7		
Press Transducer	NA5-27412		3	
Press Transducer	1B43320		1	
Press Transducer	1B53574		1	
Press Transducer	1B53573		1	
Press Transducer	1B88035		2	
Press Transducer	1B43324		1	
Press Transducer	TBD	3		
Position Transducer	NA5-27307		1	
Position Transducer	NA5-27285		1	
Position Transducer	NA5-27306		1	
Position Transducer	TBD	1		
Level Sensor	1A68710		3	
Event Network	1B44241		4	
Event Network	TBD	2		

TABLE 10.3.3-VII. (Continued)

Component	Part Number	Number Added	Number Deleted	Modify
Flow Rate	1A89104		2	
Freq to DC Converter	1B37523-501		2	
Wire Harness 404W203	1B74670			10% change to noted drawing.
Wire Harness 404W208	1B67209			5% change to noted drawing.
Wire Harness 403W200	1B67208			5% change to noted drawing.

TABLE 10.3.3-VIII. ADDITIONAL R&D INSTRUMENTATION FOR THE SYNCHRONOUS MISSION

Measurements	Fwd	Aft	Minimum Sample Rate	Function
LOX Tank (Internal)				
Twenty-one (21) Temperatures		21	24 SPM	Evaluation of LOX sloshing and heat input (for boiloff calculations)
LOX Tank (External)				
Six (6) Temperatures		6	1 SPM	Evaluation and analysis of heat input into LOX tank (for boiloff calculations) entrainment, and stratification
Auxiliary Propulsion System				
Thirteen (13) Temperatures		13	1 SPM	Thermo evaluation and verification of passive and active thermo control methods.
O_2H_2 Burner				
Three (3) Temperatures		3	4 SPS	Determination of flowrates of propellants (oxygen and hydrogen) and pressurant (Helium)
Six (6) Pressures		6	4 SPS	

TABLE 10.3.3-VIII. (Continued)

Measurements	Fwd	Aft	Minimum Sample Rate	Function
Non Propulsive Vent				
Two (2) Temperatures	2		4 SPS	Determination of vent flow-rates required to determine LH2 mass characteristics such as boiloff
LOX Tank Vent Inlet				
One (1) Temperature		1	4 SPS	Determination of LOX tank pressurant flowrate and venting characteristics
Cold Helium System				
Two (2) Temperatures	2		1 SPM	LH2 mass calculations due to uncertainty of LH2 level and LOX ullage volume
LH2 Tank (Internal)				
Thirty-five (35) Temperatures	35		24 SPM	Evaluation and analysis of fuel sloshing, vapor entrainment, heat input (for boiloff calculations) and stratification
LH2 Tank (External)				
Nine (9) Temperatures	9		1 SPM	Evaluation and analysis of fuel heat input (for boiloff calculations)
Aft Skirt (External)				
Four (4) Temperatures		4	1 SPM	Evaluation and analysis of heat input into LOX tank (for boiloff calculations)
Continuous Vent				
One (1) Temperature	1		4 SPS	Determination of vent gas energy increase in vent line

TABLE 10.3.3-VIII. (Continued)

Measurements	Fwd	Aft	Minimum Sample Rate	Function
<u>Power System</u>				
One (1) Current	1		4 SPS	Prelaunch redline requirement, determination of battery loads for performance evaluation.
One (1) Voltage	1		4 SPS	
Two (2) Temperatures	2		5 SPM	
<u>PCM RF System</u>				
Two (2) Forward Power	2		5 SPM	Prelaunch checkout performance evaluation.
Two (2) Reflected Power	2		5 SPM	
Deleted:				
One (1) Forward Power	-1		12 SPS	
One (1) Reflected Power	-1		12 SPS	
<u>Thermo Conditioning</u>				
Ten (10) Temperatures		10	1 SPM	Thermal evaluation and verification of thermal control methods.
Total	55	66		

10.3.3.4 (Continued)

A comparison of the operational telemetry lists for the S-IVB/J-2 and S-IVB/J-2S stages indicates the S-IVB/J-2S synchronous stage will require 47 additional measurements, 54 deletions and range changes for 9 measurements. During the R&D phase of the synchronous program, it is anticipated that an additional 166 measurements will be required, or a net change of 112 measurements. The deletions fall into the same category as for the LOR mission. The operational additions all fall into the same category as for the LOR mission. The R&D measurements are a general augmentation of the engine, APS and tankage instrumentation. The present instrumentation system cannot accommodate this many new measurements without additional equipment. A discussion of these changes is found in the following paragraphs.

10.3.3.4 (Continued)

Instrumentation System Evaluation for Synchronous Orbit

The present instrumentation system must be augmented to telemeter the additional synchronous orbit measurements. The ground rules for the mission are as follows:

1. Mission control parameters will not be changed in sample rate, time slot, or cross strapping.

2. The measurements listed in Table 10.3.3-VIII were used in sizing the system.

3. Continuous data coverage is required, which requires a tape recorder.

The recommended system is outlined below and shown in Figure 10.3.3-11.

- Add 2 Model 270 multiplexers
- Add 2 low level remote analog submultiplexers
- Re-program the Model 301 PCM/DDAS assembly for time sharing 3 Model 270 multiplexers in the present CP multiplexer time slot.

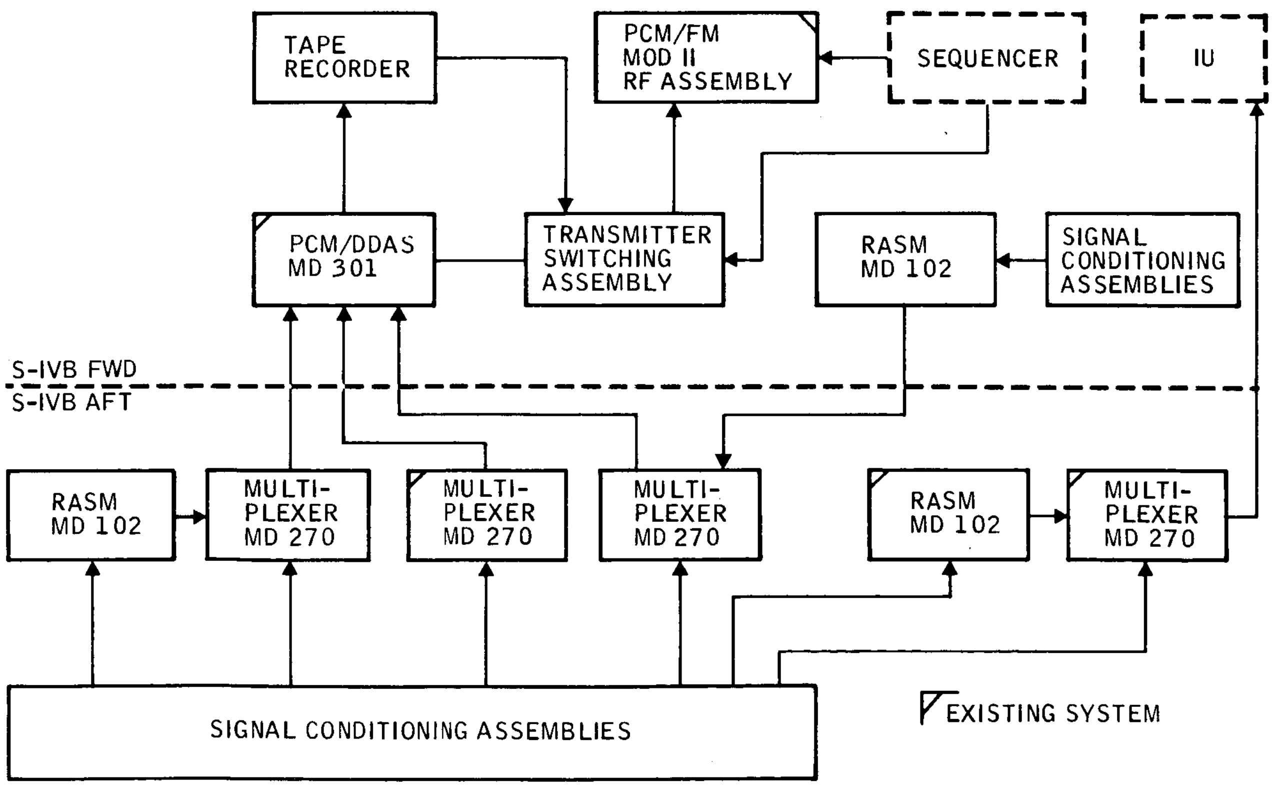


FIGURE 10.3.3-11. INSTRUMENTATION SYSTEM

10.3.3.4 (Continued)

All present measurements on the CP multiplexer will be cross-strapped to the same channels on the 2 added Model 270 multiplexers. This maintains a sample rate of 120 or 12 sps for these measurements. Prime channels 6, 7, 8, and 15 are opened up for remote analog multiplexer inputs. As a sample rate of 4 sps will satisfy the requirements of the synchronous orbit measurement, provisions are available, on each of the three multiplexers, for 4 remote submultiplexer inputs (40 low-level measurements) and 14 high-level (0-5V) inputs to be measured at 4 sps. Considering all three multiplexers, this provides for 120 temperature measurements, via remote analog multiplexing and 42 other measurements conditioned and inserted in the Model 270 multiplexer for measurement at a 4 sps rate. The complete evaluation may be found in the synchronous orbit study.

PCM RF System Evaluation

The ground rules are as follows:

1. The IU PCM/RF system will be changed to allow receiving of data throughout mission.

2. A 10 db system improvement is required to insure reception of data through mission.

3. The vehicle will roll at the rate of one revolution per hour.

The vehicle RF and/or TM systems must be modified if they are to be used to facilitate data recovery during the final portion of the transfer orbit and the final burn of the S-IVB. Calculations based on the current 18 watt configuration of the Saturn V and utilizing the TLM 18 tracking net indicate that threshold will occur at a nominal altitude of 5,000 miles. System improvement versus probability of data recovery by a single ground station for 20,000 miles altitude and arbitrary vehicle attitude is plotted in Figure 10.3.3-12.

Since the vehicle is experiencing a roll during the transfer orbit maneuver and its axis is perpendicular to the sun, the ground station may fall anywhere in the complete sphere of the radiation pattern of the vehicle. This makes the probability curve directly related to the antenna gain pattern of the vehicle. Further, since at 20,000 miles distance the earth subtends an angle of 28° with respect to the vehicle, the entire earth (and therefore all antennas on earth) could fall within a single null of the pattern. Therefore, it must be assumed that to completely preclude the loss of data, the system must be improved sufficiently to assure ground station lock in a worst-case condition. The worst nulls in the vehicle pattern go to -18 dbm with respect to an isotrophic radiator.

The curve in Figure 10.3.3-12 is based on contour plots of antenna patterns. Characteristics of the antenna patterns are as follows:

80 percent of sphere of radiation above	-6 db
90 percent of sphere of radiation above	-9 db
95 percent of sphere of radiation above	-12 db

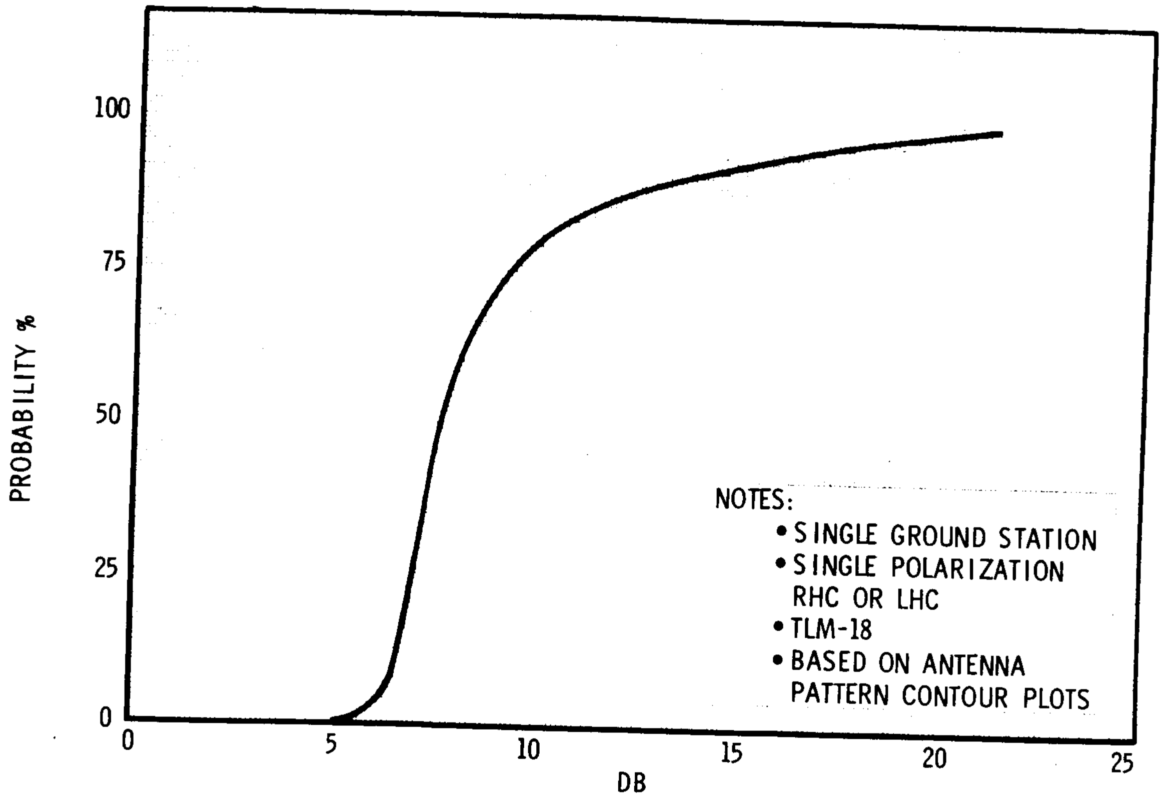


FIGURE 10.3.3-12. PROBABILITY OF DATA RECEIPT/PCM-RF SYSTEM (SYNC MISSION)

10.3.3.4 (Continued)

97.5 percent of sphere of radiation above	-18 db
100 percent of sphere of radiation above	-21 db

Using the formula: $N_n = 1 - (1 - N_1)^n$

Where: n = Number of ground receivers

N_n = Probability of receiving data with n ground receivers

N_1 = Percent of sphere of radiation above a certain db level

With the existing system characteristics shown in Table 10.3.3-IX, there is an 80 percent probability of receiving data at a single TLM-18 ground receiver. With two ground receivers there is a 96 percent probability of receiving data. With three ground receivers the probability rises to 99.2 percent.

TABLE 10.3.3-IX. EXISTING AND IMPROVED PCM RF MARGIN COMPARISONS

	Nom.	Best	Worst
1. (18W, TLM-18)			
Veh Trans Pwr	+ 42.5	+ 42.5	+ 42.5
Veh Ant. Gain	- 6.5	0	- 12
Veh RF Syst Loss	- 4	- 4	- 4
Space Attenuation	-170	-170	-170
Grnd Sta Ant. Gain	+ 28	+ 28	+ 28
Grnd Sta RF Loss	- 0.5	0	- 2
Exp Revr Sig Pwr	-110.5 dbm	-103.5	-117.5
Req Sig Pwr	-100 dbm	-100 dbm	-100 dbm
Margin	- 10.5	- 3.5	- 17.5
2. (Two 40W, TLM-18)			
Veh Trans Pwr	+ 46	+ 46	+ 46
Veh Ant. Gain	- 6.5	0	- 12
Veh RF Syst Loss	0	0	0
Space Attenuation	-170	-170	-170
Grnd Sta Ant. Gain	+ 28	+ 28	+ 28
Grnd Sta RF Loss	- 0.5	0	- 2
Exp Revr Sig Pwr	110.5 dbm	-103.5	-117.5
Req Sig Pwr	-100 dbm	-100	-100
Margin	0	+ 7	- 7

10.3.3.4 (Continued)

Six systems were analyzed in the synchronous orbit study. The recommended system adds two 1A77080 forty watt amplifiers, one at each antenna, using the Mod II RF assembly as drivers. This would increase the antenna power from a nominal value of 4.4 watts (22 watts nominal transmitter output through a nominal 7 db insertion loss) to 40 watts for an improvement of approximately 10 db. This system is shown in Figure 10.3.3-13. The major considerations for this system is listed in Table 10.3.3-X. The complete evaluation may be found in the synchronous orbit study.

A list of the hardware required for the synchronous mission is given in Table 10.3.3-XI.

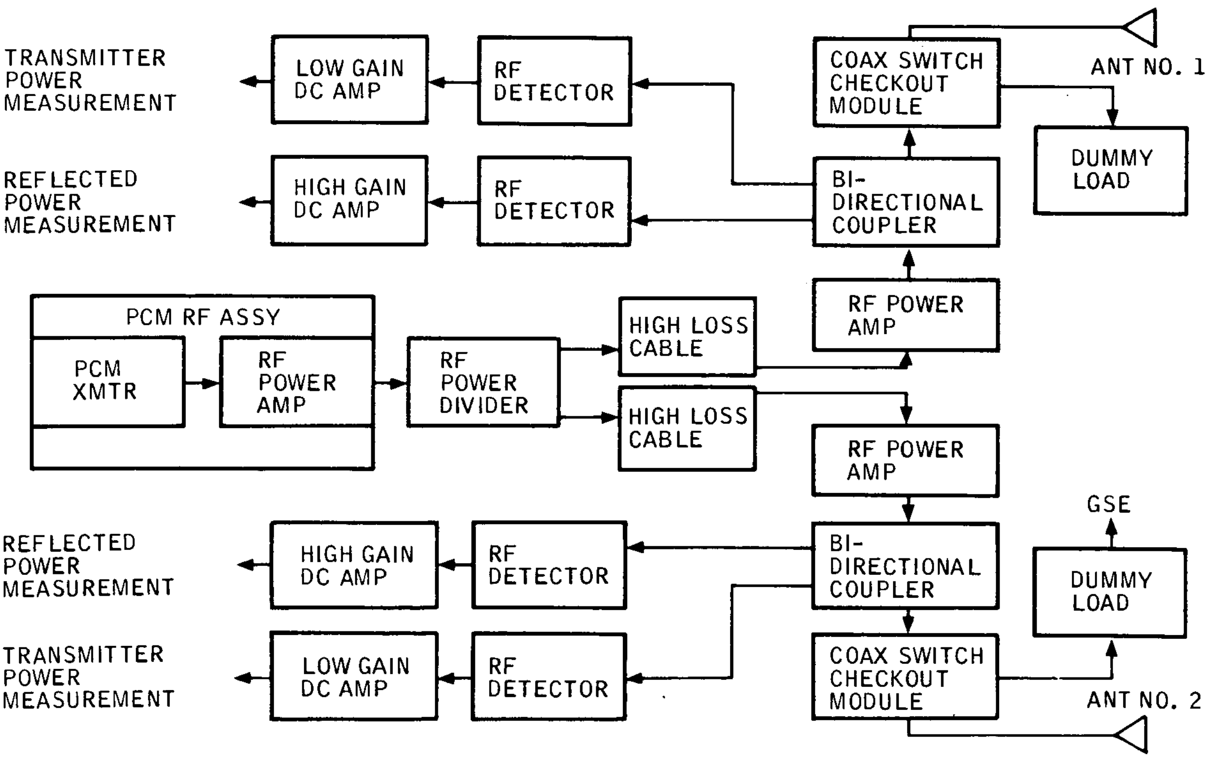


FIGURE 10.3.3-13. IMPROVED PCM RF SYSTEM

TABLE 10.3.3-X. PCM RF SYSTEM SELECTION CRITERIA FOR A SYNCHRONOUS ORBIT MISSION

	Selected System
Selected Receiving Stations Required	No
All S-IVB Meas Received During Flight	Yes
Decrease in S-IVB Meas Rate	No
Antenna Switching Required	No
Vehicle Attitude Control	No
Vehicle Attitude Sensing	No
Antenna Pattern Test	No
New Hardware	No
System Improvement	10 db
System Description	Add two power amplifiers

TABLE 10.3.3-XI. INSTRUMENTATION HARDWARE FOR THE J-2S SYNCHRONOUS ORBIT MISSION

Quantity	Part Name	*Reference Part Number
Add 3	Signal Conditioning Racks**	1B55689
Add 2	Amplifiers	1A77080
Add 2	Model 270 Multiplexers	1B55251
Add 2	Low Level Remote Analog Submultiplexers	1B54062
Add 1	Tape Recorder	1A66884-xxx
Add 110	Temperature Transducers	1A67863
Add 7	Pressure Transducers	1B43324

*This part number provided similarity information for pricing purposes and may not be the actual part number.
**These racks hold the necessary temperature bridges, amplifiers, etc. for the added transducers.

TABLE 10.3.3-XI. (Continued)

Quantity	Part Name	*Reference Part Number
Delete 1	Forward Power Transducer	1A94910-511
Delete 1	Reflected Power Transducer	1A54875-501
Add 2	Forward Power Transducers	1A95910-511
Add 2	Reflected Power Transducers	1A54875-501
Modify 1	Wire Harness 404W208	50% Increase to 1B67209
Modify 1	Wire Harness 406W200	200% Increase to 1B58865
Modify 1	Wire Harness 408W200	300% Increase to 1B58247
Modify 1	Wire Harness 404W209	25% Increase to 1B66648
Modify 1	Wire Harness 404W206	25% Increase to 1B66646
Modify 1	Wire Harness 411W219	50% Increase to 1B64944
Modify 1	Wire Harness 411W200	50% Increase to 1B58282
Modify 1	Wire Harness 411W205	50% Increase to 1B76615
Add 1	Wire Harness 405Wxxx	Equal to 1B65303
Modify 2	Wire Harness 414W200 415W200	50% Increase to 1B58284

*This part number provided similarity information for pricing purposes and may not be the actual part number.

10.3.4 Thermal Environment and Protection

The thermal environments for the S-IVB Stage were defined for the boost and orbital phases of flight. The aerodynamic heating environments were found to be less severe than those produced by the AS-501 maximum heating trajectory; therefore, no high temperature problems are indicated for the vehicle structure. Thermal analyses of the LH_2 and LOX feedlines were performed to determine the orbital minimum and maximum line temperatures in support of the engine chill-down analyses. The results indicate that most of the line segments will exceed the Rocketdyne upper temperature limit (-100°F) for the various mission phases considered unless the surface optical properties are modified.

10.3.4.1 Aerodynamic Heating

The J-2S missions as described in NASA letter R-P&VE-PA-68-L-52, dated 6 August 1968 (LOR and synchronous) are less severe from a heating viewpoint than the AS-501 design trajectory. No high temperature problems are indicated for the S-IVB vehicle structure.

10.3.4.2 Orbital Heating

The objective of this thermal analysis was to determine minimum and maximum LOX and LH_2 feedline temperatures for use in the engine chilldown analysis. Also determined were maximum orbital heating rates to the LH_2 and LOX when flowing through the feedlines. These heating rates are presented below for a vehicle in Low Earth Orbit (LEO) and in a transfer trajectory.

	LEO	Transfer Trajectory
LH_2 Feedline	2100 Btu/hr	1900 Btu/hr
LOX Feedline	500 Btu/hr	675 Btu/hr

The thermal models for both feedlines, extending from the propellant tanks to the J-2S engine interface, consist of a series of cylindrical nodes (Figure 10.3.4-1) whose temperatures are determined by use of a three-dimensional heat transfer digital computer program. Each node, representing a line segment which is partially shaded by the surrounding structure, is input with variable direct solar, albedo, and earth emission heat fluxes and allowed to reradiate to space and the surrounding structure. The portion of the LH_2 feedline enclosed in the fairing and aft skirt region, and the LOX feedline inside the thrust structure radiate to the surrounding structure. Line supports and brackets are represented by conduction paths in the thermal models.

The following assumptions were used in determining the minimum and maximum feedline temperatures.

a. Earth Orbit Parameters

1. 100 nautical mile circular orbit.
2. Maximum heating orbit - angle between orbit plane and sun vector (β) = 52 degrees.
3. Minimum heating orbit - β = 0 degree.

b. Vehicle Orientation

1. LEO - vehicle centerline parallel to velocity vector; position plane I toward earth.
2. Transfer trajectory to synchronous orbit - vehicle centerline perpendicular to sun vector with a one revolution per hour roll rate.

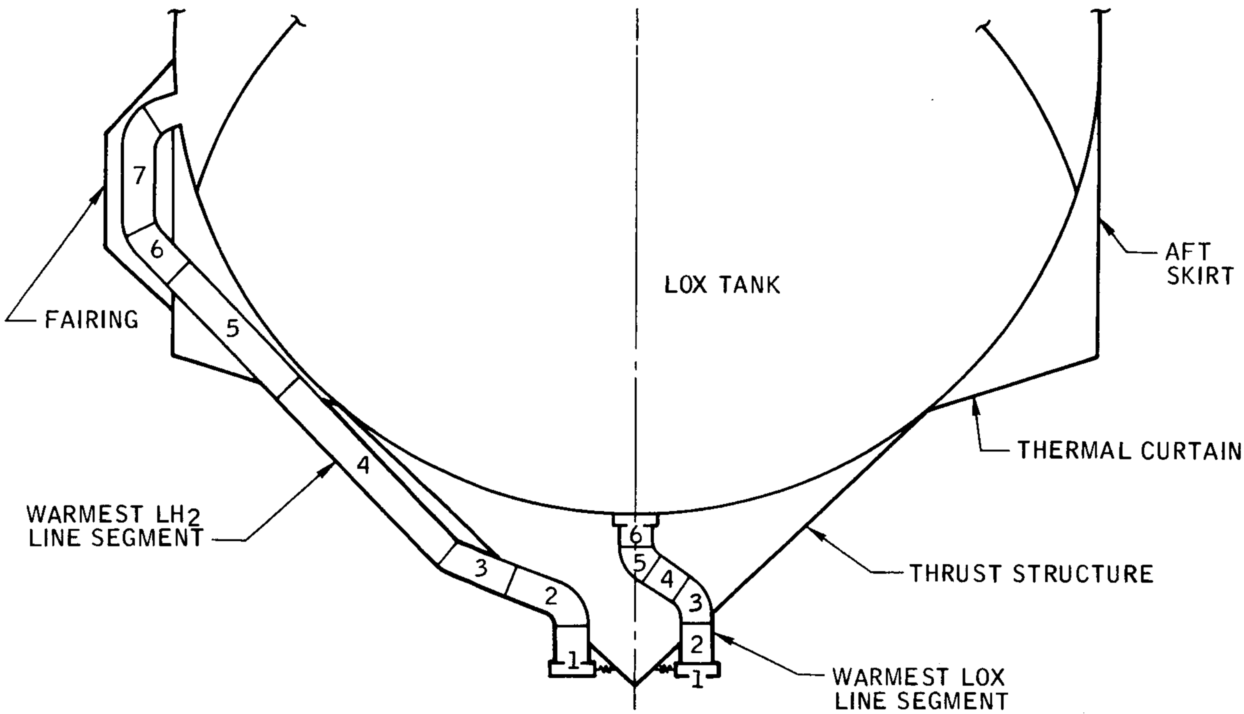


FIGURE 10.3.4-1. LH_2 AND LOX FEEDLINE THERMAL MODELS

10.3.4.2 (Continued)

c. Feedline Optical Properties

   1. Solar absorptivity (α) = 0.52.

   2. Emissivity (ϵ) = 0.29.

d. Vaporized propellant and oxidizer were assumed to occupy the lines immediately after engine cutoff.

Figures 10.3.4-2 and 10.3.4-3 present the minimum and maximum temperatures along the LH_2 and LOX feedlines that exist at the engine restart times being considered in this study. Temperature bands are presented for both lines during the earth orbit phases of the flight. These bands define the range of temperatures which occur because of the variation of the time that the vehicle is in the earth's shadow (dependent on the β angle). No temperature band is presented for the 5.3-hour transfer phase of flight since the only source of heat during this phase of the mission is direct solar radiation, and the 1 rph roll rate produces a fixed time in the sun for each line segment.

The results indicate the maximum average LOX feedline temperature will occur at the end of the 5.3-hour transfer trajectory, whereas the maximum average LH_2 line temperature will occur at the end of the 7.5-hour earth orbit period. This is because, unlike the LOX line, the LH_2 feedline runs along the entire length of the thrust structure which produces a greater shading effect on the line during the transfer orbit, when the only source of heat on the line is highly directional solar energy. During the low earth orbit phase of flight, when there is diffuse albedo and earth emitted energy incident on the line, the shading effects of the thrust structure are less pronounced, resulting in the higher average line temperature.

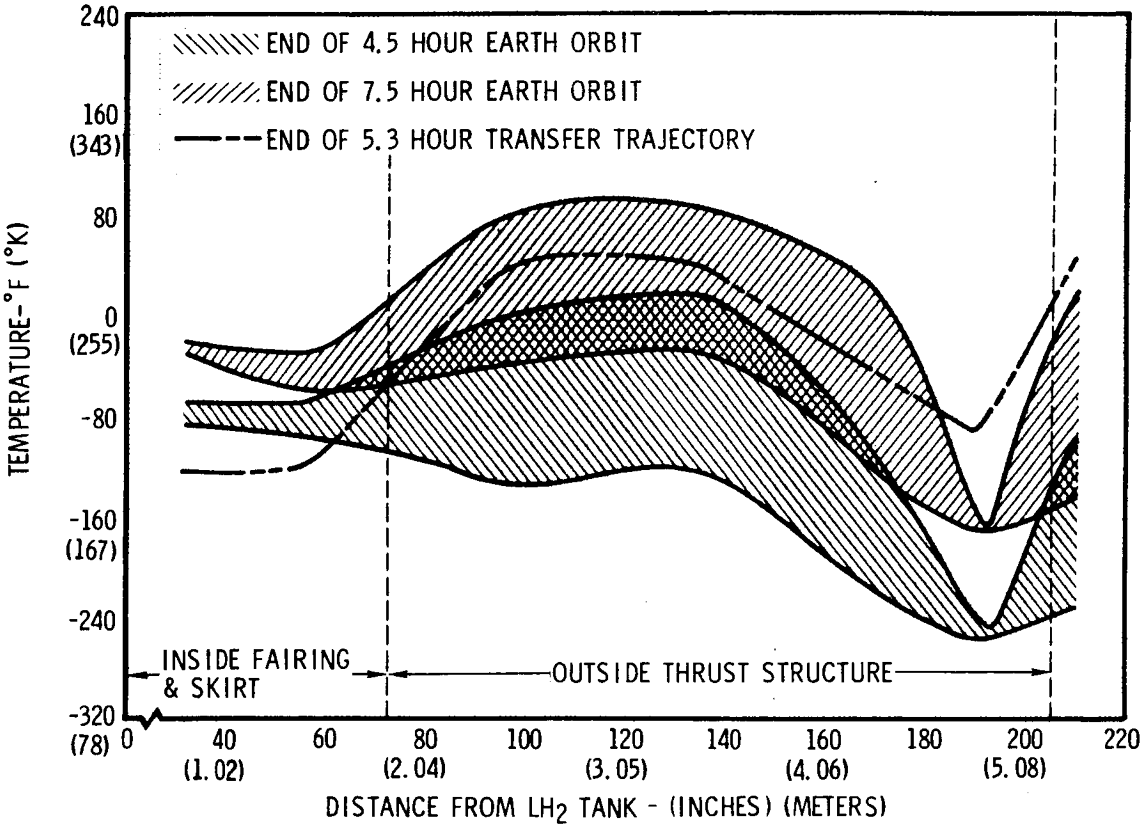


FIGURE 10.3.4-2. LH_2 FEEDLINE TEMPERATURE BAND

10.3.4.2 (Continued)

Temperature histories for the warmest LH_2 and LOX feedline segments are presented in Figures 10.3.4-4 and 10.3.4-5 respectively. The warmest LH_2 line segment is located adjacent to the thrust structure approximately 120 inches from the LH_2 tank. The warmest LOX line segment is located near the customer connect panel approximately 40 inches from the LOX tank. The LOX feedline has a more rapid warming rate in comparison to the LH_2 feedline because it is not vacuum jacketed and has less mass.

To reduce the line temperatures to values approaching the Rocketdyne -100°F upper temperature limit, the following possibilities should be considered:

a. Reduce the optical property ratio (α/ϵ) of the line segments exposed to direct solar radiation from their present high value of 1.8 to approximately 0.3 by painting those line segments white. Although there is the possibility of some degradation of the 0.3 value (i.e., an increase) due to contamination of the paint as the result of retrorocket plume impingement during separation from the S-II Stage, the final α/ϵ ratio should be considerably less than the present 1.8 value. A reduction of the α/ϵ ratio to 0.3 should maintain the line segments exposed to direct solar radiation at approximately 0°F during all of the missions presently being considered.

b. Reduce the emissivity of those line segments not exposed to direct solar radiation from the present value of 0.3 to approximately 0.1 by wrapping these segments with a low emissivity material such as aluminized mylar. This would reduce the rate of heat transfer to these line segments by radiation from the surrounding structure.

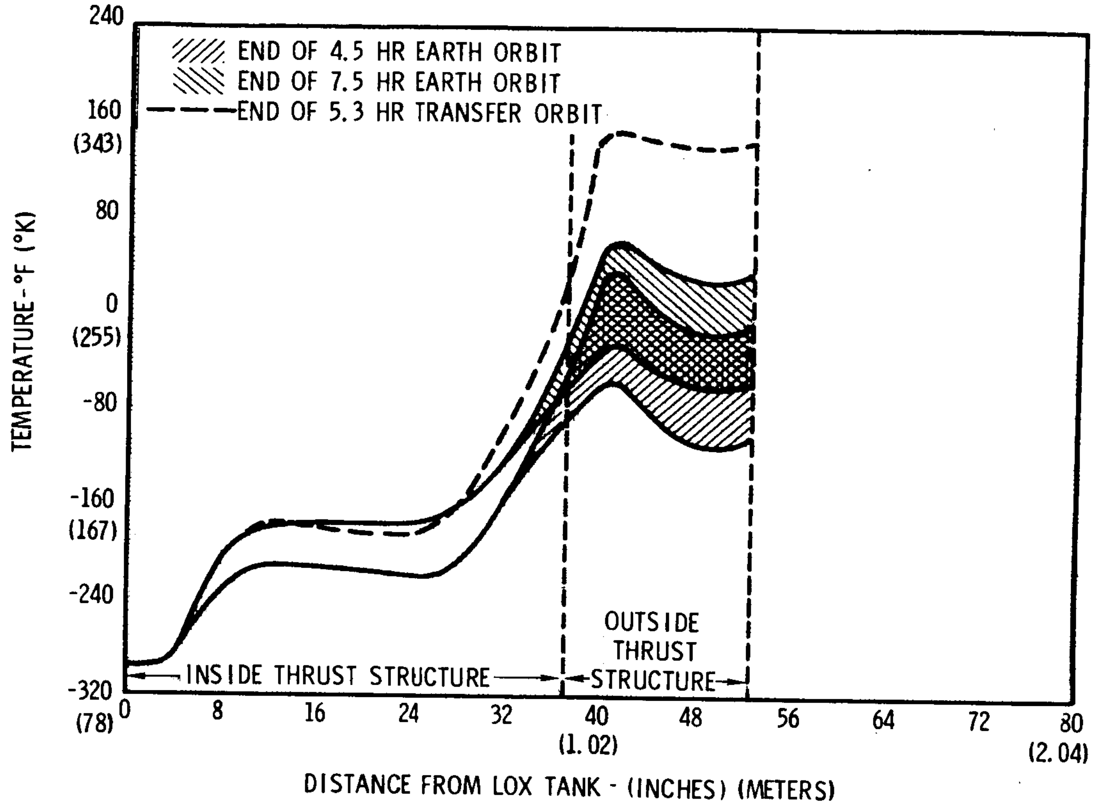


FIGURE 10.3.4-3. LOX FEEDLINE TEMPERATURE

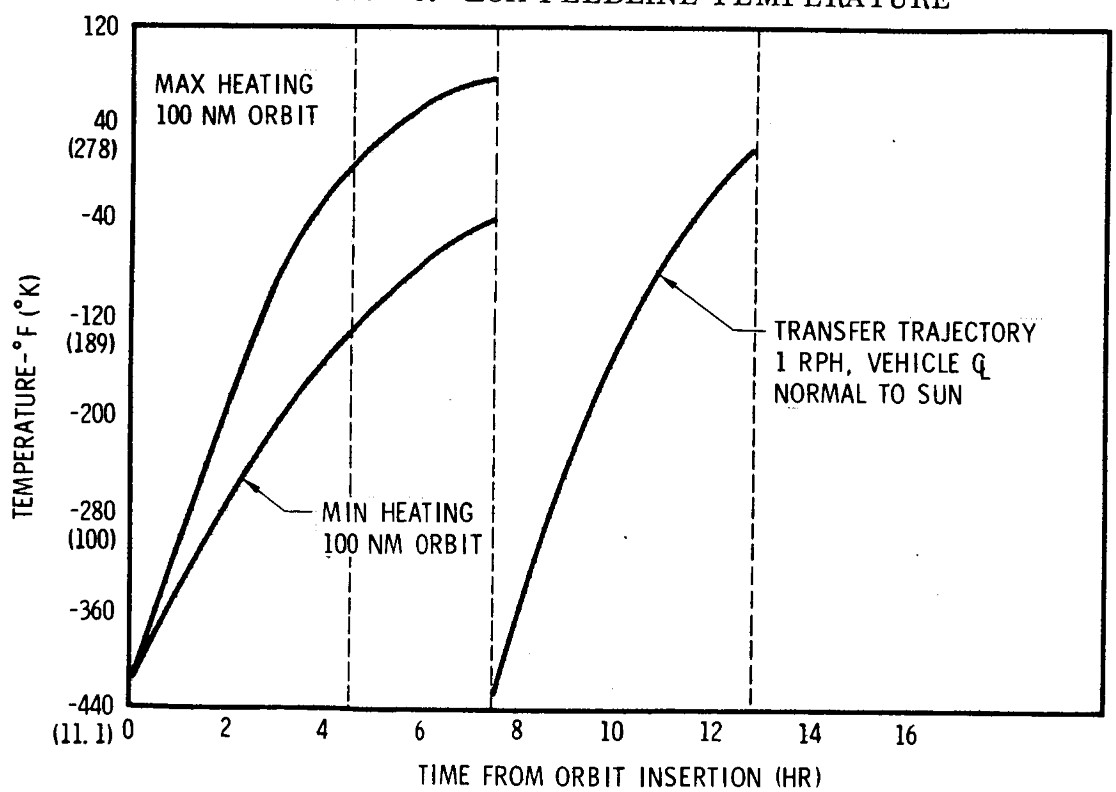


FIGURE 10.3.4-4. WARMEST LH_2 FEEDLINE SEGMENT TEMPERATURE HISTORY

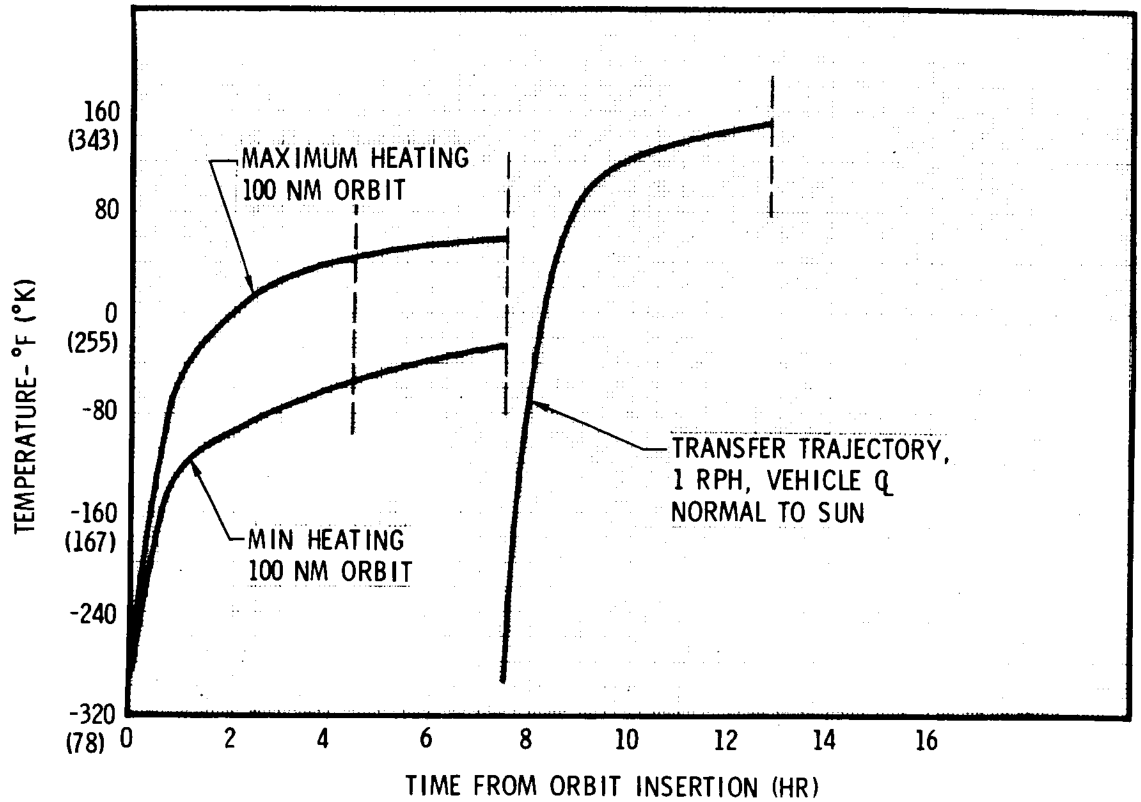


FIGURE 10.3.4-5. WARMEST LOX FEEDLINE SEGMENT TEMPERATURE HISTORY

10.3.4.2 (Continued)

c. An alternative to the first two possibilities would be to wrap all line segments with a high performance insulation system such as multiple layers of aluminized maylar. Such an insulation system, while it would have to be protected from the retrorocket plume impingement effects, should maintain most of the line segments well below -100°F.

10.3.5 Weight and Mass Properties

The weight data generated during this study examined the mass characteristic changes resulting from the incorporation of a J-2S engine as an "in-line modification" on an operational stage.

10.3.5.1 Weight Breakdown

Table 10.3.5-I is a tabulation, in NASA format, of the weights of the major subsystems of the three configurations evaluated. The baseline configuration, Vehicle 511, was chosen as the representative operational vehicle. The LOR mission and synchronous mission vehicles are modifications of the baseline vehicle incorporating a J-2S engine and mission related modifications as an "in-line change".

TABLE 10.3.5-I WEIGHT SUMMARY (J-2S IN LINE)

NASA 2nd Generation Breakdown		S-IVB-511 Baseline LOR Mission (lbs)	S-IVB-511 J-2S Engine LOR Mission (lbs)	S-IVB-511 J-2S Engine Sync. Orbit Mission (lbs)
Propellant Container	W3.3	8933	8923	9232
Forward of Tanks	W3.6	1242	1242	1242
Aft of Tanks	W3.8	1816	1801	1801
Thrust Structure	W3.9	774	809	809
Fairings & Assoc Structure	W3.10	196	174	174
Paint & Sealer	W3.15	104	104	104
Heat & Flame Prot Insulation	W3.18	182	182	182
Structure	W3.0	13248	13235	13544
Engine & Accessories	W4.1	3572	4012	4073
Purge System For Chilldown	W4.6	272	0	0
Fuel System	W4.7	1573	1616	1067
Oxidizer System	W4.8	1264	1317	1111
Cryogenic Repress System	W4.9	310	310	310
Stage Control Sys Hardware	W4.10	284	284	284
Propulsion Sys & Accessories	W4.0	7275	7539	6845
Equip & Instr Structure	W6.1	430	427	431
Environmental Control Sys	W6.2	231	231	268
Control Sys Electronics	W6.5	116	116	116
Telemetry & Measuring Sys	W6.8	1165	1126	1533
Propellant Utilization Sys	W6.10	175	175	175
Electrical System	W6.11	829	726	1060
Range Safety System	W6.12	69	69	69
Pneumatic System	W6.15	298	269	269
Auxiliary Propulsion Sys	W6.16	854	829	829
Separation System	W6.17	117	117	117
Ullage System	W6.18	212	0	0
Systems for Total Vehicle	W6.20	91	91	91
Equipment & Instrumentation	W6.0	4588	4176	4958
Stage Dry Weight	WAD	25111	24950	25347
Change from S-IVB-511 Baseline		Ø	-161	+236

10.3.5.2 Weight Substantiation

To aid in the evaluation of the study results, the substantiation for the weight changes shown in Table 10.3.5-I are presented below.

Propellant Container W3.3

	LOR	Sync Orbit
Remove chilldown systems fittings coded to propellant container	-10 lb	-10 lb
Delete existing LH2 tank baffle		-198 lb
Add new LH2 tank baffle & deflector		+220 lb
Add new LOX tank baffle		+287 lb
Change to W3.3	-10 lb	+299 lb

Aft of Tanks W3.8

	LOR	Sync Orbit
Redesign skin supports due to deletion of chilldown system	-21 lb	-21 lb
Redesign skin & stringers due to deletion of ullage rockets	+ 6 lb	+ 6 lb
Change to W3.8	-15 lb	-15 lb

Thrust Structure W3.9

	LOR	Sync Orbit
Increase stringer area on thrust structure due to higher design loads of J-2S engine	+35 lb	+35 lb
Change to W3.9	+35 lb	+35 lb

Fairings & Associated Structure W3.10

	LOR	Sync Orbit
Remove chilldown system line fairings	-22 lb	-22 lb
Change to W3.10	-22 lb	-22 lb

10.3.5.2 (Continued)

Engine & Accessories W4.1

	LOR	Sync Orbit
Remove engine start tank fill line	- 6 lb	- 6 lb
Remove engine start tank vent & relief line	- 6 lb	- 6 lb
Add engine LOX dome purge line and wiring	+ 7 lb	+ 7 lb
Remove standard J-2 engine	-3515 lb	-3515 lb
Add J-2S engine with 2 solid propellant turbine starters	+3960 lb	
Add J-2S engine with 3 solid propellant turbine starters		+4021 lb
Change to W4.1	+ 440 lb	+ 501 lb

Purge System for Chilldown W4.6

	LOR	Sync Orbit
Remove chilldown & purge system	- 256 lb	- 256 lb
Remove multipurpose support structure and assign to oxidizer system	- 16 lb	- 16 lb
Change to W4.6	- 272 lb	- 272 lb

Fuel System W4.7

	LOR	Sync Orbit
Add spacer & couplings to replace chilldown prevalve	+ 43 lb	+ 43 lb
Delete (5) ambient helium bottles, bottle supports, and plumbing		-664 lb
Add (2) cold helium bottles, bottle supports, and plumbing		+115 lb
Change to W4.7	+ 43 lb	-506 lb

10.3.5.2 (Continued)

Oxidizer System W4.8

	LOR	Sync Orbit
Add multipurpose supports previously coded to W4.6	+ 16 lb	+ 16 lb
Add spacer to replace chill-down prevalve	+ 37 lb	+ 37 lb
Delete 2 ambient helium bottles, bottle supports, and plumbing		- 264 lb
Add 1 cold helium bottle, bottle supports, and plumbing		+ 58 lb
Change to W4.8	+ 53 lb	- 153 lb

Equipment & Instrumentation Structure W6.1

	LOR	Sync Orbit
Remove panel isolators no longer required	- 3 lb	- 3 lb
Add wire support panel for thermoconditioning equipment		+ 4 lb
Change to W6.1	- 3 lb	+ 1 lb

Environmental Control Equipment W6.2

	LOR	Sync Orbit
Add heater blankets		+ 11 lb
Add 1 cold plate for tape recorder		+ 26 lb
Change to W6.2	ø	+ 37 lb

Telemetry & Measuring Equipment W6.8

	LOR	Sync Orbit
Remove LOX depletion sensors	- 7 lb	- 7 lb
Remove feed thru disconnect and wiring	-15 lb	-15 lb
Delete 54 telemetry measurements	-35 lb	-35 lb

10.3.5.2 (Continued)

	LOR	Sync Orbit
Add new J-2S telemetry measurements.	+ 18 lb	+210 lb
Add tape recorder.		+ 38 lb
Add XMTR switch assembly.		+ 1 lb
Add RF power amplifier.		+ 1 lb
Add 2 Model 270 multiplexers.		+ 40 lb
Add 3 signal conditioning racks.		+135 lb
Change to W6.8	- 39 lb	+368 lb

Electrical System W6.11

	LOR	Sync Orbit
Remove aft No. 1 battery.	-170 lb	
Remove forward No. 1 battery.		-170 lb
Add smaller aft No. 1 battery.	+ 78 lb	
Add 2 larger forward No. 1 batteries.		+378 lb
Modify wiring.	- 11 lb	+ 14 lb
Add E/I switch.		+ 9 lb
Change to W6.11	-103 lb	+231 lb

Pneumatic System W6.15

	LOR	Sync Orbit
Remove control modules associated with chilldown system.	- 27 lb	- 27 lb
Modify engine pneumatic control modules and plumbing.	- 2 lb	- 2 lb
Change to W6.15	- 29 lb	- 29 lb

10.3.5.2 (Continued)

Auxiliary Propulsion System W6.16

	LOR	Sync Orbit
Remove APS ullage engines	- 22 lb	- 22 lb
Remove engine support structure and wiring	- 3 lb	- 3 lb
Change to W6.16	- 25 lb	- 25 lb

Ullage System W6.18

	LOR	Sync Orbit
Remove entire ullage rocket system	- 212 lb	- 212 lb
Change to W6.18	- 212 lb	- 212 lb

10.3.5.3 Drop Weights

Tables 10.3.5-II and 10.3.5-III present the S-IVB drop weights for the LOR and synchronous orbit missions respectively. These drop weights are listed in the format specified by Boeing memorandum 5-9225-H-121, dated 6 September 1968.

TABLE 10.3.5-II S-IVB DROP WEIGHTS LOR MISSION (J-2S IN LINE)

Mission Event	Item Description	Item Weight (LBM)
S-II/S-IVB Separation	AFT Interstage Retro Propellant Separation Package	7,021 1,062 51
Idle Mode Start	Idle Mode LOX Idle Mode LH2	13 5
Mainstage Start	Turbine Spin Prop LOX Thrust Buildup LH2 Thrust Buildup	10 365 145
90 Percent Thrust	APS (Roll Control)	1
Engine Cutoff Command	LOX Thrust Decay LH2 Thrust Decay LOX Idle Mode LH2 Idle Mode	91 40 20 8

TABLE 10.3.5-II (Continued)

Mission Event	Item Description	Item Weight (LBM)
Start Coast	LOX Vented	270
	LH2 Vented	3,623
	APS (Attitude Control)	30
2nd Idle Mode Start	LOX Idle Mode	150
	LH2 Idle Mode	920
2nd Mainstage Start	LOX Thrust Buildup	365
	LH2 Thrust Buildup	145
	Turbine Spin Prop	10
2nd 90 Percent Thrust	APS (Roll Control)	2
2nd Engine Cutoff	LOX Thrust Decay	91
	LH2 Thrust Decay	40
	LOX Idle Mode	20
	LH2 Idle Mode	8
Trans Lunar Injection	S-IVB Dry Stage	24,950
	LOX Trapped	396
	LH2 Trapped	733
	LOX Ullage Gas	377
	LH2 Ullage Gas	425
	Cold Helium in Bottles	197
	Repress Helium (Ambient)	72
	APS Propellant	589
	Service Items	57

TABLE 10.3.5-III S-IVB DROP WEIGHTS SYNCHRONOUS ORBIT MISSION (J-2S IN LINE)

Mission Event	Item Description	Item Mass (LBM)
S-II/S-IVB Separation	AFT Interstage	7,021
	Retro Propellant	1,062
	Separation Package	51
Idle Mode Start	Idle Mode LOX	13
	Idle Mode LH2	5
Mainstage Start	Turbine Spin Prop	10
	LOX Thrust Buildup	365
	LH2 Thrust Buildup	145
Ninety Percent Thrust	APS (Roll Control)	1

TABLE 10.3.5-III (Continued)

Mission Event	Item Description	Item Mass (LBM)
J2-S Cutoff Command	LOX Thrust Decay	91
	LH2 Thrust Decay	40
	LOX Idle Mode	20
	LH2 Idle Mode	8
Start Coast	LOX Vented	400
	LH2 Vented	6,606
	APS (Attitude Control)	75
2nd Start Idle Mode	LOX Idle Mode	150
	LH2 Idle Mode	920
2nd Mainstage Start	LOX Thrust Buildup	365
	LH2 Thrust Buildup	145
	Turbine Spin Prop	9
2nd 90 Percent Thrust	APS (Roll Control)	1
2nd J2-S Cutoff Command	LOX Thrust Decay	91
	LH2 Thrust Decay	40
	LOX Idle Mode	20
	LH2 Idle Mode	8
Start Coast	LOX Vented	300
	LH2 Vented	1,800
	APS (Attitude Control)	68
3rd Start Idle Mode	LOX Idle Mode	150
	LH2 Idle Mode	920
3rd Mainstage Start	LOX Thrust Buildup	365
	LH2 Thrust Buildup	145
	Turbine Spin Prop	9
3rd 90 Percent Thrust	APS (Roll Control)	1
3rd J2-S Cutoff Command	LOX Thrust Decay	91
	LH2 Thrust Decay	40
	LOX Idle Mode	20
	LH2 Idle Mode	8
Sync Orbit Injection	S-IVB Dry Stage	25,347
	LOX Trapped	396
	LH2 Trapped	733
	LOX Ullage Gas	570
	LH2 Ullage Gas	614
	Cold Helium	120
	APS Propellants	199
	Service Items	59

10.3.5.4 Mass Characteristics

Table 10.3.5-IV presents the S-IVB dome mass characteristics for the LOR mission. This mission is considered the worst controllability case for the study.

TABLE 10.3.5-IV. S-IVB DOME MASS CHARACTERISTICS AT 1ST ENGINE START COMMAND FOR THE LOR MISSION

	Weight (lbs)	Horiz. C.G. (Inches)	Vertical C.G. (Inches)	Lateral C.G. (Inches)	Point of Attachment S-IVB Station (Inches)	Pitch Moment of Inertia (3) (Slug-ft^2)
Forward Dome (1)	11,240	597.7	0	0	554.7	5,888
Aft Dome (2)	204,008	237.2	-0.1	-0.1	286.1	143,492

(1) Includes forward dome, insulation, accessories and 10,082 lbs. of LH_2.

(2) Includes aft dome, common bulkhead, thrust structure and accessories, J-2S engine, LH_2 and LOX feed lines, and 193,273 lb of LOX.

(3) The Pitch moments of inertia of the domes are about their respective attach points.

10.3.6 Ground Support Equipment

The GSE associated with the S-IVB is divided into propulsion, electrical, and mechanical support systems. The following paragraphs are identified by the corresponding stage system that is being serviced or monitored. Where deleted functions have resulted in unused GSE, the present policy of system isolation rather than removal has been applied.

The J-2S caused changes to propulsion GSE are generally ones of deleting functions and isolating hardware. Engine start tank elimination has permitted isolation of the GH_2 supply system. GSE heat exchanger subsystems were not isolated since this would require new control circuits, the requirement for both engine and stage cryogenic helium was eliminated. Rather, no LH_2 is admitted to the GSE heat exchanger, thereby, significantly reducing prelaunch efforts and helium consumption. Propulsion GSE changes occur only at STC and KSC.

Electrical GSE changes are required where valve functions or monitoring are no longer required in the propulsion GSE. In addition, a large number of changes are required to the automatic checkout equipment at Huntington Beach and STC.

10.3.6 (Continued)

However, since these systems have inherent flexibility, most changes are readily accomplished via patch panel changes. At KSC the majority of electrical changes are in the government furnished GSE made by General Electric. The expected changes to panels in the low bay, launch umbilical tower, and launch control center have been identified. More details of KSC impact is found in Section 4.4.7 of the Resources Analysis, D5-15772-7.

Mechanical GSE is affected by certain access kit changes resulting from removal or relocation of stage and engine components.

10.3.6.1 Propulsion System Ground Support Equipment

The primary pieces of propulsion GSE consist of the following:

a. One pneumatic console, Model 321, located at Huntington Beach, and one pneumatic console, Model 321A, located at STC. Figure 10.3.6-1 is typical of each.

b. Two each pneumatic consoles, Model 319 and 320, located at STC. Three each pneumatic consoles, Model 432A and 433A, located at KSC. Figure 10.3.6-2 is typical of each.

c. Two gas heat exchangers, Model 318, located at STC. This model is similar to Model 438A below except for deletion of the GH_2 control system rack assembly.

d. Three gas heat exchangers, Model 438A, located at KSC (Figure 10.3.6-3).

The block diagrams (Figure 10.3.6-4 for STC and Figure 10.3.6-5 for KSC) show the GSE and S-IVB Stage interfaces.

The pneumatic console, Model 432A, is the ambient helium distribution console. This console is designed to receive ambient temperature helium between 3,500 psig and 6,600 psig, regulate it to lower pressures and distribute the helium through its internal system to the gas heat exchanger (Model 438A), pneumatic console (Model 433A), and the stage at the required pressure and flowrates.

The gas heat exchanger (Model 438A) consists of a dewar-type, double-walled cylindrical vessel which contains three systems of coils (Figure 10.3.6-3). The vessel is mounted on a frame base with the LH_2 fill and vent systems and the rack-type control console. The function of the gas heat exchanger is to produce cryogenic helium and hydrogen gas needed by the stage and J-2 engine for chilldown and pressurization functions during the latter portion of the launch countdown. The heat exchanger is designed to receive LH_2 and gaseous hydrogen from the facility, helium from the Model 432A console, and nitrogen from the Model 433A console. LH_2 is supplied from the facility to the heat exchanger inlet. The LH_2 passes through a 2-inch control valve and a vacuum-jacketed pipe into the bottom of the vessel.

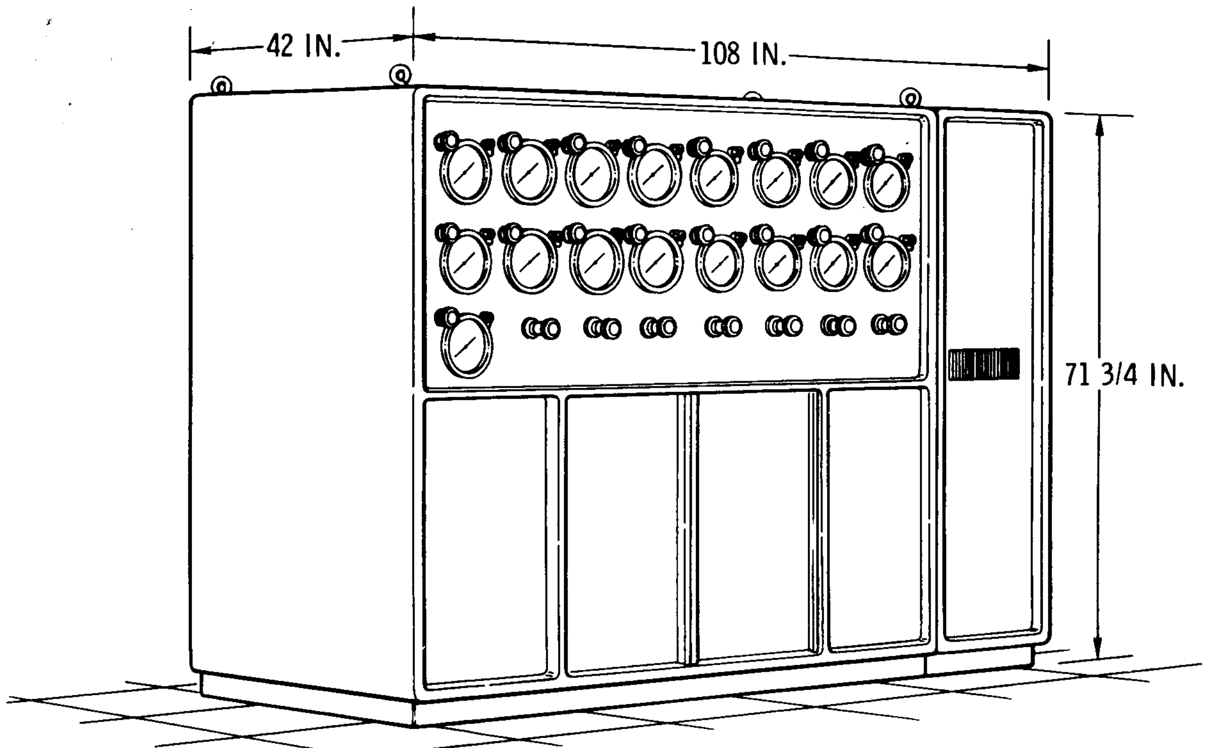


FIGURE 10.3.6-1. PNEUMATIC CONSOLE, MODEL 321

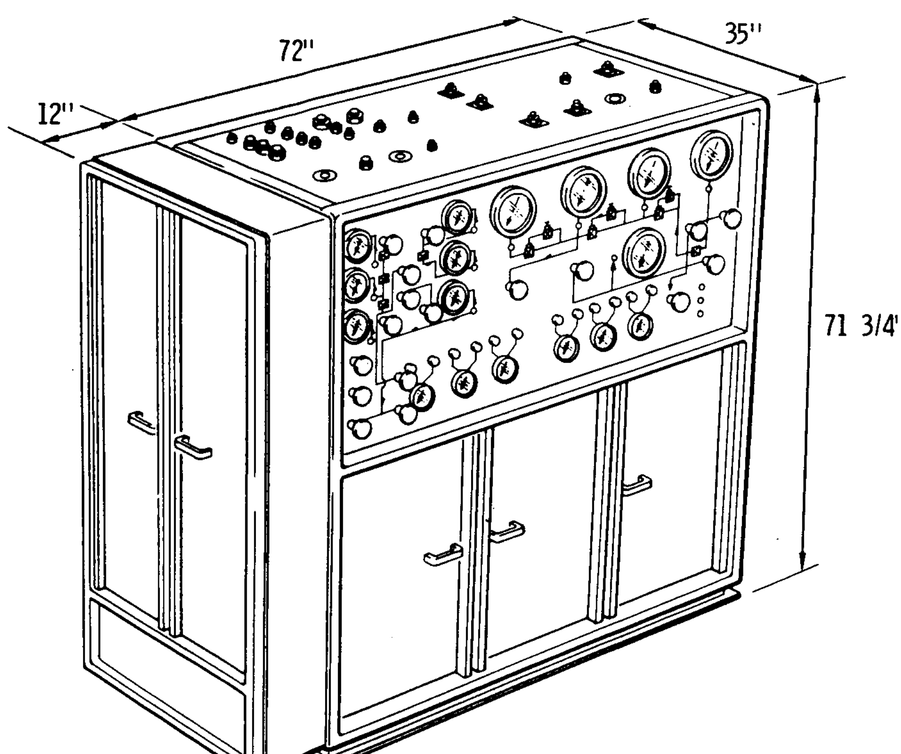


FIGURE 10.3.6-2. PNEUMATIC CONSOLE, MODELS 433A AND 320
SIMILAR TO 319 AND 432A

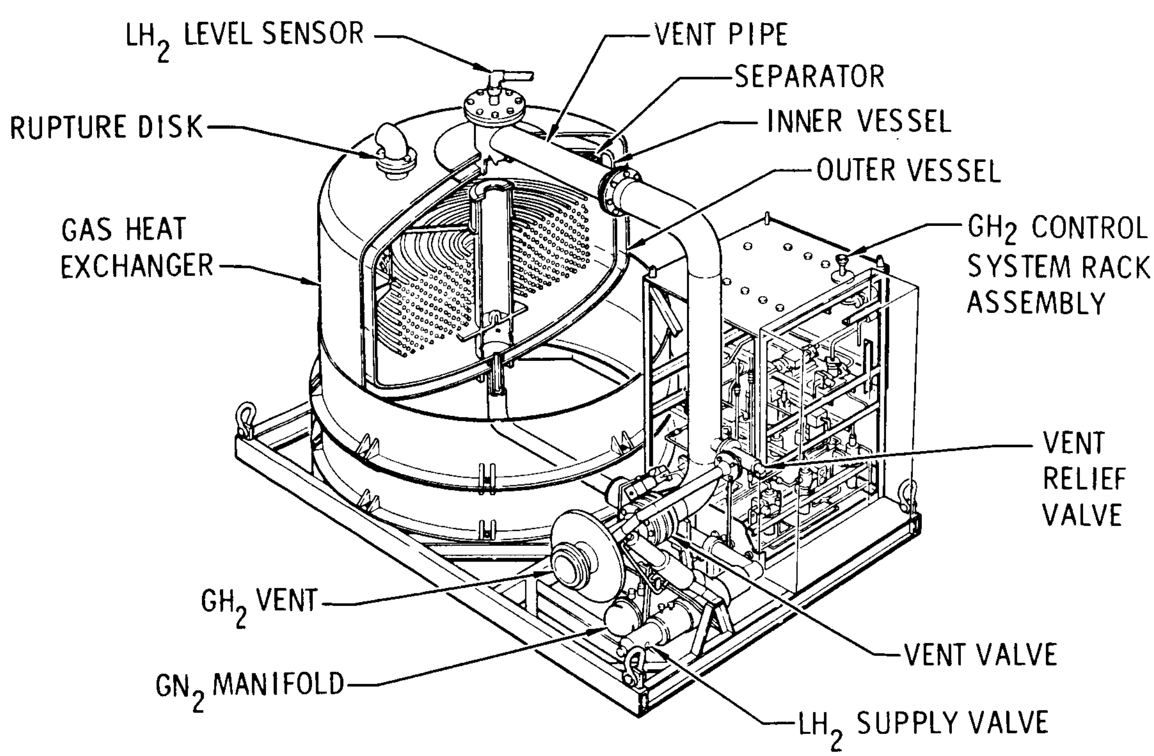


FIGURE 10.3.6-3. GAS HEAT EXCHANGER-MODEL 438A

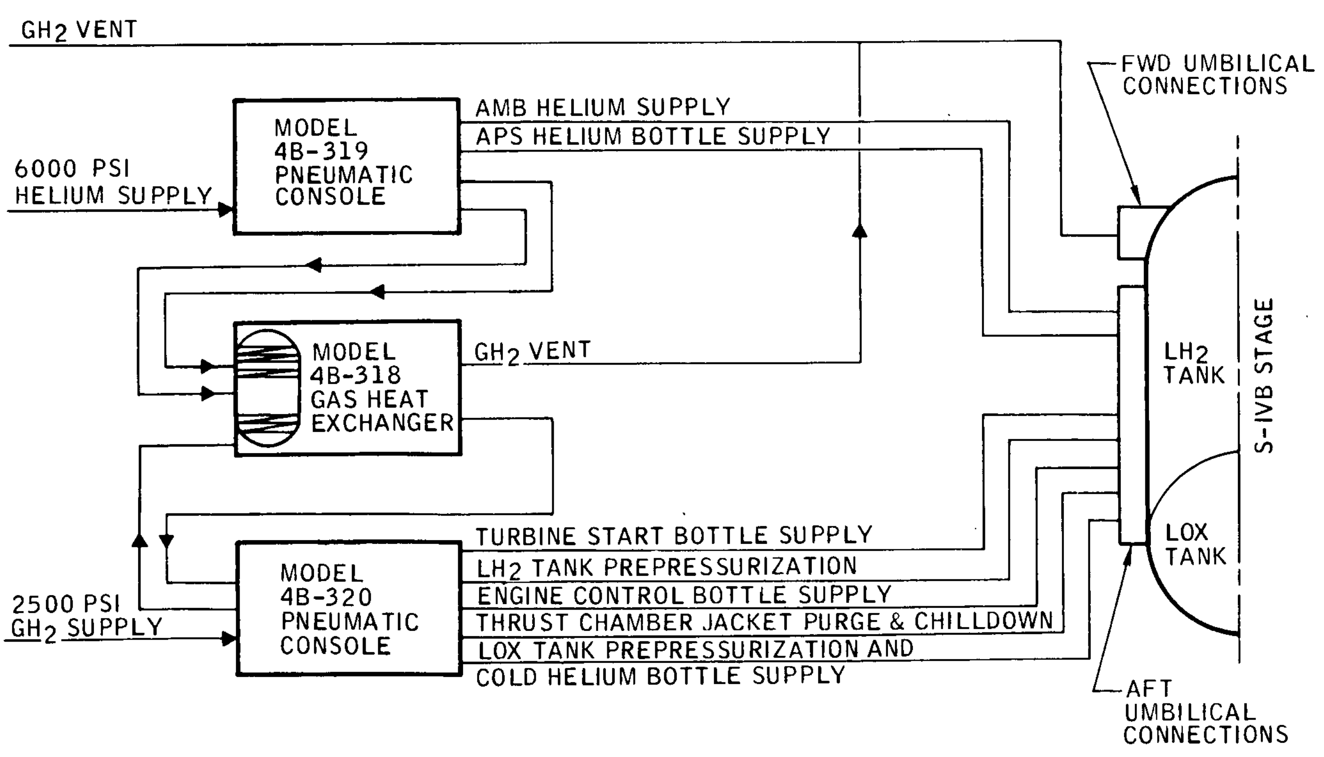


FIGURE 10.3.6-4. PROPULSION GROUND SUPPORT EQUIPMENT (STC)

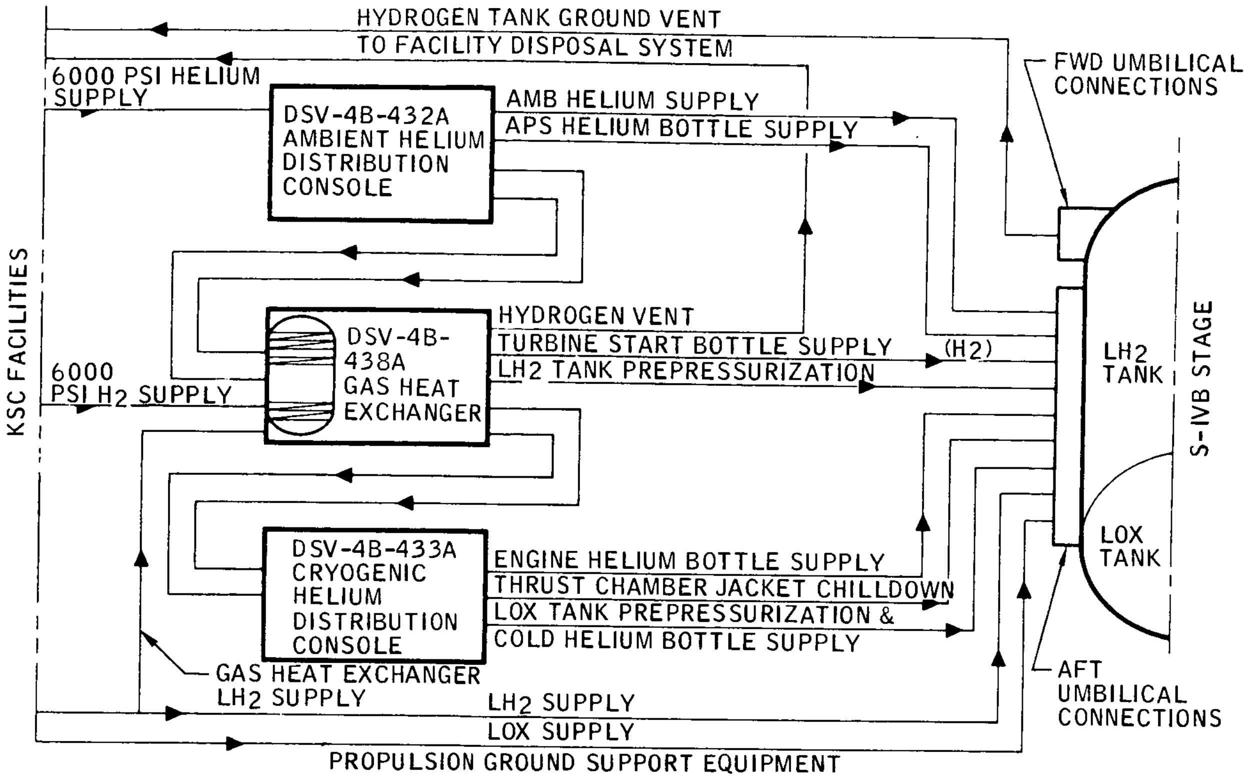


FIGURE 10.3.6-5. PROPULSION GROUND SUPPORT EQUIPMENT (KSC)

10.3.6.1 (Continued)

The LH_2 volume in the vessel is controlled by a capacitance type level sensor. The level sensor ensures that the three systems of coils are immersed in LH_2 at all times during operation. The level sensor automatically opens and closes the fill valve, as required.

The rack assembly adjacent to the heat exchanger tank contains the gaseous hydrogen system and the helium inlet and outlet control systems. The rack has a control panel for manual setup and checkout of the unit. An enclosed electrical cabinet is mounted on the base next to the rack. The rack is an open console to provide ventilation and prevent possible explosion from entrapped hydrogen. Facility supplied gaseous hydrogen at 1,600 to 6,000 psig is received by the gaseous hydrogen system. The pressure is regulated to 1,225 ±50 psig before entering the heat exchanger. The regulator is a dome-loaded type, but instead of using upstream gas to load the dome, a supply of helium from Model 432A is used for safety reasons.

All of the lines and components leaving the heat exchanger through the rack assembly are well insulated to minimize heat leak into the flowing cold gas. In addition to the gaseous hydrogen and helium systems, there is a gaseous nitrogen system to provide control pressure for the solenoid operated pneumatic valves. The 750 psig gaseous nitrogen is supplied from Model 433A console.

10.3.6.1 (Continued)

The Model 433A pneumatic console is primarily the cold helium distribution console. The cabinet is very similar to Model 432A and is illustrated in Figure 10.3.6-2.

The console receives cryogenic helium from circuits 2 and 3 of the Model 438A gas heat exchanger, and distributes the gas to the stage and J-2 engine. Model 433A console contains the gaseous nitrogen system. It is designed to receive gaseous nitrogen from the facility at 6,000 psig and regulate and distribute the gas to purge the Model 432A console, the interior of the Model 433A console, and the Model 438A electrical console. The purge maintains a positive pressure of 2-inches of water in the cabinets. The gaseous nitrogen system also supplies actuation pressure for all pneumatically-operated valves in the GSE. All cryogenic lines and components in the console are insulated.

The combination of Models 318, 319, and 320, located at STC is similar to the combination of KSC Models 438A, 432A, and 433A, and has the same combined functional capabilities.

GSE is supplied for propellant loading of the two APS modules of the S-IVB stage. The APS GSE consists of a fuel distribution console, oxidizing distribution console, and three regulator gage assemblies. The APS GSE is located on the mobile service structure at the level closest to the S-IVB APS modules. APS GSE supplied to perform complete receiving and preinstallation checkout of the APS propellant and pneumatic systems is similar in functional capabilities at Huntington Beach-VCL, STC-VCL, and APS Lab at KSC.

Table 10.3.6-I shows the systems which checkout and prepare the S-IVB stage for static firings and launch. The affected models and their locations are also shown. A discussion of J-2S impact on these systems is found in the following paragraphs.

Pneumatic Control Pressurization System

The pneumatic console pressurization system provides ambient temperature helium gas to the stage pneumatic control system and to the LOX and LH_2 repressurization storage spheres. The system must pressurize the stage pneumatic control sphere, and repressurization spheres in a 3-step operation required by stage system design, safety requirements, and engine system requirements. Implementation of the J-2S engine will not affect this system.

Auxiliary Propulsion System Pneumatic Pressurization System

The APS pneumatic pressurization system pressurizes the helium storage tank in each APS module for flight. The system also maintains propellant ullage pressure after the APS modules have been loaded and prior to pressurization of the helium tanks. Propellants are loaded onboard the APS modules before the actual launch countdown begins.

TABLE 10.3.6-I. S-IVB STAGE STATIC FIRINGS/LAUNCH SYSTEMS

System	Applicable Model Number		
	HB	STC	KSC
Pneumatic Control Pressurization System	321 321A	319 321A	432A
Auxiliary Propulsion System Pneumatic Pressurization System	321 321A	321A	432A
Cold Helium Bottle Pressurization System	321 321A	318 319 320	432A 433A 438A
Engine Turbine Start Tank Pressurization System	321 321A	318 320 321A	432A 438A
Engine Control Bottle Pressurization System	321 321A	318 319 320 321A	432A 433A 438A
Thrust Chamber Purge and Chilldown System	---	318 319 320	432A 433A 438A
LOX Tank Prepressurization System	321 321A	318 319 320 321A	432A 433A 438A
LH_2 Tank Prepressurization System	321 321A	318 319 320 321A	432A 433A 438A
Auxiliary Propulsion System Purge System	321 321A	321A	472 473
Oxidizer Dome Purge	---	320	432A

10.3.6.1 (Continued)

The APS pneumatic system is pressurized in a 3-step operation to maintain ullage pressure on the propellants during standby and to meet safety and vehicle system requirements. Implementation of the J-2S engine will not affect this system.

Cold Helium Bottle Pressurization System

The cold helium bottle pressurization system provides cryogenic helium gas to the storage spheres located in the stage LH_2 tank. The stage spheres provide helium gas for LOX tank pressurization and operation of the dual pressurization system for LOX and LH_2 tank repressurization during flight. The GSE system is required to produce the low temperature gas from an ambient supply. Pressurization of the spheres requires a high flow rate to load several storage spheres in a reasonable length of time. The cold helium spheres are pressurized in a 3-step operation to comply with stage checkout, safety, and purge requirements.

Implementation of the J-2S engine eliminates the requirement that the pressurization of the cold helium bottles, engine control bottle, LOX tank and LH_2 tank be accomplished with cold helium gas. At STC the LH_2 supply valve to Model 318 heat exchanger shall remain in the closed position at all times. At KSC the LH_2 supply valve to Model 438A heat exchanger shall remain in the closed position at all times. These changes will be accomplished by STC and KSC revisions to applicable checkout and loading procedures.

Engine Turbine Start Tank Pressurization System

The turbine start tank pressurization system supplies cryogenic hydrogen gas for chilldown and pressurization of the J-2 engine turbine start tank. The system also supplies ambient temperature helium to purge and dry the tank prior to chilldown. The implementation of the J-2S engine eliminates the requirement for this system. To save cost, the unused systems will not be removed from the GSE. At STC the GH_2 system will be disconnected and capped off at the following locations (Figure 10.3.6-6):

a. Facility in-line tee located downstream of Model 320 of GH_2 supply to heat exchanger.

b. Turbine start bottle supply outlet port.

c. 50 psig purge supply upstream of purge supply valve A-9377.

d. GH_2 vent system downstream of check valves A-9548 and A-9511.

e. 750 psig GH_2 valve control pressure at inlet port of solenoid valves A-9361, A9360, and A-9441.

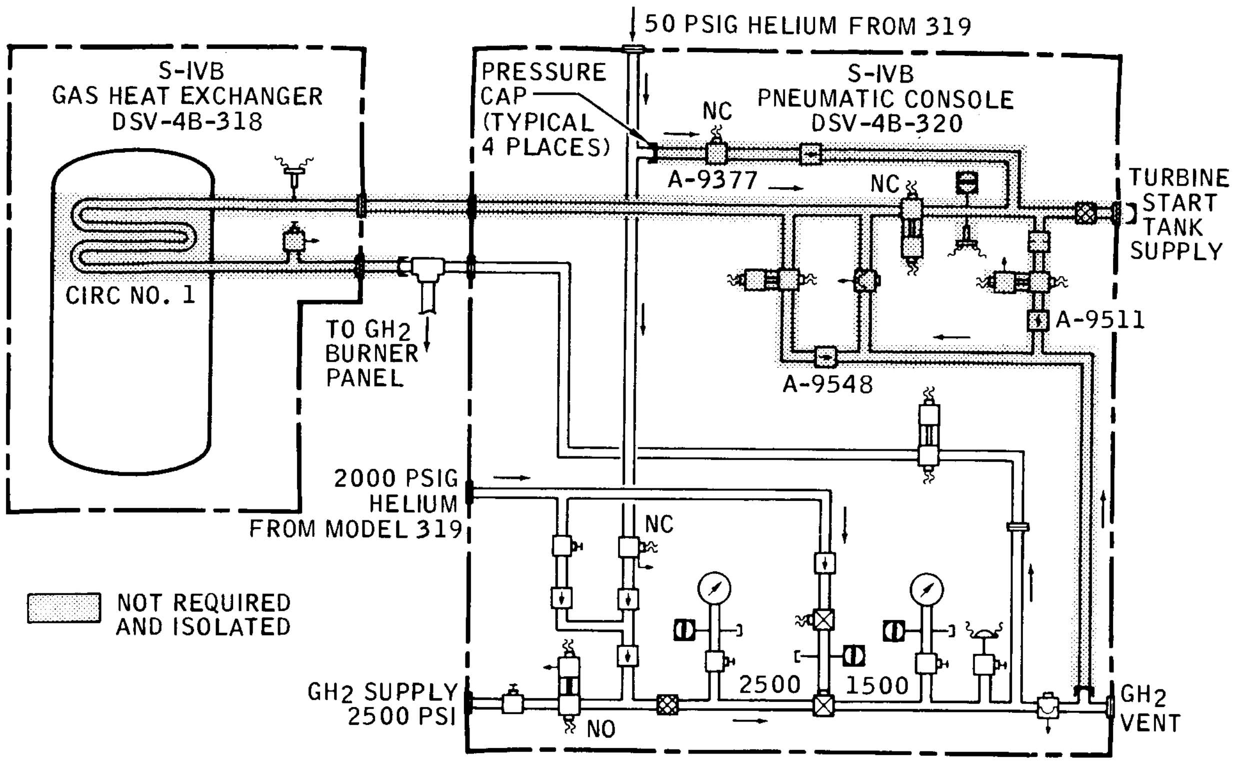


FIGURE 10.3.6-6. ENGINE TURBINE START TANK PRESSURE SYSTEM (STC)

10.3.6.1 (Continued)

At KSC the GH_2 and helium systems will be disconnected and capped off at the following locations (Figure 10.3.6-7):

a. 6,000 psig GH_2 inlet A-11714.

b. Turbine start bottle supply outlet port A-11708.

c. GH_2 vent outlet A-11709.

d. 50 psig helium system upstream of solenoid valves A-11925 and A-11964.

e. Valve control pressure at inlet ports of control solenoid valves A-11983, A-11915, and A11927.

f. Inlet port of 6,000 to 2,000 psig helium regulator A-12057.

At Huntington Beach the implementation of the J-2S engine does not affect the propulsion GSE. For maintenance procedure revisions, See Table 10.3.6-II.

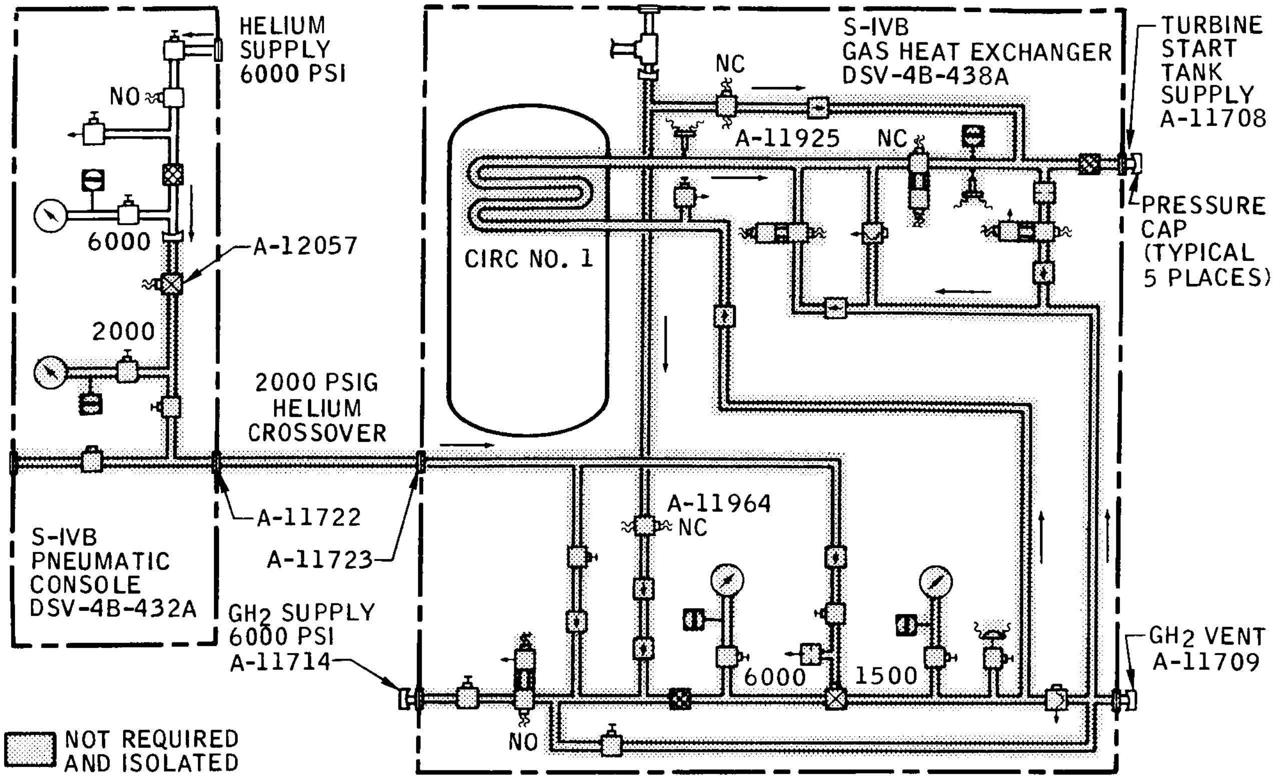


FIGURE 10.3.6-7. ENGINE TURBINE START TANK PRESSURE SYSTEM (KSC)

10.3.6.1 (Continued)

Engine Pneumatic Bottle Pressurization System

This system pressurizes the J-2 engine pneumatic control bottle with cryogenic helium gas. The engine control bottle provides helium for engine valve actuation and engine purge on the ground and during flight. Pressurization of the bottle requires a 3-step operation. The bottle is initially pressurized to 950 psia and then the pressure is dumped to purge and dry the bottle. The second step pressurizes the bottle to 1,450 psia to perform engine purges prior to propellant loading. The bottle pressure is again dumped at completion of engine purge. Final pressurization is accomplished with cryogenic helium gas. The implementation of the J-2S engine eliminates the need for cryogenic helium and hence the requirement of controlling cryogenic helium with this system. At Huntington Beach the implementation of the J-2S engine does not affect the propulsion GSE. For maintenance procedure revisions, see Table 10.3.6-II.

Thrust Chamber Jacket Purge and Chilldown System

This system chills the J-2 engine thrust chamber prior to vehicle liftoff. Chilldown is accomplished with cryogenic helium gas. Prior to chilldown, the system is purged with low pressure ambient temperature helium to remove all moisture contamination. Implementation of the J-2S engine eliminates the helium chill requirement.

TABLE 10.3.6-II. PROPULSION GSE MAINTENANCE PROCEDURE CHANGES

Model Number/Name	M.P. Dwg Number	Effort
Gas Heat Exch DSV-4B-318	1B33747	Delete pneumatic actuation press setup, para 6.1.2 and circuit No. 1 (GH_2) press. para 6.1.4. Revise depress. para 6.1.5 and securing of actuation press para 6.1.6.
Pneu Console DSV-4B-319	1B40841	No changes to maintenance procedure.
Pneu Console DSV-4B-320	1B40842	Revise filters etc., para 6.1.1, console-stage and sys press. for external leak-checks, para 6.3, console securing, para 6.4, regulators and valves operational check, para 6.5, controls test, para 6.6 and lubrication, para 6.7. Revise transducer leakchecks, Table 2.
Auto. Stage Ckout Pneu Console DSV-4B-321	1B37943	No changes to maintenance procedure.
Auto. Stage Ckout Pneu Console DSV-4B-321A	1B58201	No changes to maintenance procedure.
Pneu Console DSV-4B-432A	1B58203	Revise relief valve functional check, para 5.3, cleaning, para 6.1, pressurization, para 6.3, depressurization, para 6.4, R&R, para 9 and repair, para 10. Revise Age items and Table I.
Pneu Console DSV-4B-433A	1B58722	Revise pressurization, para 6.3 and depressurization 6.4 Age items and Table I.
Gas Heat Exch DSV-4B-438A	1B66388	Revise relief valve functional check, para 5.3, cleaning, para 6.1, pressurization, para 6.3, and depressurization, para 6.4 Age items and Table I.

10.3.6.1 (Continued)

Propulsion GSE at STC will be modified as follows, see Figure 10.3.6-8:

Eliminate the thrust chamber chilldown supply by disconnecting the cold helium supply line in Model 320 from inlet port of pneumatic A-9376 valve, and installing pressure caps on the inlet port and the disconnected tubing. At the facility interface plate, disconnect the thrust chamber chilldown supply line from the tee and install pressure caps on the tee and disconnected tubing. Disconnect the vent system downstream of check valve A-9551 and add pressure caps. Disconnect the 750 psi GN_2 valve control pressure from the inlet port of A-9362 and A-9513, and add pressure caps.

Propulsion GSE at KSC will be modified as follows, see Figure 10.3.6-9:

Eliminate the thrust chamber chilldown supply by disconnecting Circuit No. 2 cold helium supply plumbing at the upstream port of filter A-11919 in Model 438A, and install pressure plugs to the filter inlet port and plumbing. In Model 433A disconnect the tube assembly downstream of orifice A-11937 and install pressure plugs in the orifice and tube assembly. Disconnect the 750 psi GN_2 valve control pressure at the inlet port of solenoid A-11931 valve and install pressure plugs. Implementation of the J-2S engine does not affect Huntington Beach propulsion GSE. For maintenance procedure changes, see Table 10.3.6-II.

LOX Tank Prepressurization System

The LOX tank prepressurization system provides cryogenic helium gas to the stage LOX tank during the latter portion of the launch countdown. The system also maintains standby pressure on the LOX tank prior to LOX loading, and provides pressure to drain the tank in the event of an aborted launch or at the end of a countdown demonstration.

Implementation of the J-2S engine will affect this system in that ambient helium will be used exclusively. The LH_2 supply valve to Model 318 at STC or Model 438A at KSC will be in the closed position at all times.

LH_2 Tank Prepressurization System

The LH_2 tank prepressurization system performs the same function on the stage fuel tank as the LOX system performs on the LOX tank. The system supplies cryogenic helium to the LH_2 tank for pressurization during the latter portion of the launch countdown. The system also supplies tank standby pressure prior to loading, and pressure to drain the tank.

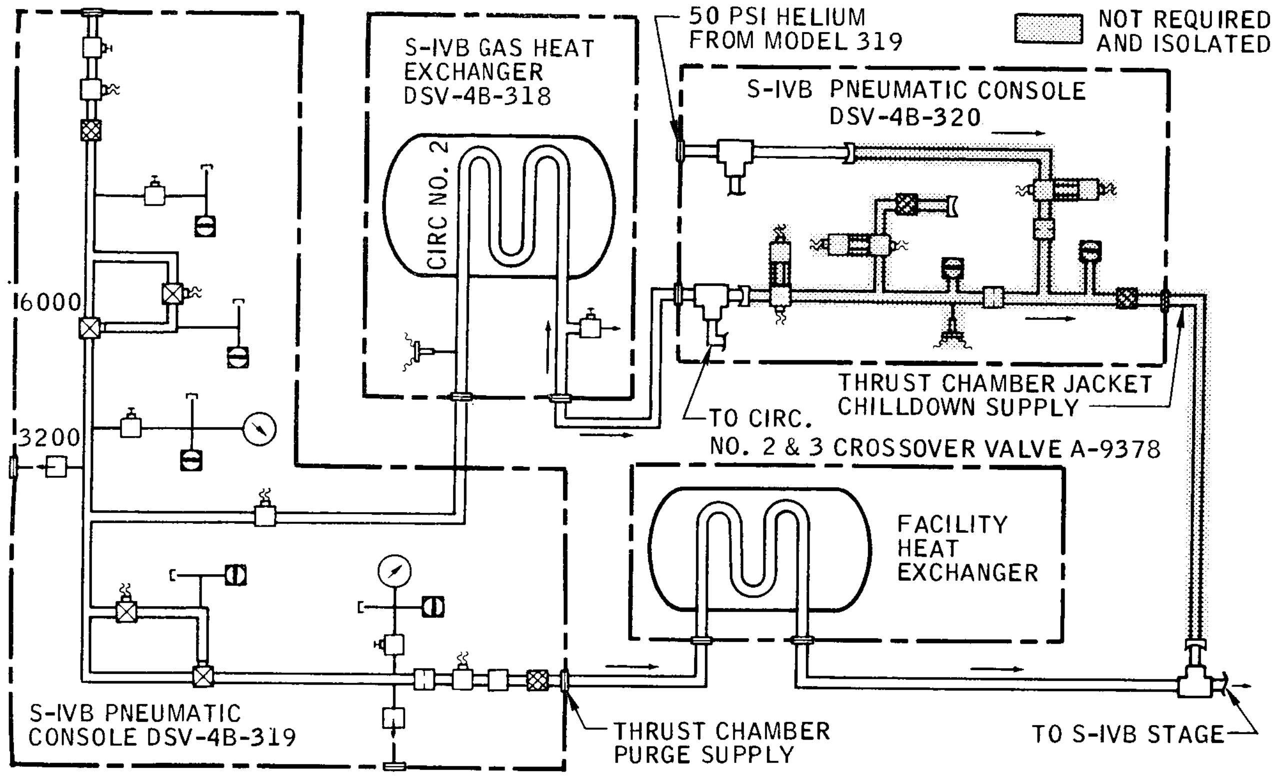


FIGURE 10.3.6-8. THRUST CHAMBER JACKET PURGE AND CHILLDOWN SUPPLY SYSTEM (STC)

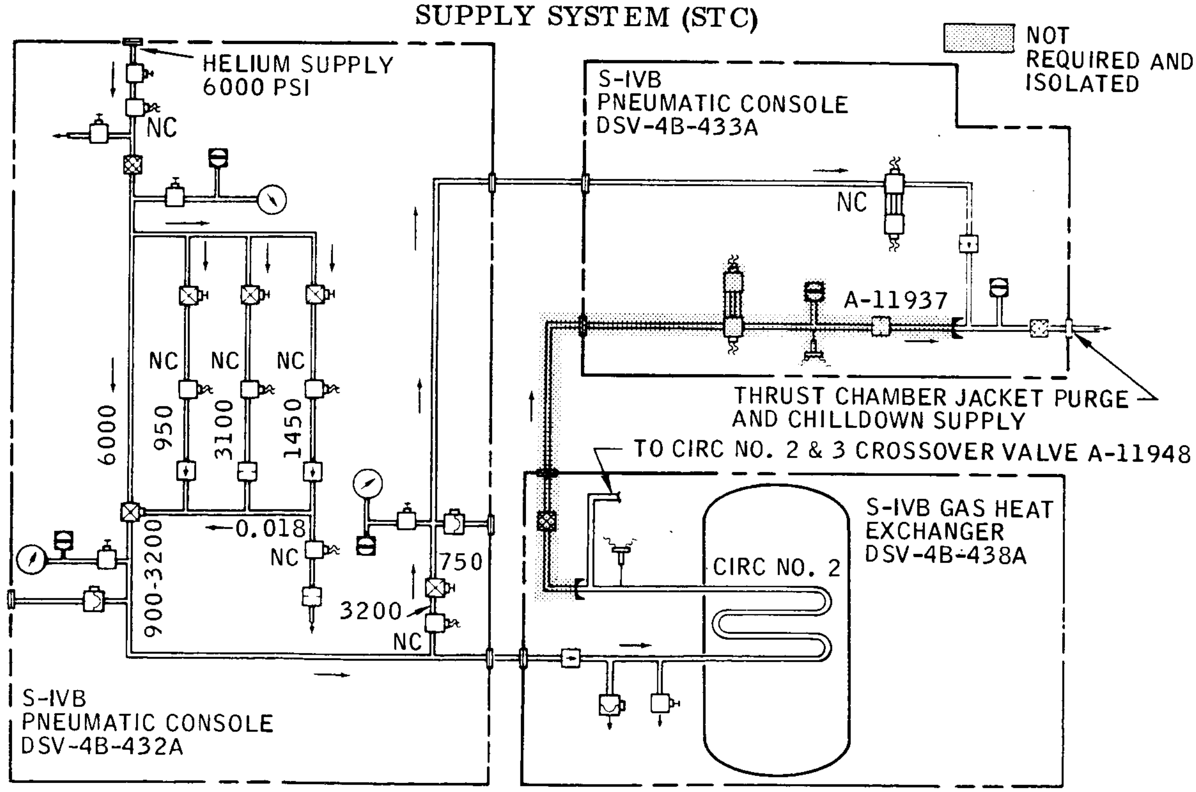


FIGURE 10.3.6-9. THRUST CHAMBER JACKET PURGE AND CHILLDOWN SUPPLY SYSTEM (KSC)

10.3.6.1 (Continued)

Implementation of the J-2S engine will affect this system in that ambient helium will be used exclusively. The LH_2 supply valve to Model 318 at STC or Model 438A at KSC will be in the closed position at all times.

APS Purge System

This system supplies gaseous nitrogen to the auxiliary propulsion system module propellant tanks and engine manifold to purge the propellant system of moisture in preparation for loading.

The system provides for circulation of propellant through the attitude control system propellant control assemblies to establish pressure and temperature requirements of the propellant prior to loading. Propellant is then supplied from the control assembly to the attitude control system module to load the tanks. The propellant is circulated through the attitude control system module tanks to ensure that the system is free of gas bubbles. The system has the capability for removal of a measured amount of propellant from the module. The system is designed to remove gas which has accumulated in the attitude control system propellant tank bladder after loading and prior to launch. The sight glass inlet tubing has been increased in diameter to improve sight glass response to actual liquid level within the bleed system. Implementation of the J-2S engine will not affect this system.

Oxidizer Dome Purge System

The purpose of this system is to supply an inert gas to purge the J-2S LOX dome and injector of moisture, contaminants, or LOX. At STC the inert gas is GN_2, and at KSC it is helium.

At STC, Model 320 may be used after revising orifice A-9958 to meet the J-2S engine purge flow requirements.

At KSC, Model 432A will be revised by removing the second stage preset 750 psig regulator A-11752 and adjacent plumbing (Figure 10.3.6-10) and replacing it with a dome-loaded regulator of the same basic setting, a loader regulator, and required plumbing in order to provide a regulator of adequate flow capability. A 750 psi helium system will be tapped off downstream of regulator A-11752. This 750 psi system requires a filter, solenoid valve, orifice, and plumbing to the added Model 432A interface. Facility plumbing must be added from this 432A interface to the stage aft umbilical interface plate. For maintenance procedure changes, see Table 10.3.6-II.

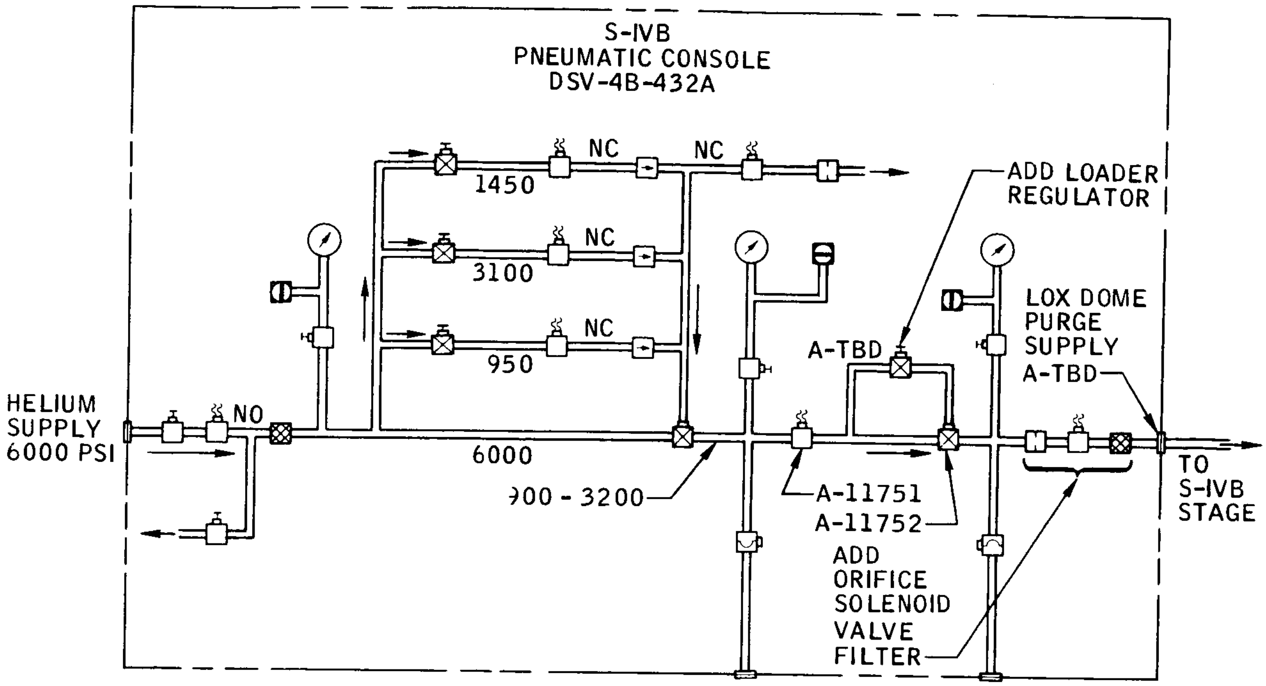


FIGURE 10.3.6-10. LOX DOME PURGE SYSTEM

10.3.6.2 Electrical GSE

The electrical GSE at MDAC Huntington Beach (HB) and STC consists of a digital computer controlled checkout system whose primary function is to verify the design integrity and operational capability of the complete S-IVB stage. A backup manual control capability is also provided to allow safing and/or shutdown of the stage in the event of failure in the automated equipment.

Each vehicle system or component that requires testing is separated or isolated, stimulated, and the response is measured and compared with a set of standards with allowable tolerances. If the measured response exceeds the allowable tolerance, further tests are made to determine whether the vehicle system or the automatic checkout system is at fault. Then an indication is given, pin-pointing the problem area.

The electrical GSE at KSC is also a digital computer controlled checkout system which has the same objectives as the system at HB and STC. The principal difference at KSC is that the backup manual control capability is much more extensive. The system is designed so that the vehicle checkout and launch could be conducted manually, if necessary.

The following discussion of GSE subsystems applies to HB, STC, and KSC. There will be an emphasis on HB and STC because the automatic checkout equipment is controlled by MDAC, whereas at the KSC Saturn V launch complex 39, the GSE is controlled by various contractors, principally General Electric.

10.3.6.2 (Continued)

Modifications in the GSE can be made with relative ease because patch panels have been provided to route commands, talkbacks, and measurements between the Automatic Checkout System (ACS) and the vehicle and mechanical GSE. Changes in the patch panels will impact a portion of the manual control and display capability. This refers to switches, lights or meters on control panels. Changes on the control panels can be economically handled by relabeling functions on the panels rather than removing or changing hardware.

Implementation of the J-2S engine into the ACS at HB and STC will affect the GSE models controlling the patch panels and the GSE manual control console models. Specifically, the GSE models affected at HB are the DSV-4B-184A & B patch panel models, the DSV-4B-233 remote pneumatic control console, and the DSV-4B-234 propulsion system digital control console.

At STC, the models affected are the DSV-4B-184C, D and E patch panel models, the DSV-4B-233A, B and C remote pneumatic control consoles, the DSV-4B-234 and 234A propulsion system control consoles, and the DSV-4B-648 engine safety cutoff rack assembly.

At KSC, the General Electric control panels affected are the 405A1 networks panel, the 406A1 recirculation panel, the 406A3 helium control panel, the 406A4 GH/GN control panel, the 406A5 engine test panel, the 406A6 engine prep panel, the 409A1 EBW and ordnance panel, and the events display panel.

KSC strip charts and recorders will be changed to remove measurements that will no longer be required and to add any new measurements. Deletion of measurements will allow for better arrangement of parameters displayed and changes to relocate the measurements on the more suitable recorders will be accomplished.

Modification of Launch Control Center (LCC) ESE panels to accommodate the J2-S will require rework of associated patch distributors. The rework will involve the removal of patch jumpers for all control and monitor functions that are deleted on the LCC panels. A deleted switch may result in up to six jumpers removed from the distributor. A deleted monitor function will normally require the removal of one to two patch jumpers for each indication removed. The addition of a switch function or indicator to a panel may require the addition of patch jumpers to route the signal to and from the panel.

There are three areas on the Launch Umbilical Tower (LUT) that will require modification. They are the LUT patch distributors, fwd and aft crossover distributors, and swing arm No. 6.

Repatching of the LUT patch distributor will be required to implement the J-2S change. The patch changes in these distributors will be more extensive than the changes to the LCC distributors. The changes will involve patch jumper removals in the various relay circuits associated with the control and monitor functions deleted. Jumpers will be added as required by any new monitoring requirements or interlocks that may be needed for the solid propellant turbine starter system.

Fwd and aft crossover distributors patching changes will be required to delete functions routed to the stage and GSE by these distributors. New monitoring requirements for the SPTS system will require patch jumper changes or additions.

Minor mods to the electrical umbilicals and pneumatic lines on swing arm No. 6 are required due to the J2-S engine systems.

10.3.6.2 (Continued)

The modifications to these manual control consoles, panels, and distributors are shown in detail in the following paragraphs in terms of vehicle system and propulsion GSE system modifications. As the J-2S engine implementation affects each of these systems, the resulting electrical GSE changes are specified. The systems under consideration are the electrical power system, the propulsion support system, the engine turbine start pressurization system, the chilldown system, the pneumatic control system, the auxiliary propulsion system, the hydraulic system, the propellant utilization system, the ordnance system, and the instrumentation and telemetry system.

With regard to mission requirements, the electrical GSE must necessarily be designed to accommodate any mission requirement since requirements may change at the last moment. Therefore, the GSE will accommodate either the LOR or the synchronous orbit missions without additional modification.

Electrical Power System

In the electrical GSE, the Vehicle External Power Racks, Model No. DSV-4B-134, provide a ground power source for all four 28 vdc vehicle power busses and also includes a 56 vdc supply. The impact of the J-2S engine on the stage has the overall effect of reducing the vehicle power requirement, therefore, the J-2S engine will not affect this GSE system.

Propulsion Support System

At HB, the propulsion support system is the Automatic Stage Checkout Pneumatic Console, Model No. DSV-4B-321, which controls the distribution of helium or nitrogen gas to meet the leak and functional checkout requirements of the stage and APS equipment during dry checkout exercises. The J-2S engine does not require a start bottle supply, therefore, the electrical GSE is affected in the following way:

Model DSV-4B-233

Remote Pneumatic Control Console-VCL 1 & 2 — Dwg 1A65727-1
Mainstage Checkout Panel (A2) — Dwg 1A78635-1

Delete	Via
Ind-Start bottle open	3 patchcord deletions
Ind-Start bottle closed	2 patchcord deletions
Switch-Start bottle supply valve	6 patchcord deletions

At STC, the propulsion support systems for VCL is the Pneumatic Console, Model No. DSV-4B-321A, while the system at the Beta complex is the Gas Heat Exchanger, Model No. DSV-4B-318, and the Pneumatic consoles, Model No. DSV-4B-319 and 320. These units are modified to eliminate the cold helium supply for thrust chamber chilldown and the cold hydrogen supply for the turbine start bottle. The start bottle supply is eliminated from Model 321A. The electrical GSE is affected as follows:

Model DSV-4B-233C

Remote Pneumatic Control Console-VCL — Dwg 1B52840-1
Mainstage Checkout Panel (A2) — Dwg 1A78635-503

Delete	Via
Same as Model 233 at HB	

10.3.6.2 (Continued)

Models DSV-4B-233A and B

Remote Pneumatic Control Console-Beta 1	Dwg 1A78634-1
Remote Pneumatic Control Console-Beta 3	Dwg 1B39212-1
Vehicle Supply Panel (A2A2)	Dwg 1A78635-501

Delete	Via
Ind-start tank GH_2 Fill-open	3 patchcord deletions
Ind-start tank GH_2 Fill-closed	3 patchcord deletions
Switch-start tank GH_2 fill supply valve	11 patchcord deletions
Ind-start tank GH_2 vent open	3 patchcord deletions
Ind-start tank GH_2 vent closed	3 patchcord deletions
Switch-start tank GH_2 vent valve	11 patchcord deletions
Ind-cold GH_2 vent open	3 patchcord deletions
Ind-cold GH_2 vent closed	3 patchcord deletions
Switch-cold GH_2 supply vent valve	6 patchcord deletions
Ind-thrust chamber C/D supply open	3 patchcord deletions
Ind-thrust chamber C/D supply closed	3 patchcord deletions
Switch-thrust chamber C/D supply valve	7 patchcord deletions
Ind-thrust chamber supply vent open	3 patchcord deletions
Ind-thrust chamber supply vent closed	3 patchcord deletions
Switch-thrust chamber supply vent valve	5 patchcord deletions
Meter-eng start tank pressure	3 patchcord deletions

Vehicle Supply Valves Panel (A1A3) Dwg 1A78638-1

Delete	Via
Ind-start tank purge supply open	3 patchcord deletions
Ind-start tank purge supply closed	3 patchcord deletions
Switch-start tank purge supply valve	8 patchcord deletions

Various measurements are also deleted from the propulsion support system at STC which cause the following patchcords to be deleted from the patch panel models:

Models DSV 4B-184C and D

Elect C/O Accessory Kit-Beta 1	Dwg 1B44043-1
Elect C/O Accessory Kit-Beta 3	Dwg 1B44047-1

Delete	Via
Ind-Eng start tank supply press.	18 patchcord deletions
Ind-Eng start tank supply temp	15 patchcord deletions
Ind-thrust cham purge & C/D temp	15 patchcord deletions
Ind-thrust cham purge & C/D OI press.	12 patchcord deletions
Ind-GH_2 Ht-Ex-1 cold output temp	15 patchcord deletions

10.3.6.2 (Continued)

At KSC, the propulsion support system consists of the pneumatic consoles, Model No. DSV-4B-432A and 433A, and the Gas Heat Exchanger, Model No. DSV-4B-438A. These consoles are similar in function to models 318, 319 and 320 at STC and therefore have similar deletions. The electrical control panels are affected in the following manner:

Helium Control Panel (406A3)

Delete:

Ind-GH bank purge open
Ind-GH bank purge closed
Switch-GH bank purge supply valve

GH/GN Control Panel (406A4)

Delete:

Ind-GH 6,000 PSI supply valve closed
Ind-GH 6,000 PSI supply valve open
Switch-GH 6,000 PSI supply valve
Meter-GH 6,000 PSI ambient sup press.
Meter-GH 1,500 PSI ambient sup press.
Ind-GH cold ht-ex-1 vent open
Ind-GH cold ht-ex-1 vent closed
Switch-GH cold ht-ex-1 vent valve
Switch-Ht exchanger outlet temp ckt No. 1

Engine Preparation Panel (406A6)

Delete:

Ind-eng start tk supply valve open
Ind-eng start tk supply valve closed
Switch-eng start tk supply valve
Ind-T/C chilldown supply open
Ind-T/C chilldown supply closed
Switch-T/C chilldown supply valve
Ind-start tk purge supply valve open
Ind-start tk purge supply valve closed
Switch-start tk purge supply valve
Ind-start tk supply vent open
Ind-start tk supply vent closed
Switch-start tk supply vent valve

10.3.6.2 (Continued)

Engine Turbine Start Tank-Pressurization System

The J-2 start tank system is used to spin the fuel turbine for the engine start sequence. The J-2S engine uses a solid propellant turbine start (SPTS) technique, thereby eliminating the start bottle and need for ground pressurization, and all associated valves and controls. The interface between this system and the electrical GSE are the umbilical lines. The deletions in the umbilical functions and their effect on the electrical GSE models are listed below:

At HB

Model DSV-4B-234

Prop Sys Digital Control Console VCL 1 & 2	Dwg 1A65728-1
Mainstage Manual Control Panel (A2)	Dwg 1A82231-503

Delete	Via
Ind-start tank vent open-cmd.	3 patchcord deletions
Switch-start tank vent pilot valve open-close	3 patchcord deletions
Ind-start tank emer. vent cmd.-on	2 patchcord deletions
Switch-start tank emer. vent valve open-close	5 patchcord deletions

At STC

Model DSV-4B-234

Propulsion System Control Console - VCL	Dwg 1A65728-1
Mainstage Manual Control Panel (A2)	Dwg 1A82231-503

Delete	Via
Same functions as model 234 at HB	

Models DSV-4B-233 A and B

Remote Pneumatic Control Console-Beta 1	Dwg 1A78634-1
Remote Pneumatic Control Console-Beta 3	Dwg 1B39212-1
Vehicle Supply Panel (A2)	Dwg 1A78635-501

Delete	Via
Ind-start tank dump open	3 patchcord deletions
Ind-start tank dump closed	4 patchcord deletions
Switch-start tank vent valve	5 patchcord deletions
Ind-start tank emer dump stand	2 patchcord deletions
Ind-start tank emer dump stage	4 patchcord deletions
Switch-start tank emer dump valve	15 patchcord deletions

10.3.6.2 (Continued)

At KSC

Engine Test Panel (406A5)

Delete:

Ind-comp test st tk discharge valve open
Ind-comp test st tk discharge valve closed
Ind-start tank sol energized
Switch-comp test start tank disch valve

Engine Preparation Panel (406A6)

Delete:

Switch-st tank vent pilot valve
Switch-st tank emer dump valve
Meter-st tank temperature
Meter-st tank pressure

Recirculation Chilldown System

The recirculation chilldown system maintains subcooled liquid propellant at the J-2 pump inlets to prevent a pump stall during engine start. The idle mode capability of the J-2S makes the entire chilldown system unnecessary. Items such as the chilldown pumps, chilldown inverters, prevalves, shutoff valves, frequency converter, and transducers are eliminated. The deletions appear to the GSE as umbilical pin function deletions which require many patching deletions and control panel changes. These changes are outlined below.

The patchcord deletions under models DSV-4B-184A and B are those which do not affect any manual control panels but affect interfaces with the computer and various digital or analog display devices:

At Huntington Beach

Models DSV-4B-184 A and B

Elect C/O Accessory Kit VCLNo. 1 Dwg 1B44042-1
Elect C/O Accessory Kit VCLNo. 2 Dwg 1B44044-1

Delete	Umbilical Pin	Via
Meas-LOX C/D pump diff. pressure	404W2J1-1, 2, 3	6 patchcord deletions
Ind - LH_2 C/D valve open	404W2J1-18	2 patchcord deletions
Ind - LOX C/D valve closed	404W2J1-34	2 patchcord deletions
Ind - LH_2 C/D pump diff press	411W1J1-1, 2, 3	6 patchcord deletions

10.3.6.2 (Continued)

Delete	Umbilical Pin	Via
Ind - LOX C/D pilot relay reset	404W15J2-N	2 patchcord deletions
Ind - LH_2 C/D pump pilot relay reset	404W15J2-P	2 patchcord deletions
Ind - LOX C/D pump inv pwr on	404W15J2-Q	2 patchcord deletions
Ind - LH_2 C/D pump inv pwr on	404W15J2-V	2 patchcord deletions

Model DSV-4B-234

Propulsion System Control Console Dwg 1A65728-1
Mainstage Manual Control Panel Dwg 1A82231-503

Delete	Via
Ind - LH_2 pre-valve closed	2 patchcord deletions
Ind - LH_2 pre-valve open	2 patchcord deletions
Ind - LOX pre-valve open	2 patchcord deletions
Ind - LOX pre-valve closed	2 patchcord deletions
Switch-LH_2 and LOX pre-valves	5 patchcord deletions
Ind - LH_2 and LOX C/D shutoff closed	3 patchcord deletions
Ind - LH_2 and LOX C/D shutoff open	2 patchcord deletions
Switch-LH_2 and LOX C/D shutoff valves	5 patchcord deletions

At the Sacramento Test Center, the following changes will be made:

Models DSV-4B-184C, D, and E

Elect C/O Accessory Kit-Beta 1 Dwg 1B44043-1
Elect C/O Accessory Kit-Beta 3 Dwg 1B44047-1
Elect C/O Accessory Kit-VCL Dwg 1B44048-1

Delete	Via
Same misc umbilical functions as the model 184 at Huntington Beach	

Model DSV-4B-234A

Propulsion System Control Console Beta 1 and 3 Dwg 1B63464-1
Mainstage Propulsion Manual Control Panel (A2) Dwg 1A82231-501

Delete	Via
Ind - LOX pre-valve open	3 patchcord deletions
Ind - LOX pre-valve closed	3 patchcord deletions
Ind - LH_2 pre-valve open	3 patchcord deletions
Ind - LH_2 pre-valve closed	3 patchcord deletions
Switch-LH_2 and LOX pre-valves	9 patchcord deletions
Ind - LH_2 and LOX C/D shutoff open	4 patchcord deletions
Ind - LH_2 and LOX C/D shutoff closed	5 patchcord deletions
Switch-LH_2 and LOX C/D shutoff valve	9 patchcord deletions

10.3.6.2 (Continued)

At Kennedy Space Center, the chilldown system deletions will have the following effects:

Recirculation Panel (406A1)

Delete entire panel composed of:

Ind -LOX prevalve closed
Ind -LOX prevalve open
Switch-LOX prevalve
Ind -LH prevalve closed
Ind -LH prevalve open
Switch-LH prevalve emer close
Ind -LOX C/D inv pwr on
Ind -LOX C/D pump ready to start
Switch-LOX C/D pump
Ind -LH C/D inv pwr on
Ind -LH C/D pump ready to start
Switch-LH C/D pump
Ind -LOX C/D valve closed
Ind -LOX C/D valve open
Switch-LOX C/D valve
Ind -LH C/D valve closed
Ind -LH C/D valve open
Switch-LH C/D valve emer close
Meter-LOX pump CAV press
Meter-LOX flow rate
Meter-LH flow rate

Networks Panel (405A1)

Delete:

Ind -LOX C/D relay reset
Ind -LH_2 C/D relay reset
Ind -LH_2 C/D inv pwr off
Ind -LOX C/D inv pwr off
Ind -Chilldown pumps safe

Engine Preparation Panel (406A6)

Delete:

Meter-Eng thrust chamber jacket temp

Events Display Panel

Delete:

Ind -prevalve emer close cmd
Ind -SIVB LOX chilldown
Ind -SIVB LH chilldown
Ind -LOX chilldown inv pwr on
Ind -LH chilldown inv pwr on

10.3.6.2 (Continued)

Pneumatic Control System

The stage pneumatic control system provides regulated gas for certain purge and control operations. Certain purge and control functions are not required on the J-2S engine, such as the start tank vent valve control, the LOX chilldown pump purge control, and the engine pump purge control. These functions were included in the discussion of the start tank system and the chilldown system.

Ordnance System

The ordnance system consists of many EBW firing units and initiators used throughout the stage to fire explosive charges or solid propellant motors. Implementation of the J-2S engine eliminates the requirement for the solid propellant ullage control rockets, thereby deleting a total of 4 EBW firing units and initiators. However, the J-2S engine presents a new requirement for EBW firing units associated with the solid propellant turbine starters. There will be 2 or 3 SPTS gas generators furnished with the engine and each unit uses 2 EBW firing units and initiators. Therefore, the impact on the electrical GSE will be an exchange of ullage rocket ignition and jettison functions with SPTS functions.

Another change resulting from the SPTS is the deletion of the gas generator which presently drives the LOX and LH_2 turbines after engine start. This change will result in further exchanges of gas generator functions with SPTS functions. At Huntington Beach, the following GSE models are affected

Models DSV-4B-184A and B

Elect C/O Accessory Kit - VCL No. 1 Dwg 1B44042-1
Elect C/O Accessory Kit - VCL No. 2 Dwg 1B44044-1

Delete	Umbilical Pin	Via
Cmd-EBW Ullage rkt fir unit disable	404W1J1-48	3 patchcord deletions
Ind-EBW Ullage rkt fir unit enable	404W2J1-51	2 patchcord deletions
Ind-EBW Ullage rkt relay reset	404W4J1-58	2 patchcord deletions
Meas-Eng spark gas gen #1 sig	404W2J1-5	No Change
Meas-Eng spark gas gen #2 sig	404W2J1-8	No Change
Meas-Eng gas gen valve closed	404W4J1-16	No Change

Add	Umbilical Pin	Via
Ind-Monitor #1 SPTS ready	404W2J1-42	2 patchcords added
Ind-Monitor #2 SPTS ready	TBD	2 patchcords added
Ind-Monitor #3 SPTS ready	TBD	2 patchcords added
Meas-Monitor #1 SPTS #1 EBW	404W2J1-5	No Change
Meas-Monitor #1 SPTS #2 EBW	404W2J1-8	No Change
Meas-Monitor #2 SPTS #1 EBW	TBD	9 patchcords added
Meas-Monitor #2 SPTS #2 EBW	TBD	9 patchcords added
Meas-Monitor #3 SPTS #1 EBW	TBD	9 patchcords added
Meas-Monitor #3 SPTS #2 EBW	TBD	9 patchcords added

10.3.6.2 (Continued)

At Sacramento Test Center, the following changes will be made.

Models DSV-4B-184C, D, and E

Elect. C/O Accessory Kit - Beta 1	Dwg 1B44043-1
Elect. C/O Accessory Kit - Beta 3	Dwg 1B44047-1
Elect. C/O Accessory Kit - VCL	Dwg 1B44048-1

Add or Delete	via
Same as the model 184 at Huntington Beach	

Model DSV-4B-648

Engine Safety Cutoff Set Rack Assy Dwg 1B49100-1

The Rocketdyne furnished gas generator temperature high-low cutoff panel assembly (G1047) will be deleted entirely. In its place will be a SPTS monitor panel which will either be furnished by Rocketdyne or built by MDAC.

At Kennedy Space Center, the following changes will be made:

EBW and Ordnance Panel (409A1)

Reidentify:

Title "Ullage Rkt Ignition" to "SPTS #1 EBWS"
Title "Ullage Rkt Jettison" to "SPTS #2 EBWS"
All "pulse sensor fired" indications to "SPTS #1 (or 2) fired."
"Pilot relays reset" indication to "SPTS Armed"

Add:

Two meter circuits for "SPTS #3 EBWS"
Three inds "SPTS #1 Ready," "SPTS #2 Ready," and "SPTS #3 Ready."
Two "SPTS #3 Fired" indications

Delete:

Ind-ullage rkt fir units enabled
Switch-ullage rkt fir units enable-disable

Events Display Panel

Delete:

Ind-Ullage rkt fir unit enabled
Ind-Ullage rkt ignition fir unit #1 fired
Ind-Ullage rkt ignition fir unit #2 fired
Ind-Ullage rkt jettison fir unit #1 fired
Ind-Ullage rkt jettison fir unit #2 fired

10.3.6.2 (Continued)

Auxiliary Propulsion System

Since the J-2S engine has an idle-mode capability, the 72-pound thrust ullage engines are no longer required. The effect on the electrical GSE is minimal since there is only one umbilical talkback involved. This impact is shown below.

At Huntington Beach and Sacramento Test Center

Models DSV-4B-184A, B, C, D, and E

Elect. C/O Accessory Kit

Delete	Umbilical Pin	via
Ind-Ullage relay reset 70 pound	404W4J1-35	2 patchcords deleted in each area

At Kennedy Space Center, one monitor is affected

APS Launch and Monitor Panel (419A1)

Delete:

Ind-70 lbs thrust ullage eng. relay reset

Hydraulic System

Engine gimbaling is accomplished with hydraulic actuator assemblies connected to an engine-driven hydraulic pump. Implementation of the J-2S engine will necessitate a new pump assembly but this change will not affect the electrical GSE in any way.

Propellant Utilization System

The propellant utilization system measures the quantity of propellant remaining in each tank and controls the engine mixture ratio according certain pre-programmed levels to achieve maximum stage payload and also to achieve simultaneous depletion of propellants. This system also provides signals to the engine cutoff logic for cutoff at certain minimum propellant levels.

The J2-S engine has the capability of running to LOX depletion thereby eliminating the requirement for the LOX depletion cutoff logic circuitry. This will cause patching changes at Huntington Beach and Sacramento Test Center and control panel changes at Kennedy Space Center.

10.3.6.2 (Continued)

At Huntington Beach and Sacramento Test Center

Models DSV-4B-184A, B, C, D, and E

Elect C/O Accessory Kits

Delete	Umbilical Pin	via
Ind-LOX point level sens #1 wet cond	404W2J1-38	2 patchcord deletions
Ind-LOX point level sens #2 wet cond	404W2J1-37	2 patchcord deletions
Ind-LOX point level sens #3 wet cond	404W2J1-36	2 patchcord deletions
Cmd-sim LOX pt lvl sens #1 wet cond	404W3J1-8	2 patchcord deletions
Cmd-sim LOX pt lvl sens #2 wet cond	404W3J1-7	2 patchcord deletions
Cmd-sim LOX pt lvl sens #3 wet cond	404W3J1-14	2 patchcord deletions
Ind-LOX Point lvl sens #4 wet cond	404W4J1-57	2 patchcord deletions
Cmd-Sim LOX pt lvl sens #4 wet cond	404W1J1-55	5 patchcord deletions

At Kennedy Space Center

Engine Test Panel (406A5)

Delete:

Ind-LOX sensor No. 1 wet
Ind-LOX sensor No. 2 wet
Ind-LOX sensor No. 3 wet
Switch-simulate LOX sens #1 wet
Switch-simulate LOX sens #2 wet
Switch-simulate LOX sens #3 wet
Switch position "LOX Sensor #4 sim wet"

Reidentify:

Ind. "LOX/LH Sensor #4 wet" to "LH Sensor #4 wet"

Events Display Panel

Delete:

Ind-LOX cutoff sens No. 1 wet
Ind-LOX cutoff sens No. 2 wet
Ind-LOX cutoff sens No. 3 wet
Ind-LOX cutoff sens No. 4 wet

<u>Instrumentation and Telemetry System</u>

The instrumentation and telemetry system transmits all stage measurements to a Digital Data Acquisition Station (DDAS) for decommutation. Then the entire data field is stored in the computer memory so that any or all measurements may be printed out for analysis. Measurement changes resulting from the J-2S engine will not have any hardware effect on this part of the electrical GSE.

10.3.6.2 (Continued)

There are a few measurements that are routed through the digital-to-analog converters in the DDAS. These measurements are then made available for oscillograph display or control panel indications. Any changes in these measurements will result in hardware changes such as patchcord deletions or monitor panel changes. The vast majority of measurements associated with the J-2S engine implementation are not routed to graphs or panels, particularly at HB and STC.

The impact of these measurement changes on the electrical GSE is shown below.

At Huntington Beach, there will be only 3 measurement exchanges and no hardware changes.

Exchange K6 "ignition phase control solenoid energized indication"
with TBD "idle mode control solenoid energized indication."

Exchange K11 "gas generator spark system ind"
with TBD "SPTS initiated ind"

Exchange K96 "start tank discharge control ind"
with TBD "mainstage start control sol on"

At STC, there will be measurements deleted at Beta 1 and 3 which affect patchcords only.

Models DSV-4B-184C and D

Elect. C/O Accessory kit - Beta 1 — Dwg 1B44043-1
Elect. C/O Accessory kit - Beta 3 — Dwg 1B44047-1

Delete	via
D218 LH_2 chilldown pump diff press	3 patchcord deletions
D219 LOX chilldown pump diff press	3 patchcord deletions
F4 LOX circ pump flow rate	6 patchcord deletions
F5 LH_2 circ pump flow rate	6 patchcord deletions

At KSC, there will be four monitor panels affected. These functions have been covered in the previous discussions on the chilldown system, start bottle system, and the ordnance system and are summarized below.

Recirculation Panel (406A1)

Delete:

D103	meter - LOX pump cav pressure
F4	meter - LOX flow rate
F5	meter - LH flow rate

10.3.6.2 (Continued)

Engine Test Panel (406A5)

Delete:

K6	ind-ignition phase cont sol energ
K11	ind-gas gen spark sys on
K96	ind-start tank disch sol energ
K105	ind-eng pump purge press not high
K117	ind-gas gen valves open
K122	ind-start tank disch valve open
K124	ind-LOX turbine bypass valve open
K125	ind-LOX turbine bypass valve closed

Engine Preparation Panel (406A6)

Delete:

C6	meter-eng start tank temp
C7	meter-eng helium cont bottle temp
C199	meter-eng thrust cham jacket temp
D17	meter-eng start tank press
D19	meter-eng helium cont bottle press
D50	meter-eng pump purge press

EBW and Ordnance Panel (409A1)

Delete:

K149	ind-Ullage rkt jtsn fir unit #1 pulse sens fired
K150	ind-Ullage rkt jtsn fir unit #2 pulse sens fired
K176	ind-Ullage rkt ign fir unit #1 pulse sens fired
K177	ind-Ullage rkt ign fir unit #2 pulse sens fired
M64	meter-Ullage rkt ign fir unit #1 charge voltage
M65	meter-Ullage rkt ign fir unit #2 charge voltage
M66	meter-Ullage rkt jtsn fir unit #1 charge voltage
M67	meter-Ullage rkt jtsn fir unit #2 charge voltage

Kennedy Space Center Test Control Center Low Bay Panel Modifications

The Test Control Center (TCC) low bay panel modifications are as follows:

Propellant Level Monitor Panel

Remove Indications:	Remove Switches:
LOX No. 1 Wet	LOX No. 1 Simulate Wet
LOX No. 2 Wet	LOX No. 2 Simulate Wet
LOX No. 3 Wet	LOX No. 3 Simulate Wet

10.3.6.2 (Continued)

Engine Test Panel

Remove Meters and Circuits:

Start Tank Press.
Pump Seal Purge

Remove Switches and Function:

Spark System Test (GG Spark)
Start Vent Open - Close
Start Tk. Emer. Vent Open - Close

Change Switch Nomenclature and Function:
STDV to Mainstage Start Solenoid

Remove Indicators:

Fuel Bleed Closed
LOX Bleed Closed
Discharge Valve Open
Discharge Valve Closed
Start Tk. Solenoid Energized
LOX TBV Open
LOX TBV Closed
GG Valves Open
GG Valves Closed
GG Spark System ON
GG No. 1 Spark OK
GG No. 2 Spark OK

Add Indicators:
Mainstage Start Solenoid Energized

Stage Pressure Panel

Remove Switch and Function:

Prevalves Close - Open
Chilldown Valves Close - Open

Remove Indicators:

Prevalves LH Closed
Prevalves LH Open
Prevalves LOX Closed
Prevalves LOX Open
Chilldown Valves LH Closed
Chilldown Valves LH Open
Chilldown Valves LOX Open
Chilldown Valves LOX Closed

Distributors

Repatch Pwr. Distributor (802-463A4)
Repatch Control Distributor (802-420A1)
Minor repatching is required to delete functions no longer required to support propulsion check in low bay.

10.3.7 Systems Test

In utilizing the J-2S engine, several major modifications to both S-IVB stage hardware and operation are planned. These changes lead to the questions of whether battleship, acceptance, and flight tests are necessary, and if so, what type should be performed and to what extent.

A summary of major changes follows:

- Chilldown will be performed by idle mode operation, thereby permitting removal of the recirculation chilldown system pumps, motors, prevalves, shutoff valves, inverters, converters, and instrumentation currently required to assure adequate propellant NPSH at engine start. Chilldown will be accomplished by a flow of propellants which are burned in the engine at a fraction of mainstage Isp.

- Propellant unsettling during separation can be tolerated during the J-2S start transient since poor quality propellants may be ingested by the J-2S at engine start. Propellant resettling and run conditions are expected to be regained by the time mainstage NPSH is required, therefore, the existing solid ullage motor rocket system will be deleted. During restart operations, idle mode chilldown provides settling thrust and APS ullaging is not necessary.

- Propellant slosh control will be accomplished after mainstage operation by a thrust decrease from mainstage thrust to idle mode and then to shutdown. This replaces present APS ullaging for this purpose, and along with the deletion of APS ullaging prior to restart, APS ullage engines can be removed.

- As a result of the J-2S LOX depletion capability, the LOX low level sensor system will be removed. This includes the sensors, conditioners, amplifiers, and other associated electrical equipment.

- The rapid propellant dump capability of the J-2S presents a convenient means of propellant tanks passivation or safing in orbit. This feature may prove useful in rendezvous and manned orbiting applications.

Four phases of testing are planned to develop and establish the S-IVB/J-2S stage as a replacement stage for the S-IVB/J-2. These phases consist of (1) chilldown development tests to be conducted at AEDC, (2) the S-IVB/J-2S system development tests to be conducted at MSFC, (3) the S-IVB/J-2S stage acceptance tests at STC, and (4) the flight verification tests at KSC.

10.3.7.1 Battleship Testing

Since the J-2S/S-IVB will be man rated, it must exhibit a high degree of reliability and therefore high confidence in design changes. Presently, no vehicle has been set aside solely for flight testing; the first launch will be for payload purposes. A successful flight program must be assured from the onset,

10.3.7.1 (Continued)

for a failure of a single Saturn V vehicle, if attributed to J-2S implementation, could obviate the program advantages this engine has to offer. Stage changes which affect reliability or which might degrade performance will be regarded with caution. There are several areas which could compromise or affect these objectives and considerations.

First, several basic analytical parameters are unknown or vary to such a degree that payload performance degradation may result from the conservative approaches required to assure mission success. The actual hardware chill requirements have not been thoroughly identified at this time and are not available for a rigorous idle mode chill optimization. Stage and engine hardware temperatures in orbit are difficult to predict, being dependent on stage orientation and mission profile. Tank pressure variations, which can result from relief setting tolerances alone, create numerous idle mode feed pressure combinations and therefore different chilldown durations. Second, simplified models have been used for the chilldown analysis, i. e., lumped masses for the engine and stage ducting, choked flow, simple resistance equations, arbitrarily chosen percents of engine mass to be chilled, etc.

Based on these models and predictions, some extensive stage modifications are proposed to exploit J-2S capabilities. If no testing is performed prior to flight, confidence cannot be expected to be very high and system reliability could possibly fall below that presently established. Therefore, a ground systems test program (a battleship program) is recommended to provide information in three vital areas. First, sufficient test data must be obtained to verify analytical predictions or to change them. Second, operational limits for safe stage performance must be established. Third, test and launch procedures must be developed in a manner which will not jeopardize flight stages during acceptance testing and launch.

The S-IVB/J-2S stage Battleship test program has been designed to integrate the S-IVB stage and J-2S engine into a compatible flight worthy S-IVB system from the initial design phase through complete engine/stage development.

The following paragraphs discuss the Battleship test planning and the test objectives which must be completed to provide a high degree of confidence and reliability in the final integration of the J-2S engine into the S-IVB stage. In the development of the objectives, emphasis has been placed on confidence level elevation and reliability gains while keeping the overall testing costs at a minimum.

Battleship Test Objectives

The testing objectives have been divided into three major groups for the Battleship tests; those of concept verification, operational limit establishment, acceptance test and launch requirements. This information will also serve to

10.3.7.1 (Continued)

augment the perpetual reliability failure analysis and Flight Information and Operations Report (FIOR) efforts when the S-IVB/J-2S stage becomes operational. Table 10.3.7-I lists the objectives applicable to each group. Each test objective is presented in the following pages, and contains a justification for the testing, a description of the test required, and the type of information to be obtained. This information should not be considered as the only data that is to be obtained, nor does it imply that the measurements listed will be the only ones required. However it does serve to indicate the magnitude, scope and importance of the objective.

The objectives and justification for each are considered preliminary at this time and will be augmented with additional objectives and/or justifications as the program is further defined.

The test requirements have been defined for both AEDC and MSFC S-IVB/J-2S Battleship tests as noted under the "test definition" section.

OBJECTIVE B-1: Determine Engine Pneumatic System characteristics for a Nominal LOR Mission and a Nominal Synchronous Orbit Mission.

This objective will require the following type of information:

a. Determine helium consumption (mass) for start, steady state, and cutoff, for each of the burns required for the above missions and the following phases of operation:

   1. Idle mode start
   2. Idle mode operation
   3. Mainstage starts
   4. Mainstage operation
   5. Mainstage cutoff (mainstage to idle mode)
   6. Engine cutoff (mainstage to zero thrust)
   7. Idle mode cutoff

b. Determine helium residuals available for propellant dumping and passivation.

c. Establish sequence, procedures, and time required to blow down (safe) the helium sphere after last burn cutoff (max, min and nominal blowdown history).

d. Establish minimum allowable pneumatic system pressures for successful operation of the phases listed under step a.

TABLE 10.3.7-I. J-2S ENGINE/S-IVB STAGE DEVELOPMENT TEST OBJECTIVES

Test	Objective	Concept Verification	Operational Limit Establishment	Acceptance Test & Launch Requirements
B-1	Determine Eng Pneu Syst Characteristics		X	
B-2	Demonstrate SPTS (2 & 3 Starts)	X	X	
B-3	Demonstrate Tapoff Turbine Concept	X		
B-4	Evaluate J-2S Mainstage Performance over PU Range		X	
B-5	Determine J-2S Eng Start Transients (I/M to MS)		X	
B-6	Determine J-2S Eng Cutoff Transients		X	
B-7	Evaluate Eng I/M for Cutoff Slosh Control	X	X	
B-8	Demonstrate LOX Depletion Cutoff	X	X	
B-9	Determine First Burn I/M Chilldown Time and Performance	X	X	
B-10	Determine Propellant Chilldown Press and Temp Limits for First Burn Idle Mode		X	
B-11	Determine Restart I/M Chilldown Time and Performance	X	X	
B-12	Determine Propellant Chilldown Press and Temp Limits for Restart Idle Mode		X	

TABLE 10.3.7-I. (Continued)

Test	Objective	Concept Verification	Operational Limit Establishment	Acceptance Test & Launch Requirements
B-13	Simulate Simultaneous Restart Idle Mode and Ambient Repress		X	
B-14	Determine LOX Tank Pressurization System Performance		X	
B-15	Determine LH_2 Tank Pressurization System Performance		X	
B-16	Establish Eng Cont He Sphere Loading and Venting Procedures			X
B-17	Demonstrate Adequate TC Jacket and Turbopump Purge			X
B-18	Demonstrate Adequate LOX Dome Purge			X
B-19	Establish Test Op and CD Procedures for Static Test and Acceptance Firing			X
B-20	Establish Test Op and CD Procedures for Launch			X
B-21	Determine Prop Tank Pressures Required for Sea Level Idle Mode Testing			X
B-22	Demonstrate Proper Hydraulic System Op		X	
B-23	Determine Engine Control Loop Characteristics		X	
B-24	Demonstrate Satisfactory PU System Closed Loop Dynamic Characteristics		X	

10.3.7.1 (Continued)

Typical data required to verify this objective includes:

a. Engine control sphere temperature

b. Engine control sphere pressure

OBJECTIVE B-2: Demonstrate Solid Propellant Turbine Starter (SPTS) concept for First, Second and Third Burn Start.

Since the SPTS concept is new to the S-IVB stage, performance data on engine start and S-IVB stage interaction is required. Additional data required is listed below:

a. SPTS chamber pressure history (pressure, burn time and characteristic pressure buildup and decay, run-to-run and engine-to-engine deviations) to be used in conjunction with engine basic performance (thrust buildup) to reconstruct test data and predict performance. Effects on rate of engine thrust buildup are also desired to determine engine side loads, stage thrust structure loading, structural and flow effects on inlet ducts (especially in transition to tapoff gas bootstrap operation).

b. Environmental operation extremes of the SPTS should be tested to obtain the characteristic buildup profile for typical temperatures within the allowable and expected flight temperatures.

c. Pump performance (stall margin) with SPTS system.

Present planning includes six three-start SPTS tests at altitude simulation, and six three-start SPTS tests during sea-level testing. These tests can be run in conjunction with other tests.

Typical data required to verify this objective includes:

a. Temperatures SPTS casing Nos 1, 2, and 3

b. Pressures SPTS chamber Nos 1, 2, and 3

c. Temperature fuel turbine inlet

d. Time from engine start command to attainment of mainstage (90% thrust)

e. Engine performance data during transient; including pump speeds, pressures, and flow rates.

OBJECTIVE B-3: Demonstrate the concept of taping off Thrust Chamber Gas and utilizing it to drive the Turbopump Turbines for first, second and third burn.

10.3.7.1 (Continued)

The tap-off-turbine concept is new to the S-IVB stage. Information is required on the performance and stability of this method. Its effect on engine start (start transients), and steady state stability (with PU perturbations). Tapoff flow will be calculated and used in conjunction with other engine performance parameters to calculate overall engine performance for test reconstruction and to make performance predictions. Environmental effects on tapoff system operation during start transient will be investigated.

Typical data required to verify this objective includes:

a. Pressure, tapoff gas

b. Temperature, tapoff gas

c. Fuel turbine inlet temperature

d. Oxid turbine inlet temperature

e. Fuel turbine inlet pressure

f. Oxid turbine inlet pressure

g. Vibration levels

h. Environmental (hardware) temperatures

OBJECTIVE B-4: Evaluate J-2S Engine Mainstage Performance Over the Mixture Ratio Excursions Expected with Open and Closed Loop PU Operation.

The following information is needed to enable effective post test reconstruction of engine performance, and to predict engine performance:

a. Steady state performance variations; including thrust, EMR, propellant consumptions and Isp.

b. Gain values for operation at expected EMR (stage effects) including variations of all interface parameters.

c. Engine response (thrust) characteristics to changes in EMR (PU effects).

d. Sea level and altitude simulation test data to derive conversion factors to predict flight performance from stage acceptance test data.

e. Thrust stability (open loop/closed loop)

Typical data required to verify this objective includes:

a. Chamber pressure

b. Thrust history (altitude versus sea level)

10. 3. 7. 1 (Continued)

c. Engine propellant flow rates

d. Engine vibration and side loads

e. Hot gas turbine system characteristics

f. Turbopump characteristics

OBJECTIVE B-5: Determine J-2S Engine Start Transients from Idle Mode to Mainstage for First Burn Start and Restart.

The following information is needed to enable effective post test reconstruction of engine performance, and to predict engine performance:

a. Start impulse

b. Performance variations; including thrust, EMR, propellant consumption, and Isp.

c. Engine response characteristics to changes in EMR.

d. Sea level and altitude simulation test data to derive conversion factors to predict flight performance from stage acceptance test data.

e. Feed system flow transients and loads

f. Thrust stability

g. Thrust alignments (stage moments)

h. Sea level and altitude side loads on restrainer links and hydraulic actuators.

Typical data required to verify this objective includes:

a. Propellant flow rates

b. Thrust history (sea level and altitude)

c. Chamber pressure

d. Engine vibration and side loads

e. Thrust rise rate (time to 90% thrust)

f. Hot gas turbine system characteristics

g. Turbopump characteristics

10.3.7.1 (Continued)

OBJECTIVE B-6: Determine J-2S Engine Cutoff Transients for Mainstage to Idle Mode, Mainstage to Zero Thrust and Idle Mode Cutoff.

The following information is needed to enable post test reconstruction of engine performance, and to predict engine and flight performance:

a. Cutoff impulse

b. Cutoff thrust decay rate and stability (time to zero thrust)

c. Sea level and altitude simulation test data to derive conversion factors to predict flight performance from stage acceptance test data.

d. Propellant consumption

e. Thrust alignment (stage moments)

f. Feed system flow transients and loads

Typical data required to verify this objective includes:

a. Propellant flow rate

b. Thrust history (sea level and altitude)

c. Chamber pressure

d. Engine vibration and side loads

e. Hot gas turbine system characteristics

f. Turbopump characteristics

OBJECTIVE B-7: Evaluate J-2S Engine Idle Mode as a means of Cutoff Slosh Control.

The application of steady state J-2S engine idle mode operation after mainstage cutoff for slosh control is dependent upon the mainstage to idle mode transient envelope. Cutoff idle mode steady state and transient repeatability are of primary concern. The following data should be obtained:

a. Steady state performance

b. Cutoff thrust transient from mainstage to steady state idle mode (rate, stability, and repeatability).

c. Thrust alignment

d. Idle mode cutoff thrust transient (cutoff impulse)

10.3.7.1 (Continued)

e. Propellant consumption

f. Engine specific impulse and mixture ratio.

Typical data required to verify this objective includes:

a. Thrust history (sea level and altitude)

b. Propellant flow rates

c. Chamber pressure

d. Engine vibration and side loads

OBJECTIVE B-8: Demonstrate the ability of the J-2S Engine to safely shutdown on Stage LOX depletion.

The J-2S utilizes LOX pump discharge pressure switches, instead of the present stage propellant tank level sensors to sense depletion. This new method of engine cutoff will affect propellant residuals and hence propellant loading schedules. Interaction between stage and engine during LOX depletion is also of significance and the following information is needed:

a. Determine engine cutoff sequence.

b. Determine residual LOX mass at engine shutdown.

c. Determine propellant consumption from engine cutoff signal to zero thrust

d. Determine variations from nominal engine cutoff in all engine systems as stated in Objective B-6.

e. Determine cutoff thrust history, impulse and repeatability.

Typical data required to verify this objective includes:

a. Propellant consumption and chamber pressure

b. Thrust history, thrust decay rate and stability (time to zero thrust)

c. Engine vibration and side loads

d. LOX feed system characteristics during depletion cutoff

e. Hot gas turbine system and turbopump characteristics

10. 3. 7. 1 (Continued)

OBJECTIVE B-9: Determine the required Idle Mode Chilldown time and performance necessary to achieve the First Burn Start.

Engine chilldown tests are necessary to determine nominal engine and turbopump cooldown time. A satisfactory chilldown must be followed by a successful attainment of mainstage operation. First burn chilldown tests should be repeated until the chilldown duration is repeatable for a given set of conditions. Questions which must be answered include:

a. Will propellant unsettling be overcome by the time run NPSH requirements are met?

b. Will a delayed thrust buildup at separation create stage control or ullage collapse problems?

c. What is the minimum duration of idle mode operation required for first burn chilldown?

Item a. can be analytically determined; both a. and b. above will be verified in flight. Item c. can be determined during the AEDC testing. The following type of information is required to satisfy the above objective:

a. NPSH conditions at the pump inlets

b. Thrust history (idle mode start to mainstage operation) and thrust stability

c. Chilldown duration

d. Engine propellant consumption, mixture ratio and specific impulse during idle mode.

e. Start impulse (including rate and stability)

Typical data required to verify this objective includes:

a. Engine chamber pressure, propellant flowrates during idle mode

b. Propellant temperatures and pressures at engine pump inlets and outlets, pump inlet duct feed hardware temperatures, and engine chilldown associated hardware temperatures.

c. Thrust (sea level and altitude)

d. Nominal chilldown duration (time required to meet specification start and mainstage requirements)

e. Engine vibration and side loads

10.3.7.1 (Continued)

OBJECTIVE B-10: Determine the effects of Operational Pressure and Temperature Limits on Chilldown during First Burn Idle Mode.

During S-IVB boost, heating will increase the propellant duct temperature prior to S-IVB/S-II separation. This is followed by a short period of zero gravity. The combination of boost heating and zero "g" could possibly cause some liquid propellant vaporization in the duct prior to mainstage start. Because of this vaporization it is necessary that the operational pressure and temperature limits be tested to insure that the J-2S engine can ingest the maximum expected vapor during idle mode and the start transient. Simulation of these conditions will also include the pressure profile through S-II maximum acceleration and into the zero "g" transient at separation.

Engine chilldown tests should be performed varying the LOX and LH_2 tank pressures and feed duct temperatures. The tests shown in Figure 10.3.7-1 should be performed to determine engine and turbopump cooldown times. The shape of the start limits "box" is only illustrative, and must be defined during the course of this test program. The following type of information is necessary for this objective:

a. NPSH conditions at the LOX and LH_2 pumps inlet, pump performance and associated transiet engine performance (including items in Objective B-5).

b. Total propellant consumption, LOX and LH_2 flow rates, and chilldown times.

Typical data required to verify this objective includes:

a. LOX and LH_2 pumps inlet and outlet pressures and temperatures.

b. Stage inlet duct temperatures, engine chilldown associated hardware temperatures, and engine injector conditioning.

OBJECTIVE B-11: Determine the required Idle Mode Chilldown Time and Performance necessary to achieve Restart.

Since idle mode chilldown operation is a new concept, considerable experimental data and test verification is needed. Some questions which must be answered include:

a. What is the minimum duration of idle mode operation required for chilldown, and the associated dispersions, sensitivity and repeatability?

b. What is the corresponding propellant consumed?

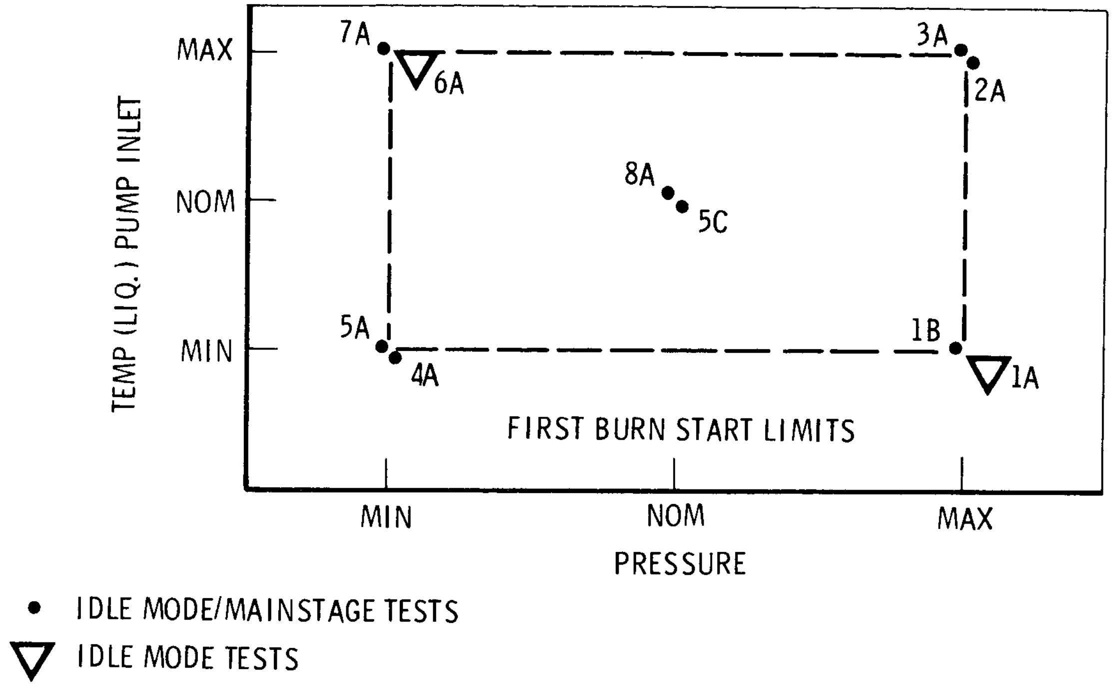


FIGURE 10.3.7-1. PRESSURE AND TEMPERATURE LIMIT TESTS FOR FIRST BURN IDLE MODE AT AEDC

10.3.7.1 (Continued)

Simplified models have been used for the chilldown analysis using lumped masses for the engine and stage ducting, choked flow, simple resistance equations, specified percents of engine mass to be chilled, etc. Based on these models and predictions, some extensive stage modifications have been proposed to exploit the J-2S capabilities (e.g., removal of the recirculation chilldown system). Testing is required to augment analytical predictions, to establish system confidence and system reliability, to avoid excessive initial performance penalties, and to identify performance margins.

Engine chilldown tests should be performed to determine the nominal engine and turbopump cooldown time. To be termed successful, a satisfactory idle mode chilldown must be followed by a successful achievement of mainstage operation. For each given set of conditions the restart idle mode chilldown should be repeated until a band of repeatability is established. The following type of information is required to satisfy this objective:

a. LOX and LH_2 turbopump inlet NPSH history from idle mode start through mainstage start.

b. Pump performance and stall margin during mainstage start.

c. Nominal restart idle mode chilldown time and performance, and stabilization.

10.3.7.1 (Continued)

d. Nominal restart idle mode thrust chamber performance, specific impulse, and mixture ratio.

e. Thrust rise rate and stability during restart idle mode.

f. Propellant flowrate characteristics and consumption during restart idle mode.

Typical data required to verify this objective includes:

a. Thrust chamber pressure and thrust.

b. Propellant flow rates.

c. Propellant pressures and temperatures in the inlet ducting and in the engine.

d. Stage inlet duct hardware temperatures, and engine chilldown associated hardware temperatures.

e. Hot gas turbine system and turbopump characteristics.

f. Engine vibration and side loads.

OBJECTIVE B-12: Determine the effects of Operational Pressure and Temperature Limits on Chilldown during Restart Idle Mode.

During S-IVB coast, orbital heating will increase the propellant duct, engine hardware, and the propellant bulk temperatures. Simulation of these extremes of pressure and temperature combinations during restart idle mode is necessary to simulate these conditions. Simulations might be conducted using the battleship stage prevalves and heat lamps to create "hot and dry" duct starts, and variations thereof. The tests will include simulations of 4.5, 5.3 and 7.5 hours of orbital heating. These engine chilldown tests should be performed by holding the engine hardware temperatures constant and varying the tank pressure and the feed duct temperatures. The tests shown in Figure 10.3.7-2 are designed to explore the limits and to establish a nominal cooldown time for all expected variations. Test numbers correspond to tests described in Table 10.3.7-3. Four additional tests, with variations in engine hardware temperatures, will be conducted to complete the survey. The following information should be obtained from these tests:

a. NPSH conditions at the LOX and LH_2 pumps inlet.

b. Propellant consumption and chilldown time.

Typical data required to verify this objective includes:

a. LOX and LH_2 flow rates

b. LOX and LH_2 pumps inlet temperature and pressure

c. LOX and LH_2 pumps discharge pressure and temperature

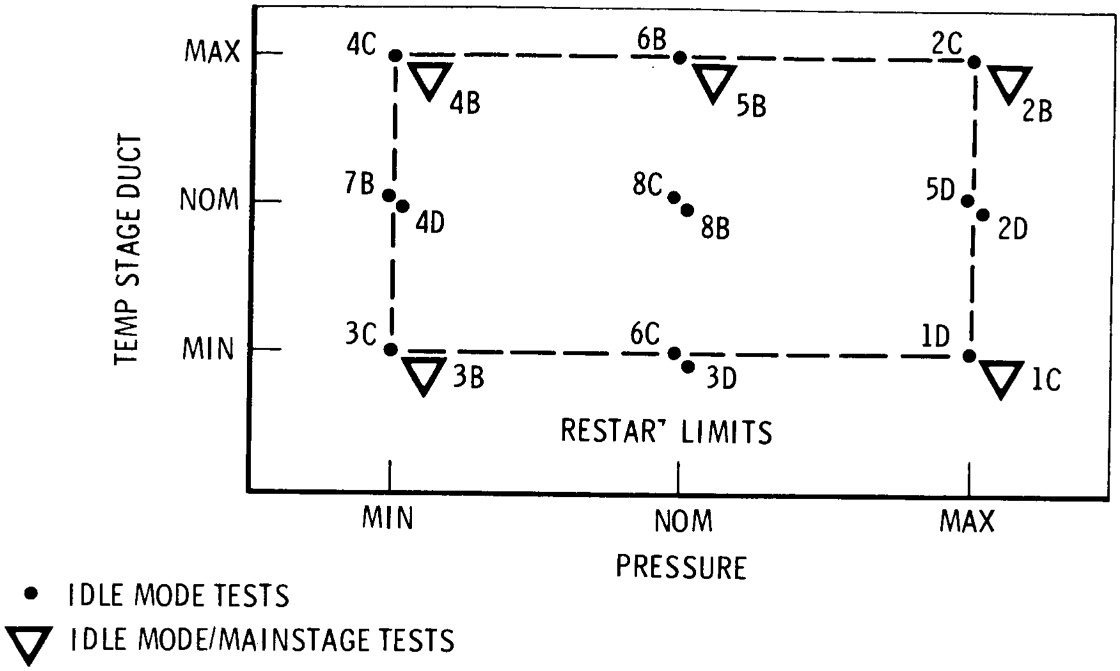


FIGURE 10.3.7-2. PRESSURE TEMPERATURE LIMIT TESTS FOR RESTART IDLE MODE AT AEDC

10.3.7.1 (Continued)

d. Stage inlet duct temperature and engine chilldown associated hardware temperatures.

OBJECTIVE B-13: Simulate Simultaneous Restart Idle Mode and Ambient Repress Start.

In the event the O_2/H_2 burner system should fail to ignite for repress, the ambient helium repress system would be used to pressurize both propellant tanks. If this were to occur, the repress signal could occur simultaneously with idle mode start or shortly thereafter. Concurrent tank pressurization and idle mode operation could be expected to yield significant differences from the completely pressurized chilldown times and thrust history.

To simulate this condition, idle mode start would be at CVS tank pressure (equivalent differential pressure) followed by closure of the CVS system and propellant tanks pressurization with the ambient helium system. Since the CVS and ambient helium systems may not be available at AEDC, equivalent venting and pressurization rates could be used. These tests should be run at the "worst case" inlet temperature conditions. The following information is necessary:

a. NPSH conditions at the pump inlets

b. Propellant consumption and chilldown time.

c. Idle mode thrust chamber performance; thrust, Isp, mixture ratio, chamber pressure, etc.

10.3.7.1 (Continued)

Typical data required to verify this objective includes:

a. Propellant flow rate

b. Stage inlet duct temperatures and engine chilldown associated hardware temperatures

c. Engine pump discharge pressures and temperatures

OBJECTIVE B-14: Determine the performance characteristics of the LOX Tank Pressurization System during Engine Mainstage Operation.

Satisfactory completion of this objective will be achieved when the following conditions have been met:

a. Because of the difference in engine start characteristics associated with the J-2S SPTS and tapoff turbine concept, determine the ullage pressure recovery rate during engine start transition.

b. Because of the higher LOX flowrate associated with the higher performance of the J-2S engine and the heat exchanger, demonstrate in-flight pressurization control at 38 to 41 psia.

c. Determine the nature of ullage gas temperature stratification with the increased propellant outflow.

Typical data required to verify this objective includes:

a. LOX tank ullage pressure

b. LOX tank ullage temperatures

c. Helium flowrate through heat exchanger

d. Inlet and outlet helium temperature at the heat exchanger

e. Overcontrol and undercontrol energy rates

f. Overcontrol and undercontrol pressure rise and collapse rates

OBJECTIVE B-15: Determine the Performance Characteristics of the LH_2 Tank Pressurization System during Engine Mainstage Operation.

The increased performance of the J-2S engine makes it necessary to evaluate the capability of the LH_2 tank pressurization system during the start transient and mainstage operation of the J-2S engine. The hydrogen gas used as a pressurant is tapped off of the fuel injector at an increased pressure due to the higher thrust chamber pressure associated with the J-2S engine. Because of the higher tapoff pressure, it is necessary to verify that the LH_2 pressurization system will function properly in providing the required tank pressure. Satisfactory

10.3.7.1 (Continued)

completion of this objective will be achieved when the following conditions have been met:

a. Because of the difference in engine start characteristics associated with the J-2S SPTS and tapoff turbine concept, determine the ullage pressure recovery rate during engine start transition.

b. Because of the higher LH_2 flowrate associated with the higher performance of the J-2S engine, demonstrate in-flight pressurization control at 28 to 31 psia.

c. Determine ullage gas temperature stratification associated with the increased propellant outflow.

d. Determine step pressurization performance

Typical data required to verify this objective includes:

a. LH_2 tank ullage pressure

b. LH_2 tank ullage temperature

c. Flowrate of the pressurant

d. Temperature of the pressurant

e. Pressure of the pressurant

f. Pump inlet temperature

OBJECTIVE B-16: Establish Loading and Venting Procedures for the Engine Control Helium Sphere.

The J-2S engine control helium is stored in a spherical tank of 4,000 cu in., under 3100 $\pm$100 psig pressure at -200° to +140°F. The loading flowrate must be limited to 0.1 lbm/sec or less. The GSE/stage/engine system must be flow tested to verify that the specified requirements are met. The following information must be obtained:

a. Mass of helium loaded

b. Maximum system flowrate

c. Time to load to specified values

d. Time to vent down

Typical data required to verify this objective includes:

a. Engine control helium sphere pressure

b. Engine control helium sphere gas temperature

OBJECTIVE B-17: Demonstrate Adequate Helium Purge for the J-2S Thrust Chamber Jacket and the Turbopumps.

10.3.7.1 (Continued)

A helium purge of the thrust chamber LH_2 jacket and both turbopumps cavities prior to firing is necessary to remove any air or water vapor present. A helium purge after a static firing removes water vapor from the turbopump cavities and accumulated hydrogen from the LH_2 jacket.

The engine manufacturer specifies that dry helium at 150 $\pm$25 psig and 50° to 150°F (125 SCFM reference) be supplied at the customer connect panel. For static testing, the purge is accomplished by flowing helium from 30 minutes prior to propellant drop and/or propellant loading until engine start, and for 15 minutes following engine cutoff. The GSE/stage/engine system must be flow tested to verify that it meets the specified requirements. Information required to complete this objective includes:

a. Mass of gas used for typical purges

b. System flow rate (orifice pressure and temperature and/or flow measurement during blowdown).

Typical data required to verify this objective includes:

a. Pressure and temperature at customer connect point

b. Purge supply pressure

OBJECTIVE B-18: Demonstrate Adequate Helium Purge to the LOX Dome.

A helium or nitrogen purge of the LOX dome is necessary before a firing to remove water vapor, and after firing to prevent accumulation of hydrogen in the dome. The engine manufacturer specifies a 30-minute purge with dry helium at 50° to 150°F (400 $\pm$25 psig) or nitrogen at 100° to 150°F (600 $\pm$25 psig) and 150 SCFM before propellant drop and/or propellant loading and a 15-minute purge at 150 SCFM after firing. The LOX dome purge is also required for the full duration of any hold period in the firing sequence, i.e., for any period after a cutoff and prior to a restart. The GSE/stage/engine system must be flow tested to verify that it meets the specified requirements. Information required to complete this objective includes:

a. Mass of gas used for typical purge

b. System flow rate (orifice pressure and temperature and/or flow measurements during blowdown).

Typical data required to verify this objective includes:

a. Pressure at customer connect panel

b. Temperature at customer connect panel

c. Purge supply pressure

10.3.7.1 (Continued)

OBJECTIVE B-19: Establish test operation and countdown procedure for Static Test and Acceptance Firings.

Establish a smooth and workable countdown procedure that will allow sufficient time to complete each necessary step of the countdown. The existing S-IVB countdown procedures will be used where applicable, the new requirements identified, and integrated into a new procedure. This procedure will be tested and verified so that conversion to actual production acceptance test procedures will be smooth and efficient. Demonstrate integration of purge procedures, redline monitoring requirements, sequencing and hold capability. Develop and/or demonstrate adequate means to eliminate the fire hazard that could occur because of the excess hydrogen dumped through the engine during extended idle mode operation.

OBJECTIVE B-20: Establish test operations and countdown procedures for Launch.

Derive new procedures emanating from the new J-2S installation and S-IVB stage modifications, and integrate into existing sequences. These procedures will be tested during simulated countdowns, where particular attention will be given to the integration of (1) new purge procedures, (2) liftoff requirements, (3) items that may require a hold, (4) launch hold time capability, (5) recycle capability, (6) recycle points, and (7) overall time to complete the countdown. Particular emphasis will be placed on obtaining a reasonably simple, but effective countdown procedure with sufficient flexibility to cope with any anticipated complexities that may evolve in an actual KSC launch countdown.

OBJECTIVE B-21: Determine the Propellant Tank Pressures required for Sea Level Idle Mode Testing.

To more adequately simulate flight conditions during MSFC Battleship testing, it is desirable that an extended idle mode capability be established for the MSFC battleship test program. It is also desirable that this testing utilize S-IVB stage system hardware, specifically the vent and relief valves, and the stage pressurization systems. Therefore, in the most part, it is undesirable to run at elevated tank pressures, i.e., pressures beyond the S-IVB stage systems capability. Testing within these pressure ranges (compensating for atmospheric pressure) must be conducted at AEDC altitude test facility to verify operational capability in this regime and to establish nominal chilldown times (and limits) prior to initiation of MSFC testing. Information required to complete this objective includes:

a. NPSH conditions at the LOX and LH_2 pump inlets

b. Propellant consumption and chilldown time

Typical data required to verify this objective includes:

a. Inlet duct temperatures and engine chilldown associated hardware temperatures

b. Pump discharge pressures and temperatures

c. Propellant flowrates

10.3.7.1 (Continued)

OBJECTIVE B-22: Demonstrate (1) proper operation of the hydraulic system and determine effect of propellant loading, chilldown and hot gimballing on hydraulic fluid and GN_2 temperature in major hydraulic system components and (2) demonstrate that the hydraulic system operates in a proper manner with the engine centered hydraulically, using the auxiliary pump. Monitor fluid and GN_2 temperature in major hydraulic system components during propellant loading and during chilldown. Repeat the test during propellant loading and the subsequent engine firing of one of the hot gimballing tests to determine if the high temperature condition produced by system operation during engine hot firing has any detrimental effect.

Instrumentation Required:

Meas. No.	Title	Range	MRD No.
C050-403	Temp - Hydraulic Pump Inlet Oil	400 to 785^{o}R	1A37147
C138-403	Temp - Accumulator GN_2	400 to 785^{o}R	1A37149
C217-401	Temp - Main Hydraulic Pump Flange	-360 to 810^{o}R	1A87937
C642-403	Temp - Reservoir Oil - (Hardware)	400 to 785^{o}R	1B32590
D043-403	Press - Accumulator GN_2	(0 to 4000 psia MVA System) 1500 to 4000 psia	1A37145
D549-403	Press - Hydraulic System - Hardwire	(1500 to 4500 MVA System) 1500 to 4500 psia	1B50156
D550-403	Press - Reservoir Oil - Hardwire	0 to 400 psia	1B50157
L504-403	Level - Reservoir Oil - Hardwire	0 to 100%	1B32589

OBJECTIVE B-23: To determine engine control loop dynamic characteristics during an engine firing condition.

Measure the frequency response and the transient response of the pitch and yaw engine actuator control system and the actuator differential pressures while gimballing during hot firing. The recorded data will be utilized to determine the following:

a. Closed loop frequency response characteristics.

b. Cross coupling effect in the nongimballing plane.

c. Dynamic response to step commands.

d. Compute control loop transfer functions.

e. Determine engine spring mass characteristics.

f. Determine actuator static load characteristics including duct/umbilical spring rate, gimbal friction, load bias effects.

g. Vehicle dynamic characteristics during gimballing.

10.3.7.1 (Continued)

The tests to be conducted during an engine hot firing condition are listed on Table 10.3.7-II.

OBJECTIVE B-24: Verify satisfactory closed loop dynamic characteristics of Propellant Utilization System.

Determine closed loop dynamic characteristics of PU system during hot firing and compare with analytical results. Offload LOX to obtain hardover PU System valve operation and transient valve response to a 5.0:1.0 commanded mixture ratio. Typical data required for this objective are:

- Mass Histories
- PU Error Signal
- PU Valve Position

S-IVB/J-2S Battleship Development Tests at Arnold Engineering Development Center

The S-IVB/J-2S Battleship development test program at Arnold Engineering Development Center (AEDC) will provide basic altitude (vacuum) simulation for detailed investigation of system chilldown requirements. Although the zero "g" aspect of the flight cannot be directly simulated, some simulation of propellant conditions at idle mode start (a zero "g" condition in flight) can be obtained by utilizing the present S-IVB prevalves and the propellant inlet ducts, to obtain variations of hot and dry inlet condition for idle mode start. This in conjunction with different tank pressures would provide different chilldown flows and times to achieve a chilldown.

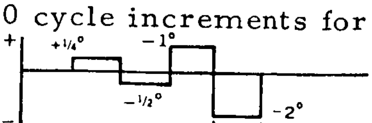
Thirty-two tests (Tables 10.3.7-III and 10.3.7-IV) have been defined for the J-2S/S-IVB test program at AEDC, and primarily include (1) first burn idle mode, (2) restart idle mode, and (3) short duration mainstage testing. The emphasis during the AEDC testing has been placed on exploring the pressure and temperature limits for first burn start and restart to establish realistic chilldown times for each. These tests are scheduled at approximately three per J-4 test cell evacuation ("air-on") every other week for seventeen weeks of testing (Figure 10.3.7-3). An "air-on" is defined as one AEDC altitude simulation with a continuous blowdown to attain the low pressures of altitude.

It is assumed that 33 seconds of mainstage and up to 1000 seconds of idle mode or combinations thereof can be run back-to-back over several hours (one "air-on" period). It is understood that steam capacity is limited to a maximum of 520 seconds of idle mode such that the remainder of a 1000 second idle mode test would be with decaying cell pressure.

TABLE 10.3.7-II. ENGINE ACTUATOR TESTS

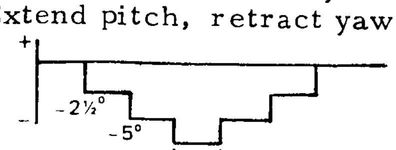
Order	Command Amplitude	"P" ACT	"Y" ACT	Frequency Range (CPS)	Sinusoidal Command	Triangular Command	Step Pattern
1.	±1/4°	X		*(0.5 to 15.0)	X		
2.	±1/2°	X		*(0.5 to 15.0)	X		
3.	±1/4°		X	*(0.5 to 15.0)	X		
4.	±1/2°		X	*(0.5 to 15.0)	X		
5.	±2°	X		0.2, 0.5, 1.0	X		
6.	±2°		X	0.2, 0.5, 1.0	X		
7.	±2°	X		0.2, 0.5, 1.0		X	
8.	±2°		X	0.2, 0.5, 1.0		X	
9.	--	X		--			X**
10.	--		X	--			X**
11.	±1°	X		*(0.5 to 15.0)	X		
12.	±1°		X	*(0.5 to 15.0)	X		
13.	±1/2°	X	X	*(0.5 to 15.0)	X***		
14.	±1/2°	X	X	*(0.5 to 15.0)	X†		
15.	--	X		--			X††
16.	--		X	--			X††
17.	±7.0°	X		0.0367 cps		X	
18.	±7.0°		X	0.0367 cps		X	

*0.5 cycle increments for 0.5 thru 3.0 cps
0.25 cycle increments for 3.25 thru 7.0 cps
0.5 cycle increments for 7.50 thru 10.0 cps
1.0 cycle increments for 11 thru 15.0 cps

** 

T = 0.9 sec

***Extend pitch, extend yaw (in phase)
†Extend pitch, retract yaw (out of phase)

†† 

T = 0.9 sec against stop

TABLE 10. 3. 7-II. (Continued)

INSTRUMENTATION REQUIRED

Meas No.	Title	Range	MRD No.
C050-403	Temp - Hydraulic Pump Inlet Oil	400 to 785°R	1A37147
C138-403	Temp - Accumulator GN_2	400 to 785°R	1A37149
C217-401	Temp - Main Hydraulic Pump Flange	-260 to 810°R	1A87937
C642-403	Temp - Reservoir Oil - (Hardwire)	400 to 785°R	1B32590
C685-401	Temp - Engine Gimbal Assembly	-310 to 785°R	1B20052
D043-403	Press. - Accumulator GN_2	1500 to 4000 psia (0 to 4000 psia for MVA System)	1A37145
D549-403	Press. - Hydraulic System - (Hardwire)	1500 to 4500 psia	1B50156
D550-403	Press. - Reservoir Oil - (Hardwire)	0 to 400 psia	1B50157
D510-403	Press. - Differential, Eng "P" Actuator (Hardwire)	-6000 to 6000 psid	1B32594
D511-403*	Press. - Differential, Eng "Y" Actuator (Hardwire)	-6000 to 6000 psid	1B32595
G504-403*	Position - Actuator Piston Pot, Pitch Plane (Hardwire)	±7. 5 deg	1B32591
G505-403*	Position - Actuator Piston Pot, Yaw Plane (Hardwire)	±7. 5 deg	1B32592
L504-403	Level - Reservoir Oil (Hardwire)	0 to 100%	1B32589
M704-B20*	Current - Servo Valve Input Signal - "P"	±50 ma	1B29435
M705-B20*	Current - Servo Valve Input Signal - "Y"	±50 ma	1B29436
N703-B03	Misc - Eng Support Link Load, Pitch Plane	±50, 000 lb	1B32665

*Δ: If the majority voting system (MVA) is incorporated on stages utilizing J-2S engines, measurements marked * will be eliminated and those marked Δ will be utilized.

TABLE 10. 3. 7-II. (Continued)

Meas No.	Title	Range	MRD No.
N704-B03	Misc - Eng Support Link Load, Yaw Plane	±50, 000 lb	1B32665
New	Rotational Gimbal Potentiometer Pitch Plane	±7. 5 deg	
New	Rotational Gimbal Potentiometer Yaw Plane	±7. 5 deg	
G0012-403Δ	Position T1 Actuator Pitch Plane	±7 deg	1B74778
G0013-403Δ	Position T2 Actuator Pitch Plane	±7 deg	1B74779
G0014-403Δ	Position T3 Actuator Yaw Plane	±7 deg	1B74780
G0015-403Δ	Position T4 Actuator Yaw Plane	±7 deg	1B74781
G0517-403Δ	Position T2 Act Pitch Plane Hardwire	±7 deg	1B74782
G0518-403Δ	Position T2 Act Yaw Plane Hardwire	±7 deg	1B74783
D0257-403Δ	Press. Differential Act, Pitch	3000 psid	1B74784
D0258-403Δ	Press. Differential Act, Yaw	3000 psid	1B74785
D0598-403Δ	Press. Differential Act, Pitch Hdw.	3000 psid	1B74786
D0599-403Δ	Press. Differential Act, Yaw Hardwire	3000 psid	1B74787
M0740-B03Δ	Current TM1-Servo Pitch Act - Hardwire	±60 ma	1B74772
M0741-B03	Current TM2-Servo Pitch Act - Hardwire	±60 ma	1B74773
M0742-B03Δ	Current TM3-Servo Pitch Act - Hardwire	±60 ma	1B74774
M0743-B03Δ	Current TM1-Servo Pitch Act - Hardwire	±60 ma	1B74775
M0744-B03Δ	Current TM2-Servo Pitch Act - Hardwire	±60 ma	1B74776
M0745-B03Δ	Current TM3-Servo Pitch Act - Hardwire	±60 ma	1B74777

*Δ: If the majority voting system (MVA) is incorporated on stages utilizing J-2S engines, measurements marked * will be eliminated and those marked Δ will be utilized.

TABLE 10. 3. 7-III. J-2S BATTLESHIP TESTS AT AEDC

Test	Run Time IM/MS/IM	LOX			LH_2			Eng	Comments
		P	T_L	T_{HW1}	P	T_{L1}	T_{HW}	T_{HW}	
1A	10/0/0	max	min	---	max	min	---	max	Establish first burn I/M time -- one hour hold -- pre-valve open
1B	1/8/2	max	min	---	max	min	---	max	Verify 1A data with engine start
1C	150/0/0	max	---	min	max	---	min	max	Establish most favorable I/M time -- prevalve closed
				280°R					Dry ducts
1D	100/8/10	max	---	min	max	---	min	max	Verify test-1C with engine start
2A	3/8/2	max	max	---	max	max	---	max	First burn max +
2B	300/0/0	max	---	max	max	---	max	max	Restart extended I/M sim (max hardware temp)
2C	150/8/10	max	---	max	max	---	max	max	Restart sim, verify 2B with start
2D	125/8/15	max	---	nom	max	---	nom	max	Restart sim
3A	1/8/3	max	max	---	max	max	---	max	First burn min I/M (max T, P)
3B	200/0/0	min	---	min	min	---	min	max	Ext'd restart I/M sim (min P, T_{HW})
3C	150/8/10	min	---	min	min	---	min	max	Restart sim, verify 3B with start
3D	125/8/15	nom	---	min	nom	---	min	max	Restart sim
4A	3/8/10	min	min	---	min	min	---	max	First burn I/M (min T, P)
4B	400/0/0	min	---	max	min	---	max	max	Ext'd restart I/M sim (min P, max T_{HW})
4C	200/8/2	min	---	max	min	---	max	max	Restart sim, verify test 4B with start
4D	175/8/5	min	---	nom	min	---	nom	max	Restart sim

TABLE 10. 3. 7-III. (Continued)

Test	Run Time IM/MS/IM	LOX			LH_2			Eng T_{HW}	Comments
		P	T_L	T_{HW1}	P	T_{L_1}	T_{HW}		
5A	1/8/2	min	min	---	min	min	---	max	First burn I/M (min T, P)
5B	200/0/2	nom	---	max	nom	---	max	max	Ext'd restart I/M sim (nom P, max T_{HW})
5C	1/33/10	nom	nom	---	nom	nom	---	max	LOR first burn sim
5D	100/33/10	max	---	nom	max	---	nom	max	LOR restart sim, LOX depl
6A	15/0/10	min	max	---	min	max	---	max	Extended first burn I/M
6B	175/33/2	nom	---	max	nom	---	max	max	Ext'd restart I/M (nom P, max T_{HW})
6C	125/33/15	nom	---	min	nom	---	min	max	Ext'd restart I/M (nom P, min T_{HW})
7A	1/5/5	min	max	---	min	max	---	max	Synchronous Mission Simulation
7B	175/5/5	min	---	nom	min	---	nom	max	
7C	100/33/10	nom	---	nom	nom	---	nom	nom	
8A	1/5/10	nom	nom	---	nom	nom	---	max	Sync mission sim
8B	100/33/2	nom	---	nom	nom	---	nom	max	Sync mission sim
8C	100/33/0	nom	---	nom	nom	---	nom	max	Sync mission sim, LOX depl
9A	1/33/10	nom	nom	---	nom	nom	---	nom	First burn I/M (nom P, T) (nom eng T_{HW})
9B	150/0/0	nom	nom	---	nom	nom	---	nom	Extended I/M (nom P, T_{HW}, and eng T_{HW})
9C	150/0/0	nom	nom	---	nom	nom	---	min	Extended I/M (nom P, T_{HW}, and min eng T_{HW})

I/M = Idle mode duration (seconds)
MS = Mainstage duration (seconds)
P = Tank pressure
T_L = Liquid temperature
T_{HW1} = Hardware temperature

TABLE 10.3.7-IV. AEDC S-IVB/J-2S BATTLESHIP DEVELOPMENT TEST PROGRAM MATRIX

Objectives		1A	1B	1C	1D	2A	2B	2C	2D	3A	3B	3C	3D	4A	4B	4C	4D	5A	5B	5C	5D	6A	6B	6C	7A	7B	7C	8A	8B	8C	9A	9B	9C
	Run Time	10/0/0*	1/8/2	150/0/0	100/8/10	3/8/2	300/0/0	150/8/10	125/8/15	1/8/3	200/0/0	150/8/10	125/8/15	3/8/10	400/0/0	200/8/2	175/8/5	1/8/2	200/0/2	1/33/10	100/33/10	15/0/10	175/33/2	125/33/15	1/8/5	175/8/5	100/33/10	1/8/10	100/33/2	100/33/0	1/33/10	150/0/0	150/0/0
B-1	Determine Engine Pneumatic System Characteristics	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
B-2	Demonstrate SPTS (1, 2 and 3 Starts)		1		2	1		2	3	1		2	3	1		2	3	1		2	3		1	2	1	2	3	1	2	3	1		
B-3	Demonstrate Tapoff Turbine Concept		X		X	X		X	X	X		X	X	X		X	X	X		X	X		X	X	X	X	X	X	X	X	X		
B-4	Evaluate J-2S Mainstage Performance Over PU Range		X		X	X		X	X	X		X	X	X		X	X	X		X	X		X	X	X	X	X	X	X	X	X		
B-5	Determine J-2S Engine Start Transients (I/M to MS)		X		X	X		X	X	X		X	X	X		X	X	X		X	X		X	X	X	X	X	X	X	X	X		
B-6	Determine J-2S Engine Cutoff Transients	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
B-7	Evaluate J-2S Engine I/M for Cutoff Slosh Control		X		X	X		X	X	X		X	X	X		X	X	X	X	X	X	X	X	X	X	X	X	X	X		X		
B-8	Demonstrate LOX Depletion Cutoff																				X									X			
B-9	Determine First Burn I/M Chilldown Time and Performance	X	X			X				X				X				X				X			X			X			X		
B-10	Determine LOX and LH_2 Chilldown Pressure and Temperature Limits for First Burn Idle Mode	X	X			X				X				X				X				X			X			X			X		
B-11	Determine Restart I/M Chilldown Time and Performance			X	X		X	X	X		X	X	X		X	X	X		X		X		X	X		X	X		X	X		X	X
B-12	Determine LOX and LH_2 Chilldown Pressure and Temperature Limits for Restart Idle Mode			X	X		X	X	X		X	X	X		X	X	X		X		X		X	X		X	X		X	X		X	X
B-13	Simulate Simultaneous Restart Idle Mode and Ambient Repress																												X	X			
B-21	Determine Propellant Tank Pressures Required for Sea Level Idle Mode Testing													X				X		X		X			X								

* Test Run Time (Idle Mode/Mainstage/Idle Mode) in Seconds.

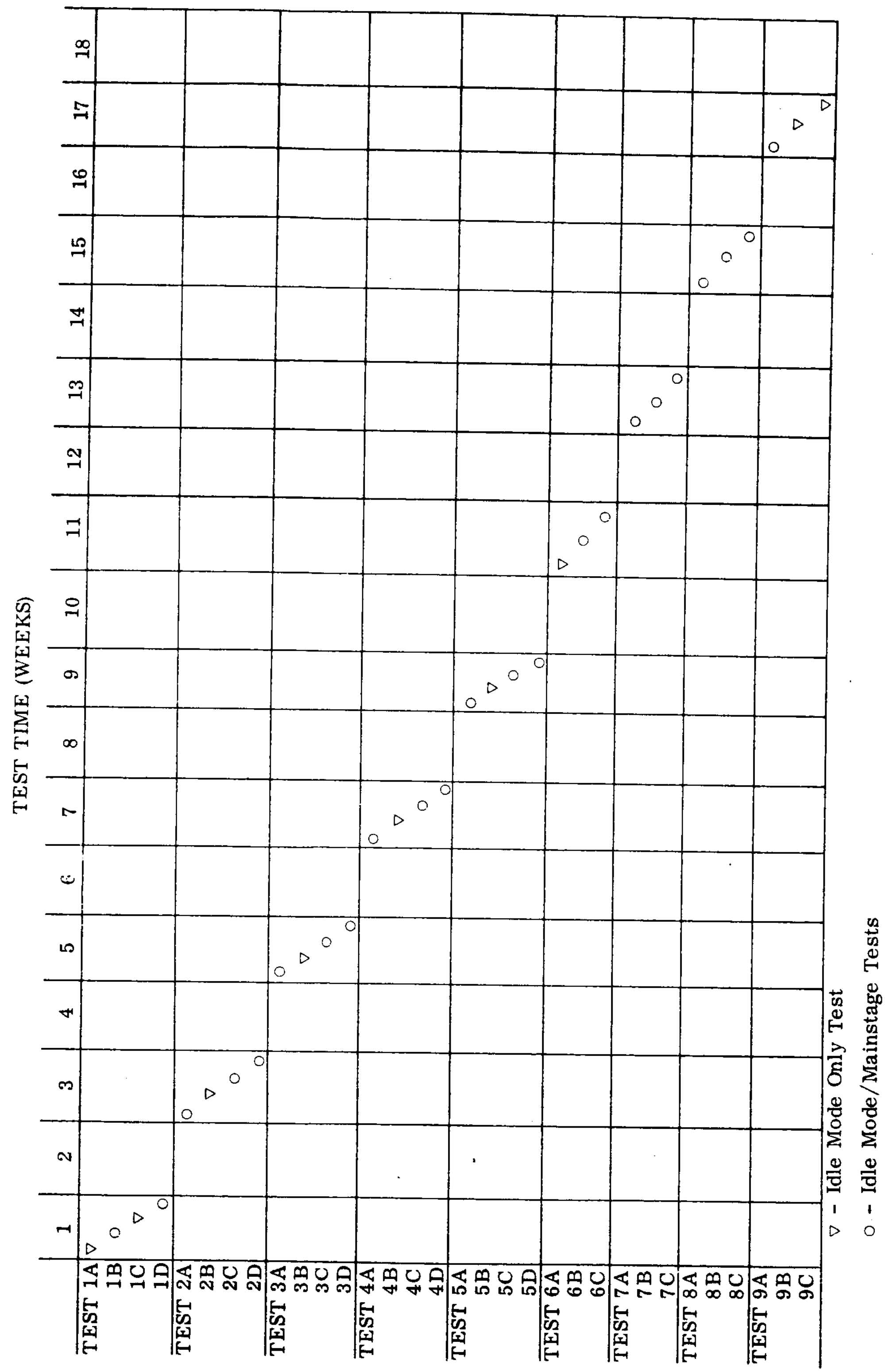


FIGURE 10.3.7-3. AEDC ATTITUDE SIMULATION TEST SCHEDULE

10.3.7.1 (Continued)

S-IVB/J-2S Battleship Development Tests At MSFC

For the MSFC S-IVB/J-2S Battleship test program, thirty-two tests have been defined (Tables 10.3.7-V and VI). These tests include numerous full duration firings, synchronous mission, and LOR mission simulations, PU excursion, and LOX depletion tests. Emphasis during these tests has been placed on S-IVB/J-2S systems testing and the establishment of acceptance test and launch requirements. The tests are scheduled over a one year period allowing significant time periods between the scheduled tests for data analysis.

It should be noted that the tests scheduled indicate numerous extended idle mode tests. These tests are designed to augment the AEDC chilldown studies utilizing the S-IVB system to integrate chilldown and mainstage testing, including complete mission simulation. The chilldown times are subject to change pending the results of the preceding S-IVB/J-2S Battleship testing at AEDC. Aside from the basic inability to simulate zero "g" during ground testing; and the lack of altitude simulation (as available at AEDC); the MSFC extended idle mode testing, at present, still faces two major constraints.

a. To conduct idle mode tests at sea level, it is presently required that the LH_2 tank pressure be a minimum of 40 psia to preclude engine deterioration. Since the S-IVB LH_2 tank relief valves are presently set for 31 to 34 psia, this would entail modification or replacement of the valves to increase LH_2 tank relief pressure upwards of 41 psia.

b. Extended idle mode testing generates excess GH_2 from the engine exhaust and unless properly disposed of, could create a fire hazard.

Idle mode testing has been safely and successfully conducted at the engine manufacturers test facility by utilizing a GN_2 ring purge around the engine exhaust. For extended idle mode testing at MSFC, this type of system is presently available at MSFC to eliminate the fire hazard problem. S-IVB LH_2 tank vent valves have also been modified at MSFC to relieve in excess of 40 psia.

As with the AEDC Battleship testing the present S-IVB propellant inlet ducting and prevalves will be used to obtain variation of hot and dry inlet conditions for idle mode start.

In light of the projected long term utilization of the S-IVB/J-2S stage testing, it is strongly recommended that an extended idle mode capability be established for the MSFC test area. Although this testing still lacks two basic ingredients of flight simulation that are fundamental to the question of adequate chilldown (i.e., zero "g" and low exhaust pressure), it is hoped that correlation can be developed between the AEDC and MSFC testing. This information, along with the later flight verification, may substantiate the value of long range MSFC flight support testing throughout the course of the S-IVB/J-2S flight program. A basic altitude (low pressure) simulation system at MSFC for long term test availability should also be considered to replace long term tie-up of the AEDC facility.

TABLE 10.3.7-V. S-IVB/J-2S BATTLESHIP TESTS AT MSFC

Test No.	Run Time	Comments
1	1/30/2	MSFC Battleship/J-2S Systems Shakedown
2A	1/150/10	Three-start Shakedown. Max restart idle mode, max slosh control idle mode
2B	250/70/10	Full duration (in three burns)
2C	250/300/10	3 SPTS demonstration
3	1/450/10	Full duration single burn (open loop PU null). Demonstrate full duration (acceptance test simulation)
4A	1/150/10	(LOR simulation) 5.5 PU
4B	200/300/10	5.0 PU
5A	1/100/10	LOR simulation 5.0 PU
5B	200/250/10	Ambient repress simultaneous with idle mode start (4.5 PU)
6A	3/150/10	LOR simulation 4.5 PU
6B	150/250/10	LOX depletion cutoff with idle mode (Ambient repress simultaneous with idle mode start) (5.5 PU)
7A	3/150/10	LOR launch countdown and flight simulation
7B	100/300/10	LOX depletion without idle mode (PU -- closed loop)

TABLE 10.3.7-V. (Continued)

Test No.	Run Time	Comments	
8A	1/150/2	LOR static test (acceptance simulation) (PU 5.5)	
8B	1/300/2	Demonstrate elect. control and instr adequacy (PU 4.5) (First acceptance test simulation)	
9	1/30/2	New Engine Shakedown Tests	
10A	1/150/10	PU 5.0 (open loop)	Sync mission simulation; pressure, temperature limits survey for Battleship test correlation with AEDC data
10B	250/70/10	PU 5.0 (open loop)	
10C	250/300/10	PU 5.0 (open loop)	
11A	3/100/5	PU 5.5 (closed loop)	Sync mission simulation; pressure, temperature limits survey for Battleship test correlation with AEDC data
11B	200/70/0	PU 5.0 (closed loop)	
11C	200/200/0	PU 4.5 (open loop)	
12A	5/150/0	PU 5.0 (closed loop)	
12B	175/70/0	Amb repress simult with idle mode start; PU 5.0 closed loop	
12C	175/250/0	Amb repress simult with idle mode start; PU 5.0 closed loop	
13A	2/70/0		Sync mission launch countdown and flight simulation
13B	125/200/0		
13C	125/150/10	LOX depletion with idle mode cutoff	
14A	1/150/2	PU 5.5 (closed loop)	Sync mission static test simulation (first acceptance test simulation)
14B	1/70/2	PU 5.0 (closed loop)	
14C	1/250/2	PU 4.0 (open loop)	
15	1/450/0	Full duration capability -- single burn closed loop LOX depletion (acceptance test simulation) PU 5.5 cutback to 5.0	

TABLE 10.3.7-VI. MSFC S-IVB/J-2S BATTLESHIP DEVELOPMENT TEST PROGRAM MATRIX

MCDONNELL DOUGLAS CORPORATION

		Run Time																														
		1/30/2*	1/150/10	250/70/10	250/300/10	1/450/10	1/150/10	200/300/10	1/100/10	200/250/10	3/150/10	150/250/10	3/150/10	100/300/10	1/150/2	1/300/2	1/30/2	1/150/10	250/70/10	250/300/10	3/100/5	200/70/0	200/200/0	5/150/0	175/70/0	175/250/0	2/70/0	125/200/0	125/150/10	1/150/2	1/70/2	1/250/2	1/450/0
		Test No.																														
Objectives		1	2A	2B	2C	3	4A	4B	5A	5B	6A	6B	7A	7B	8A	8B	9	10A	10B	10C	11A	11B	11C	12A	12B	12C	13A	13B	13C	14A	14B	14C	15
B-1	Determine Eng Pneu Syst Characteristics	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
B-2	Demonstrate SPTS (1, 2 and 3 Starts)	1	1	2	3	1	1	2	1	2	1	2	1	2	1	2	1	1	2	3	1	2	3	1	2	3	1	2	3	1	2	3	1
B-3	Demonstrate Tapoff Turbine Concept	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
B-4	Evaluate J-2S Mainstage Perf Over PU Range	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
B-5	Determine J-2S Eng St Transients (I/M to MS)	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
B-6	Determine J-2S Eng Cutoff Transients	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
B-7	Evaluate J-2S Eng I/M for Cutoff Slosh Control	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X								X	X	X	X	
B-8	Demonstrate LOX Depletion Cutoff													X															X				X
B-9	Determine First Burn I/M Chilldown Time and Perf	X	X			X	X		X		X		X		X	X	X	X			X			X			X			X	X	X	X
B-10	Determine Propellant Chilldown Press. and Temp Limits for First Burn Idle Mode	X	X			X	X		X		X		X		X	X	X	X			X			X			X			X	X	X	X
B-11	Determine Restart I/M Chilldown Time and Perf			X	X			X		X		X		X					X	X		X	X		X	X		X	X				
B-12	Determine Propellant Chilldown Press. and Temp Limits for Restart Idle Mode			X	X			X		X		X		X					X	X		X	X		X	X		X	X				
B-13	Simulate Simultaneous Restart Idle Mode for Ambient Repress																								X	X							
B-14	Determine LOX Tank Press. Syst Perf	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
B-15	Determine LH_2 Tank Press. Syst Perf	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
B-16	Establish Eng Cont He Sphere Loading and Venting Procedures	X	X			X											X																
B-17	Demonstrate Adequate TC Jacket and Turbopump Purge	X	X			X											X																
B-18	Demonstrate Adequate LOX Dome Purge	X	X			X											X																
B-19	Establish Test Op and DC Procedures for Static Test and Acceptance Firing	X	X	X	X	X									X	X														X	X	X	X
B-20	Establish Test Op and DC Procedures for Launch												X	X													X	X	X				
B-21	Determine Prop Tank Pressures Required for Sea Level Idle Mode Testing	X	X	X	X																												

* Test Run Time (Idle Mode/Mainstage/Idle Mode) in Seconds.

10.3.7.2 S-IVB/J-2S Stage Acceptance Test At STC

The S-IVB/J-2S Acceptance Test Program at the Sacramento Test Center (STC) will provide basic S-IVB/J-2S and GSE systems integration verification and flight performance prediction data. It is proposed that acceptance testing be accomplished in lieu of all systems testing and to verify S-IVB single, double and triple burn modes. The number of initial stages to be acceptance tested is subject to static firing and flight successes. For study purposes, three stages were selected as a minimum number for acceptance testing. This number is somewhat arbitrary and it is recommended that following initial acceptance tests the program be reviewed with NASA to establish whether acceptance firings be 1) continued, 2) performed intermittently at a rate not presently identifiable, or 3) discontinued entirely. The objectives and tests defined for the first three stages are shown in Table 10.3.7-VII. The test objectives listed for each test are only those associated with the S-IVB/J-2S modifications and are in addition to, and/or replace current S-IVB/J-2 stage acceptance test objectives. All current test objectives of removed systems will be deleted. Further definition of each objective is presented in this paragraph and contains a justification for the testing, a description of the tests required, and the type of information to be obtained. This information should not be construed to mean these are the only data that is to be obtained, nor does it imply that the measurements listed will be the only ones required. However, it does serve to indicate the magnitude, scope, and importance of the objectives.

Two tests have been defined for acceptance testing of the first S-IVB/J-2S stage to simulate a LOR mission. The first test, a 70-second duration run, serves as a system shakedown, an initial S-IVB/J-2S and GSE integration verification, and initial systems calibration. The 70-second run time is the minimum time required to achieve good engine steady state data for calibration purposes. Hydrogen system leak checks and a thorough data analysis will be completed prior to loading for the next test. The second test will be approximately 380-seconds duration to complete the equivalent of a mission burn time on the stage. This test will also serve as a calibration test to verify and augment the test results of the first test and demonstrate run-to-run performance repeatability. The PU system will be operated at different levels for each of the two tests above, providing two sets of data points.

A single, full duration firing is defined for acceptance testing of the second S-IVB/J-2S stage for systems reliability demonstration, and systems operational and performance verification. The S-IVB/J-2S stage systems calibration and performance data acquired will be used for comparison with the first stage testing data to verify system stage-to-stage performance repeatibility and to establish flight performance prediction techniques to apply to later stages which may not be acceptance fired.

TABLE 10.3.7-VII. S-IVB/J-2S ACCEPTANCE TEST PROGRAM

Test	Objective	Stage 1		Stage 2	Stage 3			Subs
		1/70/2*	1/380/2	1/450/2	1/150/2	1/70/2	1/250/2	1/450/2
A-1	Determine Eng Pneu Syst Characteristics	X	X	X	X	X	X	X
A-2	Demonstrate SPTS (1, 2, and 3 Restarts)	1	2	1	1	2	3	1
A-3	Demonstrate Tapoff Turbine Concept	X	X	X	X	X	X	X
A-4	Evaluate J-2S Mainstage Perf	X	X	X	X	X	X	X
A-5	Determine J-2S Eng St Transients (I/M to MS)	X	X	X	X	X	X	X
A-6	Determine J-2S Eng Cutoff Transients	X	X	X	X	X	X	X
A-7	Demonstrate First Burn I/M Chilldown Perf	X	X	X	X	X	X	X
A-8	Determine LOX Tank Press. Syst Perf	X	X	X	X	X	X	X
A-9	Determine LH_2 Tank Press. Syst Perf	X	X	X	X	X	X	X
A-10	Verify Eng Cont He Sphere Loading & Venting Procedures	X	X	X	X	X	X	X
A-11	Demonstrate Adequate TC Jacket & Turbopump Purge	X	X	X	X	X	X	X
A-12	Demonstrate Adequate LOX Dome Purge	X	X	X	X	X	X	X
A-13	Verify Test Op & CD Procedures for Acceptance Firing	X	X	X	X	X	X	X
A-14	Establish Test Op & CD Procedures for Launch	X	X	X	X	X	X	X

*Test Run Time (Idle Mode/Mainstage/Idle Mode) in Seconds.

10.3.7.2 (Continued)

Acceptance firing of the third S-IVB/J-2S stage will be performed on the first synchronous mission stage and will include three firings to simulate a synchronous mission. PU operation and engine calibration will closely simulate the performance profile of the three burn mission and will provide additional calibration data for different performance levels. As with the tests on the first stage, sufficient time will be allowed between tests to allow adequate hydrogen system leak checks and thorough data analysis.

This testing provides six stage tests to confirm the S-IVB/J-2S integration and performance and, excluding the PU variations, provides three stage-to-stage and four test-to-test performance repeatibility verification tests.

The idle mode starts on the restart mission simulation tests will be limited to a short idle mode (similiar to the first burn start simulation of 1 to 3 seconds); since extended idle mode testing is not considered practical from two aspects:

a. First burn idle mode and restart idle mode cannot be adequately simulated, even to the degree demonstrated at AEDC and MSFC, and would not add significant data or confidence for later flight of the stages.

b. The 40 psia LH2 tank pressure required for extended idle mode testing cannot be achieved on a flight stage during sea level testing without temporary modifications to the LH_2 vent and relief valves. This type of modification is not acceptable to MDAC and would violate the acceptance test criteria of verifying complete S-IVB/J-2S stage and GSE integration.

OBJECTIVE A-1: Determine Engine Pneumatic System Characteristics for Nominal LOR Mission and a Nominal Synchronous Orbit Mission.

This objective will require the following type of information:

a. Determine helium consumption (mass) for start, steady state, and cutoff, for each of the burns required for the above missions and the following phases of operation:

   1. Idle mode start
   2. Idle mode operation
   3. Mainstage starts
   4. Mainstage operation
   5. Mainstage cutoff (mainstage to idle mode)
   6. Engine cutoff (mainstage to zero thrust)
   7. Idle mode cutoff

10.3.7.2 (Continued)

b. Determine helium residuals available for propellant dumping and passivation.

c. Establish sequence, procedures, and time required to blow down (safe) the helium sphere after last burn cutoff (max, min and nominal blowdown history).

d. Establish minimum allowable pneumatic system pressures for successful operation of the phases listed under step a.

Typical data required to verify this objective includes:

a. Engine control sphere temperature

b. Engine control sphere pressure

OBJECTIVE A-2: Establish Mainstage Start Characteristics for First, Second, and Third Burn Start.

Since the SPTS concept is new to the S-IVB stage, information is required on the performance of this method, its effect on engine start, and any S-IVB stage interactions. Acceptance test data will supplement Battleship test results. The following type of information is required for the reasons noted:

a. SPTS chamber pressure history (pressure, burn time, and characteristic pressure buildup and decay, run-to-run and engine-to-engine deviations) to be used in conjunction with engine basic performance (thrust buildup) to reconstruct test data and predict performance. Effects on rate of engine thrust buildup are also desired to determine engine side loads, stage thrust structure loading, structural and flow effects on inlet ducts (especially in transition to tapoff gas bootstrap operation).

b. Pump performance (stall margin) with SPTS system.

Typical data required to verify this objective includes:

a. Temperature SPTS Casing No. 1

b. Temperature SPTS Casing No. 2

c. Temperature SPTS Casing No. 3

d. Pressure SPTS Chamber No. 1

e. Pressure SPTS Chamber No. 2

f. Pressure SPTS Chamber No. 3

10.3.7.2 (Continued)

g. Temperature fuel turbine inlet

h. Time from engine start command to attainment of mainstage (90 percent thrust).

i. Engine performance data during transient; including pump speeds, pressures, and flowrates.

OBJECTIVE A-3: Demonstrate the Concept of Tapping Off Thrust Chamber Gas and Utilizing it to Drive the Turbopump Turbines.

The tap-off-turbine concept is new to the S-IVB stage. Information is required on the performance and stability of this method. Its effect on engine start (start transients), and steady state stability (with PU perturbations). Tapoff flow will be calculated and used in conjunction with other engine performance parameters to calculate overall engine performance for test reconstruction and to make performance predictions. Acceptance test data will be used to supplement Battleship test results.

Typical data required to verify this objective includes:

a. Pressure tapoff gas

b. Temperature tapoff gas

c. Fuel turbine inlet temperature

d. Oxidizer turbine inlet temperature

e. Fuel turbine inlet pressure

f. Oxidizer turbine inlet pressure

g. Vibration levels

h. Environmental (hardware) temperatures

OBJECTIVE A-4: Evaluate J-2S Engine Mainstage Performance.

The following information must be obtained to enable effective post test reconstruction of engine performance, and to predict engine flight performance:

a. Steady state performance variations; including thrust, EMR, propellant consumptions, and ISP.

b. Gain values for operation at expected EMR (stage effects) including variations of all interface parameters.

10.3.7.2 (Continued)

c. Engine response (thrust) characteristics to changes in EMR (PU effects).

d. Sea level test data to derive conversion factors to predict flight performance from stage acceptance test data.

e. Thrust stability (open loop/closed loop).

Typical data required to verify this objective includes:

a. Chamber pressure

b. Thrust history

c. Engine propellant flowrates

d. Engine vibration and side loads

e. Hot gas turbine system characteristics

f. Turbopump characteristics

OBJECTIVE A-5: Determine J-2S Engine Start Transients from Idle Mode to Mainstage for First Burn Start and Restart.

Acceptance test data will supplement Battleship test results, providing additional data points for flight stage predictions. The following information is needed to enable effective post test reconstruction of engine performance, and to predict engine flight performance:

a. Start impulse

b. Performance variations; including thrust, EMR, propellant consumption, and ISP.

c. Engine response characteristics to changes in EMR.

d. Test data to derive conversion factors to predict flight performance from stage acceptance test data.

e. Thrust stability

f. Thrust alignments (stage moments).

10. 3. 7. 2 (Continued)

Typical data required to verify this objective includes:

a. Propellant flowrates (gpm)

b. Thrust history

c. Chamber pressure

d. Engine vibration and side loads

e. Thrust rise rate (time to 90% thrust)

f. Hot gas turbine system characteristics

g. Turbopump characteristics

OBJECTIVE A-6: Determine J-2S Engine Cutoff Transients for Mainstage to Idle Mode, Mainstage to Zero Thrust, and Idle Mode Cutoff.

Acceptance test data will supplement Battleship test results. The following information is needed to enable post test reconstruction of engine performance, and to predict engine performance.

a. Cutoff impulse

b. Cutoff thrust decay rate and stability (time to zero thrust)

c. Test data to derive conversion factors to predict flight performance from stage acceptance test data.

d. Propellant consumption

e. Thrust alignment (stage moments)

Typical data required to verify this objective includes:

a. Propellant flowrate

b. Thrust history

c. Chamber pressure

d. Engine vibration and side loads

e. Hot gas turbine system characteristics

f. Turbopump characteristics

10. 3. 7. 2 (Continued)

OBJECTIVE A-7: Demonstrate Idle Mode Chilldown for First Burn Start.

Acceptance test data will supplement Battleship test results of ground chill effects prior to first burn start. The following type of information is required to satisfy the above objective:

a. NPSH conditions at the pump inlets

b. Thrust history (idle mode start to mainstage operation) and thrust stability

c. Chilldown duration

d. Engine propellant consumption, mixture ratio and specific impulse during idle mode.

e. Start impulse (including rate and stability)

Typical data required to verify this objective includes:

a. Engine chamber pressure, propellant flowrates during idle mode

b. Propellant temperatures and pressures at engine pump inlets and outlets, pump inlet duct feed hardware temperatures, and engine chilldown associated hardware temperatures.

c. Thrust (sea level and altitude).

d. Nominal chilldown duration (time required to meet specification start (mainstage) requirements).

e. Engine vibration and side loads.

OBJECTIVE A-8: Determine the Performance Characteristics of the LOX Tank Pressurization System during Engine Mainstage Operation:

Satisfactory completion of this objective will be achieved when the following conditions have been met:

a. Because of the difference in engine start characteristics associated with the J-2S SPTS and tapoff turbine concept, determine the ullage pressure recovery rate during engine start transition.

b. Because of the higher LOX flowrate associated with the higher performance of the J-2S engine and the heat exchanger, demonstrate in-flight pressurization control at 38 to 41 psia.

c. Determine the nature of ullage gas temperature stratification with the increased propellant outflow.

10.3.7.2 (Continued)

Typical data required to verify this objective includes:

a. LOX tank ullage pressure

b. LOX tank ullage temperature

c. Helium flowrate through heat exchanger

d. Inlet and outlet helium temperature at the heat exchanger

e. Overcontrol and undercontrol energy rates

f. Overcontrol and undercontrol pressure rise and collapse rates

OBJECTIVE A-9: Determine the Performance Characteristics of the LH_2 Tank Pressurization System during Engine Mainstage Operation

The increased performance of the J-2S engine makes it necessary to evaluate the capability of the LH_2 tank pressurization system during the start transient and mainstage operation of the J-2S engine. The hydrogen gas used as a pressurant is tapped off of the fuel injector at an increased pressure due to the higher thrust chamber pressure associated with the J-2S engine. Because of the higher tapoff pressure, it is necessary to verify that the LH_2 pressurization system will function properly in providing the required tank pressure. Satisfactory completion of this objective will be achieved when the following conditions have been met:

a. Because of the difference in engine start characteristics associated with the J-2S SPTS and tapoff turbine concept, determine the ullage pressure recovery rate during engine start transition.

b. Because of the higher LH_2 flowrate associated with the higher performance of the J-2S engine, demonstrate in-flight pressurization control at 28 to 31 psia.

c. Determine ullage gas temperature stratification associated with the increased propellant outflow.

d. Determine step pressurization performance.

Typical data required to verify this objective includes:

a. LH_2 tank ullage pressure

b. LH_2 tank ullage temperature

c. Flowrate of the pressurant

d. Temperature of the pressurant

10.3.7.2 (Continued)

e. Pressure of the pressurant

f. Pump inlet temperature

OBJECTIVE A-10: Verify Loading and Venting Procedures for the Engine Control Helium Sphere.

The J-2S engine control helium is stored in a spherical tank of 4,000 cu. in., under 3,100 ±100 psig pressure at -200 to +140°F. The loading flowrate must be limited to 0.1 lbm/sec or less. The GSE/stage/engine system will be flow tested to verify the loading procedures developed in Battleship testing.

The following information must be obtained:

a. Mass of helium loaded

b. Maximum system flowrate

c. Time to load to specified values

d. Time to vent down

Typical data required to verify this objective includes:

a. Engine control helium sphere pressure

b. Engine control helium sphere gas temperature

OBJECTIVE A-11: Demonstrate Adequate Helium Purge for the J-2S Thrust Chamber Jacket and the Turbopumps.

A helium purge of the thrust chamber LH_2 jacket and both turbopumps cavities prior to firing is necessary to remove any air and water vapor present. A helium purge after a static firing removes water vapor from the turbopump cavities and accumulated hydrogen from the LH_2 jacket.

The engine manufacturer specifies that dry helium at 150 ±25 psig and 50° to 150°F (125 SCFM reference) be supplied at the customer connect panel. For static testing, the purge is accomplished by flowing helium from 30 minutes prior to propellant drop and/or propellant loading until engine start, and for 15 minutes following engine cutoff. The GSE/stage/engine system will be flow tested to verify that Battleship test derived design requirements are satisfactory for flight stage testing.

Information required to complete this objective includes:

a. Mass of gas used for typical purges

b. System flow rate (orifice pressure and temperature and/or flow measurement during blowdown).

10.3.7.2 (Continued)

Typical data required to verify this objective includes:

a. Pressure and temperature at customer connect point
b. Purge supply pressure

OBJECTIVE A-12: Demonstrate Adequate Purge to the LOX Dome.

A purge of the LOX dome is necessary before a firing to remove water vapor, and after firing to prevent accumulation of hydrogen in the dome. Nitrogen will be used for the purge. The engine manufacturer specifies a 30-minute purge with dry nitrogen at 150 SCFM, and 100° to 150°F (600 ±25 psig) before propellant drop and/or propellant loading and a 15-minute purge at 150 SCFM after firing. The LOX dome purge is also required for the full duration of any hold period in the firing sequence, i.e., for any period after a cutoff and prior to a restart. The GSE/stage/engine system will be flow tested to verify that the purge media and hardware modifications meet the purge requirement.

Information required to complete this objective includes:

a. Mass of gas used for typical purge
b. System flowrate (orifice pressure and temperature and/or flow measurements during blowdown).

Typical data required to verify this objective includes:

a. Pressure at customer connect point
b. Temperature at customer connect point
c. Purge supply pressure

OBJECTIVE A-13: Verify Test Operation and Countdown Procedure for Acceptance Firings.

Verify that the acceptance countdown procedures developed in Battleship testing allow sufficient time to complete each necessary step of the countdown. The existing S-IVB countdown procedures will be used where applicable, the new requirements identified, and integrated into a new procedure. This procedure will have been tested and verified during Battleship testing so that conversion to actual production acceptance test procedures will be smooth and efficient. Verify integration of purge procedures, redline monitoring requirements, sequencing and hold capability.

OBJECTIVE A-14: Establish Test Operations and Countdown Procedures for Launch.

Derive new procedures emanating from the new J-2S installation and S-IVB stage modifications, and integrate into existing sequences. These and Battleship-developed procedures will be tested during simulated countdowns, where particular attention will be given to integration of the following.

10.3.7.2 (Continued)

a. New purge procedures

b. Liftoff requirements

c. Items that may require a hold

d. Launch hold time capability

e. Recycle capability

f. Recycle points

g. Overall time to complete the countdown

Particular emphasis will be placed on obtaining a reasonably simple, but effective countdown procedure with sufficient flexibility to cope with any anticipated complexities that may evolve in an actual KSC launch countdown.

10.3.7.3 Flight Testing

The first S-IVB stage using the J-2S engine will be man-rated. Therefore, a high level of confidence in the design changes and overall reliability is mandatory. Verification of the S-IVB/J-2S performance, flight verification of new concepts and GSE systems integration will be obtained from the first flight. Data obtained from this flight will be compared to the data generated during extensive ground testing at AEDC and MSFC for performance verification and to provide data for future flight predictions.

Special emphasis will be placed on verification of first burn start at zero-g and restart at zero-g with "hot and/or dry" ducts. This will be the first actual demonstration of these start concepts with flight conditions. Other areas of particular interest will be propellant control and propellant conditions at the pump inlets during S-II/S-IVB separation and prior to mainstage start and restart. The early flights will have extra instrumentation for systems performance verification, for stage and engine hardware environmental temperature histories and to establish margins of performance. The data will be used to verify performance predictions and clarify environmental envelopes.

The following flight test objectives must be achieved to verify satisfactory operation of the J-2S engine and the S-IVB stage. These objectives are in addition to current J-2/S-IVB stage flight test objectives.

<u>J-2S Engine Performance and Conditioning</u>

Determine the J-2S engine ground chilldown, idle mode chilldown, start, and cutoff performance characteristics. Evaluation of the following will verify this objective:

a. J-2S engine duct and turbopump chilldown performance on the ground prior to liftoff and during idle mode operation prior to start and restart. The environmental heating of the engine during boost and orbital coast.

10.3.7.3 (Continued)

b. Engine conditioning provided by the thrust chamber jacket purge, turbopump and LOX dome purges required during prelaunch preparations.

c. The J-2S engine control helium sphere prepressurization, loading, and sphere conditions prior to and at liftoff, and at engine start command. Environmental heating of the control helium sphere during boost and orbital coast. Helium consumption during:

1. Idle mode operation
2. Mainstage start
3. Mainstage
4. Cutoff

d. The J-2S engine sequencing.

e. Engine performance including thrust, specific impulse, total impulse, and propellant consumption during:

1. Idle mode operation
2. Mainstage start
3. Mainstage
4. Cutoff

f. Propellant consumption by the J-2S engine using flow integral.

g. The performance of the solid propellant turbine start system during engine start and restarts. Effects of environmental heating and vibration during boost and orbital coast on the performance of the system.

h. The performance and stability of the turbopumps during mainstage operation. Satisfactory operation of the turbopumps will provide in-flight verification of the concept of tapping off thrust chamber combustion gases.

i. The effect of cryogenic and/or ambient repressurization upon the extended idle mode chilldown performance during restart.

j. The effect of the deletion of APS ullaging upon J-2S engine restart and the capability of idle mode operation to control the propellants prior to restart.

k. The performance of idle mode in providing control of propellant unseating and sloshing after mainstage cutoff.

10.3.7.3 (Continued)

LH$_2$ Propellant Feed System

Demonstrate the capability of the LH$_2$ feed system to provide sufficient LH$_2$ and net positive suction head during idle mode and mainstage operation of the J-2S engine. Evaluation of the following will verify this objective:

a. LH$_2$ temperature and pressure and tank ullage pressure during loading operations and at liftoff.

b. Prepressurization of the LH$_2$ tank prior to launch.

c. Transition from ground prepressurization to onboard flight pressurization system to provide tank ullage pressure during engine operation.

d. Conditions of propellant supplied to the J-2S engine LH$_2$ pump inlet during idle mode and mainstage operation.

e. Idle mode chilldown of the LH$_2$ propellant feed system and associated engine hardware.

f. Repressurization of the LH$_2$ tank prior to restarts.

g. Orbital boiloff mass.

h. Boost and orbital heating of the LH$_2$ propellant feed system.

i. Pressurization control module operation.

LOX Feed System

Demonstrate the capability of the LOX feed system to provide sufficient LOX and net positive suction head during idle mode and mainstage operation of the J-2S engine. Evaluation of the following will verify this objective:

a. LOX temperature, pressure and tank ullage pressure during loading operation and at liftoff.

b. Prepressurization of the LOX tank with ambient helium prior to launch.

c. Transition from ground prepressurization to onboard flight pressurization system to provide LOX tank ullage pressure during engine operation.

d. Pressurization control module operation.

e. Pressure and temperature of the cold helium supply.

f. J-2S engine heat exchanger performance.

10.3.7.3 (Continued)

g. Conditions of propellant supplied to the J-2S engine LOX pump inlet during idle mode and mainstage operation.

h. Idle mode chilldown of the LOX propellant feed system and associated engine hardware.

i. Boost and orbital heating of the LOX propellant feed system.

j. Repressurization of the LOX tank prior to restarts.

Pneumatic Control System

Determine the amount of stage pneumatic helium required to provide pneumatic power throughout the mission.

Evaluation of the following will verify the objective:

a. Pressure and temperature of the stage ambient helium supply.

b. Regulated control pressure.

c. Actuation of pneumatic valves.

Hydraulic Power System

The evaluation of the hydraulic power system will include the following:

a. Verification that adequate pressurized fluid flow is provided by the redesigned main hydraulic pump to the servo-actuator and that hydraulic system pressures are maintained within expected limits.

b. Verification that fluid temperature is maintained within expected limits during system operation.

c. Verification that auxiliary hydraulic pump motor air pressurization is maintained.

d. Verification of auxiliary hydraulic pump engine positioning capability during idle mode and restart.

Hydraulic Servo System

The evaluation of the hydraulic servo system will include the following:

a. Verify the adequacy of actuator artificial damping mechanism performance

b. Verify the adequacy of present compensation for thrust vector deflection errors caused by gimbal "slop" and thrust structure compression effects.

10.3.7.3 (Continued)

c. Evaluate the effects of thrust misalignment and thrust eccentricity errors on actuator performance.

d. Determine and evaluate actuator start transient loads during initial start and restart.

e. Determine and evaluate gimbal friction during engine burn after gimbal bearing has been exposed to space environment.

f. Compare critical actuator component temperatures with predicted values.

g. Verify proper pitch and yaw actuator responses to commands.

h. Evaluate the effects of IU command errors in the non-S-IVB burn modes on actuator performance.

i. Evaluate actuator deflections during non-S-IVB burn modes.

Stage Separation

Verify clearance distance between S-II/S-IVB stage during separation. Demonstrate adequacy of stage attitude control and propellant control systems during separation and prior to mainstage start. This objective will be achieved by determining the following:

a. Lateral clearance between stages.

b. Separation distance history between stages.

c. Evaluation of attitude errors and rates experienced during separation and first idle mode.

d. The effect of the deletion of the solid ullage rockets upon propellant control during separation.

e. Performance of idle mode in providing propellant control prior to mainstage start after separation.

f. Propellant conditions at the LOX and LH_2 pump inlets and ullage pressures prior to the first mainstage start.

Thrust Vector Control System

Demonstrate proper performance of the main engine control system during S-IVB flight. This objective will be achieved by evaluation of the following:

a. Response of the thrust vector control system to commands from the instrument unit.

b. Response of the control system sensors and networks.

10.3.7.3 (Continued)

c. Verification of control system stability during S-IVB flight, including controlability immediately after separation.

d. Evaluation of transient regions of flight (e.g., separation, guidance initiation).

e. Demonstrate proper main engine positioning prior to engine restart during extended idle mode operation.

Trajectory/Propulsion Compatibility

Verify compatibility of the observed trajectory and S-IVB propulsion system performance. This objective will be achieved by determining the following from trajectory data:

a. S-IVB stage thrust, specific impulse, and mass flow.

b. Vehicle mass at ignition and cutoff.

c. S-IVB stage thrust vector misalignment.

Stage Sequence of Events

Verify proper S-IVB acknowledgment of sequence commands issued from the IU. This objective will be verified by comparing IU command times to stage monitored command times.

Propellant Utilization System

Demonstrate the PU system performance for inflight propellant management as defined by the criteria listed herein.

a. Demonstrate the ability of the PU system to provide propellant management and to deplete residuals within a predetermined value.

b. Demonstrate closed-loop PU operation in the programmed mixture ratio (PMR) mode during first burn. Following a three-orbit coast and restart, control the EMR to a nominal reference mixture ratio of 5.0:1 during second burn.

Stage Aero/Thermodynamics

Determine stage Aero/Thermodynamics environments during all phases of flight. This objective will include the following:

a. Stage thermal environment and the response of structure and components subjected to cryogenics, aerodynamic heating, solar and albedo radiation,

10.3.7.3 (Continued)

and plume impingement are to be investigated. Areas to be evaluated include:

1. Forward skirt (sync mission)
2. LH2 tank (sync mission)
3. Aft skirt and interstage
4. Thrust structure
5. APS (sync mission)
6. O_2-H_2 burner (sync mission)
7. J-2S engine
8. Propellant feed ducts

b. The propellant heat input during ground hold, boost, powered flight, and orbital coast.

c. The internal pressures within the forward and aft compartments will be compared with postflight simulations and design data.

10.3.8 Reliability

The J-2S Engine features result in a decisive simplification of S-IVB stage subsystems. An evaluation of the effects of these simplifications in terms of (1) potential improvements in current S-IVB reliability predictions and (2) reduction in the number of critical failure modes are the major objectives of the reliability study.

The secondary objective of the reliability study is an evaluation of Lunar Orbit Rendezvous (LOR) and Synchronous Orbit (Sync) missions which are operationally more difficult than current two-burn and three-burn missions, primarily due to longer mission durations.

To accomplish the above objectives, criticality evaluations were made comparing J-2 and J-2S equipped stages for the candidate missions, providing reliability indices (in terms of criticality) and tabulations of relative numbers of failure modes for each mission and vehicle.

10.3.8.1 Background, Definitions, Guidelines and Assumptions

Saturn S-IVB stage modifications resulting from use of the J-2S engine include deletions of flight critical items associated with removal of engine preconditioning and ullage positioning functions and reductions in the criticality of components which no longer are required to provide control for these functions.

10.3.8.1 (Continued)

An evaluation of the impact of these changes on S-IVB Stage reliability was provided in Douglas Report DAC-56372. This evaluation was based on stage configurations, mission descriptions, and reliability models which existed in 1966. To establish an improved basis for decision, it is now appropriate to reevaluate the earlier reliability analysis to accommodate the results of S-IVB flight history acquired since the earlier report, and changes in missions, configurations, and reliability analysis procedures.

Definitions

To provide an understanding of terminology used in the analyses, the following definitions are provided:

- Probability (P) - The likelihood of occurrence, expressed as a value $0 \leq P \leq 1$; where 0 represents impossibility and 1 represents certainty.
- Stage Reliability (R) - Probability that stage loss will not occur.
- Criticality Number (CN) - The rate resulting from the failure probability of a component/system expressed as the number of stages lost per million missions.
- Reliability Analysis Model (RAM) - An MDAC-WD analysis containing diagrams, failure effect analyses, and reliability predictions developed for the S-IVB evaluation in 1968.
- Reliability Engineering Model (REM) - An MDAC-WD report containing block diagrams, failure effect analyses, and reliability predictions written for Flight AS-501 in 1966.
- Flight Critical Item (FCI) - A component piece of hardware; the failure of which may lead to loss of stage in flight.
- Mission Failure Mode (FM) - The failure of an FCI in a specific mode of required operation which causes loss of the mission or the vehicle.

Guidelines and Assumptions

The method used to evaluate the reliability of the Saturn S-IVB stage is a criticality analysis. In this analysis, criticality numbers are generated to determine the stage-loss probabilities that are chargeable to each of the Flight Critical Items. The factors which have been taken into account in calculating the criticality of the Flight Critical Items (FCI) are the following:

a. Generic Failure Rate (λ) (assumed to be constant)

b. Environmental Adjustment Factor (K)

During J-2S Main Stage	K = 1000
During J-2S Idle Mode	K = 100
During Coast	K = 10

10.3.8.1 (Continued)

c. Failure Mode Frequency Factor for each failure mode (α).

d. Operating times or cycles during each of the flight phases.

e. Conditional probabilities of stage-loss, given the designated failure mode has occurred (β).

The criticality number is that portion of an item failure probability which contributes to the probability of "stage loss" multiplied by 10^6. Thus, criticality is expressed in losses per million flight, chargeable to the FCI or subsystem being analyzed. In the Reliability Engineering Models (REM) and Reliability Analysis Models (RAM) reports, stage reliability and "no-stage-loss" probability are considered equivalent. Correspondingly, "stage-loss" probability and the total criticality number multiplied by 10^{-6} are considered equivalent.

The actual performance of the criticality number calculation is accomplished by a computer program written in Fortran IV language using the factors listed above. These factors are obtained from historical failure data for stage components found to be flight critical through failure mode and effects analyses, and operating times determined from mission time lines for the candidate missions.

Current Reliability Analysis Models (RAM) for S-IVB two-burn and three-burn missions were used as baselines for the analyses contained in this report. The baseline analyses (AS-503 and AS-504) provide the most recent information on S-IVB reliability, and their mission profiles approximate the candidate LOR and Synchronous missions to be evaluated in this study. The data used from these models were the computer printouts of the failure effects and criticality analysis, the criticality ranking of flight critical items, and critical single failure point lists. These data were first modified to reflect equal levels of analysis details and the longer mission profiles of the candidate missions to provide a criticality determination for a standard J-2 Engine equipped stage. The second step included analysis of failure effects which would be modified or removed as a result of equipment removals, reduced requirements, and subsystem changes to provide a criticality determination for a J-2S Engine equipped stage.

Detailed analyses were performed for the 25 most critical components, and those components directly changed by J-2S implementation; accounting for over 80 percent of the total stage mission criticality. The balance of stage component criticalities were adjusted using adjustment factors derived from the detail analyses; 1.26 for the Synchronous Mission and 1.13 for the LOR Mission. These factors account for the longer coasts for the candidate missions; when compared to the AS-503 and AS-504 RAM time lines.

10.3.8.2 Summary of Results

Criticality reductions due to implementation of the J-2S engine are 15 percent for the Lunar Orbital Rendezvous mission, and 16 percent for the Synchronous Orbit mission. The reductions principally reflect equipment and functional deletions in the Chilldown, Propellant Feed, Auxiliary Propulsion, and Electrical subsystems. In addition, critical failure modes causing actual mission loss are reduced approximately 25 percent as shown in the Reliability Summary Table 10.3.8-I and 10.3.8-II. Figure 10.3.8-1 also displays the stage reliabilities for the LOR and Sync missions. Reliability improvements are indicated for both missions when the J-2S is implemented. For the synchronous mission using a J-2 engine equipped stage, the reliability prediction is sisnificantly below the goal of 0.95. The J-2S equipped stage reliability prediction exceeds the goal of 0.95 for the synchronous mission.

Tables of failure mode deletions, item criticality reductions, and criticality ranking are given in DAC-56749.* Table 10.3.8-III summarizes criticality information for the 25 most critical components.

Table 10.3.8-III explains some of the differences between the results of this study and those provided in the original J-2S/S-IVB study (DAC-56372). First, Rocketdyne information dictated a constant criticality number for the J-2S engine used in this study, whereas in the earlier study the criticality numbers reflected the reducted complexity of the J-2S engine and adjustments for multiple burns. Second, criticality reductions for equipment deletions in the twenty-five most critical components amount to 2600 units for the LOR mission in this study, compared to 4800 units in the earlier study. The lower criticality number (2600) reflects knowledge gained through operational experience. Consequently, a smaller percentage reduction in stage criticality is now observed for J-2S implementation. In summary, the failure modes deleted by J-2S have been evaluated as less critical, or have lower occurrence frequencies than in the earlier report.

Finally, the analysis presented in this report evaluates changes in equipment operations and configurations which were not identified in the earlier report. These include the increased duty cycles on the hydraulic subsystems. This subsystem, therefore becomes more critical due to J-2S implementation. Evaluation of ullage pressure system, which has a considerably higher criticality for the Synchronous mission than for the LOR mission is also included in the new analysis.

10.3.8.3 Subsystem and Component Evaluations

Evaluations of S-IVB subsystems and components which have undergone major revisions for J-2S engine applications or for the Synchronous mission are presented in this section. Other items of equipment which were removed or whose functional requirements were modified as a result of J-2S application

*DAC-56749, J-2S Implementation on the Saturn V/S-IVB Stage - LOR and Synchronous Missions, March 1969.

TABLE 10.3.8-I. RELIABILITY SUMMARY: LOR MISSION

Item Code *	Subsystem	S-IVB/J-2 Criticality	S-IVB/J-2S Criticality	#Failure Modes Deleted with J-2S	#Launch Delays Deleted	#Actual Losses FM Deleted
1A	Propellant Fill & Drain	2	2	0/19	0/14	0/2
1B	Ullage Pressure	1145	1117	0/113	0/75	0/16
1C	Feed & Chilldown	901	0	58/58	24/24	12/12
1D	Pneumatics	1822	1515	21/160	8/82	6/12
1E	O_2/H_2 Burner	319	319	0/60	0/35	0/0
2G	Hydraulics	3427	4961	0/100	0/41	0/14
2H	APS	5934	3676	15/154	0/24	0/5
4J	Electrical Power	3400	1670	2/31	0/21	2/5
4K & 4L	Electrical Control & Distribution	8082	5860	55/229	21/144	10/46
4N	Propell. Utilization	3174	2727	4/12	0/10	0/0
4P	Repressurization	38	38	0/4	0/14	0/1
6T	Severance	1	1	0/8	0/0	0/0
6U	Lwr. Stg. Rvr. Thrust	1316	1316	0/33	0/0	0/0
6V	Ullage Positioning	250	0	41/41	0/0	0/0
7X	Command	40	40	0/34	0/21	0/2
		29851	23242	196/1056 19%	55/505 11%	30/116 26%
	Engine	14000	14000			
	Total Criticality	43851	37242			
	Reliability	.9562	.9628			

*Item code numbers are based on AS-503 and AS-504 RAMs.

NOTE: For the S-IVB Stage without the engine, a 22% criticality reduction results from J-2S implementation.

TABLE 10.3.8-II. RELIABILITY SUMMARY: SYNCHRONOUS MISSION

Item Code*	Subsystem	S-IVB/J-2 Criticality	S-IVB/J-2S Criticality	#Failure Modes Deleted with J-2S	#Launch Delays Deleted	#Actual Losses FM Deleted
1A	Propellant Fill & Drain	2	2	0/19	0/14	0/2
1B	Ullage Pressure	6132	5549	0/113	0/75	0/16
1C	Feed & Childown	1429	0	58/58	24/24	12/12
1D	Pneumatics	2630	2201	22/170	8/82	6/12
1E	O_2/H_2 Burner	560	560	0/60	0/35	0/0
2G	Hydraulics	3909	4109	0/100	0/41	0/14
2H	APS	7589	6742	15/154	0/24	0/5
4J	Electrical Power	5763	2566	2/31	2/21	2/5
4K & 4L	Electrical Control & Distribution	10769	8821	60/281	21/144	10/46
4N	Propell. Utilization	4418	3240	4/12	0/10	0/0
4P	Repressurization	49	49	0/4	0/14	0/1
6T	Severance	1	1	0/8	0/0	0/0
6U	Lwr. Stg. Rvr. Thrust	1316	1316	0/33	0/0	0/1
6V	Ullage Positioning	250	0	41/41	0/0	0/0
7X	Command	40	40	0/34	0/21	0/2
		44857	35194	202/1073 19%	55/505 11%	30/116 26%
	Engine	14000	14000			
	Total Criticality	58857	49194			
	Reliability	.9411	.9508			

*Item code numbers are based on AS-503 and AS-504 RAMs.

NOTE: For the S-IVB Stage without the engine, a 22% criticality reduction results from J-2S implementation.

TABLE 10.3.8-III. CRITICALITY SUMMARY - "25 MOST CRITICAL COMPONENTS"*

	LOR		SYNC	
	J-2	J-2S	J-2	J-2S
1. Engine (GFE)	14,000	14,000**	14,000	14,000**
2. Sequencer	2681	2310	3530	3365
3. Elec. Assy., P.U.	2350	2146	3140	2390
4. Repress. Control Module	0	0	3191	2843
5. Aft. Power Assy. (56V)	1470	930	2160	2051
6. Actuator, Hydraulic (2 ea)	1812	1812	1919	1919
7. Cont. Vent Control Module	519	491	1941	1708
8. Chilldown Inverters	1180	0	1842	0
9. Batt's. Fwd. & Aft #1	1039	824	1858	1240
10. APS Tank Assy.	1436	1436	1760	1760
11. Engines, 150 #APS	1131	1131	1693	1693
12. Press. Reg, APS	1307	1307	1660	1660
13. Batt, Aft #2	899	564	1474	737
14. Low Press Module APS	1122	1122	1416	1416
15. Motor, Retro Rocket (2 ea)	1316	1316	1316	1316
16. Aux. Hydraulic Pump	1084	1108	1159	1208
17. Inverter, P.U.	509	403	945	628
18. Power Control Module	627	627	914	914
19. Power Mtg. Assy. (28V)	576	440	960	480
20. Engine, 70 # APS	821	0	847	0
21. Control Relay Pkg.	660	660	700	700
22. Lox Chill Pump	300	0	442	0
23. LH_2 Chill Pump	300	0	442	0
24. Accum/Reservoir	157	157	441	441
25. LOX Tank Press. Control	313	313	416	416

* Based on 3 burn synchronous mission.

** See paragraph 10.3.8.3

10.3.8.3 (Continued)

were identified during the course of this study. These equipment removals result in major changes in criticality rankings for the candidate mission. Analysis and results for these deletions and changes are presented in DAC-56749, J-2S Implementation on the Saturn V/S-IVB Stage - LOR and Synchronous Missions, March 1969.

Engine

Criticality evaluation of the J-2S engine was based on the failure effects analysis of the J-2 engine for the SA-503 mission (MSFC Drawing 10M30828). Because a quantitative listing of component failure modes with the resultant criticalities was not available, a count was made of the failure modes which use of the J-2S will eliminate. A listing was then made of new (J-2S) hardware and the failure modes associated with it. It was found that the total number of failure modes has been reduced by 21 percent. It is estimated, then, that the criticality of the J-2 engine (14,000) will be reduced by 21 percent (to 11,000) for the J-2S engine. It is the responsibility of the engine manufacturer to determine the new criticality, therefore, while the remainder of the report states the criticality of the J-2S engine as 14,000, it should be kept in mind that a lower value will probably ultimately be used. The MDAC analysis of the engine is presented in DAC-56749, previously mentioned.

Auxiliary Hydraulic Pump

A detailed analysis has been made of the Auxiliary Hydraulic Pump (AHP) because of its changed duty cycles. The AHP must be turned on every hour during coast periods to insure proper thermal conditioning of the hydraulic fluid. In addition, during idle mode operation, the AHP is the only source of power for positioning of the engine; failure of the AHP during idle mode burn may result in an engine hardover condition and loss of the mission.

It was found that a criticality increase of 86 percent for the AHP has been incurred on the Synchronous mission due to the extended coast periods and the extensive time spent in idle mode. In contrast, there was less change in the AHP's criticality for the shorter LOR mission.

Engine-Driven Hydraulic Pump

The J-2S LOX turbopump during mainstage operation will be turning at a speed greater than the present J-2 pump. Since the main hydraulic pump is driven directly by the turbopump's shaft it must also operate at a higher rotational speed. The increase in pump speed causes an increase in the criticality of the pump. The magnitude of this increase has been identified as 34.6 percent.

10.3.8.3 (Continued)

S-IVB/J-2S Pneumatic Systems Combination

It has been proposed that a provision be made to join the pneumatic systems of the stage and the engine in some manner so that, if one or the other system failed, the mission could be continued. A reliability analysis was performed to evaluate the result of such a change on the criticality of the vehicle.

The description of the "best choice" system, and detailed analysis is presented in DAC-56749. There are potential criticality reductions of nearly 900 and 600 units for the Synchronous and LOR missions, respectively, if implementation of this connection were carried out. However, development considerations and loss of engine commonality with the S-II Stage applications are not considered merited by the system reliability increase and this system change is not recommended.

10.3.9 Summary of S-IVB Design Changes

The major modifications to the S-IVB resulting from J-2S implementation are summarized in Table 10.3.9-I. These modifications apply to both LOR and synchronous missions. Table 10.3.9-II summarizes the additional S-IVB changes dictated by requirements peculiar to the synchronous mission using the J-2S engine. A list of deleted components and other significant items is shown in Table 10.3.9-III. GSE modifications are summarized in Table 10.3.9-IV for Huntington Beach, STC, and KSC.

Proposed stage and engine instrumentation has not met with general agreement among NASA, Rocketdyne, and MDAC. MDAC does not consider the instrumentation presented in the Rocketdyne Interface Criteria Document, R-7211, to be adequate for verification of idle mode chilldown models and engine performance. If the present program policies on flight evaluation and mission support are continued in a follow-on J-2S/S-IVB program, MDAC recommends the instrumentation listed in Tables 10.3.9-V, VI, and VII. If NASA chooses to reduce the amount of preflight and postflight evaluation presently required, it may be possible to reduce this list significantly.

Tables 10.3.9-V, VI, and VII reflect only that instrumentation associated with the J-2S engine and its supporting systems. Other stage peculiar and synchronous mission measurements were discussed in paragraph 10.3.4.

TABLE 10.3.9-I. J-2S/S-IVB IMPLEMENTATION MAJOR MODIFICATIONS

Delete

LOX and LH_2 Chilldown Systems

- 2 Prevalves
- 2 Motor/Pumps
- 2 Shutoff Valves
- 2 Flow Meters
- 2 Actuation Modules
- 5 Check Valves
- 2 Inverters
- 5 Measurements
- Ducts, Diffusers, Fairings, and Mount Panels

Ullage Systems

- 2 Solid Ullage Motors
- 2 Fairings
- 2 Mount Panels
- 2 APS Ullage Engines
- 2 Jettison Devices
- 2 EBW Firing Units, Detonators and CDF Systems

LOX Low Level Sensors

- 3 Sensors and Controls
- Feed-thru Disconnect

Engine Service System

- Start Tank Vent, Relief Line and Disconnect
- Fuel Pump Drain Lines and Disconnect
- Engine Pump Purge Module and Lines

Add/Modify

- 2 Feed Duct Spacers
- LOX Dome Purge Line
- Strengthen Thrust Structure
- Re-orifice Fuel and LOX Tank Pressure Control Modules
- Replace Aft No. 1 Battery with Smaller Battery
- Redesign Main Hydraulic Pump Isolator, Reduce Pump Stroke

TABLE 10.3.9-II. J-2S/S-IVB SYNCHRONOUS MISSION MAJOR MODIFICATIONS

- Add 3 Cold Helium Spheres - Delete 7 Ambient Spheres.
- Add Baffles/Deflectors to LOX and LH_2 Tanks.
- Add Saturn IB Power Amplifiers to T/M Transmitter.
- Add Instrumentation for New Equipment.
- Add Active and Passive Thermal Control Systems.

TABLE 10.3.9-III. NET DELETIONS RESULTING FROM J-2S IMPLEMENTATION

150 Leak Paths	3 Disconnects
15 Valves	1 Pressure Switch
16 Position Indicator Switches	2 Motor/Pumps
2 Actuation Control Modules	7 Redlines

TABLE 10.3.9-IV. GSE MODIFICATIONS RESULTING FROM J-2S IMPLEMENTATION

At Huntington Beach

- Modify Model 348, LOX Tank Access Kit (Also for STC and KSC).
- Modify 4 electrical panels (patchcords and reidentification).

At Sacramento Test Center

- Delete GG overtemp panel. Add tap-off overtemp panel.
- Isolate turbine start bottle supply.
- 32 changes to 3 electrical control panels (patchcords and reidentification).

At Kennedy Space Center

- Isolate turbine start bottle supply.
- Provide 750 psig helium for LOX dome purge.
- Delete solid ullage motor handling kit.
- Add SPTS handling cart.
- 91 changes to 8 ESE panels - G. E. controlled (patchcords, switches, and reidentification).
- Modify Model 400, Aft Battery Handling Kit.
- Modify Model 1900, Component Handling Kit.

TABLE 10.3.9-V. EXISTING T/M MEASUREMENTS RETAINED (ENGINE RELATED)

A. The following existing flight measurements are listed in the J-2S Engine Interface Criteria Document.

No.	Title	No.	Title
C1	Temp - Fuel Turbine Inlet	K5	Event - Mainstage Control Sol On
C2	Temp - LOX Turbine Inlet	K7	Event - Helium Control Sol On
C7	Temp - Helium Tank	K10	Event - ASI Sparks On
C11	Temp - Electrical Control Assy	K12	Event - Engine Ready Signal
C133	Temp - LOX Pump Discharge	K13	Event - Engine Cutoff Lock-in On
C134	Temp - Fuel Dump Discharge	K14	Event - Mainstage OK Press Sw 1
C197	Temp - Instrumentation Package	K118	Event - Main Fuel Valve Open
C200	Temp - Main Fuel Injection	K119	Event - Main Fuel Valve Closed
C215	Temp - LOX Turbine Outlet	K120	Event - Main LOX Valve Open
D1	Press - Thrust Chamber	K121	Event - Main LOX Valve Closed
D8	Press - Fuel Pump Discharge	K157	Event - Mainstage OK Press Sw 2
D9	Press - LOX Pump Discharge	K158	Event - No. 1 Mainstage OK Depress
D19	Press - Helium Tank	K159	Event - No. 2 Mainstage OK Depress
D242	Press - Helium Tank Backup	M6	Volt - Engine Control Bus
F1	Flow - Oxidizer	M7	Volt - Engine Ignition Bus
F2	Flow - Fuel	T1	Speed - Oxidizer Pump
G3	Pos - Main LOX Valve	T2	Speed - Fuel Pump
G4	Pos - Main Fuel Valve		
G10	Pos - PU Valve		

TABLE 10. 3. 9-V (Continued)

B. The following flight measurements were on the J-2 engine, but they are not being provided by Rocketdyne on the J-2S engine. It is recommended that these measurements be retained:

No.	Title	No.	Title
D4	Press - Main Fuel Injection	D7	Press - Oxidizer Turbine Inlet
D5	Press - Main Oxidizer Injection	D18	Press - Engine Regulator Outlet

C. The following stage furnished measurements will be retained on the S-IVB/J-2S stage:

No.	Title	No.	Title
C3	Temp - Fuel Pump Inlet	D2	Press - Fuel Pump Inlet
C4	Temp - Oxidizer Pump Inlet	D3	Press - Oxidizer Pump Inlet
C8	Temp - Heat Exchanger Helium Inlet	G1	Pos - Actuator Piston Pitch
C9	Temp - Heat Exchanger Helium Outlet	G2	Pos - Actuator Piston Yaw
C10	Temp - Engine Area Ambient	K21	Event - Engine Start On
D104	Press - LH_2 Press Module Inlet	K140	Event - Engine Cutoff On
		M60	Volt - PU Valve Control

D. Range changes are required for the following measurements:

No.	Title	No.	Title
D1	Press - Thrust Chamber	F1	Flow - Oxidizer
D4	Press - Main Fuel Injection - Not Listed in J-2S Interface Document	F2	Flow - Fuel
D5	Press - Main Oxidizer Injection - Not Listed in J-2S Interface Document	T2	Speed - Fuel Pump
D8	Press - Fuel Pump Discharge	C7	Temperature - Helium Gas
D9	Press - Oxidizer Pump Discharge		
D104	Press - LH_2 Press Module Inlet		
D242	Press - Eng Cont Helium Sphere Backup		

TABLE 10.3.9-VI. FLIGHT MEASUREMENTS DELETED

The following measurements will be deleted when the J-2S engine is used with the S-IVB stage:

No.	Title	No.	Title
C6	Temp - GH_2 Start Bottle	K11	Event - GG Spark System On
C159	Temp - LOX Circulation Return Line	K20	Event - Eng ASI LOX Valve Open
C161	Temp - Fuel Circulation Return Line	K96	Event - Start Tank Discharge Control
C199	Temp - Thrust Chamber Jacket	K105	Event - Pump Purge Control Backup Press Switch De-Energ
D10	Press - GG Chamber	K110	Event - LOX Prevalve Closed
D17	Press - GH_2 Start Bottle	K111	Event - Fuel Prevalve Open
D50	Press - Engine Pump Purge Reg	K112	Event - Fuel Prevalve Closed
D103	Press - He Inlet LOX Recirc Pump Motor Container	K117	Event - GG Valve Open
D218	Press - LH_2 Chilldown Pump Diff	K122	Event - Start Tank Discharge Valve Open
D219	Press - LOX Chilldown Pump Diff	K123	Event - Start Tank Discharge Valve Closed
D220	Press - Ullage Control Chamber 1-4	K124	Event - Oxid Turbine Bypass Valve Open
D221	Press - Ullage Control Chamber 2-4	K125	Event - Oxid Turbine Bypass Valve Closed
D241	Press - GH_2 Start Bottle Backup Measurement	K126	Event - Oxid Bleed Valve Closed
F4	Flow - Oxid Circ Pump	K127	Event - Fuel Bleed Valve Closed
F5	Flow - Fuel Circ Pump	K131	Event - LOX C/D Purge Reg Backup P/SW De-Energ
G5	Pos - Eng GG Cont Valve	K136	Event - Closed Fuel SOV Chill System
G8	Pos - Oxid Turbine Bypass Valve	K137	Event - Open Fuel SOV Chill System
G9	Pos - Eng Start Sphere Discharge Valve	K138	Event - Open Oxid SOV Chill System
K6	Event - Ignition Phase Control Sol Energ		

TABLE 10.3.9-VI. (Continued)

No.	Title	No.	Title
K139	Event - Closed Oxid SOV Chill System	M28	Freq - LH_2 Chilldown Inverter
K149	Event - Ullage Rocket Jettison 1	M29	Freq - LOX Chilldown Inverter
K150	Event - Ullage Rocket Jettison 2	M40	Volt - LOX Chilldown Inverter AC
K176	Event - Ullage Rocket Ignition 1	M41	Volt - LH_2 Chilldown Inverter AC
K177	Event - Ullage Rocket Ignition 2	M64	Volt - Firing Unit 1 EBW Ullage Rocket Ign
K109	Event - Oxid Prevalve Open	M65	Volt - Firing Unit 2 EBW Ullage Rocket Ign
M26	Volt - Phase A-B LH_2 Chilldown Inverter	M66	Volt - Firing Unit 1 EBW Ullage Rocket Jettison
M27	Volt - Phase A-B LOX Chilldown Inverter	M67	Volt - Firing Unit 2 EBW Ullage Rocket Jettison
		K8	Event - Ignition Detected

TABLE 10.3.9-VII. ADDITIONAL FLIGHT MEASUREMENTS REQUIRED FOR J-2S ENGINE

A. The following new measurements are contained in the J-2S Engine Interface Criteria Document and will be provided by Rocketdyne:

No.	Title	No.	Title
TBD	Press - Idle Mode Chamber - Low	TBD	Event - Mainstage Cutoff Lock-In On
TBD	Event - Idle Mode Valve Open	TBD	Event - SPTS, EBW Firing Units Armed
TBD	Event - Idle Mode Valve Closed	TBD	Event - SPTS 1 Ready
TBD	Event - Fuel Bypass Valve Open	TBD	Event - SPTS 2 Ready
TBD	Event - Fuel Bypass Valve Closed	TBD	Event - SPTS 3 Ready
TBD	Event - SPTS Initiated	TBD	Pos - Fuel Bypass Valve
TBD	Event - Hot Gas Tapoff Valve Open	TBD	Pos - Hot Gas Tapoff Valve
TBD	Event - Hot Gas Tapoff Valve Closed	TBD	Pos - Idle Mode Valve

B. The following new measurements will be furnished by the stage contractor and will be placed on the stage side of the interface:

No.	Title	No.	Title
TBD	Event - Mainstage Start Command	TBD	Event - Prop Dump Sol Energize
TBD	Event - Mainstage Start Sol Energize	TBD	Event - Mainstage Cutoff On
TBD	Event - Idle Mode Cont Sol Energize	TBD	Temp - LOX Engine Interface (160 - 700°R)
TBD	Event - Helium Vent Control Command	TBD	Temp - LH_2 Engine Interface (35 - 700°R)
TBD	Event - Eng Pneu Sys Vent Command		

TABLE 10.3.9-VII. (Continued)

C. It is recommended that the following new measurements be provided on the engine for the purpose of idle mode flowrate and effectiveness evaluations, SPTS evaluation, and environmental condition determinations.

MEASUREMENT	RANGE
Temp - LOX Pump Bearing Coolant	160-700^{o}R
Temp - LH_2 Pump Bearing Sump	35-700^{o}R
Temp - Idle Mode ASI LOX Line	160-700^{o}R
Temp - Idle Mode Injector LOX Line	160-700^{o}R
Temp - Idle Mode ASI LH_2 Line	35-700^{o}R
Temp - LH_2 Film Coolant Injector	35-700^{o}R
Temp - LH_2 Injector (Range Change)	35-700^{o}R
Temp - Fuel Bypass Manifold	35-700^{o}R
Temp - SPTS Case No. 2	300-2000^{o}R
Press - Idle Mode ASI LOX Line	0-50 PSIA
Press - Idle Mode Injector LOX Line	0-50 PSIA
Press - Idle Mode ASI LH_2 Line	0-50 PSIA
Press - LH_2 Film Coolant Injector	0-50 PSIA
Press - LH_2 Injector	0-50 PSIA
Press - Fuel Turbine Inlet	0-1500 PSIA
Press - ASI Combustion Chamber	0-50 PSIA

10.4 INSTRUMENT UNIT

The Instrument Unit is a load bearing slice of the vehicle structure, 260 inches in diameter and 36 inches high. The astrionic equipment, cabling and plumbing are periferally mounted to provide clearance for the Lunar Module legs and for convenience in the structural mounting of the thermal conditioning panels.

Each IU is uniquely configured and is modified for each mission and/or vehicle. The recent history of design release activity has shown that one in four of the mandatory changes has been a mission-associated change.

Flexibility has been designed into the IU to minimize the impact of mission imposed guidance, control and sequencing variations on the vehicle stages. With the exception of the J-2S/Synchronous mission, where the 15-hour mission requires lifetime extension of the IU, the J-2S engine application has imposed minor design changes to the structure and hardware. The software impact is detailed but largely absorbed within the normal maintenance-of-design activity.

It has been assumed that the J-2S program will result in a mix of Saturn V missions and that IU's will correspondingly be of mixed configuration during the design and fabrication cycles. A specific J-2S program definition may provide for some additional efficiencies in being able to settle on a "worst case" or universal design for all missions.

The following IU modifications are in large part based on the particular requirements of each mission.

10.4.1 Stage Structural Design

The present IU structure is defined by Dwg 30Z13100-1, Structural Assembly. Principal features of this structure are depicted in Figures 10.4.1-1, 10.4.1-2 10.4.1-3. The IU structure is a cylindrical structure 260 inches in diameter and 36 inches high. The cylindrical structure consists of honeycomb sandwich construction 0.95 inch thick with upper and lower interface channel rings. It provides various pads, brackets, and insert mounting provisions, cutouts for antenna cables, a ST-124 viewport, interface bolt access, an umbilical connection and a load-carrying access door.

The honeycomb sandwich construction consists of 7075-T6 aluminum alloy face sheets bonded with METLBOND 329 adhesive to a 3.1 or 8.1-lb-per-cu-ft core. EPOCAST H-1310, Mod. 1, is used to adhesively splice the core. The brackets and pads, bonded with METLBOND 329 and EPON 934, are in most cases also bolted to the basic honeycomb structure.

Certain salient features which must be carefully considered when evaluating the IU structure for new vehicle missions and configurations are:

> The access door is load carrying and must be capable of being removed and reinstalled any time prior to flight. Also, the door frame must be capable of supporting the vehicle load when the door is removed.
>
> The 8.1-lb-per-cu-ft core is used to redistribute loads imposed by bracket, pad and mounting ring structural elements. Therefore, certain component additions or changes could require redesign of the core pattern prior to bonding.
>
> The structural buckling considerations must include not only the basic vehicle shell loads, but also the lateral loads imposed by the IU components attached to the basic honeycomb structure. The lateral loads are intensified by the dynamic environment imposed on the component and attachment dynamic response. Because of these complexities, structural capability values of the IU shell are subject to minor variations.

The present IU configuration (installation and assembly criteria) is defined by Dwg 10Z22501-1, Instrument Unit Assembly. This assembly defines each IU configuration by drawing revision level as required by the 10 IU system requirements (Navigation and Guidance, Attitude Control, Sequencing, Measurement and Telemetry, Radio and Command, Tracking, Power and Distribution, Emergency Detection, Environmental Control and Structural) combined with alignment, interface control documentation and parameters effecting component mounting surface requirements. This assembly also defines the component mounting hardware and component locations.

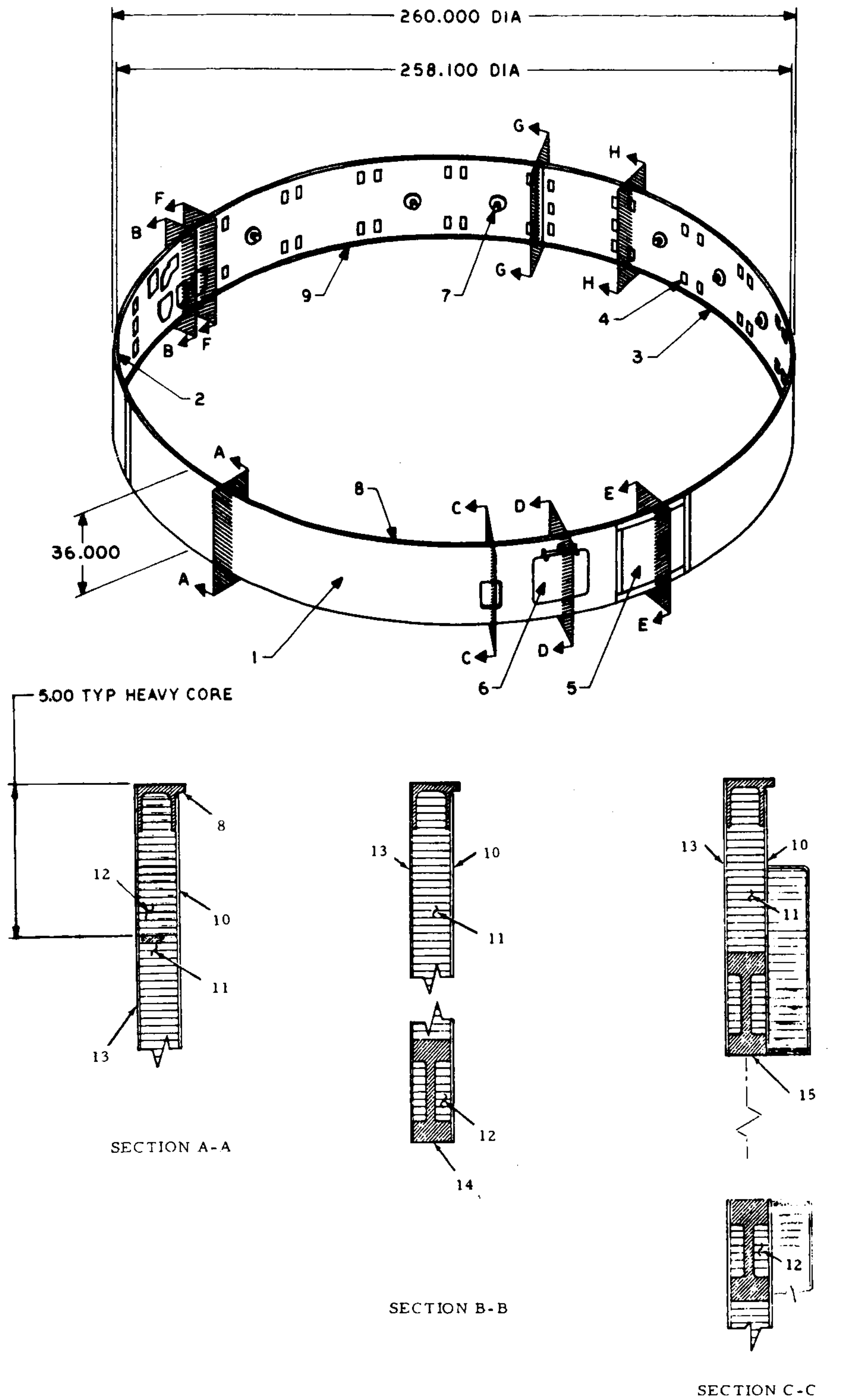
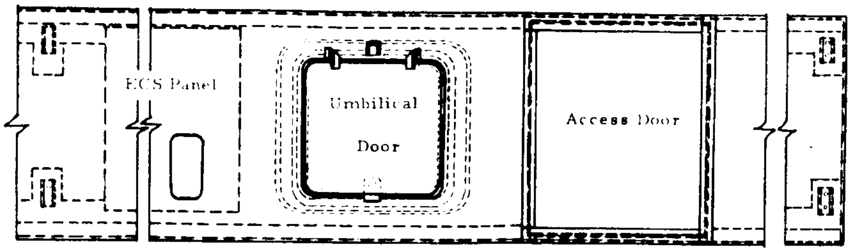
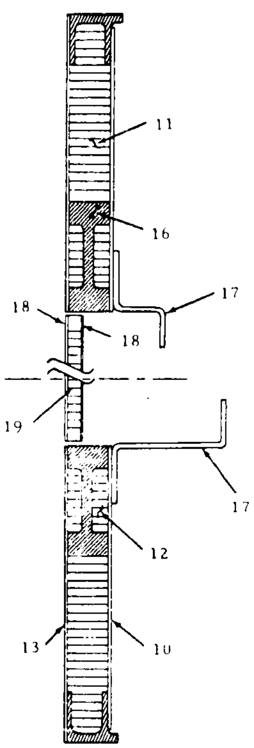


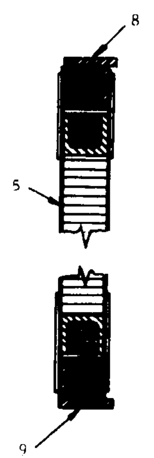
FIGURE 10.4.1-1. IU STRUCTURAL CONFIGURATION - CROSS SECTIONS



View of Umbilical Door Area



SECTION D-D



SECTION E-E

Index No.	IBM Drawing No	Description	Gage	Material	Material Specification
1	30Z13101	Segment Assembly		Honeycomb Construction	
2	30Z13103	Segment Assembly		Honeycomb Construction	
3	30Z13102	Segment Assembly		Honeycomb Construction	
4	30Z13004	Bracket (Weldment)		Honeycomb Construction	
5	30Z13109	Door Assembly		Honeycomb Construction	
6	30Z13008	Door Assembly		Honeycomb Construction	
7	30Z13050	Bracket		7075-T6	
8	30Z13110	Ring		7075-T6	QQ-A-277
9	30Z13111	Ring		7075-T6	QQ-A-277
10	30Z13105-7	Skin	0.020	7075-T6	QQ-A-283
11	30Z13031	Core	3.1 Lbs per Cubic Foot	Aluminum Honeycomb	MIL-C-7438
12	30Z13030	Core	8.1 Lbs per Cubic Foot	Aluminum Honeycomb	MIL-C-7438
13	30Z13105-5	Skin	0.030	7075-T6	QQ-A-283
14	30Z13053	Frame Assembly		7075-T6	QQ-A-283
15	30Z13037	Frame Assembly		7075-T6	QQ-A-283
16	30Z13021	Frame Assembly		7075-T6	QQ-A-283
17	30Z13005	Bracket		Reinforced Plastic	MIL-P-25421 Type II, Class I
18	30Z13105-6	Skin	0.015	7075-T651	QQ-A-283
19	30Z13035	Core	4.3 Lbs per Cubic Foot	Aluminum Honeycomb	MIL-C-7438
20	30Z13121	Filler Material		Polystyrene Beads	

NOTES:

1. ADHESIVE BETWEEN METAL-TO-METAL AND METAL-TO-CORE IS NARMCO METLBOND 329.
2. ADHESIVE BETWEEN CORES IS EPO-CAST H-1310 MOD I.
3. INSERT INSTALLATION IS MADE WITH EPON 934 (SECTION F-F).

FIGURE 10.4.1-2. IU STRUCTURAL CONFIGURATION - DOOR LOCATIONS

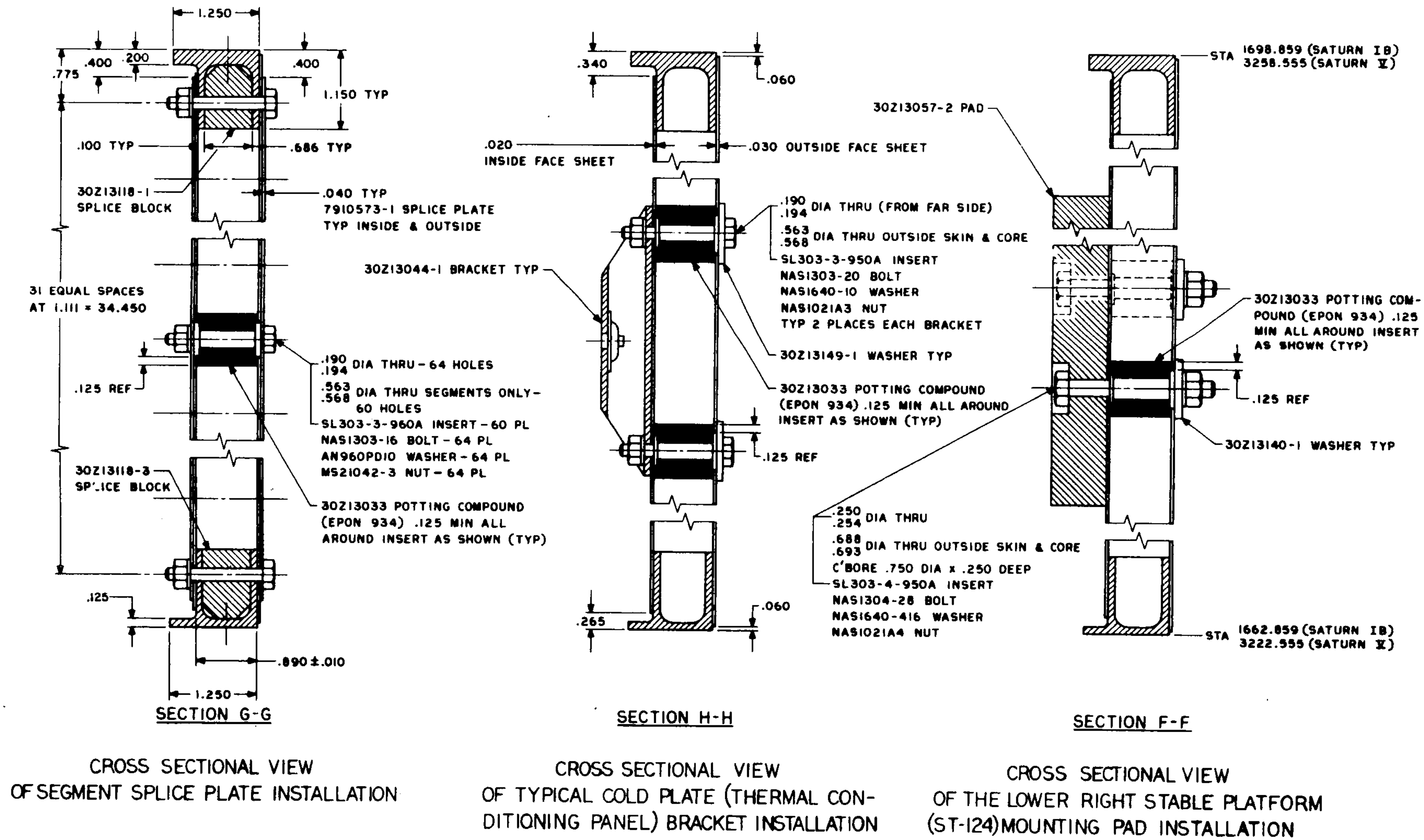


FIGURE 10.4.1-3. IU STRUCTURAL CONFIGURATION - SPLICE AND MOUNTING DETAIL

10.4.1 (Continued)

The structural design criteria is identical to that presently specified by NASA for the Saturn V IU structures. The safety factors which are applicable to the Saturn V IU structural design as minimum values are:

Yield Load	=	1.1 times limit load
Ultimate Load	=	1.25 times limit load (unmanned)
	=	1.40 times limit load (manned)

The ultimate load factor generally governs the present IU structural design, except for flange yield under tension load at the lower interface.

10.4.1.1 Structure

The major IU structural differences between the Uprated Saturn I (200 series) and the Saturn V (500 series) are that the Saturn V IU's (AS 505 and subs) have the following design modifications:

Corked insulation cold-bonded on the IU outer skin except for the umbilical door, splice plates and protrusion covers. The insulation is defined by Dwg 7916352-1, Installation of Thermal Insulation. This insulation greatly increases the load carrying capability of the IU shell structure under end boost loads with temperature.

Vibration damping compound of X-306 adhesive loaded with lead is cold-bonded under a pressure of one psi to the outer skin in the ST-124 viewpoint area (position IV) and replaces steel channels previously used for the Saturn V IU configuration. This installation is defined by Dwg 7916344-1. Vibration Damping Pad. The damping compound is more effective in reducing local vibration induced loads than the previous damping system.

Antenna insert patterns to accommodate different types and relocated antennas.

By inspection it will be noted that the 500 series will have a higher structural load carrying capability compared to the 200 series, and is used in the study.

The present IU capability to resist structural loads is shown in Figures 10.4.1.1-1, 10.4.1.1-2 and 10.4.1.1-3. Engine-out induced loads were not studied.

These curves are based on predicted structural capability, dated December 12, 1968, and are to be used as a guide for mission trends. Specific verification of any combination of loads and environments, by stress analysis, is required.

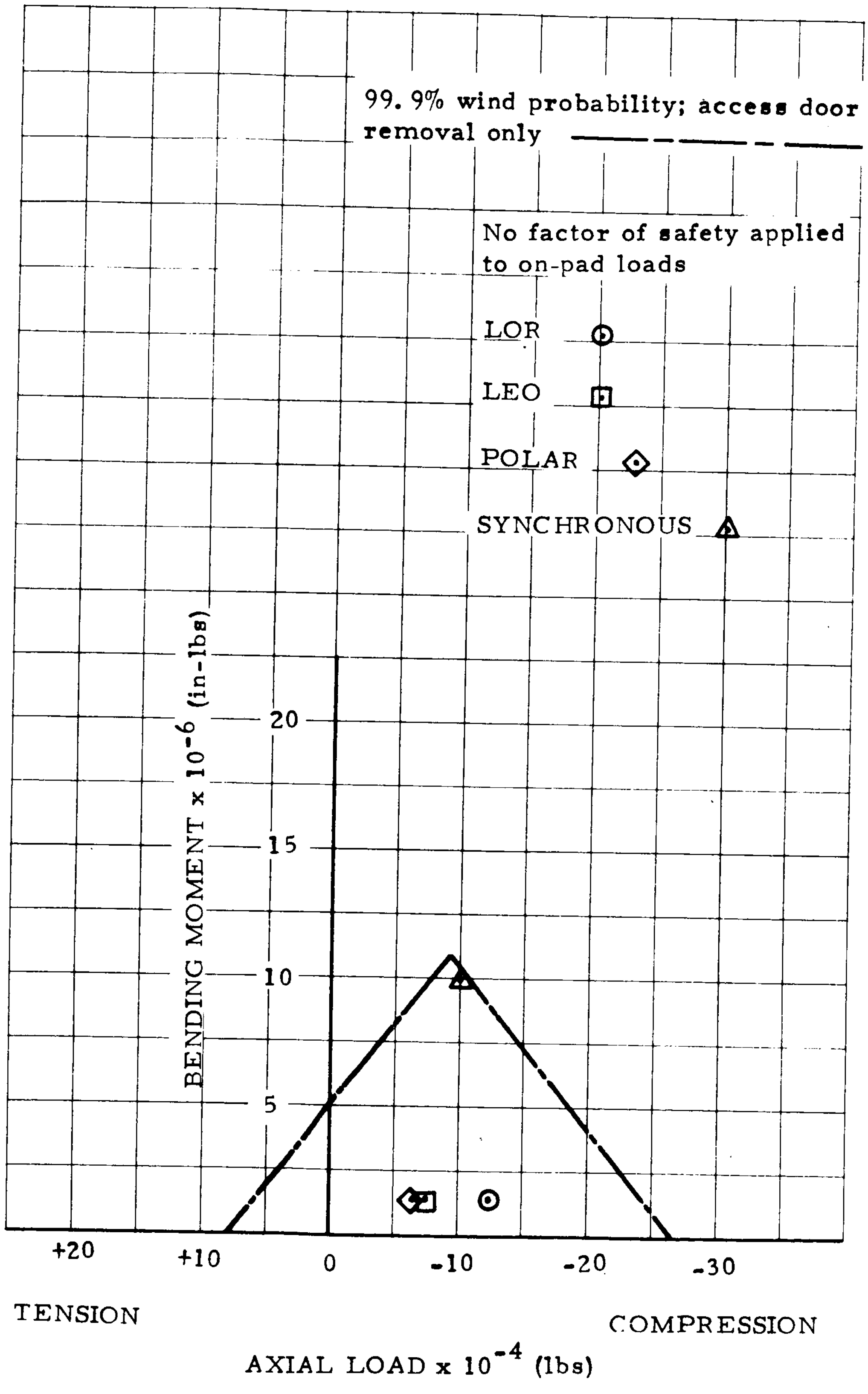


FIGURE 10.4.1.1-1. IU STRUCTURAL CAPABILITY ON PAD, LOWER INTERFACE ROOM TEMPERATURE

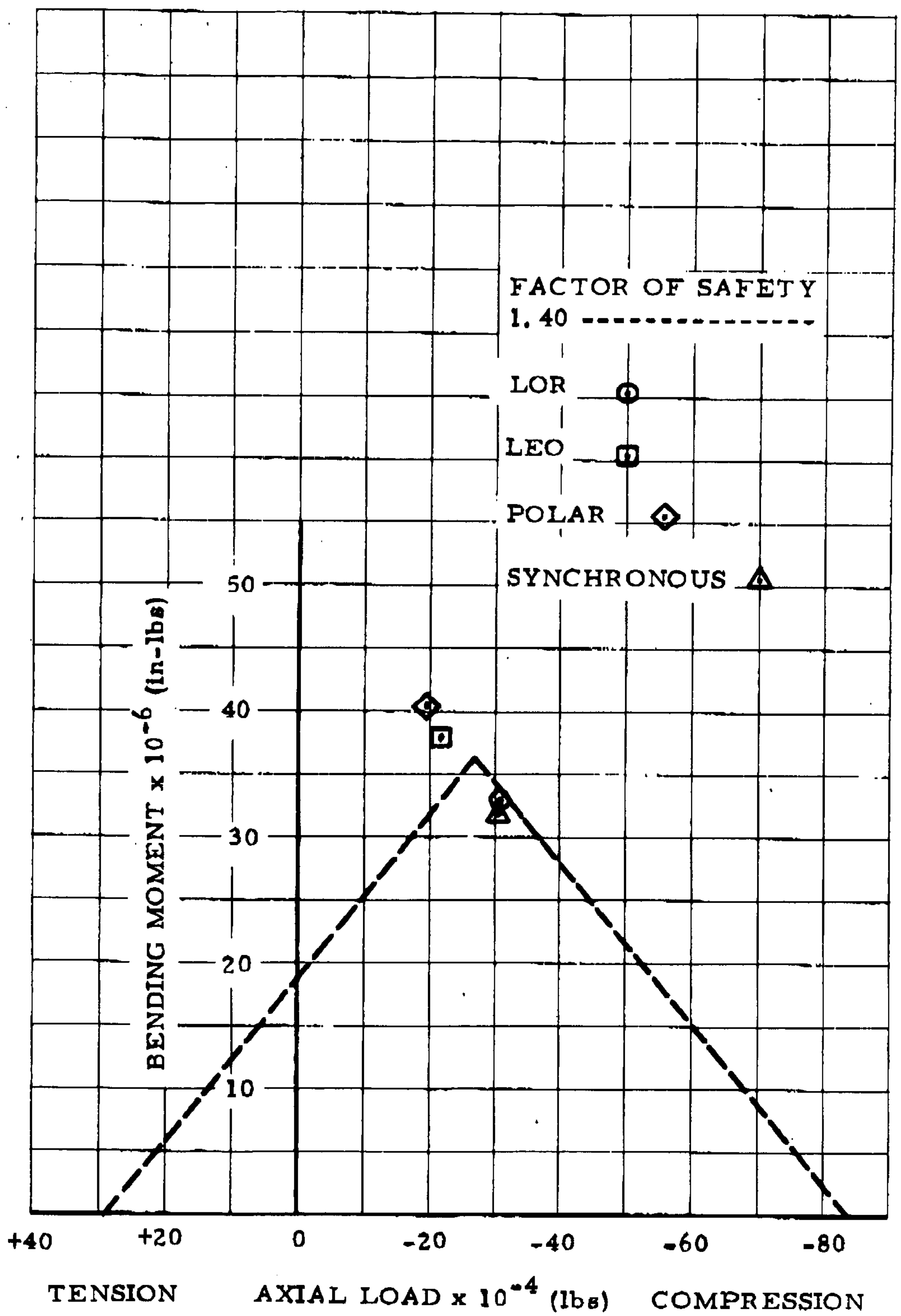


FIGURE 10.4.1.1-2. IU STRUCTURAL CAPABILITY MAX Q ALPHA, LOWER INTERFACE ROOM TEMPERATURE

15

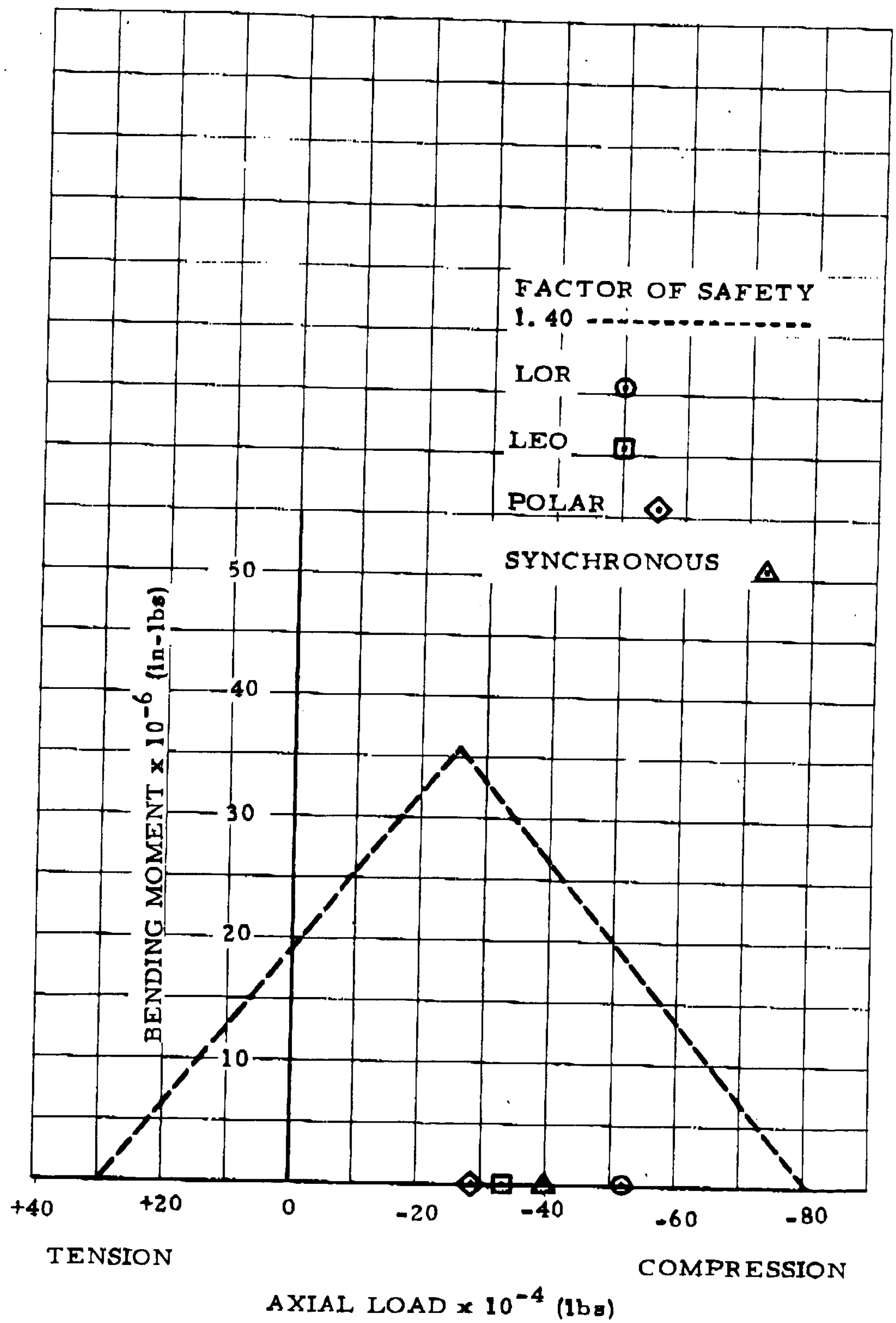


FIGURE 10.4.1.1-3. IU STRUCTURAL CAPABILITY END BOOST, LOWER INTERFACE ROOM TEMPERATURE

10.4.1.1 (Continued)

It is necessary to consider only the lower interface curves which were conservatively developed from the lower interface load allowables. The lower interface curves govern in the analysis. Values of estimated loads falling outside the required safety factor line with reference to zero axial load and bending moment indicate that structural modification is required.

The structural capabilities for Figures 10.4.1.1-1 through 10.4.1.1-3 are based upon the following longitudinal compressive and tensile values for nominal flight:

Figure 10.4.1.1-1	N_c = 324 lbs /in., N_t = 98 lbs /in.
Figure 10.4.1.1-2	N_c = 1435 lbs /in. Ult, N_t = 361 lbs /in. limit yield (flange)
Figure 10.4.1.1-3	N_c = 1400 lbs /in. Ult, N_t = 361 lbs /in. limit yield (flange)

Since the J-2S payload is not presently defined, the load distribution on the IU structure was considered uniform. This does not allow the consideration of load concentrations from payload hard points or load application points such as the LEM load concentrations presently considered in the Uprated Saturn I and Saturn V IU structural design.

The IU structure, besides providing a load transfer path between adjacent stages, also provides a component mounting area for guidance and control, telemetry, supporting ECS and other subsystems. The determination of component equipment loads to assess the capability of the IU structure requires definitions of the acoustic/vibration environment, stiffness/transmissibility characteristics and the arrangement of the component equipment. The determination of static and static equivalent dynamic equipment loads resulting from these factors to determine IU structure impact are not dealt with in this study, but are expected to be of the same order of influence as for the present Saturn V vehicles.

Since the design criteria for the IU specifies a minimum-weight structure, no handling and transportation loads were considered critical in the original design. Consequently handling and transportation fixtures were provided which introduced loads into the IU in a manner compatible with the IU flight structure. The handling and transportation design philosophy was not impacted in this application.

a. J-2S/LOR

The vehicle loads for the IU structure were obtained from Reference 10.4-13 for the lower interface (station 3222).

The combined loads, axial and bending, are shown in Figures 10.4.1.1-1, 10.4.1.1-2 and 10.4.1.1-3 for the respective on-pad, Max Q Alpha and end boost conditions of

10.4.1.1 (Continued)

the LOR. These figures depict that the LOR loads are within the IU capability. Thus, the present IU, baseline AS 511, with the addition of cork and damping compound as described in Para 10.4.1.1 is structurally adequate to withstand the running loads imposed by the LOR mission.

The access door is load carrying and must be capable of being removed and reinstalled any time prior to flight. Also, the door frame must be capable of supporting the vehicle load when the door is removed. Since the LOR on-pad loading is less than the IU capability, the access door and door frame are qualified for and capable of carrying the imposed mission loads.

The LOR IU is not subjected to acoustic, acceleration or vibration environments which are of greater severity than those imposed on the present Saturn V IU vehicles (refer to Para 10.4.4.1). Therefore, the determination of static and static equivalent of acceleration and equipment dynamic loading effects is not dealt with in this study.

The thermal environment for the LOR IU is within the adequacy of the presently flown Saturn V vehicles (AS 501, 502 and 503). Therefore, no thermal analysis is required for this IU (refer to Para 10.4.4.2).

b. J-2S/LEO

The vehicle loads for the IU structure were obtained from References 10.4-36 and 10.4-37 for the lower interface (station 3222).

The combined loads, axial and bending are shown in Figures 10.4.1.1-1, 10.4.1.1-2 and 10.4.1.1-3. The respective on-pad, Max Q and end boost loads are within the IU capability with the exception of the Max Q Alpha condition.

The Max Q Alpha load is a condition in which the bending moment is contributing to a tension running load on the interface rails which exceeds the IU structural tension capability.

Table 10.4.1.1-I gives the margins of safety for the upper and lower interface tension loads and for two Max Q Alpha flight conditions, nominal and one engine-out. This table depicts the lower interface rail during Max Q Alpha period of flight (nominal) to be the design case.

This tension condition also exists in Para 10.4.1.1.c (Polar). The design changes necessary to overcome this condition will be discussed in Para 10.4.1.1.c since the Polar tension load is a more severe case. Thus, the design change can be incorporated to accommodate either the LEO or Polar IU's and give these vehicles greater load carrying capability.

TABLE 10.4.1.1-I. J-2S/LEO MARGINS OF SAFETY

MAX Q ALPHA CONDITION	MARGINS OF SAFETY UPPER INTERFACE	LOWER INTERFACE
Nominal	+0.28	-0.25 (Tension)
Engine-Out	+0.46	+0.43

ASSUMPTION: The maximum limit tension load due to delta pressure across the IU structure is 26 lbs /in.

NOTES:

1. The previous assumption is included in the MS.
2. The negative MS indicates that the interface rail has a yield factor of safety less than 1.1 and indicates that flange yielding occurs.

$$MS = \frac{361}{481} - 1 = -0.25$$

10.4.1.1 (Continued)

The on-pad and end boost loads present no problem to the present baseline IU; AS 511. However, these IU's must be equipped with insulation (cork) and damping compound as described in Para 10.4.1.1.

The access door is load carrying and must be capable of being removed and reinstalled any time prior to flight. Also, the door frame must be capable of supporting the vehicle load when the door is removed. Since the LEO on-pad loading is less than the IU capability, the access foor and door frame are qualified for and capable of carrying the imposed mission loads.

The LEO IU is not subjected to acoustic, acceleration or vibration environments which are of greater severity than those imposed on the present Saturn V IU vehicles (refer to Para 10.4.4.1). Therefore, the determination of static acceleration and static equivalent of dynamic component loading effects are not dealt with in this study.

The thermal environment for the LEO IU is within the adequacy of the presently flown Saturn V vehicles (AS 501, 502 and 503). Therefore, no thermal analysis is required for this IU (refer to Para 10.4.4.2).

c. Polar

The vehicle loads for the IU structure were obtained from References 10.4-36 and 10.4-37 for the lower interface (station 3222).

The combined loads, axial and bending, are shown in Figures 10.4.1.1-1, 10.4.1.1-2 and 10.4.1.1-3 for the respective on-pad, Max Q and end boost conditions of the Polar IU. These figures depict that the Polar loads are within the IU capability with the exception of the Max Q Alpha condition.

The Max Q Alpha load is a condition in which the bending moment is contributing to a tension running load on the interface rails which exceeds the IU structural tension capability.

Table 10.4.1.1-II gives the margins of safety for the upper and lower interface tension loads and for two Max Q Alpha flight conditions, nominal and one engine-out. This table depicts the lower interface rail during Max Q Alpha (nominal) to be the design case.

This type of localized failure is attributed to flange yielding at the lower interface rail. There are 108 NAS 625 high-strength-steel bolts (.3125 diameter) with washers mounted through access bolt holes on the SLA through the SLA interface ring and the IU upper interface rail. Washers and MC 623-5 nuts complete the attachment through the IU upper access bolt holes.

TABLE 10.4.1.1-II. J-2S/POLAR MARGINS OF SAFETY

MAX Q ALPHA CONDITION	MARGINS OF SAFETY UPPER INTERFACE	LOWER INTERFACE
Nominal	+0.10	-0.35 (Tension)
Engine-Out	+0.36	+0.34

ASSUMPTION: The maximum limit tension load due to delta pressure across the IU structure is 26 lbs /in.

NOTES:

1. The previous assumption is included in the MS.
2. The negative MS indicates that the interface rail has a factor of safety less than 1.1 and indicates that flange yielding occurs.

$$MS = \frac{361}{557} - 1 = -0.35$$

10.4.1.1 (Continued)

The IU lower interface, on the other hand, has 216 NAS 625 high-strength-steel bolts (.3125 diameter) with washers mounted through the IU lower access bolt holes, IU lower interface rail, and S-IVB stage forward interface ring to NAS 687-A5 floating nut-plates. The nut-plates are mounted on the S-IVB stage, and therefore are not addressed in this study by IBM, although they may be impacted. The number of bolt holes is a multiple function of the 108 S-IVB forward skirt stringers; two interface bolts are on each side of a stringer.

Because fewer bolts were required at the upper interface, the IU upper interface rail is thicker than the lower (.200 vs .125 inches thick). The effect of prying forces between stage interfaces makes the IU lower interface rail marginally adequate for flange yield under current Saturn V contract loads.

There are several possible fixes for this structural design condition. The break points between increasingly more costly fixes are as follows:

> A minimum modification fix may be possible, dependent upon small interface specimen (approximately 8 x 15 inches) testing for flange yield. A special manufactured washer to reinforce the flange locally would be used to increase the flange yield capability, but it is not known how much because of the complex prying of the IU interface. (See Figure 10.4.1.1-4.) The fix need not be for the entire interface, but the locations can only be decided upon after such testing takes place.
>
> For load levels beyond which the minimum modification fix would be effective, a modification would include increasing the length of the vertical legs to provide a larger bond (adhesive) area and increasing the flange height of lower interface rails. Along with this modification, the shape of bolted access hole cutouts at the IU interface would be altered to help prevent stress concentration factors from becoming a governing factor. From Table 10.4.1.1-II the MS is -0.35 and this corresponds to a N_t of 888 lbs /in. (ultimate). A preliminary analysis indicates that the vertical leg height would be increased from the present 1.0 inch to 1.39 inch (Figure 10.4.1.1-5) and the flange height would be increased from the present 0.125 inch to approximately 0.166 inch. These values would be refined during the design phase for the Polar IU and are only presented here to indicate a representative change.
>
> For very high tension load levels, the inner skin thicknesses would also have to be increased and the number and/or strength of the interface bolts would have to be addressed. This impact is not described in this study and will be analyzed subsequent to redesign.

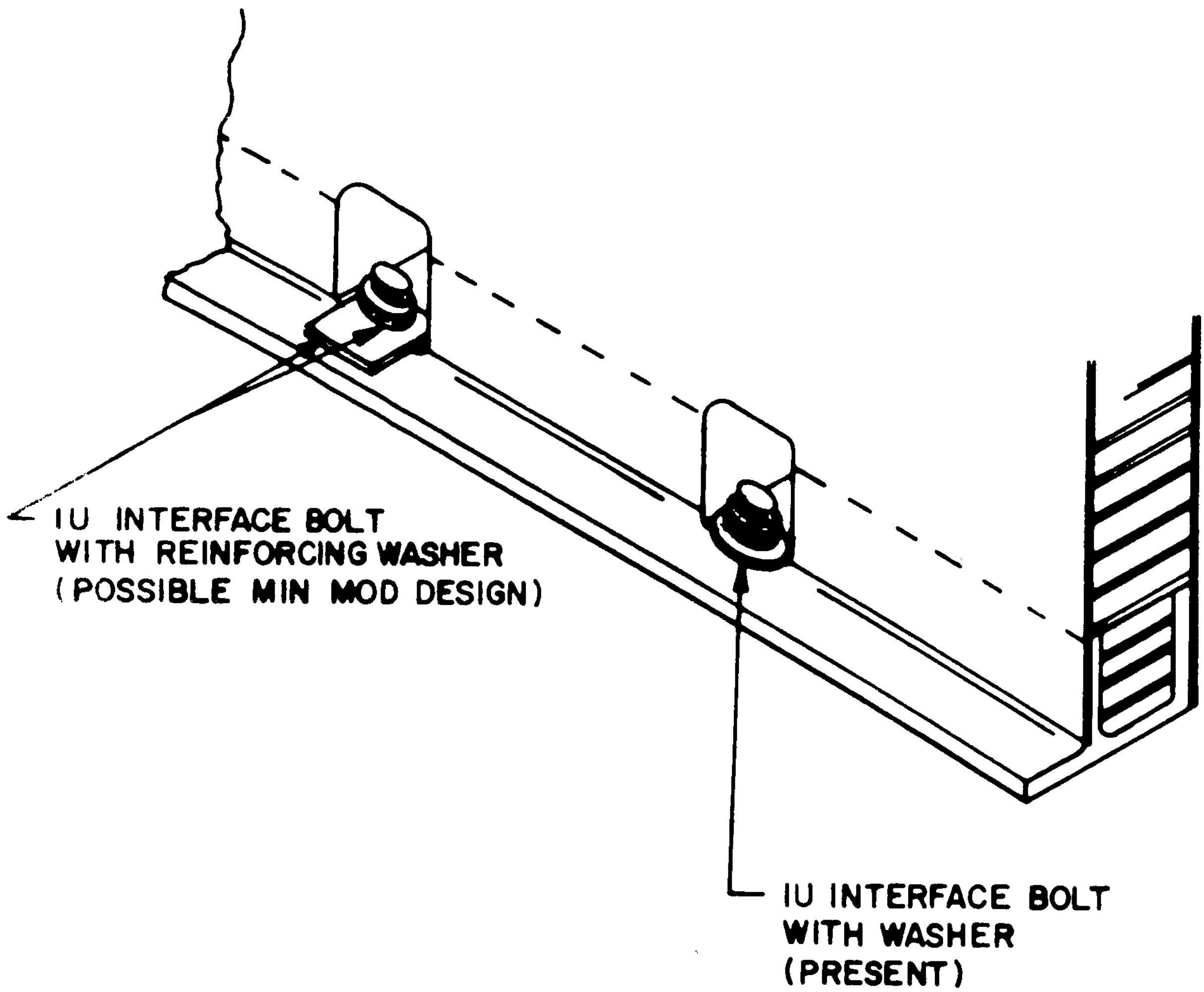


FIGURE 10.4.1.1-4. IU LOWER INTERFACE CONSTRUCTION

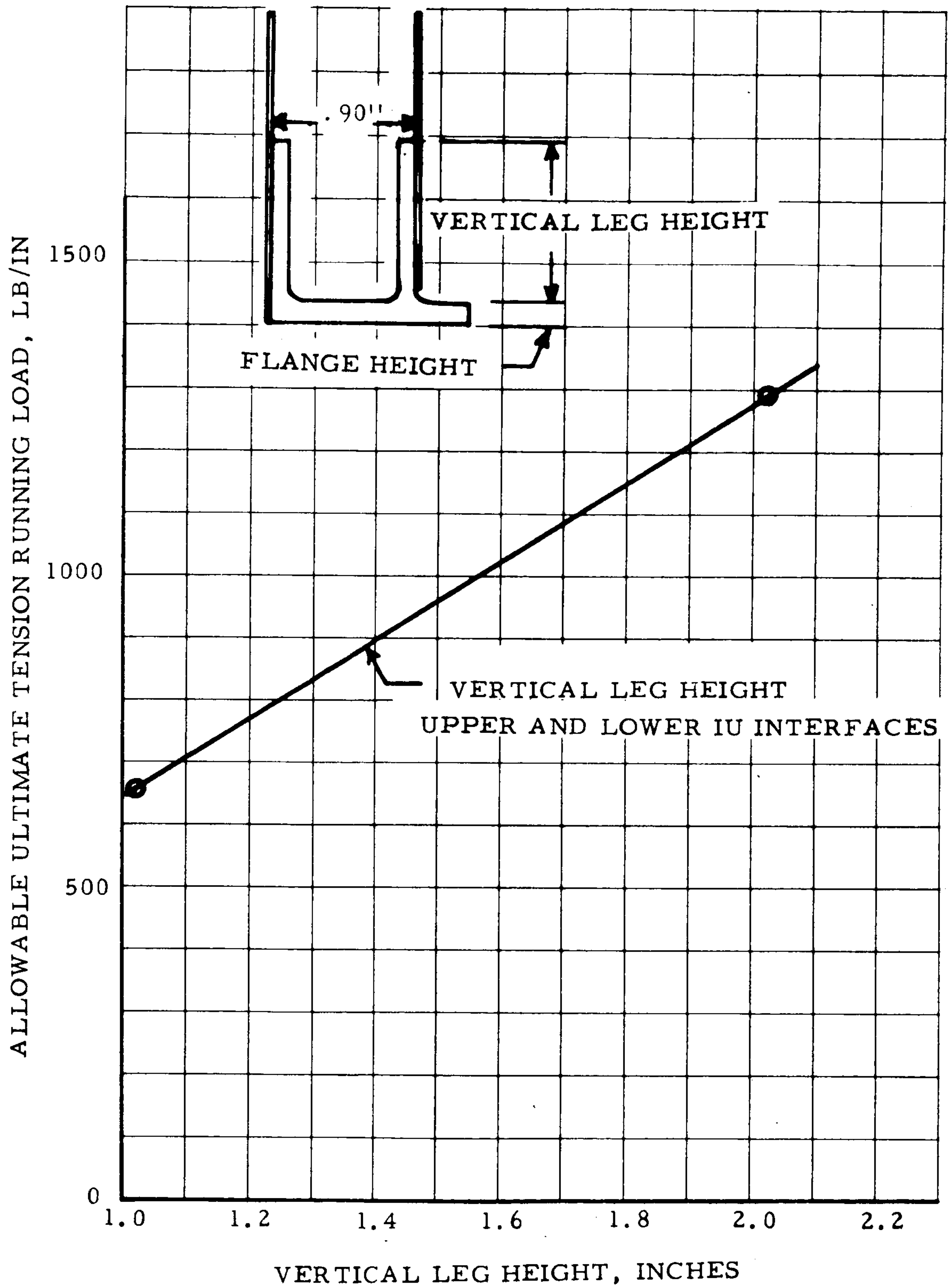


FIGURE 10.4.1.1-5. INTERFACE TENSION CAPABILITY

10.4.1.1 (Continued)

The required change would make the MS positive for a yield condition.

The on-pad and end boost loads present no problem to the present baseline IU; AS 511. However, these IU's must be equipped with insulation (cork) and damping compound as described in Para 10.4.1.1.

The access door is load carrying and must be capable of being removed and reinstalled any time prior to flight. Also, the door frame must be capable of supporting the vehicle when the door is removed. Since the Polar on-pad loading is less than the IU capability, the access door and door frame are qualified for and capable of carrying the imposed Polar mission loads.

The Polar IU is not subjected to acoustic, acceleration or vibration environments which are of greater severity than those imposed on the present Saturn V IU vehicles (refer to Para 10.4.4.1). Therefore, the determination of static acceleration and static acceleration and static equivalent of acceleration and dynamic equipment is not dealt with in this study.

The end of boost and orbital thermal environment for the Polar IU are within the adequacy of the presently flown Saturn V vehicles (AS 501, 502 and 503). Therefore, no thermal analysis is required for this IU (refer to Para 10.4.4.2).

d. Synchronous

The vehicle loads for the IU structure are obtained from Reference 10.4-39 for the lower interface (station 3222).

The combined loads, axial and bending, are shown in Figures 10.4.1.1-1, 10.4.1.1-2 and 10.4.1.1-3 for the respective on-pad, Max Q Alpha and end boost conditions of the Synchronous mission IU. These figures depict that the Synchronous loads are within the IU capability. However, structural buckling considerations must include lateral loads imposed by the IU components attached to the honeycomb structure. These lateral loads are intensified by the dynamic response of components and attachment hardware. This consideration has not been fully analyzed in the area of the side-by-side GBS panels. The predicted dynamic response of the basic structure in this area cannot be accurately analyzed by existing techniques. Therefore, a comprehensive structural analysis cannot be accomplished without the static equivalent of dynamic loads for this area being derived by structural testing.

The access door is load carrying and must be capable of being removed and reinstalled any time prior to flight. Also, the door frame must be capable of supporting the vehicle load when the door is removed. Since the Synchronous on-pad

10.4.1.1 (Continued)

loading is less than the IU capability, the access door and door frame are qualified for and capable of carrying the imposed mission loads.

The Synchronous IU is not subjected to acoustic, acceleration or vibration environments of greater severity than those imposed on the present Saturn V IU vehicles. However, the complexity of determining the static equivalent of dynamic loads and component vibration responses in the area of the side-by-side GBS panels is highly questionable as to the degree of accuracy. Thus, dynamic testing is required in order to verify analysis methods and certify the structure and components for flight worthiness. This is addressed in Para 10.4.7.3.

The structural in-line modifications for the Synchronous IU are defined in Paragraph 3.4.1.1.b (Antenna Mounting Provisions) and Para 3.4.1.2.b (GBS Panel Mounting Provisions) of Reference 10.4-39. A summary of these paragraphs follows. (See Figure 10.5.1.4-15 for IU location numbers.)

A modification will be required to add six 30Z13058-1 brackets at location 23, in a pattern identical with the existing 30Z13058-1 bracket pattern in location 22, to accommodate an additional GBS panel in location 23. The other changes required for this area are as follows:

Add 8.1 honeycomb core in location 23.

Delete four brackets, 30Z13044-1, and one bracket assembly, 30Z13050-1.

Delete 16 inserts, NAS 1398134-3.

The antenna modifications required in the basic structure are as follows:

VHF antenna cutouts and insert patterns which are presently located in location 9-10 and 21-22 would be relocated to locations 10-11 and 22-23 below the existing UHF TM antenna cutouts and insert patterns.

CCS directional antenna cutouts and insert patterns must be made at locations 5-6, 9-10, 13-14, 17-18 and 21-22. The antenna mounting provision in location 3-4 must be deleted and relocated to provide similar antenna mounting provisions in location 1-2.

The referenced ECP did cite certain modifications which are now included in the AS 511 baseline vehicle. These additional changes include the vibration damping channels in the ST-124 area, location 21, which are deleted and replaced by a vibration damping pad. This installation is defined in Reference 10.4-41. Cork thermal insulation is added to the IU outer skin except for the umbilical door, splice plates and protrusion covers. This insulation is defined in Reference

10.4.1.1 (Continued)

10.4-42, however, these are not changes to the present AS 511 baseline IU, since they have previously been incorporated.

The thermal environment for the Synchronous IU is within the capability of the presently flown Saturn V vehicles (AS 501, 502 and 503). Therefore, no thermal stress analysis is required for the Synchronous IU (refer to Paragraph 10.4.4.2.1).

A thermal cycling environment is encompassed by the Synchronous IU. This has been addressed in Paragraph 10.4.4.2.3 and presents no problems to the IU structure.

10.4.1.2 Stage Interfaces

The upper (station 3258) and lower (station 3222) interfaces and adjacent structure from the J-2S IU's are assumed to be similar to the present Saturn V with the exception of the LEO and Polar IU interfaces. These exceptions are explained in Paragraphs 10.4.1.1.b and 10.4.1.1.c for the respective LEO and Polar IU's. A basic assumption is made that the undefined payload design will satisfy the baseline vehicle interface criteria specification for the IU/S-IVB.

10.4.2 IU Systems

This section defines the impact on the IU systems resulting from the J-2S engine and mission changes. The impacts are confined to resulting hardware alterations only. Software is treated in paragraph 10.4.3. To clearly show mission related changes, the systems are treated by mission. The J-2S engine impact is entirely contained in the J-2S/LOR mission sections. Changes for the other missions then add to those but must be considered mission deltas only.

10.4.2.1 Guidance Subsystem

The Saturn V IU Guidance Subsystem consists of the ST-124M three-gimbal, stabilized inertial platform assembly providing a space-fixed coordinate reference system for navigation measurements (vehicle acceleration and velocity). The measurements are sent through the LVDA to the LVDC where they are used in calculating the vehicle's velocity and position in space relative to the earth.

10.4.2.1.1 J-2S/LOR Mission

a. Requirements

There are no requirements resulting from the vehicle modification to J-2S engines or the LOR mission that imposes new requirements on the guidance subsystem hardware. Therefore, no hardware modification is required.

10.4.2.1.2 J-2S/Synchronous Mission

a. Requirements

The present three-gimbal ST-124M platform has a $\pm$ 45° operational yaw limitation, although the gimbal stops are at approximately 60°. A large variety of yaw requirements is encountered when considering the spectrum of missions addressed for Synchronous Orbit.

The worst-case yaw of approximately 72° is encountered for an equatorial Synchronous orbit which is entered from a 44° inclined waiting orbit. Since this exceeds the present capability, a hardware or software modification is required.

b. Modification

There will be no hardware modifications to meet the large yaw requirement of certain Synchronous Orbit missions. This requirement will be met by using a yaw bias technique. This technique involves intentional offset of the ST-124M platform to take advantage of the available 90° range of yaw.

10.4.2.1.3 J-2S/LEO Mission

a. Requirements

There are no requirements resulting from the vehicle modification to J-2S engines on the LEO mission that imposes new requirements on the guidance subsystem hardware; therefore, no hardware modifications is required.

10.4.2.1.4 J-2S/Polar Mission

a. Requirements

As stated under the requirements for Synchronous Orbit, the present three-gimbal ST-124 platform has a $\pm 45^{\circ}$ operational yaw limitation. The Polar Orbit mission requires a yaw guidance command of approximately 50°. Since this exceeds present capability, some modification is required.

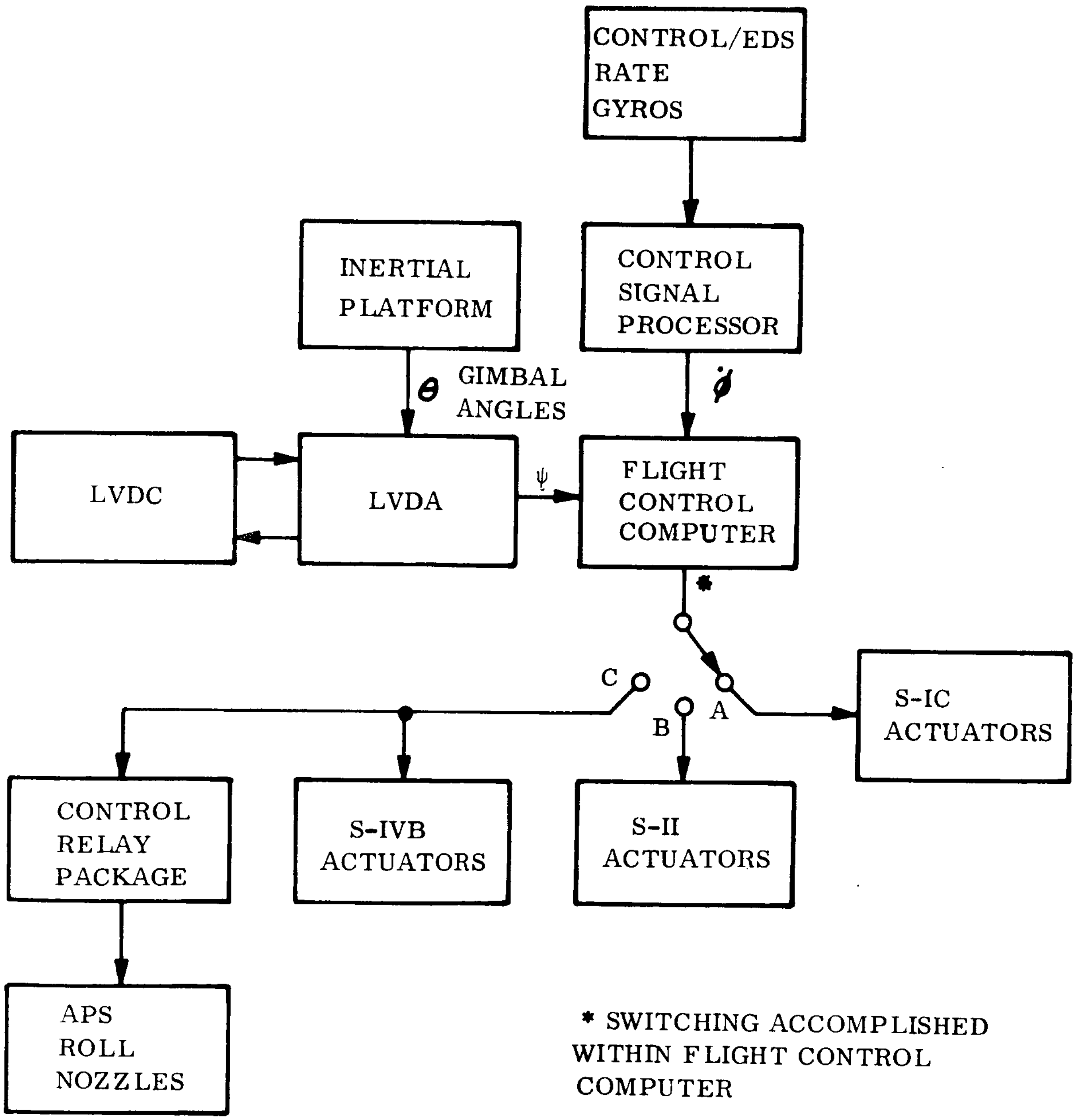
b. Modification

There will be no hardware modification to meet the large yaw requirement of the Polar mission. This requirement will be met by using a yaw bias technique. This technique involves intentional offset of the ST-124M platform to take advantage of the available 90° range of yaw.

10.4.2.2 Control Subsystem

The attitude control requirement of the Saturn Launch Vehicle can be divided into control during powered flight and control during coast flight. Attitude control during powered flight is accomplished by swivelling the propulsion engines to control thrust vector direction. During coast flight, attitude control is provided by the auxiliary propulsion system of the S-IVB stage. This auxiliary propulsion system is used also during powered flight of the S-IVB stage for roll control, which cannot be achieved with the single engine of the S-IVB stage.

The main components of the Saturn attitude control system are shown in Figure 10.4.2.2-1. The platform gimbal-angle readings ($\theta_X \theta_Y \theta_Z$) indicate the orientation of the vehicle in the navigation coordinate system (X_S Y_S Z_S). The LVDC computes the required thrust vector orientation according to the guidance scheme and generates attitude error signals ($\psi_R \psi_P \psi_Y$)--with respect to the body-fixed roll, pitch, and yaw axes--which are sent to the Flight Control Computer (FCC).



FLIGHT PHASE	SWITCH POS
S-IC	A
S-II	B
S-IVB (POWERED)	C

FIGURE 10.4.2.2-1. SATURN CONTROL SYSTEM

10.4.2.2 (Continued)

In addition to the attitude error signals, the FCC receives attitude rate signals ($\dot{\phi}_P$ $\dot{\phi}_Y$ $\dot{\phi}_R$ -- with respect to the vehicle pitch, yaw and roll axes) from the Control Signal Processor. All control signals fed into the FCC are analog signals. The FCC processes and combines these signals according to the control law to generate the control signals for the engine actuators and attitude control nozzles.

The Saturn V control law for the thrust vector deflection angle (β) is:

$$\beta = a_o \psi + a_1 \dot{\phi}$$

where a_o and a_1 are gain factors, and ψ and $\dot{\phi}$ are att.tude error angle and attitude angular rate, respectively.

Attitude control of the launch vehicle during powered flight is accomplished by swivelling of the main propulsion engines and thereby changing the orientation of the thrust vector. The S-IC and S-II stages have five engines each. The four outer engines can be swivelled in pitch and yaw by two hydraulic actuators at each engine. The S-IVB stage is propelled by a single engine which can be swivelled in pitch and yaw. In addition, the S-IVB stage is equipped with an auxiliary propulsion system consisting of six nozzles (two sets of three nozzles) mounted on the outside of the aft end of the S-IVB stage. Four of the six nozzles are required for roll control of the S-IVB stage, since roll control cannot be achieved with a single main propulsion engine.

The output to each actuator is derived from a 50 MA Servo Amplifier. The 50 MA Servo Amplifier is a current amplifier capable of supplying 50 milliamps to the actuator load.

During coast flight, attitude control of the vehicle is accomplished by means of the S-IVB stage auxiliary propulsion system which contains six attitude control engines (thrust nozzles). When fired, the nozzles produce torques about the center of gravity of the vehicle. They are controlled in a pulse-type manner (full thrust or OFF) by the spatial amplifiers in the Flight Control Computer. The attitude control engines are turned ON when the input signal to the spatial amplifier exceeds certain limits and turned OFF when the input signal falls below a threshold. The control scheme generates thrust pulses of variable duration at changing intervals which are controlled by the pseudo rate modulators in the spatial amplifier. Over a certain range of the input signal, the output of the modulator is pulse-width and frequency modulated and is a function of the input signal magnitude. Above this range, the output (thrust) is continuously ON; below this range, it is OFF.

10.4.2.2 (Continued)

A typical attitude error channel in the FCC is shown in Figure 10.4.2.2-2. The filter configuration determines a_0 of the control equation. Figure 10.4.2.2-3 depicts a typical attitude rate channel within the FCC. The filter again determines a_1 of the control equation.

Both a_0 and a_1 are capable of being changed in flight. This is accomplished by energizing relays (called switch points) within the filter with Switch Selector discretes.

10.4.2.2.1 J-2S/LOR Mission

a. Requirements

The only additional requirement to change a J-2/LOR control subsystem to a J-2S/LOR control subsystem is to have the capability of resetting switch point 5. The purpose of the switch point is to change the gain through the filter network in the FCC. The switch point will be set while in the S-IVB idle mode. This is not necessary when idle mode is merely a step to mainstage as is the case for S-IVB first burn. However, under restart conditions, the S-IVB is in idle mode for 100 seconds. During these 100 seconds idle mode the switch point will be set. For S-IVB mainstage it will be reset.

b. Hardware Modification Summary

Figure 10.4.2.2-4 shows the schematic of a typical switch point in the control subsystem. The Switch Selector issues a discrete that energizes a latching relay in the control distributor. The latching relay then sends 28 volts to the FCC, energizing a non-latching relay in the proper filter. The only existing reset capability on the switch points is a GSE function.

To provide the required in-flight reset capability, a Switch Selector discrete will be used to reset switch point 5. The hardware impact will consist of routing one wire from the Switch Selector to the Control Distributor to diode - or the GSE reset signal and the in-flight Switch Selector reset signal. The new circuitry is shown in Figure 10.4.2.2-5.

10.4.2.2.2 J-2S/Synchronous Orbit Mission

Requirements and hardware modification is identical to that discussed under the LOR mission paragraph 10.4.2.2.1.

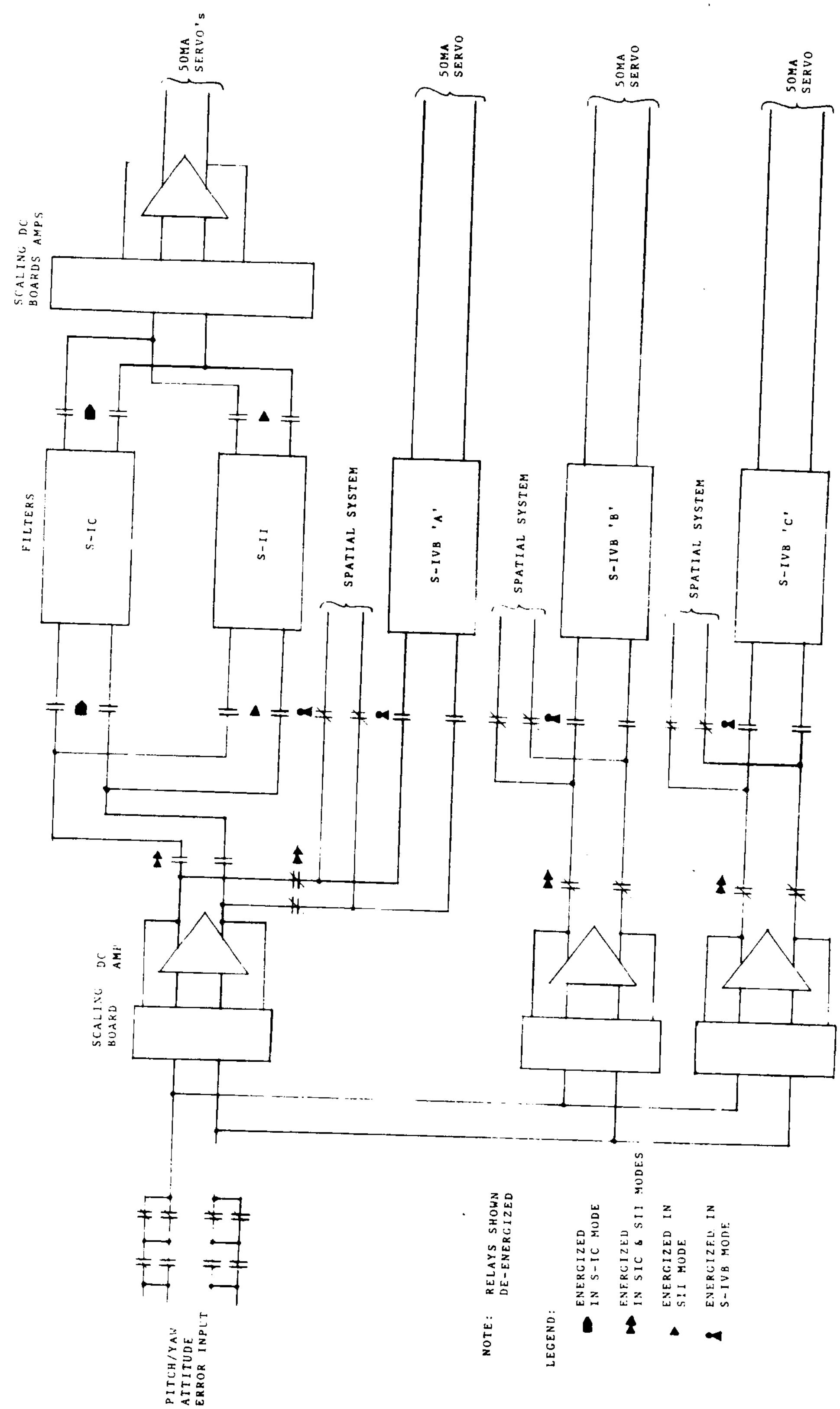


FIGURE 10.4.2.2 - 2. TYPICAL ATTITUDE ERROR CHANNEL

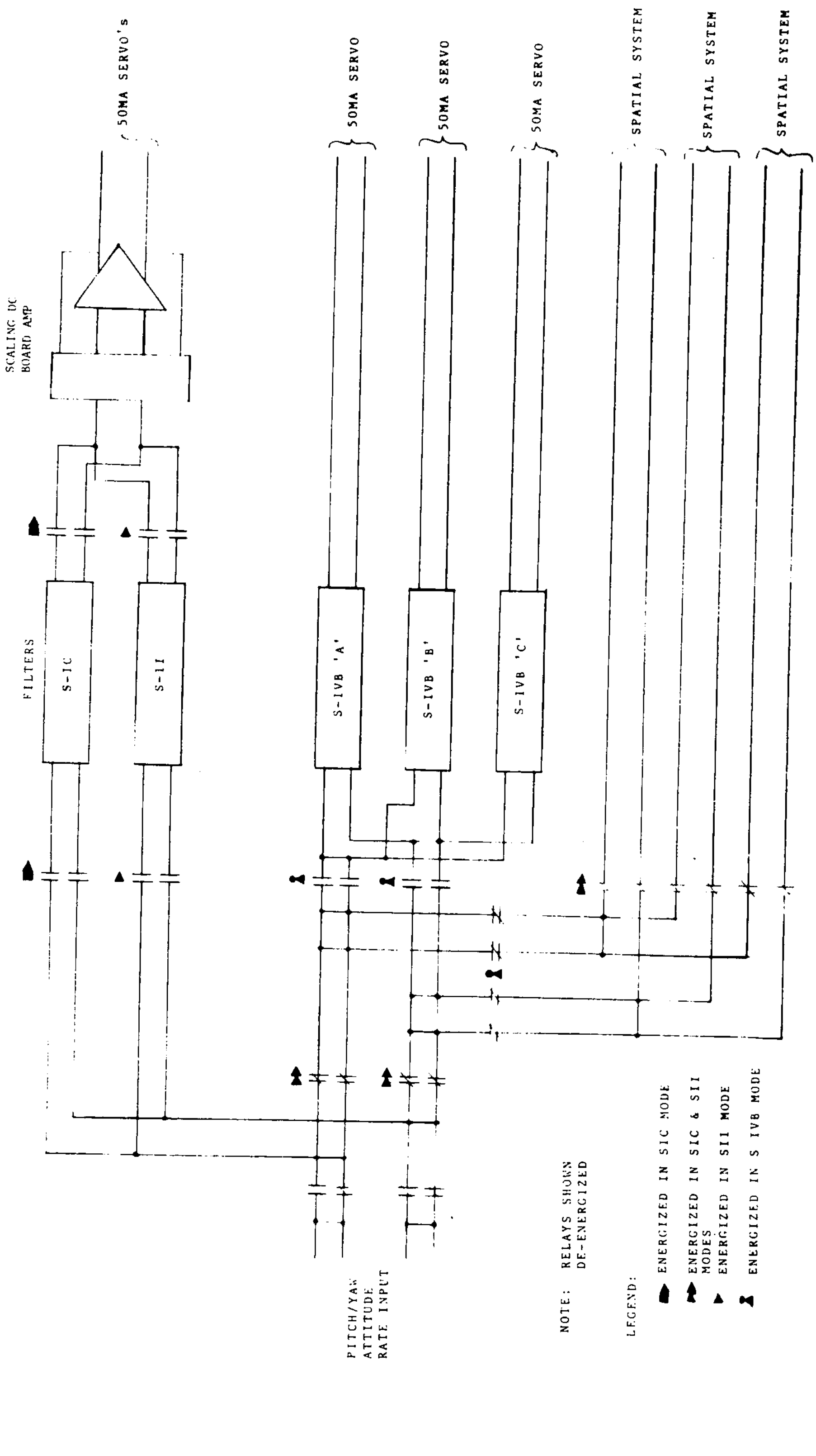


FIGURE 10.4.2.2-3. TYPICAL ATTITUDE RATE CHANNEL

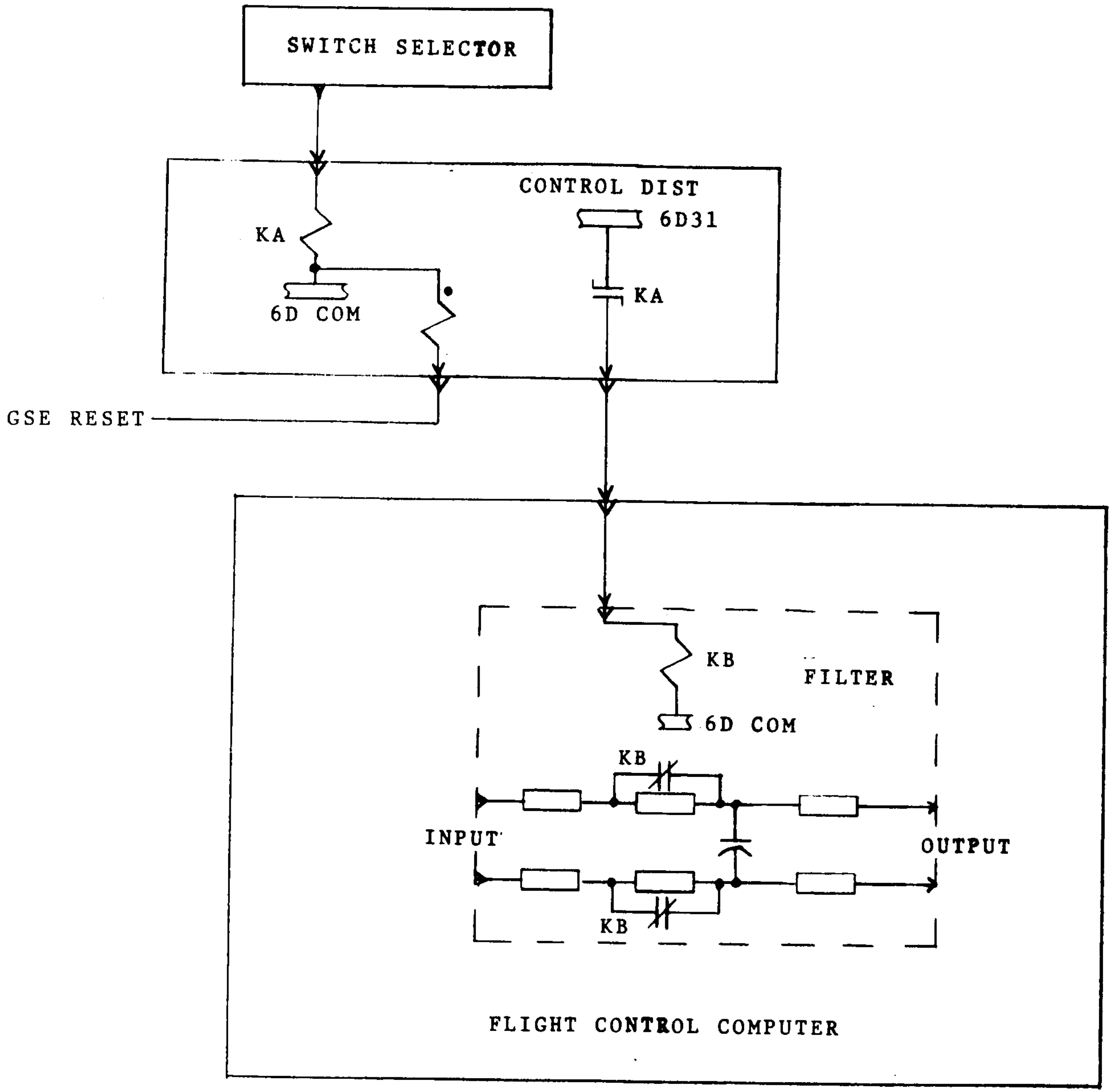


FIGURE 10.4.2.2 - 4. TYPICAL SWITCH POINT SCHEMATIC

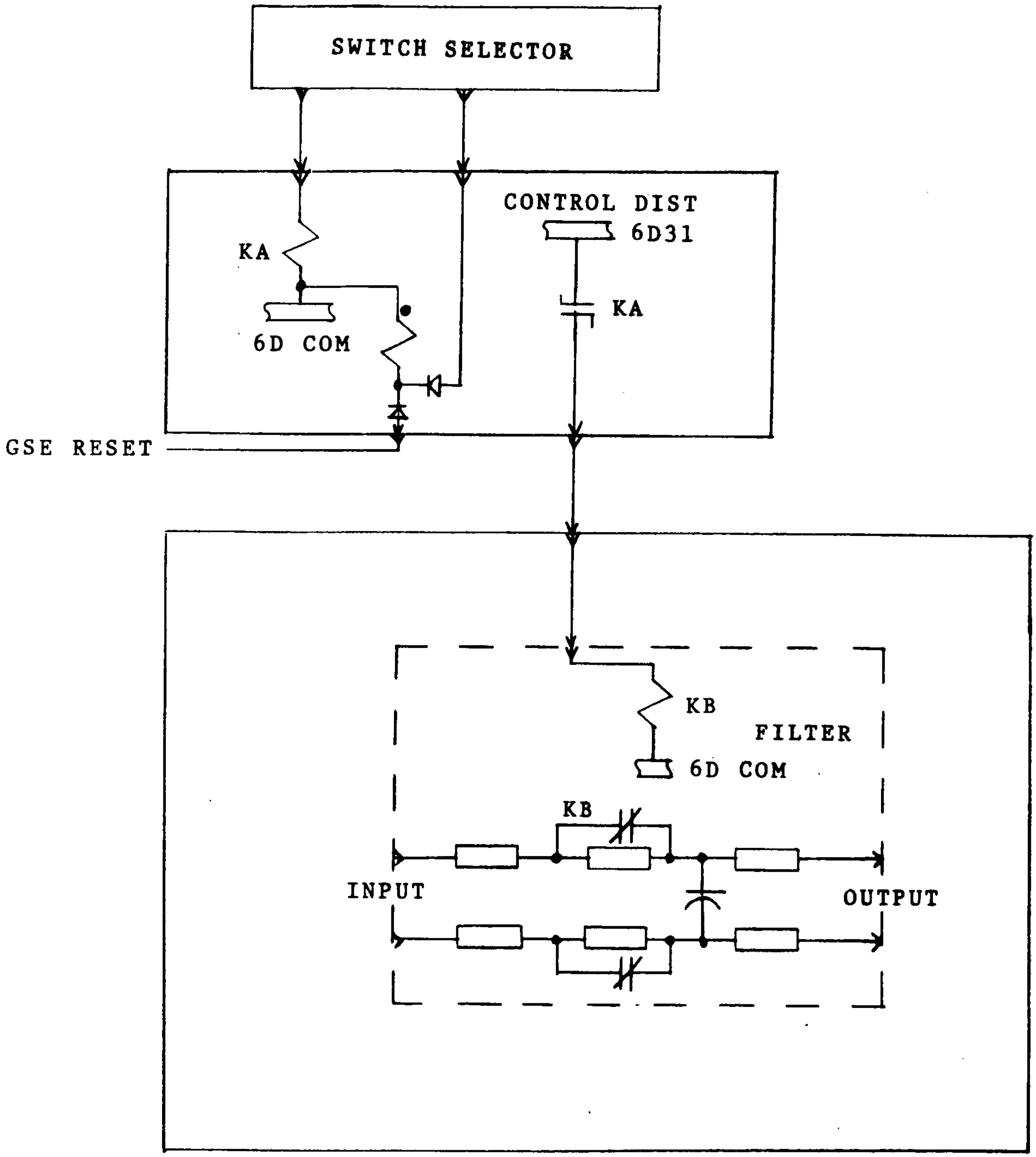


FIGURE 10.4.2.2 - 5. RESETTABLE SWITCH POINT SCHEMATIC

10.4.2.2.3 J-2S/LEO Mission

a. Requirements

The additional requirements to be met are:

1. No S-IVB stage (Dummy printed circuit cards).

2. Two states of burn for the S-II stage (mainstage and idle).

3. S-II Pitch and Yaw attitude rate channels require different filters in mainstage and idle modes.

4. S-II attitude error filters and Roll attitude rate filter require an additional switch point for idle mode.

One additional constraint for this study was no major pin function changes would be allowed.

b. Hardware Modification Summary

Since no S-IVB stage is used on the J-2S/LEO vehicle, seventeen modules used only for S-IVB operation will be removed from the FCC. Of these seventeen modules, fifteen will be replaced by blank boards; two will be replaced by the additional S-II mode Pitch and Yaw attitude rate filters. Replacing the fifteen modules with blank boards minimizes the impact to the existing FCC design and permits conversion to other mission configurations.

The existing attitude error filters and the Roll attitude rate and attitude error filter will be redesigned to add an additional switch point for S-II idle mode. This switch point will be connected to the S-II idle mode signal rather than a Switch Selector discrete. Connecting it to the idle mode signal ensures that the filter is in the proper configuration for the idle mode without additional Switch Selector functions.

The mode signals for S-II idle mode will be routed into the FCC on pins presently designated for switch points 8 and 9. These pins were chosen because the wiring presently exists in IU Networks but switch points 8 and 9 are not used in the Saturn configuration. This allows minimum impact to convert from one mission to a J-2S/LEO mission.

Presently, only six of the eight 50 MA Servo Amplifiers are loaded during coast to ensure minimum transients when going from Coast to S-IVB Burn mode. Since the J-2S/LEO mission will require going from Coast to S-II mode, the remaining two 50 MA Servo Amplifiers will also be loaded during coast.

10.4.2.2.3 (Continued)

The relays to switch the dummy loads in and out as well as the relays required to switch the idle mode filters in and out will be derived from the unused S-IVB relays. Presently, ten relays are used in the S-IVB mode only; six of these will be used for the switching described.

All of the aforementioned changes will require major modifications to the FCC wiring harness.

Typical J-2S/LEO attitude error and attitude rate channels are shown in Figures 10.4.2.2-6 and 10.4.2.2-7, respectively.

10.4.2.2.4 J-2S/Polar Mission

a. Requirements

The requirements for a J-2S/Polar control subsystem are:

1. No S-IVB stage.

2. No APS.

b. Hardware Modification Summary

The only change to the internal construction of the control subsystem components is in the FCC. The following depicts the minimum modifications to the FCC to meet the above requirements.

Since no S-IVB Burn or Coast control is needed, fifty-one modules will be removed from the FCC and replaced with blank boards. This allows versatility in case the APS is ever desired, but also eliminates the expense of flying unnecessary hardware.

The switching within the FCC will be redesigned so that relays will be energized in S-IC mode and de-energized in S-II mode. This reduces the power consumption of the FCC.

The wiring harness will be completely redesigned to remove the unnecessary wiring. Approximately 50% of the wiring harness will be eliminated; however, the FCC pin function will remain the same.

Typical J-2S Polar attitude error and attitude rate channels for the FCC are shown in Figures 10.4.2.2-8 and 10.4.2.2-9, respectively.

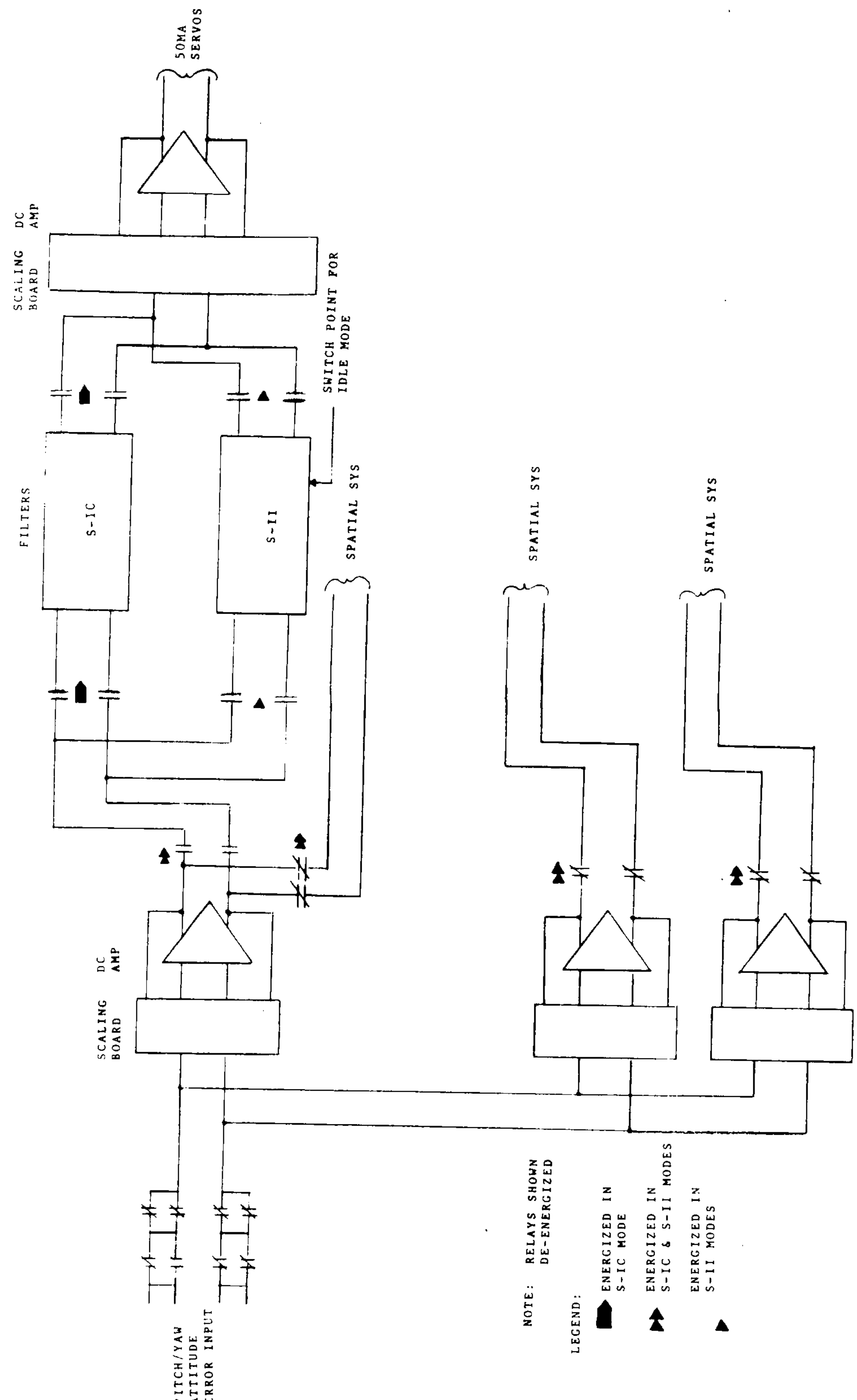


FIGURE 10. 4.2.2 - 6. J 2S/LEO ATTITUDE ERROR CHANNEL

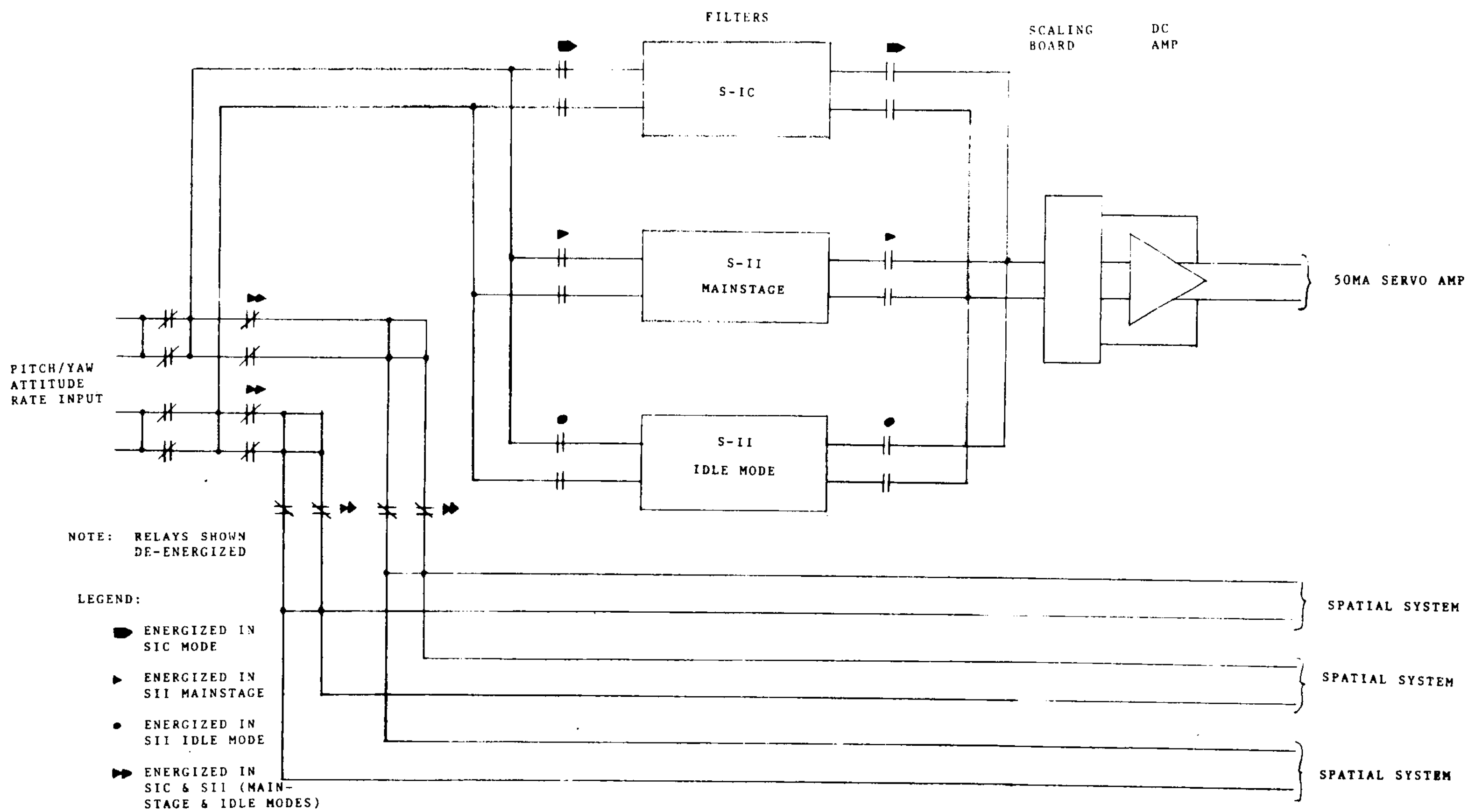


FIGURE 10.4.2.2 - 7. J 2S/LEO ATTITUDE RATE CHANNEL

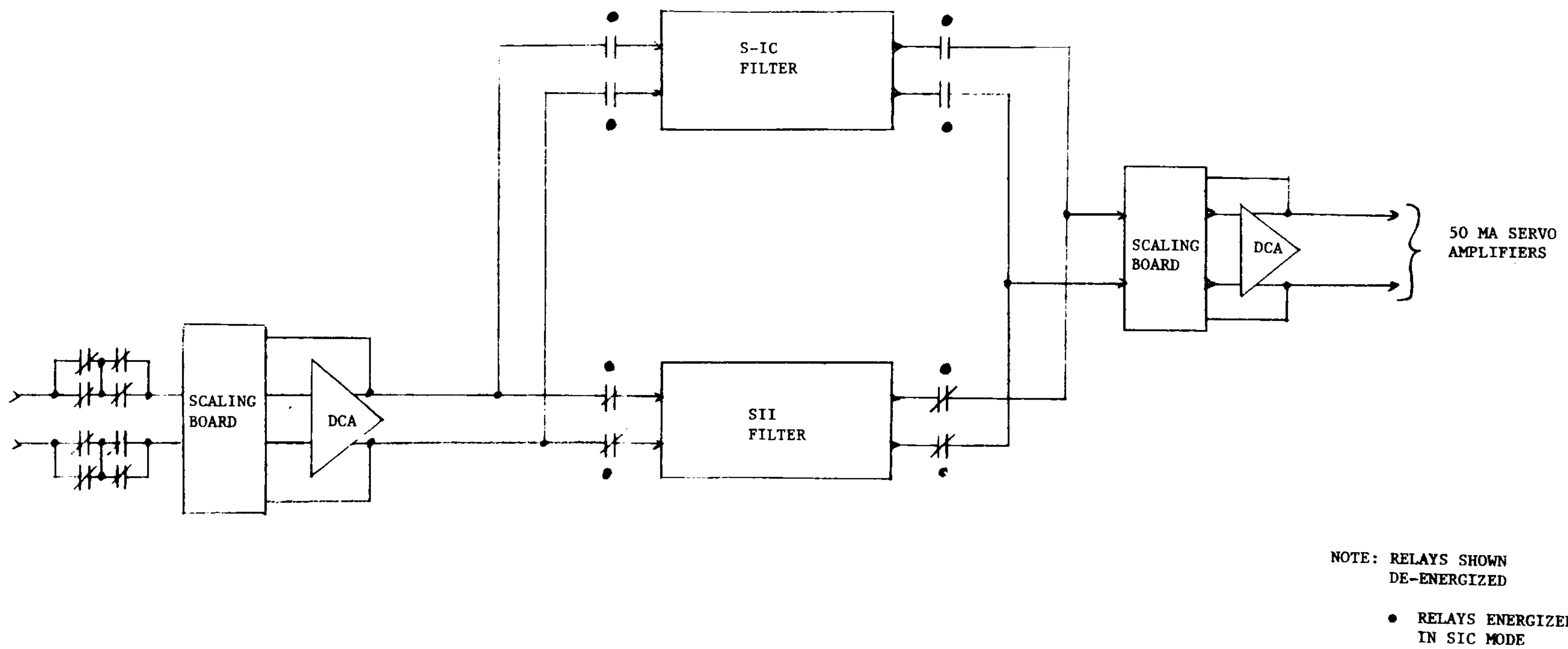


FIGURE 10.4.2.2 - 8. TYPICAL ATTITUDE ERROR CHANNEL

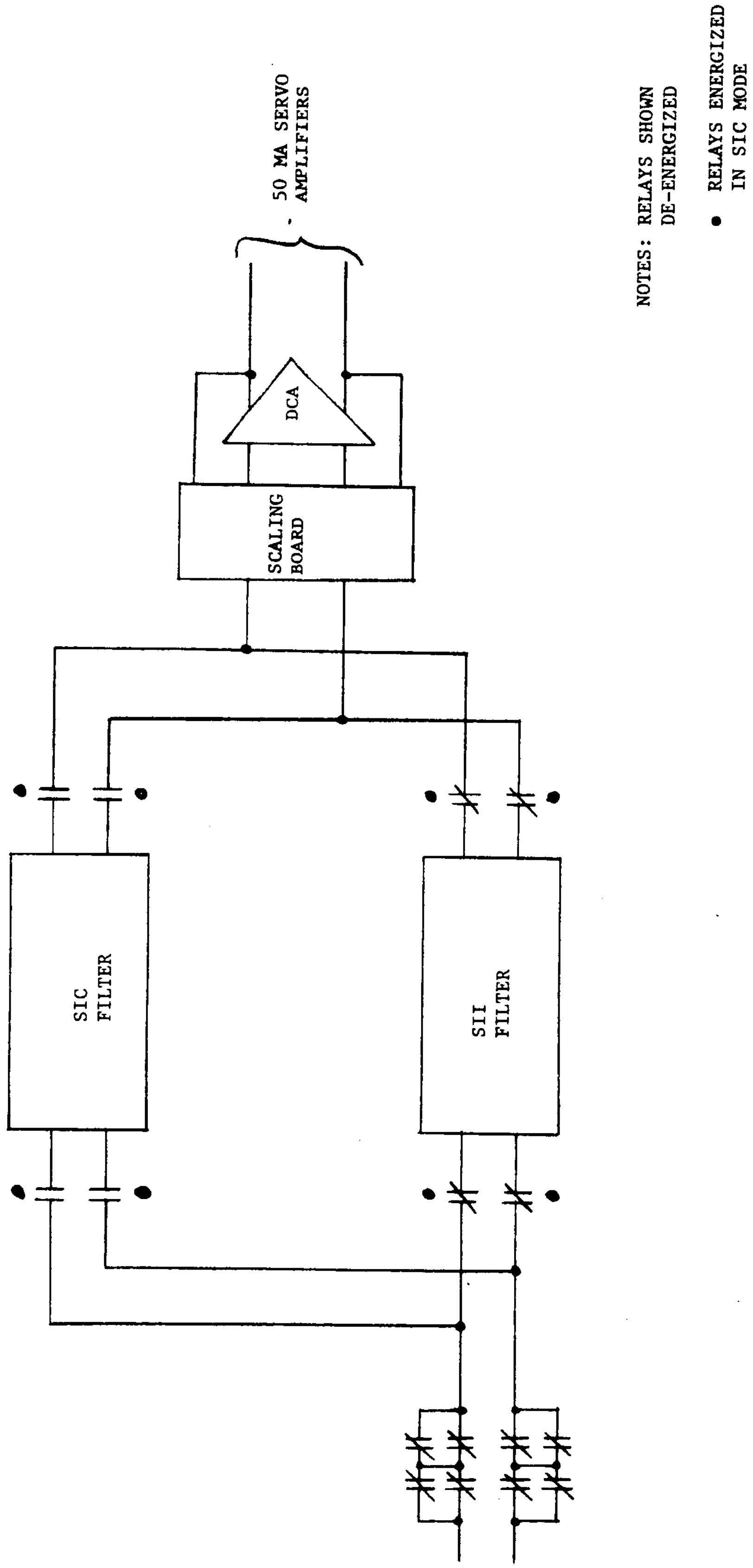


FIGURE 10.4.2.2-9. TYPICAL ATTITUDE RATE CHANNEL

10.4.2.2.4 (Continued)

A preliminary look at completely redesigning the FCC indicates that the non-recurring costs can be absorbed and the unit cost will be reduced. The approach is not feasible, however, because few Polar missions will be flown from the inventory. It is recommended that the essentially universal design described earlier be retained as baseline.

10.4.2.3 Instrumentation and Communication Subsystem

Within this subsystem are included the functions of measuring and telemetry, digital command and tracking.

10.4.2.3.1 J-2S/LOR Mission

There are no system changes that will require additional measurements; therefore, no impact on measuring and telemetry. The digital command and tracking hardware, in its present form, is not impacted by the change to J-2S engines. Using the digital command link, it is possible to address any switch selector function in either the IU or the S-IVB stage. Because a number of switch selector functions change within the stages and the IU, the functional capability of digital command will change. This will be handled with software, however, and requires no hardware change.

10.4.2.3.2 J-2S/Synchronous Mission

a. Requirements

The Synchronous orbit mission resulted in the following additional requirement on the I&C subsystem. See Reference 10.4-39.

The requirement to orient the roll axis perpendicular to the sun and roll at one revolution per hour during the Hohmann transfer, impose new requirements on the CCS system antenna configuration. The present CCS omni capability is insufficient at altitudes encountered in this mission. Further, the single CCS directional antenna cannot be pointed during the Hohmann transfer because of S-IVB stage constraints. Mission analysis, which considered the constraints and the unpredictability of vehicle - ground station positional relationship at this time, showed that the IU required a capability to radiate a donut-shaped pattern around the IU. The conclusion was reached that by using proper combinations of pitch and yaw to maintain the sun pointing during the Hohmann transfer, a 70° width of the donut-shaped pattern (nose to tail) would be sufficient.

10.4.2.3.2 (Continued)

b. Hardware Modification Summary

To meet the communications requirement imposed during the Hohmann transfer, a configuration employing six of the present CCS directional antennas is recommended. The selection of a single antenna for "pointing" purposes will be controlled by the LVDA/LVDC and switch selector. To maintain satisfactory circuit margins, two modified Power Amplifiers are used. A power divider and coax switches are added to permit antenna selection.

c. Hardware Modifications Details

The Synchronous Orbit mission constraints have imposed requirements which exceed the present production IU telemetry RF down-link capability. Other inconsistencies between requirements and capabilities were noted in the measurement and telemetry areas. The command system was reviewed to ensure an adequate up-link circuit margin.

The above areas have been fully assessed and are discussed in the subsequent sections. The I&C hardware modifications necessary to successfully accomplish the Synchronous mission are summarized below:

The current CCS Power Amplifier will be replaced by two modified Power Amplifiers.

Five additional CCS directional antennas will be added.

One additional power divider will be installed.

Two additional coaxial switches will be added.

1. Justification for Change

(a) Radio Frequency Subsystems

The modification of the RF down-link subsystem was limited to equipment in the UHF band. Limiting improvements to the UHF band was chosen because the VHF hardware did not offer as much improvement potential in gain per unit size, ground network coverage, etc. and to conform with the FCC direction to abandon the VHF spectrum in the 1970's.

Circuit margin analysis of the CCS system indicated that with a nominal 20-watt power amplifier a minimum vehicle antenna gain of +4.5 db would be required for satisfactory reception of the down-link PCM telemetry. This requirement was based on:

10.4.2.3.2 (Continued)

Synchronous altitude, 5^{o} elevation angle.

30 ft dish ground antennas.

Continuous vehicle roll at 1 revolution per hour.

The spectrum of mission constraints dictated the antenna beam pattern or coverage necessary: a donut-shaped radiation pattern around the vehicle with 360^{o} coverage in roll and a 60^{o} coverage from nose to tail (centered about $0+90^{o}$). Essentially, this is an omnidirectional antenna system with a required minimum gain of +4.5 db.

A basic problem in achieving this type of omni coverage is that any final design will turn out to be non-omnidirectional. The use of multiple elements is mandatory to minimize the shadowing effects of the vehicle. The use of elements around such a large vehicle, although posing problems, provides an adequate radiation pattern. Feeding these antennas in parallel simplifies the sytem design and operational use, but usually produces interference pattern lobing with deep nulls and wastes power through power division to the antennas and resultant transmission not beamed at earth stations. The optimum omnidirectional antenna system for this type of application, then, is not omnidirectional at all but provides for maximum energy transfer in only one direction at any given time. The antenna system chosen for the Synchronous mission utilizes six CCS directional antennas spaced a nominal 60^{o} apart around the IU. These antennas operated in the low gain configuration provide the desired beam width and gain. Two power amplifiers are used to minimize RF power losses in the distribution system. See Figure 10.4.2.3-1 for a schematic of the CCS RF component layout. A functional block diagram of the instrumentation is seen in Figure 10.4.2.3-2.

Power amplifier and antenna switching is accomplished under control of the LVDC program. Switch selector functions are used in conjunction with coaxial switches to control the beam direction relative to the vehicle. Inherent in the design of the RF system are the following features:

Present capability of the CCS omni antenna system is retained for near-earth orbit use.

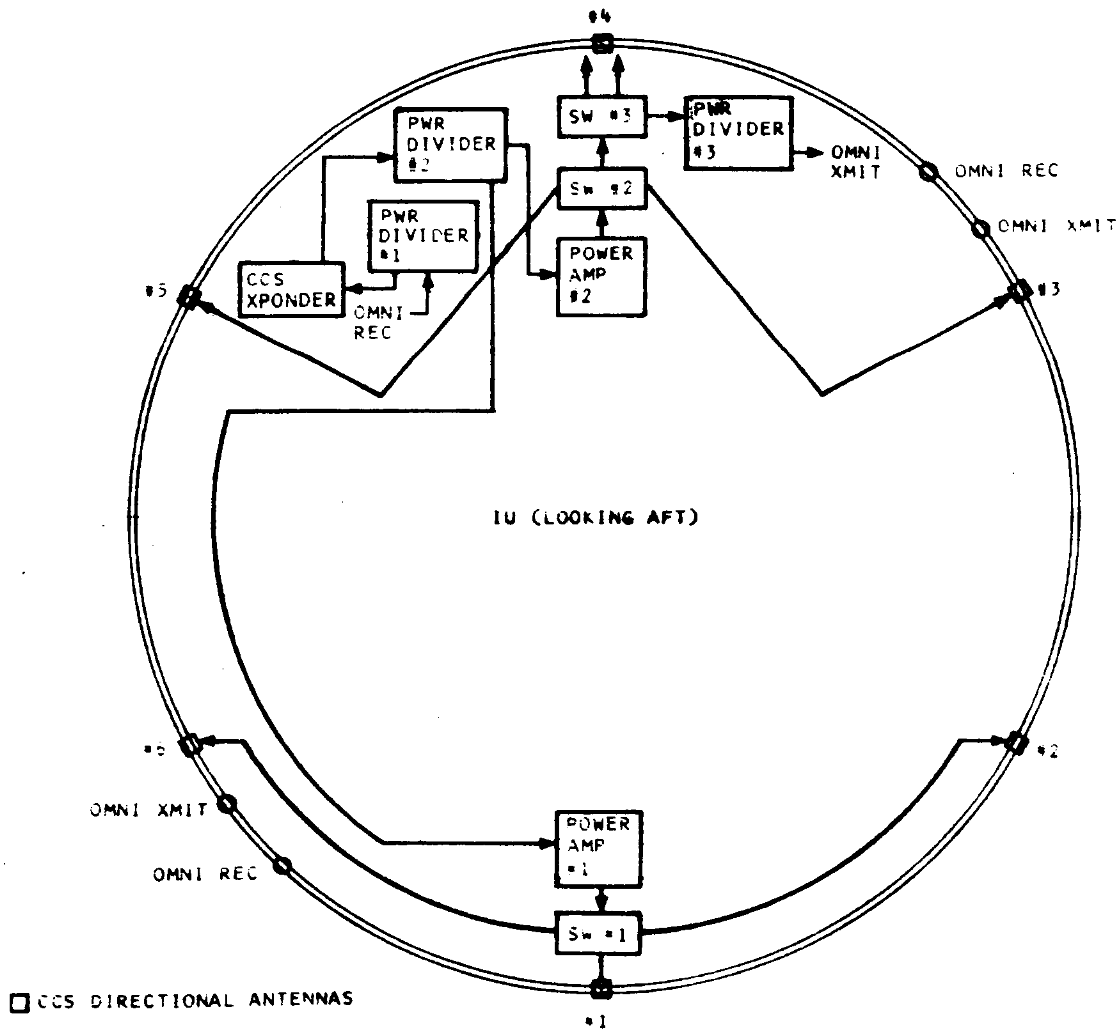


FIGURE 10.4.2.3 - 1. CCS RF COMPONENT LAYOUT

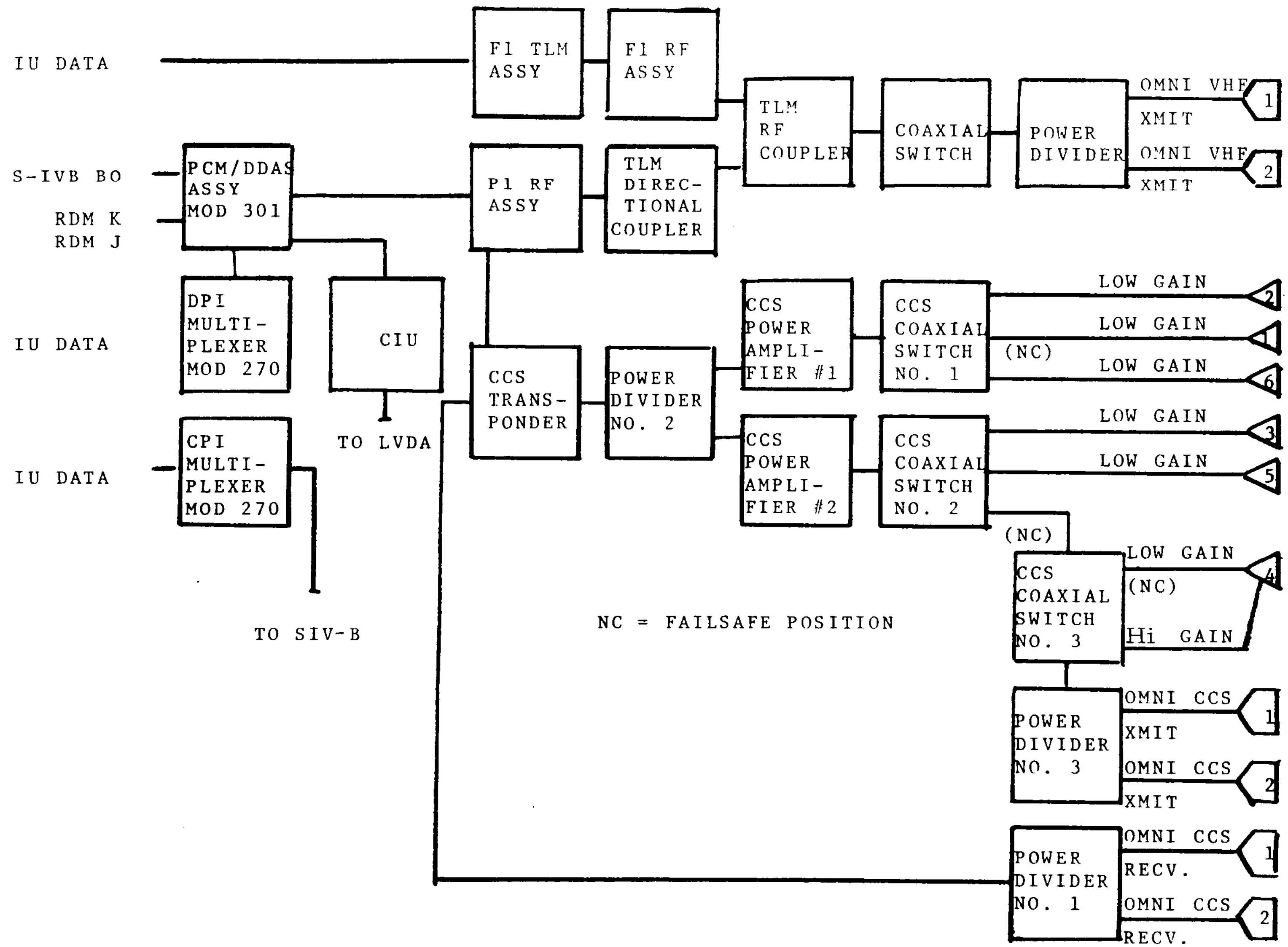


FIGURE 10.4.2.3 - 2. MODIFIED CCS FUNCTIONAL BLOCK DIAGRAM

10.4.2.3.2 (Continued)

Adjacent antenna switching during the transfer trajectory is accomplished by switch selector command resulting in a maximum RF blackout of 50 db. (Carrier phase lock will be maintained although PCM decommutation will become unlocked.).

It is impossible for any two adjacent antennas to radiate at the same time thereby eliminating interference nulls.

The present high gain CCS antenna capability is retained for use at Synchronous altitude after vehicle stabilization prior to the third S-IVB burn.

(b) Telemetry Subsystem

The Mission Control Data will be cross-strapped between the S-IVB and IU as implemented in the baseline vehicle. No hardware modifications are necessary in this area.

(c) Measurement Subsystem

The S-IU 511 baseline IU contains four measuring racks with a total of 80 conditioners. There are many measurements (e.g., vibration and battery temperature) which would be desirable from a post-flight analysis standpoint. The addition of these measurements would have resulted in the requirement for a fifth measuring rack. However, since none of these could be considered in the mandatory category where (a mandatory measurement is defined here as one which, if not obtained, might jeopardize the lives of the crew), it was not recommended that these measurements be added to the IP&C List for the Synchronous Orbit mission. The following measurement changes represent an extension of the baseline measurement philosophy:

PCM Measurements Considered for Addition:

Parameter	Measurement Name	Characteristics
M	6D---Bus Voltage	Similar to M12-601
M	6D---Battery Current	Similar to XM16-601
J	CCS Amp #2 Helix Current	Similar to J82-603

10.4.2.3.2 (Continued

Parameter	Measurement Name	Characteristics
K	CCS Switch #1 Antenna #2 Pos	Similar to VK131-603
K	CCS Switch #1 Antenna #6 Pos	Similar to VK131-603
K	CCS Switch #2 Antenna #3 Pos	Similar to VK131-603
K	CCS Switch #2 Antenna #5 Pos	Similar to VK131-603
K	CCS Switch #3 Antenna #4 High Gain Pos	Similar to VK131-603
K	CCS Switch #3 Omni Antenna Pos	Similar to VK131-603
K	CCS Power Amplifier #1 On Indication	Similar to VK150-601
K	CCS Power Amplifier #1 Inhibit	Similar to VK153-603
K	CCS Power Amplifier #2 On Indication	Similar to VK150-601
K	CCS Power Amplifier #2 Inhibit	Similar to VK153-603
K	CCS Transponder On Indication	Similar to VK150-601
K	CCS Transponder Inhibit	Similar to VK153-603

10.4.2.3.2 (Continued)

The following measurements are no longer required:

VK131-603	CCS Switch Pos Omni
VK132-603	CCS Switch Pos Directional Lo Gain
VK150-601	CCS Transmitter and Amp On Indication
VK153-603	CCS Transmitter and Power Amplifier Inhibit

It is possible that to add the battery and helix current measurements will require a new measuring rack not considered necessary at the time the referenced ECP was prepared. See Reference 10.4-39.

10.4.2.3.3 J-2S/LEO Mission

a. Requirements

The J-2S/LEO mission uses the S-IC and S-II stages to boost a space station into Low-Earth Orbit. A constraint in this study was to consider the mission manned. Furthermore, the J-2S equipped S-II will use the idle mode to circularize the orbit. This is equivalent to lunar injection restart of a typical manned Saturn V. Prior to restart, the vehicle is checked using Mission Control Data. These parameters are cross-strapped between the S-IVB and IU PCM systems on baseline vehicles to assure reliability. Therefore, provisions must be made to cross-strap the S-II and IU PCM system.

There are no system changes within the IU brought about by the J-2S/LEO mission that will require additional measurements; therefore, no impact on measuring and telemetry. Some measurements associated with ECS going to the S-IVB stage and associated with S-IVB J-2 propulsion are no longer required; however, for several reasons hardware should not be changed.

These are not required but may cost more to remove than to leave wired. The actual number of LEO IU's will determine required effort.

C-26	Temperature, IU/S-IVB Exit Coolant
F-10	Flow Rate, S-IVB Inlet Coolant
K-20	S-IVB Burn Mode

10.4.2.3.3 (Continued)

These TM channels are time shared with other, still-required, stage propulsion measurements and therefore only the source can and will be removed. The wiring in cables is best left spare.

G1-403	Position Pitch Actuator S-IVB Engine
G2-403	Position Yaw Actuator S-IVB Engine
H1-403	Valve Current Pitch Actuator S-IVB Engine
H2-403	Valve Current Yaw Actuator S-IVB Engine

Auxiliary Propulsion System (APS) associated measurement will be used with the S-II APS system.

The digital command and tracking hardware in its present form is not impacted by the change to J-2S engines or the LEO mission. Using the digital command link, it is possible to address any switch selector function in either the IU or the S-IVB stage. Because the S-II stage has assumed the S-IVB stage restart function, it will now become addressable via the digital command system and its switch selector. The new IU switch selector functions will also be available. This will be handled with software, however, and requires no hardware change.

b. Hardware Modification Summary

No modification to IU hardware is required. However, to accomplish the IU S-II stage PCM cross-strapping requirement, it is assumed that interconnecting cables through the space station will utilize the same interface cables and pins that now accomplish cross-strapping with the S-IVB PCM.

This is discussed further in paragraph 10.4.2.4.3 b, Electrical Subsystem, LEO mission, Hardware Modification Summary.

10.4.2.3.4 Polar Mission

a. Requirements

The J-2S/Polar mission uses the S-IC and S-II stages to boost a space station directly into a 100 n.m. orbit. No S-II stage restart, extended idle or APS mode are presently required. Therefore, the S-II lifetime is very short and no orbital checkout is required. As a result, the IU and S-II PCM cross-strapping required for LEO mission is not a requirement for Polar missions.

10.4.2.3.4 (Continued)

There are no system changes within the IU brought about by the J-2S/Polar mission that will require additional measurements; therefore, no impact on measuring and telemetry. Some measurements associated with ECS going to the S-IVB stage and associated with S-IVB propulsion are no longer required; however, for several reasons hardware should not be changed.

These are not required but may cost more to remove than to leave wired. The actual number of Polar mission IU's will determine required effort.

C-26	Temperature, IU/S-IVB Exit Coolant
F-10	Flow Rate, S-IVB Inlet Coolant
K-20	S-IVB Burn Mode

These TM channels are time shared with other, still-required stage propulsion measurements and therefore only the source can and will be removed. The wiring in cables is best left spare.

G1-403	Position Pitch Actuator S-IVB Engine
G2-403	Position Yaw Actuator S-IVB Engine
H1-403	Valve Current, Pitch Actuator S-IVB Engine
H2-403	Valve Current, Yaw Actuator S-IVB Engine
H3-404	S-IVB Aux Propulsion IP/IIIP
H4-404	S-IVB Aux Propulsion I-II/I-IV
H5-404	S-IVB Aux Propulsion III-II/III-IV

The digital command and tracking hardware in its present form is not impacted by the change to J-2S engines or the Polar Mission.

b. Hardware Modification Summary

The removal of the requirement to cross-strap the IU and a booster stage PCM's creates a problem. Because the CP1 Model 270 Multiplexer is not exclusively redundant with DP1 Multiplexer, the CP1 is required and must now be used with some IU TM system as other stage PCM's are no longer available. It is therefore required that the CP1 be wired to the IU PCM/DDAS Assembly Model 301

10.4.2.3.4 (Continued)

and receive sample rate and frame synchronous from the IU PCM. No internal changes are required within either the CP1 or the PCM.

10.4.2.4 Electrical Subsystem

Within this subsystem are included the functions of 28 vdc power distribution, sequencing, and interconnection required by other system modifications.

10.4.2.4.1 J-2S/LOR Mission

a. Requirements

There are no system changes required to accomplish the J-2S/LOR mission that cause an impact on the 28 vdc power distribution system.

There is an additional sequencing requirement within the control system FCC to use special filter networks during S-IVB extended idle mode. These filters must then be switched out during S-IVB mainstage burn. In-flight resettable filter switching for switch point 5; therefore, becomes a new requirement, because in the present IU networks the filter switching is not resettable except from GSE.

The only requirement for systems interconnection modification is to implement FCC switch point No. 5 reset from the Switch Selector and isolate the Switch Selector from the GSE reset bus.

b. Hardware Modification Summary

A spare channel of the Switch Selector will be wired to the Control Distributor, to the reset coil of FCC switch point 5 relay. Also wired to that reset is a GSE controlled reset bus. A diode in each of the two lines will isolate the two driving sources (See Figures 10.4.2.2-4 and 10.4.2.2-5).

10.4.2.4.2 J-2S/Synchronous Mission

a. Requirements

The IU lifetime extended to 15 hours imposes new requirements on the IU 28 vdc power distribution system. It is recommended that the three 350 ampere-hour batteries of the baseline IU be replaced with four redesigned 470 ampere-hour batteries. This approach will require development and qualification of the new batteries. An alternate configuration using six of the present IU batteries was studied. While this six-battery approach would eliminate the need for battery development and qualification, the network modifications for the six-battery configuration which would include the addition of a Power Distributor, are significantly more complex than with the four-battery configuration.

10.4.2.4.2 (Continued)

The sequencing and interconnection modification for the LOR mission are also required for the Synchronous mission. In addition, there is a requirement for interconnection of the communication system modifications and the four-battery system. Sequencing using the LVDA/DC and switch selector is also required for the antenna switching.

b. Hardware Modification Summary

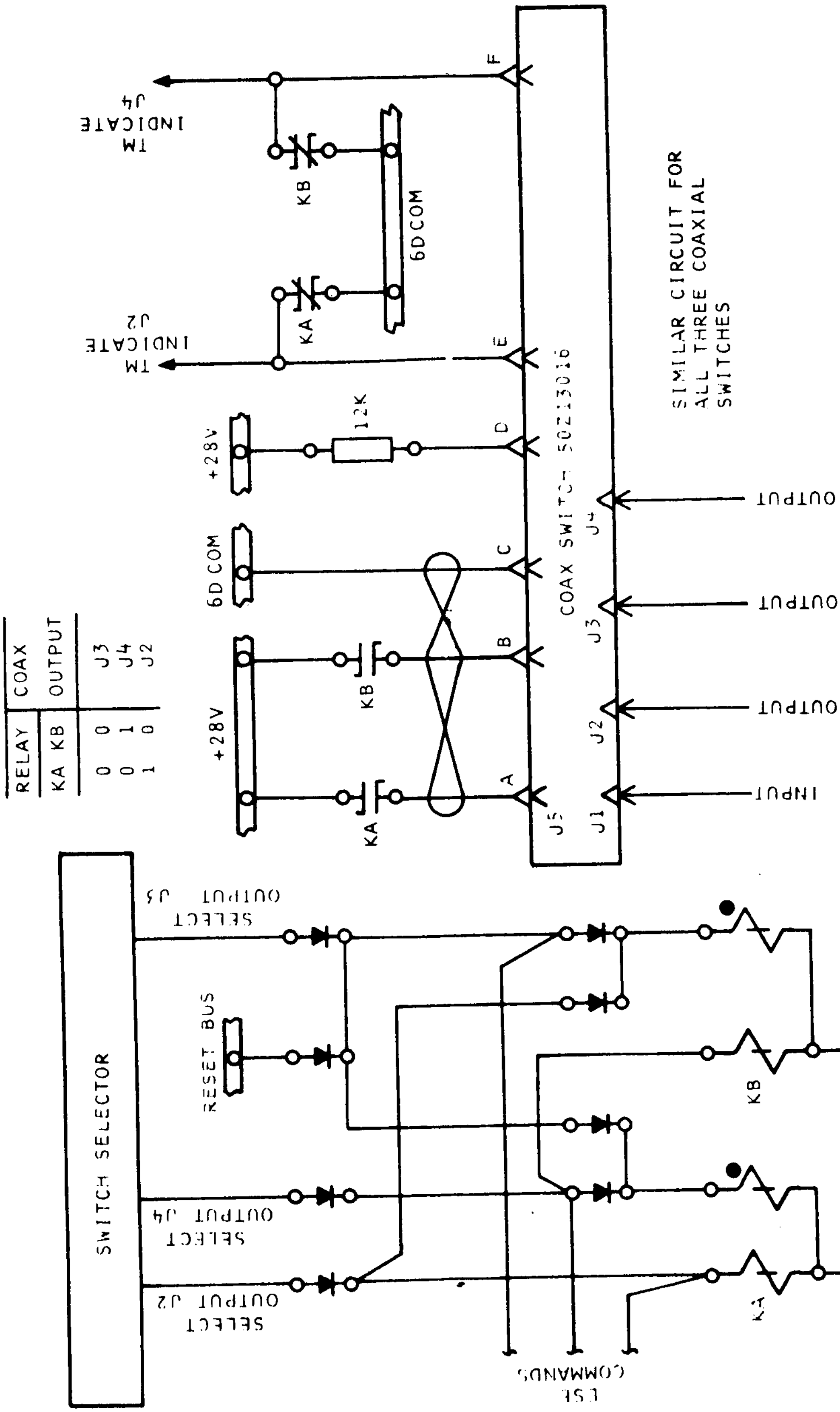
1. Communication System Modification

The communication system for a synchronous orbit mission will consist of the existing VHF telemetry link, the existing CCS telemetry up-link, and a modified CCS telemetry down-link. The modified down-link includes: the CCS transponder, two CCS power-amplifiers, three coax switches, two power dividers, six low/high gain antennas, and two omni antennas. See Para 10.4.2.3.2 for a complete telemetry system description. The electrical system cabling and sequencing logic must be modified due to the new system requirements and the relocation of components.

The sequencing logic for the three coax switches will be similar to the CCS coax switch sequencing logic used on AS 511. Three switch selector commands will be used to select one of the three outputs of each coax switch. ESE simulate commands will be provided for each of the switch selector outputs. The position of each coax switch will be monitored by two telemetry measurements. Each measurement will indicate one of the energized positions. The absence of both measurements will indicate the de-energized position of the switch. A total of nine ESE commands will be required to simulate the nine switch selector commands (See Figure 10.4.2.4-1).

To prevent antenna pattern interaction, only one power amplifier at a time will be active. The power amplifiers will be enabled and inhibited by switch selector commands. Two separate commands are required for each power amplifier to prevent loss of communication during the transition from the inhibited to the enabled stage. A 15-second warm-up period is required before the standby amplifier can be operational. A total of four switch selector commands and four ESE simulate commands will be required for the power amplifier enabling and inhibiting. Telemetry measurements will monitor the status of each power amplifier enabling/inhibiting circuit. Additional measurements may be required by I&C on one or both of the power amplifiers (See Figure 10.4.2.4-2).

Approximately fifteen switch selector commands will be required to initiate the desired sequencing and inhibiting of the power amplifiers and antennas. Approsimately ten relays and thirty-five diodes will be required in the distributors for the sequencing and inhibiting logic.



RELAY		COAX
KA	KB	OUTPUT
0	0	J3
0	1	J4
1	0	J2

FIGURE 10.4.2.4 - 1. COAX SWITCH LOGIC

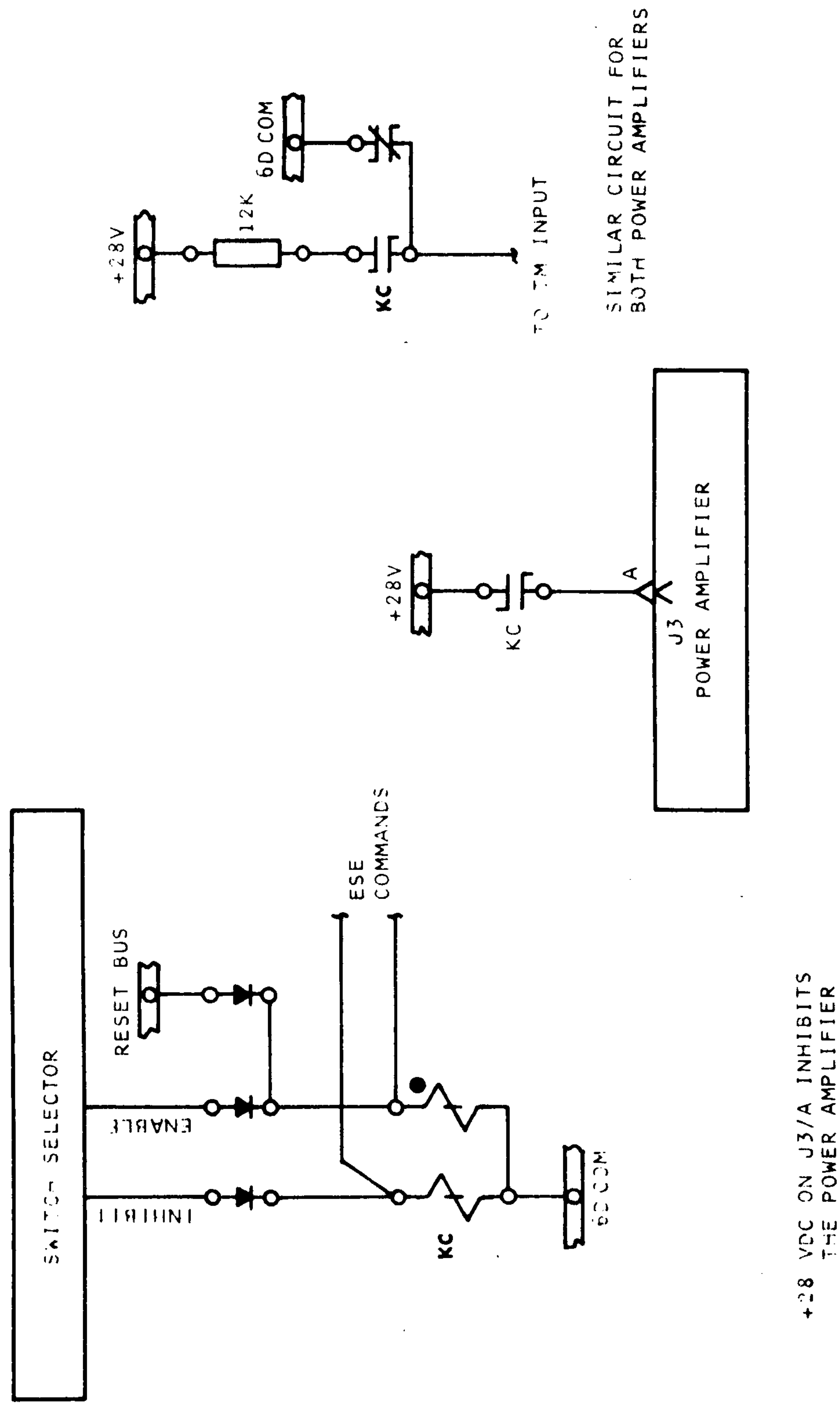


FIGURE 10.4.2.4 - 2. POWER AMPLIFIER INHIBITING

10.4.2.4.2 (Continued)

c. Power System Modification

The synchronous orbit mission requires an extension of the IU power system lifetime from 6.8 hours to 15 hours. Several methods of extending the power system lifetime were investigated. Systems that were studied included:

Four uprated IU batteries

Six of the present IU batteries

Five of the present IU batteries.

The four-uprated battery system is recommended for synchronous orbit mission power system.

1. Electrical Load Profile

Electrical load profiles were prepared using the nominal power requirements of the IU components. The loads were distributed to meet the following criteria:

Battery discharge depth shall not exceed 80% at end-of-mission.

Minimize the number of flight-critical buses.

Isolate the coolant pumps from the other IU loads.

All subsystem components should be on the same bus.

Equalize battery loading.

The nominal values were derived from test data, except for the measuring racks and the Telemetry Calibrator Power and Control Assembly. An estimated load of 1.75 amperes was used for each measuring rack. An estimated load of 0.45 amperes was used for the Telemetry Calibrator Power and Control Assembly. It should be noted that a component could draw more current than previously measured and still meet its specification. For example, the specification for the P1 RF and F1 RF assemblies allows the input current to range as high as 7.5 amperes. The average 28 vdc current for the 34 samples tested has been 3.8 amperes. The possibility of the IU components drawing more than the nominal currents has not been neglected. The six-battery power system and the four-battery power system discharge rate can increase by 25% per battery and still remain within the 80% discharge depth at the end-of-mission. In addition, the 80% discharge depth provides a built-in safety factor for any unexpected increases in the power requirements of the IU components.

10.4.2.4.2 (Continued)

The load profiles also assume that the load is equally shared when a component receives power from several batteries. This assumption is true only if the batteries have approximately equal voltages. Any appreciable differences in the battery voltages could increase the load on the battery with the highest voltage. The problem of load sharing can become critical when the batteries are operated above the 80% discharge depth.

2. Power Shutdown (Selective Shutdown)

The selective shutdown of certain systems was investigated as a method of conserving power. The VHF telemetry and C-Band transponder systems could be shut down once the vehicle was out of range of the ground stations. However, since the overall battery requirements did not change, selective shutdown of these components was not adopted.

3. End-of-Line Shutdown (Figure 10.4.2.4-3)

At the end of the required IU lifetime, which for a particular synchronous orbit mission may be from 8 to 15 hours after lift-off depending on the selected hover point, there will be some time period before the IU batteries are depleted.

During the last two hours of the required IU lifetime, the IU will participate in the initiation of the S-IVB stage passivation by issuing approximately 40 switch selector commands. However, the passivation procedure, which is relatively slow at synchronous altitude, will not be completed at the end of the required IU lifetime (The final steps will be controlled manually from the spacecraft.).

To prevent inadvertent IU or S-IVB switch selector commands and to prevent possible hazardous conditions during the period of battery depletion after the end of the required IU lifetime, power will be removed from the IU.

The system proposed to shut down the power system will prevent any single failure from causing an undesired power transfer. Two separate switch selector commands will initiate the power-down sequence to the LVDA/LVDC and transfer the motor-driven switch to the external position. The LVDA/LVDC power-down sequence must be followed to prevent spurious outputs when power is removed.

The two switch selector commands set relays KA and KB. Normally open contacts of KA and KB are logically "AND" in three circuits: (1) to enable relay coils KC, KD, KE, KF, and KG, (2) to enable power to the contacts of relay KE, and (3) to reset transfer switch relay K8. Relays KF and KG are only required for the six-battery system. Relay KE is used to initiate the LVDA/LVDC power-down sequence. The existing Halt, GCOA, GCOB, and Memory Release commands

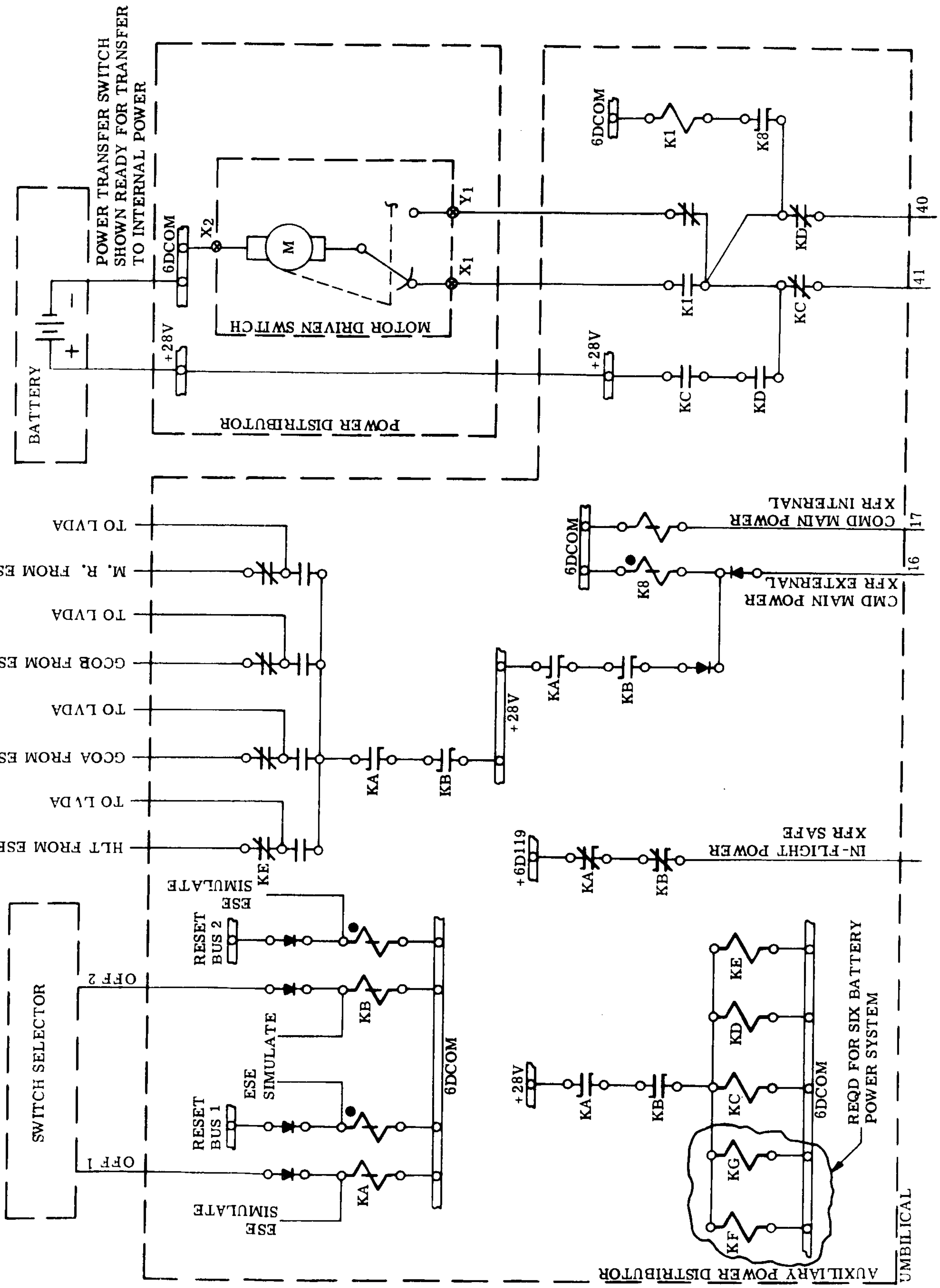


FIGURE 10.4.2.4-3. POWER TRANSFER SWITCH OPERATION, FOUR-BATTERY SYSTEM

10.4.2.4.2 (Continued)

from ESE will be wired through normally closed contacts of KE to the LVDA. The normally open contacts of KE will be wired to +28 volts through normally-open contacts of KA and KB. Relays KA, KB, and KE must be energized to initiate the power-down sequence.

Prior to lift-off, the IU is transferred to internal power. Normally-open contacts of K8 close and energize the coil of relay K1 when the set coil of K8 is energized. Contacts of K1 then transfer and apply ESE power to contacts X_1 - X_2 of the motor-driven switch, driving the switch to the internal power position. The motor wiper stops on contact Y_2. The switch is driven back to the external power position by applying power to contact Y_1 - X_2. Resetting K8 de-energizes relay K1, which then applies ESE power to motor contacts Y_1X_2. In-flight power transfer is performed by resetting relay K8 and supplying 28V power to motor contacts Y_1 - X_2. Power for the power transfer sequence is obtained directly from the battery. Contacts of relays KA and KB are "AND"ed together to provide the 28V power to the switch. Normally closed contacts of the KC and KF are used to interlock the ESE power supply from the IU battery during checkout. Normally closed contacts of KA and KB are "AND'ed together and wired to ESE. This circuit will monitor the logic state of these relays to ensure that KA and KB are reset prior to lift-off.

4. Four Uprated IU Batteries-Recommended Power System (Figure 10.4.2.4-4)

The baseline S-IU-511 uses three batteries. After considering the increased lifetime and electrical energy requirements of the synchronous mission, an attempt was made to employ the four-battery system used on R&D vehicles S-IU-501 through S-IU-503. However, it was found that the ampere-hour capacity of four of the present IU batteries was inadequate to meet the Synchronous Orbit mission requirements.

The Eagle-Picher Company was asked to recommend a battery which could supply 470 ampere-hours at a 25-ampere discharge rate. The Eagle-Picher Company indicated that the rating of the present IU battery could be increased from 350 to 470 ampere-hours by using different battery cells. The present 4240 cell would be replaced by a 4240-5 cell. This cell is presently used in the Agena program. Eagle-Picher engineering is confident that they have the ability to provide a battery for this purpose.

Several new cables will be required to connect the fourth battery into the existing power system. Cable changes to the other three batteries will be slight since the same battery case will be used for both the existing IU battery and the uprated battery. The fourth battery will be used to power the cooling system. The loads on the other three batteries will be optimized by wiring changes within the distributors. The end-of-mission discharge depth on each of the batteries will be approximately 67%.

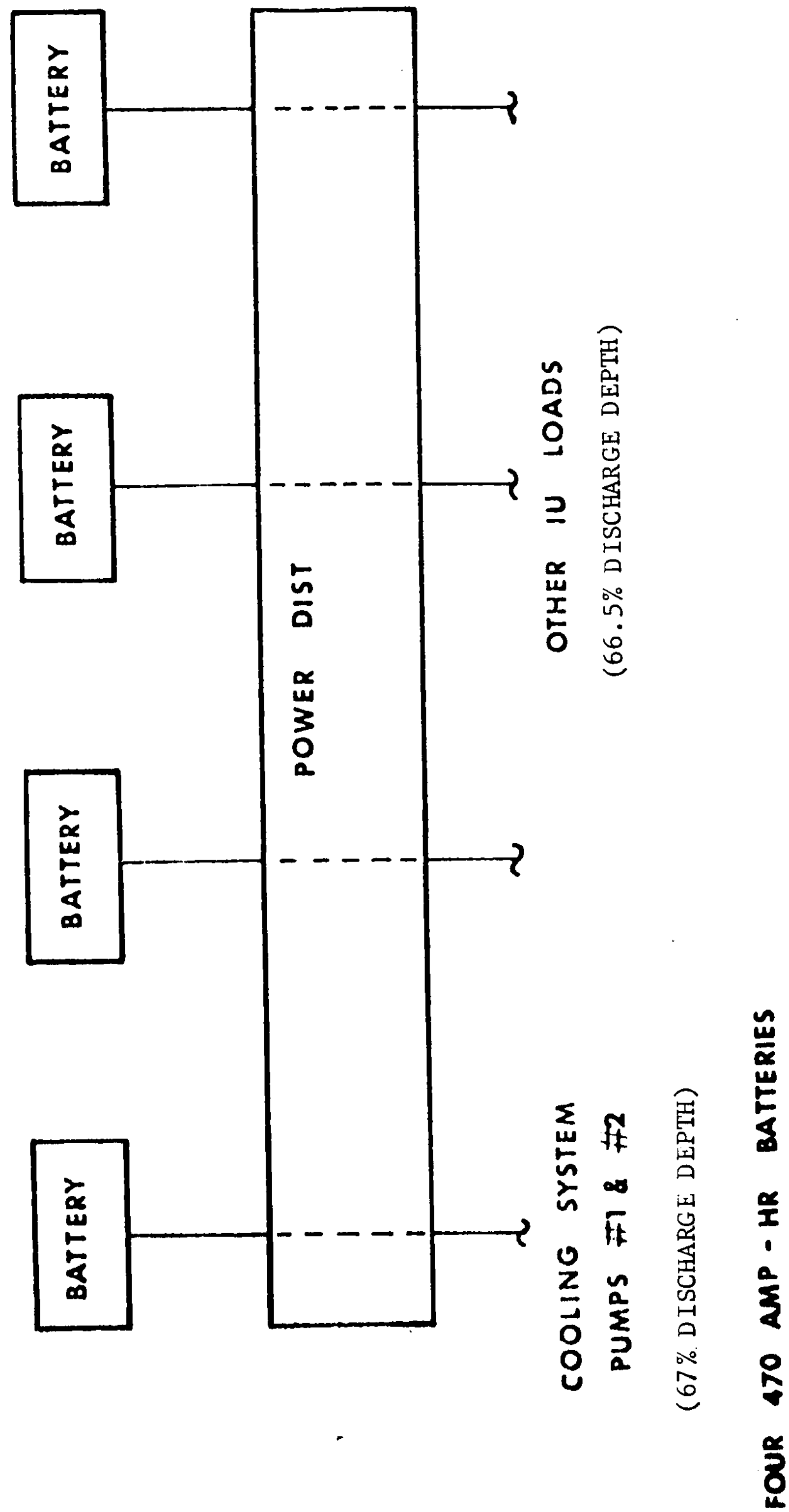


FIGURE 10.4.2.4-4. FOUR BATTERY POWER SYSTEM

10.4.2.4.2 (Continued)

The ground stations must be modified to support a four-battery configuration. ESE bus voltage monitoring capability via the umbilical and test connectors will be required.

Voltage and current measurements must be added for the fourth battery. A 20-ampere loading per battery is expected for a Synchronous Orbit mission. The accuracy of the current measurements could be increased by changing the value of the shunt in the uprated batteries. The present IU battery has a 100-ampere shunt, 100 millivolts at 100 amperes. A 50-ampere shunt, 100 millivolts at 50 amperes, is recommended for the uprated batteries.

10.4.2.4.3 J-2S/LEO Mission

a. Requirements

There are no system changes required to accomplish the J-2S/LEO mission that cause an impact on the 28 vdc power distribution system.

There are two additional requirements on the sequencing system.

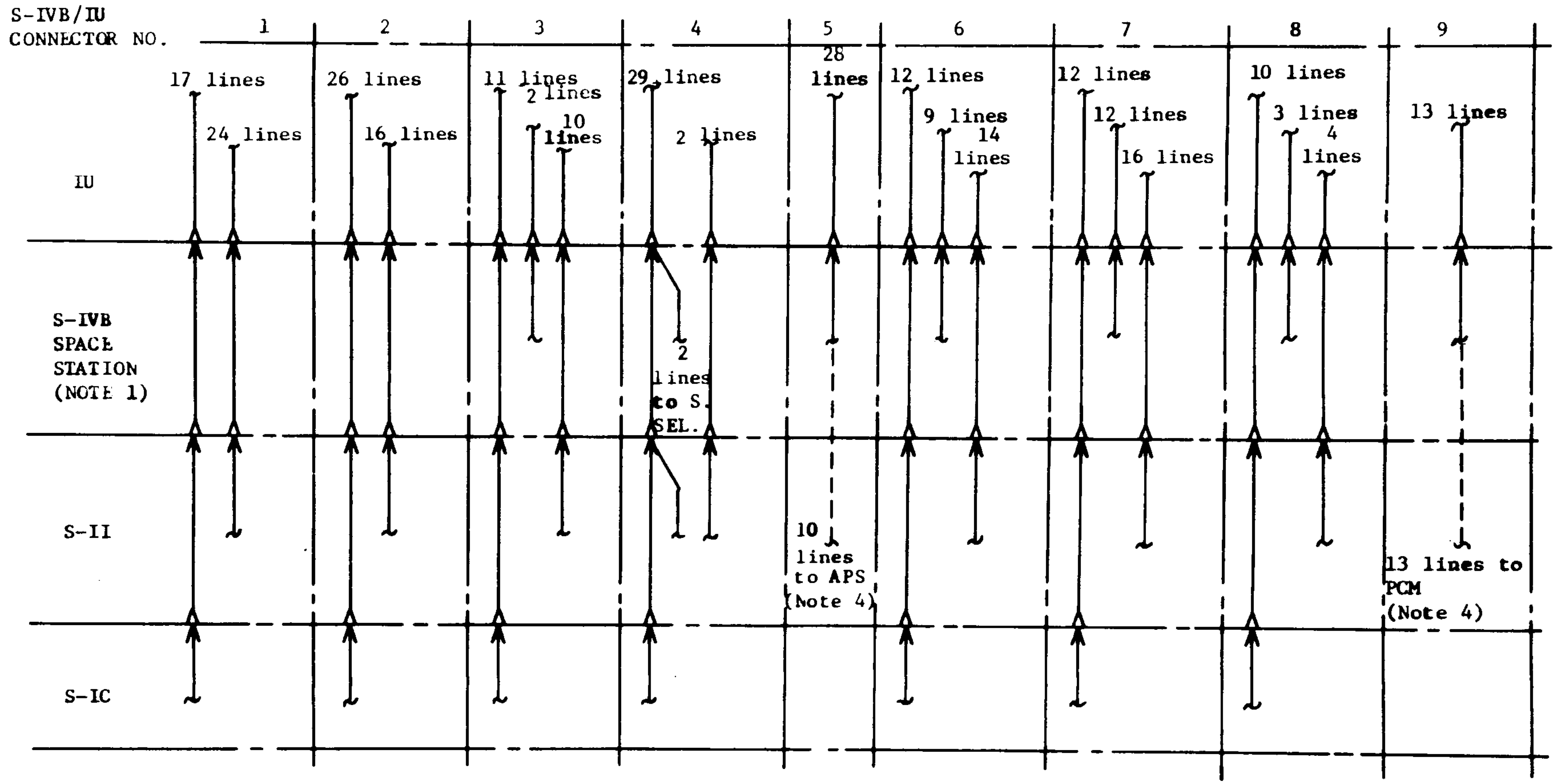
1. Make the S-II burn mode for the FCC in-flight resettable so that S-II coast mode can be accomplished without having to stage the S-II, as is presently done to remove the FCC from S-II burn mode status.

2. Provide sequencing to switch the FCC into S-II idle mode.

An interconnection requirement will impact the space station and S-II stage. All IU/S-IVB interface wires to and from the S-II and S-IC stage will require wiring through the space station. Some IU/S-IVB interface wires associated with the APS moving from the S-IVB stage to the S-II stage and the PCM cross-strapping between IU and S-II stage will require wiring through the space station and into the S-II stage. See Figure 10.4.2.4-5.

b. Hardware Modification Summary

The hardware modification necessary to accomplish the FCC in-flight resettable S-II burn mode is a switch selector output wired to the side of a new S-II burn mode reset relay. This relay will have normally closed contacts in series with the source of 28V power which puts the FCC in the S-II burn mode. Setting the relay will remove power, which is presently done only by staging the S-II. This will then put the FCC in the S-II coast mode.



NOTES

1. S-IVB stage becomes a space station.
2. All lines terminating in S-IC & S-II are wired thru space station.
3. All lines terminating in S-IVB now terminate at IU/S-IVB interface except from connector 5 and 9.
4. Dotted lines show extended lines to functions moved from S-IVB to S-II.

FIGURE 10.4.2.4 - 5. IU TO STAGES, WIRING REQUIRED FOR LEO MISSION

10.4.2.4.3 (Continued)

The requirement in the sequencing system to switch the control system FCC into a S-II idle mode is done with existing switch points 8 and 9 so that no cabling or distributor changes are required. The only impact is on the switch selector table, which is normally modified for every mission.

Definition of the IU/S-IVB interface wired on through the space station and wiring associated with moving the APS and PCM cross-strapping from S-IVB to S-II is required. See Figure 10.4.2.4-5.

1. 201 wires pass through the space station to the S-IC and S-II stages from stage connectors as indicated.

STAGE CONNECTOR	S-IC	S-II
1	17	24
2	26	16
3	11	10
4	27	2
6	12	14
7	12	16
8	10	4
	115	86

2. In addition, 23 wires normally terminating in the S-IVB stage now terminate in the S-II stage. These are:

STAGE CONNECTOR	WIRES
5(APS function)	10
9(PCM function)	13

3. 46 wires normally terminating in the S-IVB stage now terminating at the IU/S-IVB interface are:

STAGE CONNECTOR	WIRES
3	2
4	2
5	18
6	9
7	12
8	3
	46

10.4.2.4.4 J-2S/Polar Mission

a. Requirements

There are no systems changes required to accomplish the J-2S/Polar mission that cause an impact on the 28 vdc power distribution system.

There is a reduced load on the sequencing system because there is no S-IVB stage or any of its functions. The sequencing hardware, 2 switch selector functions (S-IVB Burn Mode), and a Discrete In and Interrupt to the LVDA/DC remain wired spare. If required for some other function,they are available by a simple Control Distributor change. The change in requirements associated with the Polar mission sequencing is mainly software because all S-IVB switch selector functions and time bases associated with the S-IVB stage are no longer required.

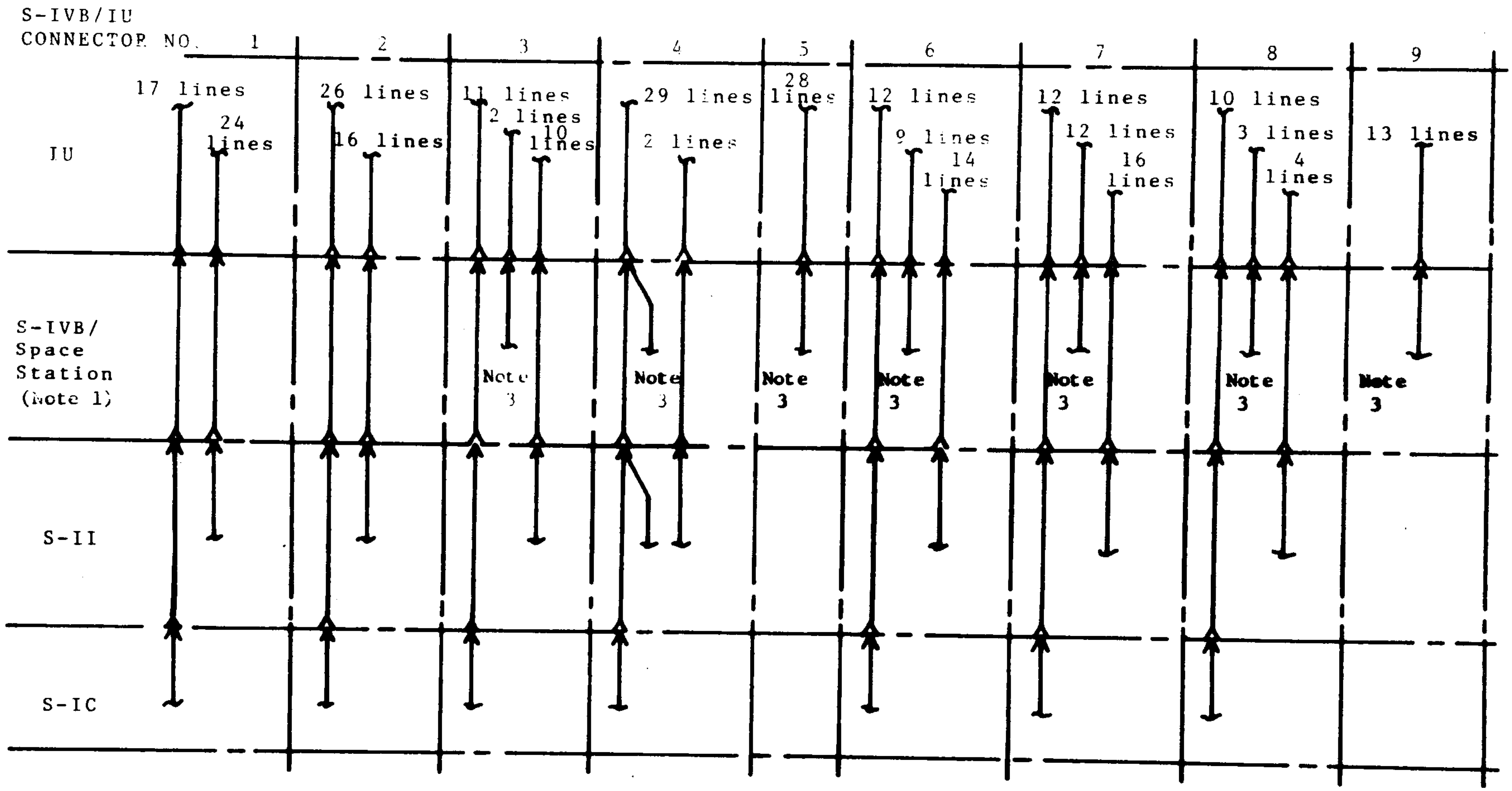
An interconnection requirement will impact the space station. All IU/S-IVB interface wires to and from the S-II and S-IC stage will require wiring through the space station. All IU/S-IVB interface wires to and from the S-IVB will now be terminated at the S-IVB interface and can be left as spares.

b. Hardware Modification Summary

Definition of the IU/S-IVB interface wired on through the space station and IU wiring terminating at the S-IVB interface is required. See Figure 10.4.2.4-6.

1. 201 wires pass through the space station to the S-IC and S-II stages as indicated:

STAGE CONNECTOR	S-IC	S-II
1	17	24
2	26	16
3	11	10
4	27	2
6	12	14
7	12	16
8	10	4
	115	86



NOTES

1. S-IVB stage becomes a space station
2. All lines terminating in S-IC& S-II are wired thru space station.
3. All line terminating in S-IVB now terminate at IU/S-IVB interface

FIGURE 10.4.2.4 - 6. IU TO STAGES, WIRING REQUIRED FOR POLAR MISSION

10.4.2.4.4 (Continued)

2. 69 wires normally terminating in the S-IVB stage now terminating at the IU/S-IVB interface are:

STAGE CONNECTOR	WIRES
3	2
4	2
5	28
6	9
7	12
8	3
9	13
	69

10.4.2.5 Emergency Detection System (EDS)

The EDS is a malfunction alarm and abort handling system. It is a redundant system making use of both triple redundant and dual redundant techniques to achieve maximum reliability.

The EDS parameters that require processing within the system are:

Angular Overrate.

Loss of thrust on two or more engines.

The present system consists of the EDS Display and Control circuitry within the spacecraft, engine thrust sensors and engine cutoff circuitry within each stage, and four units within the IU:

EDS Distributor.

Rate Gyros.

Control Signal Processor.

EDS Engine Cutoff Inhibit Times.

EDS Distributor

The EDS distributor is used to interconnect, enable, or disable all parameters and commands as they require transmission to and from the Booster and Spacecraft.

10.4.2.5 (Continued)

Rate Gyros

The Rate Gyros sense vehicle rate about the P, R, Y axis with triple redundant gyros.

Control Signal Processor

The Control Signal Processor determines from a 2 of 3 majority that an "Angular Overrate" condition exists. The overrate threshold can be switched; for example, it is 4^o/sec in pitch and yaw for most of the first stage burn and 9.2^o/sec thereafter. Roll is not switched and remains at 20^o/sec for the entire boost.

EDS Engine Cutoff Inhibit Timer

The EDS Engine Cutoff Inhibit Timer prevents engine cutoff as part of the abort sequence for the first 30-40 sec from lift-off. This is done to make sure that any engines thrusting will remain thrusting to help pull the booster away from the pad before destruct is required.

10.4.2.5.1 J-2S/LOR Mission

a. Requirements

A requirement of this mission when utilizing J-2S engines is that the S-IVB stage will be operated in the idle mode. When this is done the thrust level is so reduced from mainstage thrust levels that the thrust switches on the J-2S engine in the S-IVB will not properly indicate thrust OK or not OK. It is assumed that the sequence of events will provide the astronauts with procedural logic to interpret the displays.

b. Hardware Modifications

None required.

10.4.2.5.2 J-2S/Synchronous Mission

Requirements and hardware modifications are identical to the LOR mission paragraph 10.4.2.5.1.

10.4.2.5.3 J-2S/LEO Mission

a. Requirements

Two mission requirements must be considered as they pertain to the EDS system.

10.4.2.5.3 (Continued)

1. The S-IVB stage as a propulsion stage no longer exists.

2. The S-II will function in the idle mode.

The absence of the S-IVB stage means that some mode switching to cause the EDS system to monitor the S-IVB in place of other stages is no longer required. This is a very minor function, requiring very little circuitry, and is therefore best left unaltered.

The S-II idle mode causes the same problem discussed under the LOR mission pertaining to S-IVB idle mode. The S-II with J-2S engines will not have functional thrust OK indications while in the idle mode. It is worse in this case because of multiple (5) J-2S engines. Again, however, this will be handled by procedures in the spacecraft, therefore, no hardware modification is required.

b. Hardware Modification

None required.

10.4.2.5.4 J-2S/Polar Mission

a. Requirements

One requirement must be considered as it pertains to the EDS system. The S-IVB stage as a propulsion stage no longer exists. The rationale stated for the LEO mission is true here also. No hardware modification is required.

10.4.2.6 Environmental Control Subsystem (ECS)

The ECS consists of the Thermal Conditioning System and the Gas Bearing Supply System. The Thermal Conditioning System heats or cools, as required, the Astrionics in both the IU and S-IVB stage forward compartment. The Gas Bearing Supply furnishes GN_2 at a regulated temperature and pressure for lubrication of the gas bearing in the ST-124 Inertial Platform System.

10.4.2.6.1 J-2S/LOR Mission

There are no requirements resulting from the change to J-2S engines as used on the LOR mission that affect the ECS.

10.4.2.6.2 J-2S/Synchronous Mission

a. Requirements

10.4.2.6.2 (Continued)

The Synchronous Orbit mission will require Environmental Control System performance beyond present system capabilities. As presently defined. the mission will subject the IU to thermal environments more severe than those existing for the basic Apollo/Saturn V mission. In addition, the mission requires extension of the operational lifetime from the current 6.8 hours to 15 hours duration.

b. Hardware Modification Summary

The above areas have been fully assessed and are discussed in the following paragraphs. The ECS hardware modifications required to successfully accomplish the Synchronous Orbit mission are summarized below:

An additional GN_2 storage sphere (2 cu ft), with appurtenances (mounting panel, plumbing, etc.) will be placed in location 23

The current TCS orifice regulator assembly will be replaced by a redesigned assembly.

c. Justification for Change

Thermal Conditioning System

Mission thermal analyses have been performed to assess the temperature control parameters and assure that positive thermal control will be maintained at all times. As presently defined, the mission consists of the following flight phases:

1. Low Earth Orbit - up to 7.5 hours of velocity-oriented low earth orbit (100 n.m.) at inclinations from 28.5° to 64°.

2. Hohmann transfer - 5.5 hours of transfer trajectory with vehicle broadside to the sun and rolling at a rate of one revolutions per hour.

3. Synchronous Orbit - up to 2 hours of Synchronous Orbit at inclinations up to 64°.

In only one of these phases of flight (Hohmann transfer) is the thermal environment known and fixed. The Low Earth Orbit can range from "cold" (maximum time in earth shadow, 28.5° inclination) to "hot" (zero time in shadow, 64° inclination) with all-inclusive cases possible as a function of inclination and date and time of launch. The Synchronous Orbit can also range from cold (vehicle parallel to solar vector) to hot (vehicle broadside to solar vector) for the 2 hours of orbit, with all intermediate conditions again possible. Without definite knowledge concerning a particular mission at a particular date it became necessary to examine the extreme

10.4.2.6.2 (Continued)

cases. The results are compiled below, showing net system heat load.

Flight Phase	"Hot" Case i=64° in earth orbit, vehicle broadside to sun in Synchronous Orbit	"Cold" Case i=28.5° in Low Earth Orbit, vehicle axis parallel to solar vector in Synchronous Orbit
Ascent	6.0 kw	
Low Earth Orbit	2.21	1.05
Hohmann Transfer	2.05	2.05
Synchronous Orbit	2.65	-0.6

Total energy expended in "hot" case - 36 kwh

The IU coldplate heat gains/losses were determined by means of a numerical transient analysis in which each vehicle coldplate position was broken down into a system of 28 nodes. The thermal energy incident on the system was obtained from a separate computer program with the synchronous orbit mission profile. The S-IVB coldplate heat losses for the maximum cold case (Low Earth Orbit and Synchronous Orbit) and heat losses for the Hohmann transfer period were provided by McDonnell-Douglas. Coldplate heat gain/loss profiles for the hot low-earth and synchronous orbit phases were unavailable. It became necessary to approximate these values to complete the evaluation of the TCS performance. The conservative numbers used were (net, average values): No heat gain (0 kw) in Low Earth Orbit, and 0.5 kw gain in Synchronous Orbit.

As observed from system heat loads, acceptable heat loads (0-9 kw) are maintained during all flight phases for the worst hot case. The total energy expended (maximum) over 15 hours duration (maximum for the Synchronous Orbit mission) is 36 kilowatt-hours. The capacity of the present cooling system is approximately 45 kilowatt-hours, i.e., there is sufficient water stored in the water accumulator (146 lbs) to provide for 45 kwh of cooling--an ample amount for the Synchronous Orbit mission.

For the worst "cold" case, as illustrated in the table, net positive system heat loads are maintained throughout synchronous orbit injection. In the final flight

10.4.2.6.2 (Continued)

phase however, the heat load drops to -0.6 kw, a net loss. This condition is identical to that which currently exists on operational Saturn V missions in cold translunar coast trajectory, and is not considered unacceptable. A heat loss rate of 0.6 kw is equivalent to about a 2°F/hour M/W temperature decrease. Owing to the relatively short duration of the Synchronous Orbit flight phase, temperature requirements will be met at all times.

The limiting factor in ECS operating life is the consumption of expendable fluids--water for Thermal Conditioning System (TCS) cooling, GN_2 for TCS pressurization, and GN_2 for the Gas Bearing Supply (GBS) to the Inertial Platform. It was established in the preceding paragraphs that there is sufficient water stored in the water accumulator to complete the Synchronous mission in the worst (hot) case.

With regard to the GN_2 for the TCS pressurization, the maximum use rate of .097 lbs/hr defines 2.18 lbs of GN_2 required to satisfy a 150% safety factor for a 15 hour mission. The present storage sphere (165 cu in.) has 1.2 lbs of usable GN_2 at lift-off. Thus, two possibilities exist: (1) increase the storage capacity (enlarge present sphere or add an additional sphere), or(2) decrease the GN_2 use rate to a level that can be supplied by the current storage configuration. Assessment has confirmed that the latter may be accomplished with a nominal hardware change by resizing the orifices within the orifice regulator assembly.

Gas Bearing Supply System

Additional GN_2 for the ST-124M Inertial Platform will be required for the Synchronous Orbit mission due to the extended-life requirement. The present storage sphere (2 cu ft) has about 26 lbs of usable GN_2 at lift-off. The 2.2 lbs/hr use rate defines 49.5 lbs of GN_2 required for 150% safety factor for 15 hours. Assessments have established that the least impact approach is the addition of a second 2 cu ft sphere to operate in parallel with the current sphere.

10.4.2.6.3 J-2S/LEO Mission

a. Requirements

This mission will have a space station substituted for the S-IVB stage. This relieves the requirement to provide thermal conditioning to the S-IVB stage forward compartment and requires that the inlet and exit the S-IVB stage presently uses, be capped in the IU.

10.4.2.6.3 (Continued)

b. Hardware Modification Summary

At present, ECS mating with the S-IVB is done with quick disconnects. The inlet is fitted with a quick disconnect adaptor which will be capped with a quick disconnect socket and then a cap. The exit is presently fitted with a flexhose, adaptor and quick disconnect socket. This will be deleted and a quick disconnect assembly and cap substituted.

10.4.2.6.4 J-2S/Polar Mission

Requirements and hardware modifications are identical to those stated for the LEO mission in paragraph 10.4.2.6.3.

10.4.3 Flight Software

This paragraph contains the flight software and attitude control system studies directed toward identifying modifications to baseline systems due to J-2S implementation. The LOR, Synchronous, LEO, and Polar missions were examined and the results and recommendations are included in the following paragraphs. The LOR was used as the baseline mission for both the flight program and attitude control system evaluations.

The LVDC flight program may be separated into three major divisions: the boost major loop, the minor loop, and the orbital flight program. Guidance and navigation calculations are performed in the major loop as well as timekeeping and other repetitive functions which do not occur on an interrupt basis. The ST-124M platform gimbal angle sampling, accelerometer sampling, and control system computations are done in the flight software. A self-test and data storage routine for use during earth parking orbit and a telemetry time-sharing routine that is used while the vehicle is over receiving stations make up the orbital flight program.

First stage guidance generates steering commands during the time from guidance reference release (GRR) to initiation of Iterative Guidance Mode (IGM). The basic scheme is a time-tilt profile. A fourth-degree time polynomial, obtained from a least-square curve fit of the nominal baseline attitude versus time, is used to determine the commanded steering angle. Rate limiting to keep the rate changes within safe bounds prevents the commanded attitude changes from exceeding one degree per second. Tilt arrest, the time at which time-tilt guidance ends, occurs approximately 140 seconds after lift-off. After tilt arrest, all guidance commands are frozen until the initiation of Iterative Guidance Mode (IGM).

IGM is designed to direct the Saturn V vehicle to the desired end conditions in a near optimum fashion. The thrust vector steering law on which IGM is based is approximately the optimum steering function for planar motion of a point mass vehicle over a flat earth. The basic equation for the attitude is applied in both the pitch and yaw planes in IGM to provide three-dimensional guidance. In addition, the guidance scheme employed for Saturn V missions permits up to three distinct thrust levels during boost to orbit, and two thrust levels during out of orbit burn. Since IGM is a path adaptive scheme, the vehicle state (position, velocity, and acceleration) is a required input to the guidance equations. IGM thus compensates for non-nominal vehicle performance and "homes" in on the desired end conditions. For the baseline mission (LOR), IGM will be implemented during two periods for this flight. The first (or boost) period begins during S-II burn and terminates at first S-IVB cutoff when the vehicle enters an earth parking orbit. The second period, out of (parking) orbit, begins after S-IVB reignition. It is completed upon insertion of the vehicle into the target orbit at S-IVB second cutoff. For the LOR mission, this orbit will be a translunar ellipse. The equations and logic needed to implement IGM are essentially the same for both periods. These equations are found in Reference 10.4-46.

10.4.3 (Continued)

The Boost Navigation routine determines position, velocity, and acceleration relative to the inertial (X_S, Y_S, Z_S) coordinate frame as needed for inputs to the boost guidance equations. The computations for position and velocity are carried in one-and-one-half precision. Boost Navigation is used from Guidance Reference Release (GRR) to T5 + 100 seconds (S-IVB cutoff +100) and from Time Base 6 alert to T7 + 15 seconds. Time Base 6 (chilldown) alert occurs at first orbital navigation cycle after T6-9 seconds. The baseline mission time line is given in Table 10.4.3-I.

The evaluation of gravitation is a three-step process. In the first step, the Y component of the vehicle position in the geocentric equatorial coordinate system is computed. In the second step, the gravitational acceleration is computed from a zonal harmonic expansion of the earth potential. In the third step, the gravitational components are transformed back into the navigation coordinate system.

The gravitation model evaluates only the spherical earth and first oblateness terms during most portions of flight. This is done by setting the third and fourth terms to zero. The accuracy of the navigation routine is not seriously degraded by neglecting all but the first two terms in the potential expansion. However, the third and fourth terms are used during orbital navigation since the total time in orbit is several hours and S-IVB restart depends on navigation.

Four missions were evaluated to determine impact on the flight software due to the inclusion of J-2S engine characteristics. These missions: LOR, Synchronous, LEO, and Polar, are treated separately in each of the following paragraphs. The LOR mission was used as a baseline for the software analysis and definitions of the various functions refers to an LOR type mission unless otherwise stated. The remaining missions were evaluated as to the deviations from the baseline. This direction was chosen because the majority of existing documentation refers to a LOR type mission. The approximate mission time lines are given in Table 10.4.3-I, 10.4.3-II and 10.4.3-III.

Attitude control system studies were directed to determining modifications to the flight control system due to the J-2S engine. The AS 504 flight control system was considered to be representative of the AS 511 J-2 LOR baseline flight control system. The analysis and conclusions are included with the guidance studies in the following paragraphs.

TABLE 10.4.3-I. APPROXIMATE MISSION TIME LINE LOR MISSION

LOR

Time Base	Time (Seconds)	Event
TB0	-17	Guidance Reference Release
TB1	0	Liftoff
	85	Q Max
TB2	149	Center Engine Cutoff
	153	Start Chi Freeze
TB3	161	S-IC Cutoff
	165	S-II Ignition
	167	Shift MR to 5.5
	193	Interstage Separation
	198	LES Separation
	426	MR Shift to 4.7
TB4	543	S-II Cutoff
	543	S-IVB Ignition
TB5	676	S-IVB Cutoff
TB6	2906	Restart Preparation
	3476	S-IVB Ignition
TB7	3793	S-IVB Cutoff

LOR - J-2S

Time Base	Time (Seconds)	Event
TB0	-17	Guidance Reference Release
TB1	0	Liftoff
	85	Q Max
TB2	149	Center Engine Cutoff
	153	Start Chi Freeze
TB3	161	S-IC Cutoff
	165	S-II Ignition (idle)
	166	S-II Mainstage
	168	S-II MR Shift (5 - 5.5)
	193	Interstage Separation
	198	LES Separation (end chi freeze)
	396	S-II MR Shift (5.5 - 4.7)
TB4	498	S-II Cutoff
	498	S-IVB Ignition (idle)
	499	S-IVB Mainstage
TB5	617	S-IVB Cutoff (1st burn)
	3 Rev ~ 4.5 hrs	Parking Orbit Coast
TB6	3317	S-IVB (idle)
	3417	S-IVB Mainstage
TB7	3708	S-IVB Cutoff (2nd burn)

TABLE 10.4.3-II. APPROXIMATE MISSION TIME LINE

SYNCHRONOUS			LOW EARTH ORBIT		
Time Base	Time (Seconds)	Event	Time Base	Time (Seconds)	Event
TB0	-17	Guidance Reference Release	TB0	-17	Guidance Reference Release
TB1	0	Liftoff	TB1	0	Liftoff
	84	Q Max		82	Q Max
TB2	149	Center Engine Cutoff	TB2	149	Center Engine Cutoff
TB3	161	S-IC Cutoff	TB3	161	S-IC Cutoff
	165	S-II Ignition (idle)		165	S-II Ignition (idle)
	166	S-II Mainstage		166	S-II Mainstage
	168	S-II MR Shift (5 - 5.5)		168	S-II MR Shift (5 - 5.5)
	193	Interstage Separation		193	Interstage Separation
	198	LES Separation		198	LES Separation
	396	S-II MR Shift (5.5 - 4.7)		396	S-II MR Shift (5.5 - 4.7)
TB4	489	S-II Cutoff	TB4	486	S-II Cutoff (1st burn)
	489	S-IVB Ignition (idle)		$\sim$45 min	Transfer Coast
	490	S-IVB Mainstage			
	493	S-IVB MR Shift (5 - 5.5)			
TB5	545	S-IVB Cutoff (1st burn)	TB5	3065	Local Horizontal Tracking
	5 Rev $\sim$7.5 hrs	Parking Orbit Coast			
TB6	5045	S-IVB Reignition (idle)	TB6	3165	S-II Reignition (idle)
	5145	S-IVB Mainstage			
	5147	S-IVB MR Shift (5 - 5.5)			
TB7	5353	S-IVB Cutoff (2nd burn)	TB7	3305	S-II Cutoff (2nd burn)
	5.3 hrs	Transfer Orbit			
TB8	8533	S-IVB Reignition (idle)			
	8633	S-IVB Mainstage			
	8635	S-IVB MR Shift (5 - 5.5)			
TB9	8725	S-IVB Cutoff (3rd burn)			

TABLE 10.4.3-III. APPROXIMATE MISSION TIME LINE POLAR ORBIT

Time Base	Time (Seconds)	Event
TB0	-17	Guidance Reference Release
TB1	0	Liftoff
	79	Q Max
TB2	149	Center Engine Cutoff
TB3	161	S-IC Cutoff
	165	S-II Ignition (idle)
	166	S-II Mainstage
	168	MR Shift to 5.5
	193	Interstage Separation
	198	LES Separation
	396	MR Shift to 4.7
TB4*	497	S-II Cutoff

*Orbital phase past time base 4 has not been defined.

10.4.3.1 Boost Major Loop

The major loop contains the navigation, guidance, timekeeping, and other repetitive operations of the flight program. The boost navigation routine combines gravitational acceleration with measured ST-124M platform data to compute position and velocity. Guidance in the first stage, referred to as "time-tilt", generates attitude commands as a function of time. This "open-loop" guidance is not affected by vehicle dynamics and the rate of turn of the vehicle about each axis of the reference frame is limited to one degree per second by the flight control system of the vehicle. Just prior to first stage cutoff, the attitude commands are "frozen" until the initiation of the Iterative Guidance Mode (IGM) in the second stage. This freeze is to reduce flight perturbations caused by staging. The IGM routine is primarily used to compute the steering angle commands and the "time-to-go" for the remaining burn time of the vehicle. When mission design requires another burn after insertion into earth parking orbit, onboard computations determine when the restart preparation should begin in order to inject the vehicle into the desired trajectory. IGM guidance is normally used for this second burn if guidance is required.

The accelerometer processing routine accomplishes two main objectives: It accumulates velocities as measured by the ST-124M platform and detects velocity measurement errors through "reasonableness" tests. The "reasonableness" test is designed to detect large errors in the change velocity. A velocity change is considered reasonable if it falls within ±50 percent of the expected change, enlarged by a Reasonable Test Constant (RTC), as discussed in Reference 10.4-31.

10.4.3.1.1 J-2S/LOR

The LOR mission design trajectory is given in Reference 10.4-53. The mission profile is injection into a 72-hour lunar transfer trajectory via a 100-nautical mile earth parking orbit by a three-stage vehicle with an Apollo payload configuration. Two burns of the S-IVB and a three-revolution parking orbit (4-1/2 hours) were used. The LOR mission was analyzed and no modifications to the flight software are necessary to include J-2S engines. The presettings which are mission dependent will change as they do from mission to mission. Although no modifications to the software are required for the J-2S engine, a modification to the cutoff logic is recommended (paragraph 10.4.3.5.5) to utilize the characteristics of the J-2S engine to improve insertion conditions.

10.4.3.1.2 J-2S/Synchronous Orbit

Evaluation of software major loop changes for the Synchronous Orbit profile was made and no problem areas peculiar to the J-2S engine were found. The major modifications necessary for a Synchronous mission are defined in Reference 10.4-39. No changes are necessary to the flight program, Reference 10.4-39,

10.4.3.1.2 (Continued)

other than IGM presettings, to include the J-2S engine characteristics in a Synchronous mission. However, the recommended changes to the cutoff loop paragraph 10. 4. 3. 5. 5 would be a desirable modification.

10.4.3.1.3 J-2S/LEO

The LEO mission, Reference 10.4-10, involves a two-stage vehicle insertion into an elliptical orbit at 100 nautical miles. A second burn of the second stage using idle mode thrust is used to circularize the orbit at 300 nautical miles. The analysis and simulation was divided into three distinct sections: ascent, coast, and idle mode circularization. The coast and targeting will be discussed under the Orbital Program section.

The basic simulation tool used in guidance analysis was the "6-Degree-of-Freedom" simulator. The "6-D" is a simulation of the Saturn V vehicle, LVDC flight program, and the LVDC/LVDA interfaces. A mathematical model of the atmosphere is used to calculate atmospheric density and speed of sound as functions of altitude. These computations are used in calculating the aerodynamic forces on the S-IC stage. The wind model simulates atmospheric winds in addition to the nominal rotating atmosphere. This is accomplished by assuming that the wind vector is perpendicular to the radius vector. The direction of the wind vector is determined by adding an input wind azimuth to the vehicle flight azimuth. Gravitational effects are calculated as functions of radius, geocentric latitude and position. The control law simulated is the same control law used in the Flight Control System. The simulation uses engine position, center of gravity position, engine vacuum thrust, and nozzle deflections to calculate forces and moments about the body axes caused by engine thrust. The vehicle control gains, moments of inertia, moment arms, and applied forces and torques are used to compute angular accelerations from which vehicle attitude rates and attitudes are determined.

The simulator's accelerometer processing section receives velocity components from the integration routines and differences these values from those of the previous computation cycle to determine the velocity change. This velocity is quantized and added to the previous accumulation of velocity. The resulting velocity components are input into the flight program.

AS 503 C' filter and actuator characteristics were used in the "6-D" for guidance and control system evaluation. Time base switching is simulated using mass, thrust values, and guidance signals. Vehicle events normally sequenced by the flight program through the Switch Selector are simulated realistically. The simulator uses the specified time-tilt polynomial guidance scheme for the S-IC stage and the Iterative Guidance Mode (IGM) for the S-II.

10.4.3.1.3 (Continued)

Also included in the simulator are orbital navigation and second restart and burn capabilities. The vehicle was considered as a rigid body in the "6-D", thus eliminating the effects of bending. Slosh dynamics were not implemented in the simulation.

The ascent phase was simulated utilizing "time-tilt" guidance in the S-IC stage and two-stage IGM guidance in the second stage. The "abort-to-orbit," Reference 10.4-46, guidance was utilized for the S-II stage to facilitate entrance into the high-speed-cutoff loop and to take advantage of the "time-to-go" calculations that are based on the velocity required for achieving the desired orbit. This modification may be accomplished through guidance constants and involves no impact on the flight software major loop.

Simulations have been made of the J-2S second S-II burn to circularize at a 300 nautical mile altitude. The initial simulations used the present Saturn V "abort-to-orbit" logic with the AS 503 C' control filter and actuator model since the data was readily available. A constant idle mode thrust and mass rate was used to investigate the capability of the simulated accelerometer package to sense the small level of acceleration.

The resulting acceleration is shown relative to time in Figure 10.4.3.1-1. The reciprocal of this acceleration is input to the M/F filter and the output, shown in Figure 10.4.3.1-2, is input to the guidance equations. The resulting transient in the commanded pitch attitude can be seen in Figure 10.4.3.1-3. The accelerometer quantization level of .05 m/s constitutes a large percentage of the approximately .68 m/s^2 acceleration during idle mode thrust.

It is readily seen from Figure 10.4.3.1-3 that the M/F filter response can adversely effect the guidance equations and that the AS 503 C' filter used in this case is unsatisfactory. A filter with a slower response, such as the AS 504 E (Reference 10.4-47), should be used to smooth the inverse acceleration input to the guidance equations. Figures 10.4.3.1-4 and 10.4.3.1-5 illustrate the resulting inverse acceleration and commanded attitude when the 504 E filter was used. An alternative method would be to replace the present method with a linear regressive curve. This technique would ensure smooth input to the guidance equations. This method, however, is dependent on predicting the thrust level during idle mode and the change in M/F more accurately than is normally possible. There would be no impact on the flight program if the M/F filter was replaced, while minor modifications would be necessary to implement the alternative method.

In the guidance equations that correct for altitude deviations (Section 3, Reference 10.4-46), gravity becomes the dominant term for the idle mode circularization. This results in a commanded attitude that places the vehicle almost along the radius vector. The attitude correction can be bypassed by operating in a "chi-tilde"

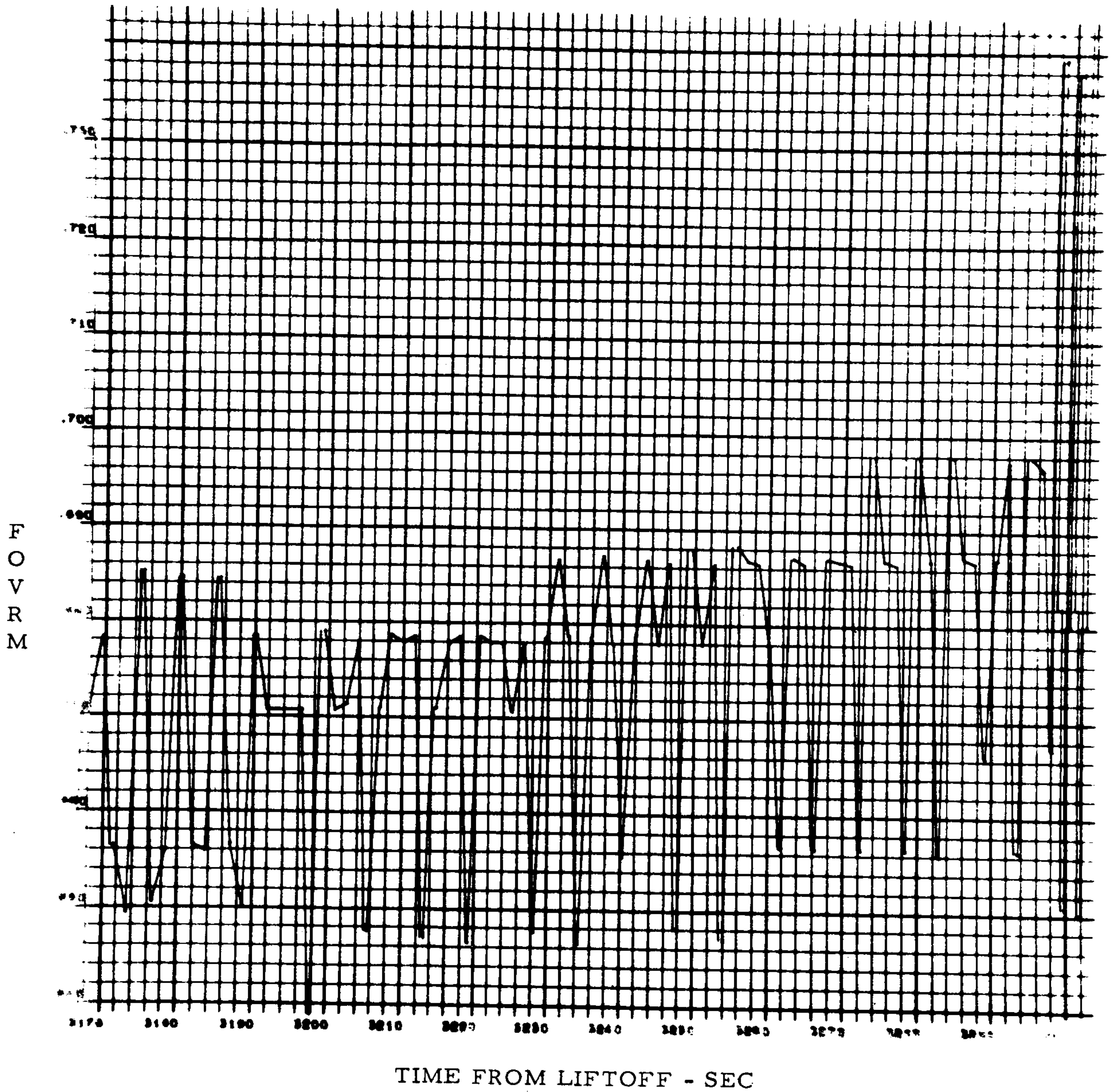


FIGURE 10.4.3.1 - 1. SENSED IDLE MODE ACCELERATION

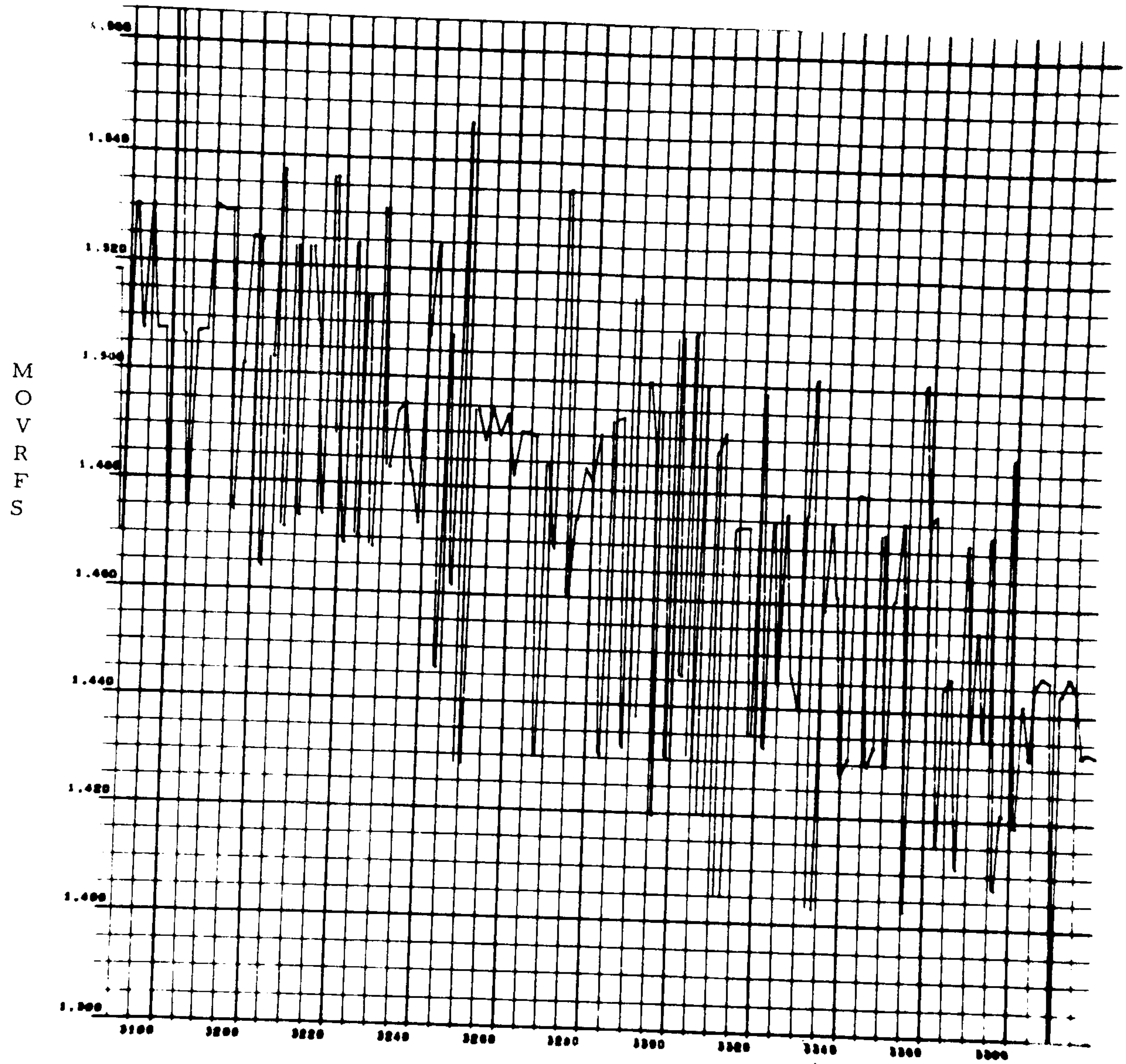


FIGURE 10.4.3.1-2. SMOOTHED INVERSE ACCELERATION (503 C' M/F FILTER)

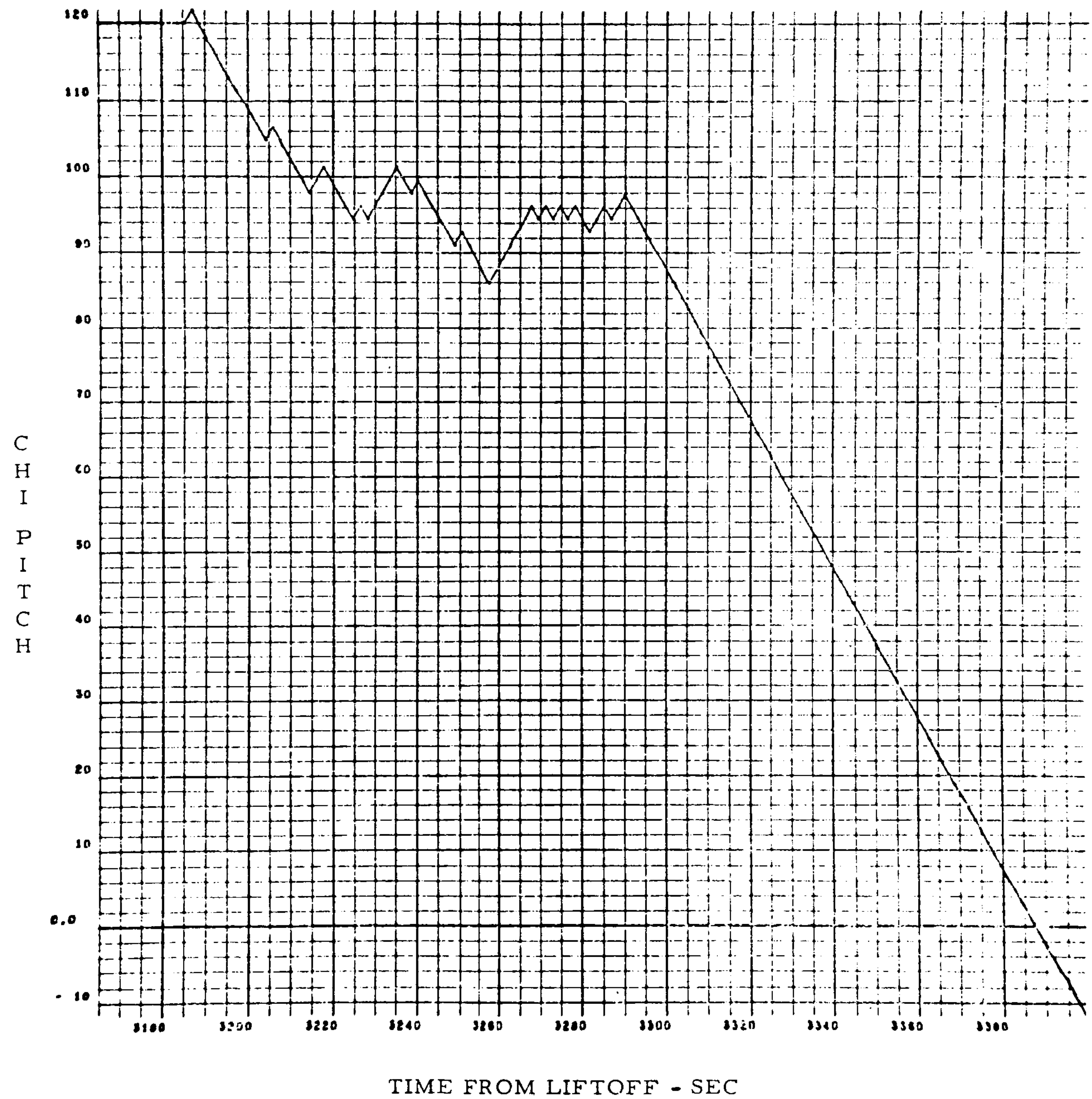


FIGURE 10.4.3.1 - 3. COMMANDED PITCH ATTITUDE (USING 503 C' M/F FILTER)

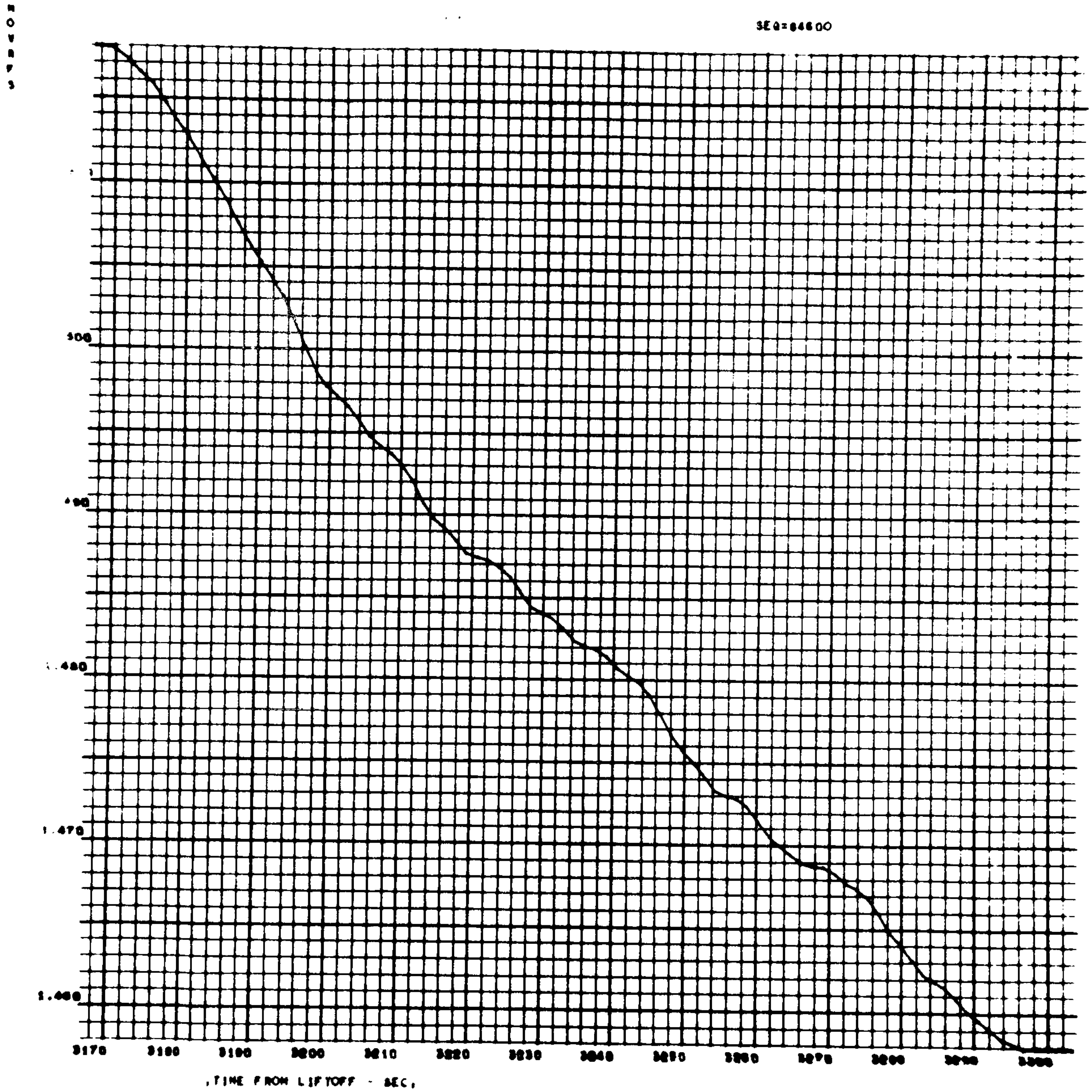


FIGURE 10.4.3.1 - 4. SMOOTHED INVERSE ACCELERATION (USING 504 E M/F FILTER)

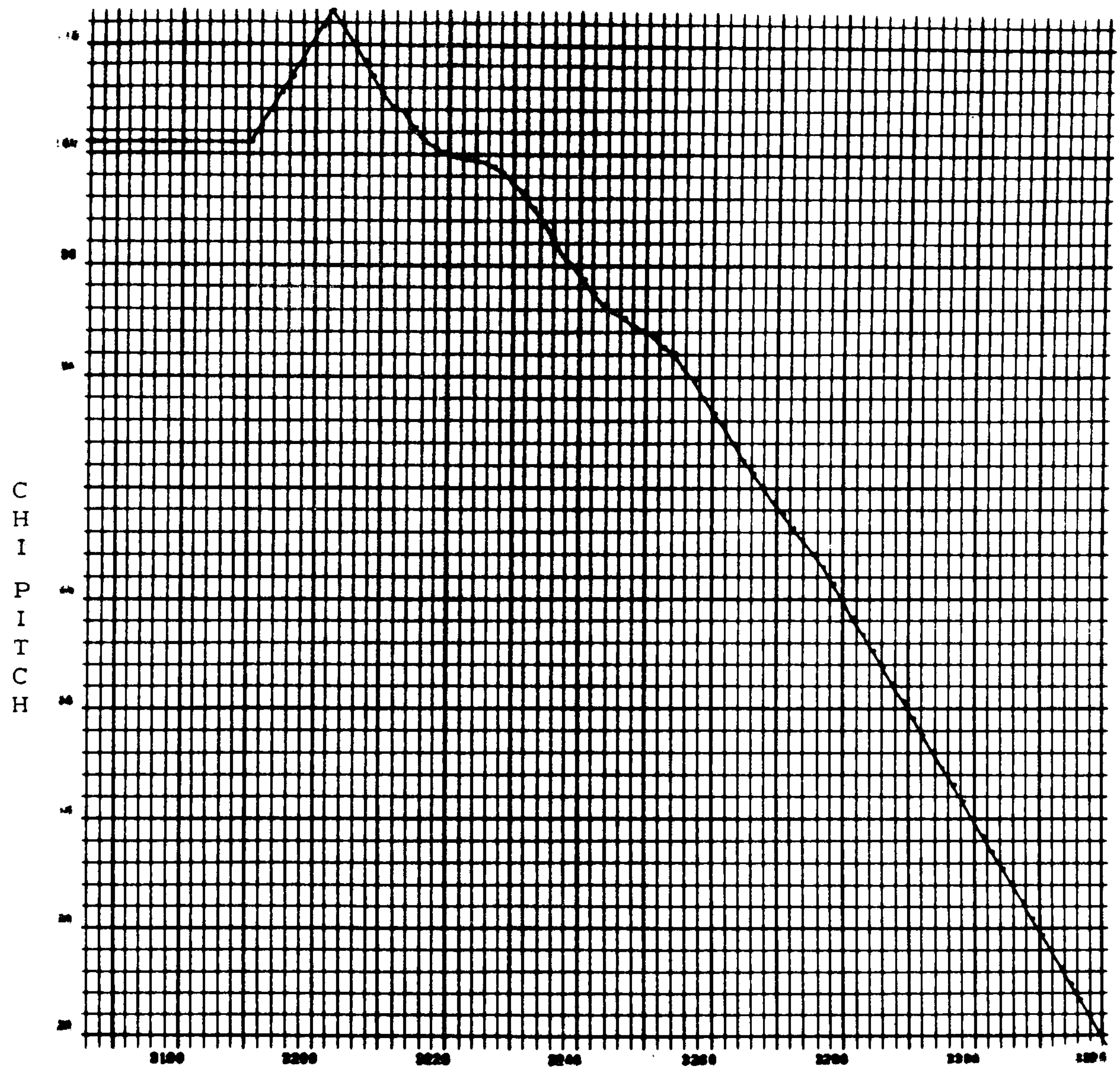


FIGURE 10.4.3.1 - 5. COMMANDED PITCH ATTITUDE (USING "504E" M/F FILTER)

10.4.3.1.3 (Continued)

(Reference 10.4-46) mode which assumes the desired altitude has been attained, as in this case, and velocity is the prime concern. This can be accomplished by adjusting a presetting. The commanded pitch attitude using "chi-tilde" guidance is illustrated in Figure 10.4.3.1-6. Figure 10.4.3.1-7 is a commanded pitch attitude from a simulation that had a limit on the altitude correction terms. When this limit was reached, guidance automatically switched to a "chi-tilde" mode. Table 10.4.3.1-I compares the terminal conditions of the techniques discussed above. No significant difference was recorded and the automatic "chi-tilde" would require a minor logic change. Therefore, the presetting technique is recommended as there would be no impact on the flight program except scaling of inverse acceleration. Table 10.4.3 - II shows the sequence of events for the LEO mission.

10.4.3.1.4 J-2S/Polar Orbit

The Polar Orbitmission profile is direct injection into a 100-nautical mile polar orbit by a two-stage vehicle. One burn of the S-IVB is required. The ground track is over Cuba and Panama by boost turning both stages. The mission was flown to avoid expended stage impact upon land masses.

The present Saturn platform is a three-gimbal platform with gimbal order from inner to outer of pitch, yaw, roll. Complete freedom of movement about pitch and roll exists, but the platform tumbles if the middle gimbal exceeds $\pm 60^o$. Therefore, study efforts were first directed towards testing the adequacy of platform alignment techniques for meeting yaw requirements. The most promising technique was that known as yaw biasing. Rather than align the platform yaw (Z_S) axis down the launch azimuth, the yaw axis is aligned to make an angle, δ , with the launch azimuth. The launch plane is converted from a plane of zero yaw to one of variable yaw. This is the key to the utility of a yaw bias (or roll offset). The standard alignment giving a design limit at most $\pm 45^o$ (Reference 10.4-31), about the boost plane is not adequate for large perturbations. With an appropriate choice of δ, these limits can be transformed and made relative to the boost plane.

Simulations were made using the yaw biasing technique with offset gimbal angles. The yaw biasing technique involves a redefinition of the platform coordinate system. The coordinate system is rotated through an offset δ about the X_s axis, illustrated in Figure 10.4.3.1-8. Thus the YAW (Z_S) axis is offset δ degrees from the firing azimuth (Z_a). This modification does not affect the vehicles trajectory, but merely modifies the platform measure of that trajectory. The gimbal angles θ_p, θ_R, θ_Y define the vehicle orientation in the "S" coordinate system (whose Z_S is offset δ degrees from the launch azimuth). Similarly, α_p, α_Y, α_R define the vehicle orientation in the "A" coordinate system (whose Z_A passes through the launch azimuth). Based on the simulation results, it can be concluded that a yaw bias of 10^o will handle this mission: instead of a maximum yaw attitude of 49.3^o, the platform senses one of 40.3^o. For a yaw bias of 20^o,

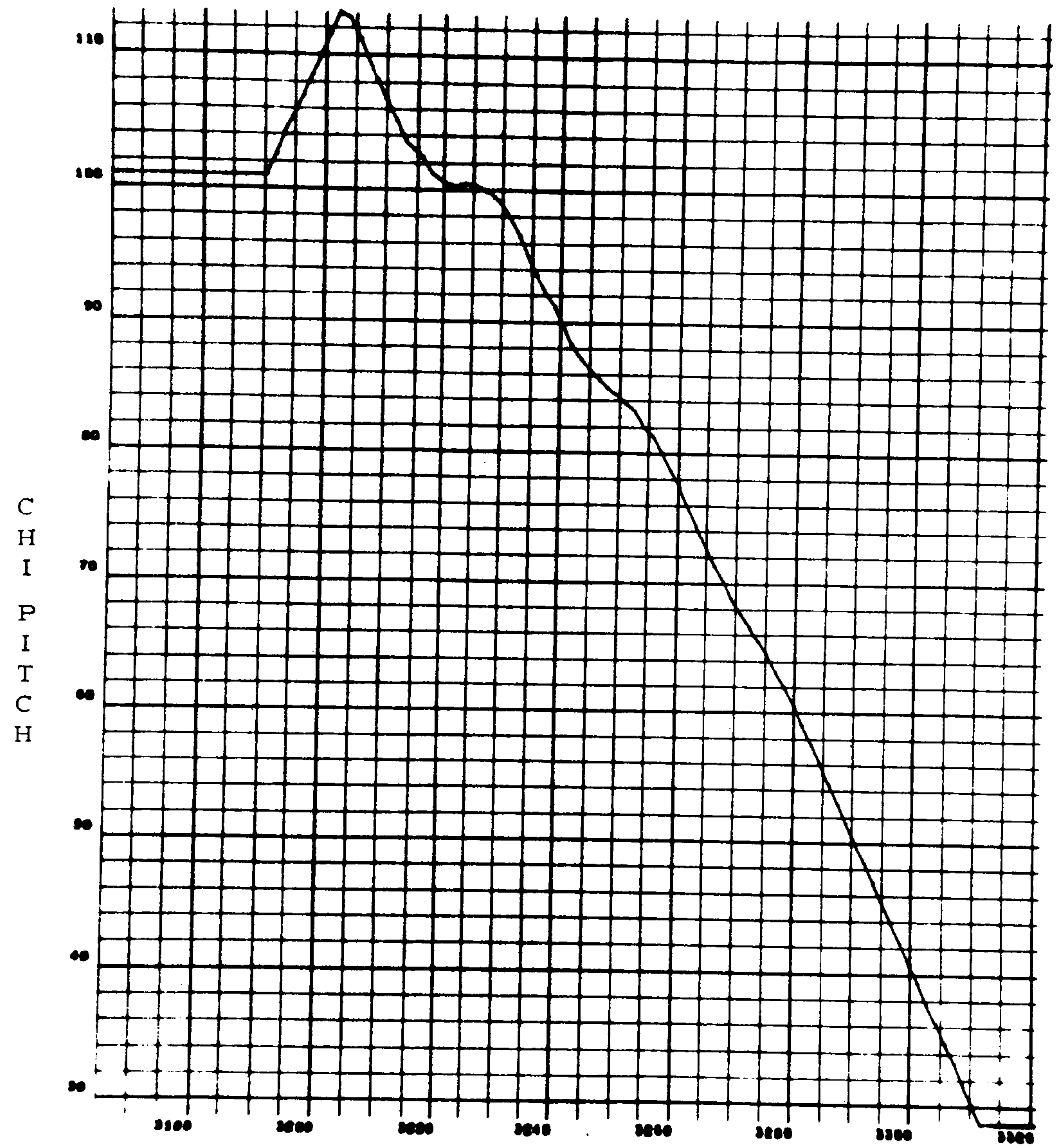


FIGURE 10.4.3.1 - 6. COMMANDED PITCH ATTITUDE USING CHI - TILDE

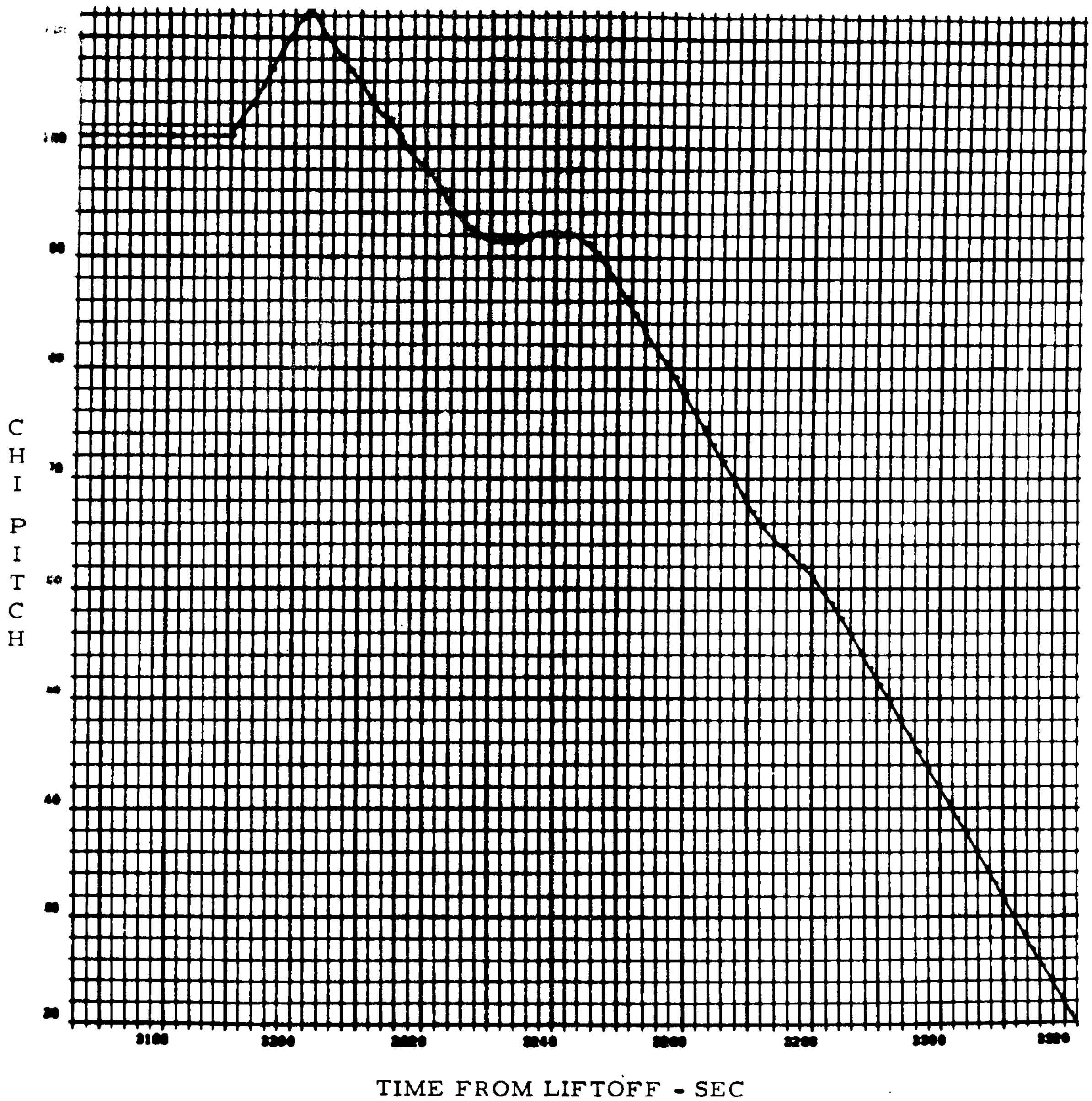


FIGURE 10.4.3.1 - 7. COMMANDED PITCH ATTITUDE USING AUTOMATIC CHI - TILDE

TABLE 10.4.3.1-I. IDLE MODE CIRCULARIZATION TERMINAL CONDITIONS

	Reference Trajectory	"Chi-Tilde" Guidance	Automatic "Chi-Tilde" Guidance	Linear Regressive	504 E M/F Filter
Terminal Radius	6919589. m	6921814	6921589	6921840	6921777
Terminal Velocity	7584.999 m/s	7583.10	7583.09	7583.03	7583.07
Inclination	33.0997 deg	33.1006	33.1008	33.1005	33.1006
Descending Node	57.1402 deg	57.1405	57.1391	57.1406	57.1407
Eccentricity	.00126	.00205	.00209	.00208	.00202

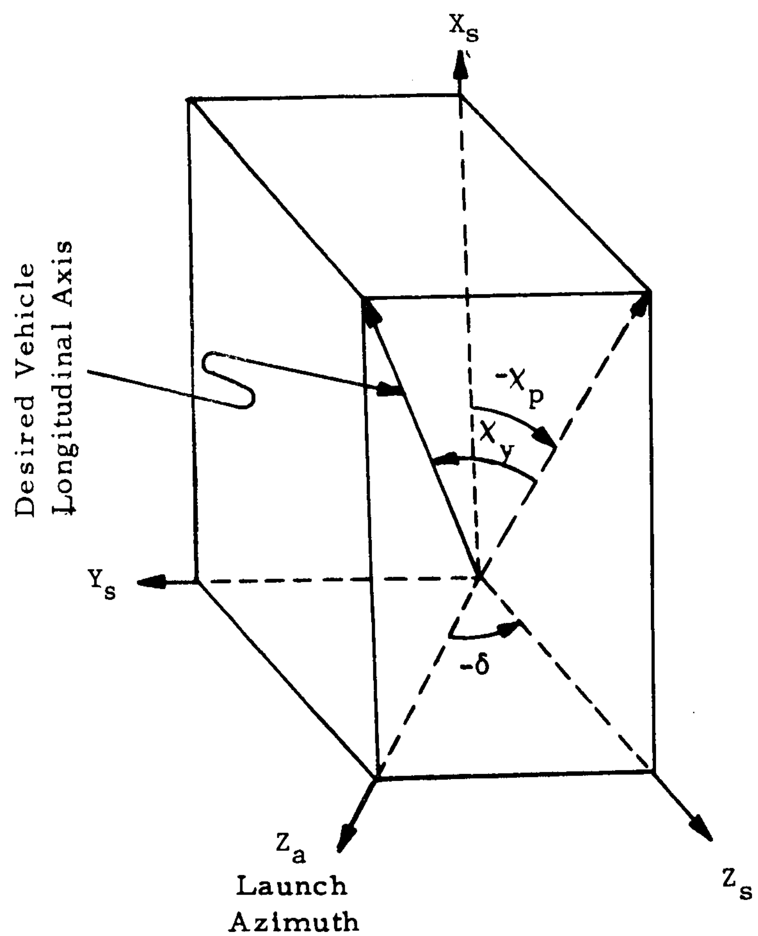


FIGURE 10.4.3.1 - 8. PLATFORM ORIENTATION WITH OFFSET δ

10.4.3.1.4 (Continued)

this is reduced to 31.0 degrees.

The principal advantage of the yaw biasing technique is that no new hardware is required; its principal drawback is the complexity added to mission planning and software development. To minimize the effect of yaw biasing on mission analysis, it may be desirable to incorporate a routine in the flight program which converts "standard" gimbal angle to "offset" gimbal angles. The required routine has been defined and is given in Figure 10.4.3.1-9. It should be used during pre-IGM guidance prior to the $\pm 45^{\circ}$ limiting of yaw in Minor Loop Support. During IGM the use of the proper matrix relating IGM coordinates to platform coordinates automatically includes the offset.

To achieve a polar orbit, a yaw command of approximately 50° at a rate of 1°/sec is required during S-IC guidance after passing the region of maximum dynamic pressure. This may be handled by initiating the yaw guidance command computation at a preset time in Time Base 1 as

CHIY = CHIY + YAWRT (DTNOM)

where CHIY = 0, initially
YAWRT = 1°/sec
DTNOM = nominal computation cycle length

This may be incorporated into the time tilt logic.

Two-stage IGM guidance is needed to guide the S-II into the desired orbit. In order to minimize the flight program impact, it is desirable to use abort-to-orbit two-stage IGM guidance (Reference 10.4-46). In order to use abort-to-orbit IGM in Time Base 3, it is necessary to set certain flags in IGM and define the presettings for this logic properly.

The "6-D" simulation of the polar mission was primarily used to verify the adequacy of abort-to-orbit IGM and the yaw biasing technique. The simulation was based on a preliminary reference trajectory provided by the Boeing Company. The firing azimuth is 140°. The vehicle inserts into a 100 nautical mile circular orbit with a descending node of 2.4270935 degrees. The insertion parameters achieved in the simulation are compared with those of the reference trajectory in Table 10.4.3.1-II. The time table of the sequence of events is given in Table 10.4.3-III.

10.4.3.2 Boost Minor Loop

The minor loop is initiated by an interrupt approximately every 40 milliseconds during boost flight and requires approximately 21 milliseconds to complete the cycle.

Δ = preset yaw bias

PLTOST = ≠0, use yaw bias;
= 0, don't use yaw bias

χ_p = pitch guidance command

χ_y = yaw guidance command

χ_R = roll guidance command

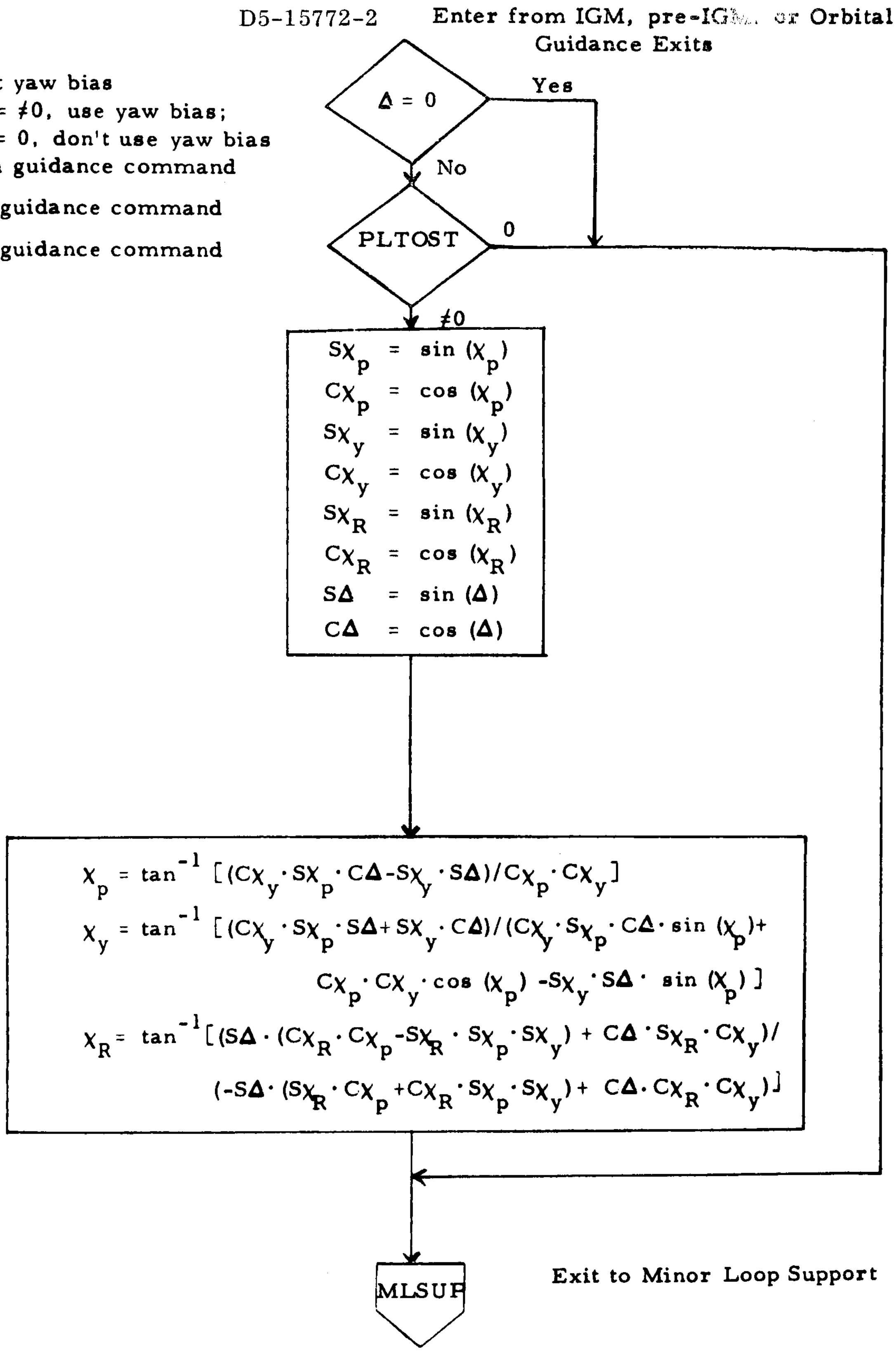


FIGURE 10.4.3.1 - 9. CALCULATION OF OFFSET GIMBAL ANGLES

TABLE 10.4.3.1-II. POLAR ORBIT INSERTION CONDITIONS

Insertion Conditions	Units	Reference Trajectory Data	6-D Results	Deviation
Burn Time	sec	497.220	497.225	.005
Radius	m	6562066.9	6561956.0	110.9
Velocity	m/sec	7796.8860	7796.8164	.0696
Inclination	deg	89.998175	89.996902	.001273
Descending Node	deg	2.4270935	2.4280844	.0009909
Path Angle	deg	.00099461	-.00319987	.00419448
Eccentricity	N/D	.00079255	.00075983	.00003272

10.4.3.2 (Continued)

Launch vehicle attitudes in the form of angular displacements measured by the ST-124M platform gimbal resolvers are compared with the desired vehicle attitude as determined by the guidance computations (Reference 10.4-31). The desired vehicle attitude angles are recomputed each minor loop by adding a fixed increment to the previously desired attitude.

The boost minor loop was evaluated for the four missions being considered and no impact to the flight software minor loop was found.

10.4.3.3 Orbital Program

During the orbital coast periods, the LVDC will perform seven functions: navigation, guidance, attitude control, event sequencing, data management, ground command processing, and data compression. Navigation, guidance and attitude control are carried out on a regular basis under control of the minor loop interrupt. Orbital navigation is performed every 8 seconds and orbital guidance is performed every second. Event sequencing is carried out by the switch selector processing routine (Reference 10.4-46) in response to the switch selector interrupt. Ground commands are processed in response to the command decoder interrupt which has an interrupt priority below that of the minor loop and the switch selector. During orbital flight when the vehicle is not over a ground station, the data compression executive program (Reference 10.4-32) provides the framework within which the interrupt controlled orbital flight program operates. Estimate time line for the J-2S/LOR Mission is given in Table 10.4.3-1.

10.4.3.3.1 J-2S/LOR Orbital Program

The LVDC Equation Defining Document for the AS 505 Flight Program (Reference 10.4-46) was used as a reference in evaluating the orbital program of the LOR mission. The orbital program that is defined in this reference was the latest documented scheme at the time of this evaluation and was designed for a lunar trajectory and no impact resulted in the LOR. When the time from GRR becomes equal to or greater than a time presetting, the $\underline{S}$ vector, which is a unit vector a predetermined number of degrees ahead of the radius vector in the flight plane, and the $\dot{\underline{S}}$ vector, a vector 90 degrees ahead of the $\underline{S}$ vector in the flight plane, are calculated. When the dot product of the $\underline{S}$ vector and the target vector $\underline{T}$, becomes negative, the $\underline{S}$ vector is nearly colinear with the target vector and the $S \cdot T$ calculations begin. This logic ensures that the $\underline{S}$ vector will be inside of the cone, defined by rotating the nodal vector about the target vector, when the $S \cdot T$ calculations are begun. When the cosine of the angle between the $\underline{S}$ and $\underline{T}$ vectors is equal to or less than the cosine of the angle between the nodal vector and the target vector, the Time Base 6 is begun. The $\underline{S}$ vector is fixed at this time and stored for later use in the end condition calculations.

10.4.3.3.1 (Continued)

The transition from orbital navigation and control to boost mode is made on the basis of predicted time to go until the start of Time Base 6. For the J-2S/LOR mission, Time Base 6 is defined as the estimated time for initiating idle mode thrust. A fixed attitude (local horizontal) guidance scheme is used until the full idle mode thrust level is attained. No modifications are necessary to the LOR orbital program other than changing presettings. Each computation cycle during Time Base 5, time to go to the beginning of Time Base 6 is calculated based on the period of the parking orbit, the position of the target vector and the vehicle radius vector in the flight plane, and the number of revolutions to be traversed before the start of Time Base 6. When time to go is less than 9 seconds, a flag is set which causes the program to return to the boost mode of operation, (boost navigation , 25 minor loops per second) after the next orbital navigation cycle. These targeting constants which vary from mission to mission constitute the only impact on the LOR mission. The estimated time line for the baseline and J-2S/LOR missions are given in Table 10.4.3-I.

10.4.3.3.2 J-2S/Synchronous Orbital Program

The orbital program for a Synchronous Orbit. Reference 10.4-39, does not need any modifications when utilizing J-2S engines during boost phases. An advantage, however, of using the J-2S engine characteristics is the elimination of a previous constraint of 0.5 hours minimum time for chilldown between parking orbit insertion and restart. This allows a restart at the first descending node and access to certain hover points with an increased payload. This is graphically represented in Figure 10.4.3.3-1.

10.4.3.3.3 J-2S/LEO Orbital Program

The AS 505 orbital program was utilized as a baseline program for the LEO mission. Navigation, guidance, attitude control, event sequencing, data management, ground command processing, and data compression as defined in Reference 10.4-46 would be adequate for the LEO mission. There are , however, modifications necessary to the targeting equations to allow the normal calculations of the restart state and estimation of the terminal targeting for a J-2S/LEO mission. Preparations for restart begin when the vehicle position satisfies a geometric relationship with the target vector for the selected restart opportunity. In a lunar mission, the target vector is a unit vector which lies in the earth-moon plane, at an angle σ from the vector through perigee of the target orbit. The target vector is defined by its right ascension and declination, which vary with launch time and are fixed at lift-off. This type of relationship is meaningless for a LEO type mission. The target vector for the LEO mission would be defined as a unit vector through apogee and the negative target vector (aim vector) would be through the perigee. An eccentricity vector that has a direction colinear with perigee and a

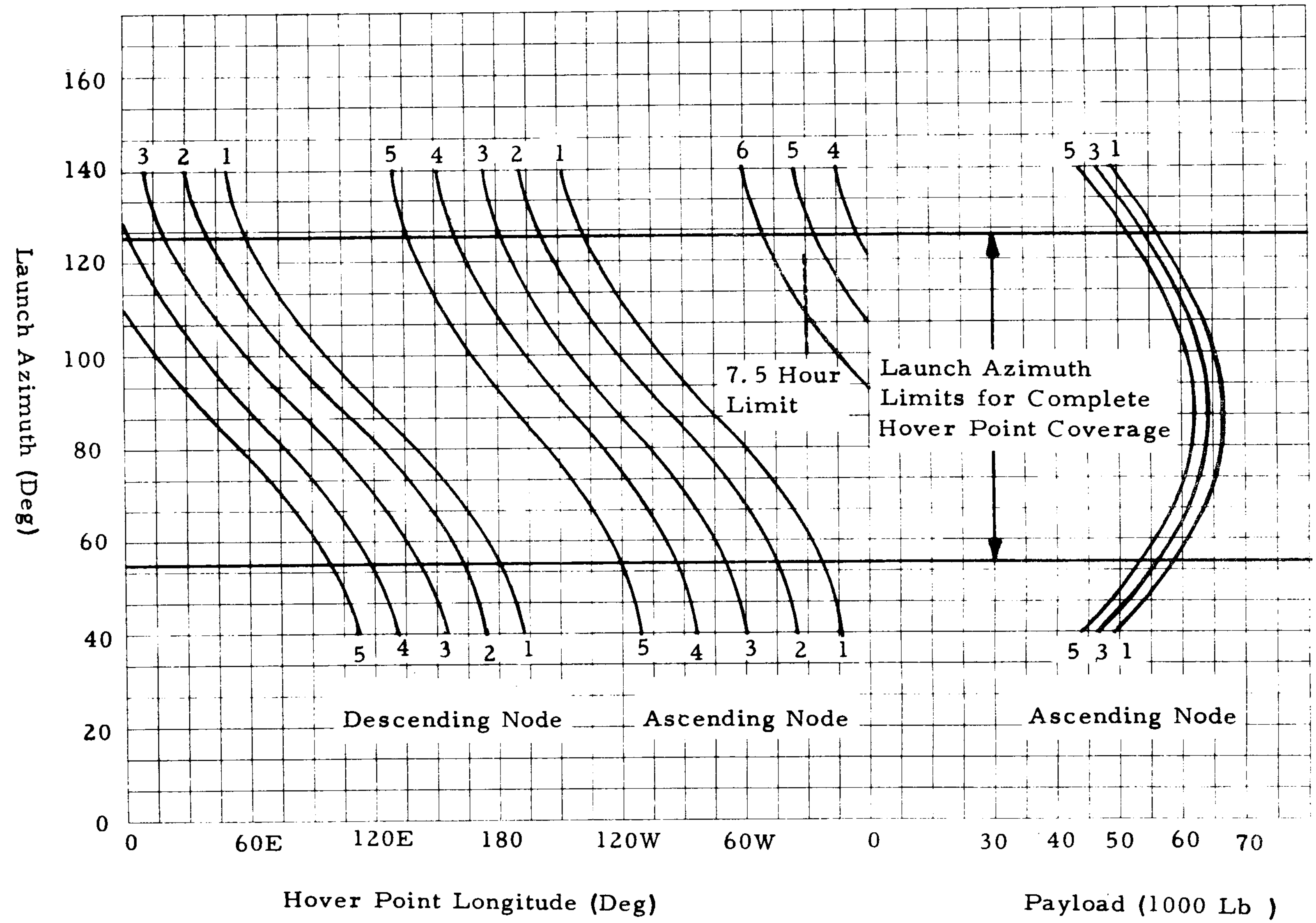


FIGURE 10.4.3.3 - 1. SYNCHRONOUS ORBIT CAPABILITY STUDY EQUATORIAL SYNCHRONOUS MISSION PERFORMANCE CAPABILITY

10.4.3.3.3 (Continued)

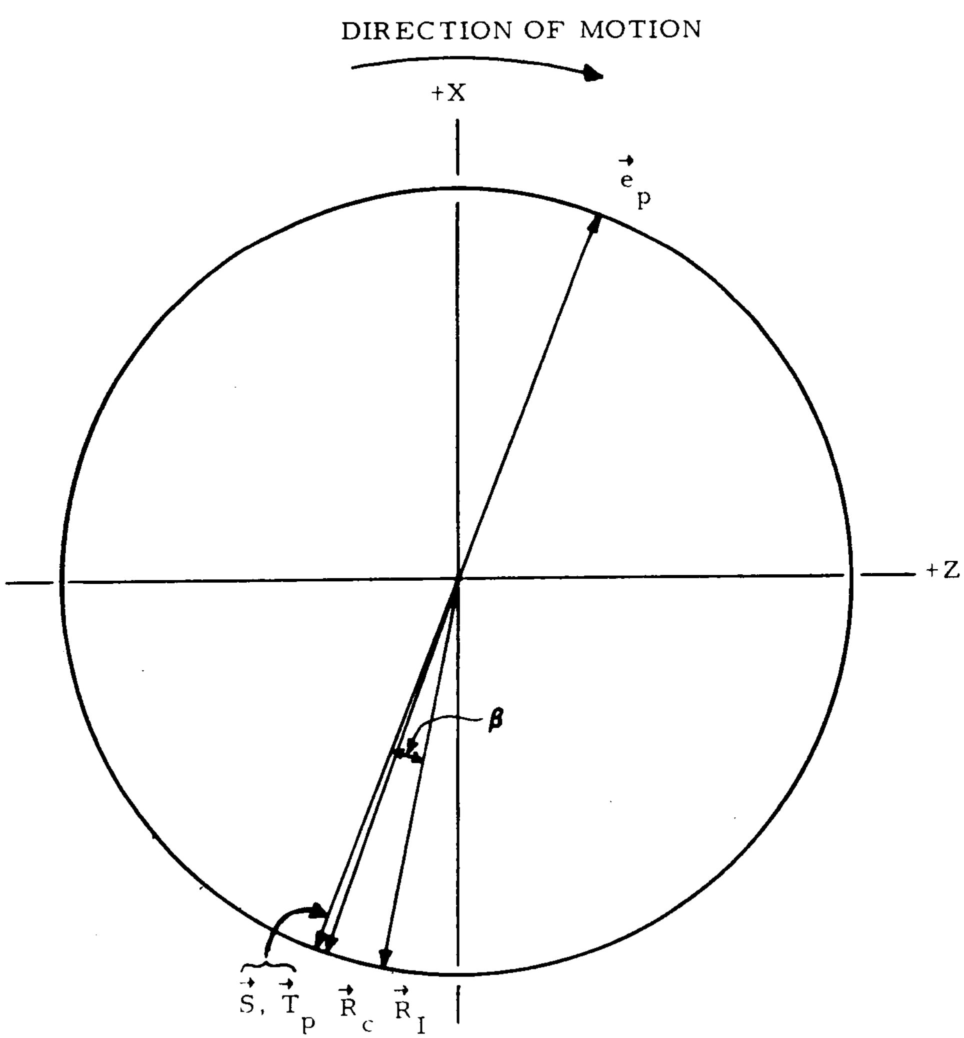
magnitude equal to the eccentricity of the transfer orbit can be used as the aim vector. This vector is derived in Reference 10.4-48 and could easily be computed onboard. The unitized negative $\underline{e}$ would constitute a target vector colinear with the line of apsides in the direction of apogee. This new target vector, $\underline{T}$, would then be utilized in the targeting equations. Figure 10.4.3.3-2 illustrates a planar relationship of the above described vectors and the radius vectors at ignition and cutoff as given in the reference trajectory. The targeting constants would necessarily be changed, but these constants are usually changed from mission to mission and would not impact the orbital flight program. The modifications to the targeting equations as discussed above were defined for implementation in the flight program as shown in Figure 10.4.3.3-3. The logic can easily be inserted in the flow illustrated in Reference 10.4-46.

10.4.3.3.4 J-2S/Polar Mission Orbital Program

Telemetry station acquisitions and losses are determined as a function of vehicle position and equations onboard determine whether the vehicle is within range of a station. In the Polar mission, station acquisition will largely be determined by ships at sea instead of the present tracking sites. The ground trace of the Polar mission is assumed to cross minimum land masses and will be out of range of most of the present tracking sites. The preliminary mission design did not include the orbital phase of the Polar trajectory. No modifications are expected, however, to the orbital program and techniques described in the previous sections. Changing targeting constants, described in paragraph 10.4.3.3.1, is the only forseeable impact on the orbital flight program for a Polar mission.

10.4.3.4 Program Verification

The present flight program verification utilizes two different six-degree-of-freedom simulations: a System/360 digital simulation (6-D), and a Simulation Laboratory simulator. The System/360 simulator is a bit-by-bit simulation of the launch vehicle digital computer/data adapter (LVDC/LVDA) and a detailed simulation of the rigid body vehicle dynamics. Some important features are run-to-run repeatability; program trace capability; extensive capability for perturbing input data to the LVDC/LVDA; and automatic checking of such items as scaling overflows, telemetry timing, and switch selector timing. The Simulation Laboratory simulator consists of a flight-type LVDC/LVDA, a flight-type switch selector, a simplified vehicle model on the GE-235 digital computer, an IBM 1800 telemetry recording/reduction computer, a platform simulator, and associated interfacing, exercising, and controlling equipment. Some important features are use of flight-type hardware, use of telemetry output for analysis, and real-time non-repeatable operation.



$\vec{e}_p$	Vector whose direction is along perigee and magnitude is eccentricity
$\vec{R}_I$	Radius vector at ignition Reference 10.4-10
$\vec{R}_c$	Radius vector at cutoff Reference 10.4-10
$\vec{T}_p$	Target vector
$\vec{S}$	Nodal vector
β	Angle between psuedo nodal vector and radius vector

FIGURE 10.4.3.3 - 2. PLANAR RELATIONSHIP OF LEO TARGETING

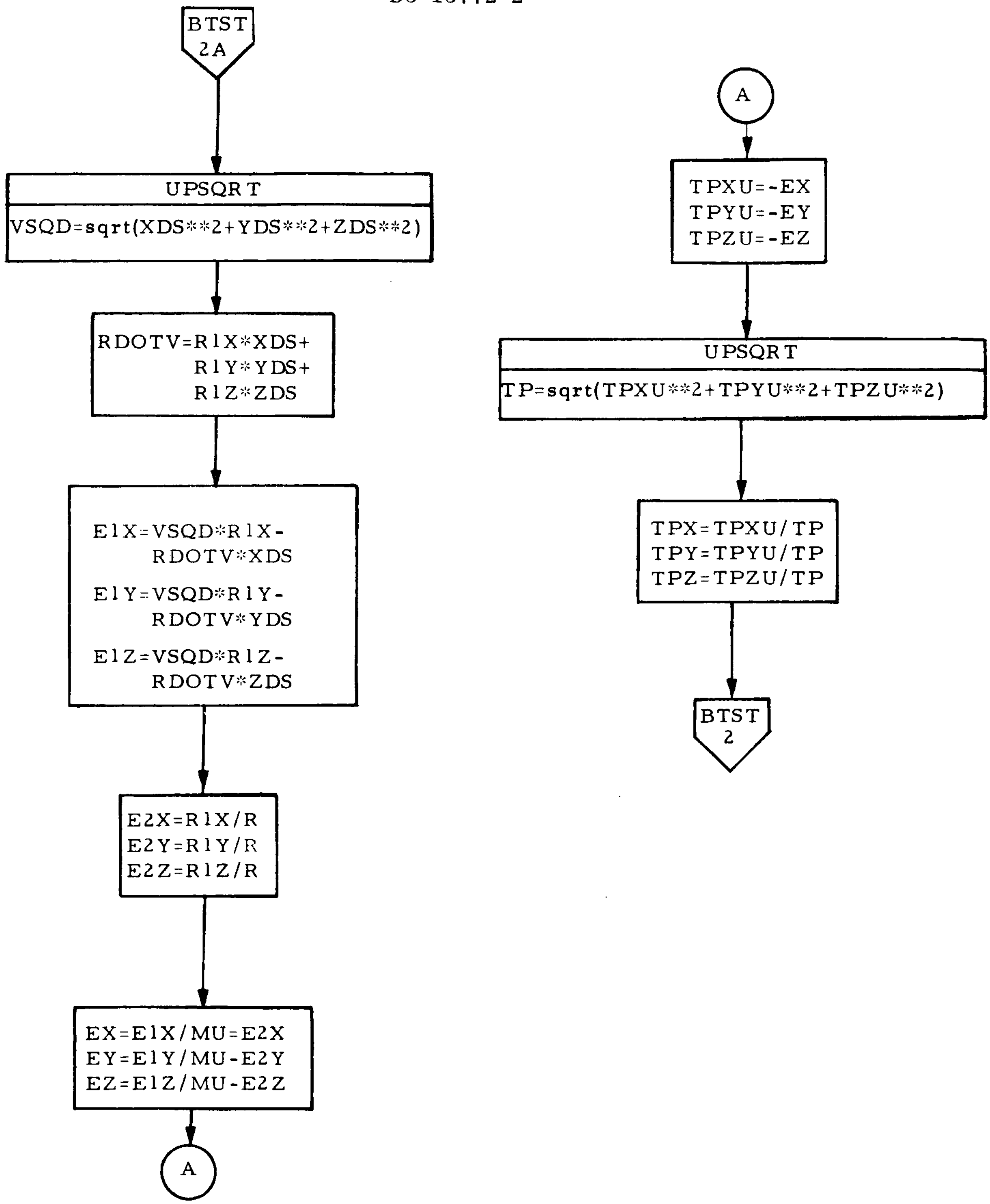


FIGURE 10.4.3.3-3. FLIGHT PROGRAM TARGETING FOR LEO

10.4.3.4 (Continued)

The navigation portion of the flight program determines the position, velocity, and acceleration of the vehicle. Accurate navigation is necessary for proper functioning of all guidance after first stage. The accuracy of the navigation is checked by comparing the flight program navigation quantities to the 6-D navigation quantities.

The accelerometer processing section of the program reads the duplexed accelerometer pulse counters in the LVDA and differences these readings from those of the previous computation cycle to determine the velocity change. Each increment is then processed through the accelerometer reasonableness tests to determine if the change is unreasonably high or low. Any reasonable increment is added into the previous accumulation of velocity increments for use in the boost navigation. Any unreasonable increment is ignored, and replaced by an increment based on a prestored thrust and mass profile. Additionally, the length of the computation cycle and the time since GRR are computed in this program section.

During the mission, nominal operation of accelerometer processing is verified through comparing the velocity accumulation of the LVDC to that of the 6-D. Detailed operation of the routine is verified by perturbing the accelerometer readings to force all accelerometer channel selection and reasonableness test paths, and tracing the operation of the program. The prestored acceleration profile is checked by forcing continual zero changes in the down-range and vertical accelerometers and by checking the velocity accumulation between the flight program and the 6-D. This perturbation causes the prestored profile $(F/M)_C$ to be used for most of the flight. The parameters of the orbit achieved are also checked to verify the adequacy of the profile.

During certain portions of flight (staging intervals, etc.) the computed F/M is noisy and a preset value is used along with the reasonableness test constants (RTC's). The preset value of F/M, the value of the RTC's, and the performance parameters of the $(F/M)_C$ routines are changed several times throughout flight. The changes in these constants used in accelerometer processings are verified by checking the values of the RTC's and the performance parameters of the $(F/M)_C$ routine once per computation cycle for one full boost period. At each point that any of the values change, the times of the change and the updated values are checked against the desired time and values.

The computation of the time since GRR will be verified on all cases for the mission by a comparison of flight computer and simulator times. The calculation of the length of the computation cycle will be verified indirectly by verifying that boost navigation, which uses this time, is accurate.

During the mission, the accuracy of boost navigation is verified by comparison of LVDC and 6-D position and velocity every computation cycle during boost.

10.4.3.4 (Continued)

Interaction of the guidance and navigation, accuracy of the guidance scheme, and performance of the high-speed loop are verified by comparison of the orbit attained to the preset target orbit.

Orbital navigation accuracy is verified by comparing LVDC and 6-D position and velocity components every flight program navigation pass. The once-per-second position computation is verified indirectly by verification of orbital guidance accuracy. The C-Band switching and telemetry acquisition and loss logic are verified at each change in status by hand computations to determine if the change should have occurred.

The start of TB6 is a function of position components and is verified with hand computations. The TB6 start time is also compared with the time calculated from the time-to-go to restart preparation. Again, as in boost navigation, the same check of accuracy is made.

Yaw and roll guidance is verified beginning at GRR. The nominal pitch guidance commands are checked by comparing the commanded attitudes to ideal values obtained by evaluating the time tilt polynomial to verify that an error no larger than 0.1° exists between them.

The orbital guidance maneuvers are verified by checking to see that the specified attitude is commanded and that the maneuvers are executed at the programmed times.

The following paragraphs attempt to distinguish the amount of deviation in program verification each of the evaluated missions require. The baseline requirements are described in Appendix 8, Revision 760, to Contract NAS8-14000.

10.4.3.4.1 J-2S/LOR Program Verification

The requirements of program verification for the LOR mission are adequately covered in the baseline referenced. There would be no impact to the present flight program verification procedures.

10.4.3.4.2 J-2S/Synchronous Program Verification

The verification programs must be modified for a Synchronous mission and new programs developed for data reduction. Preparation and planning for flight evaluation will take into consideration the changed hardware and mission lifetime extension and their effects on present data reduction programs and analysis techniques. The 15-hour lifetime of the IU is expected to add approximately 75 percent more computer time and 10-percent more manpower than the requirements referenced in the baseline.

10.4.3.4.3 J-2S/LEO Program Verification

The LEO mission consists of a two-stage ascent to 100 nautical miles, coast to 300 nautical miles and a restart of the S-II stage to circularize the orbit. Moderate modifications to the verification programs will be required for the LEO mission. The verification requirements will, however, be approximately the same as the baseline due to the two-stage configuration ascent phase.

10.4.3.4.4 J-2S/Polar Program Verification

The Polar mission requires modifications to the verification programs similar to the requirements of the Synchronous mission. The extended lifetime of the IU will not be required for this mission and the computer time necessary for evaluating this area will not be needed. Therefore, the manpower requirements are the same as those for the Synchronous and the computer requirements are approximately the same as those stated in the reference.

10.4.3.5 Guidance Analysis

Guidance is defined in Reference 10.4-31 as the computation of maneuvers necessary to achieve the desired end conditions of a trajectory. Control is defined as the execution of maneuvers determined from the guidance scheme. These general definitions were used as a basis for analyzing the performance of the present Saturn V guidance and control techniques; the objective being to evaluate the feasibility of using J-2S engines in various types of missions. Table 10.4.3.5-I summarizes software impact for the missions under considerations.

10.4.3.5.1 J-2S/LOR Guidance

Present software is aligned to a lunar trajectory and no impact will result from using J-2S engines. IGM constants will be changed but they are consistently changed from mission to mission at the present time.

10.4.3.5.2 J-2S/Synchronous Guidance

The objective in this analysis was to evaluate the advantages of the J-2S engine for a Synchronous-type mission. The advantage of releasing the time constraint, discussed in para 10.4.3.3.2 for restart is due to the absence of previoysly required chilldown operations. Synchronous Orbit software, Reference 10.4-39, will be impacted as shown in Table 10.4.3.5-I and discussed in Reference 10.4-39.

TABLE 10.4.3.5-I . FLIGHT PROGRAM IMPACT

PHASE	GUIDANCE	SOFTWARE CHANGE	CONSTANTS CHANGE
LEO First S-II Burn	Present abort-to-orbit	None	Constants
LEO Targeting	None	Moderate	Constants
LEO Idle Mode Guidance	Present abort-to-orbit with active "chi-tilde"	None	Constants
Polar Ascent (S-IC)	Modified	Major	Constants
Polar Ascent (S-II)	Present abort-to-orbit	None	Constants
Synchronous Ascent	*	*	Constants
Synchronous Orbital & Restart	*	*	Constants
LOR Ascent	None	None	Constants
LOR Orbital & Restart	None	None	Constants

*Defined in Reference 10.4-39

10.4.3.5.3 J-2S/LEO Guidance

The ascent phase of the LEO mission was simulated using "time-tilt" guidance in the first stage and a two-stage iterative guidance (abort-to-orbit) in the S-II stage. The insertion conditions are compared to the reference trajectory (Reference 10.4-10) in Table 10.4.3.5-II. No modifications to the guidance equations were required to obtain the commanded pitch attitude illustrated in Figures 10.4.3.5-1 and 10.4.3.5-2. Necessarily, guidance constants were changed to utilize the advantages of the two-stage IGM logic. These advantages include easy entry in the cutoff logic and a more accurate calculation of "time-to-go."

Simulations were made using the recommended targeting technique at various times after the transfer ellipse had been defined to investigate any degradation of the transfer ellipse due to venting, drag, and gravity. Since the target vector is orbit dependent, degradation of the transfer ellipse would alter the target vector. Table 10.4.3.5-III lists the target vector components calculated at various times of flight. The last three calculations compare vent/drag models with the reference trajectory described in Reference 10.4-10. It is readily seen from Table 10.4.3.5-III that some degradation occurs from the original transfer ellipse insertion state, and it is therefore reasonable to calculate the target vector at a later time. The present restart calculations include a time guard presetting to prevent certain parameters being calculated too early. This time guard would be utilized to delay the target vector calculation until the desired time. Table 10.4.3.5-IV compares the reference trajectory restart state to that of a simulation using the $-\underline{e}$ target vector. The program used was a navigation program (HYPERVENT) that included the AS 503 vent/drag model and the modified restart equations. The position and velocity at the termination of the first S-II burn from the reference trajectory was input to the program and the restart state was calculated 2678 seconds later which was very close to the reference trajectory.

Planar simulations of the J-2S circularization study indicated that near optimum thrusting will occur when a thrust tangential to the velocity vector is applied. Therefore, only minimum guidance will be required to fulfill the mission requirements. In these simulations, no IGM guidance was used and thrust was obtained by linear interpolation of the build-up curves previously illustrated (Figure 10.4.3.5-3). The thrust initiation was varied as a function of time from perigee and the thrust direction was along the velocity vector except in one series of runs the attitude was aligned above and below the velocity vector by a constant angle which varied from 0 to +3 degrees. The results of these basic simulations are given in Tables 10.4.3.5-V through 10.4.3.5-VIII and indicate that the reference trajectory is not an optimum burn in the S-II idle mode circularization phase. A more nearly optimum trajectory would be a burn across apogee keeping the vehicle aligned to the flight-path angle. This would result in less fuel expenditure and small guidance manuevers. The results of the simulation aligning the vehicle to the flight-path angle, along the same arc as the reference, are compared to the reference trajectory terminal conditions as illustrated in Table 10.4.3.5-IX.

TABLE 10.4.3.5 - II. LEO TERMINAL CONDITIONS

	Simulation	Reference
Radius (m)	6563270	6559974
Velocity (m/s)	7898.01	7898.02
Eccentricity	.026605	.026596
Flight Path Angle (deg)	-.0039	-.00005
Perigee (m)	6563270	6559974
Apogee (m)	6922045	6918446
Inclination (deg)	33.1392	33.0901
Descending Node (deg)	57.3921	57.3750

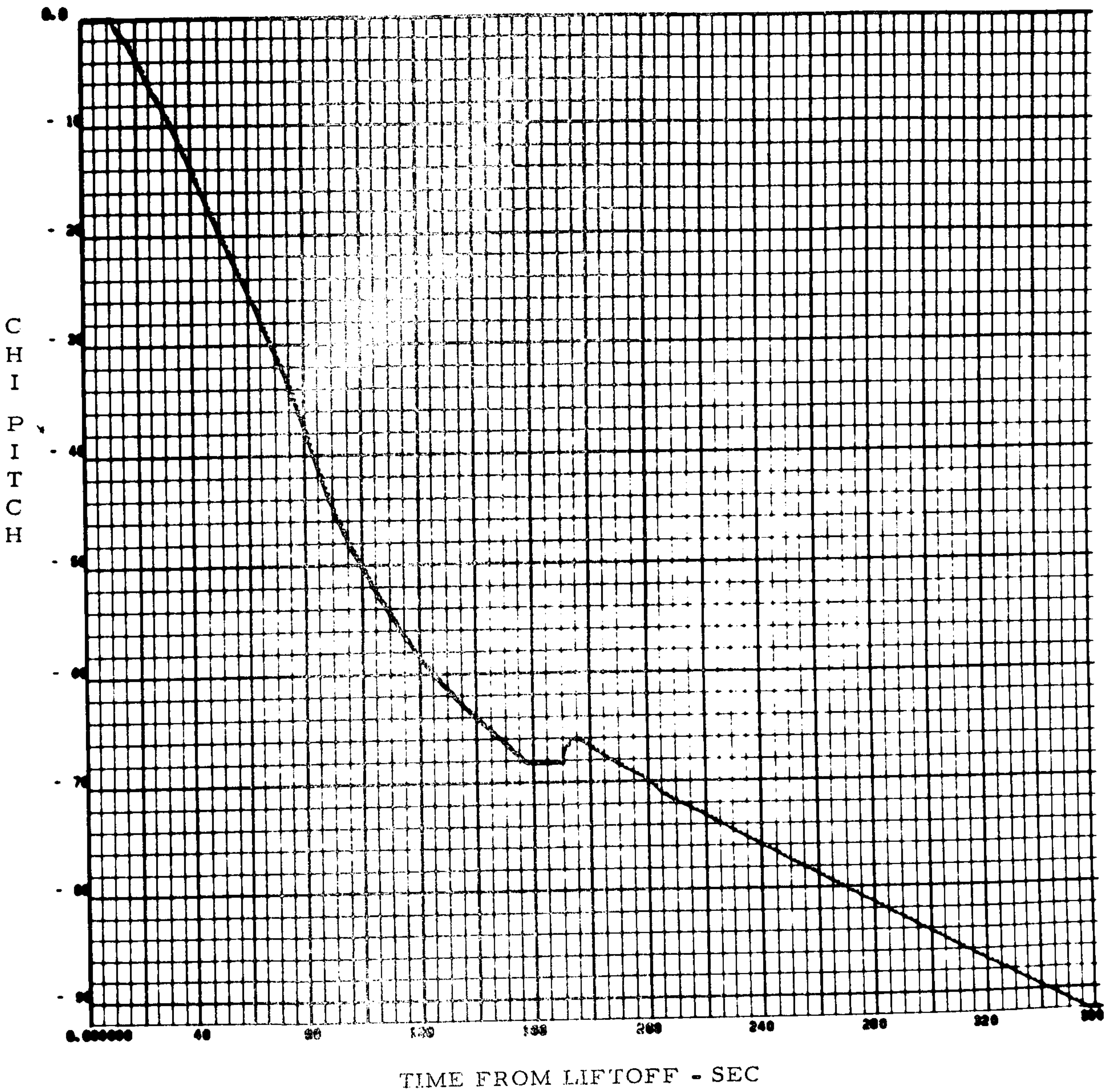


FIGURE 10.4.3.5-1. COMMANDED PITCH ATTITUDE LEO ASCENT (FROM 0 TO 360 SEC)

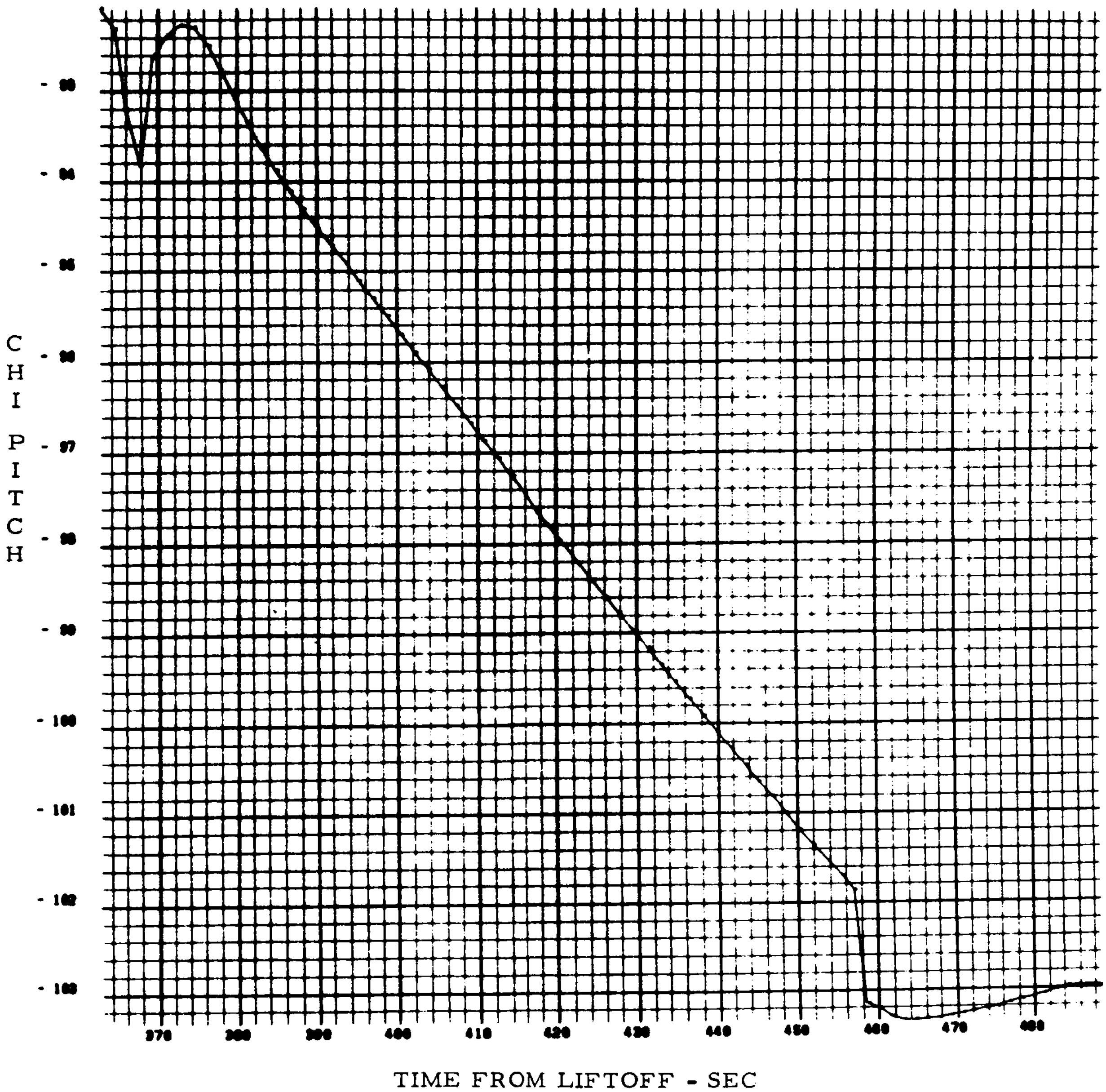


FIGURE 10.4.3.5-2. COMMANDED PITCH ATTITUDE LEO ASCENT (FROM 360 TO 490 SEC)

TABLE 10.4.3.5 - III. VARIATION OF TARGET VECTOR (-e)

TIME	TPX	TPY	TPZ
489	-.26230547	-.96496322	.00646442
697	-.271177418	-.96233840	.00662765
1009	-.29045546	-.95686312	.00696568
2305	-.28342734	-.95897502	.00594097
3165	-.27718179	-.96080326	.00524073
3165*	-.27642584	-.96102073	.00522847
3165**	-.29575303	-.95524869	.00548497
3166***	-.29573089	-.95525587	.00548442

*Simulation results - no venting thrust

**Simulation results - 503 C' vent/drag model

***Simulation results - 503 C' vent/drag model + 10%

TABLE 10.4.3.5 - IV. RESTART STATE

	Reference Trajectory	$(-\vec{e})$ Hypervent	H-R Difference
Calculated restart time (sec)	3165	3164	-1 sec
X (m)	-1292364.6	-1307321.7	-4957.1
Y (m)	-6796459.5	-6781667.4	14792.1
Z (m)	27964.5	28126.5	162.0
R (m)	6906600.58	6906583.46	-17.12
$\dot{X}$ (m/s)	-7362.704	-7368.389	-5.685
$\dot{Y}$ (m/s)	1381.592	1403.235	21.643
$\dot{Z}$ (m/s)	98.220	98.316	.096
V (m/s)	7501.4436	7501.4595	.0159
Inclination (deg)	33.08767	33.08758	-.00009
Descending Node (deg)	57.16318	57.16334	.00016
Eccentricity	.0250800	.02508092	.0000009
Flight Path Angle	.13034933	.13198541	.00163608
True Anomaly (deg)	174.92586	174.86216	-.06370
Radius at Perigee (m)	6569303.76	6569292.09	-11.67
Radius at Apogee (m)	6907296.87	6907297.35	.48
Energy	-59103259.2	-59154669.9	51410.7

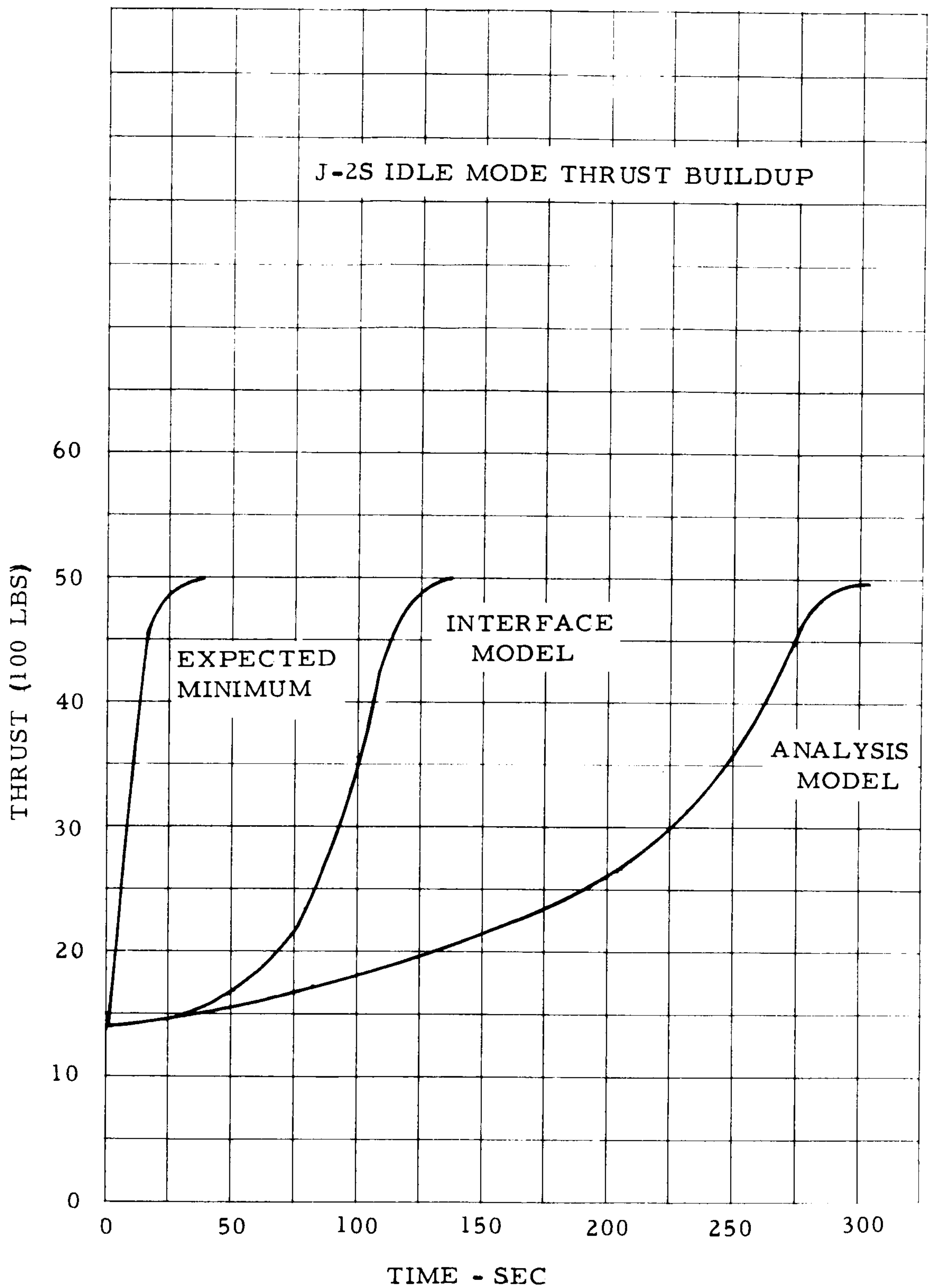


FIGURE 10.4.3.5-3. J-2S IDLE MODE THRUST BUILDUP

TABLE 10.4.3.5-V. THRUST TANGENT TO VELOCITY VECTOR CONSTANT THRUST 5000#/ENGINE

THRUST INITIATION			Burn Time	TERMINAL CONDITIONS				
Time Before Apogee	Time From Perigee	Angle From Perigee		Eccentricity	Apogee	Perigee	Angle From Perigee	Radius
140.0	2613.0	171.308	152.4	.00189627	6933.40	6907.16	180.856	6919.32
120.0	2633.0	172.549	152.7	.00130508	6929.06	6910.99	182.084	6919.21
100.0	2653.0	173.790	152.4	.00071467	6924.59	6914.70	183.306	6919.02
80.0	2673.0	175.030	152.0	.00013269	6920.06	6918.23	184.521	6918.74
70.0	2683.0	175.651	151.7	.00017872	6920.01	6917.54	185.123	6918.56
60.0	2693.0	176.271	151.4	.00047059	6921.66	6915.14	185.724	6918.37

TABLE 10.4.3.5-VI. THRUST TANGENT TO VELOCITY VECTOR 12 SEC BUILD UP TO 5000#/ENGINE

THRUST INITIATION			Burn Time	TERMINAL CONDITIONS				
Time Before Apogee	Time From Perigee	Angle From Perigee		Eccentricity	Apogee	Perigee	Angle From Perigee	Radius
110.0	2643.0	173.169	156.4	.00088126	6925.87	6913.67	182.965	6919.08
100.0	2653.0	173.790	156.7	.00058583	6923.57	6915.46	183.572	6918.96
90.0	2663.0	174.410	156.5	.00029232	6921.29	6917.25	184.180	6918.82
80.0	2673.0	175.030	156.2	.00003458	6919.13	6918.65	184.782	6918.66
70.0	2683.0	175.651	155.9	.00030437	6920.63	6916.41	185.383	6918.48

TABLE 10.4.3.5 - VII. 250 SEC BUILD UP TO 5000#/ENGINE

THRUST INITIATION			Burn Time	TERMINAL CONDITIONS				
Time Before Apogee	Time From Perigee	Angle From Perigee		Eccentricity	Apogee	Perigee	Angle From Perigee	Radius
* 220	2533	---	302.6	.00101762	6928.66	6914.57	185.211	6920.01
200	2553	---	302.0	.00044884	6923.97	6917.76	186.414	6919.60
180	2573	---	301.2	.00020144	6921.26	6918.47	187.605	6919.11
160	2593	---	300.3	.00074763	6923.92	6913.57	188.790	6918.54
140	2613	---	299.2	.00132492	6926.54	6908.21	189.962	6917.88
** 220	2533	---	302.7	.00105194	6929.02	6914.46	185.218	6920.06
200	2553	---	302.0	.00046941	6924.11	6917.61	186.414	6919.64
180	2573	---	301.2	.00018538	6921.15	6918.58	187.605	6919.14
160	2593	---	300.3	.00073868	6923.86	6913.63	188.789	6918.56

*Circum Perential
**Tangential to Velocity Vector

TABLE 10.4.3.5-VIII. THRUST VECTOR ABOVE AND BELOW VELOCITY VECTOR 12 SEC BUILD UP TO 5000#/ENGINE

THRUST INITIATION				TERMINAL CONDITIONS				
Time Before Apogee	Time From Perigee	Angle From Perigee	Burn Time	Eccentricity	Apogee	Perigee	Angle From Perigee	Radius
80 (+3°)	2673.0	175.030	155.8	.00071484	6923.09	6913.20	184.757	6918.26
80 (+2°)	2673.0	175.030	155.9	.00047946	6921.73	6915.09	184.763	6918.40
80 (+1°)	2673.0	175.030	156.0	.00024477	6920.31	6916.92	184.769	6918.53
80 (0°)	2673.0	175.030	156.2	.00003458	6919.13	6918.65	184.782	6918.66
80 (-1°)	2673.0	175.030	156.5	.00023394	6920.86	6917.62	184.800	6918.80
80 (-2°)	2673.0	175.030	156.8	.00046806	6922.76	6916.29	184.819	6918.93
80 (-3°)	2673.0	175.030	157.1	.00070237	6924.62	6914.90	184.838	6919.07

TABLE 10.4.3.5 - IX. FLIGHT-PATH ANGLE GUIDANCE TERMINAL CONDITIONS

	Simulation	Reference
Radius (m)	6919380	6919589
Velocity (m/s)	7584.289	7584.999
Inclination (deg)	33.09346	33.09974
Descending Node (deg)	57.15370	57.14019
Eccentricity	.0015003	.0012609
Perigee (m)	6898810	6902166
Apogee (m)	6919540	6919595
Flight Path Angle (deg)	.01505	.00262

10.4.3.5.3 (Continued)

There are several alternative methods for guidance as discussed in paragraph 10.4.3.1.3 and above. The impact on the flight software would obviously depend on the scheme chosen. With the assumption that minimum modifications are desirable, the recommended technique would be to utilize the "chi-tilde" guidance and a mission designed inverse acceleration filter. This would result in no impact on the flight software and reasonable terminal conditions could be achieved. This technique, therefore, is the recommended scheme. The impact on the flight software is given in Table 10.4.3.5-I for the LEO mission using this technique.

10.4.3.5.4 Polar Analysis

One area of the flight program requires modification for a two-stage Polar mission. The large commanded yaw attitude could approach the yaw limit of $\pm 60^\circ$ of the three-gimbal platform. To prevent the platform from tumbling, a software limit $\pm 45^\circ$ is in the present flight programs. The study effort was concentrated on testing the adequacy of a platform alignment technique for meeting the $\pm 45^\circ$ yaw requirements. The yaw biasing technique, discussed in paragraph 10.4.3.1.4, does not affect the trajectory of the vehicle, but merely changes the platform's measurement of the trajectory. To implement this method, it is desirable to incorporate a routine in the flight program to convert "standard" gimbal angles ($\theta = 0^\circ$) to "offset" gimbal angles ($\theta \neq 0^\circ$) for use during pre-IGM guidance prior to the $\pm 45^\circ$ limiting of yaw in Minor Loop Support. During IGM the use of the proper matrix relating IGM coordinates to platform coordinates automatically includes the offset. The necessary logic changes are illustrated in Figure 10.4.3.1-9. The Polar mission impact on the flight software is summarized in Table 10.4.3.5-I.

10.4.3.5.5 Improved Cutoff Prediction

The cutoff calculations are begun when the predicted time-to-go calculated in IGM becomes less than a presetting. In order to maximize the frequency at which the cutoff calculations are executed and thereby increase the accuracy of the cutoff, only the essential routines are included in the high-speed loop. These are Accelerometer Processing, Boost Navigation, S-IVB Cutoff Backup and the Cutoff Time Prediction Routine. The prediction routine uses a table of velocities and corresponding times to establish a second-order polynomial which expresses velocity as a function of time. This polynomial is then solved to predict the time at which cutoff velocity will be reached. The cutoff velocity is less than the insertion velocity by a preset bias. This bias compensates for thrust tailoff and system delays which cause some velocity to be gained after the cutoff command is issued.

If the engine is shut down from idle mode thrust (5000 lbs) rather than mainstage thrust (230,000 - 265,000 lbs), the velocity gained during thrust decay after cutoff is decreased by a factor of approximately 50. Thus, the magnitude of possible discrepancy between the predicted velocity gained and the actual velocity gained and its effect on the insertion conditions is considerably smaller. Any method that

10.4.3.5.5 (Continued)

could be recommended for adding logic to the present cutoff scheme would be dependent on thrust decay data to determine the method feasibility. Uncertainty in the amount of time involved and thrust levels from mainstage to idle and from idle to cutoff precludes any recommendation that could be made. With appropriate data available, more extensive investigation into cutoff schemes and interactions with the control system could be made.

The apparent advantages of using an idle mode cutoff are: reduced tailoff impulse uncertainty; reduced criticality of high-speed-loop timing; and providing an ullage function. An obvious, but minor, disadvantage is the inclusion of two shutdown signals in the cutoff logic. In the preliminary analysis, see Figure 10.4.3.5-4, the transient from mainstage to idle mode was represented by two steps. The mainstage level before the step was 202,000 pounds. The first step was to 25,300 pounds, while the final step was to expected steady-state idle mode thrust (5000). As the thrust decreases, F/M decreases and the IGM "time-to-go" increases. This possible problem area should be investigated thoroughly but is not expected to be critical due to the expected short transient time and since the actual transient will be smoother than the step representation. When a cutoff scheme is approved for implementation, it would be recommended for all of the missions being considered in this report.

10.4.3.6 Control Analysis

Attitude control system studies were performed with the objective of determining modifications to the baseline vehicle/mission (AS 511 J-2/LOR) flight control system (FCS) necessitated by J-2S implementation. This did not include final design efforts in terms of optimizing control gain programs and shaping network configurations. The AS 504 FCS was considered to be representative of the AS 511 J-2/LOR FCS for these studies.

Vehicle parameters were derived from data supplied by the participating stage contractors. No changes are to be made to the engine actuators for the J-2S application. There is no difference between mainstage and idle mode thrust vector control frequency response; therefore, AS 504 design characteristics have been assumed in this study.

Three vehicle/mission profiles and corresponding payloads—three-stage LOR with Apollo payload, three-stage Synchronous with Apollo payload, and two-stage LEO with a S-IVB/Apollo shaped payload—were considered. No detailed studies were necessary on the two-stage Polar vehicle/mission with S-IVB/Apollo shaped payload because from a FCS standpoint it is similar to the LEO boost phase.

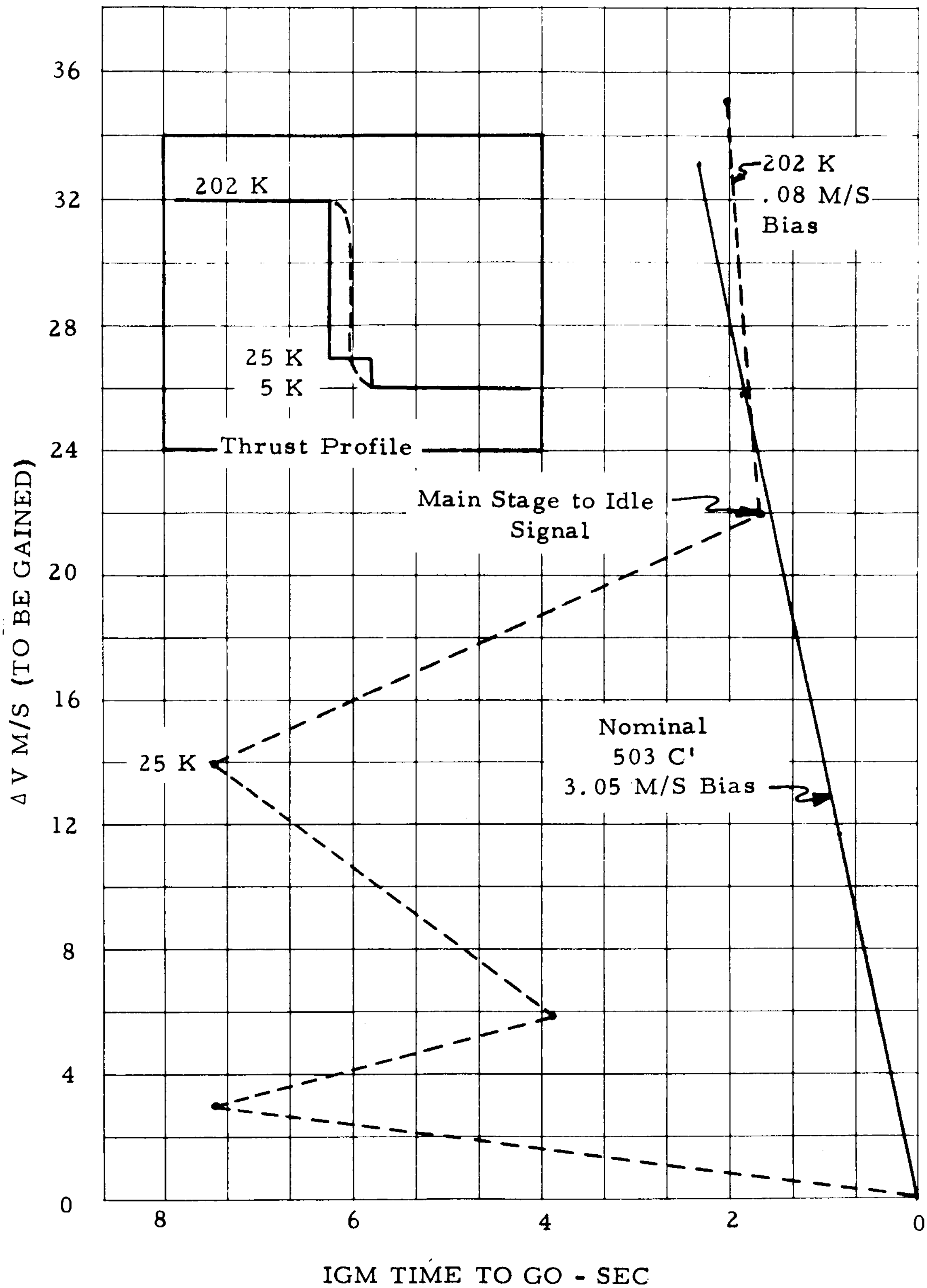


FIGURE 10.4.3.5-4. IDLE MODE CUT-OFF

10.4.3.6.1 J-2S/LOR Mission

Since the J-2S/LOR vehicle, mission, and payload are similar to those of the AS 504, the existing baseline FCS is adequate for S-IC burn, S-II burn, S-IVB mainstage burn, and S-IVB coast modes. Therefore, control studies were oriented toward investigation of requirements during idle mode operation.

a. S-IVB Idle Mode

Pitch/yaw axis control studies were performed for the extended S-IVB idle mode preceding second mainstage burn. Due to uncertainties in the idle mode thrust profile, two candidate control systems were considered.

Gimballed J-2S control using existing S-IVB burn pitch/yaw control scheme.

APS (Auxiliary Propulsion System) control using existing S-IVB coast pitch/yaw control scheme.

Also, as a result of the thrust profile uncertainty, two possible thrust profiles were considered (See Figure 10.4.3.6-1).

The constant 5000 pounds thrust, which is an ideal case, and the ramped thrust, which was obtained from Reference 10.4-30, represent bounds of the actual J-2S idle mode thrust profile.

1. Gimballed J-2S Control

Due to the rapid variation in the ramped thrust profile, time response (rather than "frozen-point" frequency response) analysis was performed to evaluate vehicle performance with gimballed J-2S pitch/yaw control. An analog simulation of a single axis model of the S-IVB stage was utilized to perform the analysis. The simulation model included:

Rigid body dynamics.

Two AS 504 fuel slosh modes.

AS 504 fourth order actuator model with position and rate limiting.

Attitude error and attitude rate feedback with AS 504 S-IVB shaping networks.

Timevarying thrust profile.

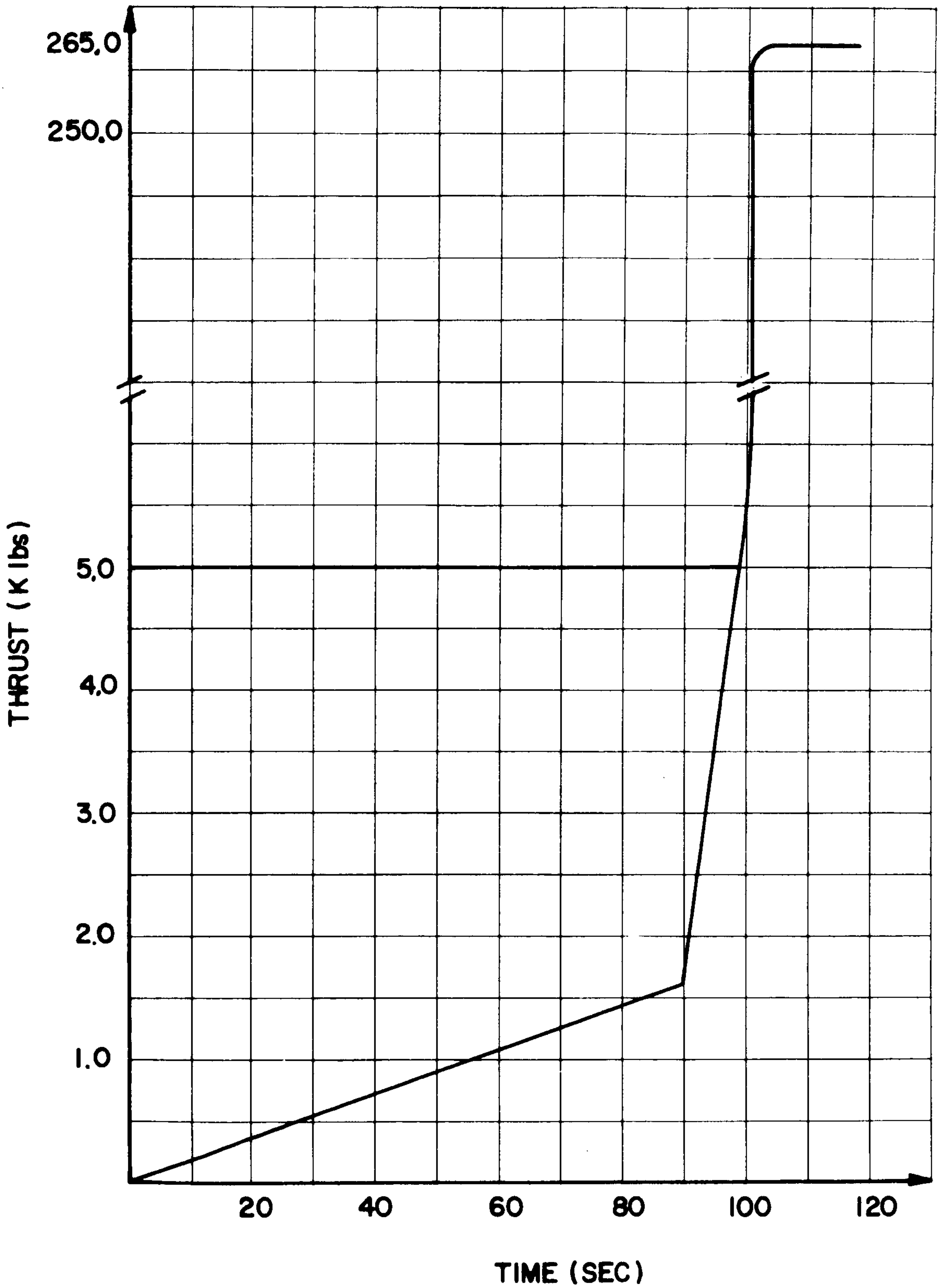


FIGURE 10.4.3.6-1. S-IVB IDLE MODE THRUST PROFILES

10.4.3.6.1 (Continued)

Effects of a 3.47 inches center of gravity offset in the form of an equivalent β_{ma} (engine misalignment angle).

Idle mode vehicle response to realistic initial conditions of ψ (attitude error) = 0.8^o and $\dot{\phi}$ (attitude rate) = 0.04^o/sec were obtained for both the ramped thrust profile and the constant 5000 pound thrust profile with various values of control gains. Attitude error and engine displacement response are shown in Figure 10.4.3.6-2 for the ramped thrust and in Figure 10.4.3.6-3 for the constant thrust. The responses in part (a) of each figure were obtained using control gains of a_o (attitude error channel gain) = 0.81 and a_1 (attitude rate channel gain) = 0.97, which are the probable S-IVB first mainstage burn gains. The responses in part (b) of each figure were obtained using gains of a_o = 2.0 and a_1 = 4.0 with a gain switch back to 0.81 and 0.97 at second mainstage burn. The responses indicate a slowly varying, low damped system as would be predicted by examination of the control mode natural frequency and damping.

$$f_n = \frac{1}{2\pi}\sqrt{a_o C_2}$$

$$\zeta = \frac{a_1}{2}\sqrt{C_2/a_o}$$

where C_2 = control moment coefficient.

Since C_2 is directly proportional to control thrust, the low level of J-2S thrust during idle mode results in a very low control mode natural frequency and damping. The effect on performance of increasing the gains and thus increasing the control frequency and damping can be seen by comparing the responses. The vehicle response is "poor" even for the maximum thrust of 5000 pounds and control gains of 2.0 and 4.0 (f_n = 0.03 and ζ = 0.19). However, the gimballed J-2S control system is adequate since the idle mode lasts a short period of time during which no significant demands on the control system are expected; i.e., idle mode initial conditions are expected to be low (on the limit cycle of the S-IVB coast control system); no significant disturbance torques are expected, and no sizeable maneuvers will be performed during idle mode.

No extensive frozen point stability analysis was performed for the S-IVB Idle Mode, but one Nyquist plot was obtained for a thrust of 5000 pounds and gains of a_o = 2.0 and a_1 = 4.0 A brief discussion of the Nyquist criterion and the digital Nyquist computer program is included in paragraph 10.4.3.6.3.a. In addition to the data used in the time response analysis, two AS 504 flexible body modes were included in the Nyquist. The Nyquist plot indicated a low rigid body phase margin (19^o), as would be expected from the time responses, and gain stabilized bending modes.

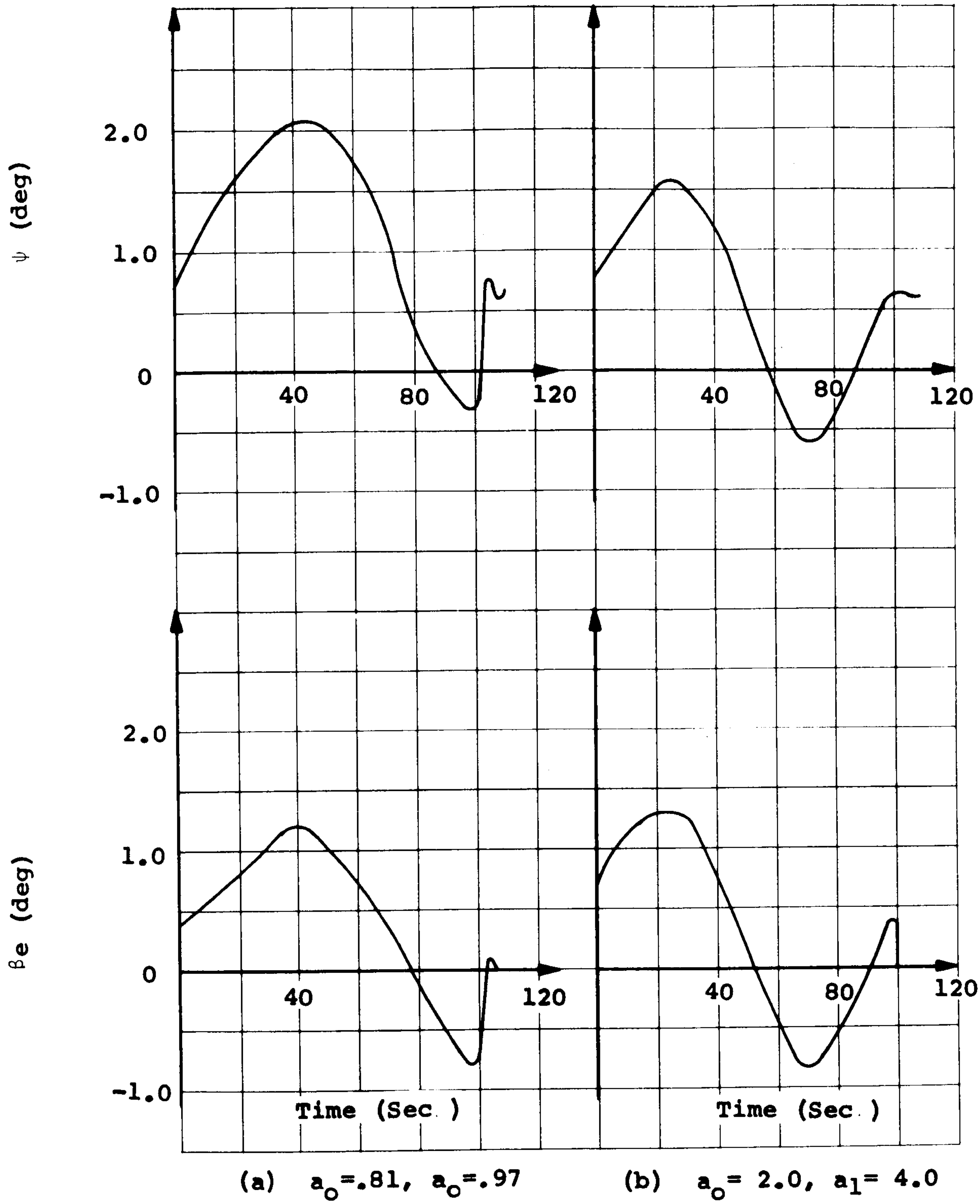


FIGURE 10.4.3.6-2. LOR S-IVB IDLE MODE TIME RESPONSE WITH THE RAMPED THRUST PROFILE

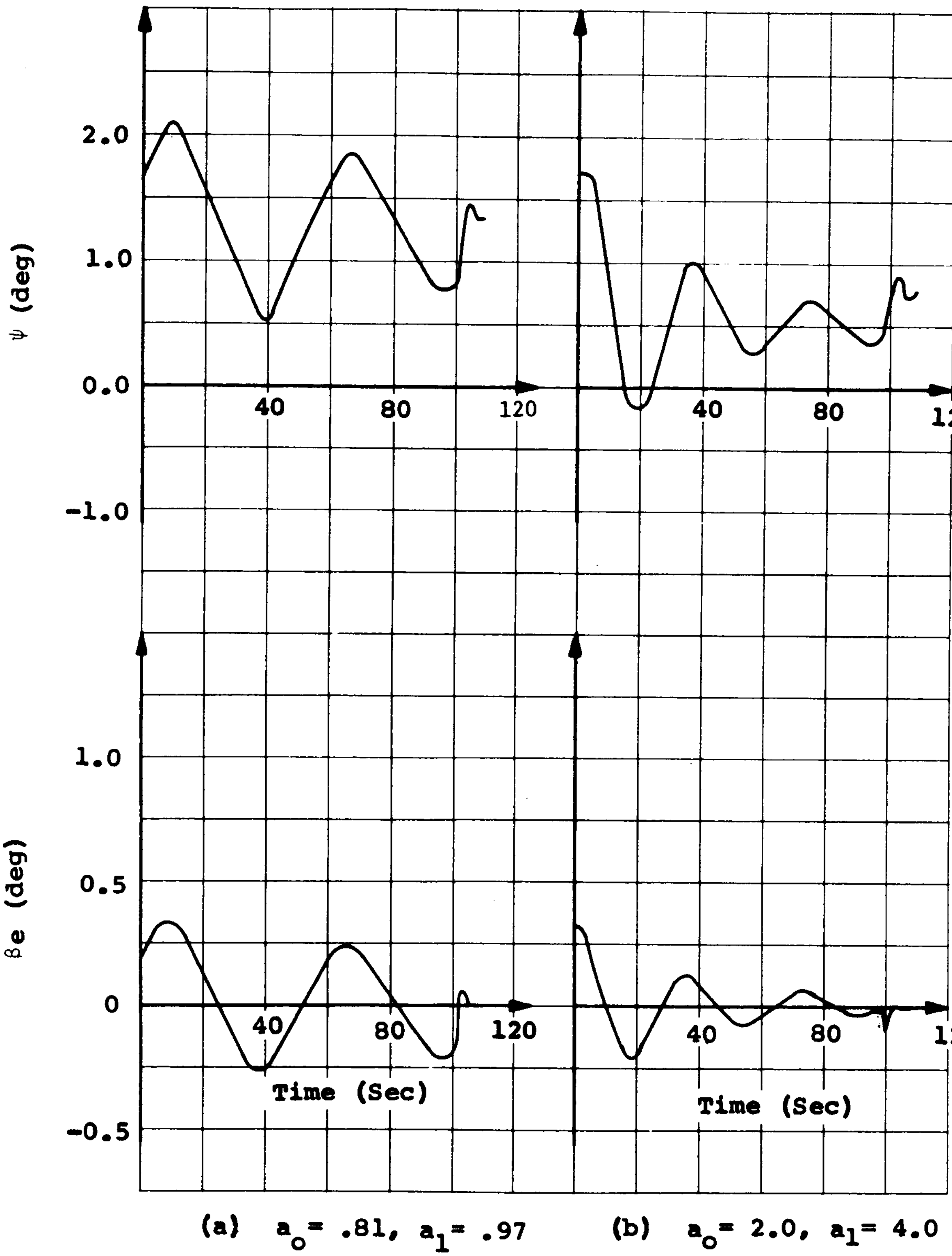


FIGURE 10.4.3.6-3. LOR S-IVB IDLE MODE TIME RESPONSE WITH CONSTANT 5000 POUNDS THRUST

10.4.3.6.1 (Continued)

2. APS Control

Due to the uncertainty in the thrust profile, use of the APS for control during idle mode was investigated. The J-2S engine would be commanded to zero.

To determine APS control adequacy and fuel requirements, a digital simulation of a three-axis rotational model of the S-IVB stage was employed. A disturbance torque resulting from the ramped thrust profile acting through a center of gravity (cg) offset of 3.47 inches together with gravity tradient torques and realistic initial conditions of $\psi = 0.8^{\circ}$ and $\dot{\phi} = 0.04^{\circ}/sec$ were included in the simulation.

Results of the simulation indicated that APS control torque is sufficient, and that the fuel requirement is 4.6 pounds for the 100 second idle mode. This amount of fuel is relatively small when compared to the Saturn V APS fuel capacity of 614 pounds. However, the fuel requirement is sensitive to the idle mode thrust level and cg offset. For thrust levels above the ramped thrust profile and/or cg offsets greater than 3.47 inches, the fuel requirement would increase.

3. Summary

Results of the analysis considering APS control for S-IVB idle mode operation indicate that APS control torque capability is sufficient, and a relatively small amount of fuel is required for the assumed conditions.

Results of the analysis of gimballed J-2S control of S-IVB idle mode operation indicate that the dominant vehicle control mode is lightly damped with a low natural frequency. However, the gimballed J-2S control system is adequate.

Gimballed J-2S is recommended for idle mode pitch and yaw axis control because it is adequate and costs nothing to use. The existing S-IVB burn control scheme with increased gains will be utilized. Control gains will be increased at the end of first S-IVB mainstage burn and switched back to the lower values at second mainstage burn start command. Therefore, the only modifications required to the baseline FCS for the J-2S/LOR vehicle/mission are those needed to provide reset capability for a switch point in the S-IVB shaping networks.

10.4.3.6.2 Synchronous Mission

The J-2S Synchronous mission is equivalent to the J-2S/LOR mission from a FCS standpoint except that the Synchronous mission includes two extended idle modes. The first precedes second S-IVB mainstage burn, and the second precedes third S-IVB mainstage burn.

10.4.3.6.2 (Continued)

Since the vehicle parameters for the first extended idle mode are almost identical to those for the LOR idle mode, no additional analysis was performed. Vehicle responses (See Figures 10.4.3.6-4 and 10.4.3.6-5) were obtained for the second idle mode employing the same analog simulation described in paragraph 10.4.3.6.1.

These responses indicate a higher control mode natural frequency and damping than those for the LOR (or first Synchronous) idle mode. The moment of inertia is less and the control moment arm is greater, resulting in a higher control moment coefficient (C_2) and thus a higher control frequency and damping. From these responses it is evident that the same control gains can be used for both idle modes. Thus, the FCS for the J-2S/Synchronous mission will be identical to the FCS for the J-2S/LOR mission. The increased control gains for S-IVB idle mode will be switched in at first and second S-IVB mainstage cutoff and switched back to the lower gains at second and third S-IVB mainstage start commands.

10.4.3.6.3 J-2S/LEO Mission

As a result of the difference in payload, the vehicle dynamics for the J-2S/LEO mission differ from those of the baseline AS 511 J-2/LOR mission. Therefore, attitude control studies were performed for all flight stages to determine any modifications required to the baseline FCS.

a. S-IC Burn

Stability analysis was performed for the pitch/yaw axis during S-IC burn via the Nyquist criterion. A representative Nyquist plot for a Saturn attitude control system, open at the actuator, is shown in Figure 10.4.3.6-6. Vehicle stability can be related to gain and phase margins, which are designated on the plot. Gain margin is the increase or decrease in gain that would cause the system to be marginally stable. Phase margin is the phase lead or lag that would cause the system to be marginally stable. Acceptable margins are generally considered to be 6 db (a factor of 2 to 1) gain margins, 30 degrees rigid body phase margin, and 45 degrees first bending mode phase margin. The second and higher bending modes are gain stabilized with a gain margin of at least 6 db.

A digital Nyquist computer program of a pitch/yaw axis attitude control system model was employed to obtain Nyquist plots for S-IC burn. The model contained:

Rigid body dynamics.

Four bending modes.

Four AS 504 fuel slosh modes.

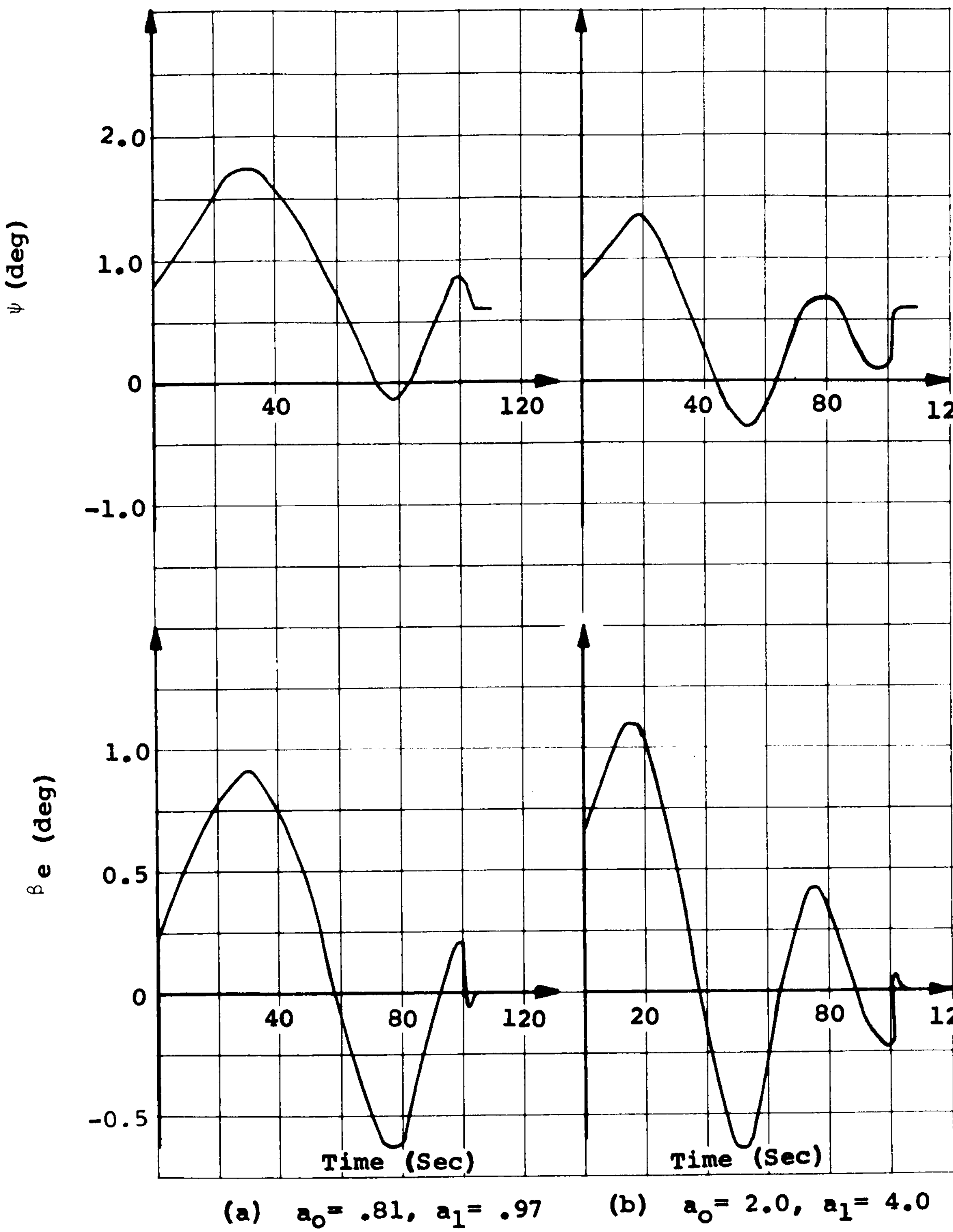


FIGURE 10.4.3.6 - 4. SYNCHRONOUS S-IVB IDLE MODE TIME RESPONSE WITH RAMPED THRUST PROFILE

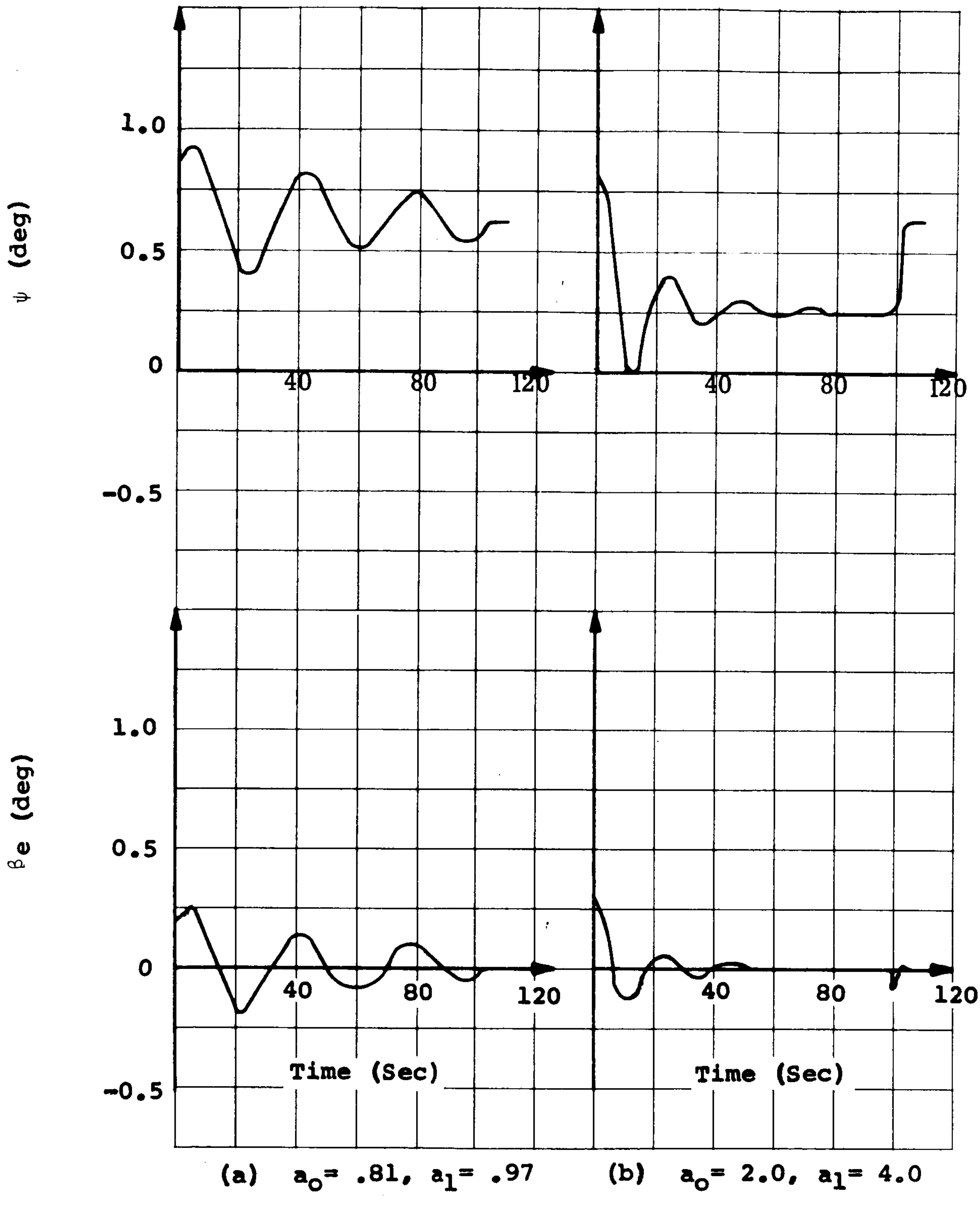


FIGURE 10.4.3.6 - 5. SYNCHRONOUS S-IVB IDLE MODE TIME RESPONSE WITH CONSTANT 5000 POUNDS THRUST

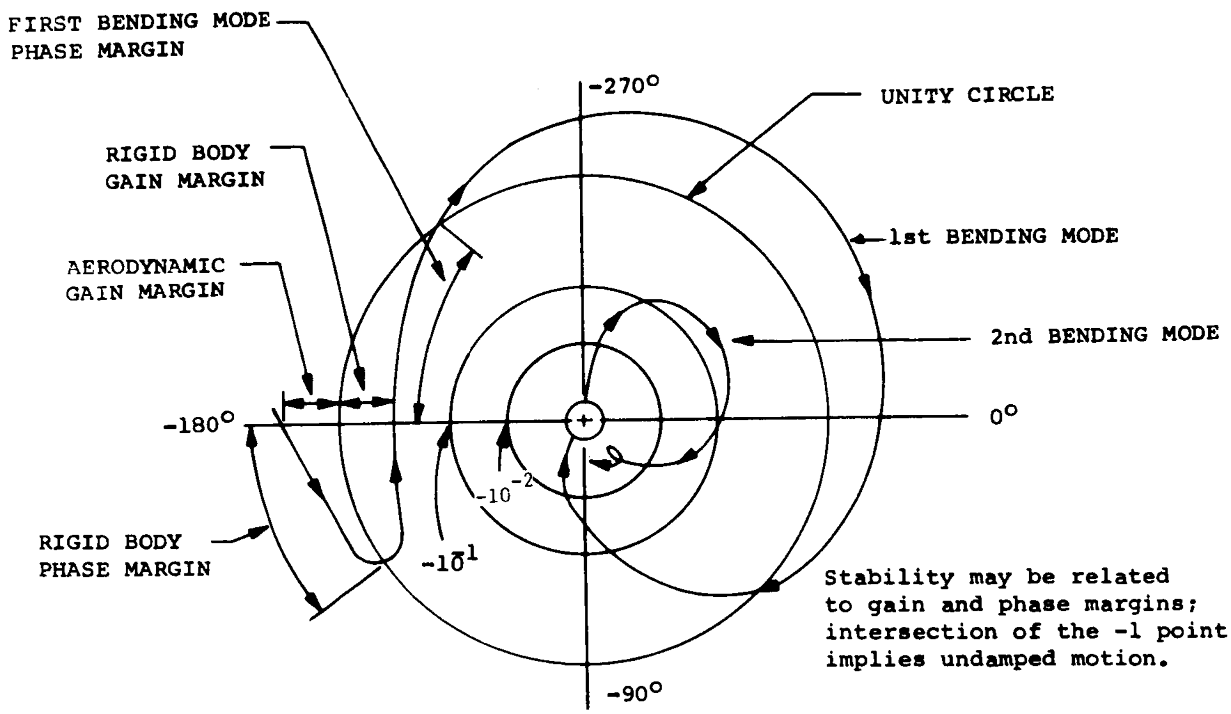


FIGURE 10.4.3.6-6. DESIGN OBJECTIVES WITH RESPECT TO NYQUIST PLOT OF ATTITUDE CONTROL SYSTEM MODEL OPEN LOOP FREQUENCY RESPONSE (OPEN AT THE ACTUATOR)

10.4.3.6.3 (Continued)

AS 504 linear actuator model.

Attitude error and attitude rate feedback.

AS 504 control gains.

Shaping networks similar to those of the AS 504.

No roll stability analysis was performed since the roll axis stability problem is less severe than that for the pitch/yaw axis.

A summary of the rigid body stability margins for lift-off, maximum dynamic pressure, and cutoff periods of flight is presented in Table 10.4.3.6-I. The AS 504 S-IC margins are included for comparison (The Nyquist plot for maximum dynamic pressure is shown in Figure 10.4.3.6-7). The results indicate that stability margins for the LEO S-IC stage pitch/yaw axis obtained on a preliminary design basis are comparable to the AS 504 stability margins. Thus, the baseline S-IC FCS will provide adequate J-2S LEO S-IC burn stability.

b. S-II Mainstage Burn

Stability analysis was performed for the pitch/yaw axis during S-II mainstage burn using the digital Nyquist program of a S-II pitch/yaw model. The S-II model included:

Rigid body dynamics.

Four bending modes.

Two AS 504 fuel slosh modes.

AS 504 linear actuator model.

Attitude error and attitude rate feedback with control gains and AS 504 shaping networks.

Nyquist plots were obtained at S-II mainstage ignition and cutoff. With control gains of $a_0 = 0.80$ and $a_1 = 1.40$ at ignition and $a_0 = 0.40$ and $a_1 = 0.90$ at cutoff, stability margins comparable to those for AS 504 S-II burn were obtained. A summary of the rigid body stability margins are presented in Table 10.4.3.6-II with the AS 504 margins shown for comparison. These results indicate that the baseline S-II control scheme will be adequate for LEO S-II mainstage burn.

TABLE 10.4.3.6-I. COMPARISON OF AS 504 S-IC AND J-2S/LEO S-IC STABILITY MARGINS

STABILITY MARGIN	LIFT-OFF		MAX-Q		CUTOFF	
	LEO	AS 504	LEO	AS 504	LEO	AS 504
Aerodynamic Gain Margin (db)	22.2	22.6	6.1	10.6	5.0	5.1
Rigid Body Phase Margin (deg)	27.0	28.3	27.1	26.4	28.2	33.1
Slosh Gain Margin (db)	7.1	6.1	3.3	3.6	3.4	8.5
1st Bending Mode Phase Margin (deg)	87.3	59.3	93.3	62.4	115.2	66.0
2nd Bending Mode Gain Margin (db)	12.3	6.3	11.8	11.6	42.2	29.0
3rd Bending Mode Gain Margin (db)	29.8	9.8	25.2	10.7	13.9	13.0
4th Bending Mode Gain Margin (db)	7.3	16.6	24.2	11.8	13.1	22.8

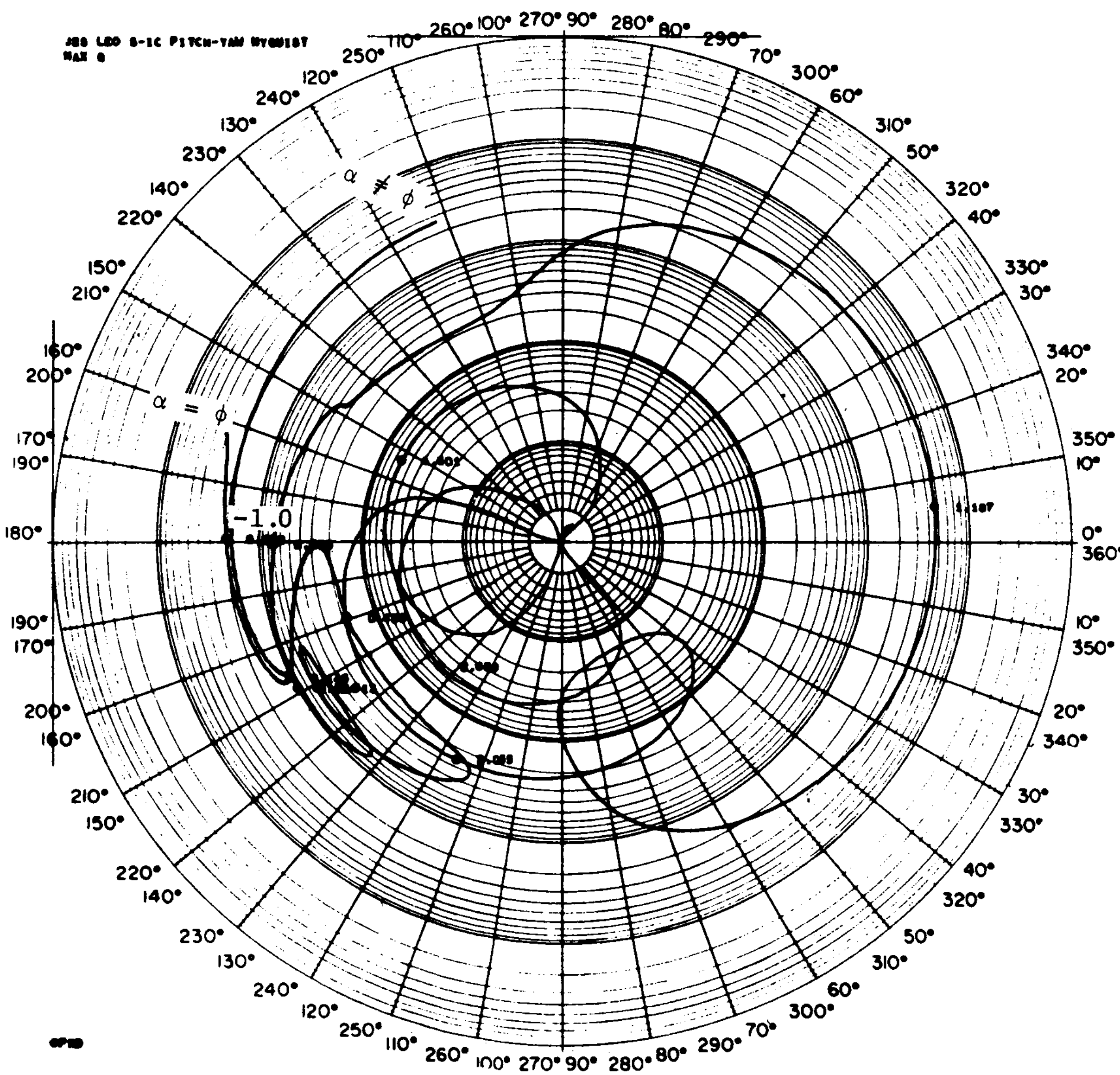


FIGURE 10.4.3.6 - 7. J-2S/LEO S-IC MAX Q PITCH YAW NYQUIST PLOT (OPEN AT THE ACTUATOR)

TABLE 10.4.3.6-II. COMPARISON OF AS 504 S-II AND J-2S/LEO S-II MAINSTAGE STABILITY MARGINS

STABILITY MARGIN	IGNITION		CUTOFF	
	LEO	AS 504	LEO	AS 504
Rigid Body Phase Margin (deg)	43.5	42.9	41.8	40.5
Rigid Body Gain Margin (db)	15.8	15.4	9.2	11.7
1st Bending Mode Phase Margin (deg)	167.0	77.0	gain stabilized 6.1 db	115.0
2nd Bending Mode Gain Margin (db)	47.0	7.5	25.5	18.0
3rd Bending Mode Gain Margin (db)	26.5	15.4	21.8	43.0
4th Bending Mode Gain Margin (db)	16.4	15.4	36.3	23.0

10.4.3.6.3 (Continued)

c. S-II Coast Flight Control

1. FCS Analysis

The proposed coast FCS utilizes Reaction Control System (RCS) modules mounted on the aft skirt of the S-II stage. An alternate approach, assuming the payload is a workshop/space station with a self-contained RCS system, is to utilize the WACS (Workshop Attitude Control System) thrusters during S-II coast phase. See Reference 10.4-49 for a description of the WACS system. Analysis was performed to predict fuel requirements for this alternate approach. A digital simulation of a three-axis rotational model of the S-II coast configuration was used in the analysis.

12 thrusters of 25 lbs thrust each, mounted on the aft end of the Workshop (near S-II Workshop interface).
Note: the S-IVB APS used as S-II/RCS has six thrusters of 150 lbs thrust each.

Spatial amplifiers with pseudo-rate modulation.

Yaw-roll channel coupling.

Reaction jet minimum on time logic.

Limiting of attitude error signals.

Reaction control system.

Gravity gradient disturbance torques - no aerodynamic torques.

Fuel requirements were obtained for the 3000-second coast using as initial conditions the attitude errors and attitude rates at S-II/S-IVB separation of the AS 501 flight and guidance commands of zero in yaw and roll and track local horizontal in pitch. Figure 10.4.3.6-8 (a) shows the fuel requirements as a function of moment arm (or longitudinal center of gravity location) and Figure 10.4.3.6-8 (b) shows the fuel requirements as a function of pitch/yaw moment of inertia. For nominal center of gravity location and pitch/yaw moment of inertia, the required WACS fuel is 203 lbs (The required fuel for RCS modules mounted on the aft of the S-II stage is 7.06 lbs for the nominal conditions.); however, the fuel consumption using the WACS is very sensitive to longitudinal center of gravity location and the pitch/yaw moment of inertia.

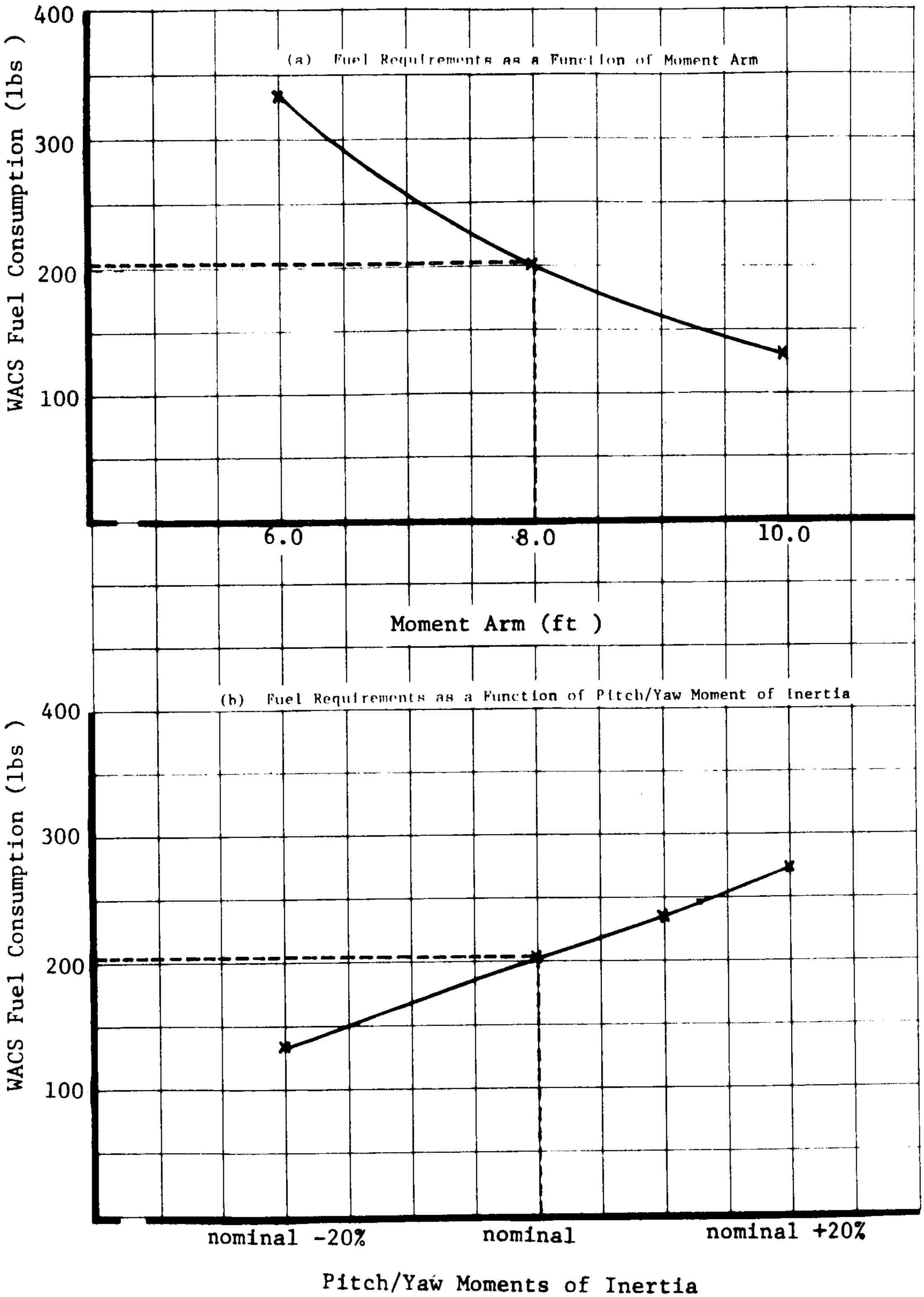


FIGURE 10.4.3.6-8. FUEL CONSUMPTION FOR WACS CONTROL OF S-II COAST

10.4.3.6.3 (Continued)

2. Implementation Impact

Use of either the WACS or the S-II RCS for S-II coast control require minor modifications to the FCS in the IU. The baseline S-IVB APS FCS is used for the S-II RCS and the WACS FCS will exist whether or not it is used for S-II coast control.

The advantages of using the WACS for S-II coast control when the payload is a workshop/space station are:

Elimination of the impact of mounting the RCS on the S-II stage and the associated costs for qualification of the design against structural and environmental constraints.

Elimination of the additional testing required for S-II RCS.

The disadvantages of utilizing the WACS for S-II coast control are:

Increase in WACS fuel requirement.

Less control torque available (400 ft -lb in pitch, 800 ft -lb in yaw and 1080 ft -lb in roll) than with S-II APS; however, this control torque is adequate (assuming nominal cg location defined above).

When the J-2S/LEO mission payload is defined, the alternative of using the payload thrusters for total vehicle flight control during the coast phase may be attractive. The implementation impact points out the fact that a trade-off exists between the advantages of eliminating the S-II RCS and the additional fuel requirement on the WACS. The ultimate decision of which system to use for S-II coast control will have to consider WACS fuel availability and final center of gravity and moment of inertia data for the S-II coast configuration.

d. S-II Idle Mode

1. Pitch/Yaw Stability Analysis

As a first step in determining whether S-II mainstage pitch/yaw control gains and shaping networks would be sufficient for S-II idle mode, a stability analysis was performed. The analysis was made with the same digital Nyquist program and data employed in the S-II mainstage cutoff analysis with the exception that idle mode thrust was considered to be 5000 lbs per engine.

With S-II mainstage pitch/yaw gains and shaping networks the idle mode Nyquist

10.4.3.6.3 (Continued)

(see Figure 10.4.3.6-9) exhibits a very low rigid body phase margin and an unstable first bending mode. Further investigation showed that the attitude rate shaping network for idle mode would have to be different from the mainstage network to obtain desirable rigid body stability margins and phase stabilize the first flexible body mode. With a second over fourth order attitude rate shaping network and control gains of $a_o = 2.0$ and $a_1 = 4.0$, desirable rigid body and flexible body stability margins were obtained as indicated by the Nyquist plot in Figure 10.4.3.6-10.

2. Pitch/Yaw Continuous Time Response

A continuous time response analysis was performed to evaluate the effect of a time varying idle mode thrust and thrust unbalance. A single axis digital simulation of the S-II stage pitch/yaw axis was used. The simulation contained:

Rigid body dynamics.

Four bending modes.

AS 504 linear actuator with position and rate limits.

Attitude error and attitude rate feedback with control gains and shaping networks.

Time varying thrust profiles.

Disturbance torques resulting from thrust unbalance.

The idle mode thrust profile and limits are shown in Figure 10.4.3.6-11. These estimated characteristics and limits were obtained by memorandum from the participating stage contractor, North American Rockwell Corporation.

Vehicle response to realistic initial conditions of $\psi = 0.8^o$ and $\dot{\phi} = 0.04^o$/sec were obtained for two cases:

Case I - each of the four control engines was assumed to have the nominal thrust profile shown in Figure 10.4.3.6-11.

Case II - a worst case thrust unbalance was considered. Each of two adjacent control engines (positive pitch) was given the upper limit thrust shown in Figure 10.4.3.6-11. Each of the opposite two control engines was assumed to have the lower-limit thrust.

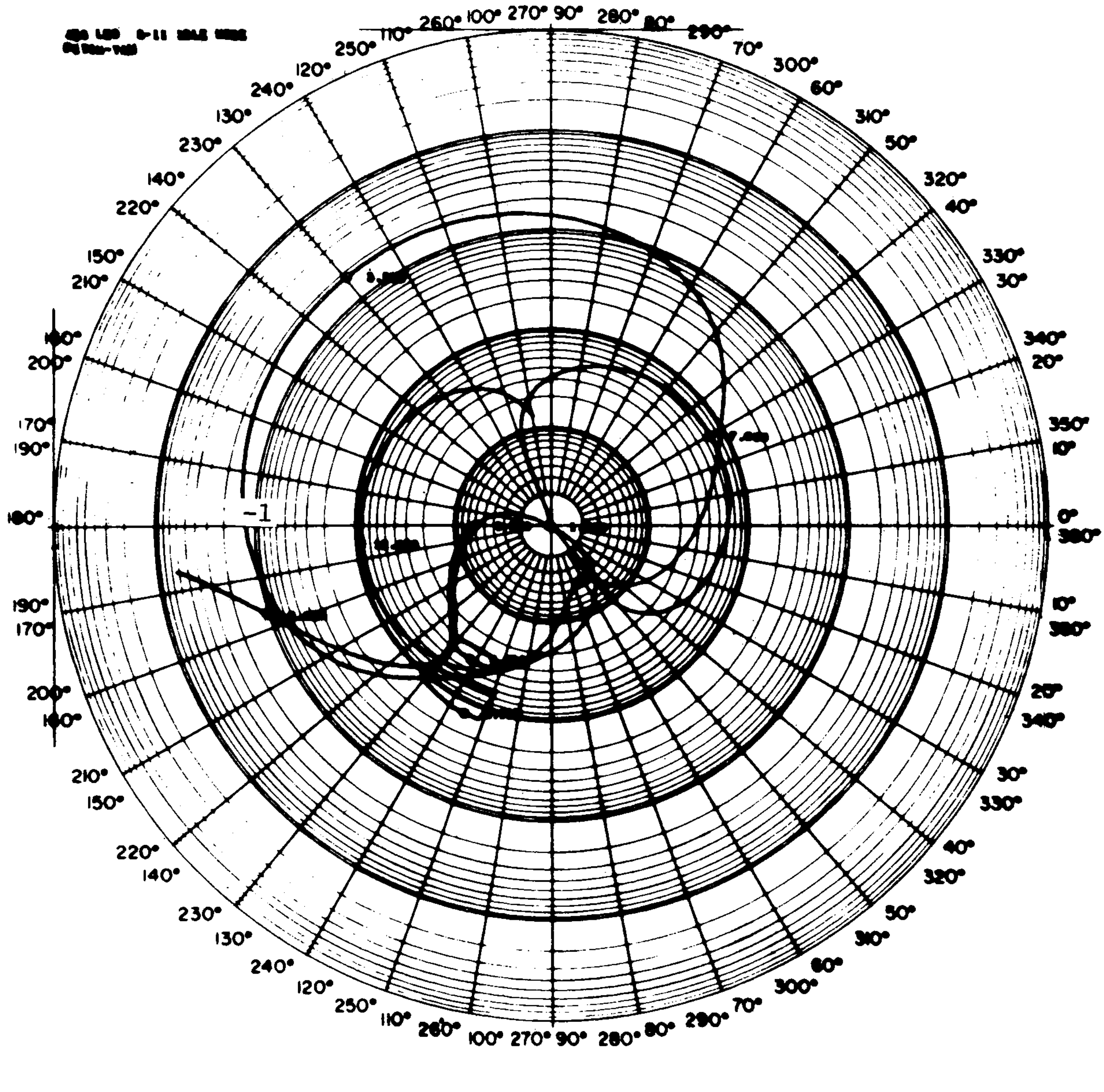


FIGURE 10.4.3.6-9. J-2S/LEO S-II IDLE MODE PITCH YAW NYQUIST (OPEN AT THE ACTUATOR) WITH S-II MAINSTAGE CUT OFF CONTROL GAINS AND SHAPING NETWORKS

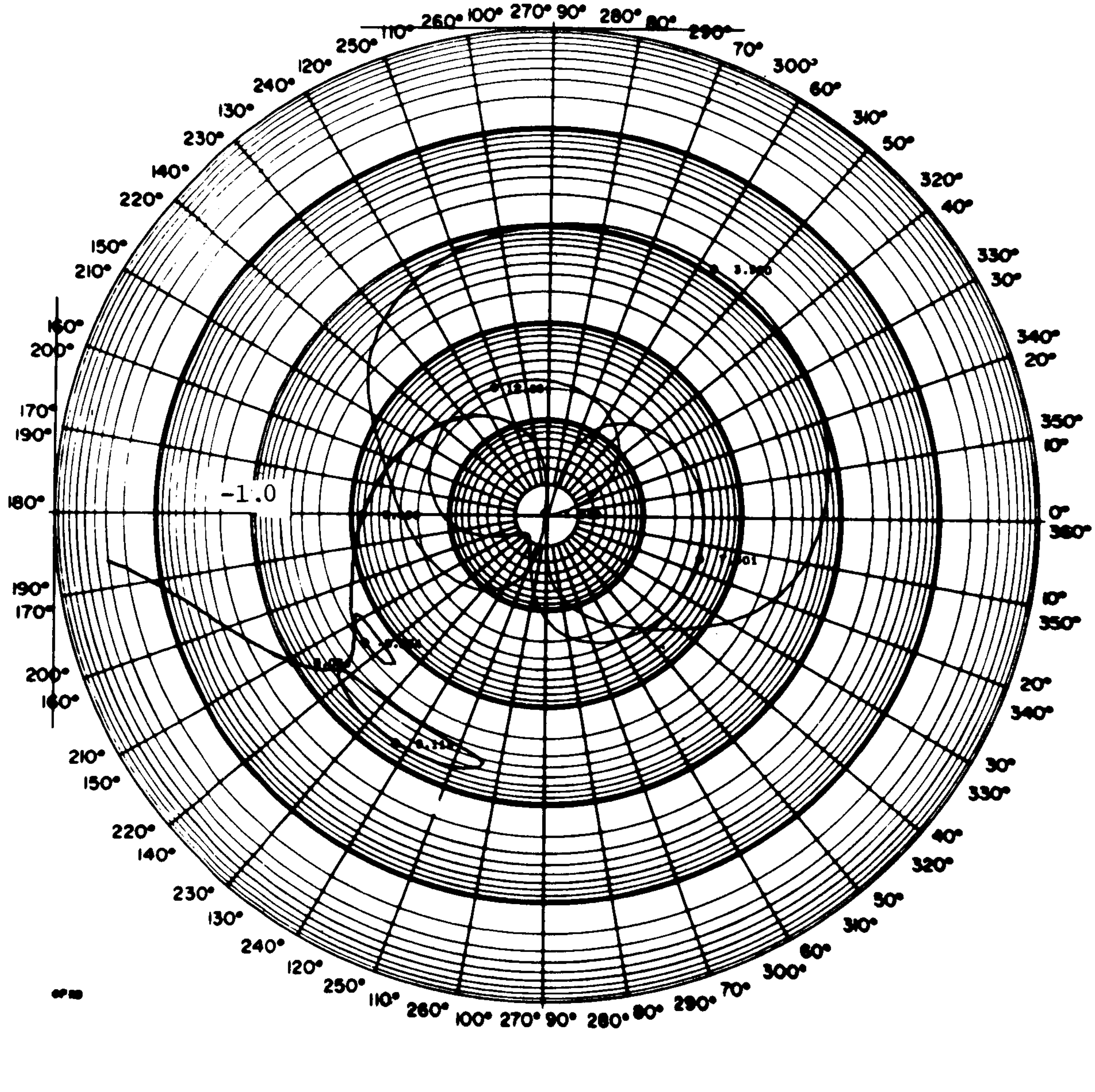


FIGURE 10.4.3.6-10. J-2S/LEO S-II IDLE MODE PITCH YAW NYQUIST PLOT (OPEN AT ACTUATOR) WITH $a_o = 2.0$, $a_1 = 4.0$ AND IDLE MODE ATTITUDE RATE SHAPING NETWORK

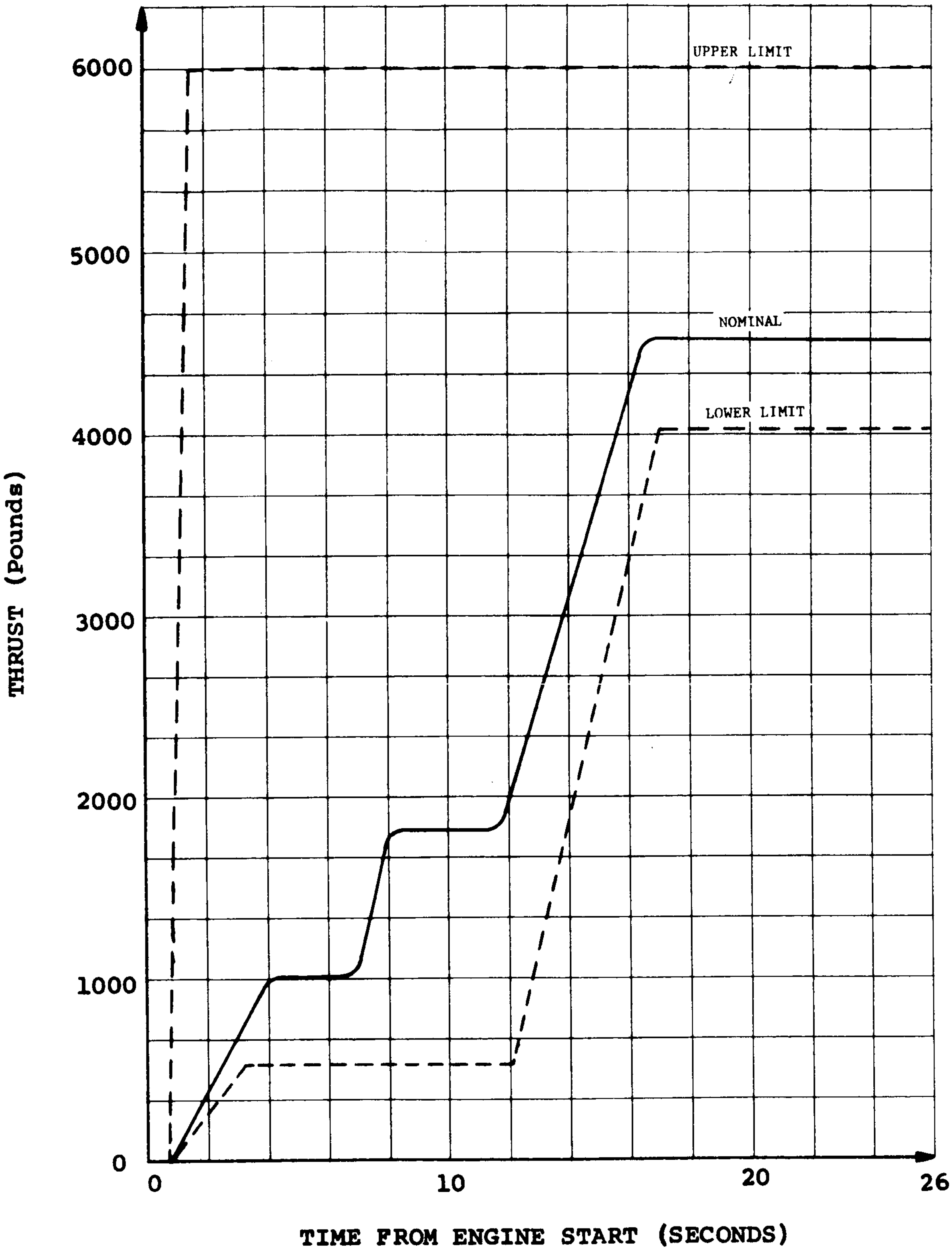


FIGURE 10.4.3.6-11. S-II IDLE MODE THRUST PROFILE AND LIMITS

10.4.3.6.3 (Continued)

Attitude error and engine deflection responses for Case I and Case II are shown in Figures 10.4.3.6-12 and 10.4.3.6-13, respectively. The control gains and shaping networks used were the ones defined by the stability analysis. The results for Case II show that the control system can overcome the worst-case thrust unbalance. The response in both cases is slow and lowly damped due to the low thrust; however, the responses settle out in 40 to 50 seconds which would be adequate for S-II idle mode.

The results of the pitch/yaw stability analysis and continuous time response analysis indicate that a switch at mainstage cutoff to higher attitude error gains and to different attitude rate shaping networks will be required for the idle mode.

3. Roll Stability Analysis

For the S-II idle mode, stability analysis was also performed for the roll axis. The digital Nyquist program was used for a roll axis model containing:

Rigid body dynamics.

Four AS 504 torsional modes.

AS 504 linear actuator model.

Attitude error and attitude rate feedback with control gains and shaping networks.

When the AS 504 S-II roll shaping networks were used, the first two torsional modes were not gain stabilized. However, adequate stability margins were obtained with gains of $a_o = 0.50$ and $a_1 = 1.0$ and shaping networks similar to those used for S-IC roll in the AS 504 FCS. These same shaping networks and gains of $a_o = 0.25$ and $a_1 = 0.20$ were used to obtain adequate stability margins at S-II mainstage cutoff.

The results indicate that the same roll shaping networks (similar to those used for AS 504 S-IC roll) can be used for S-II mainstage burn and S-II idle mode with a gain switch at mainstage cutoff.

e. Summary

Stability analysis indicated that margins comparable to the AS 504 stability margins are obtained for LEO S-IC burn and S-II mainstage burn using control gains and shaping networks similar to those of the baseline AS 511 FCS.

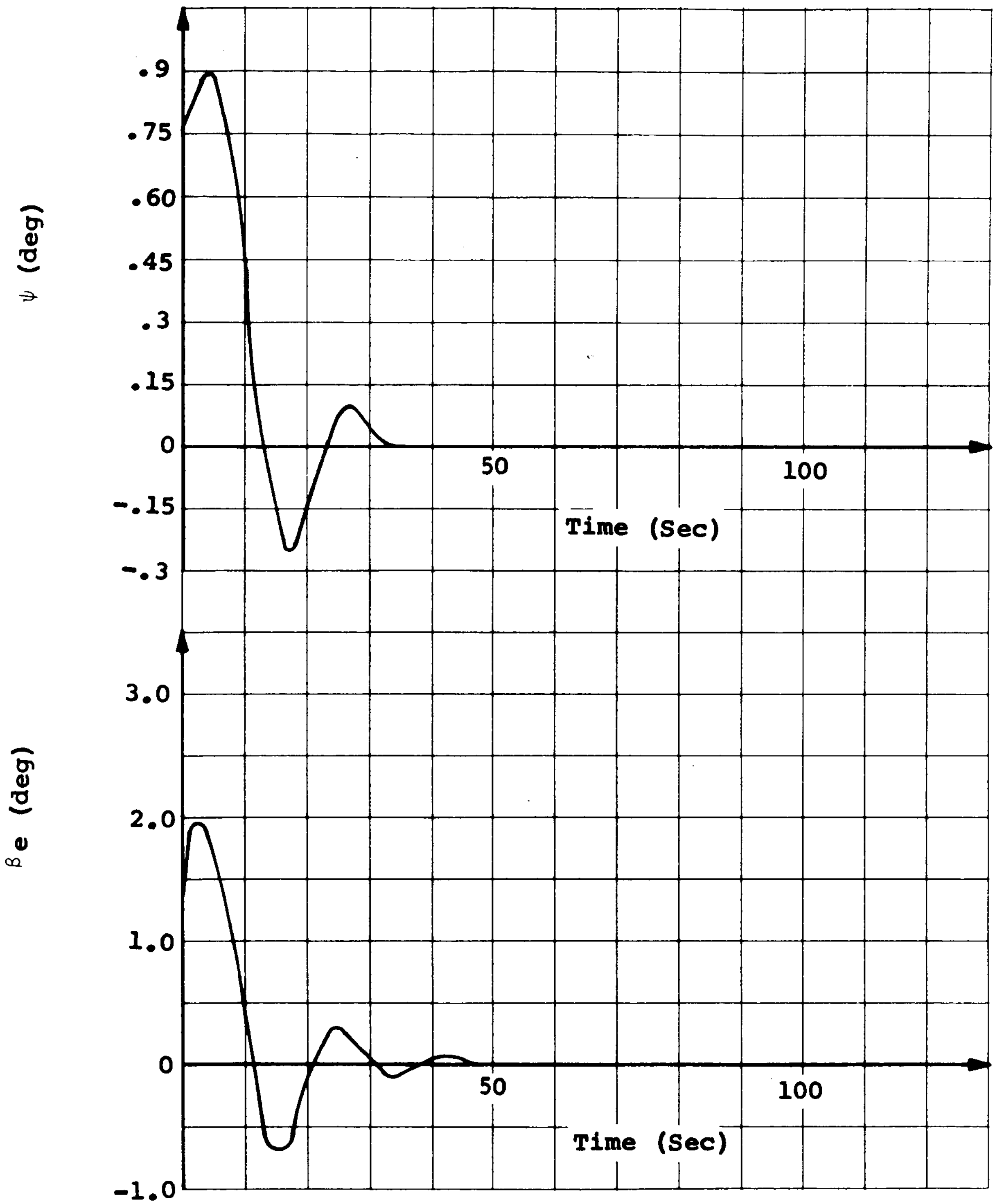


FIGURE 10.4.3.6-12. S-II IDLE MODE TIME RESPONSE WITH THE NOMINAL THRUST PROFILE

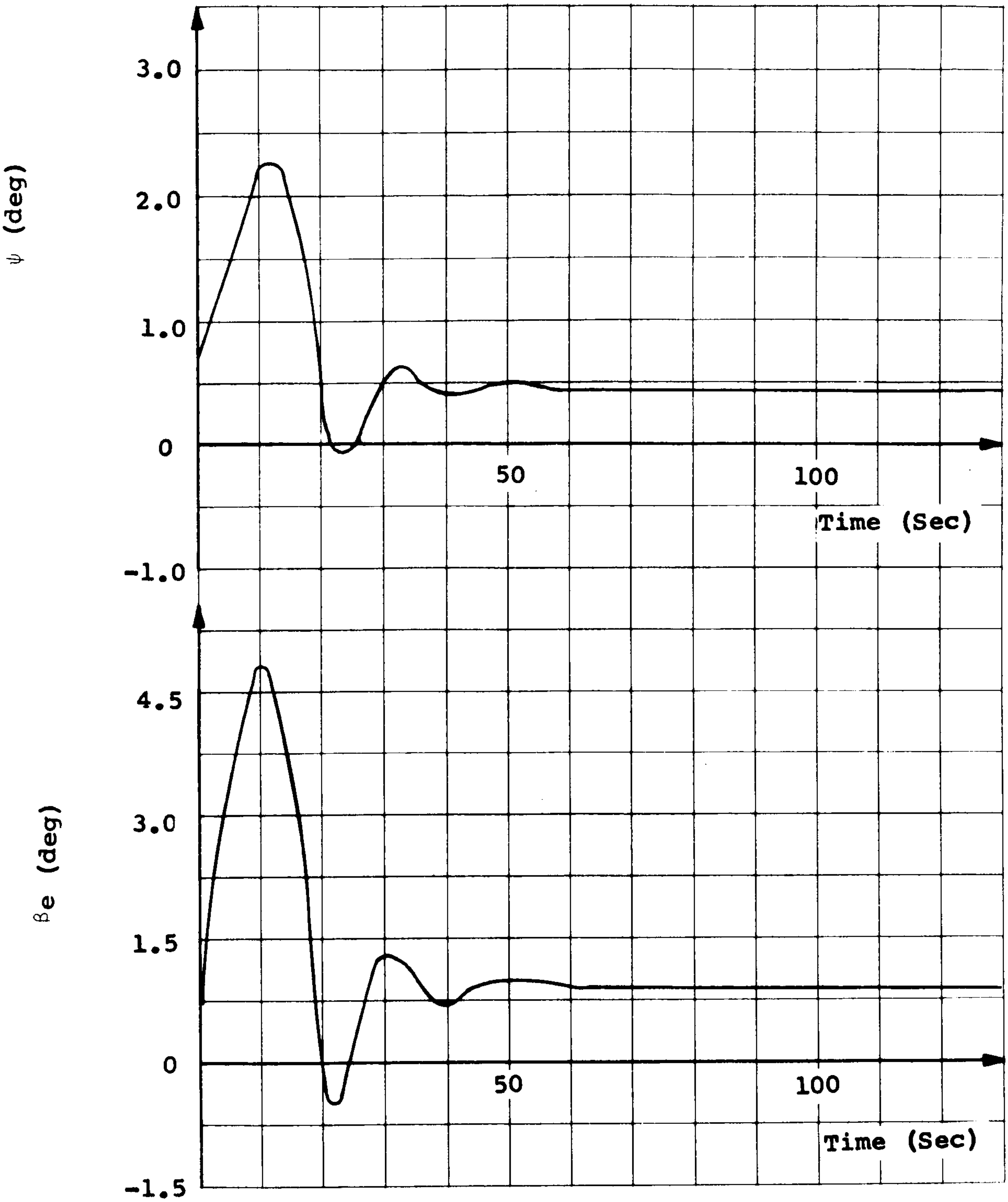


FIGURE 10.4.3.6-13. S-II IDLE MODE TIME RESPONSE WITH WORST-CASE THRUST UNBALANCE

10.4.3.6.3 (Continued)

Continuous time response analysis and stability analysis have shown that adequate vehicle response and stability are achieved for LEO S-II idle mode using S-II main-stage attitude error and roll attitude rate shaping networks with increased gains and specific S-II idle mode pitch and yaw attitude rate shaping networks and gains.

Therefore, with the exception of normal control gain and shaping network design:

No modifications are required to the baseline S-IC FCS.

No modifications are required to the baseline S-II FCS for S-II mainstage burn mode.

One additional gain switch will be required in the S-II pitch and yaw attitude error shaping networks for S-II idle mode.

An additional pitch and yaw attitude rate shaping network will be required for S-II idle mode.

One gain switch will be required in the S-II roll shaping networks for S-II idle mode.

The gains in the pitch/yaw attitude error networks, the gains in the roll networks and the pitch/yaw attitude rate networks will be switched at S-II mainstage cutoff.

10.4.3.6.4 J-2S/Polar Mission

The two-stage Polar vehicle/mission with the S-IVB/Apollo-shaped payload is very similar to the vehicle/payload of the LEO mission. Primary differences are the absence of the RCS, no extended S-II stage coast and steering in the yaw axis during S-IC and S-II mainstage burns. These differences do not significantly affect the boost FCS performance and, therefore, the LEO mission S-IC and S-II mainstage burn FCS analyses (see paragraphs 10.4.3.6.3.a and 10.4.3.6.3.b) are applicable to the Polar mission. The results of the LEO mission analyses indicated that adequate stability margins are obtained using control gains and shaping networks similar to those of the baseline AS 511 FCS. Since the Polar mission does not require an S-II coast phase or an extended S-II idle mode, minor control gain and shaping network changes will be the extent of the modifications to the baseline AS 511 FCS required for the J-2S/Polar mission.

10.4.4 Environmental Analysis

10.4.4.1 Vibration

The J-2S IU is located at the same station location (station 3222) as the present Saturn V IU's. Also the lift-off acoustic environment is generated by the S-IC stage F-1 engines as on present Saturn V vehicles. Thus, the J-2S on-pad and lift-off acoustic and associated vibration environments will be of similar magnitude to those seen on the AS 501, 502 and 503 vehicles and will present no impact on the J-2S IU structure or components, provided the J-2S components are of the same type as the present IU components and are vibration qualified to the present Saturn V environment and vibration specifications. Since the Synchronous IU has the side-by-side GBS panels, this will have to be dynamically tested to determine the coupling effect of the two panels on components and structure.

The maximum dynamic pressures and the associated static accelerations at the J-2S IU station location 3222 are lower than, or approximately equal to, the pressures of AS 501, 502 and 503 and to the acceleration profiles utilized in Reference 10.4-38. Therefore, the J-2S in-flight environment (aerodynamic induced acoustic, vibration and static acceleration) will be similar to the presently imposed environments of the AS 501, 502 and 503 IU's and should not present any problems to the J-2S IU's assuming the components are of the same type as the present IU components or are vibration/acceleration qualified to the present Saturn V environment and vibration/acceleration specifications. Since the Synchronous IU has side-by-side GBS panels, this will have to be dynamically tested to determine the coupling effect of the two panels on components and structure.

The acoustic environment imposed by the J-2S engines on the IU will be no more severe than that for the present Saturn V J-2 engines during S-II burn (Reference 10.4-35). The associated IU vibration environments as depicted on the AS 501, 502 and 503 vehicles during S-II powered flights were negligible. Therefore, the J-2S IU vibration environments will be negligible during S-II burn.

10.4.4.2 Thermal

10.4.4.2.1 Structural Environment

The thermal environments as defined by Reference 10.4-8 for the J-2S improvement studies are less severe during the boost phase of flight than the AS 501, 502 and 503 design trajectory heating. The AS 501 through 503 IU's were not insulated with cork and may be considered a worst-case condition as compared to the insulated (.090 in. layer of cork added to the outside skin of the IU) J-2S IU's.

This layer of cork reduces the maximum structural temperature at the end of first stage boost (Figure 10.4.4.2-1); thus, increasing the IU load-carrying capability at end boost.

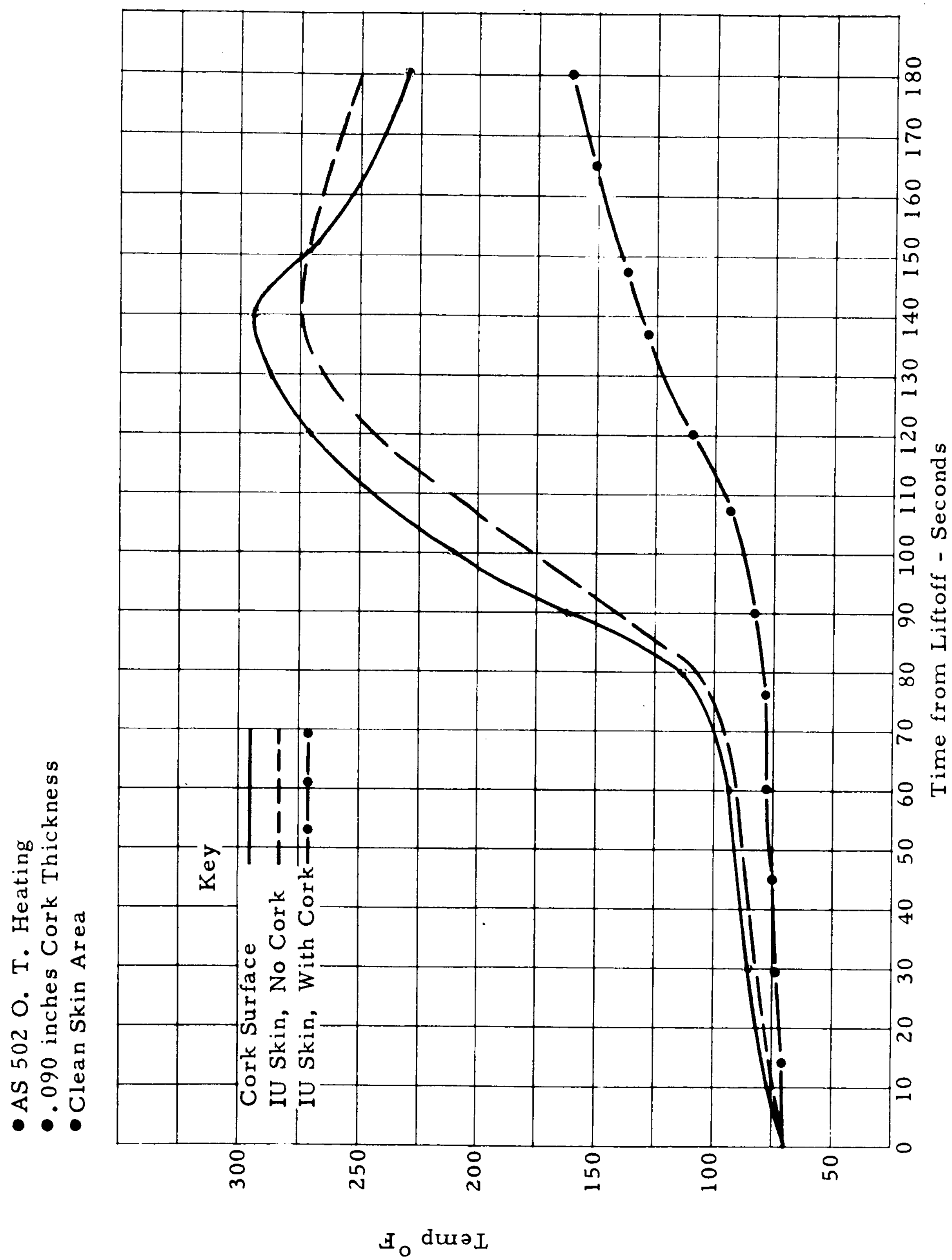


FIGURE 10.4.4.2-1. THERMAL EFFECTS OF CORK DURING BOOST

10.4.4.2.1 (Continued)

The J-2S IU structures are capable of withstanding the end boost structural temperatures as seen from a cursory look, and no analysis was performed.

10.4.4.2.2 ECS Environment

Addition of the cork will have a secondary effect on the operation of the ECS. During boost, the structural temperature will be reduced (Figure 10.4.4.2-1); therefore, ECS heat loads will be reduced also. This will result in slightly lower coolant temperatures prior to and immediately following sublimator start-up and also lower sublimator heat rejection requirements. During orbit, the cork will also act as an insulator, reducing ECS heat losses during the cold portions of the orbit and reducing ECS heat gains during the hot portions of the orbit. The heat gain/loss reduction is expected to be small, and no analyses have been performed to establish the magnitudes. However, assessments can still be made on the effect of the cork for particular mission thermal analyses.

For the 15-hour Synchronous mission, the effect of the cork will be benificial. Previous analyses, documented in "Synchronous Orbit ECP" (Reference 10.4-39) defined the ECS requirement as maintaining a net positive heat load at all times while not exceeding the cooling capacity of the water accumulator (equivalent of 46 kilowatt-hours of sublimator cooling). The analyses demonstrated that the above requirements would be met, while utilizing 80% of the 46 KWH for "hot" case conditions.

During all phases of the "hot" case including Ascent, LEO, Hohmann Transfer and Synchronous, there was a net heat flux from the structure into the ECS. This value was extremely small, 160 watts for LEO and 100 watts for Hohmann Transfer and Synchronous. The cork will reduce these heat gains towards zero, thus increasing slightly the 20% safety factor on the water supply.

For the "cold" case, the ECS loses heat through the structure to space for LEO and Synchronous resulting in a net cooling requirement of approximately 19 KWH.

For the "cold" set of conditions, the cork will reduce the heat losses to bring the cooling requirement somewhere between the maximum value of 36 and the minimum (uncorked) value of 19 KWH.

Thus the cork will improve the ECS heat balance in all cases for the Synchronous by reducing heat gains under hot conditions and thereby conserving water, and reducing heat losses under cold conditions, further assuring the positive system heat load required for proper ECS operation.

10.4.4.2.3 Orbital Environment

The thermal cycling environment which would be encompassed during the 15-hour Synchronous mission does not present any thermal stress problems to the IU structure (Reference 10.4-40); and therefore, no thermal stress analysis is required.

10.4.5 Weight and Mass Properties

The operational weight for the J-2S IU's are shown in Table 10.4.5-I. Also shown are the physical characteristics of the J-2S IU's (LOR, LEO, Polar and Synchronous).

The data in the table reflects the following IU structural modifications:

Remove Vibration Damping Channels.	-63.5 lb
Install Vibration Damping Pad (X-306).	+60.0 lb
Install Cork Thermal Insulation.	+64.0 lb
Install IU/SLA Mating Hardware. (This is a new IU weight reporting requirement).	+ 6.0 lb
Install additional paint for cork.	+11.0 lb
	+78.5 lb

TABLE 10.4.5-I. IU PHYSICAL CHARACTERISTICS

	VEHICLE	
	LOR, LEO, POLAR	SYNCHRONOUS
Weight	4301 (-0, +100) lbs *	4675 (-0, +100) lbs *
X Axis cg	3244.7 in.	3244.2 in.
Y Axis cg	-7.1 in.	-4.8 in.
Z Axis cg	-12.1 in.	-15.0 in.
Roll I_m	1935.1 kg m sec^2	2095.8 kg m sec^2
Pitch I_m	1058.7 kg m sec^2	1150.9 kg m sec^2
Yaw I_m	940.9 kg m sec^2	992.6 kg m sec^2
NOTES:	AS 509 Baseline IU dated 11/20/68, is identical to AS 511.	AS 506 Baseline IU analysis, dated 11/20/68, used to establish Synchronous criteria.
	IU I_m were taken about the IU center of mass.	IU I_m were taken about the IU center of mass.
	Axes are vehicle sign convention.	Axes are vehicle sign convention.
	Design modification weights for LEO and Polar are not included in this table and will be determined subsequent to redesign.	
	*Weight variations account for minor variations prior to launch.	*Weight variations account for minor variations prior to launch.

10.4.6 Electrical Support Equipment/Ground Support Equipment

The purpose of the Electrical Support Equipment/Ground Support Equipment (ESE/GSE) paragraph is the definition of ESE/GSE modifications required by the J-2S engines used on a Saturn V vehicle for a LOR, LEO, Polar Orbit or Synchronous Orbit mission. The ESE/GSE modifications were determined by considering the AS 505 IU as a baseline.

10.4.6.1 Component Acceptance Test

The component acceptance test ESE/GSE will require no modification to acceptance test the components to be implemented on the LOR, LEO, Synchronous or Polar missions.

10.4.6.2 System Test

The system test ESE/GSE modifications are discussed below according to IU subsystems. In each case, the missions are listed in a manner such that the mission creating the greatest amount of impact is easily determined.

10.4.6.2.1 Instrumentation and Communications (I&C)

a. J-2S/LOR, LEO and Polar Missions

The I&C subsystem is not impacted by these missions or the implementation of the J-2S engine. Therefore, no ESE/GSE modifications are required.

b. J-2S/Synchronous Mission

The I&C subsystem does require modification for this mission as described in paragraph 10.4.2.3. Therefore, the following ESE/GSE modifications are required:

In order to verify proper operation of the antenna switching network of the onboard CCS system, a switching matrix must be installed in the RF ground station.

The capability to monitor one additional helix current and six antenna positions must be incorporated into the RF ground station. The helix current measurement will be similar to the one now monitored and the antenna position indicators will be similar to ones now implemented in the ground station.

Five additional cables must be added to the RF ground station to monitor the CCS antenna outputs.

10.4.6.2.1 (Continued)

These modifications represent approximately 50% change to the CCS section of the RF ground station.

10.4.6.2.2 Electrical Subsystem

a. J-2S/LOR, LEO and Polar Missions

The electrical subsystem is neither impacted by these missions nor the implementation of the J-2S engine; therefore, no ESE/GSE modifications are required.

b. J-2S/Synchronous Missions

The IU lifetime extension requirements of a Synchronous mission require adding a fourth battery to the IU. The ground power source has the capability to supply four vehicle buses, but the capability to monitor the bus voltage of the fourth bus must be added. This will require removal of one jumper in the ground power source. The addition of six directional antennas with switching capabilities and the replacement of the existing power amplifier with two uprated power amplifiers with switching capabilities will require adding a total of 13 ESE simulate command lines. Nine ESE lines will be required to switch the antennas and four lines will be required to switch the amplifiers.

10.4.6.3 Software

The automatic checkout program modifications resulting from implementing a J-2S engine to perform a LOR, LEO, Polar or Synchronous mission with a Saturn V vehicle are divided into two categories:

Subsystems Automated Checkout Programs.

IU Overall Checkout Program.

Each type is listed below with a discussion of the changes to be expected with each type.

10.4.6.3.1 Subsystems Automated Checkout Programs

The subsystems automated checkout programs used to check out the following subsystems of AS 505 will require modifications for each of the missions under study. In most cases, only parameter changes will be required.

However, some modifications will require portions of the checkout program to be rewritten.

10.4.6.3.1 (Continued)

a. Control Subsystem

1. J-2S/LOR, Polar and Synchronous Missions

(a) A_1 Gain

(b) A_0 Gain

(c) Control System Nulls

(d) Control Relay Redundancy

(e) Engine Deflection

2. J-2S/LEO Mission

(a) Control Computer Comparators

(b) A_1 Gain

(c) Control Computer APS

(d) Control Computer Relay Redundancy

(e) A_0 Gain

(f) Control System Nulls

(g) Engine Deflection

b. Electrical Subsystem

1. J-2S/LOR, LEO, Polar and Synchronous Missions

(a) Power Distribution and Control

(b) General Networks

(c) Simulated Plug Drop

10.4.6.3.2 IU Overall Checkout Program

The IU Overall Checkout Program is a general program applicable to all Saturn V vehicles with little or no modification from one vehicle to another. The mission requirements of the missions under study are such that the basic checkout program will not require modification. All modifications due to new or changed mission requirements can be loaded into the basic program through the use of user controlled data tables. These tables can be updated and maintained to the latest mission requirements to be inserted into the basic program without impacting that program.

10.4.6.4 Simulation Laboratory

The Simulation Laboratory hardware will require minor modifications. The Flight Control Computer (FCC) is checked out in the Simulation Laboratory, but since the FCC will be modified in a manner to cause least impact, the test console functions will only require new name plates.

10.4.6.5 Flight Evaluation and Analysis

The additional Flight Evaluation and Analysis for the missions under study was determined by considering the mission requirements of AS 503 versus the mission requirements of the missions under study.

10.4.6.5.1 J-2S/LOR Mission

No additional Flight Evaluation and Analysis will be required for the LOR mission.

10.4.6.5.2 J-2S/LEO and Polar Missions

The LEO and Polar Orbit missions will require an additional 25% effort to evaluate and analyze the flight data. The major portion of this effort results from an additional 100% effort in evaluating and analyzing the Guidance and Control flight data.

10.4.6.5.3 J-2S/Synchronous Mission

Due to the life extension requirements of the IU for the Synchronous Orbit mission, it is projected that an additional 100% effort will be required to evaluate and analyze the flight data. All systems must be evaluated to determine the performance during the extended life of the IU.

10.4.6.6 KSC

The KSC ESE/GSE requirements are essentially the same as those at Huntsville; therefore, the same modifications will be required at KSC. However, the ESE/GSE at KSC must be modified by the various support contractors that control the ESE/GSE.

10.4.6.6 (Continued)

The IU hardware modifications will likewise necessitate modification to the automated subsystem checkout procedures at KSC. The checkout programs to be modified are listed below with the missions that require the change.

10.4.6.6.1 Electrical Subsystem

a. J-2S/LEO, LOR, Polar and Synchronous Missions

1. Launch Vehicle Operations for Space Vehicle Overall Test #1 (Plugs In), V-20010.

2. Launch Vehicle Operations for Space Vehicle Overall Test #2 (Plugs Out), V-20012.

3. Switch Selector Functional Verification, V-21107.

4. Power Distribution and Control Switching Test, V-21263.

10.4.6.6.2 Control Subsystem

a. J-2S/LEO Mission

1. Flight Control Computer Comparator Test, V-23169.

2. Flight Control Computer Redundancy Test, V-23171.

3. APS Gain Test, V-23175.

4. FCC Systems Gain Test, V-23176.

b. J-2S/LOR, Polar and Synchronous Missions

1. Flight Control Computer Redundancy Test, V-23171.

2. APS Gain Test, V-23175.

3. FCC Systems Gain Test, V23176

As discussed in the paragraph on Huntsville test software, the IU overall test program requires no modifications to the basic program.

10.4.7 Testing Requirements

The test requirements to be used for testing the J-2S vehicles will be essentially the same as those used on AS 505. The test specifications will however, change to reflect the IU hardware design changes.

10.4.7.1 Development

No new test requirements will be necessary for the J-2S vehicle IU's. Existing test specifications can be used with some modifications to reflect the hardware design changes. The specification modifications will result in less than 1% change in the existing specification documents for the LOR missions, approximately 10% changes for LEO and Polar Orbit missions and approximately 25% change for the Synchronous Orbit missions. The test specifications will be determined and released once the hardware design changes are released.

10.4.7.2 Component Qualification

The component qualification status of the changing components will be unaffected except in the Synchronous Orbit case. The batteries to be used for Synchronous Orbit will require qualification since they are built with entirely new cells. These battery cells have been used on the Agena program and qualification should present no problems.

10.4.7.3 IU Static and Dynamic Test Requirements

10.4.7.3.1 J-2S/Synchronous Mission

No significant difference exists dynamically between the Synchronous Orbit IU configuration and the S-IU 511 baseline study IU. However, in the area of location 23, an increase in dynamic loads is anticipated as a result of the addition of another gas bearing supply panel in this highly loaded region. Most of the dynamic load increase will occur at frequencies below 100 Hertz because of coupling effects between the two adjacently mounted GBS panels.

The predicted response of the basic structure to the increased dynamic loading cannot be analyzed accurately by use of existing techniques. Even though our present analysis technique will not reveal coupling between responsive panels, it is expected to occur and result in a slightly higher GBS panel response. However, the resulting dynamic load on the IU structure and the levels of vibration exposure of the GBS panel and Astrionic equipment components are expected to be within the present component qualification environment.

Since the above assessment is based on engineering judgement rather than rigorous analysis, an IU structure certification test is proposed to confirm that the GBS

10.4.7.3.1 (Continued)

panel coupling effects will result in component environments that are within present qualification levels and to confirm that the structure can withstand the increased loads.

A grazing incident progressive wave acoustic noise test would be conducted on a portion of the IU consisting of locations 21 through 24. Maximum vehicle body loads would be applied to the segment simultaneously with the acoustic environment. A similar test was performed to evaluate a vibration damping modification.

Test hardware would consist of a portion of IU structure segment 30Z13102 which would be loaded with dummy components using flight type mounting panels and brackets. Both IU 511 and J-2S configuration would be tested and the resulting vibration responses would be evaluated to determine coupling effects.

This test would be subcontracted to a test contractor. The contractor would set up and conduct the test. He would also furnish all necessary test equipment, collect the data and submit a data report to IBM.

The acoustic environment specification for this test is shown in Table 10.4.7.3-I. The specification was generated from Saturn Apollo flight data. The Overall Sound Pressure Level and the Sound Pressure Level versus Frequency Spectrum would be based on composite worst-case environments considering both lift-off and Max Q Alpha for Saturn Apollo flights 501, 502 and 503. The normal 3 db tollerance was added to the worst-case environment to complete the specification.

Acoustic noise, vibratory acceleration and structural strain measurements will be required for test evaluation.

Three microphones, 18 accelerometers and 18 strain gages will be required. Data from these measurements will be recorded during the tests for later spectral analysis.

The costs of these tests would be minimized by utilizing existing test fixtures, GFE dummy components and MSFC supplied transportation for transporting the test specimen and test fixtures between IBM and the test contractor. The following dummy components would be GFE:

50M22100	Dummy ST-124 Platform	GFE
SK80-0541-1	Dummy GN_2 Heat Exchanger	GFE
50M22108-1	Dummy Retroreflector Assy	GFE
20Z42023	Dummy GBS Panel Assy	GFE

TABLE 10.4.7.3-I. 1/3 OCTAVE BAND ACOUSTIC TEST LEVELS, REF $2x10^{-5} n/m^2$

GEOMETRIC MEAN FREQUENCY HZ	LEVEL DB	TOLERANCE
5.0	128.0	SPL 0 +4 DB
6.3	129.5	OASPL 0 +4 DB
8.0	130.5	
10.0	132.0	
12.5	133.5	
16.0	134.5	
20.0	135.5	
25.0	136.5	
31.5	137.0	
40.0	138.0	
50.0	141.0	
63.0	143.0	
80.0	144.0	
100.0	144.5	
125.0	145.0	
160.0	145.0	
200.0	144.0	
250.0	142.0	
315.0	141.5	
400.0	141.5	
500.0	140.5	
630.0	140.5	
800.0	140.0	
1000.0	140.0	
1250.0	139.5	
1600.0	138.5	
2000.0	138.0	
2500.0	137.5	
3150.0	137.0	
4000.0	135.0	
5000.0	133.5	
6300.0	131.5	
8000.0	128.0	
10000.0	125.0	
OASPL	155.0	

10.4.7.3.1 (Continued)

50Z13019	Dummy CCS TM Antenna	GFE
50Z12504	Dummy C-Band Antenna	GFE
50Z12500	Dummy VHF TM Antenna	GFE
50Z13018	Dummy "S" Band CCS Air Antenna	GFE
50Z12574	Dummy "C" Band Transponder	GFE

10.4.7.3.2 J-2S/LOR Mission

The LOR mission IU structure will require no additional static and dynamic testing since there are no structure modifications or additional hardware to be added to the IU.

10.4.7.3.3 J-2S/Polar and LEO Missions

The combined loads, axial and bending, for the respective on-pad, Max Q Alpha, and end boost conditions of the Polar and LEO are within the IU capability with the exception of the Max Q Alpha condition. The Max Q Alpha load is a condition in which the bending moment is contributing to a tension running load on the interface rails which exceeds the IU structural tension capability. Since the Polar tension load is a more severe case, the design changes for the Polar case will overcome this condition for Polar and LEO.

As a result of the interface rail modifications discussed in paragraph 10.4.1.1, it is recommended that a test be performed on the interface rail to determine the flange yield on the 8 in. x 15 in. specimen.

10.4.8 Summary of IU Design Changes

The large majority of changes, which in effect convert the J-2/LOR IU to the J-2S/LOR IU, fall within the category of mission-to-mission changes which, in the current program, require a steady state "maintenance-of-design" work force in the engineering design and release cycle. The basic universality of the IU design, the electrical support equipment, and the digital computer analytical programs, makes accommodation to new missions very straightforward. An exception is the Synchronous mission where lifetime extension requires modifications that were not designed in, but again are easily implemented in exercising the growth potential of the IU.

10.4.8.1 J-2S/LOR IU Changes

Minor changes to the Flight Control System include provision for a reset capability into one of the two switch points in the S-IVB pitch and yaw attitude error and attitude rate filters. The changes are confined to the Flight Control Computer and the Control Distributor.

Flight programs and electrical support programs require minor parametric value changes and sequence reprogramming because of the idle mode and minor variations in event sequences .

10.4.8.2 J-2S/Synchronous IU Changes

It is assumed that all changes made to the J-2S/LOR IU cover all engine-imposed changes. Additional changes which are mission-imposed, follow.

Additional Switch Selector commands will be required to initiate the desired sequencing and inhibiting of the power amplifiers and antennas. Minor rework will be required in the distributors for the sequencing and inhibiting logic.

Five additional CCS directional antennas are employed. To maintain satisfactory circuit margins, two modified Power Amplifiers are used. A power divider and two coax switches are added to permit antenna selection.

A four uprated IU battery configuration was chosen for the J-2S/Synchronous mission which extends the IU power system lifetime from 6.8 to 15 hours. The batteries are a new design and result in a non-recurring cost impact.

Extensive rework of the flight program results from the mission. Yaw biasing of the three gimbal ST-124 platform and retargeting equation implementation are unique to the application.

The Environment Control System is modified by the addition of an additional two cu ft GN_2 storage sphere and mounting appurtenances and a redesigned TCS orifice regulator assembly.

Extensive logic changes are required to the Electrical Support Equipment automated checkout program with no change to the basic program. The ESE hardware changes include network modifications for accommodation of the new batteries and antennas.

10.4.8.3 J-2S/LEO IU Changes

Moderate redesign of the Flight Control Computer (FCC) is required to eliminate the S-IVB circuitry and to add the S-II idle mode pitch and yaw attitude rate filters. Blank boards will replace populated boards to minimize the redesign impact. Major modification is required of the FCC wiring harness.

An alternate provision for a Reaction Control System for the J-2S/Polar vehicle during the Hohmann transfer is the use of the Auxiliary Propulsion System of the space station with control provided by the IU. The changes to the IU are minor since identical instrumentation of the function is required in either approach. The cost effectiveness of the integrated payload/vehicle approach appears to be significant. Further study is recommended after further development of the space station.

IU PCM/DDAS Telemetry links to the S-II stage and transmission of data via the S-II PCM telemetry system will be required. The design impact is minor.

The LEO mission will impose careful optimization of the guidance filters. Implementation is unique for restart state and estimation of the terminal targeting.

Additional gain switching will be required in the S-II pitch and yaw attitude error shaping networks for the S-II idle mode and one gain switch will be required in the S-II roll shaping network for S-II idle mode.

The IU Environmental Control System liquid coolant lines to the S-IVB stage will be capped off.

10.4.8.4 J-2S/Polar IU Changes

Because the S-II RCS is not used, the elimination of all S-IVB curcuitry, including APS circuitry from the FCC, results in extensive rework. The design impact is minimized by use of dummy cards rather than total repackaging of the FCC.

Both the LEO and Polar missions impose excessive tension loads which require minor structural modification.

10.5 ASTRIONIC SYSTEM INTEGRATION

The Astrionic System Integration section is a consolidation of changes to the Astrionics of the stages where like subsystems for the entire vehicle may be evaluated by inspection without cross referencing. Detailed explanation and analysis may be found in earlier sections under corresponding Stage/Systems Description. Attention is invited to the fact that in many cases, the text is verbatim extraction from preliminary stage reports provided by stage contractors.

This section is divided into four subsections: the J-2S deltas to the J-2 /LOR vehicle configuration, J-2S/Synchronous, J-2S/LEO, and J-2S/Polar. The J-2S/LOR section will contain essentially all changes due to the conversion to J-2S engines. The three sections following the J-2S/LOR section will contain mission oriented changes with separate identification of additional modifications directly resulting from J-2S engine impact. Assumptions made are:

AS 511 Saturn vehicle configuration is basically identical to AS 505 (IU only).

Any in-line modifications are compared with present Saturn V operation.

Two-stage vehicles will be manned flights and will have an undefined 22 ft. diameter payload assumed to be a space station.

If a system is modified, deleted or shaded, the changes are so noted by a shaded overlay with an explanatory note on each figure.

10.5.1 J-2S/LOR Astrionic System Interface

10.5.1.1 Purpose

This section of the J-2S Improvement Study (Astrionic System Integration) defines the Astrionic System differences between the three-Stage Saturn V vehicle AS 511 configured with J-2S engines on the S-II and S-IVB stages and the AS 511 vehicle configured with conventional J-2 engines.(Figure 10.5.1.1-1) Mission definition for the baseline study is a Lunar Orbit Rendezvous described by Figure 10.5.1.1-2.

If an existing system is changed, only the changes or additions are described in this document and the Astrionic System Handbook and/or Saturn V Flight Manual is referenced for a more detailed overall description. Each illustration in this section is referenced to a corresponding illustration in the Astrionic System Handbook or the document from which it was taken.

Ref 10.4-32, Fig. 1-3

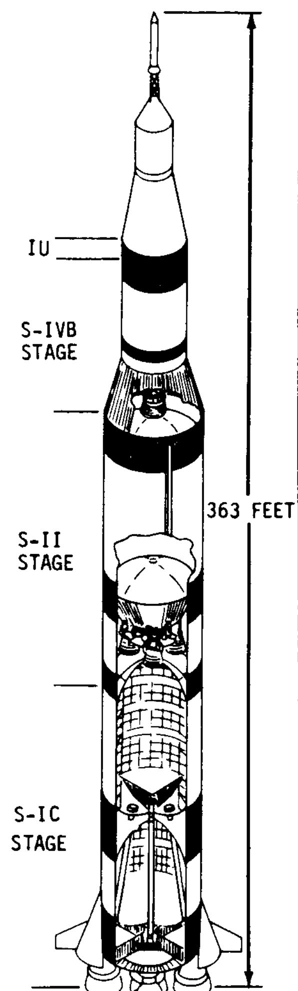
SATURN V LAUNCH VEHICLE

SOLID ULLAGE ROCKET AND RETROROCKET SUMMARY				
STAGE	TYPE	QUANTITY	NOMINAL THRUST AND DURATION	PROPELLANT GRAIN WEIGHT
S-IC	RETROROCKET	8	75,800 POUNDS• 0.541 SECONDS	278.0 POUNDS
S-II	ULLAGE	4	[illegible] POUNDS‡ 3.75 SECONDS	[illegible] POUNDS
	RETROROCKET	4	34,810 POUNDS‡ 1.52 SECONDS	268.2 POUNDS
S-IVB	ULLAGE	2	[illegible] POUNDS‡ [illegible] SECONDS	[illegible] POUNDS

ENGINE DATA					
STAGE	QTY	ENGINE MODEL	NOMINAL THRUST EACH	NOMINAL THRUST TOTAL	BURN TIME
S-IC	5	F-1	1,526,500	7,632,500	150 SEC
S-II	5	J-2	[illegible]	[illegible]	[illegible] SEC
S-IVB	1	J-2	[illegible]	[illegible]	[illegible] SEC

STAGE DIMENSIONS		
	DIAMETER	LENGTH
S-IC Base (including fins)	63.0 FEET	138 FEET
S-IC Mid-stage	33.0 FEET	
S-II Stage	33.0 FEET	81.5 FEET
S-IVB Stage	21.7 FEET	59.3 FEET
Instrument Unit	21.7 FEET	3.0 FEET

SATURN V STAGE MANUFACTURERS	
STAGE	MANUFACTURER
S-IC	THE BOEING COMPANY
S-II	NORTH AMERICAN-ROCKWELL
S-IVB	McDonnell Douglas Astronautics Co.
S-IU	INTERNATIONAL BUSINESS MACHINE CORP.

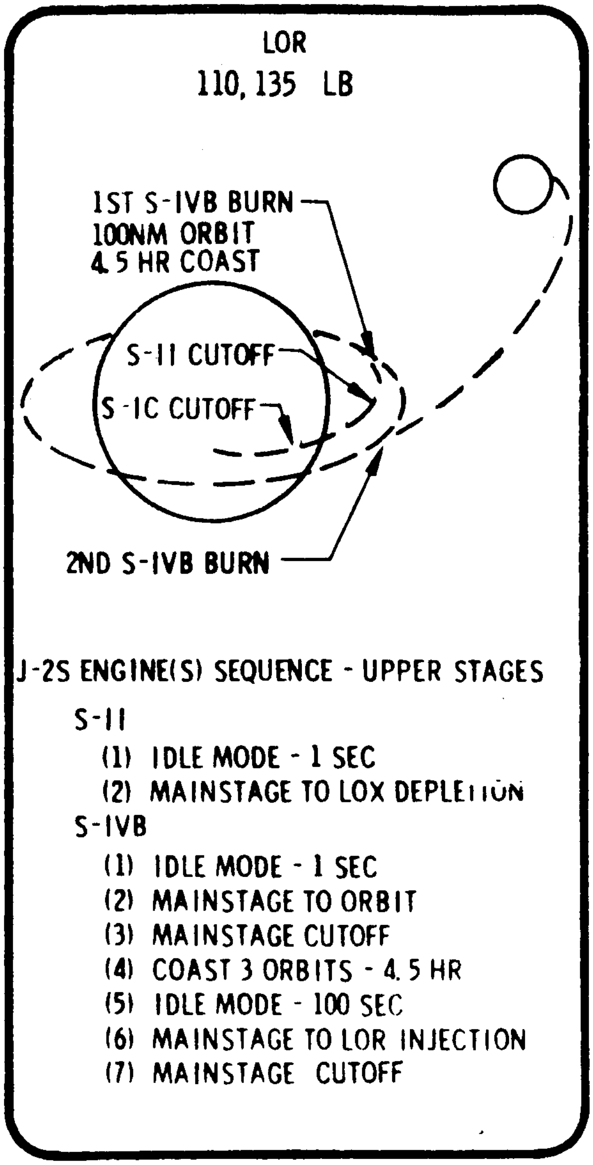


PRE-LAUNCH LAUNCH VEHICLE GROSS WEIGHT ≈ 6,368,000 POUNDS

• MINIMUM VACUUM THRUST AT 120°F
(shaded) AREA CHANGED
‡ NOMINAL VACUUM THRUST AT 60°F

NOTE: THRUST VALUES, WEIGHTS, AND BURN TIMES ARE ALL APPROXIMATIONS.

FIGURE 10. 5. 1. 1-1. SATURN LAUNCH VEHICLE



Ref 10.4-1

FIGURE 10.5.1.1-2. LOR MISSION PROFILE

10.5.1.2 J-2S/LOR Astrionic System Functional Description

The Astrionic System includes that integrated group of components and/or subsystems which provide the following vehicle functions during flight:

Navigation, guidance, and control of the vehicle

Measurement of vehicle parameters

On-board data management

Data transmission between vehicle and ground stations (up and down)

Tracking of the launch vehicle

Checkout and monitoring of vehicle functions

Detection of emergency situations

Generation of electrical power for system operation

Power and signal distribution

Thermal conditioning of components

Separation of stages

Propellant management

Most of the astrionic system components are located in the Instrument Unit (IU) which is mounted on top of the S-IVB Stage. Additional components such as telemetry, distributors, thrust vector control subsystem, electrical networks, are located in the vehicle stages.

The overall Astrionic System of the J-2S AS 511 vehicle is shown in the simplified block diagram, Figure 10.5.1.2-1.

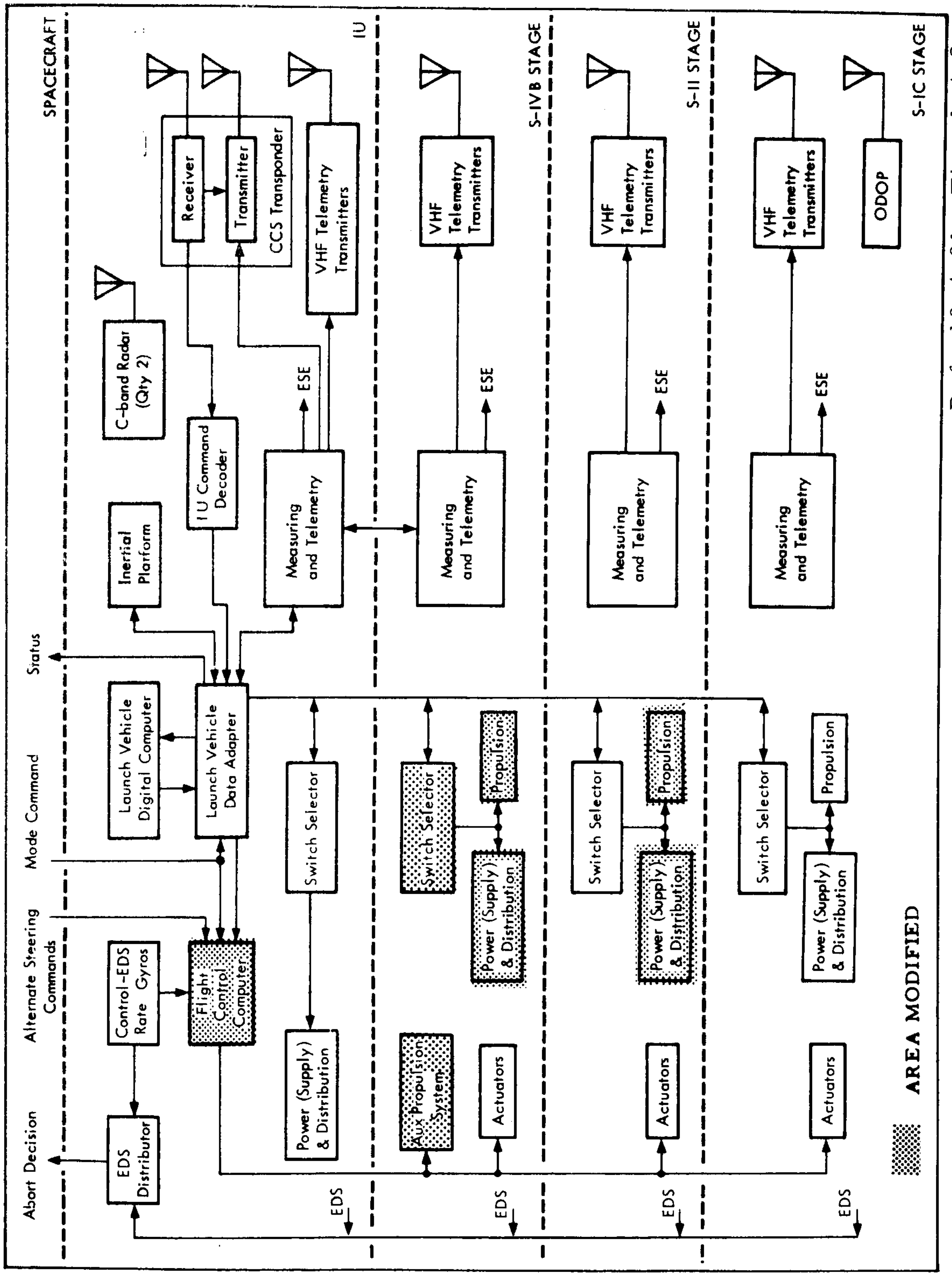
a. Navigation, Guidance and Control

Function and Description

The G&C system provides the following basic functions during flight (Reference Figure 10.5.1.2-2.

Stable positioning of the vehicle to the commanded position with a minimum amount of sloshing and bending.

A first stage tilt attitude program which gives a near zero lift trajectory through the atmosphere.



Ref 10.4-31, Fig. 1-4.2

FIGURE 10.5.1.2-1. ASTRIONIC SYSTEM

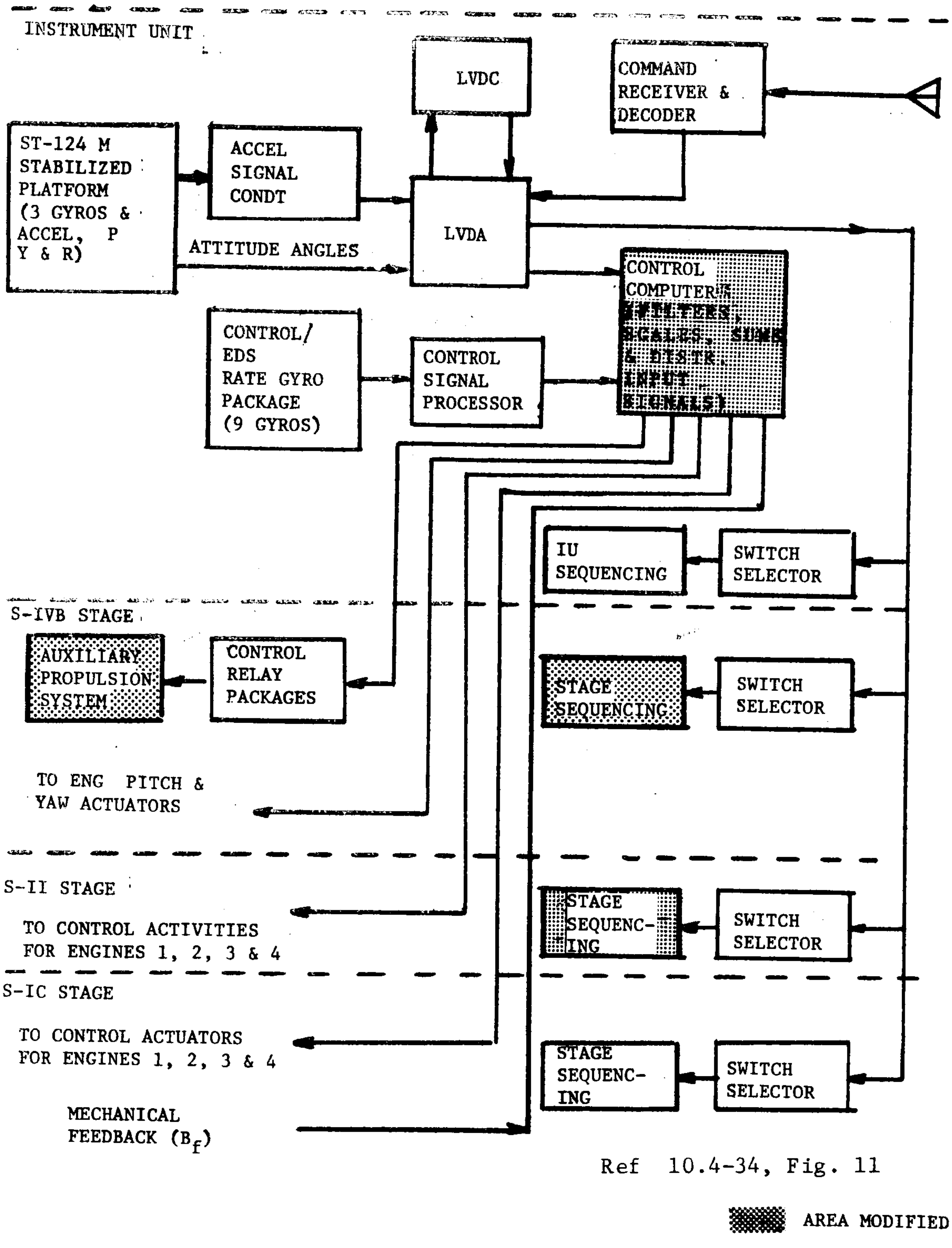


FIGURE 10.5.1.2-2. J-2S NAVIGATION, GUIDANCE AND CONTROL SYSTEM

10.5.1.2 (Continued)

Provides steering commands during S-II and S-IVB burns which guide the vehicle to a predetermined set of end conditions while maintaining a minimum propellant trajectory for earth orbit insertion.

Maintains the proper vehicle position during earth orbit.

Provides guidance during the second S-IVB burn, placing the vehicle in the proper orbit.

G&C Hardware

The Stabilized Platform (ST-124M) is a three-gimbal configuration with gas bearing gyros and accelerometers mounted on the stable element. Gimbal angles are measured by redundant resolvers and inertial velocity is obtained from integrating accelerometers (See Figure 10.5.1.2-2).

The Launch Vehicle Data Adapter (LVDA) is an input-output device for the Launch Vehicle Digital Computer (LVDC). The LVDA/LVDC components are digital devices which operate in conjunction to carry out the flight program. The flight program performs the following functions: (1) processes the inputs from the ST-124M, (2) performs navigation calculations, (3) provides the first stage tilt program, (4) calculates IGM steering commands, (5) calculates attitude errors, and (6) issues launch vehicle sequencing signals.

The Control/EDS Rate Gyro package contains nine rate gyros (triple redundant in three axes). Their outputs go to the Control Signal Processor (CSP) where they are voted and sent to the Flight Control Computer (FCC) for damping vehicle angular motion.

The FCC is an analog device which receives attitude error signals from the LVDA/LVDC and vehicle angular rate signals from the CSP. These signals are filtered and scaled, then sent as commands to the S-IC, S-II and S-IVB engine actuators and to the Auxiliary Propulsion System (APS) Control Relay packages. The Control Relay packages accept FCC commands and relay these commands to operate propellant valves in the APS.

The Switch Selectors in each stage are used to relay Sequencing Commands from the LVDA/LVDC to other locations in the vehicle.

b. Measurements and Data Transmission

Each vehicle stage is equipped with a Measuring and Telemetry System, including RF transmitter and antennas. For efficient utilization of available bandwidth and to obtain the required accuracy, three different modulation techniques are used in

10.5.1.2 (Continued)

each stage telemetry system. These three are: frequency modulation/frequency modulation (FM/FM), pulse code modulation/frequency modulation (PCM/FM), and single sideband/frequency modulation (SS/FM) which is employed in R and D flights only.

Telemetry data is radiated from the vehicle to ground stations in the VHF band (225-260 MHz). The PCM/FM System of the S-IVB stage and the IU are interconnected to provide a redundant transmission path and to make S-IVB measurements available to the LVDA. All flight control data is transmitted through the PCM/FM System.

The Telemetry System of each stage has a separate output via coaxial cable to the electronic support equipment, which is used with the digital data acquisition system for vehicle checkout before launch.

The IU command system permits data transmission from ground stations to the IU for insertion into the LVDC.

c. Tracking

The ODOP Transponder is located in the S-IC stage of the launch vehicles. The IU is equipped with two C-Band Radar Transponders and CCS Transponder (S-Band tracking).

d. Emergency Detection System

The Emergency Detection System (EDS) collects special measurements from each stage of the launch vehicle. Based on these measurements, critical states of the vehicle which may require mission abort are detected, and the information is sent to the Spacecraft for display and/or initiation of automatic abort.

10.5.1.3 J-2S/LOR Electrical Interface

The J-2S Improvement Study Launch Vehicle is identified to the basic Saturn V AS 511 configuration. The baseline vehicle is configured for Lunar Orbit Rendezvous mission. The electrical interface is shown in Figure 10.5.1.3-1.

10.5.1.4 J-2S/LOR Astrionic Subsystems

10.5.1.4.1 Navigation and Guidance

The hardware for the J-2S Guidance System will not require modification. The basic guidance philosophy as given in the Astrionic System Handbook will not be changed for J-2S. The flight program will require programming changes to guide

Ref. 10.4-34, Fig. 1-4

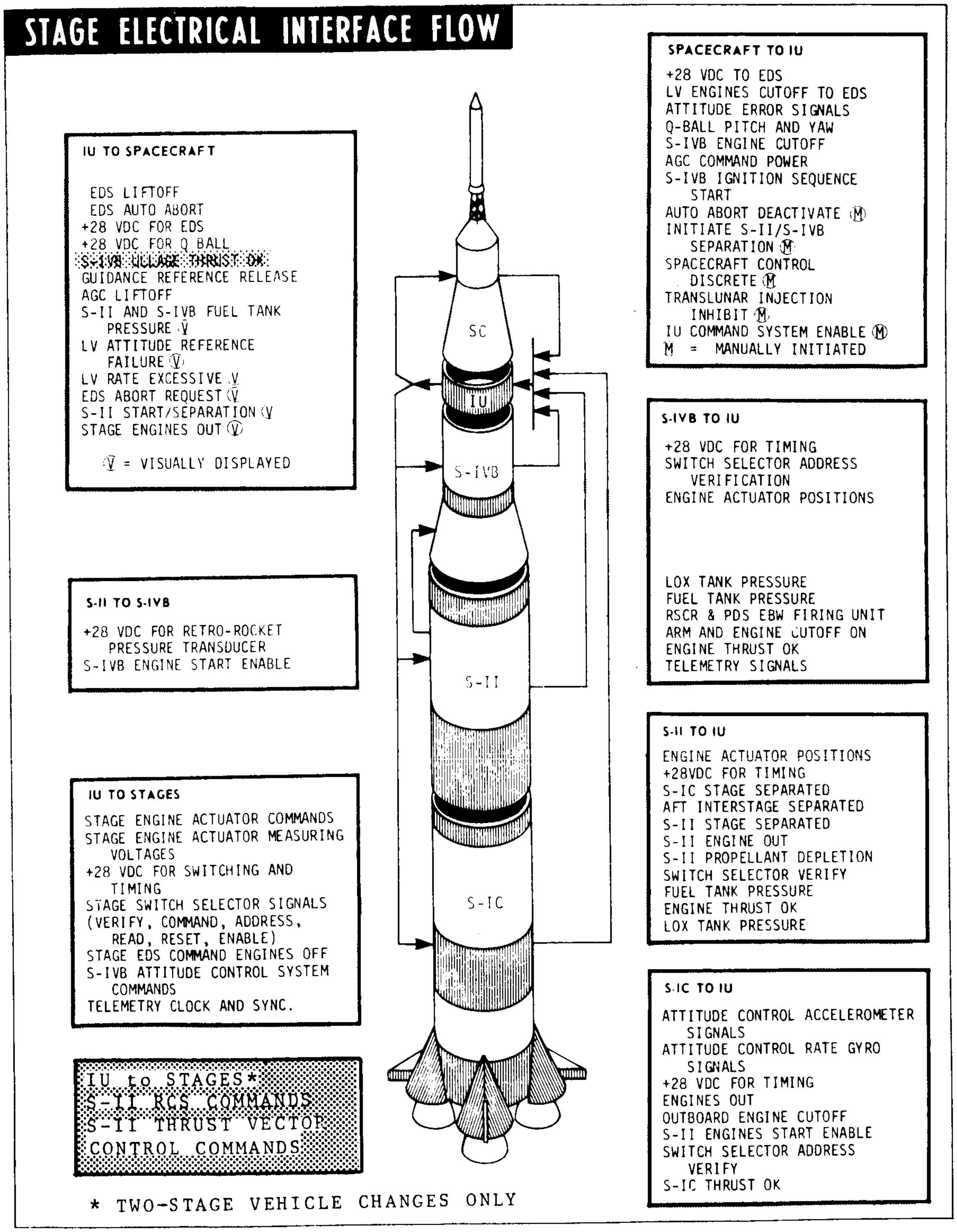


FIGURE 10.5.1.3-1. STAGE ELECTRICAL INTERFACE FLOW

10.5.1.4.1 (Continued)

the vehicle with the J-2S stages. Program sequence changes are discussed in more detail in paragraph 10.5.1.4.3 and paragraph 10.5.1.4.10.

10.5.1.4.2 Attitude Control

The Saturn V Vehicle Control System Analysis is discussed further utilizing vehicle simulation techniques in paragraph 10.5.1.4.10.

a. S-II Attitude Control

The S-II Attitude Control System will be identical to the J-2 configuration. Electrical control systems changes (deletion of LOX and Fuel Recirculation Systems) are discussed in paragraph 10.5.1.4.7.

b. S-IVB Attitude Control

The S-IVB Flight Control System incorporates two systems. During powered flight thrust vector steering is accomplished by gimbaling the J-2S engine for pitch and yaw control and by operating the APS engines for roll control. Steering during coast flight is by use of APS engines only. (Reference Figure 10.5.1-6).

The APS engines are located in two modules 180° apart on the aft skirt of the S-IVB stage (Reference Figure 10.5.1.4-2). Each module contains four engines; three 150-lb thrust control engines and one 70-lb thrust ullage engine. The APS modules provide three-axis stage attitude control and stage propellant-settling control. With the idle-mode capability of the J-2S engine, the 70-lb thrust ullage engines used for propellant control are no longer required. The engines are located in the Auxiliary Propulsion System as shown in Figure 10.5.1.4-3. Engine gimbaling is accomplished with hydraulic actuator assemblies connected to an engine-driven hydraulic pump. Implementation of the J-2S engine will necessitate a new pump assembly. A more detailed description of these changes may be found in paragraph 10.5.1.4.7.

c. IU Attitude Control

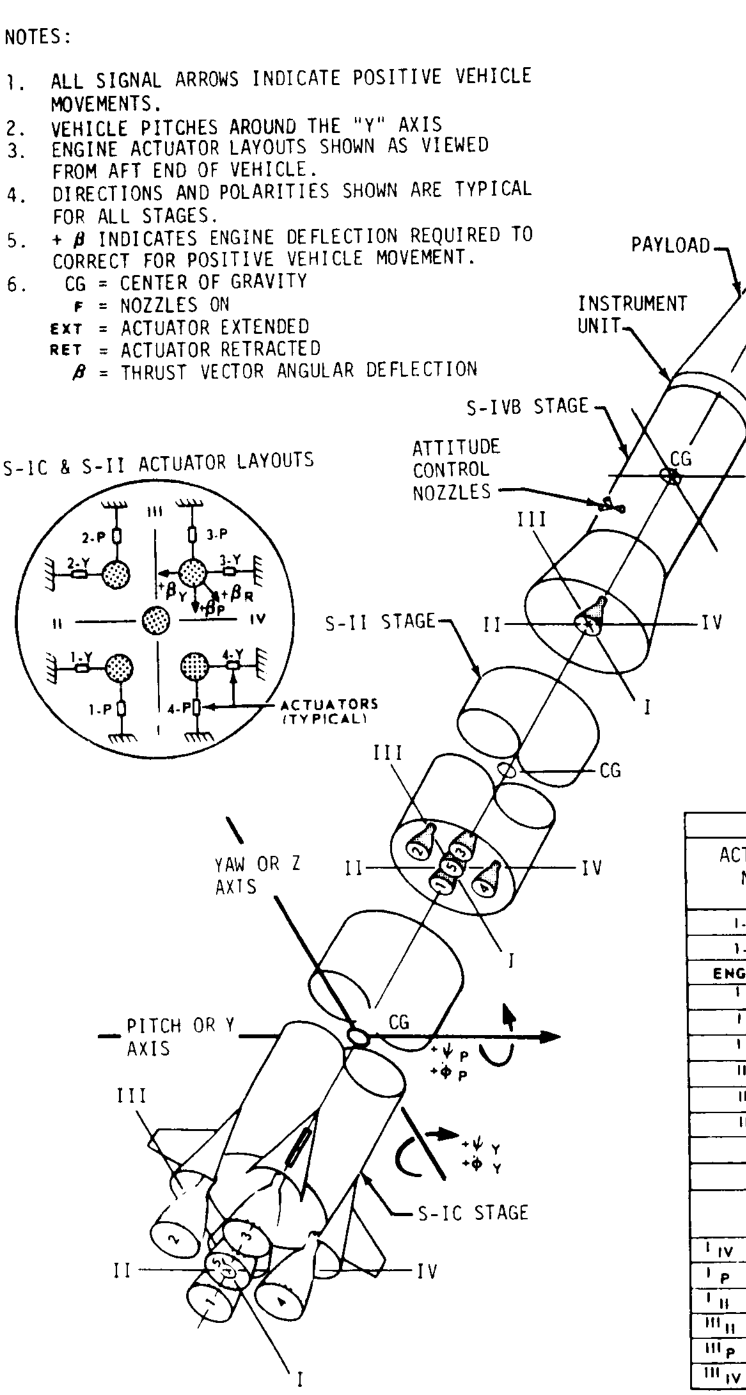
The IU Basic Attitude Control System will be identical to the J-2 configuration. The only flight control system hardware impact for the LOR mission are the changes required to include a reset capability into one of the two switch points in the S-IVB pitch and yaw attitude error and attitude rate filters. This switch point reset capability is necessary to implement a switch to higher control gains for idle mode. The incorporation of a resetable switch point into the Flight Control Computer for the S-IVB filters requires the following modifications (reference Figure 10.5.1.4-4.

Ref 10.4-32, Fig. 7-5

SATURN V ENGINES, ACTUATORS AND NOZZLE ARRANGEMENT

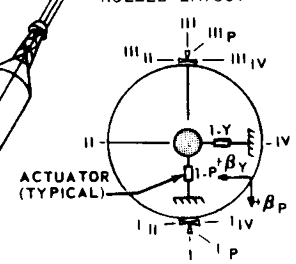
NOTES:

1. ALL SIGNAL ARROWS INDICATE POSITIVE VEHICLE MOVEMENTS.
2. VEHICLE PITCHES AROUND THE "Y" AXIS
3. ENGINE ACTUATOR LAYOUTS SHOWN AS VIEWED FROM AFT END OF VEHICLE.
4. DIRECTIONS AND POLARITIES SHOWN ARE TYPICAL FOR ALL STAGES.
5. + β INDICATES ENGINE DEFLECTION REQUIRED TO CORRECT FOR POSITIVE VEHICLE MOVEMENT.
6. CG = CENTER OF GRAVITY
   F = NOZZLES ON
   EXT = ACTUATOR EXTENDED
   RET = ACTUATOR RETRACTED
   β = THRUST VECTOR ANGULAR DEFLECTION



S-IC & S-II ACTUATOR LAYOUTS

S-IVB ACTUATOR AND NOZZLE LAYOUT



S-IC & S-II POLARITY TABLE

ACTUATOR NO.	ACTUATOR MOVEMENT $+\phi_R$	$+\phi_Y$	$+\phi_P$
1-Y	RET	RET	
1-P	EXT		RET
2-Y	EXT	RET	
2-P	RET		EXT
3-Y	RET	EXT	
3-P	EXT		EXT
4-P	EXT	EXT	
4-Y	RET		RET

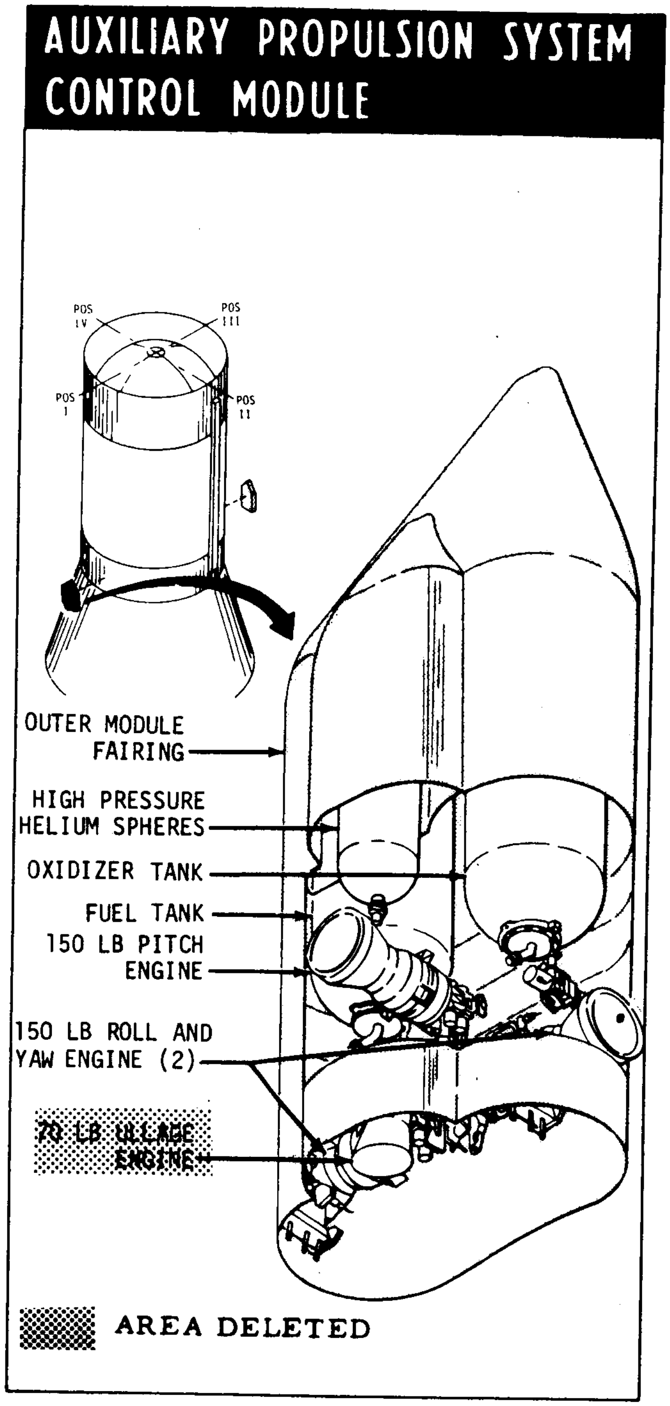
S-IVB POLARITY TABLE

ACTUATOR NO.	SIGNAL & ACTION $+\psi_R$	$-\psi_R$	$+\psi_Y$	$+\psi_P$
1-Y			EXT	
1-P				RET
ENGINE NO.				
I_{IV}		F		
I_P				
I_{II}	F			
III_{II}		F		
III_P				
III_{IV}	F			

CONDITIONS DURING COAST

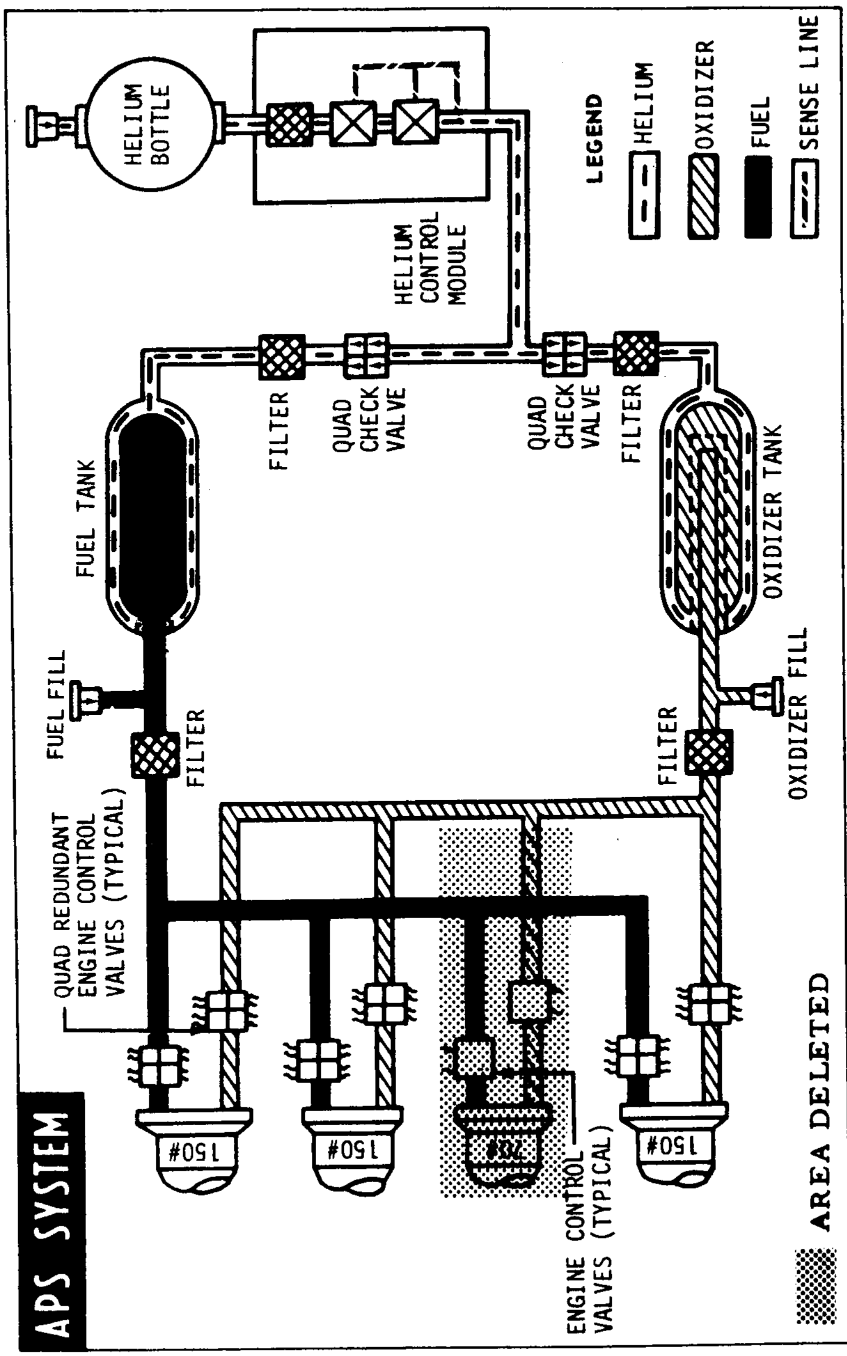
	$+\psi_R$	$-\psi_R$	$+\psi_Y$	$-\psi_Y$	$+\psi_P$	$-\psi_P$
I_{IV}		F		F		
I_P					F	
I_{II}	F		F			
III_{II}		F	F			
III_P						F
III_{IV}	F			F		

FIGURE 10.5.1.4-1. SATURN V ENGINES, ACTUATORS AND NOZZLE ARRANGEMENT



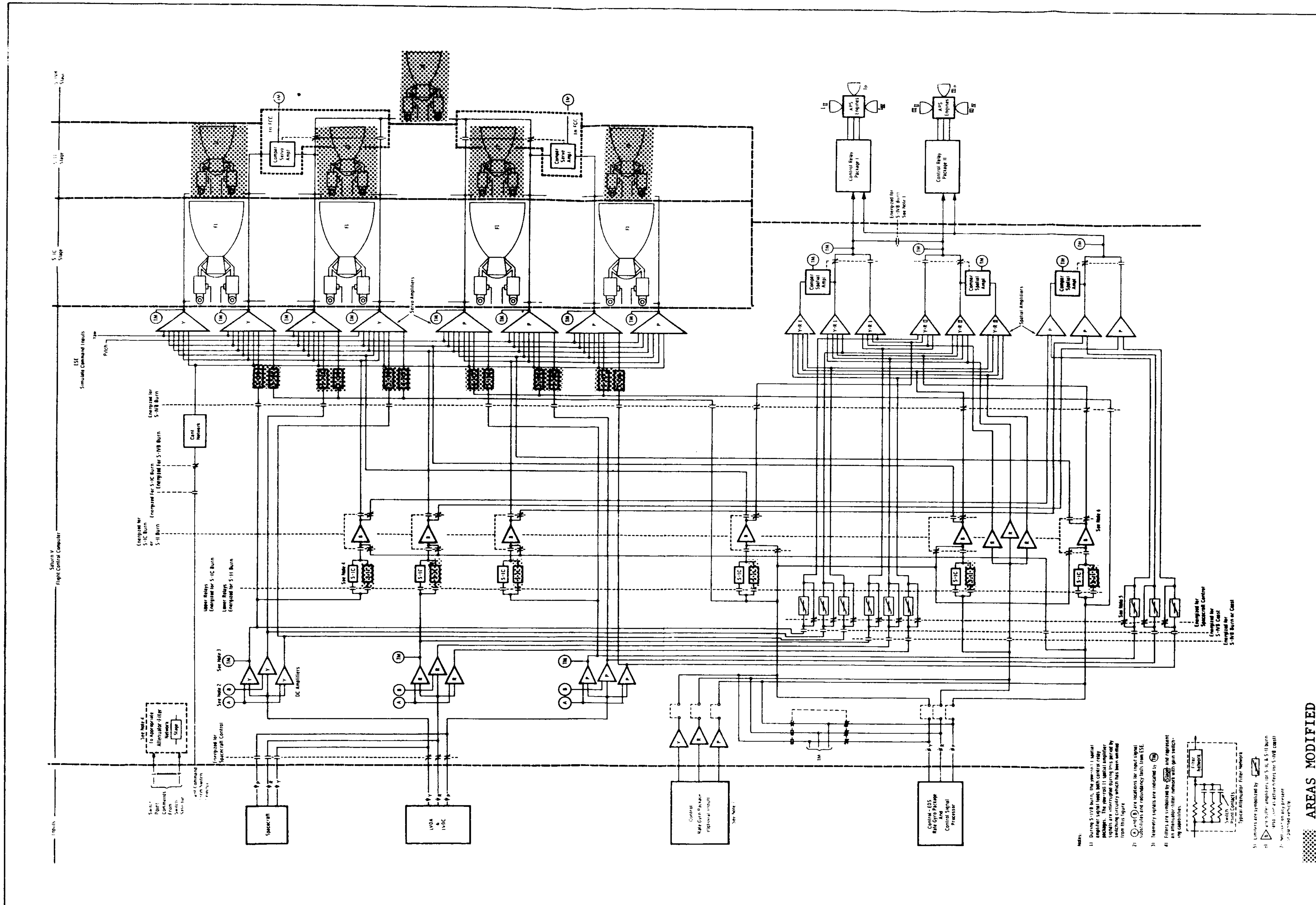
Ref 10.4-32, Fig. 6-14

FIGURE 10.5.1.4-2. AUXILIARY PROPULSION SYSTEM CONTROL MODULE



Ref 10.4-32, Fig. 6-17

FIGURE 10.5.1.4-3. APS SYSTEM



Ref 10.4-31, Fig. 3.3-1

FIGURE 10.5.1.4-4.
FLIGHT CONTROL COMPUTER

10.5.1.4.2 (Continued)

Redesign of the 12 S-IVB pitch and yaw attitude error and attitude rate filters printed circuit boards.

Redesign of motherboards #2, 3 and 4.

In-flight reset capability, which would permit setting or resetting the switch point more than one time during flight, for one switch point can be obtained by adding the following information:

Add one diode to control distributor to provide isolation between switch selector and GSE inputs to reset coils.

Add one wire to the cable from the switch selector to the control distributor to provide excitation of the reset coils.

Add one wire to the IU cables interfacing with the GSE through the umbilical to provide reset capability for the switch point.

10.5.1.4.3 Mode and Sequencing

Mode and sequence control involves most of the electrical/electronic systems in the Saturn V launch vehicle; however, in this section the discussion will deal mainly with the switch selector and associated circuitry (Figure 10.5.1.4-5).

The launch vehicle digital computer (LVDC) memory contains a predetermined number of sets of instructions which, when initiated, induce portions of the launch vehicle electrical/electronics systems to operate in a particular mode. Each mode consists of a predetermined sequence of events. The LVDC also generates appropriate discrete signals such as engine ignition, engine cutoff and stage separation.

Mode selection and initiation is accomplished through either an automatic LVDC internal command or through an external command from ground checkout equipment, IU command system, or from the flight crew in the spacecraft.

The flexibility of the mode and sequence control scheme is such that no hardware modification is required for mode and flight sequence changes. The changes are accomplished by changing the instructions and programs in the LVDC memory.

Many of the sequential operations in the launch vehicle that are controlled by the LVDC are performed through a switch selector located in each stage. The switch selector decodes digital flight sequence commands from the LVDA/LVDC and activates the proper stage circuits to execute the commands. The outputs of the switch selector drive relays either in the units affected or in the stage sequencer.

Ref 10.4-32, Fig. 7-20

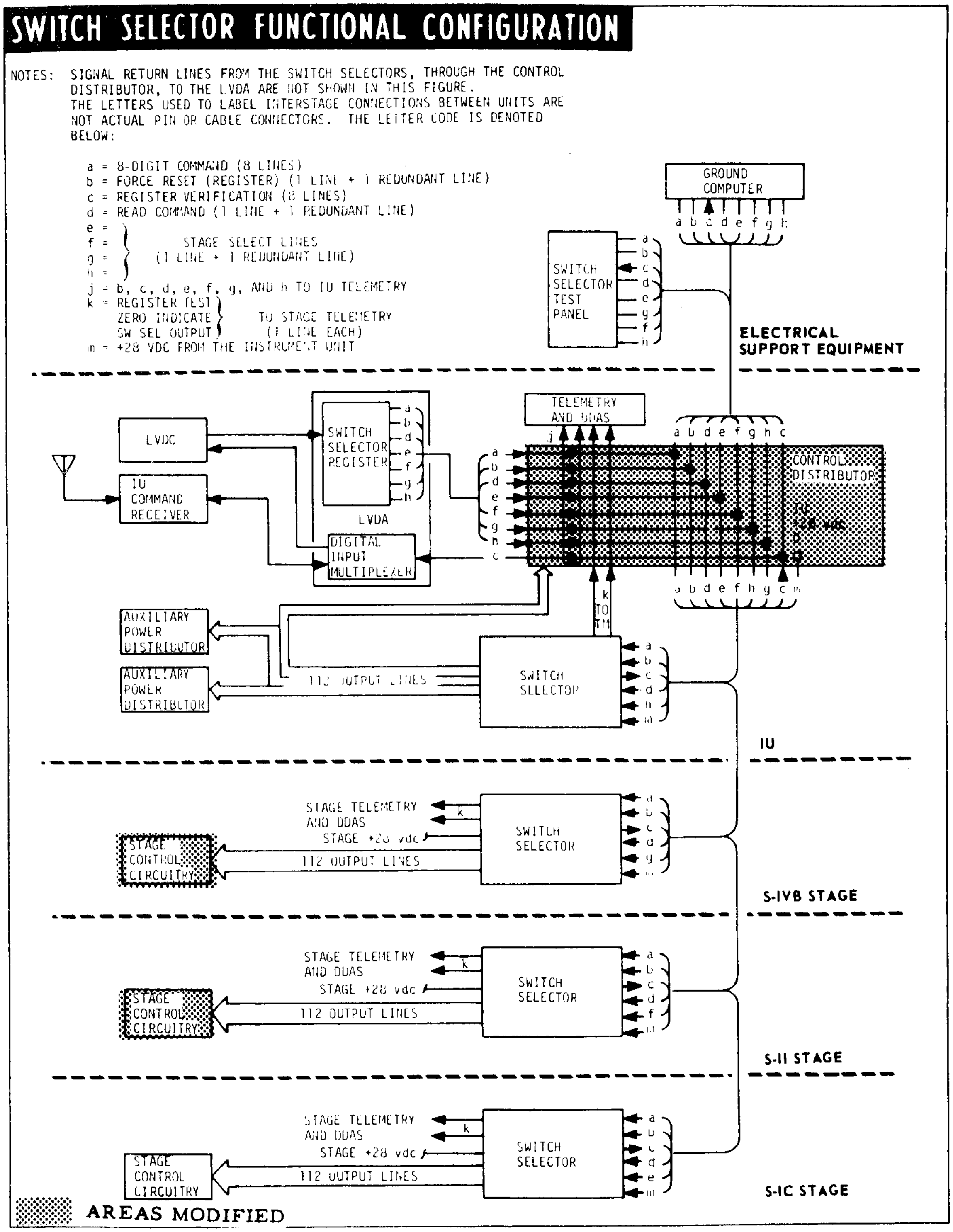


FIGURE 10.5.1.4-5. SWITCH SELECTOR FUNCTIONAL CONFIGURATION

10.5.1.4.3 (Continued)

a. S-II Mode and Sequencing

The incorporation of the J-2S engines on the S-II stage requires a modification to the Electrical Control System, mainly to provide stage electrical control for the new J-2S engine solid propellant turbine spinner (SPTS) system.

The J-2S engine requires changes to the switch selector commands for the LOR mission as follows:

Deleted S-II Switch Selector Commands

"LH_2 Recirculation Pumps Off"

"Ullage Trigger"

"Chilldown Valves Close"

"LOX Depletion Sensors Cutoff Arm"

New S-II Switch Selector Commands

"Prevalves Close Arm Reset"

"All Engines Start No. 2"

"Mainstage Start No. 1"

"Mainstage Start No. 2"

Revised Title

"All Engine Start No. 1" WAS "Engines Start"

The J-2S engine will require an "engine ready" bypass signal from the stage prior to engine start. The stage will provide signals from engine ready bypass bus No. 1 and No. 2 when commanded through the switch selector. One additional relay will be required to incorporate this change. These relays are reset by the engine control reset bus and the S-II engine start enable bus.

J-2S engine start will require an "all engine start No. 1" command followed by an "all engine start No. 2" command from the switch selector. Each command is capable of initiating engine start and two relays will be required to implement these functions. These engine start control relays are reset by the engine control relay reset bus and the S-II engine start enable bus. The S-II engine start enable bus is hot until S-IC separation during flight.

10.5.1.4.3 (Continued)

J-2S engine mainstage mode (following engine start) will require "mainstage start No. 1" command followed by "mainstage start No. 2" command from the switch selector. Each command is capable of initiating mainstage operation. Two relays are required to implement these functions. These relays are reset by the engine control reset bus and the S-II engine start enable bus.

The in-flight LOX depletion engine cutoff arm command will be provided by a signal from two of five sensors in the LOX tank without a switch selector arm command. Five relays will be required to implement this change. A hardware LOX depletion arm indication will be provided through an umbilical connector for ground checkout.

The one-engine-out capability during mainstage operation will be retained. An additional feature is required in the case of an early engine out to prevent that engine from cutting off the remaining four engines at LOX exhaustion cutoff arm (2 out of 5 dry sensors).

Assuming engine No. 1 terminates prematurely during mainstage operation, stage circuitry in conjunction with a 430 millisecond prevalve timer for engine No. 1 will prevent the other four engines from being terminated prematurely by the 2 out of 5 LOX dry sensor signal. The remaining four engines will continue to burn until LOX exhaustion produces an internal engine cutoff signal (from one of the four remaining engines). The signal will be transmitted to cut off the remaining three engines before the prevalve timer, for that engine, can run out. The normal external switch selector command and emergency engine cutoff circuitry remain unchanged. Five relays will be required to implement this change.

The reason the additional electrical control relays are required to incorporate the J-2S engine for the LOR mission is because of the redundant idle mode start, redundant mainstage start, mainstage operations, LOX exhaustion PC engine cutoff and redundant engine ready bypass functions.

b. S-IVB Mode and Sequence

Incorporation of the J-2S engine on the S-IVB stage will require the deletion of twenty-three switch selector channels, with six being added to the switch selector for engine functions. These channel assignments are required to operate the J-2S/S-IVB vehicle in LOR modes, considering the following hardware modifications: (1) the deletion of the two chilldown inverters and associated controls; (2) deletion of the four EBW firing units, four EBW pulse sensors, and controls associated with the solid ullage rocket motors; (3) deletion of the LOX depletion sensors, control units; (4) depletion of the control circuits for the LOX and LH_2 chilldown valves and prevalves, including the prevalve delay timer module;

10.5.1.4.3 (Continued)

(5) deletion of the 75-lb ullage engine and associated controls; and (6) addition of a solid propellant turbine starter unit for each engine start. In addition, the engine pump purge has been deleted and the LOX dome purge has been delegated to the GSE. The start bottle and gas generator commands have been deleted through the use of SPTS to start the engine.

The auxiliary hydraulic pump will be activated by switch selector commands from the IU at predetermined times.

The list of deleted switch selector channel assignments are found in Table 10.5.1.4-I with a list of added channel assignments in Table 10.5.1.4-II.

c. IU Mode and Sequence

Incorporation of the J-2S engine on the Saturn V vehicle will require additional switch selectors to reset switch points in the Flight Control Computer. One diode must be added to the control distributor and one cable from the switch selector to the control distributor must be added to accomplish the resetable switch points. The flight sequence associated with the J-2S/LOR mission is shown in Table 10.5.1.4-III.

10.5.1.4.4 Telemetry and Measurement

The purpose of the measuring systems is to detect the phenomena to be measured and to process and distribute this data to the input of each stage telemetry system. All measurements, regardless of their original characteristics, must be processed into electrical signals within a 0 to 5-volt range prior to delivery to the stage telemetry system. The telemetry system accepts these input signals for transmission to the ground recovery stations.

The Telemetry System for each stage of the vehicle must accept signals produced by the measuring portion of the instrumentation system, and accurately reproduce and transmit them to the ground stations. Measurement signals are accepted at a fixed input level, processed, and fed to the proper airborne antennas. In the case of checkout measurements, the signals are transmitted via breakaway cable arrangement to the ground checkout station prior to lift-off. Saturn vehicle telemetry subsystems are shown in Reference 10.4-32.

a. S-II Telemetry and Measurement

The changes, listed in this paragraph, are based on the no-change, S-II-11 Instrumentation Program and Components List, dated January 5, 1968. Approved S-II design changes, subsequent to this date, may alter the information contained in this paragraph.

TABLE 10.5.1.4-I. S-IVB SWITCH SELECTOR CHANNEL ASSIGNMENTS DELETED

Channel 58	Fuel Chilldown Pump ON
59	Fuel Chilldown Pump OFF
22	LOX Chilldown Pump ON
23	LOX Chilldown Pump OFF
91	Chilldown Shutoff Pilot Valve ON
92	Chilldown Shutoff Pilot Valve OFF
82	Prevalve Close Command ON
83	Prevalve Close Command OFF
24	Engine Pump Purge Control Valve Command ON
25	Engine Pump Purge Control Valve Command OFF
1	Start Tank Vent Valve Open ON
2	Start Tank Vent Valve Open OFF
42	70 lbs. Ullage Engine Command No. 1 ON
43	70 lbs. Ullage Engine Command No. 1 OFF
101	70 lbs. Ullage Engine Command No. 2 ON
102	70 lbs. Ullage Engine Command No. 2 OFF
55	Charge Ullage Jettison ON
88	Ullage Charging Command Reset
57	Fire Ullage Jettison ON
54	Charge Ullage Ignition ON
56	Fire Ullage Ignition ON
73	Firing Ullage Reset
11	Fuel Injection Temp. OK Bypass (Main Stage Enable)

TABLE 10.5.1.4-II. S-IVB SWITCH SELECTOR CHANNEL ASSIGNMENT ADDED

1.	Mainstage Cutoff ON
2.	Mainstage Cutoff OFF
3.	Mainstage OK Bypass ON
4.	Mainstage OK Bypass OFF
5.	Mainstage Start ON
6.	Mainstage Start OFF

TABLE 10.5.1.4-III. APPROXIMATE MISSION TIME LINE LOR MISSION

LOR			LOR - J-2S		
Time Base	Time (Seconds)	Event	Time Base	Time (Seconds)	Event
TB0	-17	Guidance Reference Release	TB0	-17	Guidance Reference Release
TB1	0	Liftoff	TB1	0	Liftoff
	85	Q Max		85	Q Max
TB2	149	Center Engine Cutoff	TB2	149	Center Engine Cutoff
	153	Start Chi Freeze		153	Start Chi Freeze
TB3	161	S-IC Cutoff	TB3	161	S-IC Cutoff
	165	S-II Ignition		165	S-II Ignition (idle)
	167	Shift MR to 5.5		166	S-II Mainstage
	193	Interstage Separation		168	S-II MR Shift (5 - 5.5)
	198	LES Separation		193	Interstage Separation
	426	MR Shift to 4.7		198	LES Separation (end chi freeze)
				396	S-II MR Shift (5.5 - 4.7)
TB4	543	S-II Cutoff	TB4	498	S-II Cutoff
	543	S-IVB Ignition		498	S-IVB Ignition (idle)
				499	S-IVB Mainstage
TB5	676	S-IVB Cutoff	TB5	617	S-IVB Cutoff (1st burn)
				3 Rev ~4.5 hrs	Parking Orbit Coast
TB6	2906	Restart Preparation	TB6	3317	S-IVB (idle)
	3476	S-IVB Ignition		3417	S-IVB Mainstage
TB7	3793	S-IVB Cutoff	TB7	3708	S-IVB Cutoff (2nd burn)

10.5.1.4.4 (Continued)

Tape recorders may be required during communication blackout at engine start. A major telemetry design and development impact would be required if this became necessary. This effort is not defined in this proposal.

Because of the J-2S engine thrust chamber dimensions, it is considered that the cameras, utilized for S-II-1 and S-II-2 interstage separation, are desirable but not mandatory for the LOR mission.

Hardware changes for the instrumentation system consist mainly of wiring harness and transducer modifications. Table 10.5.1.4-IV reflects the changes in types of measurements affected by the J-2S incorporation.

Sufficient telemeter channels are available to accommodate the additional measurements. Changes to the wire harnesses, associated with all measurements, as well as the installation drawings for the new measurements, will be required.

b. S-IVB Measurement and Telemetry

A comparison of the applicable telemetry for J-2/S-IVB and J-2S/S-IVB indicates that 24 measurements at engine interface remain the same. Functionally, 9 engine measurements remain the same; however, the measurement range is different from the J-2. Total vehicle deletions number 54, 17 are at the engine interface with the remainder in the vehicle. Nine additional measurements will be monitored from the stage. One of these measurements, C8, was flown on earlier vehicles but is not presently telemetered from vehicle 509. The remainder are engine associated measurements such as commands and talkbacks which are picked up in distributors. The additional telemetry requirements at the engine interface fall into two categories as follows: (1) those that are presently wired to the interface and (2) those which are not presently wired to the interface, most of which have been requested by McDonnell Douglas Astronautics Company. It is assumed that transducers and wiring to implement these requested measurements can be installed on the engine by Rocketdyne. The deletions occur primarily in three areas as follows: (1) ullage rocket and jettison, (2) peculiar J-2 engine measurement, and (3) chilldown system. The additions are primarily on the J-2S engine.

Since the net change in terms of numbers of parameters is only 8 measurements less for the J-2S (54 deletions and 46 additions), it will not be necessary to change the present multiplexing and RF transmission system provided the sampling rates remain approximately the same as they are presently. However, as a result of these changes, it will be necessary to change the signal conditioning rack module complement, revise cable drawings, cable network drawings, schematics, IP&CL's, and installation drawings.

TABLE 10.5.1.4-IV. STAGE MEASUREMENTS SUMMARY

TYPE	QUANTITY
Acceleration	4
Acoustic	--
Discrete Signals	157
Flowrate	10
Liquid Level	4
Miscellaneous	4
Position	36
Pressure	105
RPM	10
Strain	--
Temperature	105
Vibration	--
Voltage, Current, Frequency	33

10.5.1.4.4 (Continued)

Virtually all the additional measurement points are on the engine; therefore, transducer component testing will be small. However, new amplifiers, temperature bridges, and other networks for the additional measurements must be designed and tested if present components are not acceptable.

c. IU Telemetry and Measurement

Incorporation of the J-2S engine on the Saturn V vehicle will have no affect on the IU Telemetry and Measurement System.

10.5.1.4.5 Radio Command System

The Saturn V Vehicle Command system consists of two major functions, (1) Range Safety system for S-IC, S-II and S-IVB stages and (2) Command Communications system for the Instrument Unit.

a. S-II Radio Command System

The function of the radio command system in the S-II stage is to provide a means of terminating the flight via coded commands from ground stations. Incorporation of the J-2S engine will require no modification to the S-II radio command system.

b. S-IVB Radio Command System

The S-IVB radio command system serves the same function as the S-II and is thus not impacted by the J-2S incorporation.

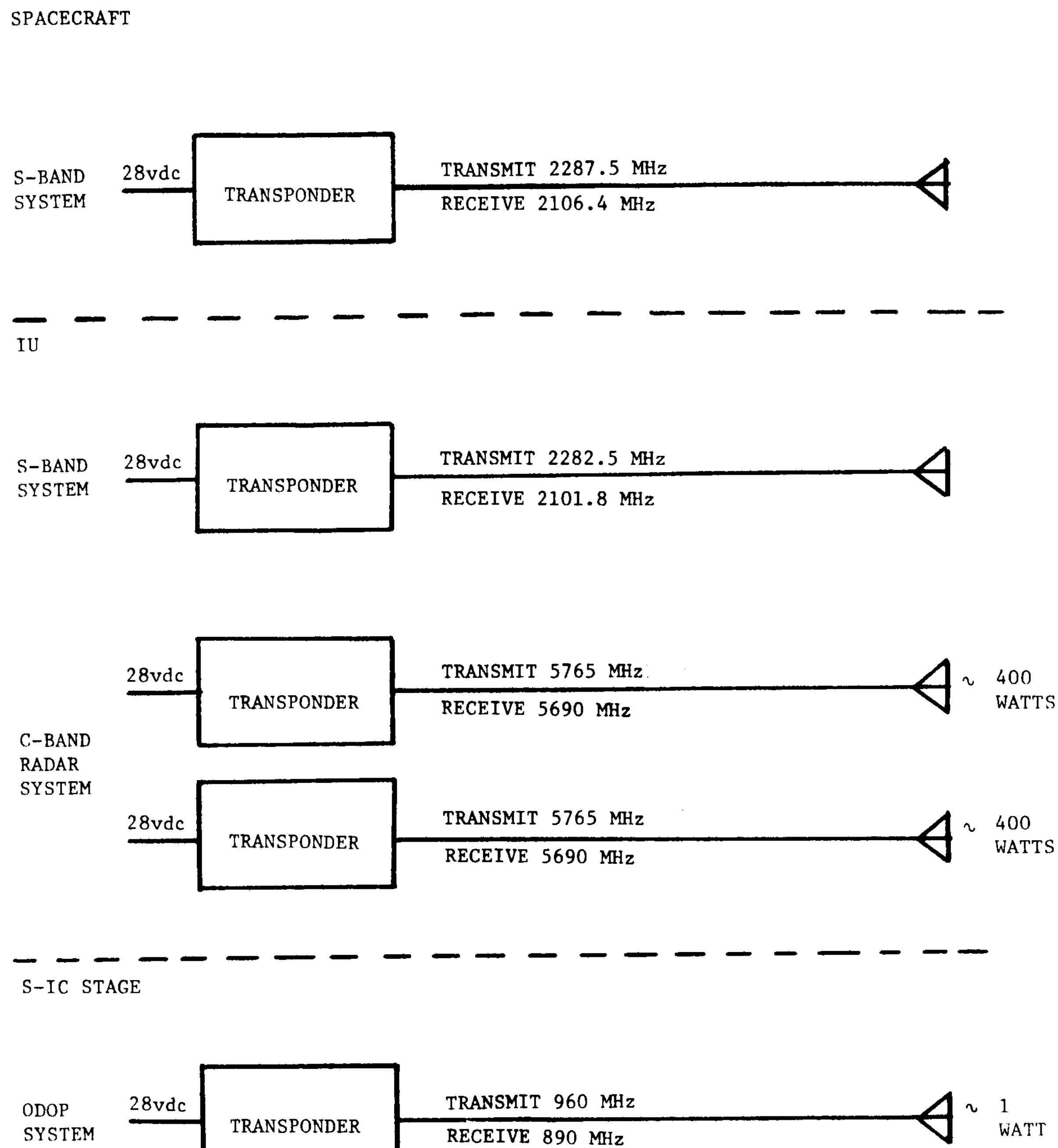
c. IU Radio Command System

The Command Communication System (CCS) provides for digital data transmission from the ground station to the LVDC. This communication link is used to update functions through the LVDC. Incorporation of the J-2S engine on the Saturn Launch vehicle will require no modifications to the CCS.

10.5.1.4.6 Tracking

The Saturn Launch Vehicle Tracking system will not change with J-2S configuration. A description of the system for AS 511 is given in Figure 10.5.1.4-6.

In the Saturn V space vehicle there is a continuous requirement to transmit information to ground stations in order to track the vehicle. This requirement is filled by the RF systems.



Ref 10.4-32, Fig. 13

FIGURE 10.5.1.4-6. SATURN VEHICLE TRACKING

10.5.1.4.6 (Continued)

The RF system functions to transmit (via RF carrier) all vehicle flight evaluation data as well as to evaluate vehicle performance (flight path) for ground receiving stations. These functions are accomplished through the use of Antenna and Tracking systems.

The principal tracking systems used are:

ODOP (offset doppler) system - used in the S-IC stage.

C-band radar - used in the IU.

S-band - used in Spacecraft and IU.

a. ODOP System (S-IC)

An offset doppler, frequency measurement system is an elliptical tracking system which measures the total doppler phase shift in a ultra-high frequency (UHF) continuous wave (CW) signal transmitted to the S-IC stage. The system uses a fixed station (ground) transmitter, a vehicle-borne transponder and three or more fixed station (ground) receivers.

b. C-Band (IU)

C-band is a pulse radar system which is used for precise tracking during launch and orbit phases. Two C-band radar transponders carried in the IU provide radar tracking capabilities independent of vehicle attitude.

c. S-Band System (SC and IU)

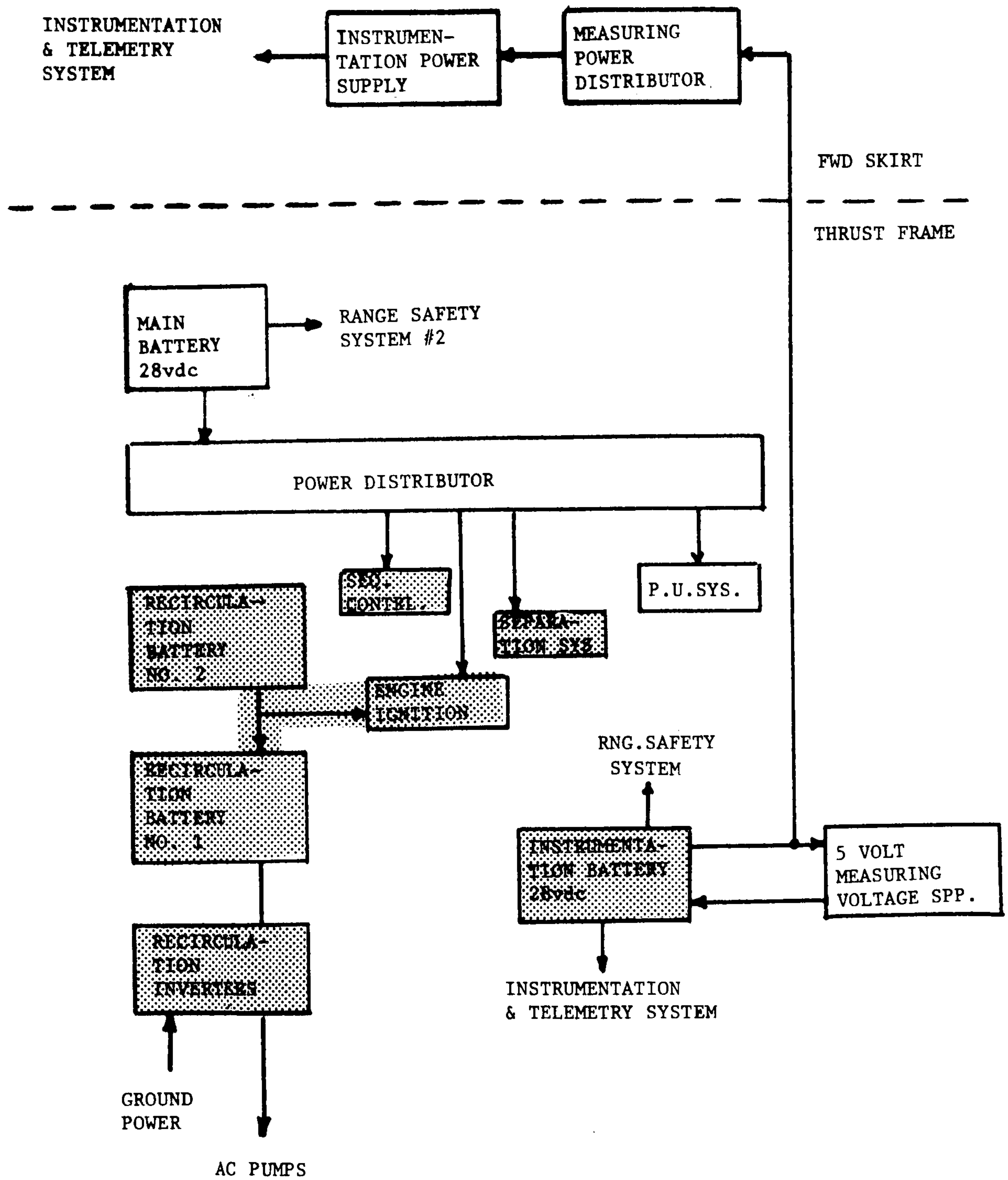
The S-band system provides tracking capability to the unified S-Band (USB) ground stations.

10.5.1.4.7 Power Supply and Distribution

a. S-II Power Supply and Distribution

Incorporation of the J-2S engine on the S-II stage for the LOR mission results in deletion of the LOX and LH_2 recirculation systems and the ullage motors. Since the LH_2 recirculation batteries are deleted (reference Figure 10.5.1.4-7) ignition power must be provided by the main battery.

Each of the five J-2S engines contains a solid propellant turbine spinner (SPTS) that is used in starting the engine. Each engine contains two SPTS ordnance initiating chains, consisting of an ordnance initiator device that is fired by a high energy pulse from an EBW firing unit. Both ordnance initiators are connected to the



Ref. 10.4-34, Fig. 29

FIGURE 10.5.1.4-7. S-II POWER AND DISTRIBUTION

10.5.1.4.7 (Continued)

solid propellant turbine spinner. The main bus shall provide power for each engine, including the ignition and SPTS devices. Dual pulse sensors are installed on the engine to be used in place of the ordnance initiators for checkout. The dual pulse sensors will be powered by the engine bus.

The main bus is required to provide the J-2S ignition load because deletion of the LH_2 recirculation batteries removes the power source presently used for engine ignition. The engine ignition load shall be added to the main bus and fifteen components shall be deleted from the Electrical Power system. The telemetry and hardwire 28 VDC supplied by the instrumentation bus for the five fuel recirculation pump valve position indicator switches shall be deleted.

The fuel recirculation system is no longer required because the J-2S engine requires no pre-conditioning prior to engine start.

The ullage motor ignition EBW Firing Units 1A and 1B are not required because the ullage motors are deleted from the S-II stage. The 28 VDC power required to operate the EBW Firing Units shall be deleted from the main and instrumentation bus. The 28 VDC power required to operate the associated pulse sensors shall be deleted from the main bus.

The ullage motors are deleted because the J-2S engine starts in a mode that provides sufficient thrust for propellant settling prior to entering the mainstage operation.

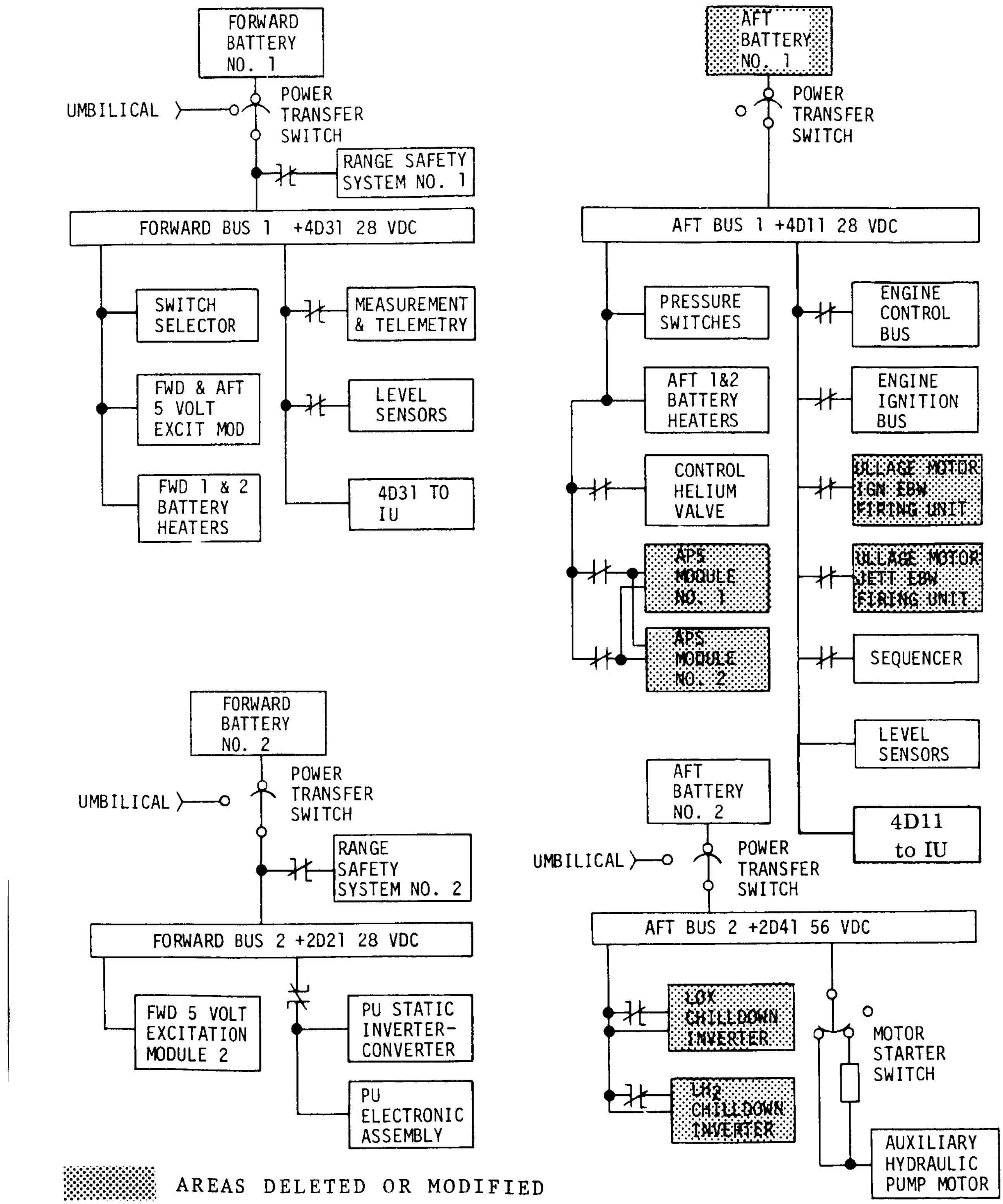
b. S-IVB Power Supply and Distribution

The impact of the J-2S engine on the power system of the J-2/LOR vehicle is minimal. Deletion of LOX and LH_2 chilldown inverters and ullage motors are the most significant factors in redistribution of battery loading (Reference Figure 10.5.1.4-8). Both chilldown inverters were supplied from aft battery No. 2, thus reducing the percent of battery life utilization. Ullage motors and associated EBW Firing Units were removed from aft battery No. 1. Considerations are being made to reduce the capacity of S-IVB batteries due to J-2S implementation.

c. IU Power Supply and Distribution

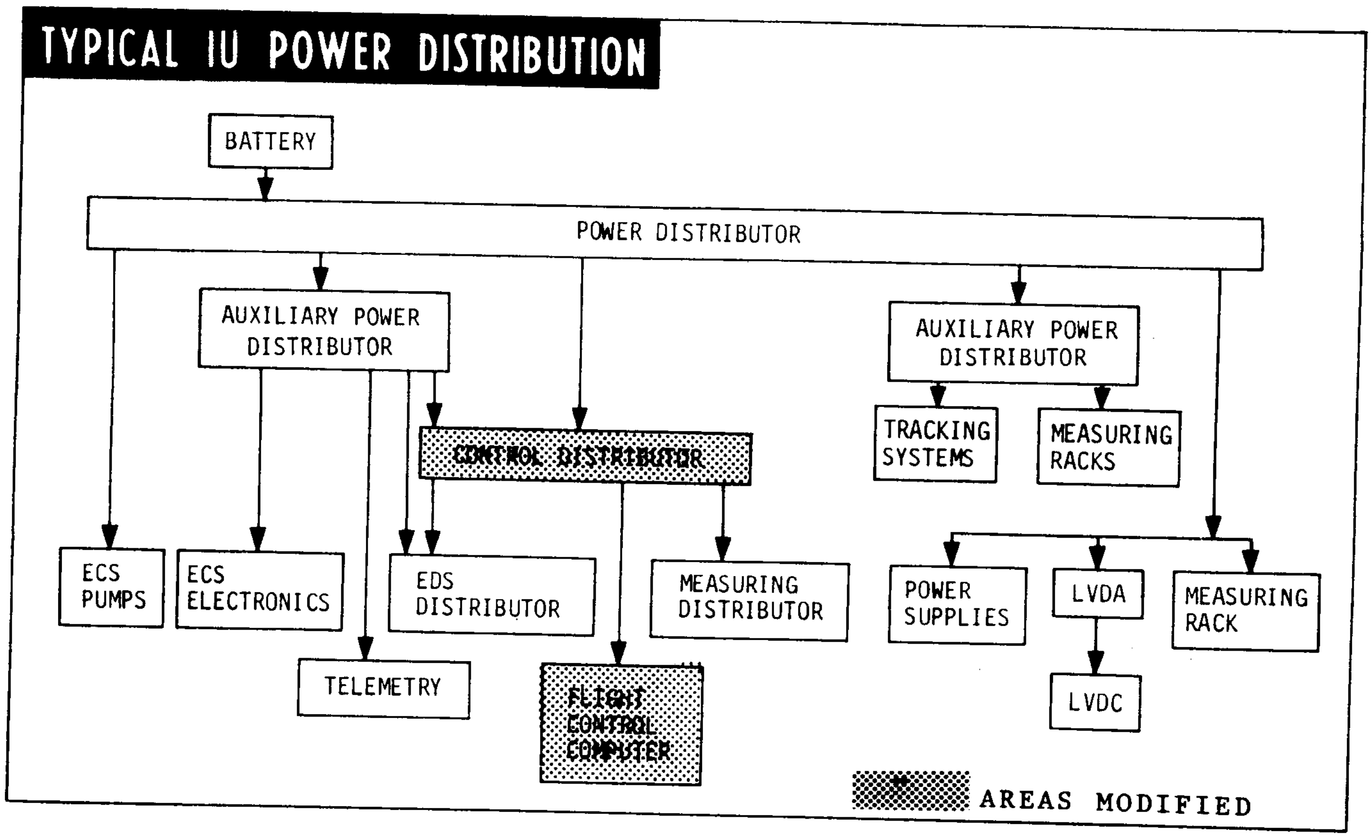
Incorporation of the J-2S engine on the Saturn V vehicle will not affect power distribution in the IU. Typical IU Power Distribution is shown in Figure 10.5.1.4-9.

ELECTRICAL POWER DISTRIBUTION



Ref. 10.4-32, Fig. 6-21

FIGURE 10.5.1.4-8. ELECTRICAL POWER DISTRIBUTION



Ref. 10.4-32, Fig. 7-10

FIGURE 10.5.1.4-9. TYPICAL IU POWER DISTRIBUTION

10.5.1.4.8 Emergency Detection System

The EDS system for the Saturn AS 511 J-2S vehicle will remain the same as present AS 511 configuration described in Reference 10.4-31, Astrionic System Handbook. Figure 10.5.1.4-10 shows a block schematic of the AS 511 J-2S EDS.

The Emergency Detection System (EDS), which is a part of the Crew Safety System, is designed to sense and react to emergency situations resulting from launch vehicle malfunctions which may arise during the mission. Protection of the Apollo crew against vehicle failure is the prime function of the EDS.

In general, the abort modes for operation of the EDS are:

Manual Abort - based on Astronauts's judgement and decision.

Automatic Abort - is initiated by excessive angular rates of the vehicle or by the loss of thrust in two or more engines in the S-IC stage during specified times of flight. The measurements are obtained from triple redundant sensors with majority voting logic.

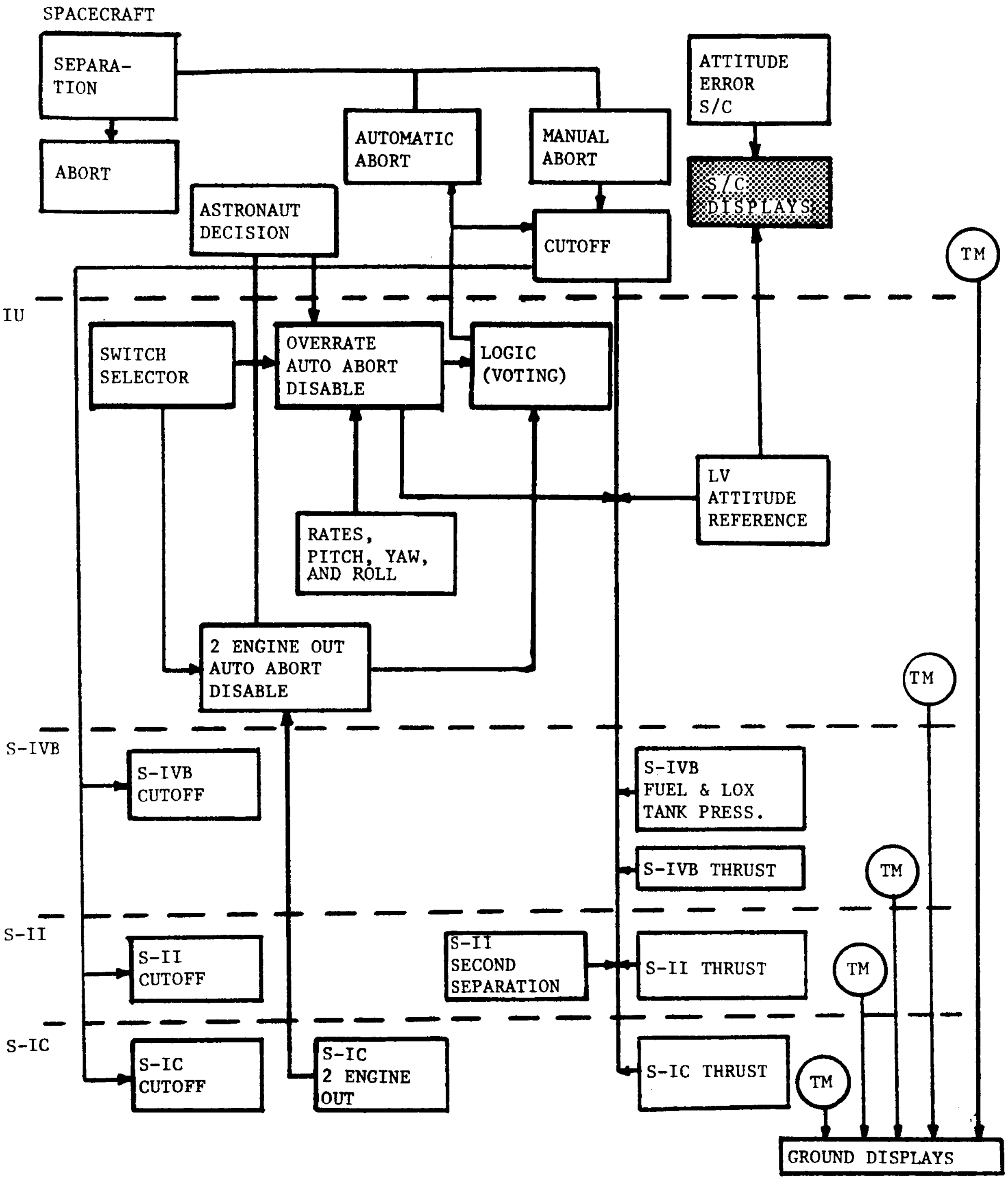
The automatic abort rate limits are: $\pm$ 4 degrees per second with a tolerance of $\pm$ .49 degrees in pitch and yaw and $\pm$ 20 degrees per second with tolerance of $\pm$ 1.5 degrees in roll.

Auto abort is automatically enabled at lift-off, provided the EDS-auto, LV-rates-auto and two-engine-out-auto switches are enabled in the spacecraft.

The automatic abort mode is active only during first stage flight from lift-off until the crew manually inhibits the automatic abort at approximately 120 seconds; therefore, an automatic abort always utilized the LES for escape (the LES is jettisoned shortly after S-II ignition by the crew).

In order to afford protection for personnel and facilities in the launch area, thrust is not terminated with aborts prior to 30 seconds of flight time. The switch selector enables the EDS cutoff circuitry at 30 seconds of flight with a timer back-up at 30 seconds.

The only EDS anomaly occurs during the idle mode of the J-2S engine. The J-2S in idle mode looks like an engine out to the "thrust OK" switches; therefore, the "thrust no OK" will be indicated in the spacecraft display. The S/C will not be able to differentiate between engine out and engine idling with present "thrust OK" lights. Correction of the problem would entail development of a more sensitive thrust OK switch which could be used in place of the present one or switched in for idle mode periods.



Ref. 10.4-34, Fig. 7

FIGURE 10.5.1.4-10. EMERGENCY DETECTION SYSTEM

10.5.1.4.8 (Continued)

The problem was not pursued further since it could also be solved with a procedure change in the spacecraft.

10.5.1.4.9 Separation

The Saturn V launch vehicle system provides for separation of an expended stage from the remainder of the vehicle. For S-IC/S-II separation, a dual plane separation technique is used wherein the structure between the two stages is severed at two different planes (Figure 10.5.1.4-11). The S-II/S-IVB separation occurs at a single plane. All separations are controlled by the launch vehicle digital computer (LVDC) located in the Instrument Unit (IU).

Stage separation sequencing includes a requirement to ensure stable flow of propellants into the engines. A small force is required to settle the propellants in their tanks prior to engine mainstage; on J-2/AS 511 vehicle this is accomplished via ullage motors on both S-II and S-IVB stages.

a. S-II Separation

With the capability of providing thrust for propellant settling prior to entering mainstage operation, the J-2S engine eliminates the requirement for ullage motors on the S-II stage. There shall be six components deleted from the S-II electrical control system as a result of stage separation modification. These components consist of four ullage motors and associated EBW firing units (reference Figure 10.5.1.4-12).

b. S-IVB Separation

For the same reason as the S-II stage separation, the S-IVB ullage rockets and associated hardware will be deleted. Deletion of the system on the S-IVB stage also includes a ullage rocket jettison system and associated hardware (reference Figure 10.5.1.4-13).

c. IU Separation

Separation is controlled by commands issued from the LVDC; however, no IU hardware will be impacted by the J-2S configuration.

10.5.1.4.10 Flight Program

a. Guidance Analysis

A flight program is defined as a set of instructions which controls the launch vehicle digital computer (LVDC) operation from seconds before lift-off until the

Ref. 10.4-32, Fig. 5-20

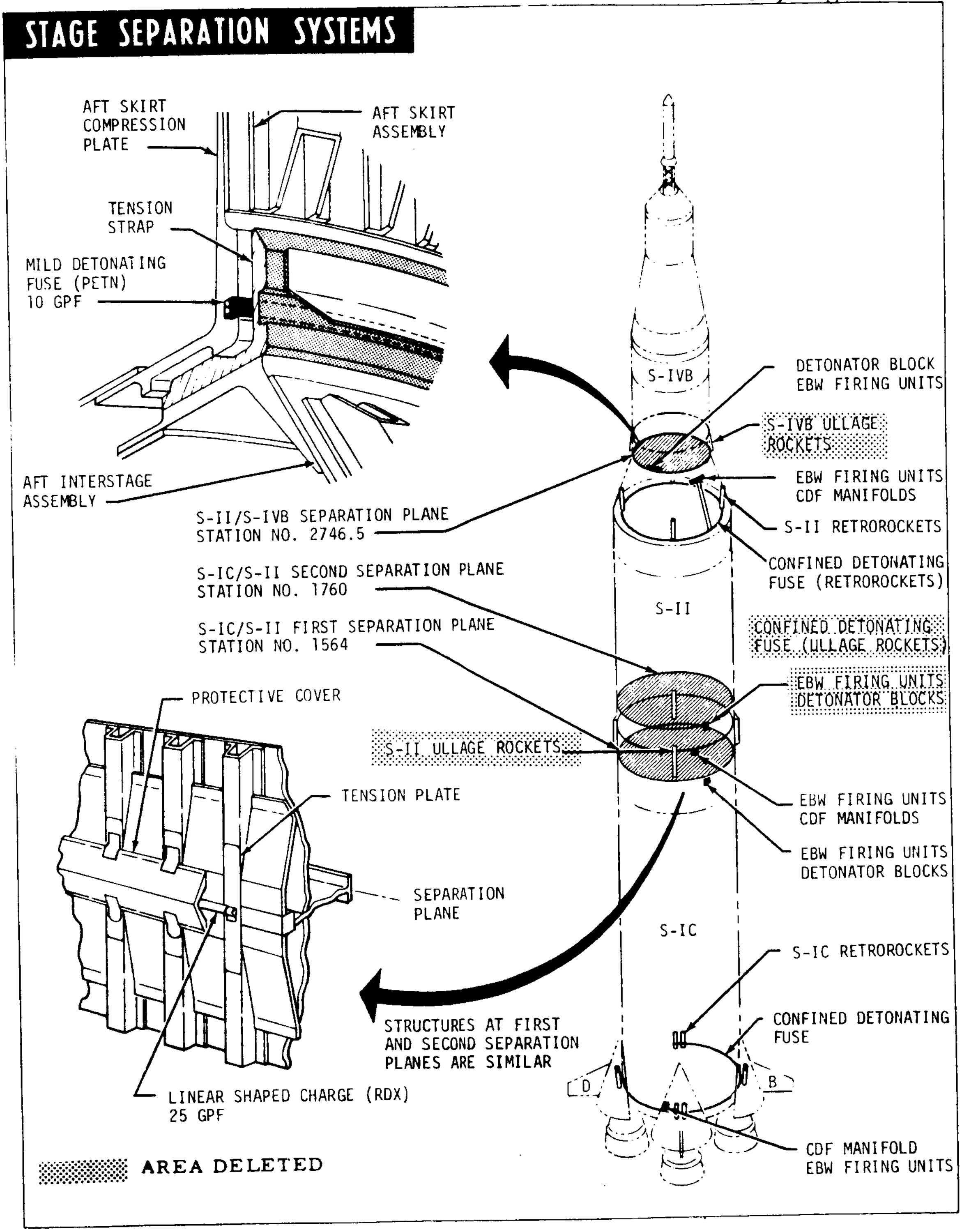
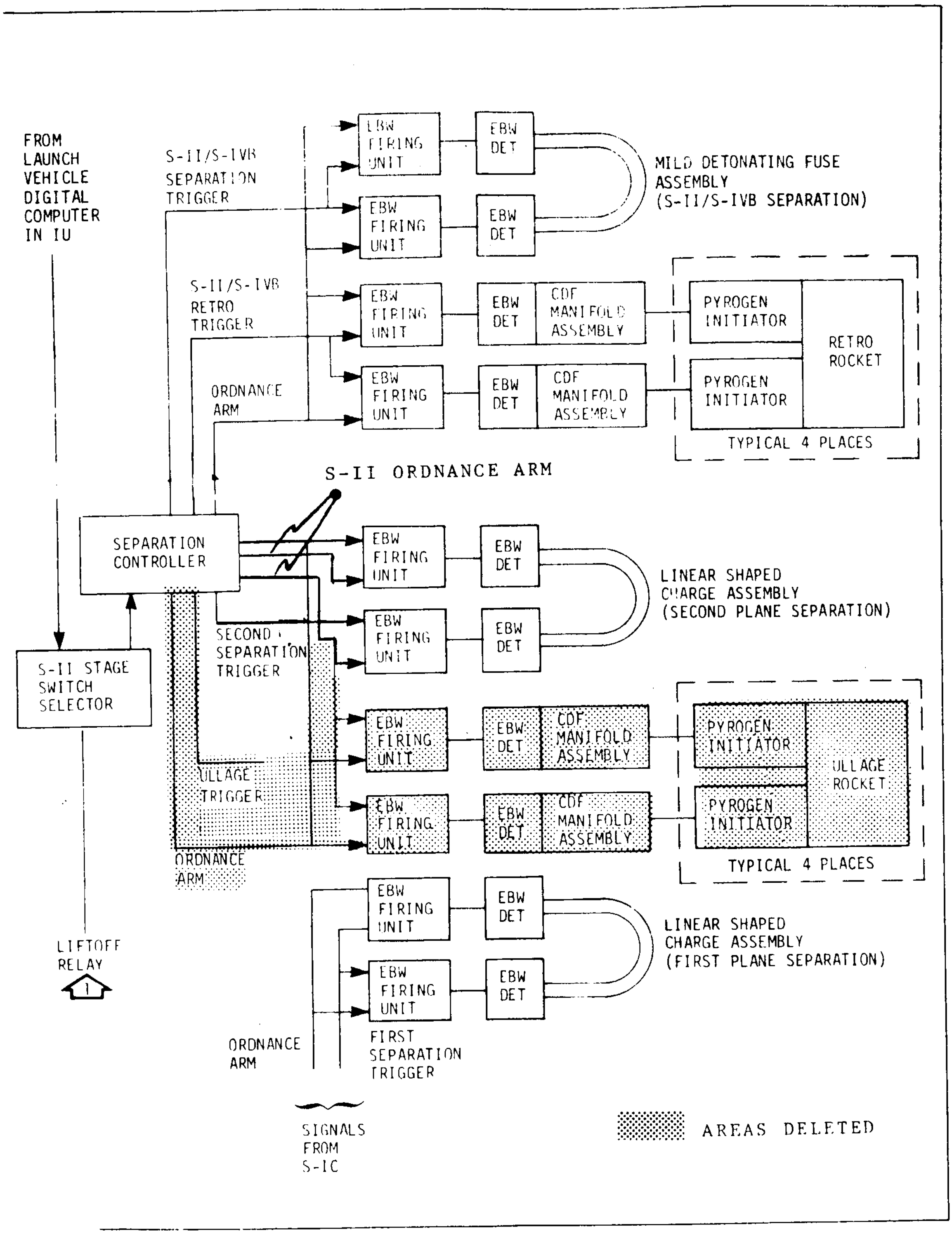
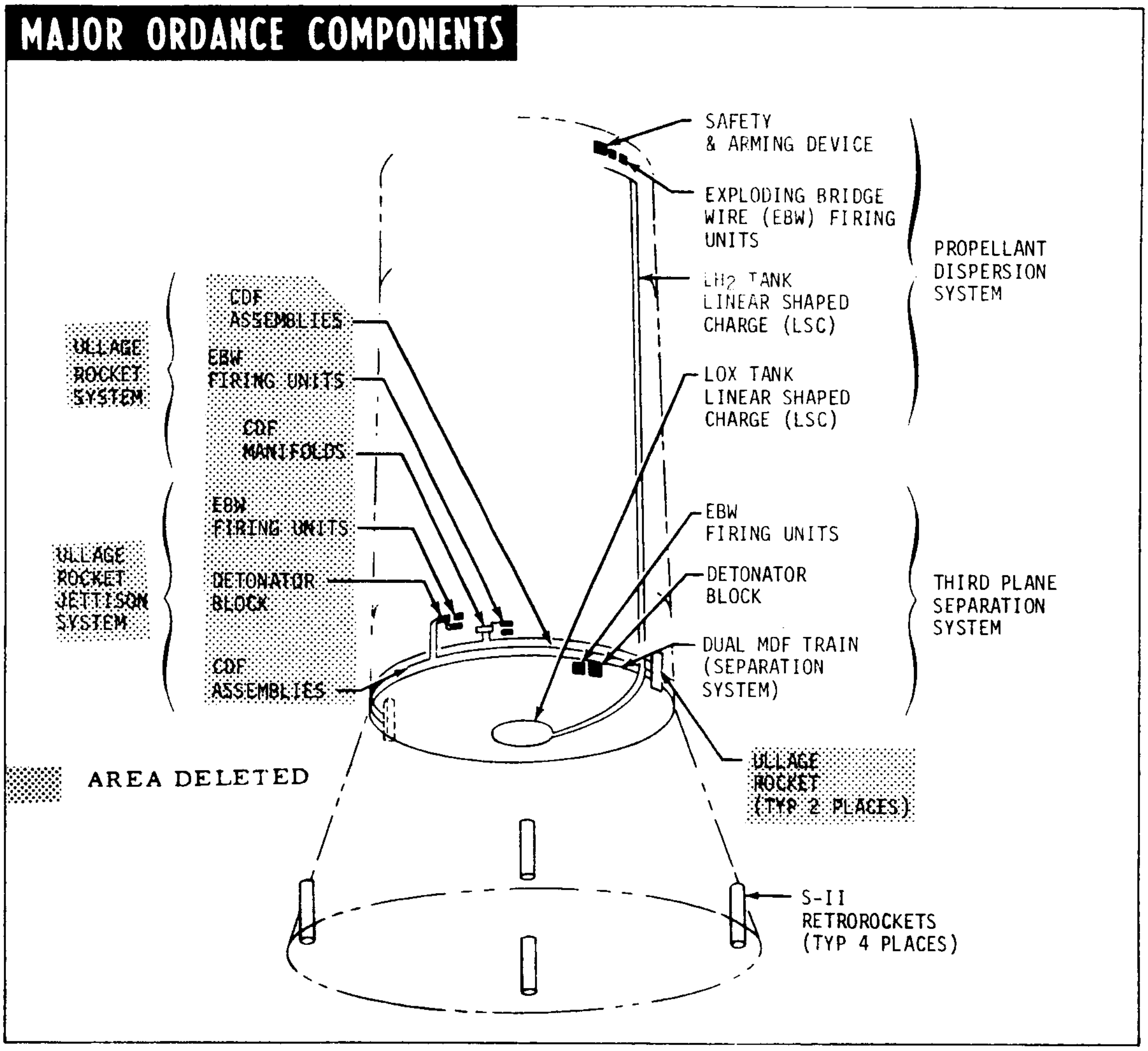


FIGURE 10.5.1.4-11. STAGE SEPARATION SYSTEMS



Ref. 10.4-32, Fig. 5-21

FIGURE 10.5.1.4-12. S-II ORDNANCE SYSTEM



Ref. 10.4-32, Fig. 6-26

FIGURE 10.5.1.4-13. S-IVB MAJOR ORDNANCE

10.5.1.4.10 (Continued)

end of the launch vehicle mission. These instructions are stored in memory within the LVDC.

The flight program performs many functions during the launch vehicle mission. These functions include: navigation, guidance, attitude control, event sequencing, data management, ground command processing and hardware evaluation. Specific definition of these functions depends on mission objectives.

For purposes of discussion the flight program is divided into five sub-elements. These are the powered flight major loop, the orbital flight program, the minor loop, interrupts and telemetry.

The powered flight major loop contains guidance and navigation calculations, timekeeping and all repetitive functions which do not occur on an interrupt basis. The orbital flight program consists of an executive routine concerned with IU equipment evaluation during orbit and a telemetry time-sharing routine to be employed while the vehicle is over receiving stations. In addition, in the orbital flight program, all navigation, guidance and timekeeping computations are carried out on an interrupt basis keyed to the minor loop. The minor loop contains the platform gimbal angle and accelerometer sampling routines and control system computations. Since the minor loop is involved with vehicle control, minor loop computations are executed at the rate of 25 times per second during the powered phase of flight. However, in earth orbit a rate of only 10 executions per second is required for satisfactory vehicle control.

1. Prelaunch and Initialization

Until just minutes before launch the LVDC is under control of the ground control computer (GCC). At T-8 minutes the GCC issues a prepare-to-launch (PTL) command to the LVDC. The PTL routine performs the following functions:

Executes an LVDC/LVDA self-test program and telemeters the results.

Monitors accelerometer inputs and calculates the platform-off-level indicators. Telemeters accelerometer outputs and time.

Performs reasonableness checks on particular discrete inputs and alerts. These discretes and alerts include PTL, guidance reference release (GRR), lift-off, S-IC fuel depletion, S-II propellant depletion and S-IVB engine cutoff.

Interrogates the error monitor register. The purpose of the error monitor register is to detect any errors in the operation of the LVDC and telemeter the results.

10.5.1.4.10 (Continued)

Keeps all ladder outputs zeroed which keeps the engines in a neutral position for launch.

Processes the GRR interrupt and transfers LVDC control to the flight program.

Samples platform-gimbal angles.

At T-22 seconds, the launch sequencer issues a GRR alert signal to the LVDC and GCC. At T - 17 seconds, a GRR interrupt signal is sent to the LVDC and GCC. With the receipt of this signal, the PTL routine transfers control of the LVDC to the flight program.

When the GRR interrupt is received by the LVDC, the following events take place:

The LVDC sets time base zero (T_0).

Gimbal angles and accelerometers are sampled and stored for use by flight program routines.

Time and accelerometer readings are telemetered.

All flight variables are initialized.

The GCC is signaled that LVDC is under control of the flight program.

During the time period between GRR and lift-off, the LVDC begins to perform navigational calculations and processes the minor loops. At T-8.9 seconds, engine ignition command is issued. At T-0 lift-off occurs and a new time base (T_1) is initiated.

2. Powered Flight Major Loop

Incorporation of the J-2S engine requires no modification to flight software. Guidance constants are expected to change as they do from mission to mission.

The major loop contains the navigation and guidance calculations, timekeeping and other repetitive operations of the flight program. Its various routines are subdivided by function. Depending upon mode of operation and time of flight, the program will follow the appropriate sequence of routines.

The accelerometer processing routine accomplishes two main objectives: it accumulates velocities as measured by the platform and tries to detect velocity measurement errors through "reasonableness" tests.

10.5.1.4.10 (Continued)

The boost navigation routine combines gravitational acceleration with measured platform data to compute position and velocity.

The "pre-iterative" guidance mode, or "time-tilt" guidance program, is that part of the flight program which is performed from GRR until the end of the S-IC burn. The guidance commands issued during the time-tilt phase are functions of time only. This phase of the program is referred to as open-loop guidance since vehicle dynamics do not affect or influence the guidance commands. When the launch vehicle has cleared the mobile launcher, the time-tilt program first initiates a roll maneuver to align the vehicle with the proper azimuth. After this command, roll and yaw commands remain at zero and the vehicle is gradually pitched about the vehicle Y axis to its predetermined boost heading. Rate limiting of the output commands prevents the angles (desired flight attitude angles) from exceeding 1^{o} per second.

The iterative guidance mode (IGM) routine, or "path adaptive" guidance, commences after second-stage ignition and continues until the end of S-IVB first burn. Cutoff occurs when the velocity required for earth orbit has been reached. IGM is used again during S-IVB second burn.

IGM is based on optimizing techniques using the calculus of variations to determine a minimum propellant flight path which satisfies mission requirements. Since the IGM considers vehicle dynamics, it is referred to as closed-loop guidance.

3. Interrupts

An interrupt routine permits interruption of the normal program operation to free the LVDC for priority work and may occur at any time within the program. When an interrupt occurs, the interrupt transfers LVDC control to a special subroutine which identifies the interrupt source, performs the necessary subroutines, and then returns to the point in the program where the interruption occurred. Table 10.5.1.4-V is a list of interrupts in the order of decreasing priority. Interrupts are not affected by incorporation of J-2S engine.

4. Telemetry Routine

A programmed telemetry feature is also provided as a method of monitoring LVDC and LVDA operations. The telemetry routine transmits specified information and data to the ground via IU telemetry equipment. In orbit, telemetry data must be stored at times when the vehicle is not within range of a ground receiving station. This operation is referred to as data compression. The stored data is transmitted on a time-shared basis with real-time telemetry when range conditions are favorable.

TABLE 10.5.1.4-V. INTERRUPTS

DECREASING PRIORITY	FUNCTION
1	Minor Loop Interrupt
2	Switch Selector Interrupt
3	Computer Interface Unit Interrupt
4	Temporary Loss of Control
5	Command Receiver Interrupt
6	Guidance Reference Release
7	S-II Propellant Depletion/Engine Cutoff

10.5.1.4.10 (Continued)

5. Discrete Back-ups

Certain discrete events are particularly important to the flight program since they periodically reset the computer time base which is the reference for all sequential events. For Saturn V vehicles, these significant events (time bases) are: (See Table 10.4.3-I).

- T_1 Lift-off (LO)
- T_2 S-IC Center Engine Cutoff (CECO)
- T_3 S-IC Outboard Engine Cutoff (OECO)
- T_4 S-II Cutoff
- T_5 S-IVB Cutoff (Boost Phase)
- T_6 S-IVB Restart
- T_7 S-IVB Cutoff (Orbital Phase)

Since switch selector outputs are a function of time (relative to one of the time bases), no switch selector output could be generated if one of the discrete signals was missed. A backup routine is provided to circumvent such a failure. The discrete back-up routine will simulate these critical signals if they do not occur when expected.

In the cases of the back-up routine for LO and CECO, special routines are established as a double safety check. In both cases, motion as well as time are confirmed before a back-up discrete is used. For LO, the back-up routine is entered 17.5 seconds after GRR. If the vertical acceleration exceeds 6.544 ft/sec^2 for four computation cycles, the vehicle is assumed to be airborne and the lift-off discrete is issued. For CECO, an assurance is made that an on-the-pad firing of the S-II stage cannot occur if T_1 is accidentally set. Before T_2 can be initiated, velocity along the downrange axis is tested for a minimum of 500 m/sec.

The execution time for any given major loop, complete with minor loop computations and interrupts, is not fixed. The average execution time for any given major loop in powered flight, complete with minor loop computation and interrupt processing, is called the normal computation cycle for that mode. The computation cycle is not fixed for two reasons. First, the various flight modes of the program have different computation cycle lengths. Second, even in a given flight mode, the uncertainties of discrete and interrupt processing and the variety of possible paths in the loop preclude a fixed computation cycle length.

Incorporation of the J-2S engine requires no modification to the present discrete back-ups.

10.5.1.4.10 (Continued)

b. Control Analysis

Since the J-2S/LOR vehicle, mission and payload are similar to those of the AS 504, the existing baseline FCS is adequate for S-IC burn, S-II burn, S-IVB mainstage burn and S-IVB coast modes. Therefore, control studies were oriented toward investigation of requirements during idle mode operation.

1. S-IVB Idle Mode

Pitch/yaw axis control studies were performed for the extended S-IVB idle mode preceeding second mainstage burn. Due to uncertainties in the idle mode thrust profile, two candidate control systems were considered:

Gimballed J-2S control using existing S-IVB burn pitch/yaw control scheme.

APS (Auxiliary Propulsion System) control using existing S-IVB coast pitch/yaw control scheme.

Also, as a result of the thrust profile uncertainty, two possible thrust profiles were considered.

The constant 5000 pounds thrust, which is an ideal case and the ramped thrust represent bounds of the actual J-2S idle mode thrust profile.

Due to the rapid variation in the ramped thrust profile, time response (rather than "frozen-point" frequency response) analysis was performed to evaluate vehicle performance with gimballed J-2S pitch/yaw control. An analog simulation of a single axis model of the S-IVB stage was utilized to perform the analysis. The gimballed J-2S control system is adequate since the idle mode lasts a short period of time during which no significant demands on the control system are expected; i.e., idle mode initial conditions are expected to be low (on the limit cycle of the S-IVB coast control system), no significant disturbance torques are expected and no maneuvers will be performed during idle mode.

Results of the analysis considering APS control for S-IVB idle mode operation indicate that APS control torque capability is sufficient, and a relatively small amount of fuel is required. However, possible misalignment between the J-2S thrust vector and S-IVB center of gravity could cause sizable APS propellant consumption. It is concluded that the gimballed J-2S utilizing the existing S-IVB burn control scheme with increased gains will be used as the S-IVB idle mode control system. The increased gains would be switched in at the end of first S-IVB mainstage burn, and switched back to the lower gains at second mainstage burn start command. A final design effort will be required to determine the "best" idle mode control gains.

10.5.1.4.11 Instrument Unit

The IU is a cylindrical structure 6.6 meters (260 in.) in diameter and 0.9 meters (36 in.) in height, mounted on top of the S-IVB stage as illustrated in Figure 10.6.1.4-14. The IU is attached to the S-IVB stage and to the Spacecraft/Lunar Module Adapter (SLA) and remains attached to both after Command and Service Module (CSM) and Lunar Module (LM) separation.

The IU contains the equipment necessary to:

Perform guidance and control of the vehicle from lift-off through CSM and LM separation.

Aid in radar tracking of the vehicle.

Provide a command link for ground control of the vehicle.

Provide temperature control for the electronic equipment in the IU and the S-IVB stage forward skirt. Instrumentation is provided to monitor performance of this equipment.

Incorporation of the J-2S engine on the Saturn V vehicle will require hardware modifications to two IU components. Flight Control Computer and Control Distributor (reference Figure 10.5.1.4-15).

10.5.1.4.12 Environmental Control

a. S-II Environmental Control

Incorporation of the J-2S engine on the S-II stage will require no modification to the environmental control system.

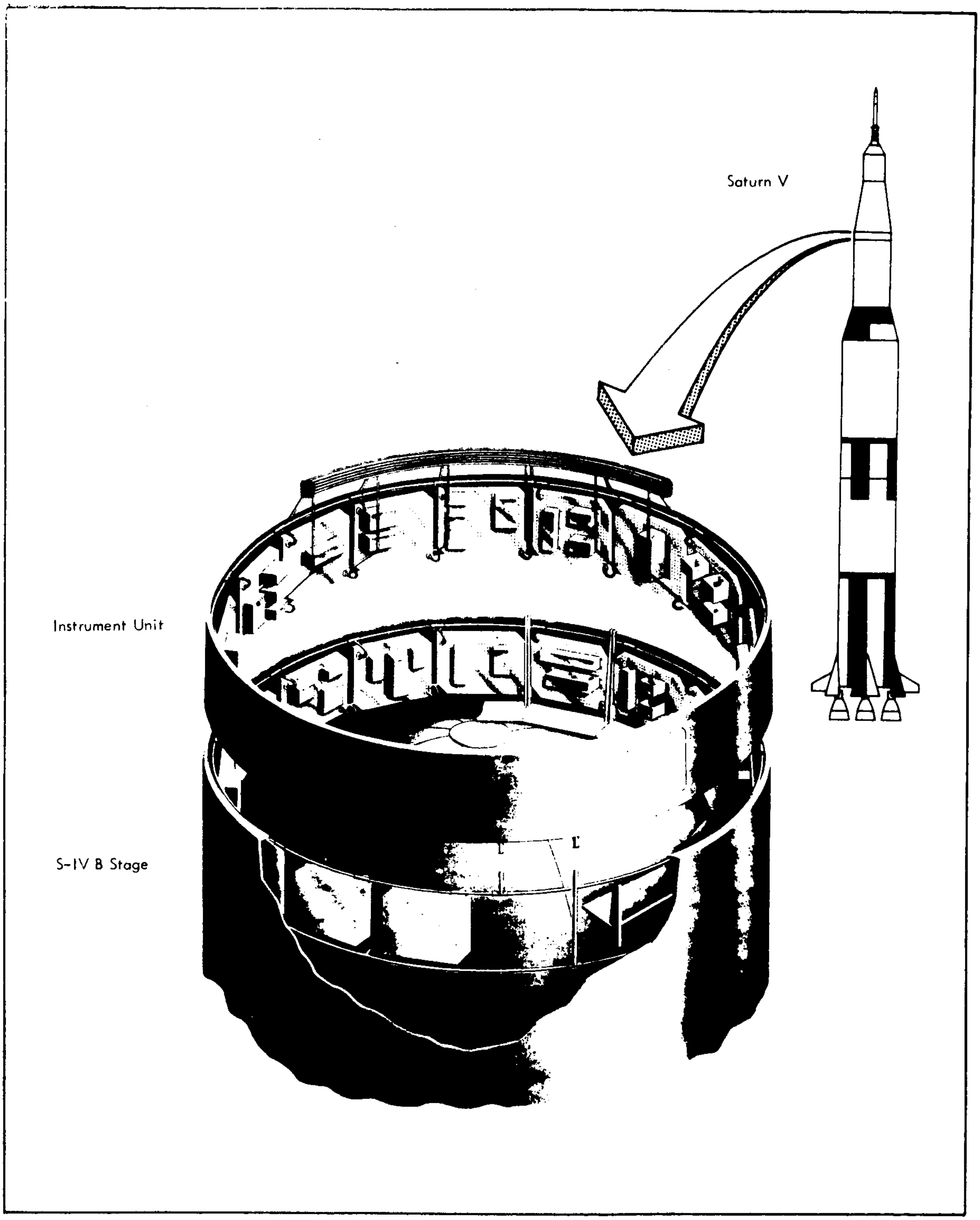
b. S-IVB Environmental Control

Forward Skirt Thermoconditioning

The electrical/electronic equipment in the S-IVB forward skirt area is thermally conditioned by a heat transfer subsystem using a circulating coolant for the medium. Principal components of the system, located in the S-IVB stage forward skirt area, are a fluid distribution subsystem and cold plates. The coolant is supplied to the S-IVB by the IU thermo-conditioning system, starting when electrical power is applied to the vehicle, and continuing throughout the mission.

Forward Skirt Area Purge

The forward skirt area is purged with GN_2 to minimize the danger of fire and explosion while propellants are being loaded or stored in the stage, or during



Ref. 10.4-31, Fig. 12.1-1

FIGURE 10. 5. 1. 4-14. INSTRUMENT UNIT

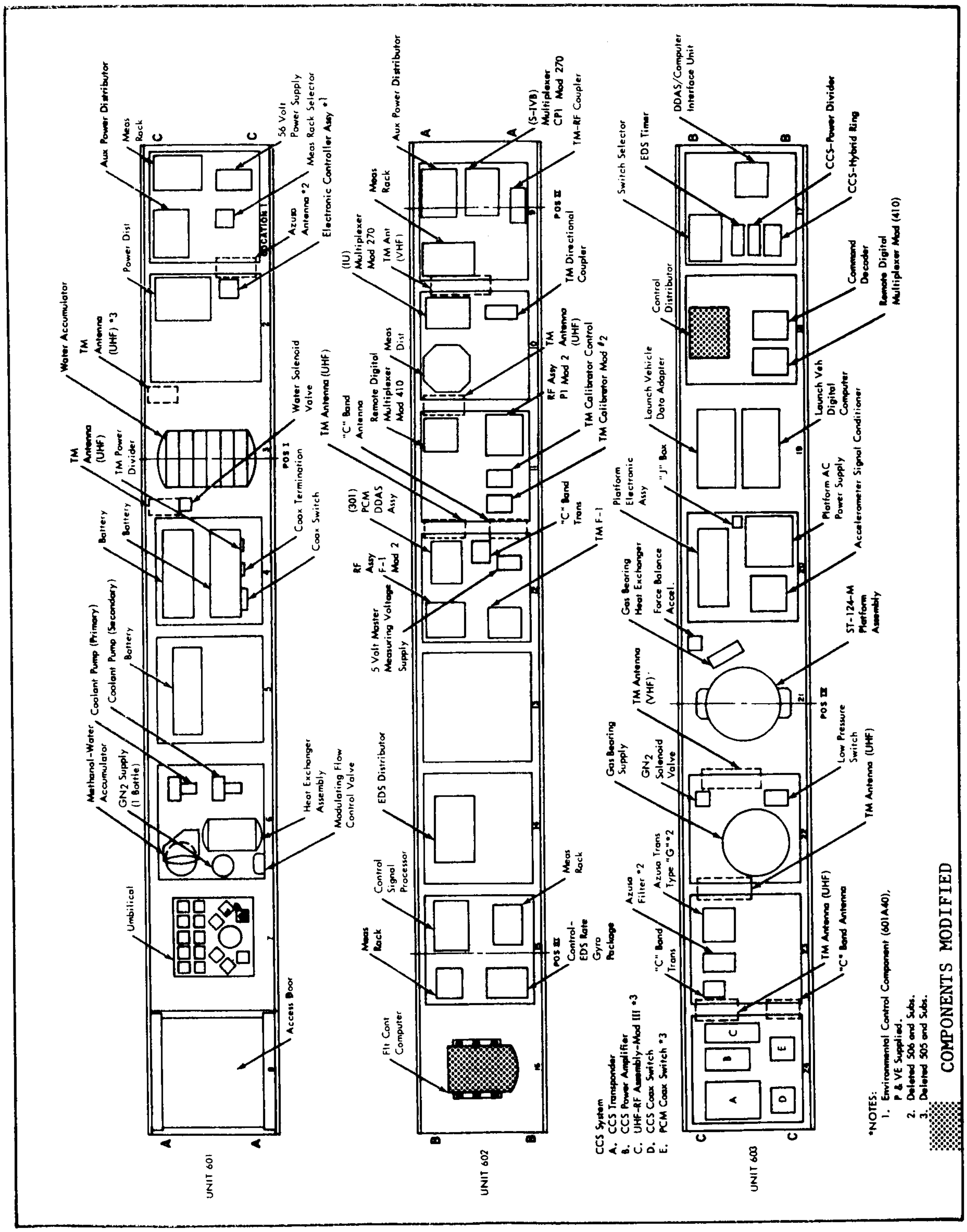


FIGURE 10.5.1.4-15. INSTRUMENT UNIT COMPONENTS

Ref. 10.4-31, Fig. 12.1-3

10.5.1.4.12 (Continued)

other hazardous conditions. The purge is supplied by the IU purge system which purges the entire forward skirt/IU/adapter area. The total flow rate into this area is approximately 3500 SCFM.

The thermal conditioning system contains five thermal conditioning panels (cold plates) in S-IVB stage. The S-IVB cold plate dimensions are 68.58 centimeters by 109.22 centimeters (27 in. by 43 in.).

"Electronic Equipment Ambient Conditioning"

"The ambient space surrounding the S-IVB/IU equipment is conditioned to maintain a minimum temperature of + 40°F by the use of conditioned air prior to loading cryogenics and after loading with GN_2. The latter maintains an inert atmosphere.

The S-IVB aft skirt mounted equipment and the APS modules are also conditioned by a separate system. The same ground rules apply as in the forward section."

"Eight components on S-IVB aft skirt panels No. 2, 3, 4, 8, 17 and 19 have external electric heating in addition to radiation shields. The heater blankets are form-fitted to the electronic component. They are made of solid and foam polyurethane. The heater elements on the inside walls of the blanket are controlled by thermostats which contact the component to regulate its temperature." (Reference 10.4-43.)

The heater thermostat "on" and "off" command controls thermal conditioning of the 8 components as required during the mission. The sequencer "on" command applies 28 volts due to a heater power but at a predetermined time. This enables all heater blankets simultaneously. The thermostat control devices on each heater blanket will cycle power on and off to control the temperature within the required band.

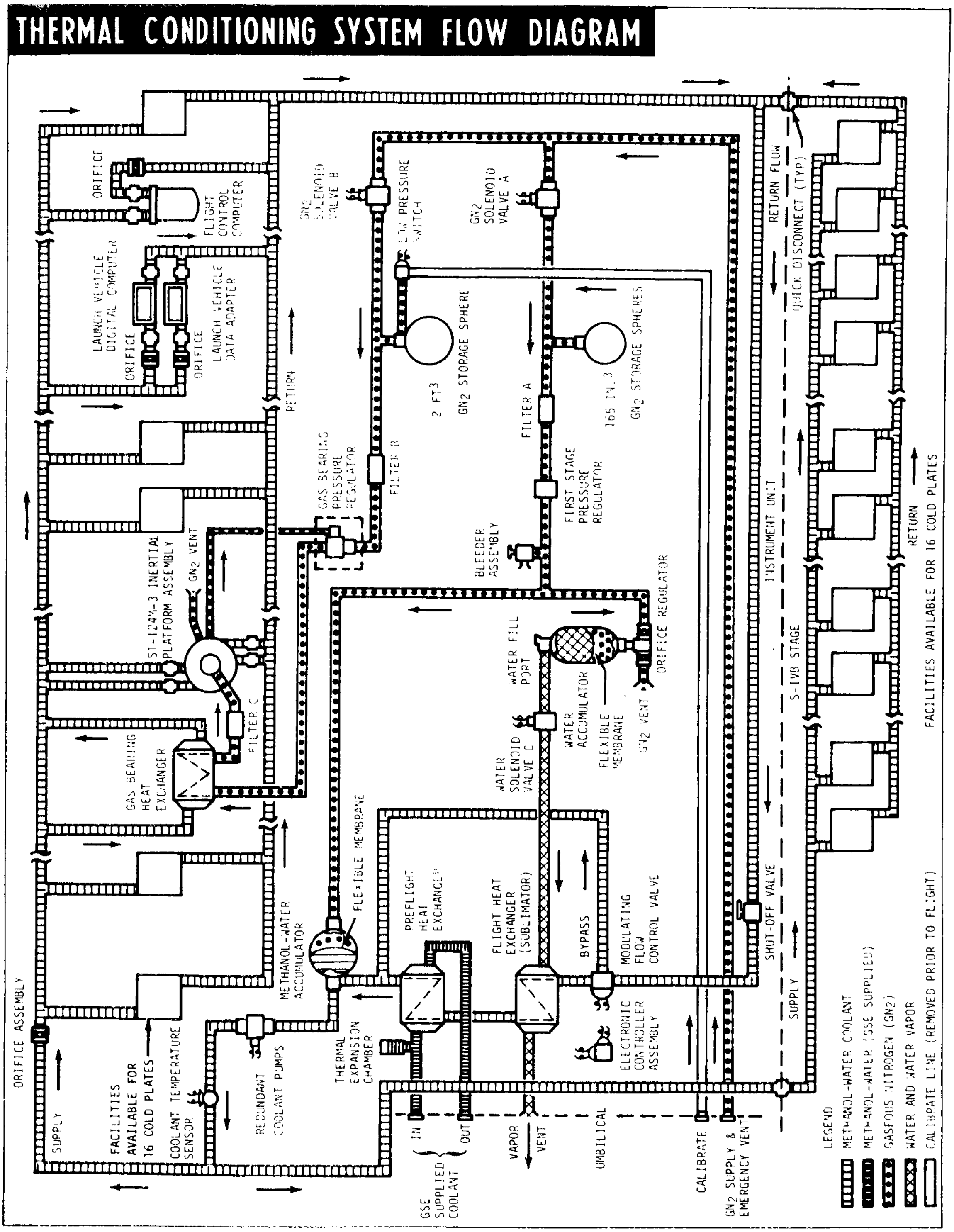
The radiation shielding is required by components which require passive thermal control only.

c. IU Environmental Control

The environmental control system (ECS) maintains an acceptable operating environment for the IU equipment during preflight and flight operations. The ECS is composed of the following (reference Figure 10.5.1.4-16).

The thermal conditioning system (TCS) which maintains a circulating coolant temperature to the electronic equipment of 59° ±1°F.

Preflight purging system which maintains a supply of temperature and pressure-regulated air/GN_2 in the IU/S-IVB equipment area.



Ref. 10.4-32, Fig. 6.2-18

FIGURE 10.5.1.4-16. IU ENVIRONMENTAL CONTROL

10.5.1.4.12 (Continued)

Gas bearing supply system which furnishes GN_2 to the ST-124M3 inertial platform gas bearings.

Hazardous gas detection sampling equipment which monitors the IU/S-IVB forward interstage area for the presence of hazardous vapors.

Incorporation of the J-2S engine on the Saturn V Launch vehicle will require no modification to the IU Environmental Control System.

10.5.1.4.13 Electrical Support Equipment

a. S-II Electrical Support Equipment

The S-II ESE at Seal Beach, California, and Mississippi Test Facility (MTF) consist of an Automatic Checkout System whose primary purpose is to perform a comprehensive checkout of the Saturn S-II stage (with J-2S engine implementation) prior to shipment to KSC. A series of ESE end items are used to test the Saturn S-II stage after manufacture under conditions which most nearly simulate actual flight of the stage. Seal Beach test sites provide a thorough check of the electrical, mechanical, fluid and telemetry systems. MTF test sites perform these same functions and in addition provide structural testing through the static firing.

The ESE is designed to provide as automatic testing as practicable; however, manual functions are retained as backup to automatic functions and also to provide certain functions not feasible to automatic control.

Checkout Station Requirements

The automatic checkout system for the J-2S engine implementation consists of a controlling computer complex linked to a number of specialized checkout stations. These checkout stations in turn are linked to the functional systems on the stage. Reference Figure 10.5.1.4-17.

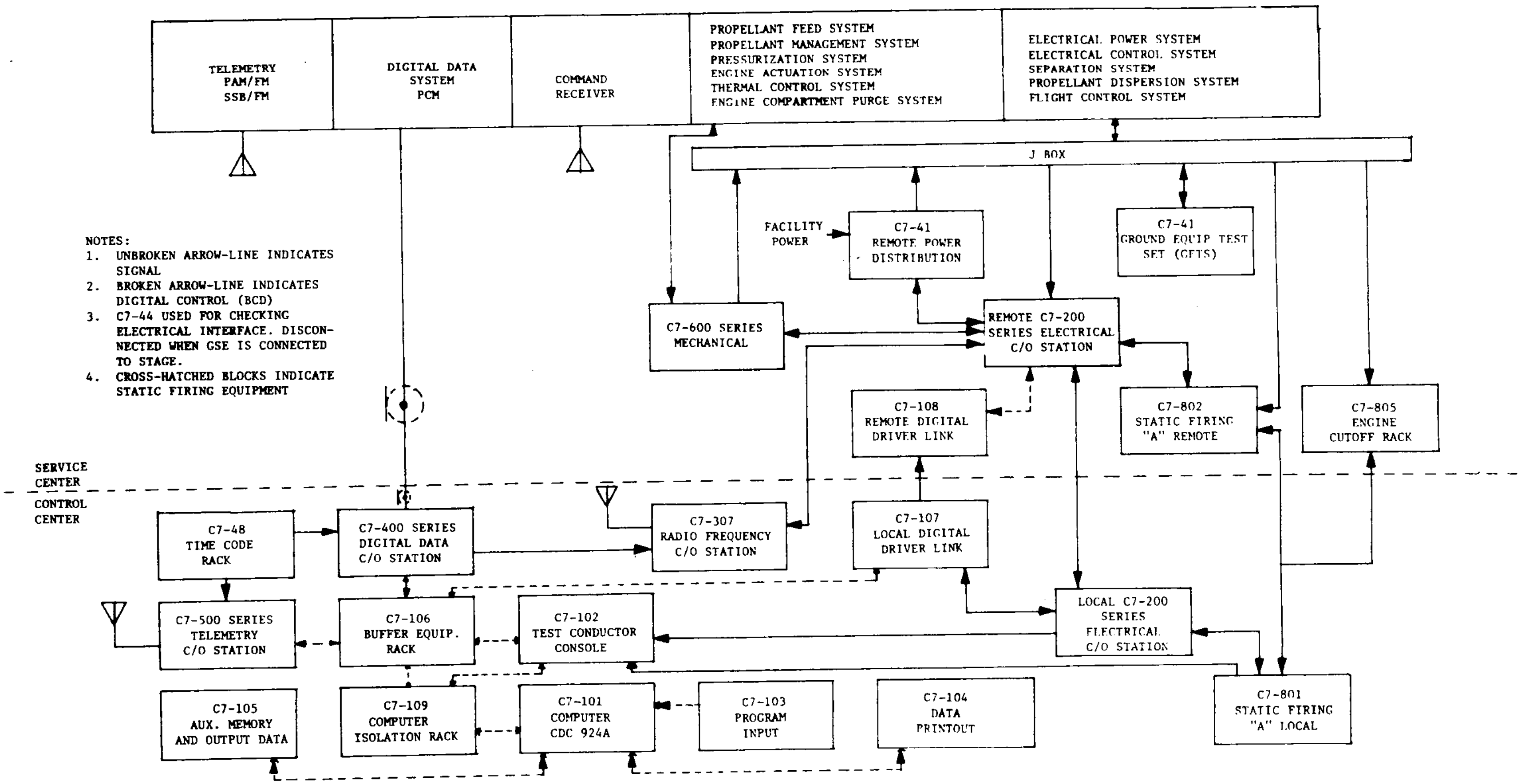
Associated with its particular function, the checkout station must provide the following:

Provide stimuli to the stage.

Provide control signals to associated checkout stations and the stage.

Monitor GSE and stage system responses.

Store system responses and results of evaluation as required.



Ref 10.4-33, Fig. 7.7.3-1

FIGURE 10.5.1.4-17. ELECTRICAL SUPPORT AUTOMATIC CHECKOUT

10.5.1.4.13 (Continued)

Translate system responses into a suitable format for facilitating evaluation.

Evaluate system responses. Perform further testing or initiate corrective action on the basis of evaluated results.

Provide visual readouts and complete records of test procedure and system responses.

Provide self-check for maintenance and machine confidence.

Computer Complex Checkout Station (C7-100)

The C7-100 checkout station consists of the C7-101 checkout computer and its peripheral equipment (C7-103 Program Input Rack, C7-104 Data Printout Rack, C7-105 Auxiliary Memory Rack, and C7-110 High Speed Data Printout Rack), C7-102 Test Conductor Console, C7-106 Buffer Equipment Rack, C7-107 Local Digital Driver Link Rack (MTF only), C7-108 Remote Digital Driver Link Rack (MTF only), C7-109 Isolation and Drive Rack and C7-111 Magnetic Tape Transport Rack (MTF only). The purpose of the computer complex checkout station is to link the automatic operations of the GSE checkout stations and S-II stage systems. Via the checkout stations (discussed subsequently), the checkout computer will control the sending of stimuli to the S-II stage systems and the monitoring of responses from the S-II stage systems.

C7-100 Station Changes

There are no changes to the C7-100 Computer complex end items because of J-2S engine implementation except for the C7-102 Test Conductor Console. Five functions are deleted from the hazardous monitor panel due to deletion of the recirculation system. These five functions are:

Recirculation System DC greater than 64V.

Recirculation System DC greater than 60V.

Recirculation System DC Low.

Recirculation System He Press Bottle greater than 1600 psig.

Recirculation System He Bottle greater than 825 psig.

These changes require removal of the nameplates and deactivation of the cabling in the J-boxes which connect the hazard functions to the C7-102.

10.5.1.4.13 (Continued)

Electrical Checkout Station (C7-200)

The electrical checkout station is composed of the C7-201 Automatic Control Rack, C7-202 Manual Control and Display Rack, C7-204 Signal Distribution Rack, C7-205 Special Data Rack, C7-208 Station Control and Display Rack, C7-209 Local Control and Display Rack, C7-210 Stage Substitutes Rack, C7-211 Scanner Rack, C7-212 Discrete Display Rack and C7-213 Interlock Relay Rack.

The system provides hardware for the electrical checkout of many stage systems. Checkout can be done in a manual mode or automatic mode.

In the automatic mode, hardware is provided for electrical checkout under computer control allowing the computer to select and direct station equipment operation and to transform data into a format for computer analysis and status recognition. Display of stimuli and responses provides rapid station status as well as easy troubleshooting and fault isolation. Interlocks of stage power, stimuli and responses, and associated limit-condition sensing of the power busses, assure safe checkout operation. The electrical checkout station is used in testing the following stage systems:

Propellant dispersion
Separation
Electrical
Engine
Pressurization
Measurement
Propellant management
Flight control
Thermal control

C7-200 Station Changes

The C7-202 Manual Control and Display Rack changes are in the switch selector control and display drawers and are due to the addition and deletion of switch selector commands. These consist of deleting two commands, revising two commands and adding seven new commands. The changes are accomplished by changing nomenclature on the display panels and by adding and rewiring diodes in the control and display drawers.

The C7-204 Signal Distribution Rack changes are in the engine system drawers and consist of deleting twenty-one commands, revising eleven commands and adding twenty-five commands. Twenty-one commands which are added will use the spared positions,

10.5.1.4.13 (Continued)

and four new commands will require additional hardware (add four flip-flops, four relays and four relay drivers). This will be accomplished by repatching and adding approximately twenty new patchcords. These changes are due to the deletion of the recirculation system, helium injection system and start tank system. Added were functions to engine mainstage operation, engine ready and engine ignition detect signals. These modifications require three new rack harnesses and one new output patch panel assembly.

The C7-209 Local Control and Display Rack will be changed by revising the nomenclature on six switch positions on the engine stimuli control panels and by revising 26 legend light nomenclatures on the engine stimuli display panels. C7-209 Rack changes are due to deletion of recirculation system, helium injection system and start tank system. Principal additions are functions for engine mainstage operation, engine cutoff, and engine ignition detect.

Changes to the C7-211 Scanner Rack consist of adding approximately 60 patchcords. C7-211 Rack changes are due to the deletion of the recirculation system, start tank system and helium injection system commands and measurements along with addition of the engine mainstage operation, engine ready, engine detect and engine start system signals. In some cases the C7-211 Rack provides a lamp driver for stage measurements with a 4.02 K resistor in series.

The C7-212 Discrete Display Rack changes require the revision of 60 legend lights and the addition of 30 new legend light connections. The deletions are due to recirculation and helium injection systems. The additions are due to new engine cutoff commands.

The C7-213 Interlock Rack changes consist of revising the patch panels. Approximately 100 patchcords are deleted and 90 are added. Interlock requirements for J-2S engine implementation have not yet been fully defined. C7-213 Rack changes are due to deletion of recirculation pump start interlock and start tank vent open commands. The additions are not yet clearly identified; however, the changes entail only patching changes in the C7-213 Rack.

No changes result to the C7-201 Automatic Control Rack, C7-205 Special Data Rack, C7-208 Station Local Control and Display Rack and the C7-210 Stage Substitutes Rack.

Range Safety Command Receiver (RSCR) Checkout Station (C7-307)

The RSCR checkout station provides equipment and circuitry for checking the radio command receivers on the stage. Test signals from the rack are transmitted to the stage radio command receivers via hardwire or air link. Signals indicating response of stage receivers are returned to the rack for comparison with specific requirements. This signal comparison is performed by personnel operating the rack.

10.5.1.4.13 (Continued)

The rack verifies proper operation of stage installed receivers, power dividers, filters and hybrid and interconnecting coaxial cable links. The command antenna subsystem test verifies the operation of each command antenna and its associated voltage standing wave ratio (VSWR) measurement. Partial automatic rack operation is provided by theElectrical Checkout Station and the Digital Data Acquisition Station.

C7-307 Changes

Due to J-2S engine incorporation, there are no changes to the C7-307 Rack.

Digital Data Acquisition System Checkout Station (DDAS) C7-400

The DDAS checkout station is composed of four racks, C7-401 Automatic Control and Display Rack, C7-402 Local Control and Display Rack, C7-403 PCM (DRS-1) Rack and C7-406 Computer Adapter Rack. This station provides the hardware and circuitry to retrieve data from the pulse code modulated (PCM) telemetry system of the stage. Code pulse trains from the PCM systems are regenerated to remove transmission noise and are converted from serial to parallel format to produce information suitable for computer processing. Capability for recording and limited playback is made available through control and feedback lines to the tape recorder rack (C7-516) of the telemetry checkout station. The station receives, demodulates and decommutates PCM data from either the stage or from magnetic tapes, and routes the data to display at the station or the telemetry ground station tape recorder (C7-516).

C7-400 Station Changes

There are no changes to the C7-400 Checkout Station due to J-2S engine incorporation.

Telemetry System Checkout Station (C7-500)

The Telemetry Checkout Station is composed of 9 racks, C7-510 Automatic Control and Display Rack, C7-511 PCM Format Rack, C7-512 Oscillograph Rack, C7-513 Decommutation Rack, C7-514 Discriminator Rack, C7-515 Receiver Rack, C7-516 Tape Recorder Rack, C7-518 Single Sideband Rack and C7-519 Telemetry Receiver Station Model TRS-1. The purpose of the C7-500 station is to check the airborne telemetry systems of the stage. The primary functions include:

Receive, monitor and detect RF carrier frequencies

Provide for the magnetic recording and playback of undecoded intelligence transmitted via the PAM/FM/FM, SSB/FM and PCM telemetry systems

10.5.1.4.13 (Continued)

Decode PAM/FM/FM and SSB/FM data

Convert analog data to the PCM format for computer access during automatic checkout

Provide for local monitoring of a single channel PCM data

Provide for local control of the stage RF equipment and telemetry calibrations

Process and record a limited amount of telemetered data for post-test evaluation.

C7-500 Station Changes

The C7-500 Telemetry Checkout Station is unaffected by changes to incorporate the J-2S engine.

Static Firing Control Station (C7-800)

The C7-800 Station exists at MTF only and is activated only for static firing tests. The station consists of the C7-801 (Local Static Firing Rack), the C7-802 (Remote Static Firing Rack) and the C7-805 (Engine Cutoff Rack). The C7-801 and C7-802 racks permit manual control, display and signal distribution of all 28 VDC parameters required for a static firing of the S-II stage (except engine gimballing, which is computer controlled). These racks also provide for automatic sequencing (with manual override capability) during the interval from completion of propellant loading to initiation of propellant detanking. In addition, the racks are integrated with the facility programming bay to provide a time-oriented countdown. The C7-801 rack consists of four bays and is located in the test control center. The C7-802, consising of three bays, is located in the ground service center and is remotely controlled by the C7-801 rack.

The C7-805 rack permits the monitoring of certain engine parameters and provides automatic cutoff of all engines in the event of a dangerous condition during static firing. The rack indicates the cause of cutoff by means of auxiliary monitoring devices. The C7-805 rack, consisting of two bays, is located in the ground service center and is remotely operated and selftested from the C7-801 rack.

C7-800 Station Changes

Changes to the C7-801 Local Static Firing "A" Rack consist of adding seven new switch selector channels, adding 10 new switches with indicator lights, deleting 13 switches and nine associated indicator lights, adding one new switch panel and adding four new diode boards. Because of the deletion of the start tanks, two start tank meter panel drawers are deactivated.

10.5.1.4.13 (Continued)

C7-801 Rack modifications are required because of deletion of the switch selector commands, recirculation system, helium injection system and start tank system. The new additions are for switch selector commands, engine mainstage operation commands and solid propellant turbine spinner (SPTS) system and engine ready and ignition detection signals. Changes to the C7-802 Remote Static Firing "A" Rack include 17 relay function additions and 31 relay function deactivations. These are accomplished by approximately 100 patchboard deletions and approximately 75 patchcord additions. These changes are required due to deletion of the recirculation system, helium injection system and start tank system. The additions are due to new engine mainstage operation commands and engine relay isolation test commands. The C7-805 Engine Cutoff Rack is modified by deactivating the five drawers for monitoring the Gas Generator temperature.

Cable and J-Box Requirements

The cabling and J-box installations provide multiconductors for the interconnection and the transmission of electrical power and signals between the S-II stage and GSE and between GSE end items. The installed cabling is capable of sustaining the maximum load requirements during any checkout phase, protecting where necessary circuits with fuses or resistors and isolating rack interconnections with diodes. Mississippi Test Facility cabling, in addition to the above, has the capability of sustaining load conditions of a static firing.

Cabling change requirements are made for all four sites (two at Seal Beach and two for MTF). In practice, however, only one site will be modified at each location. The cabling information gives all necessary information for choosing the respective sites.

Accpetance Stand No. 1 Cable Installation (C7-35)

Acceptance Stand No. 2 Cable Installation (C7-40)

MTF Firing Control Center Cable Installation (C7-38)

Changes to these equipments require revision of the terminal distribution rack wire lists in the control centers and test stands. Changes to cabling and terminal distribution racks consist of moving jumper wires and equipment cable wires. Changes to this equipment can be summarized as follows:

C7-35 (A2) Move 110 jumper wires, revise 1300 terminations
C7-40 (A1) Move 200 jumper wires, revise 860 terminations
C7-38 (C2) Move 55 jumper wires, revise 560 terminations
C7-38 (C1) Move 160 jumper wires, revise 340 terminations

10.5.1.4.13 (Continued)

Electrical Terminal Distributor Station 8 and Station 9 (SDD 154)

Cable Installation, Station 8 (SDD 196)

Cable Installation, Station 9 (SDD 197)

Changes to the SDD cabling consist mainly of moving jumper wires, revising equipment cable wires, fuses and diodes. These SDD cabling changes can be summarized as follows:

SDD 154 Station 8 Move 131 jumpers, revise 338 terminations, add 16 fuses

SDD 154 Station 9 Move 131 jumpers, revise 338 terminations, add 16 fuses

SDD 196 Station 8 Move 160 jumpers, revise 100 terminations

SDD 197 Station 9 Move 160 jumpers, revise 100 terminations

Instrumentation Drag-On Cables A2 (SDD 345)

Instrumentation Drag-On Cables A1 (SDD 346)

Changes to the instrumentation drag-on cables at MTF consist of the deactivation of existing measurements and the addition of new measurements which affect cable routine and receptacle box harnesses. For each of SDD 345 and SDD 346 cable sets, there are five cable assembly revisions, eight harness assemblies changed and 10 cables deleted.

Stage Station Test Electrical Harness (C7-43) Field Site Installation of Stage Mounted ESE

Changes to the C7-43 consist of revising the cable sets. This entails the deletion of six cables from each of the eight cable sets, in support of deletion of the re-circulation system. Changes to the Field Site Installations of stage mounted GSE consist of revising drawings depicting installation of "carry-on" instrumentation cables and drag-on instrumentation cables to show new mountings, brackets, cable clamp locations and drilling details of Saturn J-2S Instrumentation. These drawings affect MTF, KSC, and the C7-43.

Power Distribution System, Test Stand A2 (C7-80)

Power Distribution System, Test Stand A1 (C7-81)

The power distribution systems for test stands A2 and A1 will be revised to show deactivation of the 56 VDC recirculation system and the engine ignition battery simulator.

10.5.1.4.13 (Continued)

Other ESE Items

Digital Events Recorder (C7-77)

The digital events recorder detects and records changes of state

$$(ON = +8^{+26}_{-0}, \; OFF = {}^{+7.5}_{-0})$$

on the S-II discrete command and measurement lines. When a change is detected, the state to which it changes, the time of the change to the nearest millisecond and an identifying number are recorded on a hard copy printout and punched paper tape or magnetic tape (MTF only).

C7-77 Changes

There are no hardware changes to this equipment -- only revision to j-box/TDR input lines and a revision to the Interface Control Document (ICD). The ICD is used to correlate the identifying number with the discrete on the input line. Revision of the ICD's is necessary to delete 48 functions (using the spares) and revise 49 functions.

Remote Distribution Rack (C7-41)

The C7-41 Remote Distribution Rack is used to monitor and transfer the power from facility power to GSE power, and from GSE power to stage power. It is provided with interrupt circuits and interlocks to protect all stage and GSE busses. Provision is made for isolating the C7-200 station, the C7-800 station and the C7-603 Pneumatic Checkout Console Set during testing and servicing at the static firing sites (MTF) and Seal Beach.

C7-41 Changes

The changes to the C7-41 Rack can be summarized as follows:

Deactivate the entire 56 VDC recirculation power system and the electrical control, sensing and power transfer system for the 56 VDC rectifier.

Deactivate the electrical control and power switching for the stage recirculation bus and battery simulator circuits.

Deactivate the engine ignition load bank and engine ignition battery simulator.

10.5.1.4.13 (Continued)

Mississippi Test Facility, in addition to the above changes, requires deactivation of the isolation valve control and summary indicating circuits. Indicators on the isolation valve drawers are changed to show spare. C7-41 changes are due to deletion of the recirculation system. The LOR mission does not use a 56 VDC battery since the recirculation system has been removed; however, this battery system can be used for the Thrust Vector Control System (TVCS) of the LEO mission. If a C7-41 hardware change is made for both the LOR mission and LEO mission simultaneously, the hardware changes can be minimized.

Time Code Rack (C7-48)

The C7-48 Time Code Rack provides the time code signals for the various systems requiring a synchronous timing signal, e.g., strip chart recorders, tape recorders, local and remote time displays, C7-101 Computer and C7-77 Digital Events Recorder. This serves as the master time synchronization throughout the ESE computers checkout complex.

C7-48 Changes

There are no changes to this equipment because of the J-2S engine modification.

Ground Equipment Test Set (C7-44)

The Ground Equipment Test Set (GETS) is used to verify the functional readiness of the ground support equipment checkout stations. It receives the electrical stimuli at the stage umbilical connectors (like the stage) from the ESE, processes these stimuli and sends back responses to the ESE. Signal acceptance and response generation is such that the electrical functional characteristics of the stage checkout equipment are verified for proper operation. The equipment has the capability of processing analog and digital signals and encoded discrete commands which simulate the IU. GETS performs hazardous condition simulation, hazardous monitor switching, flight control simulation, decoder switching, GETS control switching, stage systems simulation, time delay patching, electrical power operation and audio communication network operation.

C7-44 Changes

Changes to the C7-44 will require the addition of two new integrated program board assemblies, the addition of two new static firing program board assemblies, the revision of the self-test program boards and a revision of the IU command decoder for the addition of seven new switch selector commands and deletion of two switch selector functions. Patch board rework will require approximately 1000 new patchboards.

10.5.1.4.13 (Continued)

Engine Sequence Recorder (SDD 273)

Power Supply 56 VDC (SDD 337) (Station 8 and Station 9 Only), S-II Ordnance

The SDD 273 Oscillograph Recorder is used to monitor the engine valve positions during engine sequence testing. These recorders are used in lieu of the C7-205 Special Data Rack for engine valve analog measurements.

SDD 337 VDC power supply provides the electric power to operate the five LH_2 recirculation pump motors during checkout and simulated prelaunch operations.

SDD 273 and SDD 337 Changes

The SDD 273 recorder requires a change in documentation for the reidentification of new engine function assignments on each channel. The SDD 337 56 VDC power supply will be disconnected and documented as spare on the power distribution drawings and system schematics.

S-II Ordnance Changes

The S-II Ordnance requires only changes in documentation due to addition of the engine SPTS system.

C7-100 Station Software Changes

The principal software items affected by Electrical ESE changes are:

C7DXXX-127-XXX	GSE Integrated Test
C7DXXX-250-XXX	Electrical Power System Checkout
C7DXXX-251-XXX	Flight Measurements System Checkout
C7DXXX-252-XXX	Stage Networks Acceptance Program
C7DXXX-253-XXX	Flight Control System Checkout
C7DXXX-254-XXX	Pressurization System Checkout
C7DXXX-255-XXX	Simulated Flight System Checkout

ESE Operational Specifications

Operational Specification Changes

MA0701-1012-111 C7-200 Station Manual Checkout (8 or 9)

Affects C7-202, C7-204, C7-209, C7-211, C7-212, C7-213.

10.5.1.4.13 (Continued)

Total change approximately 20%.

Specification is easier to run because of:

Deletion of 56 VDC power supply switching test (Recirc).

Addition and deletion of switch selector cards.

Addition and deletion of C7-204 commands controlled by C7-209 and C7-202.

MA0701-1014-210 Static Firing Countdown (A2)

Affects C7-41, 44, 204, 211, 212, 123, 801, 802, 805.

Total Change approximately 30%.

Essentially same ease of performing specification as before.

Addition and deletion of engine system functions (including recirc system).

Changes in switch selector functions.

Changes in the auto sequence test.

MA0701-1013-211 Static Firing Countdown (A1)

Affects C7-41, 44, 204, 211, 212, 213, 801, 802, 805.

Total change approximately 30%.

Essentially same ease of performing specification as before.

Addition and deletion of engine system functions including recirculation system.

Changes in switch selector functions.

Changes in auto sequence test.

10.5.1.4.13 (Continued)

MA0701-1027-111 ACE/GSE Activation Station 8 and 9

Affects C7-41.

Total change approximately 5%.

Essentially easier specification to run than before.

Deactivation of 56 VDC power supply system of C7-41.

MA0705-1015-111 Semi Automatic Activation C7-100 Station (8 and 9)

Affects C7-102 hazardous monitor.

Total change approximately 5%.

Testing is easier since there are fewer hazards to check.

Deletion of recirculation power and pressurization conditions.

b. S-IVB Electrical Support Equipment

The S-IVB ESE at Huntington Beach (HB) and Sacramento (STC), California consists of a digital computer controlled checkout system whose primary function is to verify the design integrity and operational capability of the complete S-IVB stage. A back-up manual control capability is also provided to allow safing and/or shutdown of the stage in the event of failure in the automated equipment.

Modifications in the ESE can be made with relative ease because patch panels have been provided to route commands, talkbacks and measurements between the automatic checkout system (ACS) and the vehicle and ESE. Changes in the patch panels will impact a portion of the manual control panels. Changes on the control panels can be economically handled by relabeling functions on the panels rather than removing or changing hardware.

The modifications to these manual control consoles and panels are shown in detail in the following paragraphs in terms of vehicle system and propulsion ESE system modifications. As the J-2S engine implementation affects each of these systems, the resulting ESE changes are specified. The systems in question are: the electrical power system, the propulsion support system, the engine turbine start pressurization system, the chilldown system, the pneumatic control system, the auxiliary propulsion system, the hydraulic system, the propellant utilization system, the ordnance system and the instrumentation and telemetry system.

10.5.1.4.13 (Continued)

1. Electrical Power System

In the electrical power system ESE, the Vehicle External Power Racks provide a ground power source for all four 28 VDC vehicle power busses and also includes a 56 VDC supply.

The impact of the J-2S engine on this system has the overall effect of reducing the vehicle power requirement. Therefore, the J-2S engine will not affect this system.

2. Propulsion Support System

At HB, the propulsion support system is the Automatic Stage Checkout Pneumatic Console which controls the distribution of helium or nitrogen gas to meet the leak and functional checkout requirements of the stage and APS equipment during dry checkout exercises. The J-2S engine does not require a start bottle supply, therefore, the ESE is affected in the following way:

Remote Pneumatic Control Console (Mainstage Checkout Panel-A2)

Delete:

Ind.-Start bottle open	3 patchcords deleted
Ind.-Start bottle closed	2 patchcords deleted
Switch-Start bottle supply valve	6 patchcords deleted

At STC, the propulsion support system is the pneumatic console, the Gas Heat Exchanger and the Beta Pneumatic consoles. These units are modified to eliminate the cold helium supply for thrust chamber chilldown and the cold hydrogen supply for the turbine start bottle. The ESE is affected as follows:

Remote Pneumatic Control Console (Mainstage Checkout Panel-A2)

Delete:

Ind.-Start bottle open	3 patchcords deleted
Ind.-Start bottle closed	2 patchcords deleted
Switch-Start bottle supply valve	6 patchcords deleted

Remote Pneumatic Control Console-Beta 1 (Vehicle Supply Panel-A2A2)

Remote Pneumatic Control Console-Beta 3 (Vehicle Supply Panel-A2A2)

Delete:

Ind. - Start tank GH_2 Fill-open	3 patchcord deletions

10.5.1.4.13 (Continued)

Ind.-Start tank GH_2 Fill-closed	3 patchcord deletions
Switch-Start tank GH_2 Fill supply valve	11 patchcord deletions
Ind.-Start tank GH_2 Vent-open	3 patchcord deletions
Ind.-Start tank GH_2 Vent-closed	3 patchcord deletions
Switch-Start tank GH_2 Vent valve	11 patchcord deletions
Ind.-Cold GH_2 Vent-open	3 patchcord deletions
Ind.-Cold GH_2 Vent-closed	3 patchcord deletions
Switch-Cold GH_2 supply vent valve	6 patchcord deletions
Ind.-Thrust chamber C/D supply open	3 patchcord deletions
Ind.-Thrust chamber C/D supply closed	3 patchcord deletions
Switch- Thrust chamber C/D supply valve	7 patchcord deletions
Ind.-Thrust chamber supply vent-open	3 patchcord deletions
Ind.-Thrust chamber supply vent-closed	3 patchcord deletions
Switch-Thrust chamber supply vent valve	5 patchcord deletions
Meter-Eng start tank pressure	3 patchcord deletions

<u>Vehicle Supply Valves Panel (A1A3)</u>

Delete:

Ind.-Start tank purge supply-open	3 patchcord deletions
Ind.-Start tank purge supply-closed	3 patchcord deletions
Switch-Start tank purge supply valve	8 patchcord deletions

Various measurements are also deleted from the propulsion support system at STC which cause the following patchcords to be deleted from the patch panel models:

<u>Elect C/O Accessory Kit-Beta 1</u>

<u>Elect C/O Accessory Kit-Beta 3</u>

Delete:

Ind.-Eng. start tk bot supply press	18 patchcords deleted
Ind.-Eng. start tk bot supply temp	15 patchcords deleted
Ind.-Thrust chamber purge and C/D temp	15 patchcords deleted
Ind.-Thrust chamber purge and C/D OI press	12 patchcords deleted
Ind.-GH_2 Ht-Ex-1 cold output temp	15 patchcords deleted

10.5.1.4.13 (Continued)

At KSC, the propulsion support system consists of the pneumatic consoles and the Gas Heat Exchanger. These consoles are similar in function to the models at STC and therefore have similar deletions. The electrical control panels are affected in the following manner:

Helium Control Panel (406A3)

Delete:

Ind. -GH bank purge-open
Ind. -GH bank purge-closed
Switch-GH bank purge supply valve

GH/GN Control Panel (404A4)

Delete:

Ind. -GH 6000 PSI supply valve-closed
Ind. -GH 6000 PSI supply valve-open
Switch-GH 6000 PSI supply valve
Meter-GH 6000 PSI ambient supply press.
Meter-GH 1500 PSI ambient supply press.
Ind. -GH cold Ht-Ex-1 vent-open
Ind. -GH cold Ht-Ex-1 vent-closed
Switch-GH cold Ht-Ex-1 vent valve
Switch-Ht. exchanger outlet temp ckt #1

Engine Preparation Panel

Delete:

Ind. -Eng start tank supply valve-open
Ind. -Eng start tank supply valve-closed
Switch- Eng start tank supply valve
Ind. -T/C chilldown supply-open
Ind. -T/C chilldown supply-closed
Switch-T/C chilldown supply valve
Ind. -Start tank purge supply valve-open
Ind. -Start tank purge supply valve-open
Switch-Start tank purge supply valve
Ind. -Start tank supply vent-open
Ind. -Start tank supply vent-closed
Switch-Start tank supply vent valve

10.5.1.4.13 (Continued)

3. Engine Turbine Start Tank Pressurization System

The start tank pressurization system is used to spin the fuel turbine for the engine start sequence. The J-2S engine uses a solid propellant turbine start (SPTS) technique, thereby eliminating the start bottle and all associated valves and controls. The interface between this system and the ESE are the umbilical lines. The deletions in the umbilical functions and their effect on the ESE models are listed below:

(a) At HB

Prop Sys Digital Control Console VCL 1 and 2 (Mainstage Manual Control Panel-A2

Delete:

Ind. -Start tank vent open-cmd	3 patchcords deleted
Switch-Start tank vent pilot valve open-close	3 patchcords deleted
Ind. -Start tank emergency vent command-on	2 patchcords deleted
Switch-start tank emergency vent valve open-close	5 patchcords deleted

(b) At STC

Propulsion System Control Console-VCL (Mainstage Manual Control Panel-A2)

Delete same functions as at HB above.

Remote Pneumatic Control Console-Beta 1 and Beta 3 (Vehicle Supply Panel-A2)

Delete:

Ind. -Start tank dump-open	3 patchcords deleted
Ind. -Start tank dump-closed	4 patchcords deleted
Switch-Start tank vent valve	5 patchcords deleted
Ind.-Start tank emergency dump stand	2 patchcords deleted
Ind. -Start tank emergency dump stage	4 patchcords deleted
Switch-Start tank emergency dump valve	15 patchcords deleted

10.5.1.4.13 (Continued)

(c) At KSC

Engine Test Panel (406A5)

Delete:

Ind.-Comp test st tk discharge valve-open
Ind.-Comp test st tk discharge valve-closed
Ind.-Start tank sol energized
Switch-Comp test start tank discharge valve

Engine Preparation Panel (406A6)

Delete:

Switch-Start tank vent pilot valve
Switch-Start tank emergency dump valve
Meter-Start tank temperature
Meter-Start tank pressure

4. Chilldown System

The thrust chamber chilldown system maintains the proper percentage of saturated liquid propellant at the turbine inlets to prevent a turbine stall during engine start. The idle mode capability of the J-2S makes the entire chilldown system unnecessary. Items such as the chilldown pumps, chilldown inverters, prevalves, valves, frequency converter and transducers are eliminated. The deletions appear to the ESE as umbilical pin function deletions which require many patching deletions and control panel changes. These changes are outlined below.

The patchcord deletions under Electrical C/O Accessory Kit VCL #1 and 2 are those which do not affect any manual control panels but affect interfaces with the computer and various digital or analog display devices:

(a) At HB

Electrical C/O Accessory Kit VCL #1

Electrical C/O Accessory Kit VCL #2

Delete the following umbilical functions:

10.5.1.4.13 (Continued)

Umb. Pin	Function	
404W2J1-1, 2, 3	Meas-LOX C/D pump diff pressure	6 patchcord deletions
404W2J1-18	Ind.-LH_2 C/D Valve open	2 patchcord deletions
404W2J1-34	Ind.-LOX C/D Valve closed	2 patchcord deletions
411W1J1-1, 2, 3	Ind.-LH_2 C/D pump diff pressure	6 patchcord deletions
404W15J2-N	Ind.-LOX C/D pilot relay reset	2 patchcord deletions
404W15J2-P	Ind.-LH_2 C/D pump pilot relay reset	2 patchcord deletions
404W15J2-Q	Ind.-LOX C/D pump inverter power on	2 patchcord deletions
404W15J2-V	Ind.-LH_2 C/D pump inverter power on	2 patchcord deletions

Propulsion System Control Console (Mainstage Manual Control Panel)

Delete:

Ind.-LH_2 pre-valve-closed	2 patchcords deleted
Ind.-LH_2 pre-valve-open	2 patchcords deleted
Ind.-LOX pre-valve-open	2 patchcords deleted
Ind.-LOX pre-valve-closed	2 patchcords deleted
Switch-LH_2 and LOX pre-valves	5 patchcords deleted
Ind.-LH_2 and LOX C/D shutoff closed	3 patchcords deleted
Ind.-LH_2 and LOX C/D shutoff open	2 patchcords deleted
Switch-LH_2 and LOX C/D shutoff valves	5 patchcords deleted

(b) At STC, the following changes will be made:

Elect. C/O Accessory Kit-Beta 1

Elect. C/O Accessory Kit-Beta 3

Elect. C/O Accessory Kit VCL

Umbilical function deletions are same miscellaneous functions as the Associated Model at HB.

10.5.1.4.13 (Continued)

Propulsion System Control Console Beta 1 and 3 (Mainstage Propulsion Manual Control Panel-A2)

Delete:

Ind.-LOX pre-valve-open	3 patchcords deleted
Ind.-LOX pre-valve-closed	3 patchcords deleted
Ind.-LH_2 pre-valve-open	3 patchcords deleted
Ind.-LH_2 pre-valve-closed	3 patchcords deleted
Switch-LH_2 and LOX pre-valves	9 patchcords deleted
Ind.-LH_2 and LOX C/D shutoff-open	4 patchcords deleted
Ind.-LH_2 and LOX C/D shutoff-closed	5 patchcords deleted
Switch-LH_2 and LOX C/D shutoff valve	9 patchcords deleted

At KSC, the chilldown system deletions will have the following effects:

Recirculation Panel (406A1)

Delete entire panel.

Networks Panel (405A1)

Delete:

Ind.-LOX C/D relay reset
Ind.-LH_2 C/D relay reset
Ind.-LH_2 C/D inverter power off
Ind.-LOX C/D inverter power off
Ind.-Chilldown pumps safe

Engine Preparation Panel (406A6)

Delete:

Meter-Eng. thrust chamber jacket temp

Events Display Panel

Delete:

Ind.-Prevalve emergency close cmd
Ind.-S-IVB LOX chilldown
Ind.-S-IVB LH chilldown
Ind.-LOX chilldown inverter power on
Ind.-LH chilldown inverter power on

10.5.1.4.13 (Continued)

5. Pneumatic Control System

The stage pneumatic control system provides regulated gas for certain purge and control operations. Certain purge and control functions are not required on the J-2S engine, such as the start tank vent valve control, the LOX chilldown pump purge control, and the engine pump purge control. These functions were included in the discussion of the start tank system and the chilldown system need not be discussed further.

6. Ordnance System

The ordnance system consists of many EBW firing units and initiators used throughout the stage to fire explosive charges or solid propellant motors. Implementation of the J-2S engine deletes the requirement for the solid propellant ullage control rockets. This deletes a total of four EBW firing units and initiators because of the ignition and jettison systems. The J-2S engine has a new requirement for EBW firing units; however, with the solid propellant turbine start (SPTS) technique. There will be three SPTS gas generators furnished with the engine and each unit uses two EBW firing units and initiators. Therefore, the impact on the ESE will be an exchange of ullage rocket ignition and jettison functions with SPTS functions.

Another change resulting from the SPTS technique is the deletion of the gas generator which presently drives the LOX and LH_2 turbines after engine start. This change will result in further exchanges of gas generator functions with SPTS functions.

(a) At HB, the following ESE models are affected:

Electrical C/O Accessory Kit - VCL #1

Electrical C/O Accessory Kit - VCL #2

Delete the following umbilical functions:

404W1J1-48	CMD-EBW Ullage rkt fir unit disable	3 patchcords deleted
404W2J1-51	Ind.-EBW Ullage rkt fir unit enable	2 patchcords deleted
404W4J1-58	Ind.-EBW Ullage rkt relay reset	2 patchcords deleted

10.5.1.4.13 (Continued)

Add the following umbilical functions:

404W2J1-42	Ind. -Mon #1 SPTS ready	2 patchcords added
TBD	Ind. - Mon #2 SPTS ready	2 patchcords added
TBD	Ind. -Mon #3 SPTS ready	2 patchcords added
TBD	Meas -Mon #2 SPTS #1 EBW	9 patchcords added
TBD	Meas -Mon #2 SPTS #2 EBW	9 patchcords added
TBD	Meas -Mon #3 SPTS #1 EBW	9 patchcords added
TBD	Meas -Mon #3 SPTS #2 EBW	9 patchcords added

(b) At STC, the following changes will be made.

Electrical C/O Accessory Kit-Beta 1

Electrical C/O Accessory Kit-Beta 3

Electrical C/O Accessory Kit VCL

Additions and deletions are the same as for the Electrical C/O Accessory Kits at HB.

Engine Safety Cutoff Set Rack Assembly

The Rocketdyne furnished gas generator temperature high-low cutoff panel assembly (G1047) will be deleted entirely. In its place will be a SPTS monitor panel which will either be furnished by Rocketdyne or built by McDonnell-Douglas. This will be determined at a later date.

(c) At KSC, the following changes will be made:

EBW and Ordnance Panel

Re-identify:
Title "Ullage RKT Ignition" to "SPTS #1 EBWS"

10.5.1.4.13 (Continued)

Title "Ullage RKT Jettison" to "SPTS #2 EBWS"
All "Pulse Sensor Fired" indications to "SPTS #1 (or 2) Fired"
"Pilot Relays Reset" indication to "SPTS Armed"

Add:

Two meter circuits for "SPTS #3 EBWS"
Three Inds "SPTS #1 Ready," "SPTS #2 Ready" and "SPTS #3 Ready"
Two "SPTS #3 Fired" indications

Delete:

Ind.-Ullage RKT fir units enabled
Switch-Ullage RKT fir units enable-disable

Events Display Panel

Delete:

Ind.-Ullage RKT fir unit enabled
Ind.-Ullage RKT ignition fir unit #1 fired
Ind.-Ullage RKT ignition fir unit #2 fired
Ind.-Ullage RKT jettison fir unit #1 fired
Ind.-Ullage RKT jettison fir unit #2 fired

7. Auxiliary Propulsion System

Since the J-2S engine has an idle-mode capability, the 72-lb thrust ullage engines are no longer required. The effect on the electrical GSE is minimal, however, since there is only one umbilical talkback involved. This impact is shown below.

(a) At HB and STC

Electrical C/O Accessory Kit

Delete the following umbilical function.

404W4J1-35	Ind.-Ullage relay reset 70-lb	2 patchcords deleted in each area

(b) At KSC, one monitor is affected.

APS Launch and Monitor Panel (419A1)

10.5.1.4.13 (Continued)

Delete:

Ind.-70-lb thrust ullage engine relay reset

8. Hydraulic System

Engine gimballing is accomplished with hydraulic actuator assemblies connected to an engine driven hydraulic pump. Implementation of the J-2S engine will necessitate a new pump assembly but this change will not affect the ESE in any way.

9. Propellant Utilization System

The propellant utilization system measures the quantity of propellant remaining in each tank and controls the engine mixture ratio according to certain pre-programmed levels to achieve maximum stage payload and also to achieve simultaneous depletion of propellants. This system also provides signals to the engine cutoff logic for cutoff at certain minimum propellant levels.

The J-2S engine has the capability of running to LOX depletion thereby eliminating the requirement for the LOX depletion cutoff logic circuitry. This will cause patching changes at HB and STC and control panel changes at KSC.

(a) At HB and STC

Electrical C/O Accessory Kits

Delete the following umbilical functions:

404W2J1-38	Ind.-LOX point level sens #1 wet cond on	2 patchcords deleted
404W2J1-37	Ind.-LOX point level sens #2 wet cond on	2 patchcords deleted
404W2J1-36	Ind.-LOX point level sens #3 wet cond on	2 patchcords deleted
404W3J1-8	Cmd.-Sim LOX pt lvl sens #1 wet cond on	2 patchcords deleted
404W3J1-7	Cmd.-Sim LOX pt lvl sens #2 wet cond on	2 patchcords deleted
404W3J1-14	Cmd.-Sim LOX pt lvl sens #3 wet cond on	2 patchcords deleted
404W4J1-57	Ind.-LOX point level sens #4 wet cond on	2 patchcords deleted

10.5.1.4.13 (Continued)

404W1J1-55	Cmd -Sim LOX pt lvl sens #4 wet cond on	5 patchcords deleted

(b) At KSC

Engine Test Panel (406A5)

Delete:

Ind.-LOX sensor No. 1 wet
Ind.-LOX sensor No. 2 wet
Ind.-LOX sensor No. 3 wet
Switch-Simulate LOX sensor #1 wet
Switch-Simulate LOX sensor #2 wet
Switch-Simulate LOX sensor #3 wet

Re-identify:

Ind.-"LOX/LH Sensor #4 Wet" to "LH Sensor #4 Wet"

Delete switch position "LOX Sensor #4 Sim Wet"

Events Display Panel

Delete:

Ind.-LOX cutoff sensor #1 wet
Ind.-LOX cutoff sensor #2 wet
Ind.-LOX cutoff sensor #3 wet
Ind.-LOX cutoff sensor #4 wet

10. Instrumentation and Telemetry System

The instrumentation and telemetry system transmits all stage measurements to a Digital Data Acquisition Station (DDAS) for decommutation. Then the entire data field is stored in the computer memory so that any or all measurements may be printed out for analysis. Measurement changes resulting from the J-2S engine will not have any hardware effect on this part of the ESE.

There are a few measurements that are routed through the digital-to-analog converters in the DDAS. These measurements are then made available for oscillograph display or control panel indications. Any changes in these measurements will result in hardware changes such as patchcord deletions or monitor panel changes. The vast majority of measurements associated with the J-2S engine

10.5.1.4.13 (Continued)

implementation are not routed to graphs or panels, particularly at HB and STC.

The impact of these measurement changes on the ESE is shown below.

(a) At HB, there will be only three measurement exchanges and no hardware changes.

Exchange K6 "ignition phase control sol energized ind." with TBD "idle mode control sol energized ind."

Exchange K11 "gas generator spark system ind." with TBD "SPTS initiated ind."

Exchange K96 "start tank discharge control ind." with TBD "mainstage start control sol on"

(b) At STC, there will be four measurements deleted at Beta 1 and 3 which affect patchcords only.

Electrical C/O Accessory Kit - Beta 1

Electrical C/O Accessory Kit - Beta 3

Delete:

D218	LH_2 chilldown pump diff pressure	3 patchcords deleted
D219	LOX chilldown pump diff pressure	3 patchcords deleted
F4	LOX circ pump flow rate	6 patchcords deleted
F5	LH_2 circ pump flow rate	6 patchcords deleted

(c) At KSC, there will be four monitor panels affected. These functions have been covered in the previous discussions on the chilldown system, start bottle system and the ordnance system but will be repeated here for clarity.

Recirculation Panel (406A1)

Delete:

D103	Meter-LOX pump cav pressure
F4	Meter-LOX flow rate
F5	Meter-LH flow rate

10.5.1.4.13 (Continued)

Engine Test Panel (406A5)

Delete:

K6	Ind.-Ignition phase cont sol energ.
K11	Ind.-Gas gen spark sys on
K96	Ind.-Start tank disch sol energ
K105	Ind.-Eng. pump purge press not high
K117	Ind.-Gas gen valves open
K122	Ind.-Start tank disch valve open
K124	Ind.-LOX turbine bypass valve open
K125	Ind.-LOX turbine bypass valve closed

Engine Preparation Panel (406A6)

Delete:

C6	Meter-Eng start tank temp
C7	Meter-Eng helium cont bottle temp
C199	Meter-Eng thrust cham jacket temp
D17	Meter-Eng start tank press
D19	Meter-Eng helium cont bottle press
D50	Meter-Eng pump purge press

EBW and Ordnance Panel (409A1)

Delete:

K149	Ind.-Ullage RKT jettison fir unit #1 pulse sensor fired
K150	Ind.-Ullage RKT jettison fir unit #2 pulse sensor fired
K176	Ind.-Ullage RKT ignition fir unit #1 pulse sensor fired
K177	Ind.-Ullage RKT ignition fir unit #2 pulse sensor fired
M64	Meter-Ullage RKT ignition fir unit #1 charge voltage
M65	Meter-Ullage RKT ignition fir unit #2 charge voltage
M66	Meter-Ullage RKT jettison fir unit #1 charge voltage
M67	Meter-Ullage RKT jettison fir unit #2 charge voltage

c. IU Electrical Support Equipment

The purpose of the IU ESE paragraph is the definition of ESE modifications required by the J-2S engines used on a Saturn V vehicle for a LOR mission. The ESE modifications were determined by considering the changes necessary to uprate a baseline AS 505 IU for use on the LOR mission. Implementation of the J-2S engines on a Saturn V vehicle and the mission requirements of a LOR mission

10.5.1.4.13 (Continued)

will require no hardware modifications to the ESE since the LOR mission requirements add no additional hardware or hardware requirements. The software will require minor changes in the automated subsystem checkout programs and the IU overall checkout program.

1. Automated Subsystem Checkout Programs

The following subsystem checkout programs will require test parameter changes and/or minor program rewrite for the LOR mission.

Control Subsystem

A_1 Gain
A_0 Gain
Control Systems Nulls
Control Relay Redundancy
Engine Deflection

Electrical Subsystem

Power Distribution and Control
General Networks
Simulated Plug Drop

2. IU Overall Checkout Program

The IU overall checkout program is a general program applicable to all Saturn V vehicles with little or no modification from one vehicle to another. The mission requirements of the LOR mission are such that the basic checkout program does not require modification. All changes due to the LOR mission can be loaded into the basic program via user controlled data tables. These tables can be updated and maintained to the latest mission requirements to be inserted into the basic program without impacting that program.

Similar programs at KSC will also require updating.

10.5.1.4.14 Propellant Management

The propellant management system monitors propellant mass for control of propellant loading, utilization and depletion. Components of the system include continuous capacitance probes, propellant utilization valves, liquid level sensors and electronic equipment.

10.5.1.4.14 (Continued)

The propellant utilization provides for control of propellant consumption during engine burn periods by controlling the usage ratio of the LOX to assure minimum propellant residuals at termination of flight. The subsystem is capable of reducing the stage propellant residual to less than 0.25 percent of the total propellant mass loaded (or a three sigma of 575 lbs), under the assumption that the engine burns until a propellant depletion signal is received. The propellant utilization subsystem is activated during flight for first and second burn, beginning shortly after engine start and is deactivated after engine cutoff. During second burn, normal cutoff will be initiated by either the IU or by a signal from the tank depletion sensors.

a. S-II Propellant Management

Incorporation of the J-2S engine on the S-II stage will require no additional hardware impact to the propellant utilization system.

b. S-IVB Propellant Management

Since the response of the propellant utilization system depends on the mainstage performance of the engine, a comparison was made between the J-2 and J-2S engine performance characteristics. A comparison between the J-2 engine propellant utilization valve characteristics presently being used for closed loop propellant utilization system operation and the estimated J-2S engine propellant utilization valve characteristics indicates that the nonlinear shape of propellant utilization valve characteristics will not change significantly. The propellant utilization valve LOX corrective flow gain for the J-2S engine is near 83.5 lbs LOX per second per EMR at a bridge gain ratio of 5.0:1.0. This gain is based on the J-2S engine weight flow versus engine mixture ratio characteristics presented in the J-2S Interface Criteria Document of May 7, 1968. The gain is also shown to be independent of the calibrated thrust level of the J-2S engine. The corresponding LOX corrective flow gains for the J-2 engines presently being used for closed loop propellant utilization system operation are within one db of the J-2S values. This small gain difference indicates that, although the J-2S engine thrust and flow characteristics have been increased over the characteristics of the J-2 engine, the LOX corrective flow gain does not display any significant change. Therefore, a ± 4.5 db adjustment presently available in the propellant utilization system motor loop feedback network provides adequate propellant utilization system gain adjustment capability without propellant utilization system modifications.

The transfer of the LOX cutoff responsibility to the J-2S engine results in a change in the amount of trapped and unavailable LOX which must be used in revising the end point of the LOX calibration. No other changes are required.

10.5.1.4.14 (Continued)

Closed loop propellant utilization system operation is recommended by the S-IVB stage contractor for the J-2S/S-IVB stage during the LOR mission. No propellant utilization system modifications are required to utilize the J-2S engine. However, the open loop low EMR command for engine restart will not be required. Recommended propellant utilization system operation is based on the Boeing Company design trajectories and status report.

The recommended modes of propellant utilization system operation for the LOR mission are presented below:

Burn Period	Mode of Operation	Commanded EMR
First	Open Loop	5.0/1.0
Second	Closed Loop	5.0/1.0

The open loop mode of operation during first burn eliminates the need of an LH_2 boiloff bias and, therefore, the sequencing associated with the bias.

c. IU Propellant Management

Since the IU only issues sequence commands to the stage for Propellant Management, no hardware changes will be associated with J-2S incorporation.

Utilizing the closed loop propellant utilization system recommended by the S-IVB stage contractor, however, will require software modifications to the IU software. Present Saturn vehicle design has the capability of placing into orbit, with the open loop propellant utilization system, payloads specified in the J-2S study. The open loop system is a preset mixture ratio to be activated by the IU Switch Selector at a predetermined time of flight. Continuing with the open loop propellant utilization system as flown on AS 503 would be more appropriate relative to guidance software modifications.

10.5.2 J-2S/Synchronous Astrionic System Interface

10.5.2.1 Purpose

This paragraph of the J-2S Improvement Study (Astrionic System Integration Synchronous Mission) defines the differences between the three-stage Saturn V vehicle J-2S/SA 511 (Reference Figure 10.5.2.1-1) configured for the Synchronous Orbit Mission (Reference Figure 10.5.2.1-2) and the three-stage Saturn V vehicle J-2S/AS 511 configured for Lunar Orbital Rendezvous (defined in paragraph 10.5.1).

If an existing system is changed only the changes or additions are described in this document and the Astrionic Handbook, Saturn V Flight Manual and/or the J-2S/LOR Astrionic System Report (para. 10.5.1) is referenced for a more detailed description. Modification peculiar to J-2/J-2S interface will be so noted. Each illustration in this paragraph is referenced to a corresponding illustration in the document from which it was taken.

10.5.2.2 J-2S Astrionic System (Synchronous Orbit)

The overall Astrionic System of the J-2S/Synchronous AS 511 Vehicle is shown in simplified block diagram Figure 10.5.2.2-1. The major portion of the Astrionic System is located in the Instrument Unit (IU), which is mounted on top of the S-IVB stage. Basic Astrionic System Functions are explained in paragraph 10.5.1.2 of the J-2S/LOR section.

The operational lifetime of the IU is limited by the capacity of the power supply (batteries) and the water supply of the environmental control system.

Characteristics of the Synchronous Orbit mission vehicle are:

> The IBM study report entitled "IU Lifetime Extension to Meet Requirements of a Saturn V Synchronous Orbit Mission" (Reference 10.4-39) is used as a basis for the mission description and constraints.
>
> The Synchronous Orbit mission will consist of from 0.5 to 7.5 hours in low-earth waiting orbit, 5.5 hours in Hohmann transfer to Synchronous altitude, and 2.0 hours at Synchronous altitude. Thus, three S-IVB burns are required with a maximum IU lifetime of 15 hours.
>
> An unlimited hover point capability is required for the equatorial case and for Synchronous Orbits inclined up to 64°. The IU modifications must meet the requirements of this spectrum of missions to permit flexibility in mission planning.

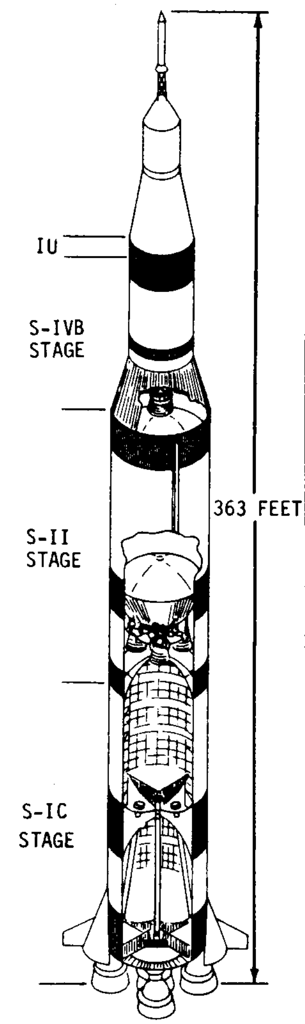
SATURN V LAUNCH VEHICLE

SOLID ULLAGE ROCKET AND RETROROCKET SUMMARY				
STAGE	TYPE	QUANTITY	NOMINAL THRUST AND DURATION	PROPELLANT GRAIN WEIGHT
S-IC	RETROROCKET	8	75,800 POUNDS • 0.541 SECONDS	278.0 POUNDS
S-II	ULLAGE	4	[illegible] POUNDS ‡ 3.75 SECONDS	[illegible] POUNDS
	RETROROCKET	4	34,810 POUNDS ‡ 1.52 SECONDS	268.2 POUNDS
S-IVB	ULLAGE	2	[illegible] POUNDS ‡ [illegible] SECONDS	[illegible] POUNDS

ENGINE DATA					
STAGE	QTY	ENGINE MODEL	NOMINAL THRUST EACH	NOMINAL THRUST TOTAL	
S-IC	5	F-1	1,526,500	7,632,500	
S-II	5	J-2	265,000	1,325,000	[illegible] SEC
S-IVB	1	J-2	265,000	265,000	[illegible] SEC

STAGE DIMENSIONS		
	DIAMETER	LENGTH
S-IC Base (including fins)	63.0 FEET	138 FEET
S-IC Mid-stage	33.0 FEET	
S-II Stage	33.0 FEET	81.5 FEET
S-IVB Stage	21.7 FEET	59.3 FEET
Instrument Unit	21.7 FEET	3.0 FEET

SATURN V STAGE MANUFACTURERS	
STAGE	MANUFACTURER
S-IC	THE BOEING COMPANY
S-II	NORTH AMERICAN-ROCKWELL
S-IVB	McDonnell Douglas Astronautics Co.
S-IU	INTERNATIONAL BUSINESS MACHINE CORP.



PRE-LAUNCH LAUNCH VEHICLE GROSS WEIGHT ≈ 6,368,000 POUNDS

• MINIMUM VACUUM THRUST AT 120°F

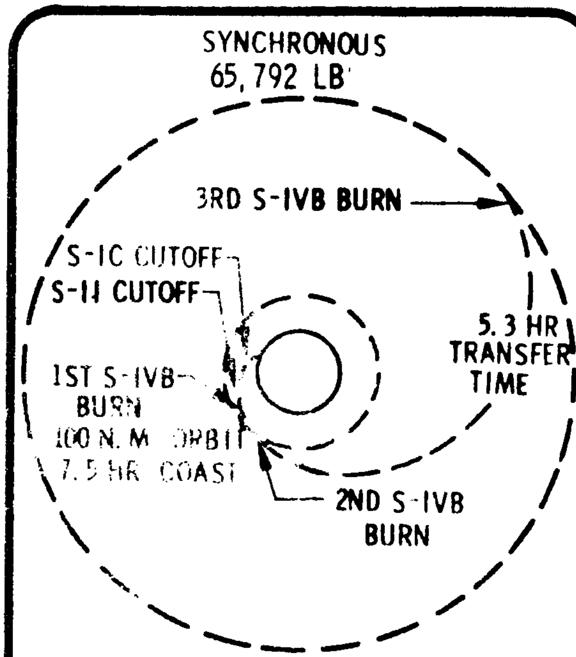
(shaded) AREA CHANGED

‡ NOMINAL VACUUM THRUST AT 60°F

NOTE: THRUST VALUES, WEIGHTS, AND BURN TIMES ARE ALL APPROXIMATIONS.

Ref. 10.4-32, Fig.1-3

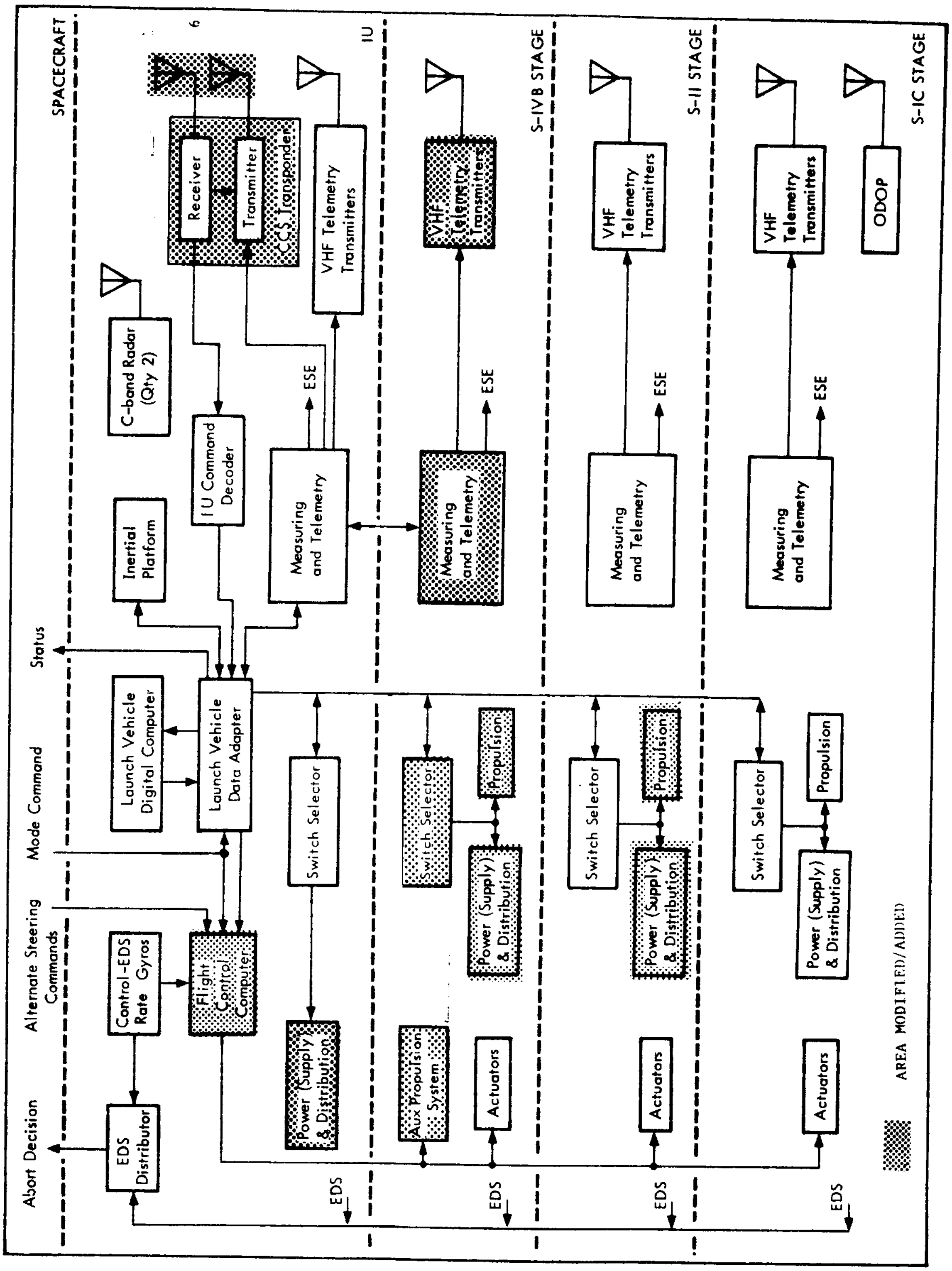
FIGURE 10.5.2.1-1. J-2S/LOR SATURN V LAUNCH VEHICLE



S-IVB
(1) IDLE MODE - 1 SEC
(2) MAINSTAGE TO ORBIT
(3) MAINSTAGE CUTOFF
(4) COAST 5 ORBITS - 7.5 HR
(5) IDLE MODE - 100 SEC
(6) MAINSTAGE TO TRANSFER TRAJECTORY
(7) MAINSTAGE CUTOFF
(8) COAST - 5.3 HR
(9) IDLE MODE - 100 SEC
(10) MAINSTAGE TO ORBIT
(11) MAINSTAGE CUTOFF

Ref 10.4-1

FIGURE 10.5.2.1-2. SYNCHRONOUS MISSION



Ref 10.4-31, Fig. 1.4-2

FIGURE 10. 5. 2. 2-1. SYNCHRONOUS ORBIT ASTRIONIC SYSTEM BLOCK DIAGRAM

10.5.2.2 (Continued)

During the Hohmann transfer, the vehicle will be in total sunlight and will be oriented with its roll axis perpendicular to the sun (S-IVB constraint).

During the Hohmann transfer, the vehicle will roll at 1 revolution per hour (S-IVB constraint).

The second S-IVB burn and third S-IVB burn must be such that ground coverage is maintained during the burns.

The IU must have a continuous communication capability during the Hohmann transfer and at Synchronous altitude.

10.5.2.3 J-2S/Synchronous Electrical Interface

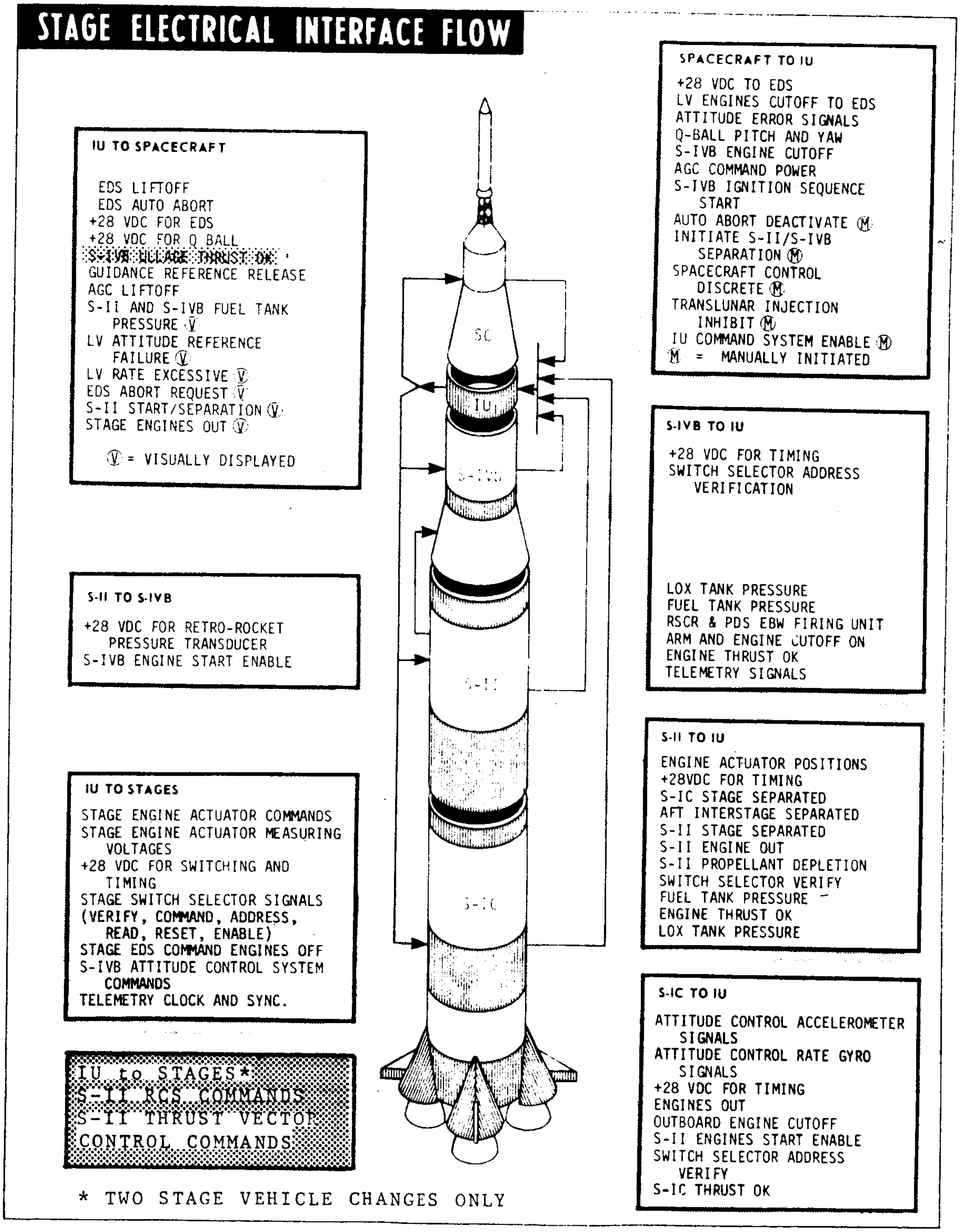
The J-2S/Synchronous Improvement Study launch vehicle is identified to the basic J-2S/LOR vehicle profile. The electrical interface is shown in Figure 10.5.2.3-1.

10.5.2.4 J-2S/Synchronous Astrionic Subsystems

10.5.2.4.1 Navigation and Guidance

The hardware for the J-2S/Synchronous Guidance System will not require modification. The basic guidance philosophy as given in the Astrionic System Handbook will not be changed for the J-2S/Synchronous mission. The flight program will require programming changes to guide the vehicle with the J-2S stages and to meet mission constraints. Program changes are discussed in more detail in paragraphs 10.5.2.4.3 and 10.5.2.4.10.

There will be no hardware impact modifications to meet the large yaw requirements of certain Synchronous Orbit missions. This requirement will be met by using a yaw bias technique. This technique involves intentional offset of the ST-124M Platform to take advantage of the available 90° range of yaw. (Reference 10.4-39)



Ref 10.4-34, Fig. 1-4

FIGURE 10.5.2.3-1. STAGE ELECTRICAL INTERFACE FLOW

10.5.2.4.2 Attitude Control

a. S-II Attitude Control

Performance of the S-II stage will be identical to the J-2S/LOR mission; therefore, no modifications will be required.

b. S-IVB Attitude Control

Converting from J-2S/LOR to J-2S/Synchronous missions will require no hardware modifications additional to those discussed in the J-2S/LOR report, paragraph 10.5.1.4.2 b.

c. IU Attitude Control

Converting from J-2S/LOR to J-2S/Synchronous missions will require no hardware modifications additional to those discussed in the J-2S/LOR report, paragraph 10.5.1.4.2 c.

10.5.2.4.3 Mode and Sequence

a. S-II Mode and Sequence

Converting from the J-2S/LOR to the J-2S/Synchronous mission will require no hardware impact additional to those described in the J-2S/LOR report, paragraph 10.5.1.4.3 a.

b. S-IVB Mode and Sequence

Converting from J-2S/LOR to the J-2S Synchronous mission requires twenty-two switch selector assignments in addition to those discussed in paragraph 10.5.1.4.3b, twelve of which are required for passivation. The remaining ten controls are related to the Electrical Equipment Heater Blankets and Tape Recorder. These additional switch selector channel assignments are listed in Table 10.5.2.4-I. See Figure 10.5.2.4-1.

Additional switch selector channel assignments for the J-2S/Synchronous Orbit are listed in Table 10.5.2.4-I.

c. IU Mode and Sequencing

Converting from J-2S/LOR to S-2S/Synchronous Orbit will require no additional hardware changes to the basic IU Mode and Sequencing System. However, as a result of modifications to the communication system modification the electrical system

Ref 10.4-32, Fig. 7-20

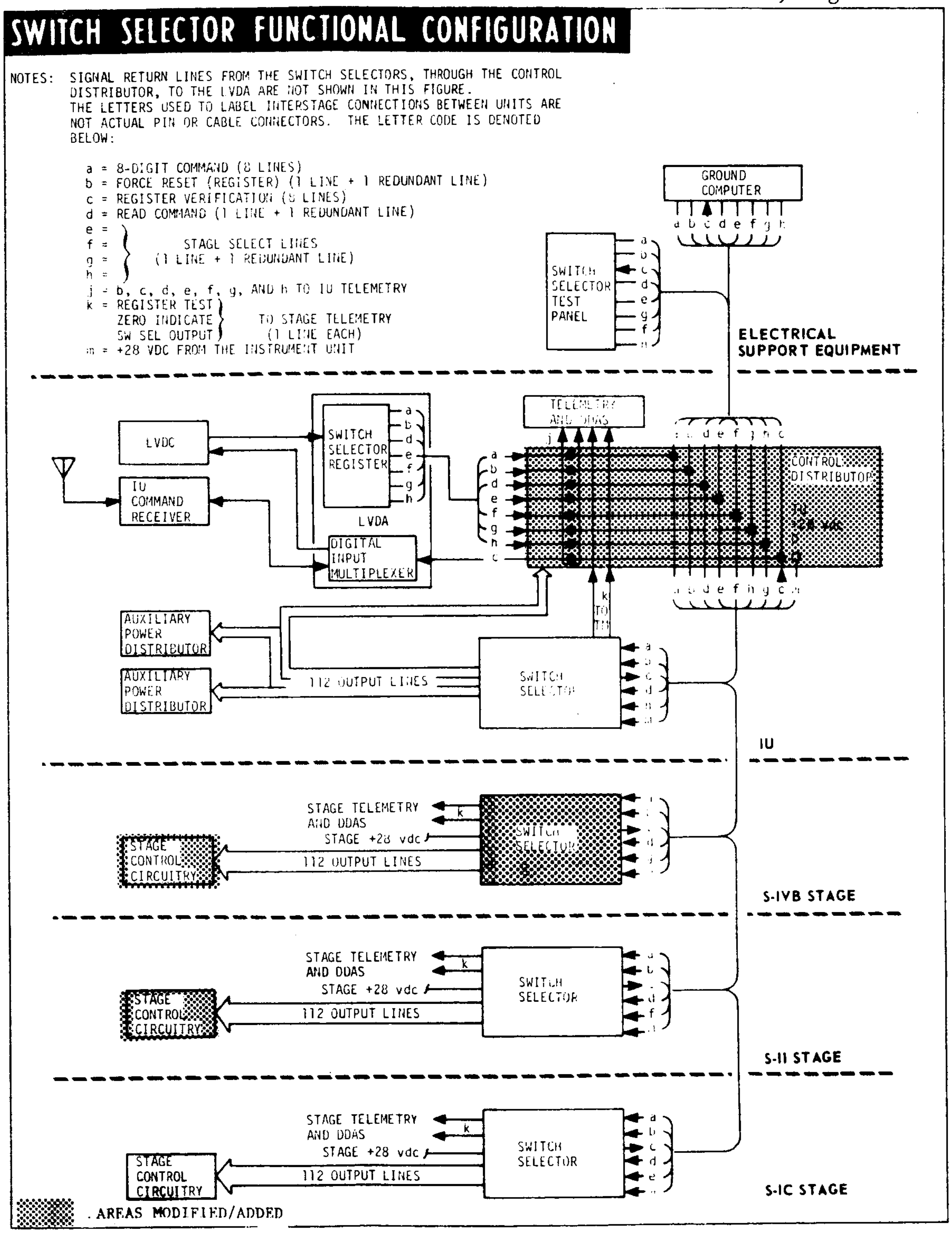


FIGURE 10.5.2.4-1. SYNCHRONOUS ORBIT SWITCH SELECTOR FUNCTIONAL CONFIGURATION

TABLE 10.5.2.4-I. ADDITIONAL SWITCH SELECTOR CHANNEL ASSIGNMENTS FOR THE SYNCHRONOUS MISSION

THERMAL CONDITIONING

1. Electrical Equipment Heaters Enable
2. Electrical Equipment Heaters Disable

TELEMETRY

3. Slow Record ON
4. Slow Record OFF
5. Record Playback ON
6. Record Playback OFF
7. Fast Record ON
8. Fast Record OFF
9. Emergency Playback Reverse ON
10. Emergency Playback Reverse OFF
11. Emergency He Dump ON*
12. Emergency He Dump OFF*
13. Pneumatic System Vent ON*
14. Pneumatic System Vent OFF*
15. Propellant Dump Solenoid Valve Closed*
16. Propellant Dump Solenoid Valve Open*
17. Idle Mode Control Valve (Solenoid) Open*
18. Idle Mode Control Valve (Solenoid) Closed*
19. Mainstage Start Control Valve Open*
20. Mainstage Start Control Valve Closed*
21. Mainstage Control Valve Open*
22. Mainstage Control Valve Closed*

* Required for In-Orbit Passivation

10.5.2.4.3 (Continued)

cabling and sequencing logic must be modified due to the new system requirements and the relocation of components.

The sequencing logic for the three coax switches will be similar to the CCS coax switch sequencing logic used on S-IU-506. Three switch selector commands will be used to select one of the three outputs of each coax switch. ESE simulate commands will be provided for each of the switch selector outputs. The position of each coax switch will be monitored by two telemetry measurements. Each measurement will indicate one of the energized positions. The absence of both measurements will indicate the de-energized position of the switch. A total of nine ESE commands will be required to simulate the nine switch selector commands. See Figure 10.5.2.4-1.

To prevent antenna pattern interaction, only one power amplifier at a time will be active. The power amplifiers will be enabled and inhibited by switch selector commands. Two separate commands are required for each power amplifier to prevent loss of communication during the transition from the inhibited to the enabled stage. A 15-second warm-up period is required before the standby amplifier can become operational. A total of four switch selector commands and four ESE simulate commands will be required for the power amplifier enabling and inhibiting. Telemetry measurements will monitor the status of each power amplifier enabling/inhibiting circuit.

Approximately fifteen switch selector commands will be required to initiate the desired sequencing and inhibiting of the power amplifiers and antennas. Approximately ten relays and thirty-five diodes will be required in the distributors for the sequencing and inhibiting logic.

Also, two separate switch selectors commands will be required to shut down IU power at the end of IU lifetime.

Refer to paragraph 10.5.2.4.10 for the Flight Sequence table.

10.5.2.4.4 Measurement and Telemetry

a. S-II Measurement and Telemetry

Converting from J-2S/LOR to J-2S/Synchronous mission will require no modification additional to those discussed in the J-2S/LOR report, paragraph 10.5.1.4.4 a.

10.5.2.4.4 (Continued)

b. S-IVB Measurement and Telemetry

Measurements associated with the S-IVB J-2S engine are described in J-2S/LOR report, paragraph 10.5.1.4.4b. A comparison of the operational telemetry lists for the S-IVB/J-2 and S-IVB/J-2S stages indicate the S-IVB/J-2S stage will require 47 additional measurements, 54 deletions, and range changes for 9 measurements. During R&D phases of the Synchronous program, it is anticipated that an additional 166 measurements will be required or a net change of 112 measurements. The deletions fall into the same category as for the LOR mission. The operational additions all fall into the same category as for the LOR mission. The R&D measurements are a general augmentation of the engine, APS and tankage instrumentation. The present instrumentation system cannot accommodate this many new measurements without additional equipment. A brief discussion of these changes is found in the following paragraphs.

The present system must be augmented to telemeter the additional Synchronous Orbit measurements. The recommended system is outlined below and shown in Figure 10.5.2.4-2.

Addition of 2 Model 270 Multiplexers.
Addition of 2 Low Level Remote Submultiplexers.
Re-Program the Model 301 PCM/DDAS Assembly for Time Sharing 3 Model 270 Multiplexers in the Present CP Multiplexer Time Slot.

All present measurements on the CP multiplexers will be cross-strapped to the same channels on the two added Model 270 Multiplexers. This will maintain a sample rate of 120 or 12 sps for these measurements. Prime channels 6, 7, 8 and 15 are opened up for Remote Analog Multiplexer inputs. As a sample rate of 120 sps will satisfy the requirements of the Synchronous Orbit measurement, provisions are available, on each of the three multiplexers, for 4 remote submultiplexer inputs (40 low-level measurement) and 14 high-level (0-5V) inputs to be measured at 4 sps. Considering all three multiplexers, this will provide for 120 temperature measurements, via remote analog multiplexing and 42 other measurements conditioned and inserted in the Model 270 Multiplexer for measurement at a 4 sps rate.

The vehicle RF and/or TM systems must be modified if they are to be used to facilitate data recovery during the final portion of the transfer orbit and the final burn of the S-IVB. Calculations based on the current 18-watt configuration of the Saturn V and utilizing the TLM 18 tracking net indicate that threshold will occur at a nominal altitude of 5,000 miles.

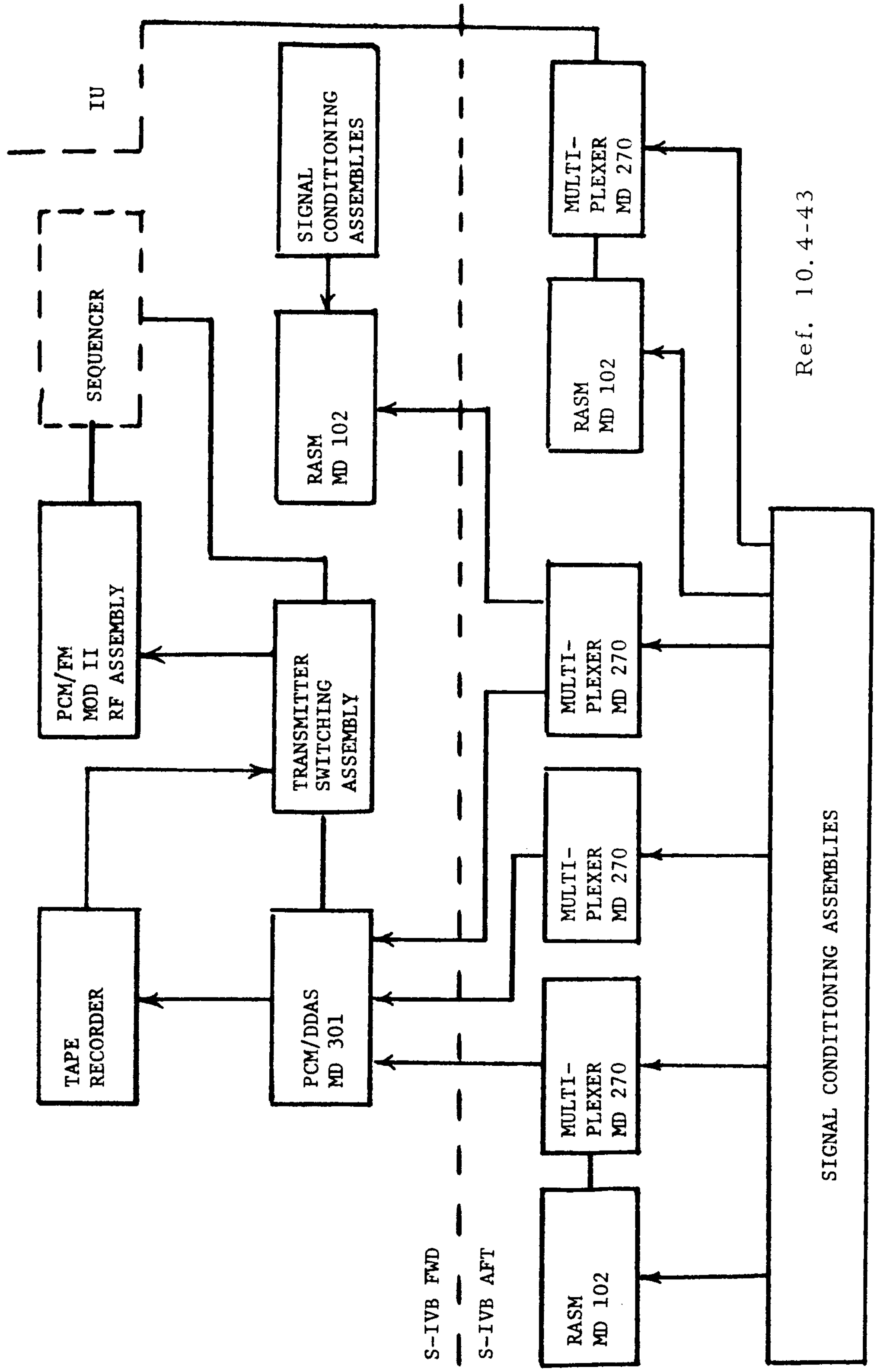


FIGURE 10.5.2.4-2. S-IVB SYNCHRONOUS ORBIT MEASUREMENT AND TELEMETRY

10.5.2.4.4 (Continued)

Since the vehicle is experiencing a roll during the transfer orbit maneuver and its axis is perpendicular to the sun, a ground station may fall anywhere in the complete sphere of the radiation pattern of the vehicle. Further, since at 20,000 miles distance the earth subtends an angle of 28° with respect to the vehicle, the entire earth (and therefore all antennas on earth) could fall within a single null of the pattern. Therefore, it must be assumed that to completely preclude the loss of data, the system must be improved sufficiently to assure ground station look in a worst-case condition. The worst nulls in the vehicle pattern go to - 18 dbm with respect to an isotrophic radiator.

With the existing system characteristics, there is an 80% probability of receiving data at a single TLM-18 ground receiver. With two ground receivers there is a 96% probability of receiving data. With three ground receivers the probability rises to 99.2%.

Six systems were analyzed in the Synchronous Orbit study. The S-IVB contractor recommended system adds two 1A77080 40 watt amplifiers, one at each antenna, using the Mod II RF assembly, as drivers. This would increase the antenna power from a nominal value of 4.4 watts (22 watts nominal transmitter output through a nominal 7 db insertion loss) to 40 watts for an improvement of approximately 10 db. This system is shown in Figure 10.5.2.4-3.

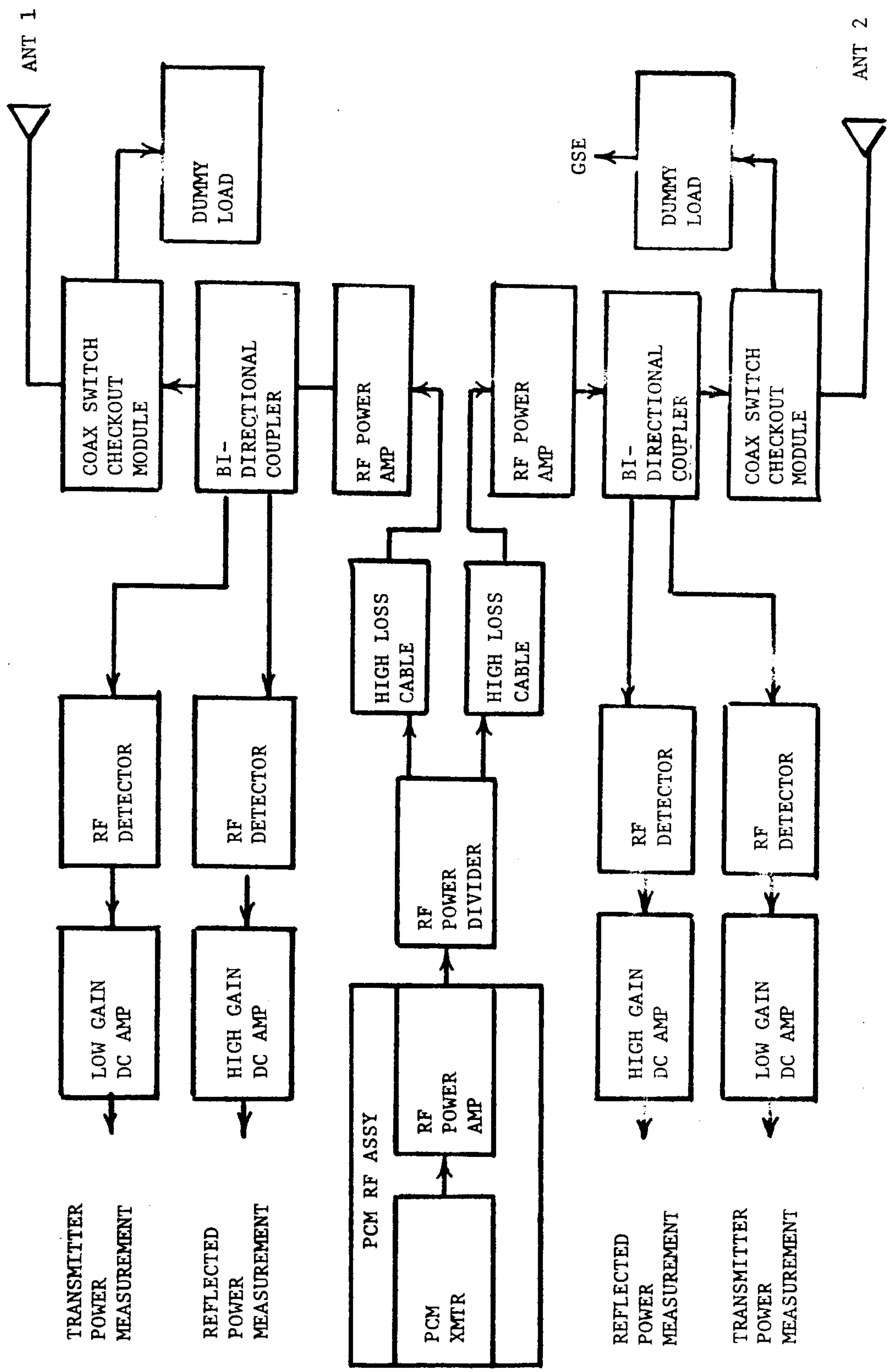
c. IU Measurement and Telemetry

The S-IU-511 baseline IU contains four Measuring Racks with a total of 80 signal conditioner slots. The S-IU-506 IP&C list contains measurements which use 73 of the 80 signal conditioners.

There are many measurements (e.g., vibration and battery temperature) which would be desirable from a postflight analysis standpoint. The addition of these measurements would have resulted in the requirement for a fifth Measuring Rack. However, since none of these could be considered in the mandatory category (a mandatory measurement is defined here as one which if not obtained might jeopardize the lives of the crew) it was not recommended that these measurements be added to the IP&C list for the Synchronous Orbit mission.

Hardware modification for the J-2S/Synchronous mission CCS telemetry are discussed in paragraph 10.5.2.4.5 c IU Radio Command System.

The Mission Control Data will be cross-strapped between the S-IVB and IU as implemented in the baseline vehicle. No hardware modifications are necessary in this area.



Ref 10.4-43

FIGURE 10.5.2.4-3. S-IVB SYNCHRONOUS ORBIT RADIO FREQUENCY SYSTEM

10.5.2.4.5 Radio Command

a. S-II Radio Command

Converting from J-2S/LOR to J-2S/Synchronous will require no hardware modifications to the S-II Radio Command. More detail is given in paragraph 10.5.1.4.5.

b. S-IVB Radio Command

Converting from J-2S/LOR to J-2S/Synchronous will require no hardware modification to the S-IVB Radio Command.

c. IU Radio Command System

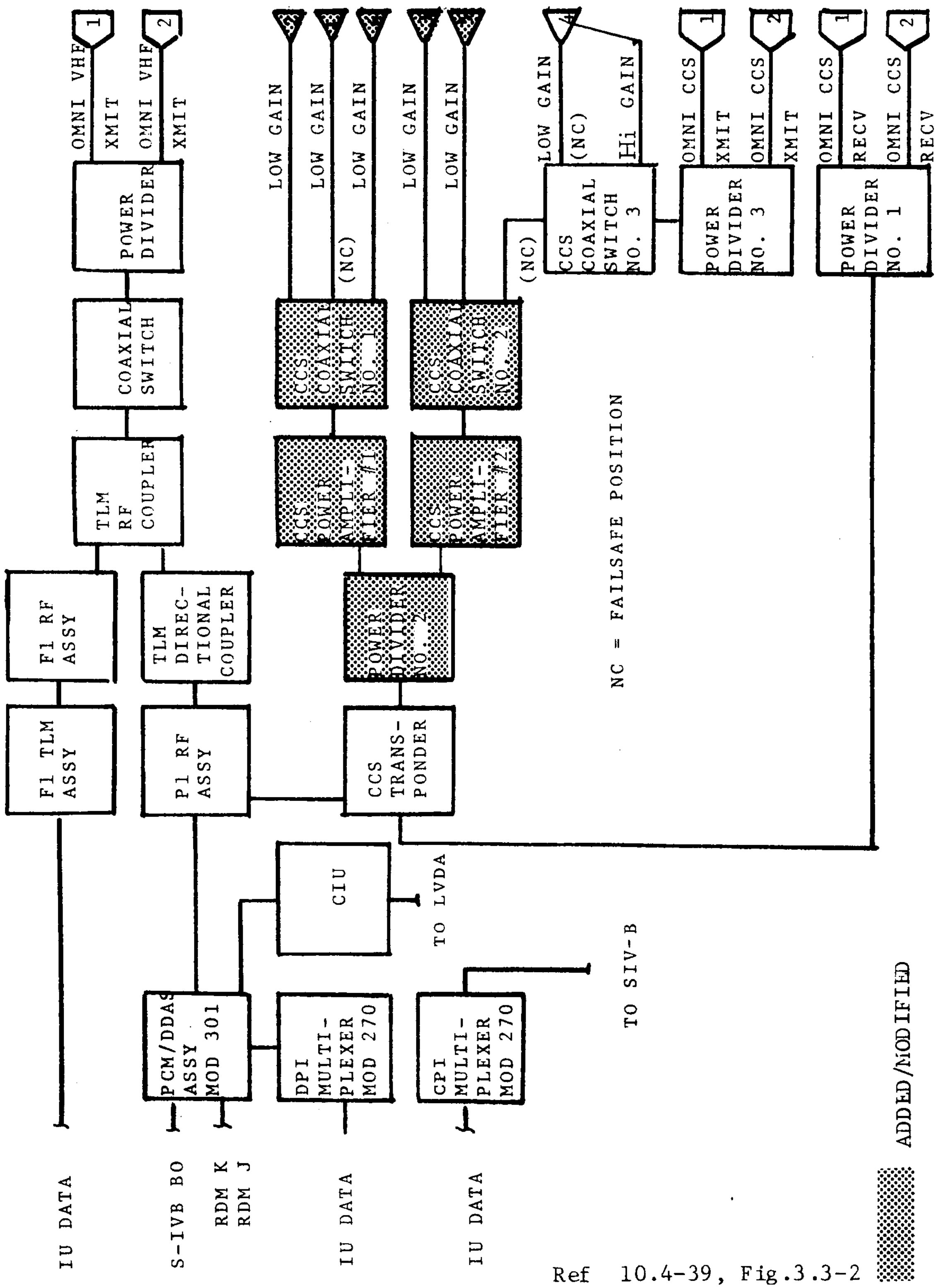
The combination of S-IVB stage constraints (to be oriented with the roll axis perpendicular to the sun and roll at one revolution per hour) for a continuous communication capability during Hohmann transfer impose new requirements on the CCS system antenna configuration. The present CCS omni capability is insufficient at altitudes encountered in this mission. Further, the single CCS directional antenna cannot be pointed during the Hohmann transfer because of S-IVB stage constraints. Mission requirements based on vehicle to ground station positional relationships, showed that the IU required a capability to communicate in a donut-shaped pattern around the IU. The conclusion was reached that by using proper combinations of pitch and yaw to maintain the sun pointing during the Hohmann transfer, a 70° width of the donut-shaped pattern (nose to tail) would be sufficient.

To meet the communications requirement imposed during the Hohmann transfer, a configuration employing six of the present CCS directional antennas is recommended. The selection of a single antenna for "pointing" purposes will be controlled by the LVDA/LVDC and Switch Selector. To maintain satisfactory circuit margins, two modified Power Amplifiers are used. A power divider and coax switches are added to permit antenna selection.

IU hardware measurement and telemetry modifications as a result of converting from J-2S/LOR to J-2 S/Synchronous mission are summarized as follows (Figure 10.5.2.4-4).

The current CCS Power Amplifier will be replaced by two modified Power Amplifiers.

Five additional CCS directional antennas will be added.



Ref 10.4-39, Fig.3.3-2

FIGURE 10. 5. 2. 4-4. SYNCHRONOUS ORBIT CCS FUNCTIONAL BLOCK DIAGRAM

10.5.2.4.5 (Continued)

One additional power divider will be installed.

Two additional coaxial switches will be added.

The Modification of the RF down-link subsystem was limited to equipment in the UHF band. Limiting improvements to the UHF band was chosen because the VHF hardware did not offer as much improvement potential in gains per unit size, ground network coverage, etc. and to conform with the FCC direction to abandon the VHF spectrum by 1970.

Circuit margin analysis of the CCS system indicated that with a nominal 20-watt power amplifier a minimum vehicle antenna gain of + 4.5 db would be required for satisfactory reception of the down-link PCM telemetry. This requirement was based on:

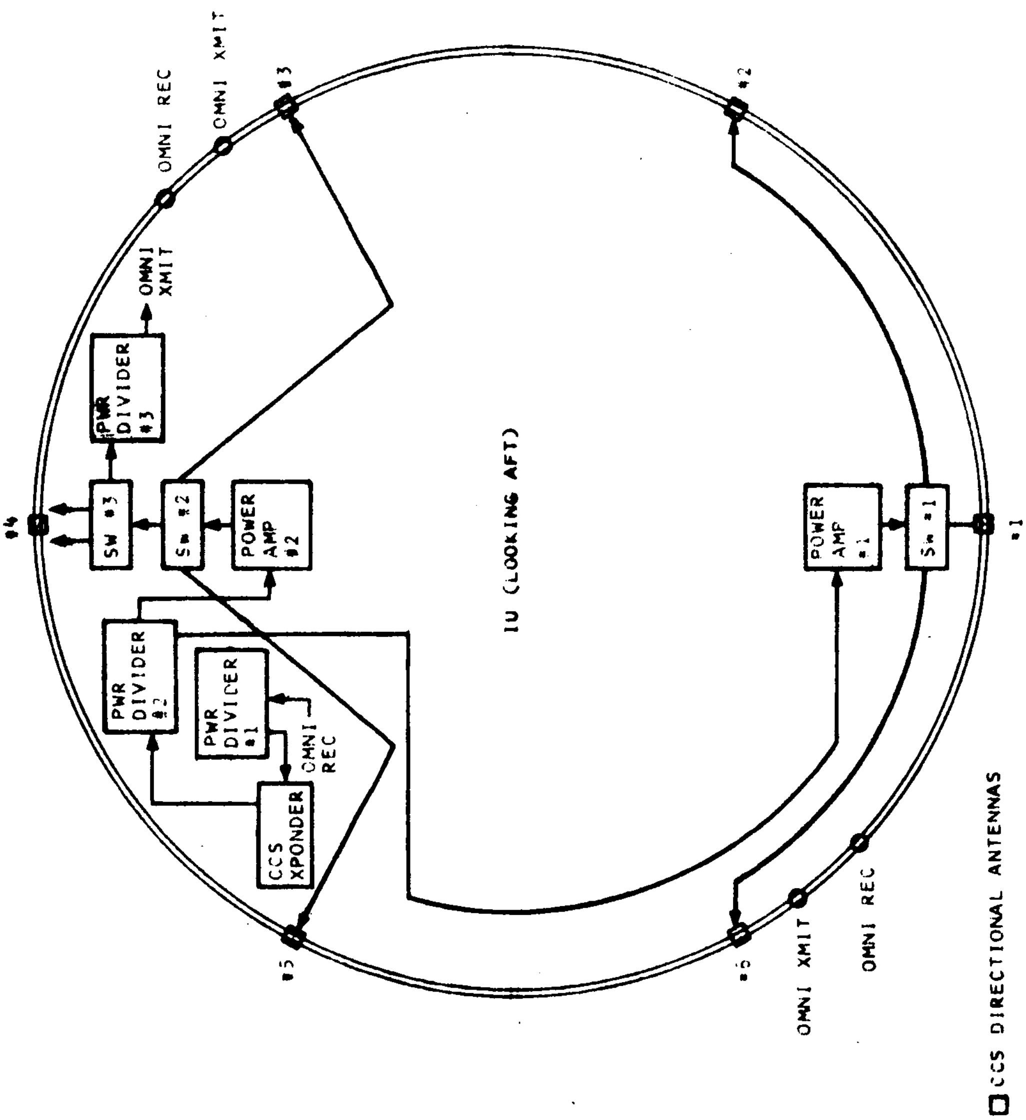
Synchronous altitude, 5^{o} elevation angle.

30-foot dish ground antennas.

Continuous vehicle roll at one revolution per hour.

The spectrum of mission constraints dictated the antenna beam pattern or coverage necessary. Essentially, this is an omni-directional antenna system with a required minimum gain of + 4.5 db.

A basic problem in achieving this type of omni coverage is that any final design will turn out to be non-omnidirectional. The use of multiple elements is mandatory to minimize the shadowing effects of the vehicle. The use of elements around such a large vehicle, although posing problems, provides an adequate radiation pattern. Feeding these antennas in parallel simplifies the system design and operational use, but usually produces interference pattern lobing with deep nulls and wastes power through power division to the antennas and resultant transmission not beamed at earth stations. The optimum omnidirectional antenna system for this type of application, then, is not omnidirectional at all but provide for maximum energy transfer in only one direction at any given time. The antenna system chosen for the Synchronous mission utilizes six CCS directional antennas spaced a nominal 60^{o} apart around the IU. These antennas operated in the low gain configuration provide the desired beam width and gain. Two power amplifiers are used to minimize RF power losses in the distribution system. See Figure 10.5.2.4-5 for a schematic of the CCS RF component layout. A functional block diagram of the instrumentation is seen on Figure 10.5.2.4-4.



Ref 10.4-39, Fig. 3.3-1

FIGURE 10. 5. 2. 4-5. CCS ANTENNA LOCATION

10.5.2.4.5 (Continued)

Power amplifier and antenna switching is accomplished under control of the LVDC program. Switch selector functions are used in conjunction with coaxial switches to control the beam direction relative to the vehicle. Inherent in the design of the RF system are the following features:

Present capability of the CCS "omni" antenna system is retained for near-earth orbit use.

Adjacent antenna switching during the transfer trajectory is accomplished by switch selector command resulting in a maximum rf blackout of 50 db. (Carrier phase lock will be maintained although PCM decommutation will become unlocked.)

It is impossible for any two adjacent antennas to radiate at the same time, thereby eliminating interference nulls.

The present high gain CCS antenna capability is retained for use at Synchronous altitude after vehicle stabilization prior to the third S-IVB burn.

The basic Command Communication System (CCS) is not impacted by mission constraints. Because a number of switch selector functions change within the stages and the IU, the functional capability of the digital command will change. This is handled with software, however, and requires no hardware changes.

10.5.2.4.6 Tracking

No hardware modifications will be required in the S-II and S-IVB Tracking system as a result of the J-2S/Synchronous Orbit mission. The IU Tracking system requires no modifications for the J-2S implementation and was not addressed in the Synchronous Orbit Study.

10.5.2.4.7 Power Supply and Distribution

a. S-II Power Supply and Distribution

Converting from the J-2S/LOR to the J-2S/Synchronous Orbit mission requires no additional hardware impact on the S-II Power Supply and Distribution.

10.5.2.4.7 (Continued)

b. S-IVB Power Supply and Distribution

For the Synchronous mission it will be necessary to change the present Forward No. 1 Battery from a 1A59741-507 to two parallel 1A59741-503 batteries rated at 366 amp-hr each (Figure 10.5.2.4-6). This results primarily from the additional power required for long range telemetry transmission. The Forward Bus No. 1 and the 366 amp-hr battery were originally sized for an R&D Data Acquisition System, so it is capable of handling the increased load. The additional 366 amp-hr battery is primarily for the increased stage operational time. This concept has been tested and used on the S-IV with this type of battery. Additional testing with this particular battery will be done for increased confidence in the paralleling technique. The percent of utilization is greater than is normally used (84%) with the battery heaters operating with a 100% duty cycle. This is a smaller margin of safety than is normally used. However, with a constant current drain of approximately 25 amp, the battery heaters will be activated infrequently, if at all. Assuming no heater activation, the percent of utilization is 76% which is acceptable. An E/I switch for each battery provides separation on external power. This prevents the batteries from bucking each other under no load and dissipating energy. On GSE command to internal power, the batteries are simultaneously paralleled and share the load.

Pending a thermal analysis for the Synchronous mission, the 1A57941-507 battery is recommended for Aft No. 1 Battery. If thermal analysis indicates the need for more battery heater power, it may be necessary to recommend a 1A59741-503 for the Aft No. 1 Battery. No structural support changes are required since the 1A59741-503 is the same physical size as the 1A59741-507. Battery modifications require modification to wiring harnesses and drawings.

c. IU Power Supply and Distribution

A four uprated IU battery configuration was chosen as J-2S/Synchronous Orbit power system (Figure 10.5.2.4-7). The four batteries will extend IU Power system lifetime from 6.8 to 15 hours.

Several new cables will be required to connect the fourth battery into the existing power system. Cable changes to the other three batteries will be slight since the same battery case will be used for both the existing IU battery and the uprated battery. The fourth battery will be used to power the cooling system. The loads on the other three batteries will be optimized by wiring changes within the distributors. The end-of-mission discharge depth on each of the batteries will be approximately 67%. A 20-ampere for loading per battery is expected for a Synchronous Orbit mission.

ELECTRICAL POWER DISTRIBUTION

Ref 10.4-32, Fig. 6-21

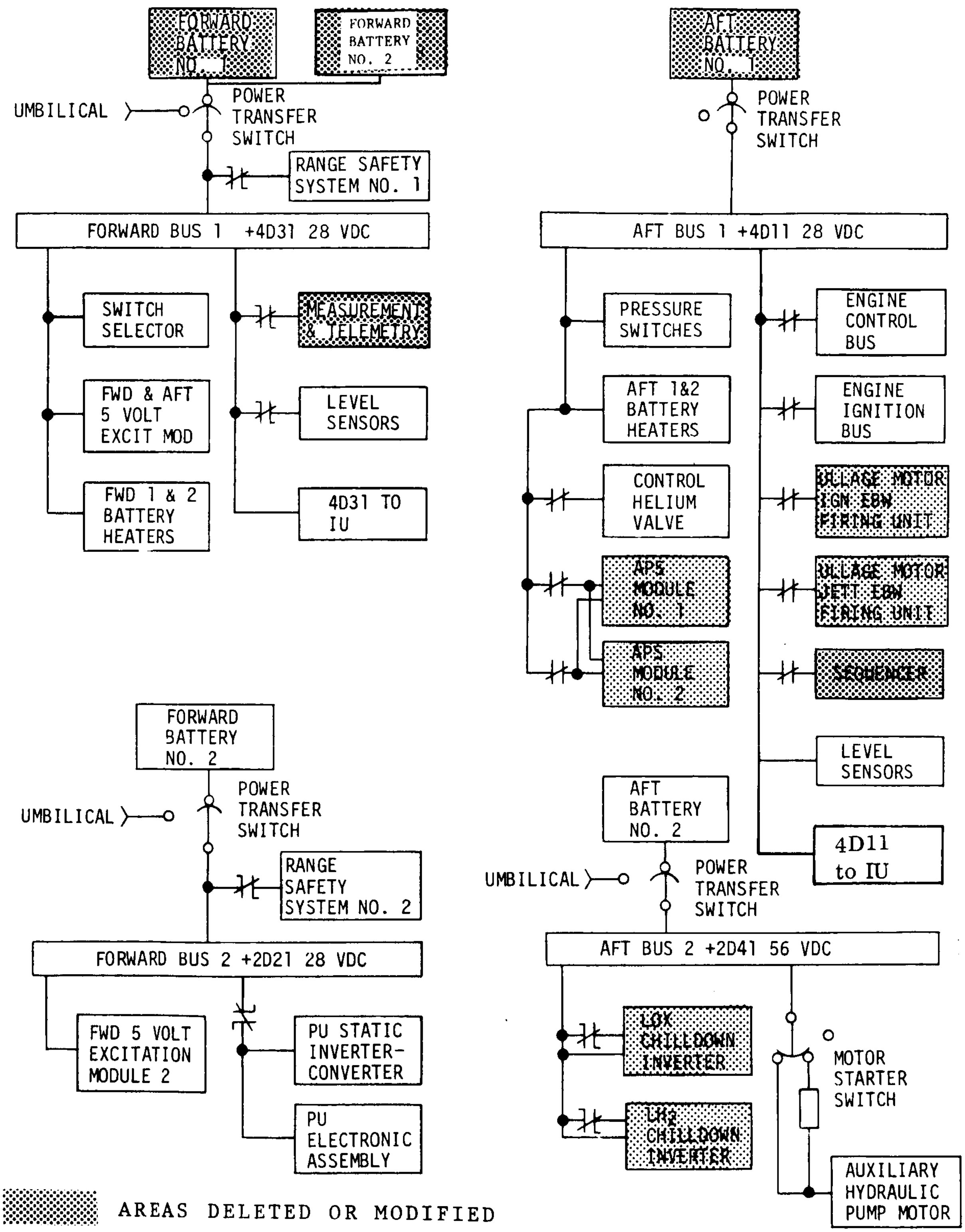
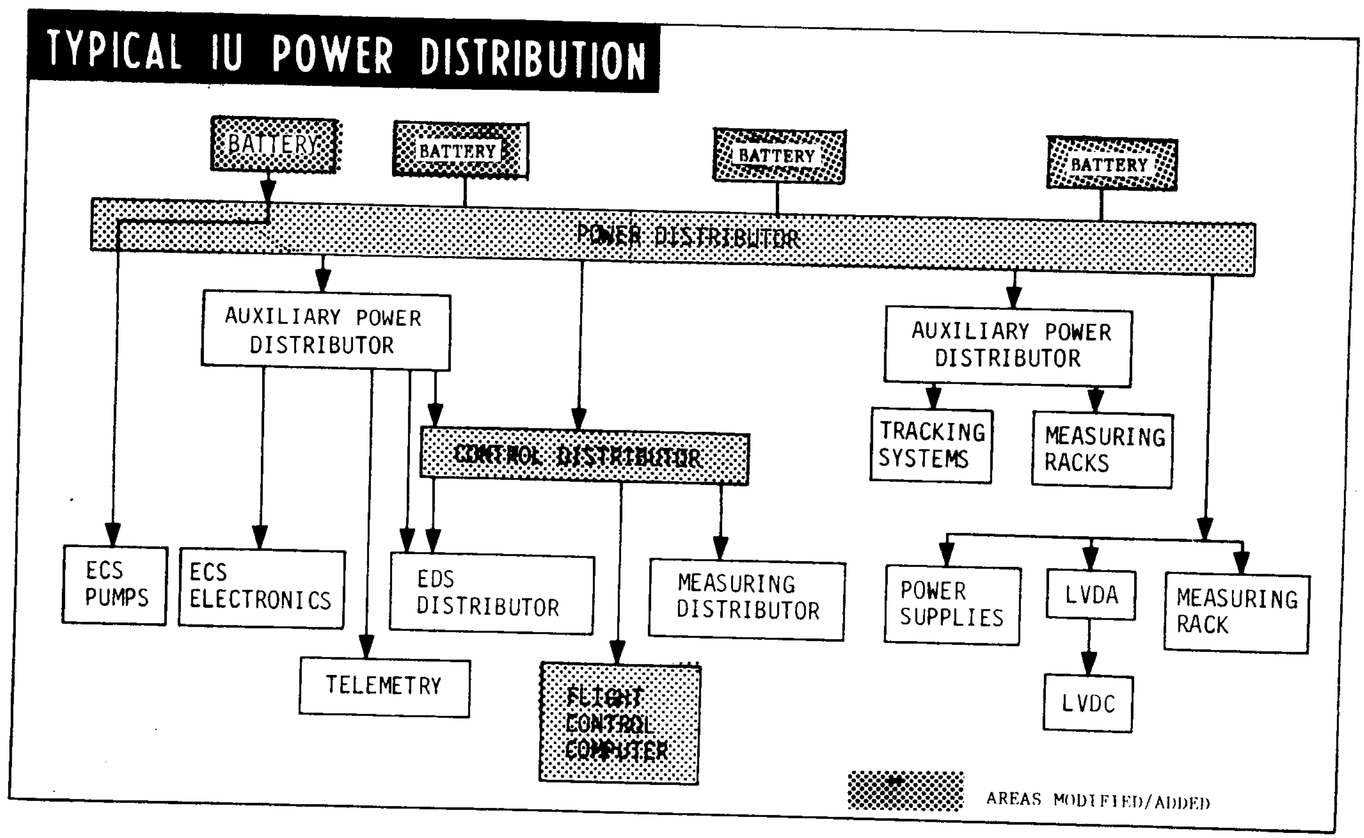


FIGURE 10.5.2.4-6. S-IVB SYNCHRONOUS ORBIT POWER AND DISTRIBUTION



Ref 10.4-32

FIGURE 10.5.2.4-7.. SYNCHRONOUS ORBIT IU POWER DISTRIBUTION

10.5.2.4.7 (Continued)

The ground stations must be modified to support a four-battery configuration. ESE bus voltage monitoring capability via the umbilical and test connectors will be required.

Voltage and current measurements must be added for the fourth battery.

Electrical load profiles were prepared using the nominal power requirements of the IU components. The loads were distributed to meet the following criteria:

Battery discharge depth shall not exceed 80% at end-of-mission.

Minimize the number of flight-critical buses.

Isolate the coolant pumps from the other IU loads.

All subsystem components should be on the same bus.

Equalize battery loading.

The nominal values were derived from test data, except for the measuring racks and the Telemetry Calibrator Power and Control Assembly. An estimated load of 1.75 amperes was used for each measuring rack. An estimated load of 0.45 amperes was used for the Telemetry Calibrator Power and Control Assembly. It should be noted that a component could draw more current than previously measured and still meet its specification.

The four battery power system discharge rate can increase by 25% per battery and still remain within 80% discharge depth at the end-of-mission. In addition, the 80% discharge depth provides a built-in safety factor for any unexpected increases in the power requirements of the IU components.

The load profiles also assume that the load is equally shared when a component receives power from several batteries. This assumption is true only if the batteries have approximately equal voltages. Any appreciable differences in the batteries' voltages could increase the load on the battery with the highest voltage. The problem of load sharing can become critical when the batteries are operated above the 80% discharge depth.

At the end of the required IU lifetime, which for a particular Synchronous Orbit mission may be from 8 to 15 hours after lift-off depending on the selected hover point, there will be some time period before the IU batteries are depleted.

10.5.2.4.7 (Continued)

During the last two hours of the required IU lifetime, the IU will participate in the initiation of the S-IVB stage passivation by issuing approximately 40 switch selector commands. However, the passivation procedure, which is relatively slow at Synchronous altitude, will not be completed at the end of the required IU lifetime (The final steps will be controlled manually.).

To prevent inadvertent IU or S-IVB switch selector commands and to prevent possible hazardous conditions during the period of battery depletion after the end of the required IU lifetime, power will be removed from the IU.

The system proposed to shut down the power system will prevent any single failure from causing an undersized power transfer. Two separate switch selector commands will initiate the power down sequence to the LVDA/LVDC and transfer the motor driven switch to the external position. The LVDA/LVDC power down sequence must be followed to prevent spurious outputs when power is removed.

10.5.2.4.8 Emergency Detection System

Converting from J-2S/LOR to J-2S/Synchronous mission will require no additional hardware impact to the Saturn Vehicle Emergency Detection System.

10.5.2.4.9 Separation

a. S-II Separation

Converting from J-2S/LOR to J-2S/Synchronous mission will require no impact additional to that discussed in paragraph 10.5.1.4.9 a , J-2S/LOR report.

b. S-IVB Separation

The S-IVB Separation system will require no impact additional to that discussed in paragraph 10.5.1.4.9 b , J-2S/LOR report.

c. IU Separation

The IU Separation will require no impact as a result of J-2S/LOR to J-2S/ Synchronous conversion.

10.5.2.4.10 Flight Program

a. Guidance Program Analysis

Synchronous Orbit studies were conducted to determine impact of mission constraints on the Flight Program. Areas of consideration were:

10.5.2.4.10 (Continued)

Yaw Biasing of Three Gimbal Platform.

Navigation Accuracy.

Scaling.

IGM.

Additional S-IVB Restart, Coast.

Retargeting, Restart Preparation.

Antenna Switching.

Orbital Guidance.

Passivation.

Memory Requirements.

Discussion of Synchronous Orbit problem area and conclusions may be found in the IBM Synchronous Orbit ECP. Once the Synchronous Orbit program is defined, the incorporation of J-2S engines will not require modification. An advantage, however, of using the J-2S engine characteristics is to eliminate a previous constraint of 0.5 hours minimum time for chilldown between orbit insertion and restart. This allows a restart at the first descending node and access to certain hover points with increased payload (Table 10.5.2.4-II).

b. Control Analysis

The J-2S synchronous mission is equivalent to the J-2S/LOR mission from a Flight Control System standpoint except the synchronous mission includes two extended idle modes. The first precedes second S-IVB mainstage burn and the second precedes third S-IVB mainstage burn.

Since the vehicle parameters for the first extended idle mode are almost identical to those for the LOR idle mode, no additional analysis was performed. Vehicle responses were obtained for the second idle mode employing the same analog simulation described in paragraph 10.4.3.6.

These responses indicate a higher control mode natural frequency and damping than those for the LOR (or first Synchronous) idle mode. The moment of inertia is less and the control moment arm is greater, which results in a higher control moment coefficient. Results were that the same control gains can be used for

TABLE 10.5.2.4-II. APPROXIMATE MISSION TIME LINE

SYNCHRONOUS

Time Base	Time (Seconds)	Event
TB0	-17	Guidance Reference Release
TB1	0	Liftoff
	84	Q Max
TB2	149	Center Engine Cutoff
TB3	161	S-IC Cutoff
	165	S-II Ignition (idle)
	166	S-II Mainstage
	168	S-II MR Shift (5 - 5.5)
	193	Interstage Separation
	198	LES Separation
	396	S-II MR Shift (5.5 - 4.7)
TB4	489	S-II Cutoff
	489	S-IVB Ignition (idle)
	490	S-IVB Mainstage
	493	S-IVB MR Shift (5 - 5.5)
TB5	545	S-IVB Cutoff (1st burn)
	5 Rev. ~7.5 hr	Parking Orbit Coast
TB6	5045	S-IVB Reignition (idle)
	5145	S-IVB Mainstage
	5147	S-IVB MR Shift (5 - 5.5)
TB7	5353	S-IVB Cutoff (2nd burn)
	5.3 hr	Transfer Orbit
TB8	8533	S-IVB Reignition (idle)
	8633	S-IVB Mainstage
	8635	S-IVB MR Shift (5 - 5.5)
TB9	8725	S-IVB Cutoff (3rd burn)

10.5.2.4.10 (Continued)

both idle modes. Thus, the FCS for the J-2S/Synchronous mission will be identical to the FCS for the J-2S/LOR mission. The increased control gains for S-IVB idle mode will be switched in at first and second S-IVB mainstage cut-off and switched back to the lower gains at second and third S-IVB mainstage start commands.

10.5.2.4.11 Instrument Unit

Components modified on the Instrument Unit as a result of J-2S/LOR to J-2S/ Synchronous mission conversion are shown in Figure 10.5.2.4-8.

10.5.2.4.12 Environmental Control

a. S-II Environmental Control

Converting from the J-2S/LOR mission to the J-2S/Synchronous Orbit mission will require no additional hardware impact on the S-II Environmental System.

b. S-IVB Environmental Control

Synchronous Orbit mission constraints impose additional thermal control requirements on the S-IVB Environmental Control System (Reference 10.4-44).

Three cold panels will be added to the forward skirt of the S-IVB for the Synchronous Orbit mission. The S-IVB sequencer, 56 volt power supply distribution panel, 28 V power distribution panel, attitude control relay (2), signal conditioning panels (6) and liquid level control units must be provided with low emissivity covers to compensate for low temperature constraints. The voltage excitation module assembly must have a large heat sink mounted to overcome a high temperature problem. Low ε and/or α / ε paint will be used on the pneumatic power control module, ambient helium fill module, ambient helium bottles (1/2 cu ft). Forward APS tank support and APS quick disconnect mounting bracket shall be modified with fiberglass to compensate for low temperature constraints.

The S-IVB Switch Selector, remote analog submultiplexer, remote digital submultiplexer and multiplexer will require a 15 watt heater blanket on each unit. The 50 amp motor driven switches on aft panels 17 and 19 will require 5 watt heater blankets on each unit to compensate for low temperature constraints. Recalibration of the cold helium bottle transducer or installation of a heater blanket on the unit will be required to overcome low temperature constraints.

Ref 10.4-31, Fig. 12.1-3

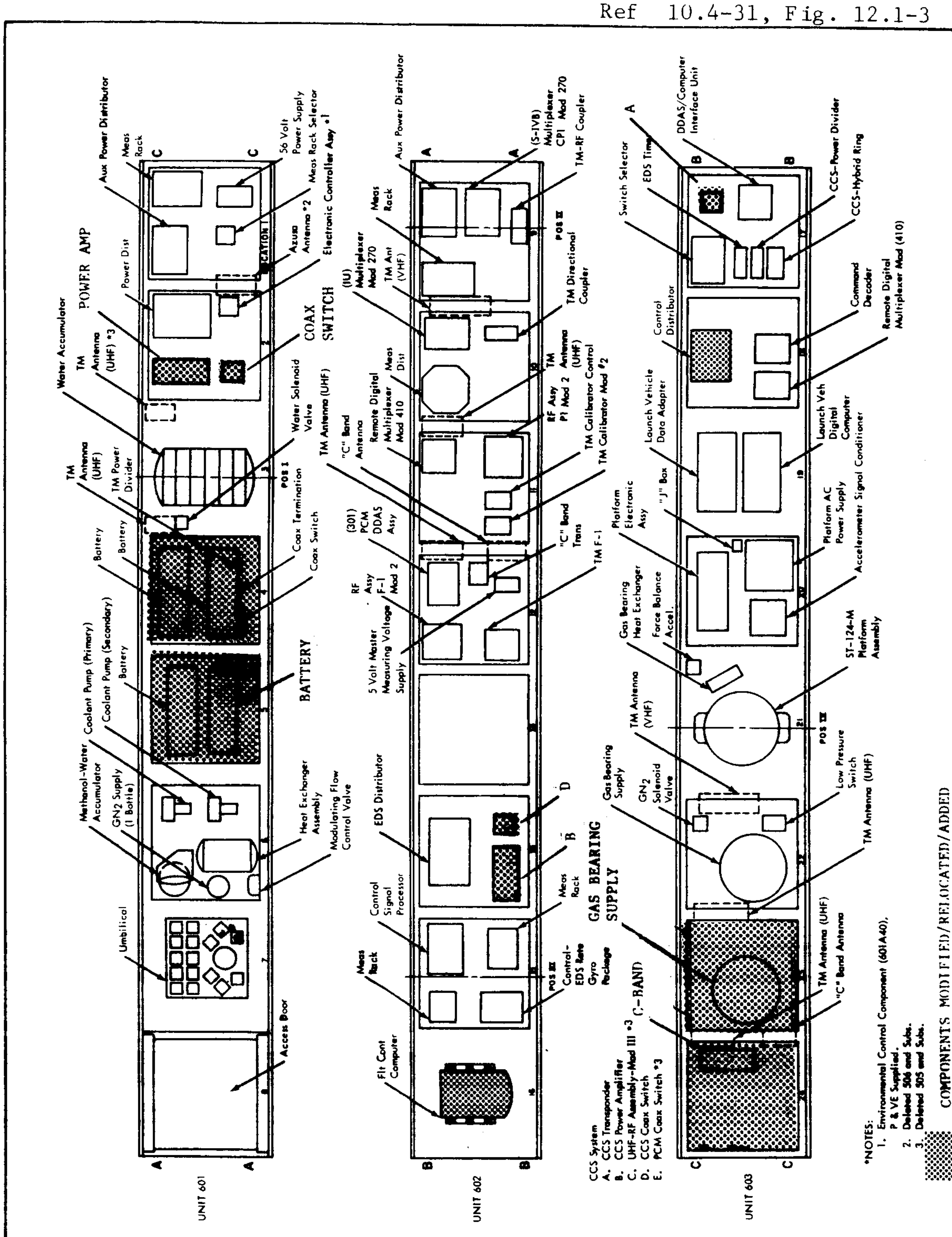


FIGURE 10.5.2.4-8. SYNCHRONOUS ORBIT INSTRUMENT UNIT

10.5.2.4.12 (Continued)

c. IU Environmental Control

The ECS hardware modifications required to successfully accomplish the Synchronous Orbit mission are summarized below: (Reference Figure 10.5.2.4-9)

1. An additional GN_2 storage sphere (2 cu ft) with appurtenances (mounting panel, plumbing, etc.) will be placed in location 23.

2. The current TCS orifice regulator assembly will be replaced by a redesigned assembly.

Mission thermal analyses have been performed to assess the temperature control parameters and assure that positive thermal control will be maintained at all times. As presently defined, the mission consists of the following flight phases:

1. Low Earth Orbit - Up to 7.5 hours of velocity-oriented low earth orbit (100 n.m.) at inclinations from 28.5^o to 64^o.

2. Hohmann transfer - 5.5 hours of transfer trajectory with vehicle broadside to the sun and rolling at a rate of one revolution per hour.

3. Synchronous Orbit - Up to 2 hours of Synchronous Orbit at inclinations up to 64^o.

In only one of these phases of flight (Hohmann transfer) is the thermal environment known and fixed. The Low Earth Orbit can range from "cold" (maximum time in earth shadow, 28.5^o inclination) to "hot" (zero time in shadow, 64^o inclination) with all-inclusive cases possible as a function of inclination and date and time of launch. The Synchronous Orbit can also range from cold (vehicle parallel to solar vector) to hot (vehicle broadside to solar vector) for the 2 hours of orbit, with all intermediate conditions again possible. Without definite knowledge concerning a particular mission at a particular date it became necessary to examine the extreme cases. Worst case thermal environments were investigated for both hot and cold environments.

The total energy expended (maximum) over 15 hours duration (maximum for the Synchronous Orbit mission) is 36 killowatt-hours. The capacity of the present cooling system is approximately 45 killowatt-hours, i.e., there is sufficient water stored in the water accumulator (146 lbs) to provide for 45 kwh of cooling -- an ample amount for the Synchronous Orbit mission. Also, owing to the relatively short duration of the Synchronous Orbit flight, temperature requirements will be met at all times.

Ref 10.4-32, Fig. 7-5

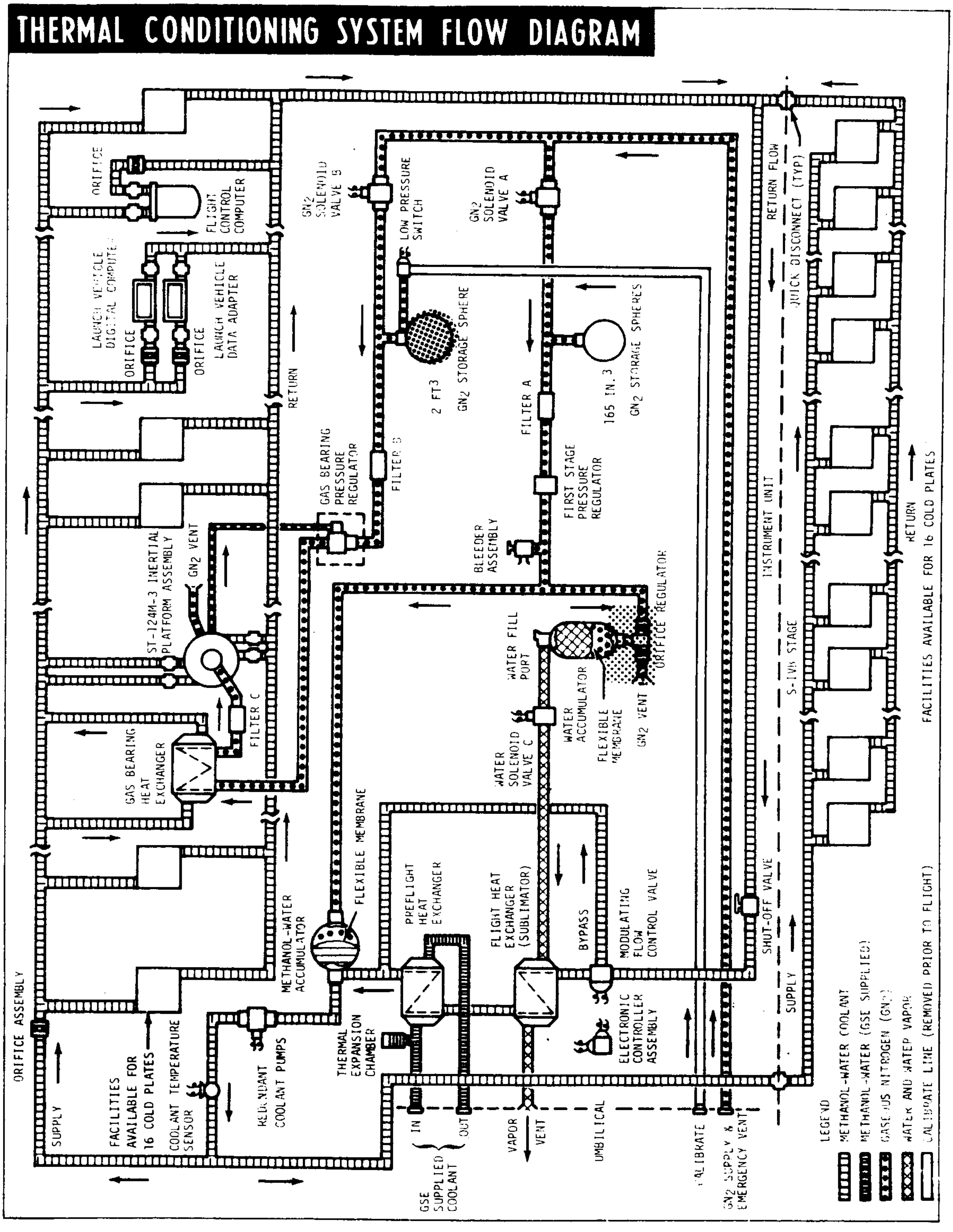


FIGURE 10.5.2.4-9. THERMAL CONDITIONING SYSTEM FLOW DIAGRAM

10.5.2.4.12 (Continued)

The limiting factor in ECS operating life is the consumption of expendable fluids -- water for Thermal Conditioning System (TCS) cooling, GN_2 for TCS pressurization, and GN_2 for the Gas Bearing Supply (GBS) to the Inertial Platform. It was established that there is sufficient water stored in the water accumulator to complete the Synchronous mission in the worst (hot) case.

With regard to the GN_2 for the TCS pressurization, the maximum use rate of .097 lbs/hr defines 2.18 lbs of GN_2 required to satisfy a 150% safety factor for a 15 hour mission. The present storage sphere (165 cu in.) has 1.2 lbs of usable GN_2 at lift-off. Thus, two possibilities exist: (1) increase the storage capacity (enlarge present sphere or add an additional sphere) or (2) decrease the GN_2 use rate to a level that can be supplied by the current storage configuration. Assessment has confirmed that the latter may be accomplished with a nominal hardware change by resizing the orifices within the orifice regulator assembly.

Additional GN_2 for the ST-124M Inertial Platform will be required for the Synchronous Orbit mission due to the extended life requirement. The present storage sphere (2 cu ft) has about 26 lbs of usable GN_2 at lift-off. The 2.2 lbs/hr use rate defines 49.5 lbs of GN_2 required for 150% safety factor for 15 hours. Assessments have established that the least impact approach is the addition of a second 2 cu ft sphere to operate in parallel with the current sphere.

10.5.2.4.13 Electrical Support Equipment

a. S-II Electrical Support Equipment

Converting from the J-2S/LOR to the J-2S/Synchronous mission will require no hardware impact additional to that discussed in paragraph 10.5.1.4.13 a J-2S/LOR report.

b. S-IVB Electrical Support Equipment

The S-IVB J-2S/Synchronous Orbit study was performed by comparing the Saturn V J-2/Synchronous Orbit vehicle configuration to the J-2S/Synchronous Orbit configuration.

Modifications to the Saturn V J-2S/Synchronous Orbit vehicle are identical to those described in paragraph 10. 5. 1. 4. 13 b.

10.5.2.4.13 (Continued)

c. IU Electrical Support Equipment

The purpose of the IU ESE paragraph is the definition of ESE modifications required by the J-2S engines used on a Saturn V vehicle for a Synchronous Orbit mission. The ESE modifications were determined by considering the changes necessary to uprate a baseline AS 505 IU for use on the Synchronous mission. Implementation of the J-2S engines on a Saturn V vehicle and the mission requirements of the Synchronous mission will necessitate changes in the ESE hardware and checkout programs used to verify IU systems operation.

1. ESE Hardware Modifications

   (a) Instrumentation and Communication Subsystem (I&C)

   The I&C subsystem does require modification for this mission as described by paragraph 10.5.2.4.5; therefore, the following ESE modifications are required.

   In order to verify proper operation of the antenna switching network of the onboard CCS system, a switching matrix must be installed in the RF ground station.

   The capability to monitor one additional helix current and six antenna positions must be incorporated into the RF ground station. The additional measurements are similar to the ones now being monitored.

   Five additional cables must be added to the RF ground station to monitor the CCS antenna outputs.

   (b) Electrical Subsystem

   The IU lifetime extension requirements of a Synchronous mission require adding a fourth battery to the IU. The ground power source has the capability to supply four vehicle busses, but the capability to monitor the bus voltage of the fourth bus must be added to the ground power supplies. This will require removal of one jumper in the ground power source. The change to six directional antennas and the replacement of the existing power amplifier with two uprated power amplifiers with switching capabilities will require adding a total of thirteen ESE simulate command lines. Nine lines will be required to switch the antennas and four lines will be required to switch the amplifiers.

10.5.2.4.13 (Continued)

2. Checkout Program

The automated checkout program modifications resulting from implementing a J-2S engine to perform a Synchronous mission with a Saturn V vehicle are divided into two categories.

Subsystems Automated Checkout Programs.

IU Overall Checkout Program.

The automated checkout programs needing modification are listed below with the subsystem requiring the program. In some cases, only parameter changes are required but there are some changes that will require minor program rewrite.

(a) Control Subsystem

A_1 Gain.

A_0 Gain.

Control Systems Nulls.

Control Relay Redundancy.

Engine Deflection.

(b) Electrical Subsystem

Power Distribution and Control.

General Networks.

Simulated Plug Drop.

The IU overall program can be modified for the Synchronous mission with no change to the basic program. The IU overall program is a general program applicable to all Saturn V vehicles with little or no modification to the basic program from one vehicle to another. All modifications due to the IU hardware changes and the Synchronous mission requirements can be loaded into the basic program via user-controlled data tables.

KSC will require changes similar to the ones mentioned above.

10.5.2.4.14 Propellant Management

a. S-II Propellant Management

Converting from J-2S/LOR to J-2S/Synchronous mission will require no additional modifications to the S-II Propellant Utilization (PU) system.

b. S-IVB Propellant Management

The S-IVB contractor recommended modes of PU system operation for the Synchronous mission are presented below:

BURN PERIOD	MODE OF OPERATION	COMMANDED EMR
First	Open Loop	5.5/1.0
Second	Closed Loop	5.35/1.0
Third	Open Loop	5.0/1.0

The open loop mode of operation during first burn eliminates the need of an LH_2 bias for first coast. The closed loop EMR for second burn reflects the results of past Synchronous mission studies. Open loop operation during third burn eliminates the need of an LH_2 bias for second coast and eliminates thrust variations due to PU valve motion toward the end of flight. The elimination of LH_2 boiloff biases also eliminates the sequencing associated with the biases.

c. IU Propellant Management

The IU will require no hardware modifications as a result of J-2S/Synchronous PU system. Open and closed loop PU software impact is discussed in paragraph 10. 5. 1. 4. 14 c, J-2S/LOR mission.

10.5.3 J-2S/LEO Astrionic System Interface

10.5.3.1 Purpose

This section of the J-2S Improvement Study (Astrionic System Integration LEO Mission) defines the differences between the two-stage Saturn V vehicle J-2S/AS 511 (Reference Figure 10.5.3.1-1) configured for the Low Earth Orbit mission (Reference Figure 10.5.3.1-2) and the three-stage Saturn V vehicle J-2S/AS 511 configured for the Lunar Orbit Rendezvous (defined in section 10.6.1).

If an existing system is changed, only the changes or additions are described in this section, and the Astrionic Handbook, Saturn V Flight Manual, and/or the J-2S/LOR section 10.6.1 is referenced for a more detailed description. Modification peculiar to the J-2/J-2S interface will be so noted. Each illustration in this section is referenced to a corresponding illustration in the document from which it was taken.

10.5.3.2 J-2S Astrionic System (LEO Mission)

The overall Astrionic System of the J-2S/LEO AS 511 vehicle is shown in the simplified block diagram, Figure 10.5.3.2-1. Vehicle characteristics of the LEO mission are:

a. The S-IVB stage is configured as an undefined space station.

b. The S-II stage must place Saturn vehicle in waiting orbit; therefore, the S-II must possess the capability of restarting J-2S engines and maneuvering the vehicle with a reaction control system. (Reaction control may be provided by the space station RCS in lieu of the RCS located in the S-II, see paragraph 10.4.3.6.3c. for detailed discussion.)

The Astrionic System includes that integrated group of components and/or subsystems which provide the following vehicle functions during flight:

Navigation, guidance, and control of the vehicle.

Measurement of vehicle parameters.

On-board data management.

Data transmission between vehicle and ground stations (up and down).

Tracking of the launch vehicle.

Checkout and monitoring of vehicle functions.

Detection of emergency situations.

Generation of electrical power for system operation.

Power and signal distribution.

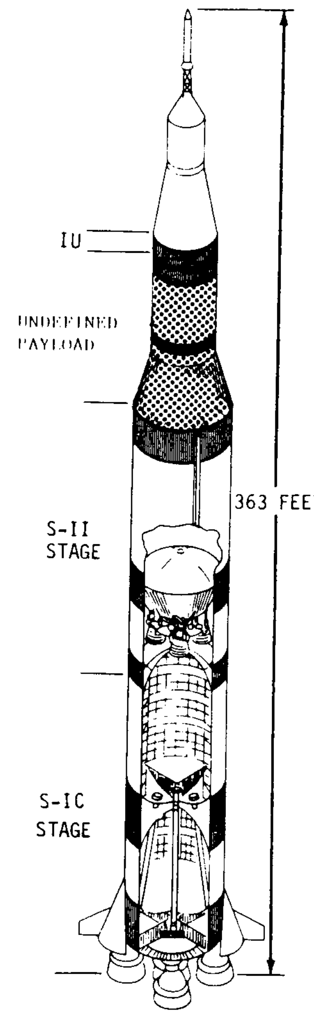
SATURN V LAUNCH VEHICLE

SOLID ULLAGE ROCKET AND RETROROCKET SUMMARY				
STAGE	TYPE	QUANTITY	NOMINAL THRUST AND DURATION	PROPELLANT GRAIN WEIGHT
S-IC	RETROROCKET	8	75,800 POUNDS• 0.541 SECONDS	278.0 POUNDS
S-II	ULLAGE	4	23,000 POUNDS‡ 3.75 SECONDS	336.0 POUNDS
	RETROROCKET	4	34,810 POUNDS‡ 1.52 SECONDS	268.2 POUNDS

ENGINE DATA					
STAGE	QTY	ENGINE MODEL	NOMINAL THRUST EACH	NOMINAL THRUST TOTAL	
S-IC	5	F-1	1,526,500	7,632,500	
S-II	5	J-2	265,000	1,325,000	360 SEC

STAGE DIMENSIONS		
	DIAMETER	LENGTH
S-IC Base (including fins)	63.0 FEET	138 FEET
S-IC Mid-stage	33.0 FEET	
S-II Stage	33.0 FEET	81.5 FEET
Instrument Unit	21.7 FEET	3.0 FEET

SATURN V STAGE MANUFACTURERS	
STAGE	MANUFACTURER
S-IC	THE BOEING COMPANY
S-II	NORTH AMERICAN-ROCKWELL
S-IU	INTERNATIONAL BUSINESS MACHINE CORP.



PRE-LAUNCH LAUNCH VEHICLE GROSS WEIGHT ≈ 6,368,000 POUNDS

• MINIMUM VACUUM THRUST AT 120°F

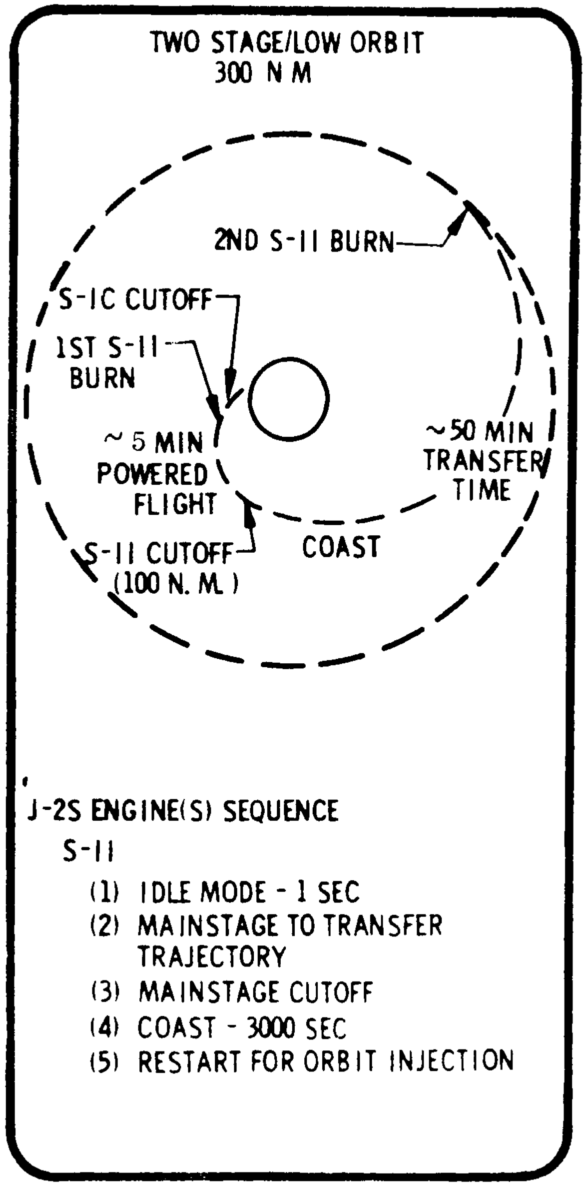
(stippled) AREA CHANGED

‡ NOMINAL VACUUM THRUST AT 60°F

NOTE: THRUST VALUES, WEIGHTS, AND BURN TIMES ARE ALL APPROXIMATIONS.

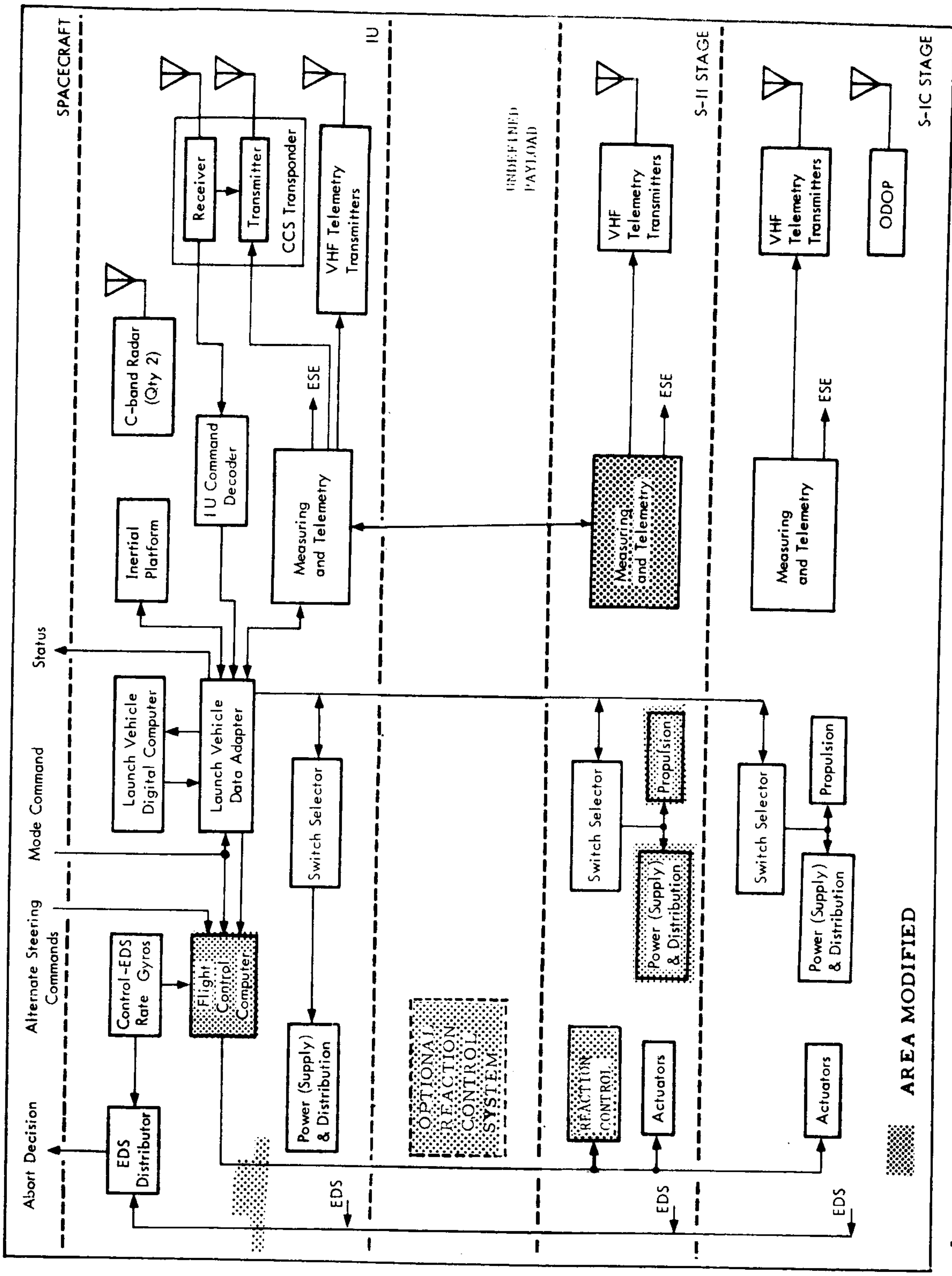
Ref 10.4-32, Fig.1-3

FIGURE 10.5.3.1-1. J-2S/LEO LAUNCH VEHICLE



Ref 10.4-1

FIGURE 10.5.3.1-2. LEO MISSION PROFILE



Ref 10.4-31. Fig. 1.4-2

FIGURE 10.5.3.2-1. ASTRIONIC SYSTEM

10.5.3.2 (Continued)

Thermal conditioning of components.

Separation of stages.

Propellant management.

Most of the astrionic system components are located in the IU which is mounted on top of the S-IVB Stage. Additional components such as telemetry, distributors, thrust vector control subsystem, electrical networks, are located in the vehicle stages.

The overall Astrionic System of the J-2S AS 511 vehicle is shown in the simplified block diagram, Figure 10.5.1.2-1.

1. Navigation, Guidance and Control

Function and Description

The G&C system provides the following basic functions during flight (Reference Figure 10.5.3.2-2.

Stable positioning of the vehicle to the commanded position with a minimum amount of sloshing and bending.

A first stage tilt attitude program which gives a near zero lift trajectory through the atmosphere.

Provides steering commands during S-II burns which guide the vehicle to a predetermined set of end conditions while maintaining a minimum propellant trajectory for earth orbit insertion.

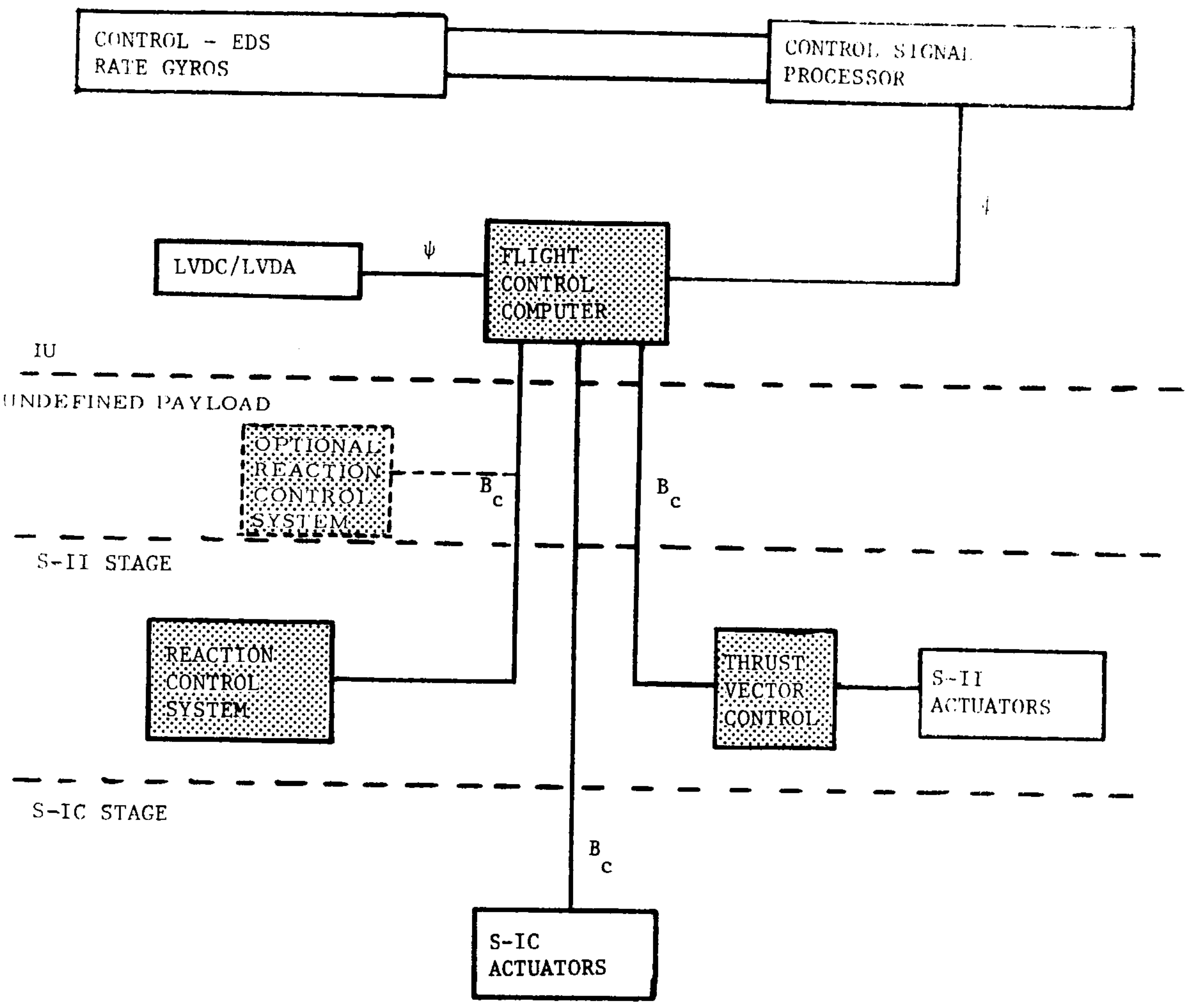
Maintains the proper vehicle position during earth orbit.

Provides guidance during the second S-II burn, placing the vehicle in the proper waiting orbit.

G&C Hardware

The Stabilized Platform (ST-124M) and the Launch Vehicle Data Adapter/Launch Vehicle Digital Computer are described in J-2S/LOR study. No additional hardware modifications are required as a result of the LEO mission.

The FCC is an analog device which receives attitude error signals from the LVDA/ LVDC and vehicle angular rate signals from the CSP.



Ref 10.4-32, Fig. 7-14

FIGURE 10.5.3.2-2. LEO NAVIGATION GUIDANCE AND CONTROL SYSTEM BLOCK DIAGRAM

10.5.3.2 (Continued)

These signals are filtered and scaled, then sent as commands to the S-IC and S-II engine actuators and to the Reaction Control System (RCS) Control Relay Packages. The Control Relay Packages accept FCC commands and relay these commands to operate propellant valves in the S-II or the space station.

2. Measurements and Data Transmission

The Measurement and Telemetry System is described in the J-2S/LOR Study. Modifications to the S-IVB stage as a result of being transformed into a space station were not addressed in this study. Other stages of the Saturn V vehicle will require no hardware changes to the basic measurement and telemetry system.

3. Tracking

The Tracking System is described in the J-2S/LOR Study. No additional modification will result from the J-2S/LEO mission.

4. Emergency Detection System

The basic Emergency Detection System will remain as described in the J-2S/LOR Study.

10.5.3.3 J-2S/LEO Electrical Interface

The J-2S/LEO Study Launch Vehicle is identified to the baseline Saturn V J-2S/LOR AS 511 configuration. The electrical interface is shown in Figure 10.5.3.3-1.

10.5.3.4 J-2S/LEO Astrionic Subsystems

10.5.3.4.1 Navigation and Guidance

The hardware for the J-2S/LEO Guidance System will require no modification. Guidance software modifications are discussed in paragraph 10.5.3.4.10, Flight Program. Flight Sequence changes are given in paragraph 10.5.3.4.3, Mode and Sequencing.

10.5.3.4.2 Attitude Control

The Saturn V vehicle LEO control system analysis is discussed further utilizing control simulation techniques in paragraph 10.5.3.4.10, Flight Program. Saturn V engine actuator and nozzle arrangements are shown in Figure 10.5.3.4-1. Figure 10.5.3.4-2 is a block diagram of LEO control system. Definition of the LEO Mission requires a passive S-IVB Stage; therefore, no discussion will be directed toward the S-IVB Stage.

Ref 10.4-34, Fig. 1-4

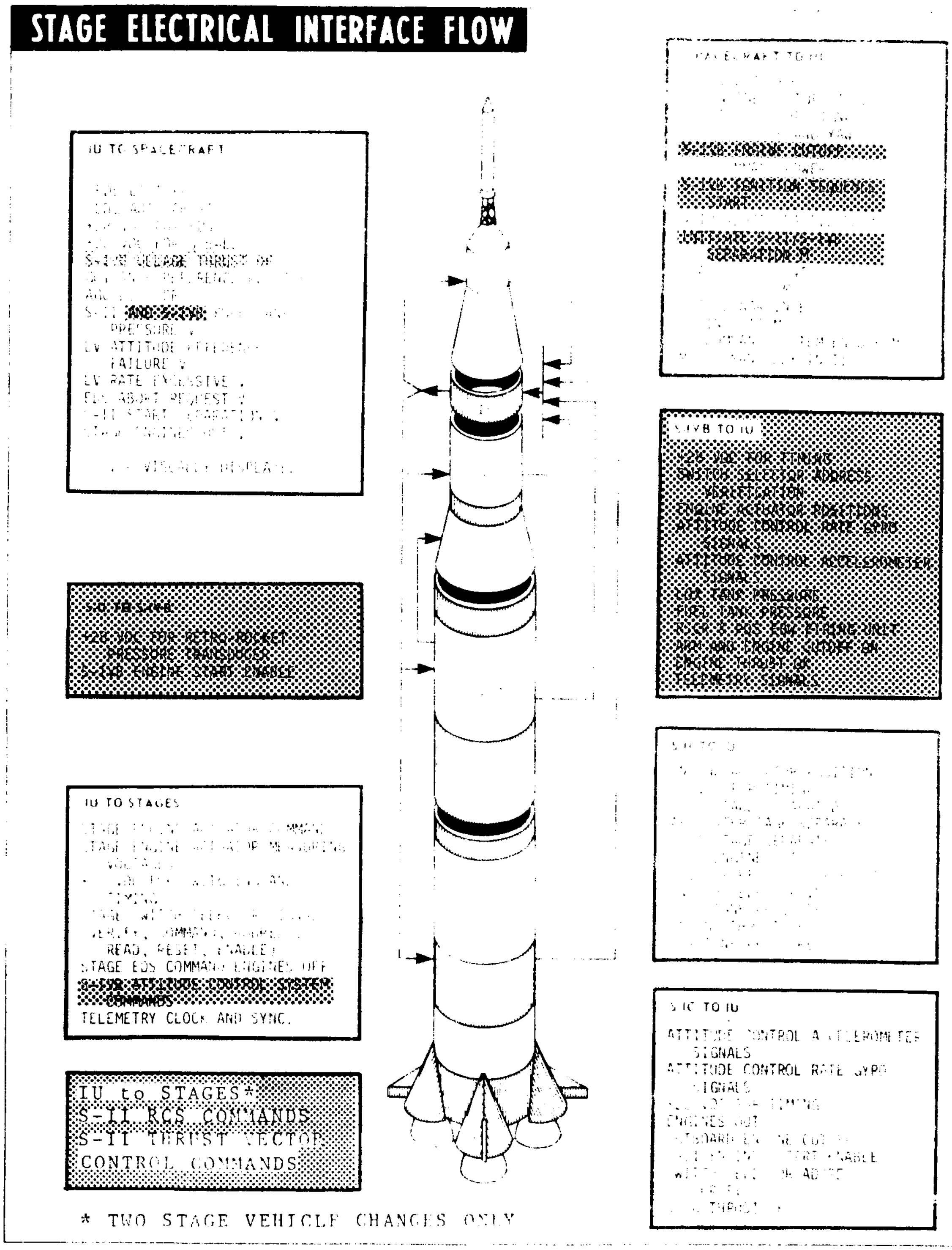


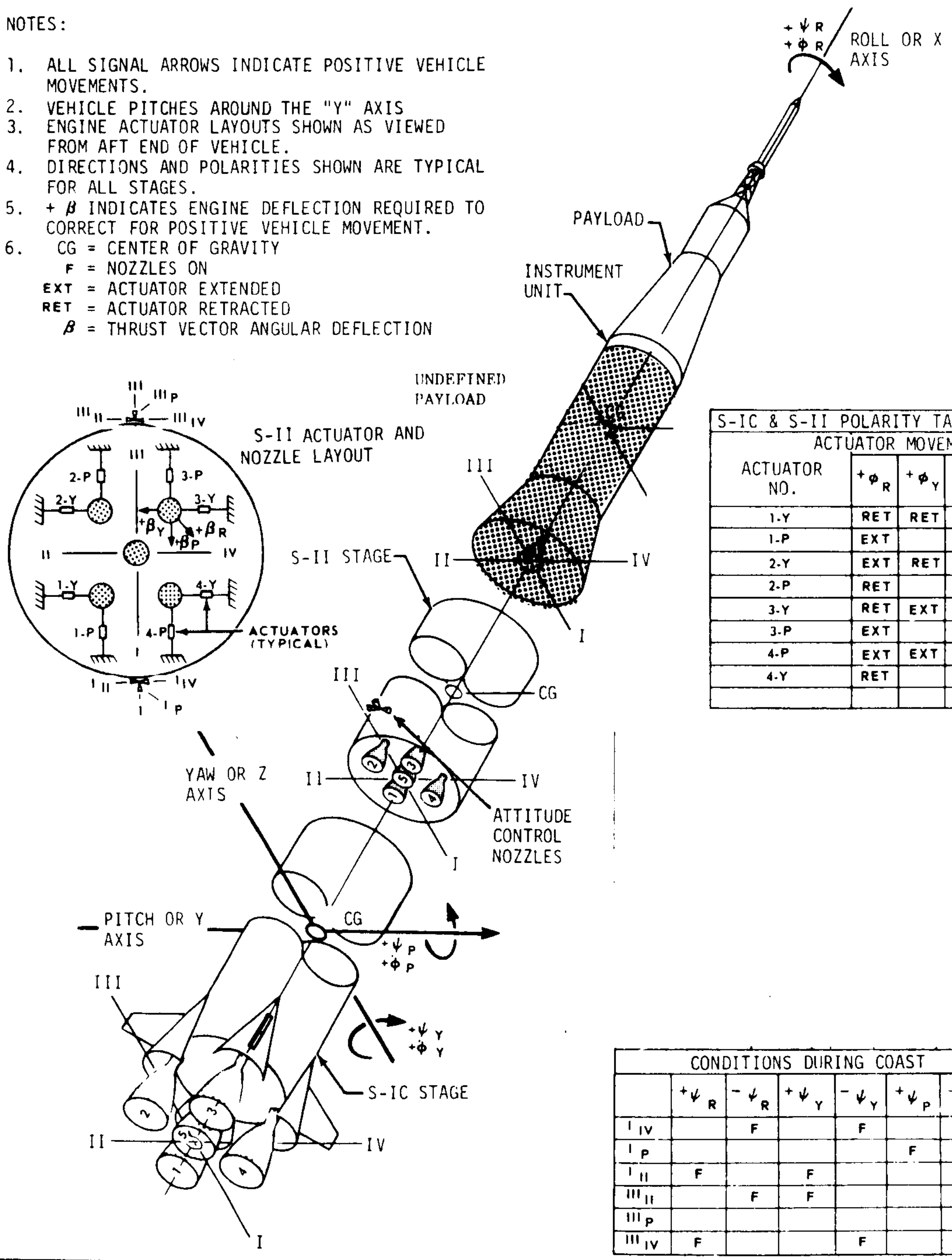
FIGURE 10.5.3.3-1. LEO MISSION VEHICLE ELECTRICAL INTERFACES

Ref 10.4-32, Fig. 7-15

SATURN V ENGINES, ACTUATORS AND NOZZLE ARRANGEMENT

NOTES:

1. ALL SIGNAL ARROWS INDICATE POSITIVE VEHICLE MOVEMENTS.
2. VEHICLE PITCHES AROUND THE "Y" AXIS
3. ENGINE ACTUATOR LAYOUTS SHOWN AS VIEWED FROM AFT END OF VEHICLE.
4. DIRECTIONS AND POLARITIES SHOWN ARE TYPICAL FOR ALL STAGES.
5. + β INDICATES ENGINE DEFLECTION REQUIRED TO CORRECT FOR POSITIVE VEHICLE MOVEMENT.
6. CG = CENTER OF GRAVITY
   F = NOZZLES ON
   EXT = ACTUATOR EXTENDED
   RET = ACTUATOR RETRACTED
   β = THRUST VECTOR ANGULAR DEFLECTION



S-IC & S-II POLARITY TABLE			
	ACTUATOR MOVEMENT		
ACTUATOR NO.	$+\phi_R$	$+\phi_Y$	$+\phi_P$
1-Y	RET	RET	
1-P	EXT		RET
2-Y	EXT	RET	
2-P	RET		EXT
3-Y	RET	EXT	
3-P	EXT		EXT
4-P	EXT	EXT	
4-Y	RET		RET

CONDITIONS DURING COAST						
	$+\psi_R$	$-\psi_R$	$+\psi_Y$	$-\psi_Y$	$+\psi_P$	$-\psi_P$
I_{IV}		F		F		
I_P					F	
I_{II}	F		F			
III_{II}		F	F			
III_P						F
III_{IV}	F			F		

FIGURE 10.5.3.4-1. SATURN V ENGINES, ACTUATORS AND NOZZLE ARRANGEMENT

10.5.3.4.2 (Continued)

a. S-II Attitude Control System

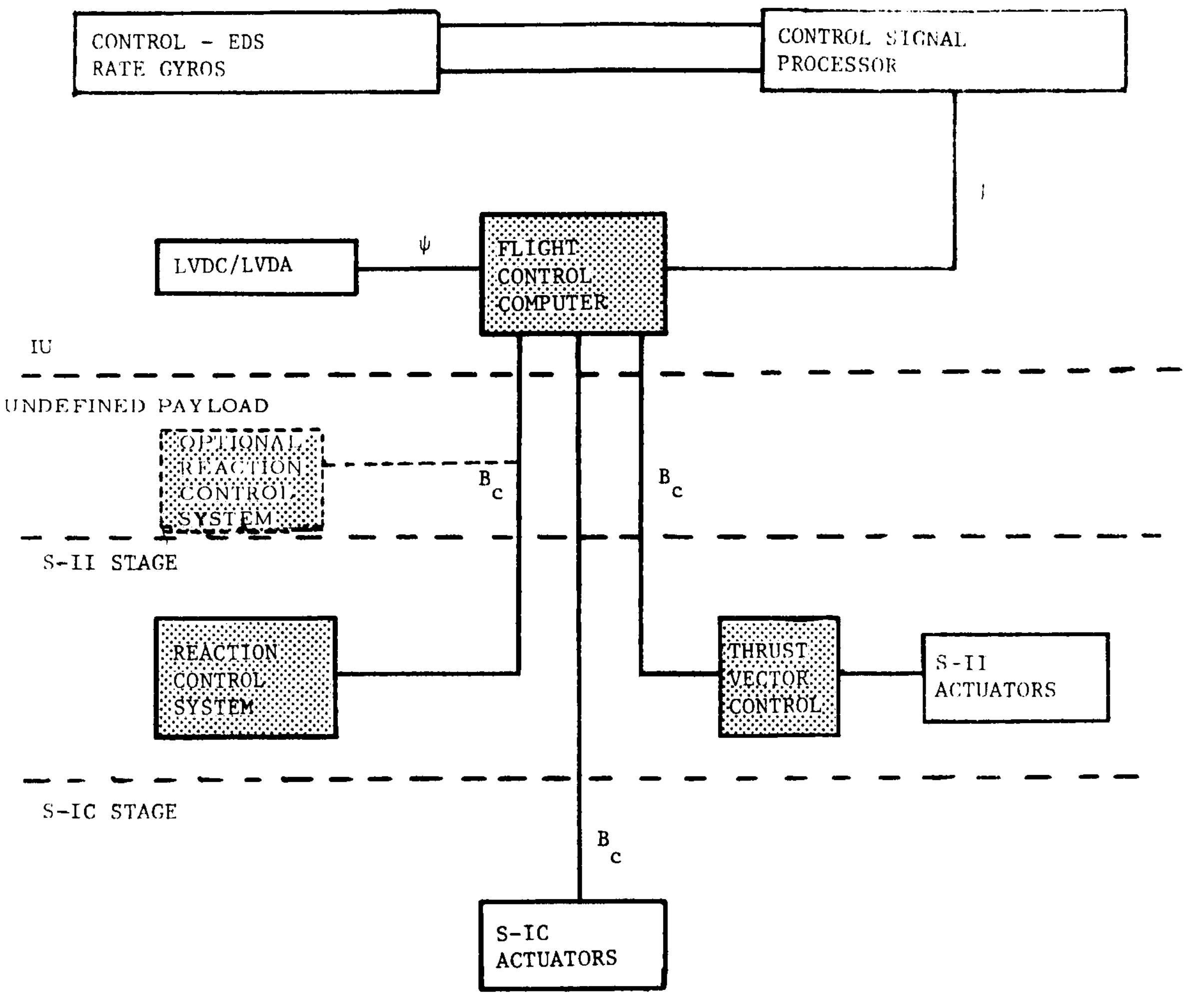
Changing the Saturn V, S-II stage from a J-2S/LOR to a J-2S/LEO will require major modification to the S-II Attitude Control System.

A Reaction Control System (RCS) will be required during the coast period for maintaining vehicle attitude control. There will be two MDAC RCS units utilized. Mounting will be on opposite sides of the S-II aft skirt area (Reference Figure 10.5.3.4-1). The RCS units will be the same as the Auxiliary Propulsion System required on the S-IVB/LOR stage (Reference Figure 10.5.3.4-3). The RCS is limited to a small body rate and ±2.5-degree attitude error. An alternative is the use of the RCS system of the space station as discussed in paragraph 10.4.3.6.3c.

The IU computer will provide the +28 VDC and return control signal necessary to sequence the RCS engines. One set of relay contacts will be used to control an oxidizer valve and the other a fuel valve. Telemetry measurements will be provided from one normally closed set of contacts on each relay. There are three engines for each RCS unit resulting in the addition of 24 nonlatching type control relays for both RCS units. The ground servicing and checkout will be provided at KSC for the RCS units.

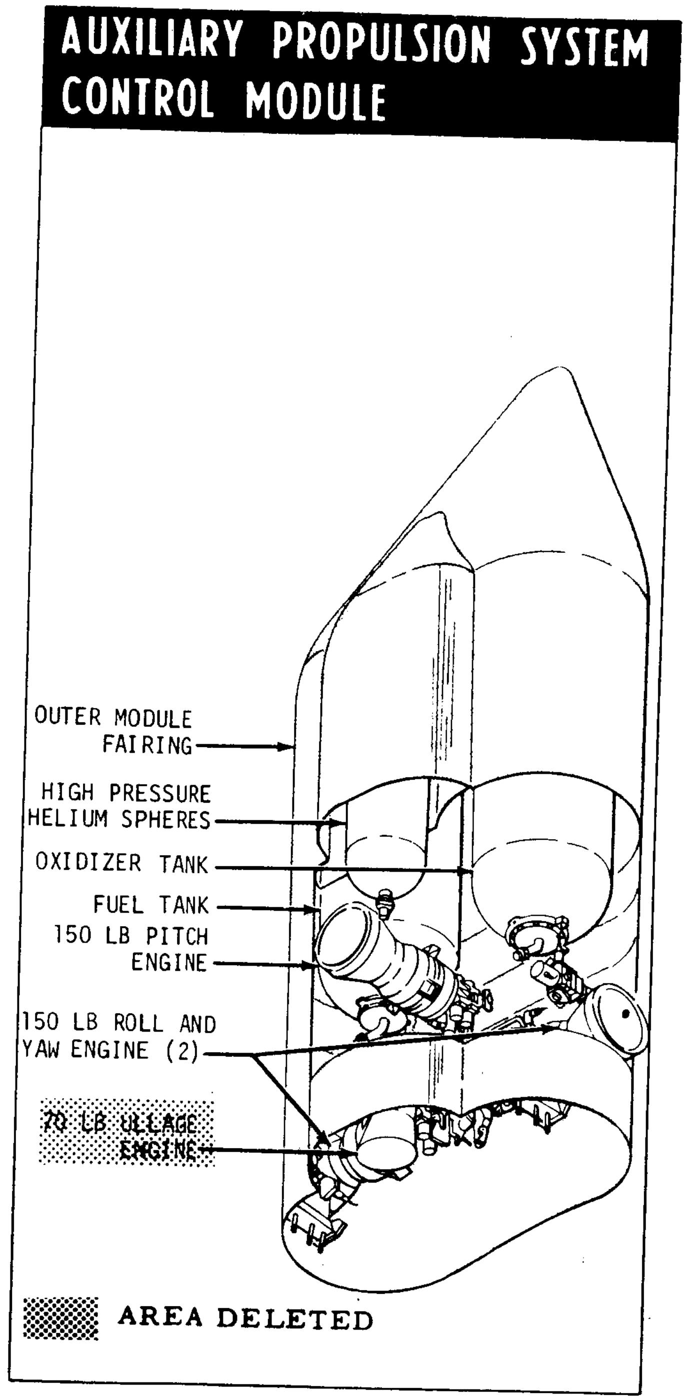
The LEO mission requirements for J-2S engine restart and idle mode operation at the end of the coast period will require a thrust vector control DC motor hydraulic system for gimballing the engines. The main turbo-pump is inoperative during idle mode operation. Another method is required to provide the hydraulic pressure necessary to gimbal the engines to complete the LEO mission. The system will be capable of providing gimballing during ground checkout. The present Douglas S-IVB DC motor auxiliary hydraulic pump system is recommended for this function. The recommended S-IVB DC motor system will require a 56 VDC battery package, power transfer switch and electrical control for start sequencing of the DC motors. Adjacent DC motor pumps will be powered from separate 56 VDC battery packages.

A balanced LH_2 tank vent system will be required during the coast period of the LEO mission. A new switch selector command will be required to arm the LH_2 tank balanced vent system prior to the coast period. The "all engine cutoff bus" will provide the signal to open the balanced LH_2 tank vent system prior to the coast period. The balanced vent system will be open for the remainder of the mission. Four latching type relays will be required to implement this new function.



Ref 10.4-32, Fig. 7-14

FIGURE 10.5.3.4-2 LEO CONTROL SYSTEM BLOCK DIAGRAM



Ref 10.4-32, Fig. 6-14

FIGURE 10.5.3.4-3. AUXILIARY PROPULSION SYSTEM CONTROL MODULE

10.5.3.4.2 (Continued)

b. IU Attitude Control

To convert from a J-2S/LOR configuration to a J-2S/LEO configuration will require major modification to the Flight Control Computer (Reference Figure 10.5.3.4-4). The additional requirements to be met are:

No S-IVB Stage.

Two states of burn for the S-II Stage (mainstage and idle).

S-II pitch and yaw attitude rate channels require different filters in mainstage and idle modes.

S-II attitude error filters and roll attitude rate filter require an additional switch point for idle mode.

An additional constraint for this study was no major pin function changes would be allowed.

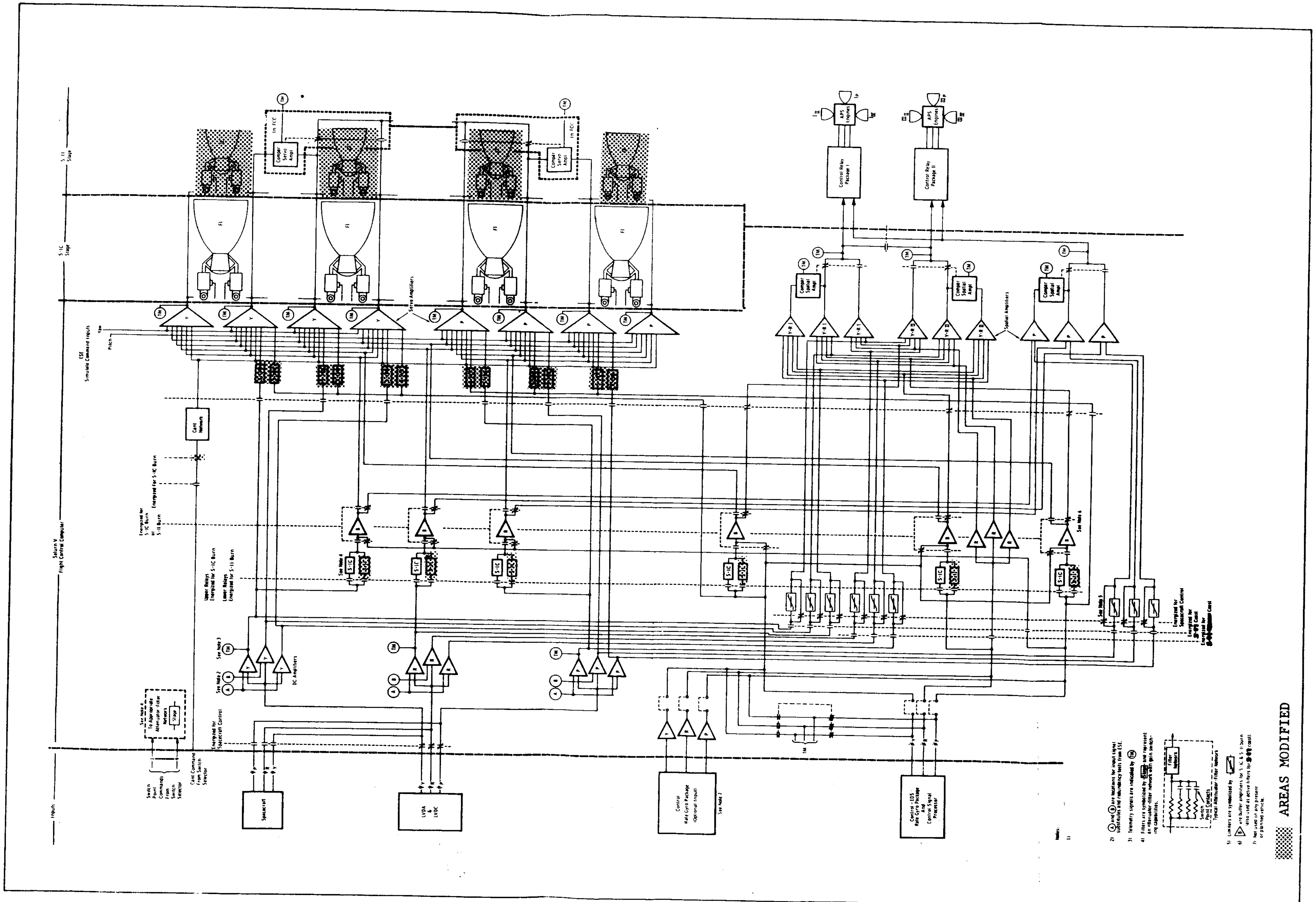
Since no S-IVB stage is used on the J-2S/LEO vehicle, 17 modules used only for S-IVB operation will be removed from the FCC. Of these 17 modules, 15 will be replaced by blank boards; two will be replaced by the additional S-II idle mode pitch and yaw attitude rate filters. Replacing the 15 modules with blank boards minimizes the impact to the existing FCC design.

The existing attitude error filters and the roll attitude rate filter will be redesigned to add an additional switch point for S-II idle mode. This switch point will be connected to the S-II idle mode signal rather than a switch selector discrete. Connecting it to the idle mode signal ensures that the filter is in the proper configuration for the idle mode without additional switch selector functions.

The mode signals for S-II idle mode will be routed into the FCC on pins presently designated for switch points 8 and 9. These pins were chosen because the wiring presently exists in IU Networks but switch points 8 and 9 are not used in the Saturn configuration. This allows minimum impact to convert from one mission to a J-2S/LEO mission.

Presently, only six of the eight 50 MA Servo Amplifiers are loaded during coast to ensure minimum transients when going from coast to S-IVB burn mode. Since the J-2S/LEO mission will require going from coast to S-II mode, the remaining two 50 MA Servo Amplifiers will also be loaded during coast.

The relays to switch the dummy loads in and out as well as the relays required to switch the idle mode filters in and out will be derived from the unused S-IVB



Ref 10.4-31, F. 3.3-5

FIGURE 10.5.3.4-4.
FLIGHT CONTROL COMPUTER

10.5.3.4.2 (Continued)

relays. Presently, 10 relays are used in the S-IVB mode only; six of these will be used for the above switching.

All of the aforementioned changes will require major modification to the FCC wiring harness.

10.5.3.4.3 Mode and Sequence (LEO)

The basic mode and sequence is described in paragraph 10.5.1.4.3 of the J-2S/LOR Astrionic System Study.

a. S-II Mode and Sequence

Converting from J-2S/LOR to J-2S/LEO missions requires modification to the S-II Sequencing system.

During the LEO mission new switch selector commands are required for mainstage cutoff, thrust vector control, LH_2 balanced vent system, restart and extended idle mode operation. The following switch selector changes are required:

<u>New S-II Switch Selector Commands</u>

"Mainstage Cutoff"

"Mainstage Cutoff Reset"

"Restart Arm"

"Restart Arm Reset"

"LH_2 Balanced Vent System Arm"

"TVC AUX Hydraulic Pumps #1 and #2 ON"

"TVC AUX Hydraulic Pumps #3 and #4 ON"

"TVC AUX Hydraulic Pumps #1 and #2 OFF"

"TVC AUX Hydraulic Pumps #3 and #4 OFF"

<u>Delete S-II Switch Selector Commands</u>

"LH_2 Step Pressurization"

10.5.3.4.3 (Continued)

Two new switch selector commands will be required for the mainstage operation cutoff (mainstage operation cutoff and mainstage operation cutoff reset). A computer mainstage cutoff backup will be provided by LH_2 and LOX tank sensors (two out of five) dry indication signal. A 10-second idle mode operation will be required prior to the 45-minute coast period to allow the flight control system to arrest the vehicle attitude transient level to a maximum of ± 2.5 degree attitude error. The Reaction Control System will provide attitude control during the 45-minute coast period and is limited to a small body rate and ± 2.5 degree attitude error. The IU computer will issue an "all engine cutoff" command that is timed 10 seconds from mainstage cutoff for insertion into the 100 nautical mile orbit. The LH_2 and LOX tank sensors will be moved up in the tanks to allow sufficient remaining propellants for restart and idle mode operation to complete the 300 nautical mile orbit mission.

The prevalves will be locked out (open) beginning with "all engine cutoff" at the 100 nautical mile altitude and will remain locked open for the remainder of the LEO mission.

The J-2S engines during the LEO mission will be cut off (zero thrust) only by the switch selector channel 18 "all engine cutoff" command or the PD and EDS emergency termination. Engine cutoff bus #1 or #2 will provide each engine with redundant cutoff signals. Blocking diodes prevent one engine from terminating another. Ten non-latching type relays will be deleted from the LOR engine cutoff circuitry for the LEO mission.

Switch selector channel 18 will apply the main and instrumentation bus to engine cutoff relays. The same +28 VDC signal will be utilized to lock open the prevalves, lock up the hydraulic accumulators and open the LH_2 balanced vent system. Switch selector commands will be required to arm the LH_2 and LOX sensor mainstage cutoff circuitry prior to switch selector command for mainstage cutoff. During mainstage operation the one engine out and emergency engine cutoff capability will be retained. In case of an early engine out during mainstage operation the prevalve for that engine will close at the runout of the 430 millisecond pre-timer. The remaining four engines will continue mainstage operation until mainstage cutoff. For the LEO mission an internal engine cutoff will not terminate the remaining engines under any circumstance.

The J-2S engines have been designed for in-flight restart and during the LEO mission this new function will require two commands (Restart Arm and Restart Arm Reset) from the switch selector.

Two new switch selector commands will be required to start the motors during the coast period at approximately 3 minutes prior to engine restart for temperature control of the hydraulic fluid and to build up the pressure in the accumulators.

10.5.3.4.3 (Continued)

The first switch selector command will start DC motors for engine #1 and #2 from separate 56 VDC battery packages. The second switch selector command shall be delayed approximately 0.5 seconds to allow the starting surge to diminish sufficiently before applying power to start the DC motors for engine #3 and #4. This process will reduce the high current drain on the batteries during motor start.

A new switch selector command will also be required to lock up the hydraulic accumulator reservoir at the time of "all engine cutoff" prior to the coast period during the LEO mission. Switch selector channel 12 command, "Unlock the hydraulic accumulator," will be required during engine restart for the LEO mission.

The following steps provide the requirements for switch selector commands and electrical control relays to implement the flight events for engine gimballing by the auxiliary DC motor pump system.

> There will be approximately a 10-second idle mode operation starting with termination of mainstage operation.
>
> The hydraulic energy in the accumulator at the termination of mainstage operation will provide for engine gimballing during the 10-second idle mode period.
>
> The hydraulic accumulators will be locked up beginning with "all engine cutoff" at the 10-second idle mode termination.
>
> The hydraulic accumulators will be unlocked shortly after commanding the thrust vector control DC motor pump system on. The DC motor pump will be turned on three minutes prior to engine restart for hydraulic fluid warm up and to charge up the accumulator.

The LH_2 tank main vent valves will not require electrical lockup during the coast period. The LH_2 step pressurization switch selector channel 7 command, during mainstage operation, will not be required for the LEO mission. A balanced LH_2 Tank Vent System will be required during the coast period of the LEO mission. The LH_2 tank will be vented through propulsive nozzles to maintain settled propellants and to control tank pressure. The thrust from the vent nozzles will be balanced to minimize stage turning movement.

b. IU Mode and Sequence

Converting from J-2S/LOR to J-2S/LEO mission requires modifications to incorporate an additional FCC switch point for the S-II idle mode. Paragraph 10.5.3.4.2b, IU Attitude Control, describes the modifications recommended to accomplish the change.

10.5.3.4.3 (Continued)

Additional switch selector events will be added to the Flight Sequence table; however, no impact will be required above normal mission modifications. A general flight sequence is given in paragraph 10.5.3.4.10.

10.5.3.4.4 Telemetry and Measurement

Converting from J-2S/LOR to J-2S/LEO missions will require no major modifications to the Saturn V, S-II, and IU stages.

a. S-II Telemetry and Measurement

Telemetry and measurement associated with the J-2S engine are discussed in paragraph 10.5.1.4.4a, S-II TM and Measurement of the J-2S/LOR Astrionic System Report. A summary of J-2S/LEO measurements is given in Table 10.5.3.4-I.

TABLE 10.5.3.4-I. STAGE MEASUREMENTS SUMMARY

TYPE	QUANTITY
Acceleration	4
Acoustic	--
Discrete Signals	185
Flowrate	10
Liquid Level	4
Miscellaneous	4
Position	36
Pressure	129
RPM	10
Strain	--
Temperature	118
Vibration	--
Voltage, Current, Frequency	37

10.5.3.4.4 (Continued)

The Reaction Control Systems will add a total of 48 new flight measurements; 24 identical measurements for each of the two systems. Use of the alternate space station reaction control system will eliminate this requirement.
The Thurst Vector Control System required for gimballing the engines during the extended idle mode operation of the LEO mission will be powered by a DC motor pump system. Each outboard engine will require one DC motor pump. The DC motor will be housed in a container that will be pressurized. The air pressure in each DC motor container will be monitored during flight in addition to the air container.

Addition of new batteries to satisfy power requirements for the LEO mission has established the requirement for the 12 new measurements.

A change to the telemetry equipment will also be required to provide the required number of low-level channels.

Two new measurements are required for the LH_2 balanced vent system for the LEO mission.

It should be noted that without sufficient number of picket ships, extensive periods will exist where the vehicle will not be within range of a ground receiving station and data will not be available during these periods.

b. IU Measuring and Telemetry

Transformation from J-2S/LOR to J-2S/LEO will require no major hardware modifications to the IU Telemetry and Measurement Subsystem. Measurements associated with the non-existant S-IVB stage will be deleted from IU measurement system. However, with the RCS system being implemented on the S-II stage a portion of the measurements will be replaced with the corresponding function measurement associated with the S-II.

One modification that must be considered in the study of two-stage vehicle missions is the possibility of routing the IU PCM/DDAS Telemetry Link to the S-II stage and transmitting data via the S-II PCM telemetry system. Flight control data is presently transmitted redundantly from the IU and S-IVB stage via two systems. One is the PCM/DDAS system in the IU, the other is PCM telemetry system located in the S-IVB stage. Groundrules for the two-stage vehicle did not define the space station capabilities, therefore, the possibility of losing the redundant S-IVB telemetry was addressed.

The S-II telemetry system is capable of accepting PCM/DDAS information from the IU and transmitting via their own PCM system. The modification would require some hardware changes to the S-II system.

10.5.3.4.5 Radio Command

No modifications will be required to the S-II stage or the Instrument Unit radio command as a result of the J-2S/LEO mission.

10.5.3.4.6 Tracking

No modifications will be required to the S-II stage or the Instrument Unit as a result of the J-2S/LEO mission.

10.5.3.4.7 Power Supply and Distribution

a. S-II Power Supply and Distribution

The J-2S engine requirements for the LEO mission shall include the capability to terminate mainstage operation and return to idle mode operation prior to the 45-minute coast period. J-2S engine restart capability and idle mode operation will be required at the end of the coast period to complete the LEO mission. Since the main and instrumentation batteries will be required to provide additional power necessary for the 45-minute coast period, restart and idle mode operation, new batteries will be required for the LEO mission (Reference Figure 10.5.3.4-5).

The battery selected for these applications will have the following characteristics:

Voltage:	26 to 31 volts for load current from 40 to 110 amperes and for a 0 to 105 degree F ambient.
Capacity:	75 ampere-hours.
Cells:	20 cells with selector for 19 cells.
Size:	19 x 11 x 8.5 inches.
Weight:	85 pounds (maximum).

The capacity requirement is based on a flight duration of approximately one hour, main battery flight usage of 48 ampere-hours, and instrumentation battery flight usage of 50 ampere-hours. The 75 ampere-hour design requirement will provide a satisfactory margin for checkout and transfer test usage.

The battery minimum voltage requirement is based on the minimum allowable of 24 volts at all using equipment and a maximum of two volts drop from the battery. The battery maximum voltage requirement is based on a maximum allowable of 31 volts for J-2S engine control and ignition busses.

10.5.3.4.7 (Continued)

A heater and a temperature transducer will be incorporated inside each battery to maintain and verify proper operating temperature. The battery case will be sealed, and the battery case and individual cells will be provided with a relief valve and vents, respectively, which permit escape of gas without loss of electrolyte in a gravity-free, vacuum environment.

If the coast period is lengthened to 105 minutes, the 75 ampere-hour battery described above could still support the main bus power requirements if the Propellant Utilization system is switched off during the coast period. This would result in a main battery flight usage of 57 ampere-hours. The increased coast period would result in an instrumentation battery flight usage of 100 ampere-hours. The 75 ampere-hour battery described above could be modified to a capacity of 110 ampere-hours with no increase in size or weight.

The 56 VDC power required by the TVC system will be provided by two battery packages previously used in J-2 configuration for recirculation. Each battery package will consist of two 28 VDC batteries which have the following characteristics:

Voltage:	26.5 to 31 volts for load currents from 110 to 170 amperes and for a 0 to 105 degree F ambient. During motor start the voltage will not exceed the limits of 18 to 33.5 volts.
Capacity:	25 ampere-hours.
Cells:	21 cells with selector for 20 cells.
Size:	16.5 x 10 x 8 inches.
Weight:	60 pounds (maximum).

The capacity requirement is based on a flight usage of 12 ampere-hours and a ground usage (checkout and transfer tests) of six ampere-hours. The 25 ampere-hour design requirement will provide satisfactory margin for mission variations.

A heater and a temperature transducer will be incorporated inside each battery to maintain and verify proper operating temperature. Stage networks will be planned to inhibit heater operation during J-2S engine ignition. The battery case will be sealed, and the battery case and individual cells will be provided with a relief valve and vents, respectively, which permit escape of gas without loss of electrolyte in a gravity-free, vacuum environment.

10.5.3.4.7 (Continued)

Power and control switching will require one additional power transfer switch to transfer the two TVC buses from ground power to batteries and four motor start switches for motor start and stop.

An alternate proposal is the use of an AC power system to supply the TVC motors. an AC system will eliminate problems with DC motor brush life and eliminate the requirement for a unique air supply to pressurize the motor throughout ground and flight operation. The AC system will require a special motor winding to match the output of the inverter but similar winding are already in use on the recirculation pump motors. The inverter can be designed to incorporate a low frequency, low voltage output during motor start to limit current inrush without reducing starting torque. However, implementation of the AC system would require extensive development and qualification programs since there is no qualified, off-the-shelf inverter of the above type available at present.

The TVC battery heater power is supplied from the +2D11 main bus and the heaters will automatically cycle on and off with temperature.

Switch selector command "all engine cutoff reset" channel 31 will be required prior to S-II engine restart to remove the TVC heater load from the +2D11 main bus. The heater circuit will remain disabled because of the small time duration for the remainder of the LEO mission. Current limiting resistors are incorporated to provide GSE monitoring of the TVC battery heater power. Two new DPDT latching type relays are required to implement this change.

The main bus shall provide 28 VDC electrical power to operate the balanced venting system.

b. IU Power Supply and Distribution

No additional hardware requirements will be required in the IU Power Supply and Distribution system as a result of the J-2S/LOR to J-2S/LEO conversion.

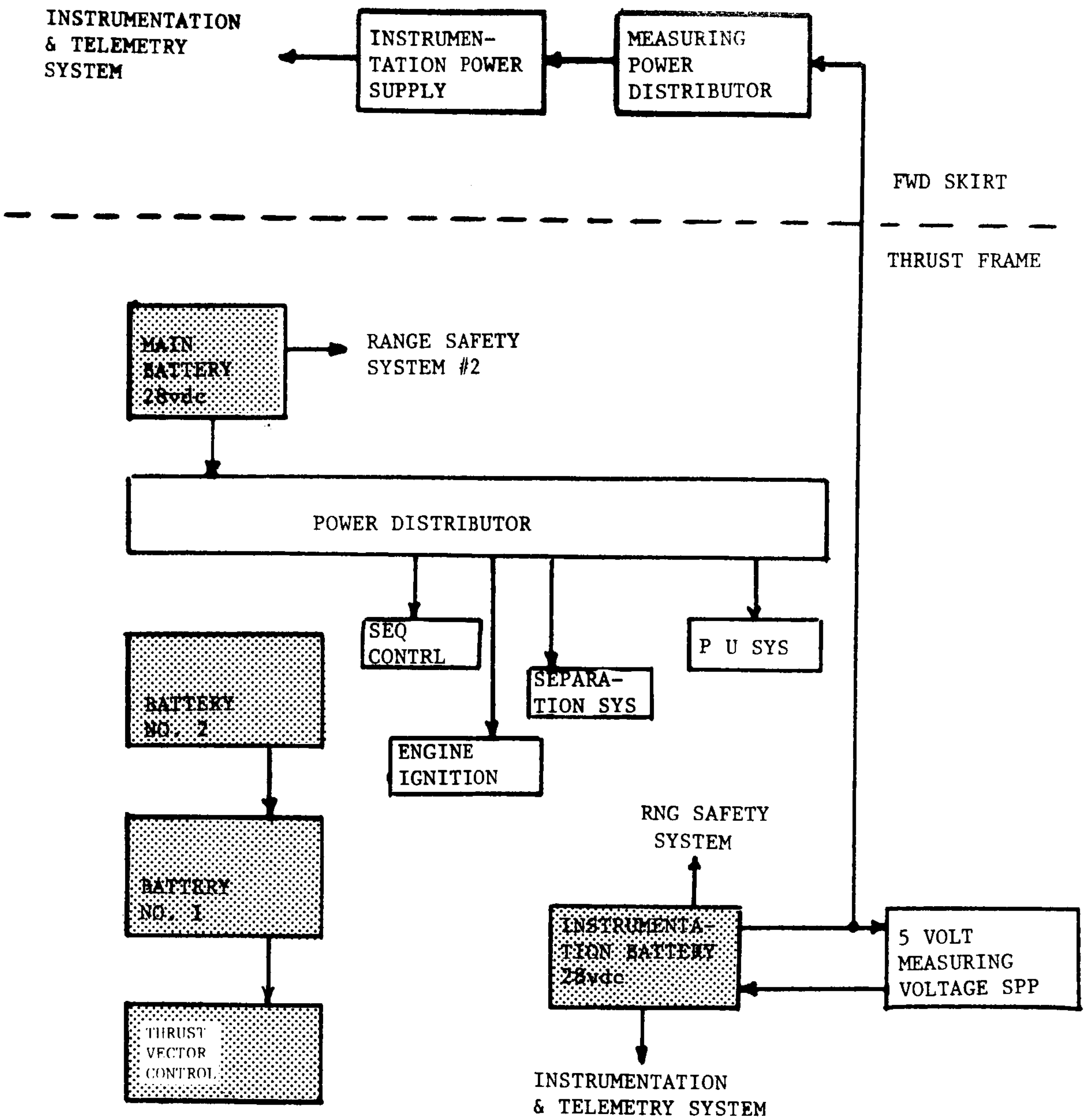
10.5.3.4.8 Emergency Detection System

During Saturn mission with a S-IVB stage coast and restart phase, the Emergency Detection System displays two functions in the spacecraft as manual abort indication:

a. LOX Tank Pressure

b. Fuel Tank Pressure

These functions are available in the S-II fuel and LOX tanks and are presently partially wired to the spacecraft. To convert from J-2S/LOR to J-2S/LEO would



Ref 10.4-34, Fig. 29

FIGURE 10.5.3.4-5. S-II POWER DISTRIBUTION

10.5.3.4.8 (Continued)

require modification in the EDS system to complete the electrical interface between the spacecraft display and these two S-II indications (Reference Figure 10.5.3.4-6).

J-2S/LEO "thrust OK" EDS switches during idle mode are affected as described in paragraph 10.5.1.4.8, J-2S/LOR mission.

10.5.3.4.9 Separation

The Saturn vehicle separation system is described in paragraph 10.5.1.4.9 of the J-2S/LOR Astrionic System Study. No additional hardware modifications will be required to the S-II stage or the IU as a result of the J-2S/LEO mission.

10.5.3.4.10 Flight Program

A description of the Saturn V Flight Program is given in paragraph 10.5.1.4.10 of the J-2S/LOR Astrionic System Study. Converting from the J-2S/LOR to the J-2S/LEO will require modification to the flight program.

a. Guidance Analysis

The LEO mission involves a two-stage vehicle insertion into an elliptical orbit at 100 nautical miles. A second burn of the second stage using idle mode thrust is used to circularize the orbit at 300 nautical miles. The analysis and simulation was divided into three distinct sections: ascent, coast and idle mode circularization.

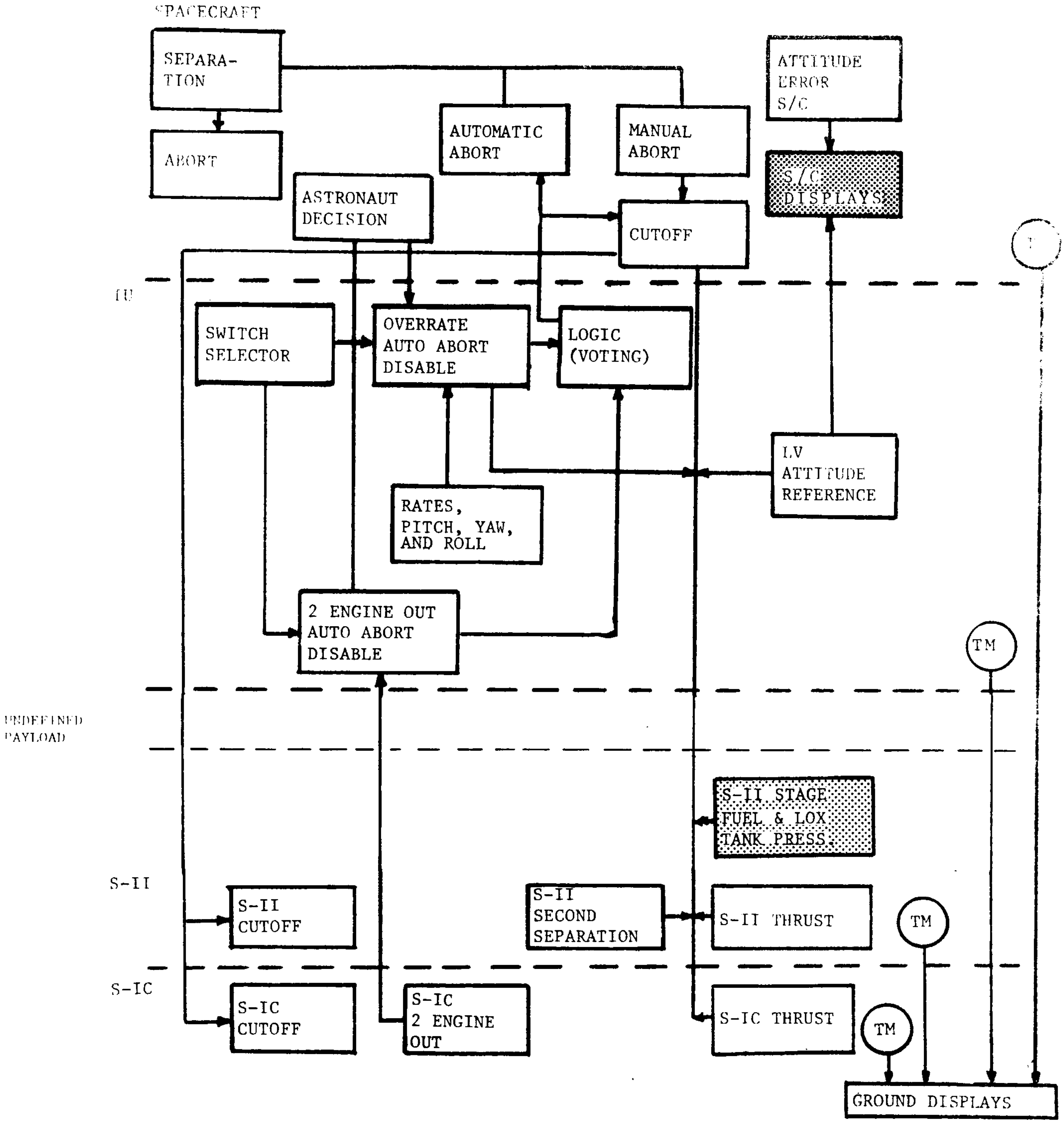
The ascent phase was simulated using the 6-D Saturn V simulator utilizing "time-tilt" guidance in the S-IC stage and two-stage IGM guidance in the second stage.

Modifications may be accomplished through guidance constants and involves no impact on the flight software major loop.

Simulations were also made of the J-2S second S-II burn to circularize at a 300 nautical mile altitude. Result of the simulation was a recommendation to replace the M/F filter. This modification was recommended as it did not require impact on the flight program except scaling of inverse acceleration.

The boost minor loop was evaluated by the LEO mission and no impact to the flight program was found.

The AS 505 orbital program was utilized as a baseline program for the LEO mission. Navigation, guidance, attitude control, event sequencing, data management, ground command processing and data compression would be adequate for the LEO mission.



Ref 10.4-34, Fig. 7

FIGURE 10.5.3.4-6. LEO EMERGENCY DETECTION SYSTEM

10. 5. 3. 4. 10 (Continued)

There are, however, modifications necessary to the targeting equations to allow the normal calculations of the restart state and estimation of the terminal targeting for a J-2S/LEO mission. Reference Table 10. 5. 3. 4-II for Mission Time Lines.

b. Control Analysis

Software simulations were utilized in the Control System Analysis to determine anomalies in transforming from the J-2S/LOR to the J-2S/LEO mission.

Control analysis for the J-2S/LEO mission has indicated that with the exception of normal control gain and shaping network design

No modifications are required to the baseline S-IC FCS.

No modifications are required to the baseline S-II FCS for S-II mainstage burn mode.

One additional gain switch will be required in the S-II pitch and yaw attitude error shaping networks for S-II idle mode.

An additional pitch and yaw attitude rate shaping network will be required for S-II idle mode.

One gain switch will be required in the S-II roll shaping networks for S-II idle mode.

The gains in the pitch/yaw attitude error networks, the gains in the roll networks and the pitch/yaw attitude rate networks will be switched at S-II mainstage cutoff.

10. 5. 3. 4. 11 Instrument Unit

A basic description of the IU is given in paragraph 10. 5. 1. 4. 11 of the J-2S/LOR report. Converting from J-2S/LOR to J-2S/LEO missions will require no impact to additional IU hardware.

10. 5. 3. 4. 12 Environmental Control System

a. S-II Environmental Control System

Converting from J-2S/LOR to J-2S/LEO mission requires no additional hardware modifications to the S-II Environmental Control System.

TABLE 10.5.3.4-II. APPROXIMATE MISSION TIME LINE

LOW EARTH ORBIT

Time Base	Time (Seconds)	Event
TB0	-17	Guidance Reference Release
TB1	0	Liftoff
	82	Q Max
TB2	149	Center Engine Cutoff
TB3	161	S-IC Cutoff
	165	S-II Ignition (idle)
	166	S-II Mainstage
	168	S-II MR Shift (5 - 5.5)
	193	Interstage Separation
	198	LES Separation
	396	S-II MR Shift (5.5 - 4.7)
TB4	486	S-II Cutoff (1st burn)
	~ 45 min.	Transfer Coast
TB5	3065	Local Horizontal Tracking
TB6	3165	S-II Reignition (idle)
TB7	3305	S-II Cutoff (2nd burn)

10.5.3.4.12 (Continued)

b. IU Environmental Control System

Converting from J-2S/LOR to J-2S/LEO mission will require no additional modification to the IU Environmental Control System. The IU ECS has the capability of meeting design functions with the undefined S-IVB stage ECS completely capped.

10.5.3.4.13 Electrical Support Equipment

a. S-II Electrical Support Equipment

Implementation of the J-2S engines on the S-II stage for a LEO mission, requires modification to the S-II stage hardware. Hence, the ESE used in the manufacturing checkout and the static firing checkout of the S-II stage must be modified to provide checkout capability of the modified hardware. The S-II ESE modifications discussed below will update the North American Rockwell (NAR) checkout facilities at Seal Beach, California and Mississippi Test Facility (MTF) to check out the S-II/J-2S stage. Similar changes will be required at KSC. However, those changes will not be discussed specifically since various support contractors control the hardware at KSC.

1. Computer Complex Checkout Station (C7-100)

The only changes to the C7-100 Computer Complex end items resulting from J-2S engine implementation are to the C7-102 Test Conductor Console. Three functions are added to the Hazardous Monitoring panel as follows:

(a) Thrust Vector Control System DC greater than 64V.
(b) Thrust Vector Control System DC greater than 60 V.
(c) Thrust Vector Control System DC low.

These changes require addition of legend plates and addition of cabling in the J-Boxes which connect the hazardous functions to the C7-102. Note that these functions can directly replace the first three functions deactivated for the LOR mission (see 10.5.1.4.13a of LOR Mission Study). Cabling changes can be minimized by combining LOR and LEO changes in this area.

2. Electrical Checkout Station (C7-200)

The C7-202 Manual Control and Display Rack changes consist of adding a new switch selector control panel to provide for the nine new switch selector commands required, and adding 17 new commands using an existing spare panel assigned for Static Firing #2. The latter commands are implemented by changing nomenclature on the panel.

10.5.3.4.13 (Continued)

The C7-204 Signal Distribution Rack changes consist of adding 17 each of the flip-flops, logic gates, relays and relay drivers, plus the necessary patching (34 patch-cords) to implement these new commands.

These modifications also require four new rack and drawer harnesses and a new patch panel assembly.

The C7-211 Scanner Rack changes are due to approximately 27 added functions to be monitored. This is accomplished by adding 49 patchcords to the A3 patch panel and 29 patchcords to the A17 patch panel. Also affected are the two new Program Board Assemblies (A3 and A17) which were added for the LOR mission.

The C7-212 Discrete Display Rack changes require the addition of approximately 12 legend lights to display new functions being monitored. These are implemented by the addition of 24 patchcords.

The C7-213 Interlock Relay Rack changes due to the LEO mission have not yet been determined. It is presently assumed that a maximum of 20 new functions may need to be interlocked. Based on this assumption, approximately 60 patchcords would be added to implement existing spare relays for these functions. Also affected are the two new Program Board Assemblies (A4 and A15) which were added for the LOR mission.

There will be no changes to the following C7-200 Station racks for the LEO mission.

C7-201	Automatic Control Rack
C7-205	Special Data Rack
C7-208	Station Control and Display Rack
C7-209	Manual Control and Display Rack
C7-210	Stage Substitutes Rack

3. Range Safety Command Receiver (RSCR) Checkout Station (C7-307)

There are no changes required for the C7-307 Rack resulting from J-2S engine incorporation.

4. Digital Data Acquisition System (DDAS) Checkout Station (C7-400)

No changes are required for the C7-400 Station due to J-2S engine incorporation.

5. Telemetry System Checkout Station (C7-500)

No changes to the C7-500 Station are required for J-2S engine incorporation.

10.5.3.4.13 (Continued)

6. Static Firing Control Station (C7-800)

Changes to the C7-801 Local Static Firing Rack consist of adding nine new switch selector commands, 11 new switch commands and 17 legend plates to spare indicator lights. The new switch selector commands will require the addition of three diode boards and wire harness changes to the Switch Selector Encoder Drawer, and nine legend plates to existing spare switchlights on the Switch Selector Panel. The 11 new switch commands will utilize spare switches on the new panel added for the LOR mission. Five of these will have associated indicator lights. The remaining 12 indicator lights will be located on various other panels of the C7-801, depending on the system involved and availability of spares.

The C7-802 changes consist of adding patchcords to implement the nine switch selector commands of the C7-801, and to connect 17 existing spare relays into the circuits required. This will require approximately 60 additional patchcords.

The C7-805 Engine Cutoff Rack is not affected by LEO mission requirements.

7. Cable and J-Box Requirements

The cabling and J-Box installations provide multiconductors for the interconnection and the transmission of electrical power and signals between the S-II stage and ESE and between ESE end items. The installed cabling is capable of sustaining the maximum load requirements during any checkout phase, protecting as necessary, circuits with fuses or resistors and isolating rack interconnections with diodes. The same capability will be retained for the LEO mission.

(a) MTF J-Box and Cable Installation (C7-35, C7-38 and C7-40)

Changes to the MTF cable sets require additional revisions to the terminal distribution rack wire lists for the control centers and test stands. Changes are summarized as follows:

C7-38 (Control Center)
- Add 75 jumper wires.

C7-35 or C7-40 (Service Center, A2 or A1)
- Add 250 jumper wires.
- Add one new cable assembly.
- Add one new umbilical cable and adapter.
- Add six drag-on cables.
- Add three stage mounted GSE harnesses.
- Add seven receptacle/harness assemblies in TRB's.

10.5.3.4.13 (Continued)

(b) Seal Beach J-Box and Cable Installation (SDD-154, SDD-196, SDD-197)

Changes to the Seal Beach cable sets consist of the following:

Add six new drag-on cables.
Add one new umbilical adapter and GETS cable.
Add two new cables for battery simulation.
Add eight receptacles in junction box.
Add one conduit hub to junction box.
Add and/or change 50 fuses.
Change 200 wire terminations.

(c) Stage Station Test Electrical Harnes (C7-43) Field Site Installation of Stage Mounted ESE

These changes require revision to stage mounted ESE drawings at Seal Beach, MTF and KSC to show new carry-on and drag-on cable clamp support locations, in addition to revision to the ESE static firing cable sets.

8. Power Distribution System, Test Stand A2 (C7-80), Test Stand A1 (C7-81)

The 56 VDC circuit required for the Thrust Vector Control System will require addition of a 56 VDC panel circuit with circuit breaker and a battery simulator circuit. With relatively minor modification, the 56 VDC Recirculation System which was deactivated for the LOR mission can be utilized for this function.

9. Digital Events Recorder (C7-77)

There are no hardware changes required for this equipment. Revision to ICD's is needed to add 21 new channels at Seal Beach and 27 new functions at MTF. These are all existing spares.

10. Remote Distribution Rack (C7-41)

As stated above, the LOR mission deactivated the 56 VDC distribution system used for the recirculation bus. The LEO mission requires two 56 VDC systems for the Thrust Vector Control Hydraulic system. This will be accomplished by using the existing 56 VDC power system and re-nomenclaturing it as TVC bus. This existing circuit fulfills one of the requirements, and a complete new circuit will be added requiring one circuit breaker, one shunt and a relay, plus interface connections.

The existing capabilities of the Main and Instrumentation power circuits are sufficient to accommodate the anticipated 60 to 80 ampere loads.

10.5.3.4.13 (Continued)

Since the spare capability of the C7-41 has been used up, an additional panel will be required to accommodate the above hardware, and an interface connection and new harness will be required.

11. Time Code Rack (C7-48)

There are no changes to this rack resulting from the LEO mission.

12. Ground Equipment Test Set (C7-44)

The changes to the C7-44 will consist of adding approximately 32 patchcords to each of two patch panels and revising one wire harness in the IU Decoder Drawer (approximately 12 wires). Also affected will be the two new patch panels added for the LOR mission.

13. Engine Sequence Recorder (SDD 273), Power Supply 56 VDC (SDD 337)

There are no changes to this equipment resulting from the LEO mission.

b. IU Electrical Support Equipment

The purpose of the IU ESE section is the definition of ESE modification required by the J-2S engines used on a Saturn V vehicle for a LEO mission. The ESE modifications were determined by considering the changes necessary to uprate a baseline AS 505 IU for use on the LEO mission. Implementation of the J-2S engines on a Saturn V vehicle and the mission requirements of a LEO mission will require no hardware modifications to the ESE since the LEO mission requirements add no additional hardware or hardware requirements. The software will require minor changes in the automated subsystem checkout programs and the IU overall checkout program.

Automated Subsystem Checkout Programs

The following subsystem checkout programs will require test parameter changes and/or minor program rewrite for the LEO mission:

1. Control Subsystem

   (a) Control Subsystem.

   (b) A_1 Gain.

   (c) A_0 Gain.

10.5.3.4.13 (Continued)

(d) Control Computer APS.

(e) Control Computer Relay Redundancy.

(f) Control System Nulls.

(g) Engine Deflection.

2. Electrical Subsystem

(a) Power Distribution and Control.

(b) General Networks.

(c) Simulated Plug Drop.

IU Overall Checkout Program

The IU overall checkout program is a general program applicable to all Saturn V vehicles with little or no modification from one vehicle to another. The mission requirements of the LEO mission are such that the basic checkout program does not require modification. All changes due to the LEO mission can be loaded into the basic program via user controlled data tables. These tables can be updated and maintained to the latest mission requirements to be inserted into the basic program without impacting that program.

Similar programs at KSC will also require updating.

10.5.3.4.14 Propellant Management

a. S-II Propellant Management

The new batteries for the LEO mission (45 minute coast period) will be capable of a two hour mission by removing Propellant Management power during the extended coast period. The power was calculated for an identical LEO mission (one main-stage operation and one three-minute idle mode period) except for a 105-minute coast period.

Power will be removed from the Propellant Utilization and Propellant Level Monitor DC power buses during the 105-minute coast period to conserve power. Power will be switched on again prior to engine restart to obtain temperature measurements.

Prior to liftoff the GSE will command power on for the Propellant Management System. A new switch selector command "Propellant Utilization Power Control Arm"

10.5.3.4.14 (Continued)

will be required to arm the power transfer switch circuit. The switch selector "all engine cutoff" Channel 18 command just prior to the 105-minute coast period will set relays causing the power transfer switch to cycle and remove power from the Propellant Utilization System.

Approximately three minutes prior to engine restart two new switch selector commands, "Propellant Utilization Power Control Reset" and "Propellant Utilization Power On" will be required to cycle the transfer switch and apply power to the Propellant Management System again. Power will be on for the remainder of the LEO mission (approximately six minutes).

Three new switch selector commands and three DPDT latching type control relays are required to implement this change.

b. IU Propellant Management

The J-2S/LEO configuration will require no hardware modification to the IU as a result of modification to the S-II Propellant Management System. A description of IU Propellant Utilization functions and recommendations is given in paragraph 10.5.1.4.14 c, IU Propellant Management.

10.5.4 J-2S/Polar Astrionic System Interface

10.5.4.1 Purpose

This section of the J-2S Improvement Study (Astrionic System Integration Polar Mission) defines the differences between the two-stage Saturn V vehicle AS 511/J-2S (Reference Figure 10.5.4.1-1) configured for the Polar Orbit mission (Reference Figure 10.5.4.1-2) and the three-stage Saturn V vehicle AS 511/J-2S configured for the Lunar Orbit Rendezvous (defined in paragraph 10.5.1).

If an existing system is changed or described to greater detail, only the changes or additions are described in this document and the Astrionic Handbook, Saturn V Flight Manual and/or the J-2S LOR Astrionic System Report (paragraph 10.5.1) is referenced for a more detailed description. Modification peculiar to J-2/J-2S interface will be so noted. Each illustration in this section is referenced to a corresponding illustration in the document from which it was taken.

10.5.4.2 J-2S Astrionic System (Polar Mission)

The overall Astrionic System of the J-2S/Polar AS 511 vehicle is shown in the simplified block diagram, Figure 10.5.4.2-1. Vehicle characteristics of the Polar mission are:

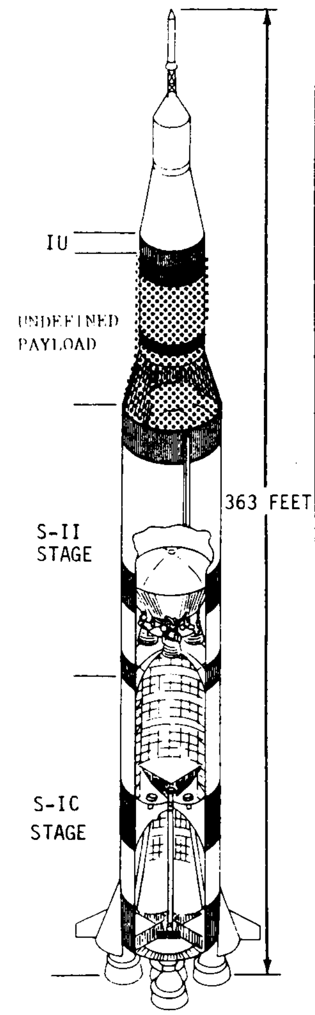
SATURN V LAUNCH VEHICLE

SOLID ULLAGE ROCKET AND RETROROCKET SUMMARY				
STAGE	TYPE	QUANTITY	NOMINAL THRUST AND DURATION	PROPELLANT GRAIN WEIGHT
S-IC	RETROROCKET	8	75,800 POUNDS• 0.541 SECONDS	278.0 POUNDS
S-II	ULLAGE	4	23,000 POUNDS‡ 3.75 SECONDS	[illegible] POUNDS
	RETROROCKET	4	34,810 POUNDS‡ 1.52 SECONDS	268.2 POUNDS

ENGINE DATA					
STAGE	QTY	ENGINE MODEL	NOMINAL THRUST EACH	NOMINAL THRUST TOTAL	
S-IC	5	F-1	1,526,500	7,632,500	
S-II	5	J-2	265,000	1,325,000	[illegible] SEC

STAGE DIMENSIONS		
	DIAMETER	LENGTH
S-IC Base (including fins)	63.0 FEET	138 FEET
S-IC Mid-stage	33.0 FEET	
S-II Stage	33.0 FEET	81.5 FEET
Instrument Unit	21.7 FEET	3.0 FEET

SATURN V STAGE MANUFACTURERS	
STAGE	MANUFACTURER
S-IC	THE BOEING COMPANY
S-II	NORTH AMERICAN-ROCKWELL
S-IU	INTERNATIONAL BUSINESS MACHINE CORP



PRE-LAUNCH LAUNCH VEHICLE GROSS WEIGHT ≈ 6,368,000 POUNDS

• MINIMUM VACUUM THRUST AT 120°F

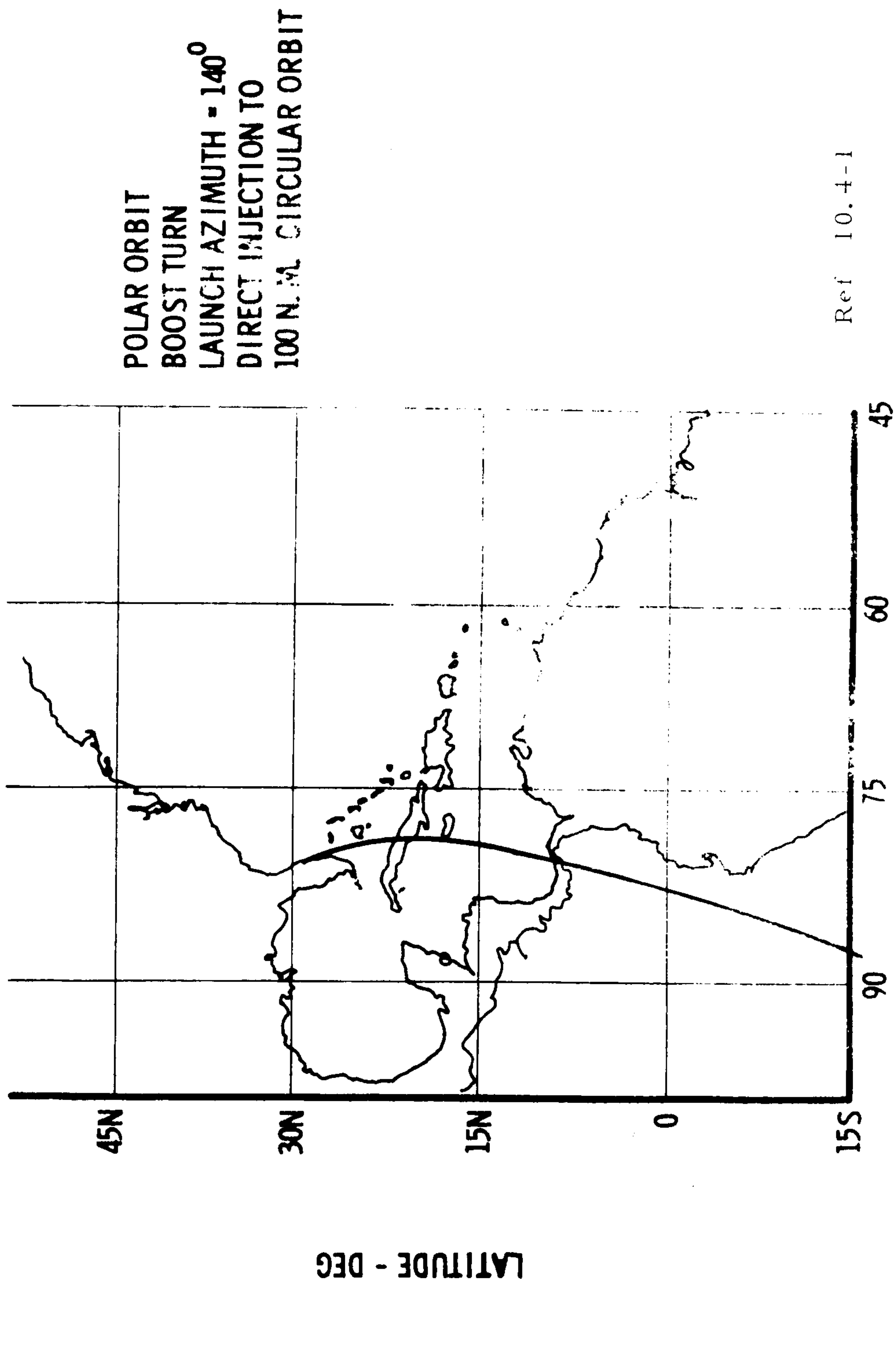
░ AREA CHANGED

‡ NOMINAL VACUUM THRUST AT 60°F

NOTE: THRUST VALUES, WEIGHTS, AND BURN TIMES ARE ALL APPROXIMATIONS.

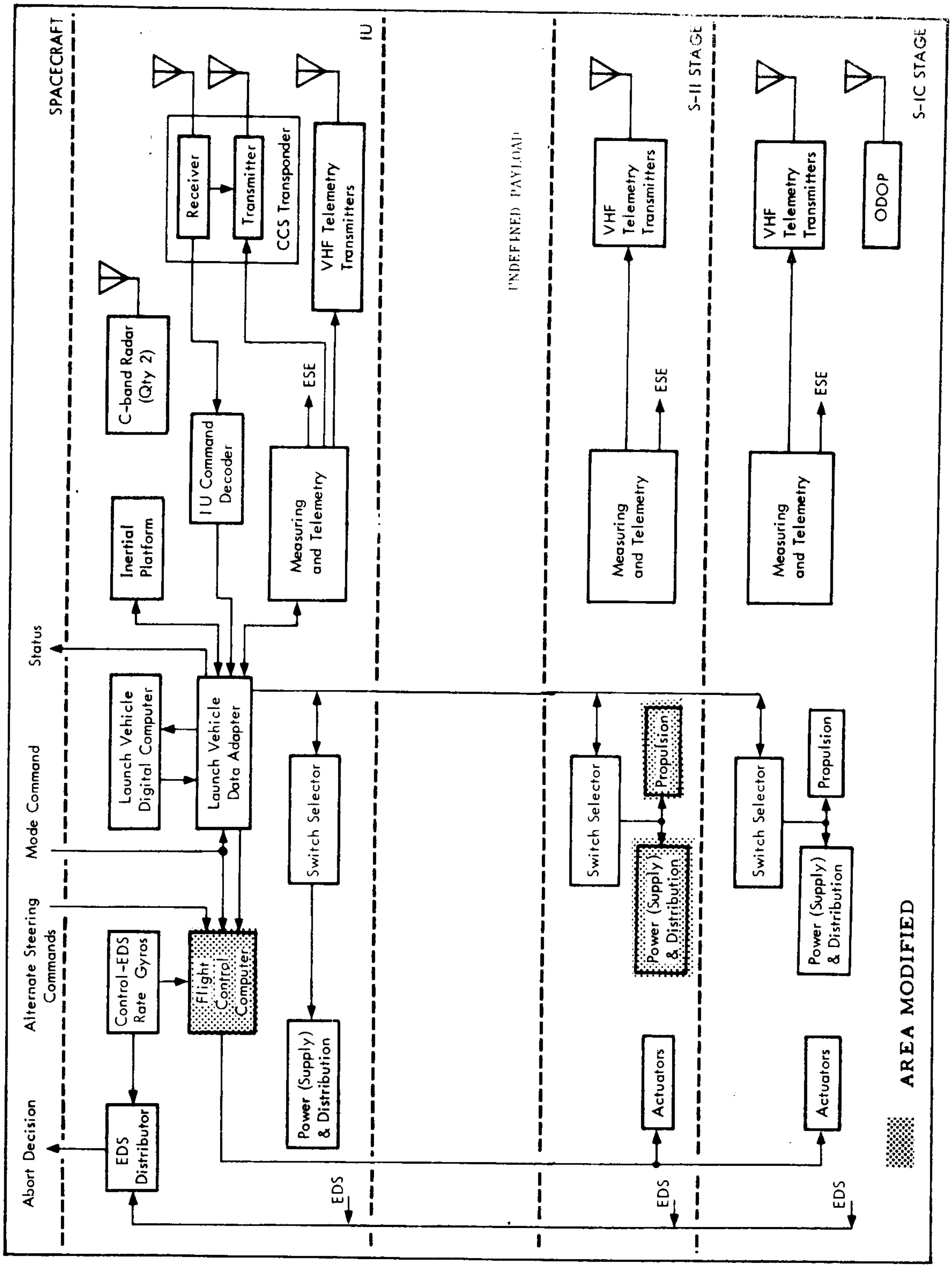
Ref 10.4-32, Fig. 1-3

FIGURE 10.5.4.1-1. J-2S/POLAR VEHICLE



Ref 10.4-1

FIGURE 10.5.4.1-2. POLAR ORBIT LAUNCH TRACE



Ref 10.4-31, Fig. 1.4-2

FIGURE 10.5.4.2-1. POLAR ASTRIONIC SYSTEM

10.5.4.2 (Continued)

The S-IVB stage is configured as an undefined space station.

The S-II stage must place Saturn vehicle into a 100 nautical mile Polar Orbit.

The S-II stage will have no RCS system nor will it have restart.

The major portion of the Astrionic System is located in the IU, which is mounted on top of the undefined S-IVB stage. During flight, the Astrionic System performs, or is associated with functions described in paragraph 10.5.1.2.

10.5.4.3 J-2S/Polar Electrical Interface

The J-2S/Polar study launch vehicle is identified to the baseline Saturn V J-2S/LOR SA 511 configuration. The electrical interface is shown in Figure 10.5.4.3-1.

10.5.4.4 J-2S/Polar Astrionic Subsystems

10.5.4.4.1 Navigation and Guidance

The hardware for the J-2S/Polar Guidance System will require no modification. Guidance software modifications are discussed in paragraph 10.5.4.4.10, Flight Program. Flight sequence changes are given in paragraph 10.5.4.4.3, Mode and Sequencing.

10.5.4.4.2 Attitude Control

Definition of the J-2S Polar mission requires an undefined 22 ft space station between the IU and S-II stage; therefore, no discussion will be directed toward the S-IVB stage. The S-II stage function for Polar mission is to place the vehicle payload into a 100 nautical mile earth orbit utilizing the same physical configuration described in paragraph 10.5.1.4.2a J-2S/LOR mission.

Figure 10.5.4.4-1 shows a block diagram of the control system configured for two-stage Polar mission. The J-2S/Polar mission vehicle control system analysis is discussed further in paragraph 10.5.4.4.10, Flight Program. Saturn V engine actuator arrangements are shown in Figure 10.5.4.4-2.

a. S-II Attitude Control System

Definition of J-2S/Polar mission will require no modifications to the S-II stage in addition to those discussed in paragraph 10.5.1.4.2a, J-2S/LOR mission. The RCS and TVC system discussed in paragraph 10.5.3.4.2a, will not be necessary for J-2S/Polar mission due to the single S-II burn and the termination of the S-II stage at 100 nautical miles.

Ref. 10.4-34, Fig. 1-4

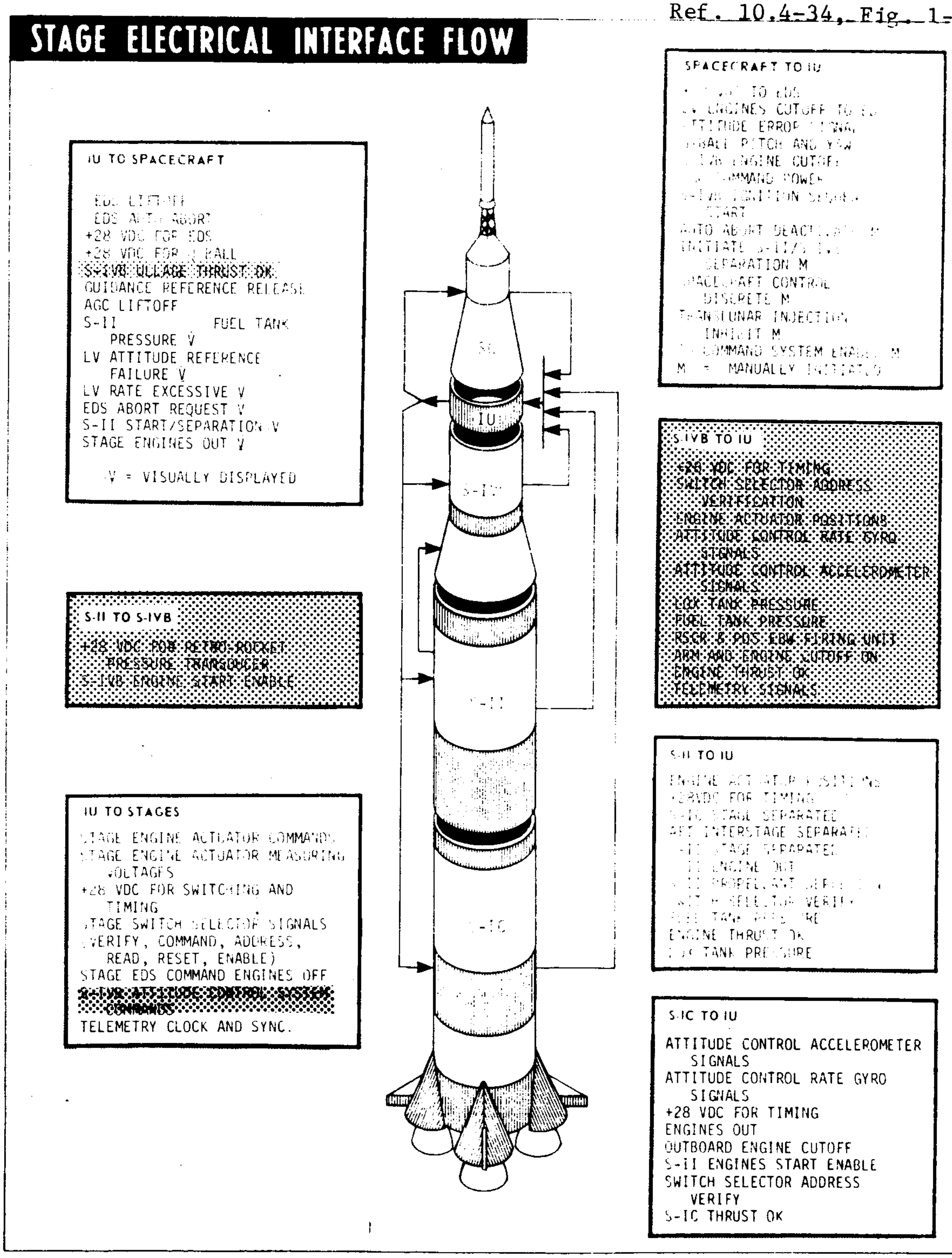
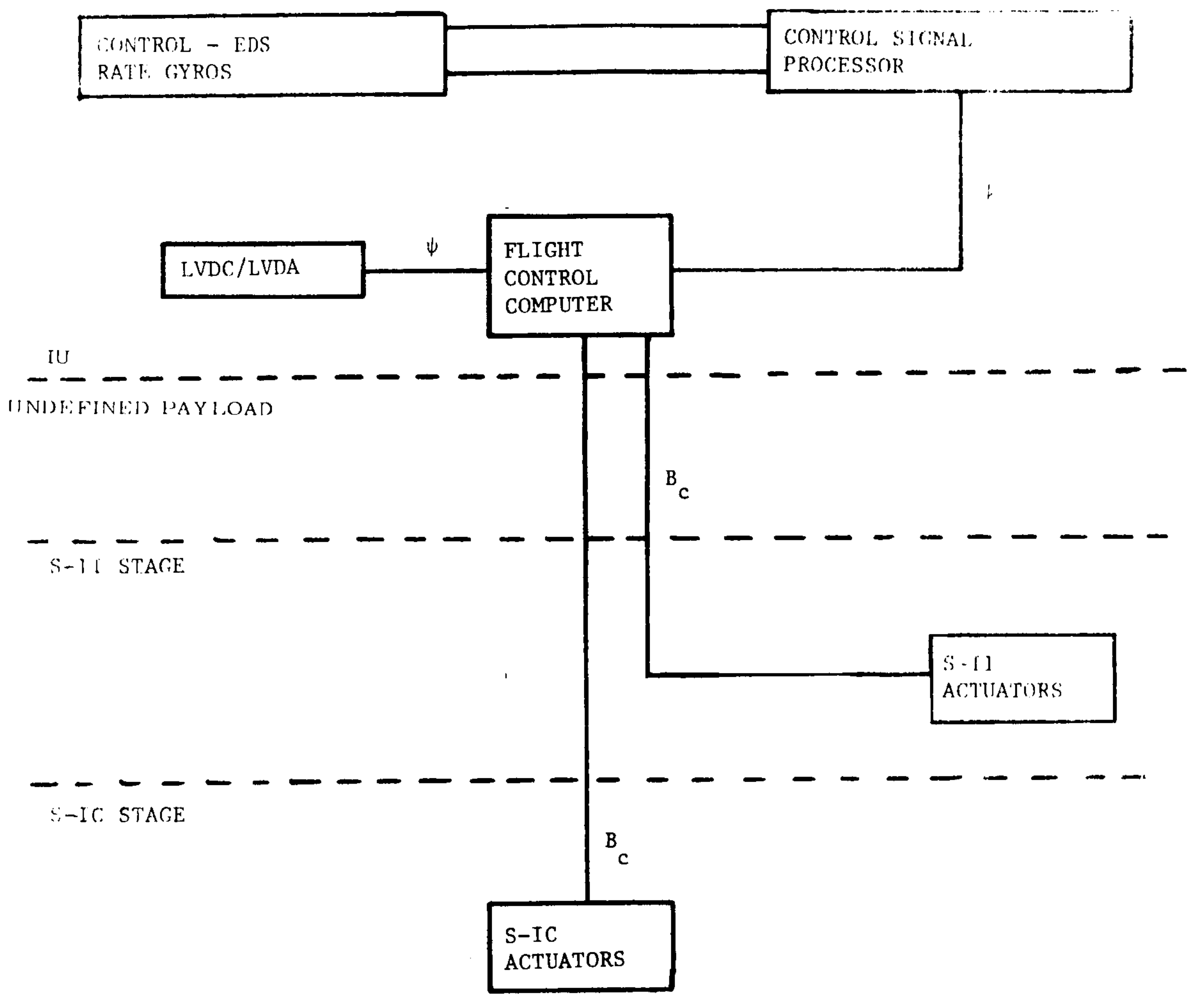


FIGURE 10.5.4.3-1. STAGE ELECTRICAL INTERFACE FLOW



Ref. 10.4-32, Fig. 7-14

FIGURE 10.5.4.4-1. POLAR VEHICLE CONTROL SYSTEM

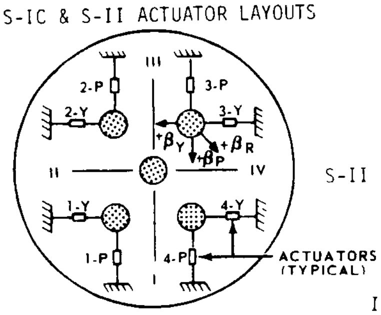
Ref. 10.4-32, Fig. 7-15

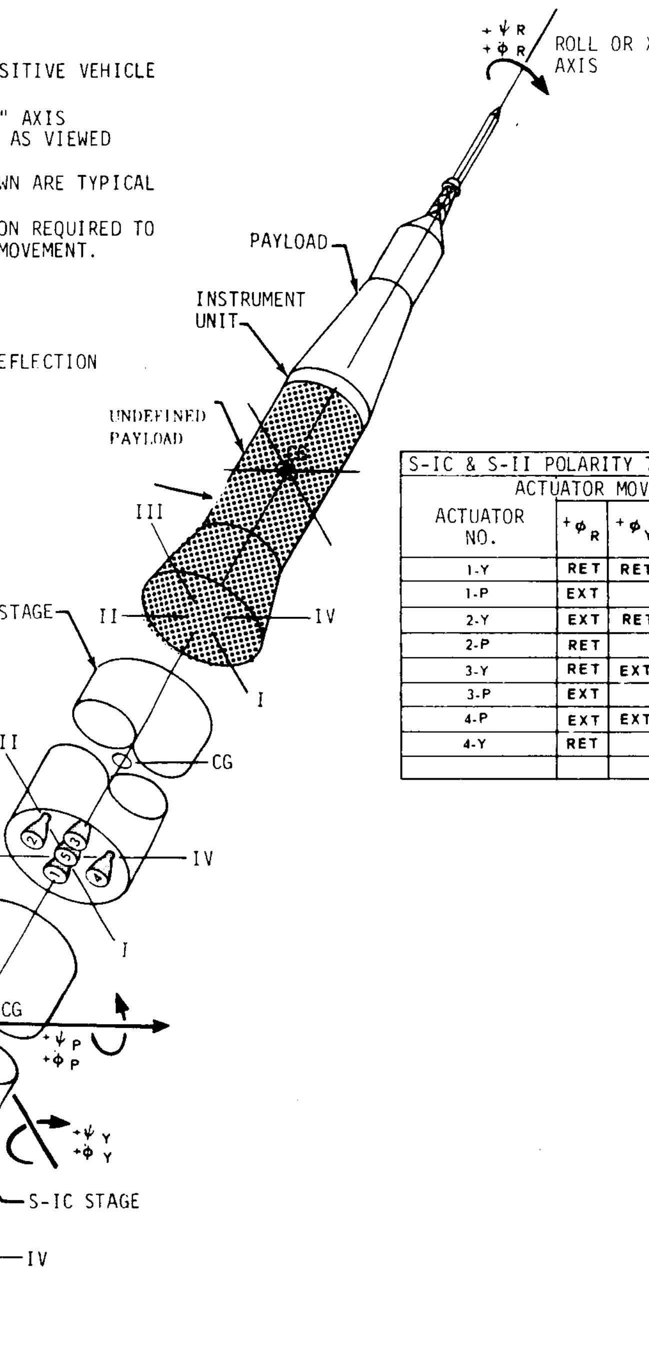
SATURN V ENGINES, ACTUATORS AND NOZZLE ARRANGEMENT

NOTES:

1. ALL SIGNAL ARROWS INDICATE POSITIVE VEHICLE MOVEMENTS.
2. VEHICLE PITCHES AROUND THE "Y" AXIS
3. ENGINE ACTUATOR LAYOUTS SHOWN AS VIEWED FROM AFT END OF VEHICLE.
4. DIRECTIONS AND POLARITIES SHOWN ARE TYPICAL FOR ALL STAGES.
5. + β INDICATES ENGINE DEFLECTION REQUIRED TO CORRECT FOR POSITIVE VEHICLE MOVEMENT.
6. CG = CENTER OF GRAVITY
   F = NOZZLES ON
   EXT = ACTUATOR EXTENDED
   RET = ACTUATOR RETRACTED
   β = THRUST VECTOR ANGULAR DEFLECTION

S-IC & S-II ACTUATOR LAYOUTS





S-IC & S-II POLARITY TABLE

ACTUATOR NO.	ACTUATOR MOVEMENT $+\phi_R$	$+\phi_Y$	$+\phi_P$
1-Y	RET	RET	
1-P	EXT		RET
2-Y	EXT	RET	
2-P	RET		EXT
3-Y	RET	EXT	
3-P	EXT		EXT
4-P	EXT	EXT	
4-Y	RET		RET

FIGURE 10.5.4.4-2. POLAR MISSION ACTUATOR AND NOZZLE ARRANGEMENT

10.5.4.4.2 (Continued)

b. IU Attitude Control

The IU Attitude Control System will require no hardware modifications due to the J-2S engines being installed on S-II stage, configured for Polar mission. However, the Polar mission Flight Control Computer will require significant modifications to compensate for deletion of S-IVB stage functions. Fifty-one of the 90 modules in the Flight Control Computer associated with S-IVB burn and orbit control will be deleted. In J-2S/LEO configuration a number of these modules were redefined for the S-II RCS system, however, Polar mission will not require the use of any S-IVB FCC functions. Reference Figure 10.5.4.4-3.

10.5.4.4.3 Mode and Sequence (Polar)

The basic Mode and Sequence system is described in paragraph 10.5.1.4.3 (a and c), J-2S/LOR Mode and Sequence. Reference Figure 10.5.4.4-4.

a. S-II Mode and Sequence

Converting from J-2S/LOR to J-2S/Polar mission requires no hardware modification to the Mode and Sequence additional to those discussed in paragraph 10.5.1.4.3a, J-2S/LOR Mode and Sequence.

b. IU Mode and Sequence

IU Mode and Sequence will require no hardware modification as a result of J-2S/Polar mission. Flight sequence changes will require software changes discussed further in paragraph 10.5.4.4.10.

10.5.4.4.4 Telemetry and Measurement

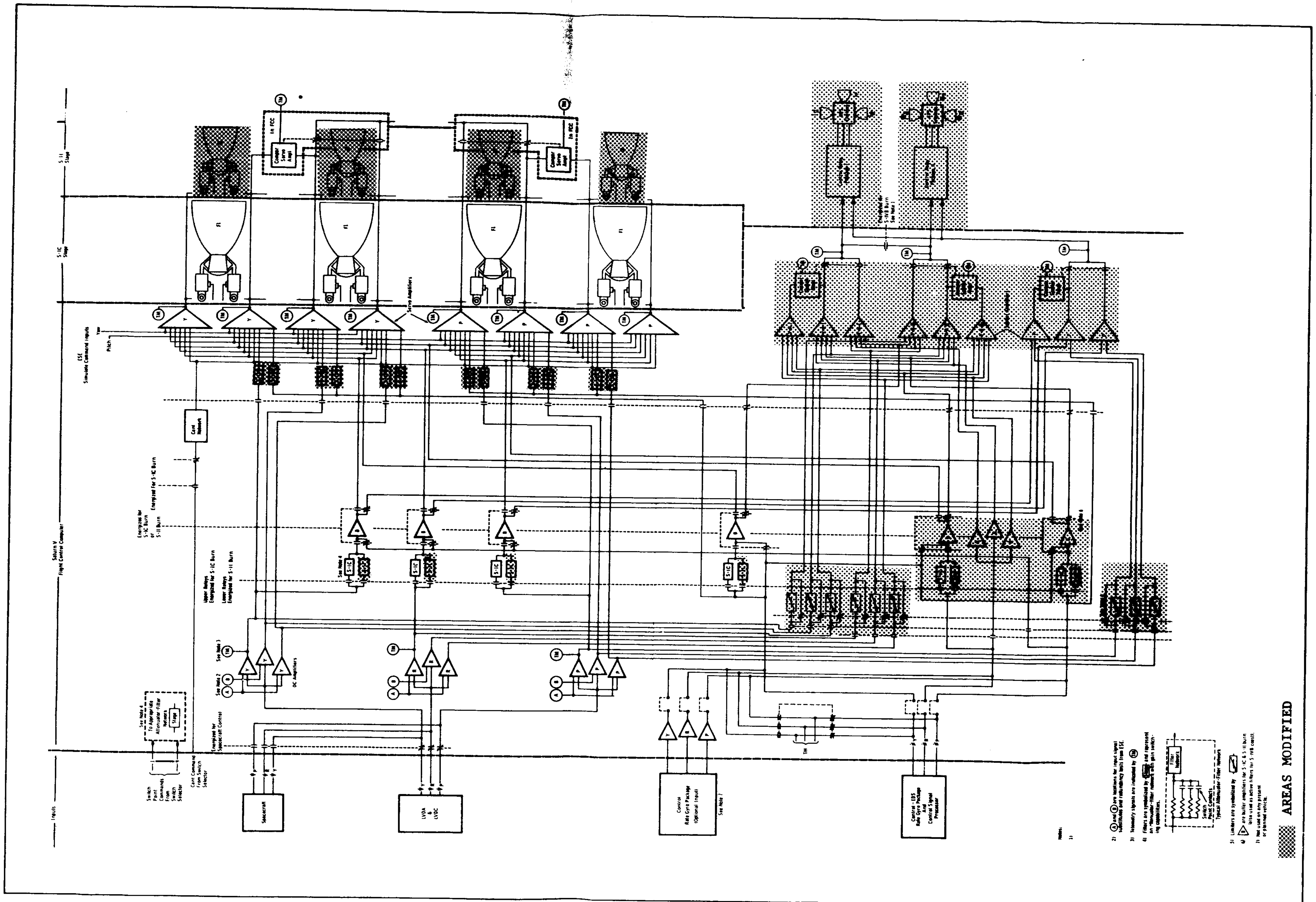
Converting from J-2S/LOR to J-2S/Polar missions will require no major modifications to the Saturn V, S-II and IU stages.

a. S-II Telemetry and Measurement

Telemetry and Measurement modifications associated with the J-2S/LOR to J-2S/Polar conversion are identical to paragraph 10.5.1.4.4a. J-2S/LOR mission.

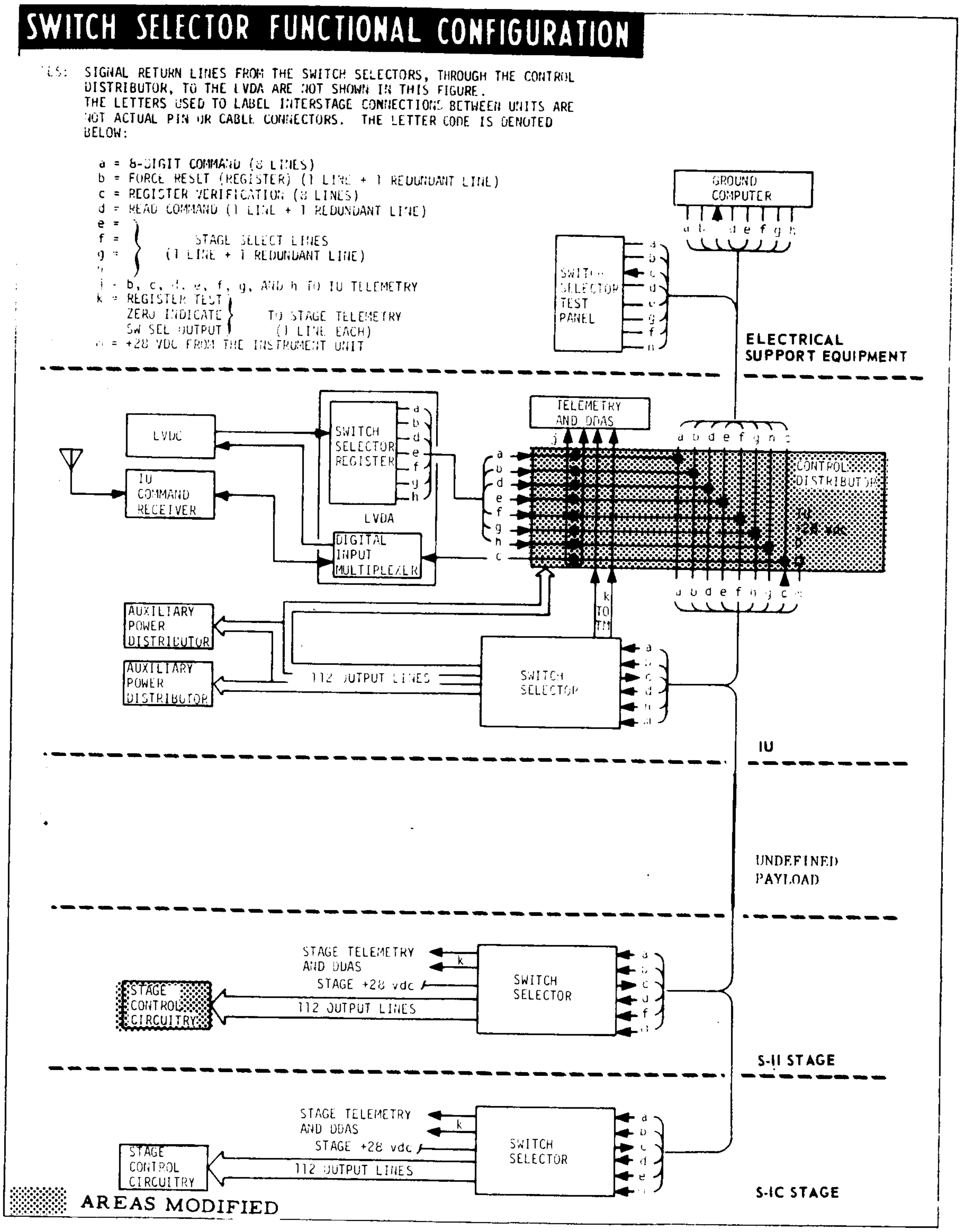
b. IU Telemetry and Measurement

The IU Telemetry and Measurement system will require no hardware modification as a result of the J-2S/LOR to J-2S/Polar conversion. Measurements associated with S-IVB functions will be deleted from IU telemetry. Without S-II RCS system the measurements need not be employed for J-2S Polar mission.



Ref. 10.4-31, Fig. 3.3-5

FIGURE 10.5.4.4-3.
POLAR VEHICLE
FLIGHT CONTROL COMPUTER



Ref 10.4-32, Fig.7-20

FIGURE 10.5.4.4-4. POLAR MISSION SWITCH SELECTOR

10.5.4.4.4 (Continued)

The undefined space station of the two-stage vehicle, however, presents a problem in redundant transmission of IU PCM/DDAS telemetry via the S-IVB telemetry system (Reference paragraph 10.5.3.4.4b) J-2S/LEO IU Telemetry and Measurement). Rather than modify the telemetry system to enable redundant flight control information transmission via the S-II stage for the short duration of boost into the 100 nautical mile orbit, the IU will use minimum modification and transmit all IU data through IU telemetry system.

10.5.4.4.5 Radio Command

No modifications will be required to the S-II stage or the IU Radio Command as a result of the J-2S/Polar mission.

10.5.4.4.6 Tracking

No modifications will be required for the S-II stage or the IU as a result of the J-2S/Polar conversion.

10.5.4.4.7 Power Supply and Distribution

a. S-II Power Supply and Distribution

Converting from J-2S/LOR to J-2S/Polar mission requires hardware impact on the S-II Power Supply and Distribution System. Modifications are identical to those described in paragraph 10.5.1.4.7a.

b. IU Power Supply and Distribution

No additional hardware requirements will be required in the IU Power Supply and Distribution system as a result of the J-2S/LOR to J-2S/Polar conversion.

10.5.4.4.8 Emergency Detection System

No modification will be required to the S-II stage or IU Emergency Detection System as a result of the J-2S/Polar mission.

10.5.4.4.9 Separation

The Saturn vehicle Separation system is described in paragraph 10.5.1.4.9 of the J-2S/LOR Astrionic System Study. No additional hardware modifications will be required to the S-II stage or the IU as a result of the J-2S/Polar mission.

10.5.4.4.10 Flight Program

a. Guidance Analysis

One area of the flight program requires modification for a two-stage Polar mission. The large commanded yaw attitude approaches the yaw limit of $\pm 60^{\circ}$ of the three-gimbal platform. To prevent the platform from tumbling, a software limit $\pm 45^{\circ}$ is in the present flight programs. The study effort was concentrated on testing the adequacy of a platform alignment technique for meeting the $\pm 45^{\circ}$ yaw requirements. The yaw biasing technique does not affect the trajectory of the vehicle, but merely changes the platform measurement of the trajectory. To implement this method, it is desirable to incorporate a routine in the flight program to convert "standard" gimbal angles to "offset" gimbal angles for use during pre-IGM. During IGM the use of the proper matrix relating IGM coordinates to platform coordinates automatically includes the offset. Reference Table 10.5.4.4-I for Polar mission flight sequence.

b. Control Analysis

The J-2S/Polar Flight Control System Analysis is the same as is discussed in paragraph 10.5.3.4.10 b.

10.5.4.4.11 Instrument Unit

Converting from J-2S/LOR to J-2S/Polar mission will require no IU components be modified additional to those shown in paragraph 10.5.1.4.11, Figure 10.5.1.4-15.

10.5.4.4.12 Environmental Control System

a. S-II Environmental Control

Converting from J-2S/LOR to J-2S/Polar will require no additional hardware modifications to the S-II ECS.

b. IU Environmental Control

Converting from J-2S/LOR to J-2S/Polar will require no additional hardware modification to the IU ECS.

10.5.4.4.13 Electrical Support Equipment

a. S-II Electrical Support Equipment

Modification to the S-II Electrical Support Equipment are identical to those described in paragraph 10.5.1.4.13a.

TABLE 10.5.4.4-I. APPROXIMATE MISSION TIME LINE POLAR ORBIT

Time Base	Time (Seconds)	Event
TB0	-17	Guidance Reference Release
TB1	0	Liftoff
	79	Q Max
TB2	149	Center Engine Cutoff
TB3	161	S-IC Cutoff
	165	S-II Ignition (idle)
	166	S-II Mainstage
	168	MR Shift to 5.5
	193	Interstage Separation
	198	LES Separation
	396	MR Shift to 4.7
TB4*	497	S-II Cutoff

*Orbital phase past time base 4 has not been defined.

10.5.4.4.13 (Continued)

b. IU Electrical Support Equipment

Modification to the IU Electrical Support Equipment are identical to those described in paragraph 10.5.1.4.13 c.

10.5.4.4.14 Propellant Management

a. S-II Propellant Management

The J-2S/Polar S-II Propellant Management system will function the same as a J-2S/LOR system therefore will require no hardware modifications.

b. IU Propellant Management

Converting from J-2S/LOR to J-2S/Polar will require no hardware modifications to the IU to accommodate the Saturn vehicle Propellant Utilization system.

10.6 VEHICLE FINAL WEIGHTS

Final vehicle weights reflecting the systems and structural modifications which result from J-2S engine installation and mission environmental requirements are given in this section.

Fuel residual summaries, stage dry weights, and interstage weights for the four design vehicles are shown in Tables 10.6-I through 10.6-IX. These data were compiled from information supplied by the associate stage contractors. The Baseline SA-511 weights are shown for comparison.

For the LOR, LEO and synchronous orbit missions, the S-IC weights remain the same as the Baseline SA-511 weights. Two-stage polar mission loads exceeded the structural capability of the S-IC fuel tank side wall. Fuel tank weight for this vehicle reflects the 74 pounds of additional structure required to increase structural capability.

Tables 10.6-X through 10.6-XIII show the final drop weights for each of the design vehicles.

TABLE 10.6-I

RESIDUAL SUMMARY - S-IC STAGE

DESCRIPTION	WEIGHT (LBS.)				
	SA-511 BASELINE	LOR MISSION	SYNCH. MISSION	LEO MISSION	POLAR MISSION
DRY WEIGHT	289,409	289,409	289,409	289,409	289,483
LOX IN TANK	1,861	1,861	1,861	1,861	1,861
LOX BELOW TANK	25,737	25,737	25,737	25,737	25,737
LOX PRESSURIZATION GAS	5,451	5,451	5,451	5,451	5,451
FUEL IN TANK	17,119	17,119	17,119	17,119	17,119
FUEL BELOW TANK	13,367	13,367	13,367	13,367	13,367
FUEL PRESSURIZATION GAS	512	512	512	512	512
HELIUM IN BOTTLE	188	188	188	188	188
SERVICE ITEMS	3,350	3,350	3,350	3,350	3,350
TOTAL AT SEPARATION	356,994	356,994	356,994	356,994	357,068

TABLE 10.6-II RESIDUAL SUMMARY - S-II STAGE

DESCRIPTION	WEIGHT (LBS)				
	SA-511 BASELINE	LOR MISSION	SYNCH. MISSION	LEO MISSION	POLAR MISSION
DRY WEIGHT	80,686	82,548	82,548	84,649	82,548
LOX IN TANK	4,210	2,545	2,545	1,000	2,545
LOX BELOW TANK	1,764	1,353	1,353	1,353	1,353
LOX PRESSURIZATION GAS	3,294	3,992	3,992	3,992	3,992
FUEL IN TANK	3,149	2,940	2,940	2,000	2,940
FUEL BELOW TANK	282	241	241	241	241
FUEL PRESSURIZATION GAS	1,330	1,594	1,594	1,594	1,594
SERVICE ITEMS	105	70	70	70	70
TOTAL AT SEPARATION	94,820	95,283	95,283	94,899	95,283

TABLE 10.6-III RESIDUAL SUMMARY - S-IVB STAGE

DESCRIPTION	WEIGHT (LBS)		
	SA-511 BASELINE	LOR MISSION	SYNCH. MISSION
DRY WEIGHT	25,084	24,950	25,347
LOX IN TANK	2,996 *	29	29
LOX BELOW TANK	369	367	367
LOX PRESSURIZATION GAS	347	377	570
FUEL IN TANK	1,240 *	685	685
FUEL BELOW TANK	45	48	48
FUEL PRESSURIZATION GAS	553	497	614
HELIUM IN BOTTLE	189	197	120
APS PROPELLANT	123	589	199
AFT FRAME AND DETONATION PACKAGE	-51	-51	-51
ULLAGE ROCKET CASES	-130	0	0
SERVICE ITEMS	62	57	59
TOTAL AT SEPARATION	30,827 *	27,745	27,987

* INCLUDES FLIGHT PERFORMANCE AND GEOMETRY RESERVES TOTALING 3,471 LBS.

TABLE 10.6-IV S-IC STAGE DRY WEIGHT (LBS)

DESCRIPTION	SA-511 BASELINE	LOR MISSION	SYNCH. MISSION	LEO MISSION	POLA R MISSION
Stage Structure	(140,835)	(140,835)	(140,835)	(140,835)	(140,909)
Structural Fuel Container	22,334	22,334	22,334	22,334	22,408
Structural Oxidizer Container	35,404	35,404	35,404	35,404	35,404
Structure Forward of Tanks	5,638	5,638	5,638	5,638	5,638
Structure Between Tanks	13,173	13,173	13,173	13,173	13,173
Thrust Structure	47,503	47,503	47,503	47,503	47,503
Fairings & Associated Structure	9,024	9,024	9,024	9,024	9,024
Non-movable Aerodynamic Control Surfaces	2,034	2,034	2,034	2,034	2,034
Base Heat Protection	5,259	5,259	5,259	5,259	5,259
Paint and Sealer	466	466	466	466	466
Propulsion System and Accessories	(139,525)	(139,525)	(139,525)	(139,525)	(139,525)
Liquid Rocket Engine and Accessories	93,423	94,423	93,423	93,423	93,423
Fuel System	13,508	13,508	13,508	13,508	13,508
Oxidizer System	23,872	23,872	23,872	23,872	23,872
Stage Control System	8,722	8,722	8,722	8,722	8,722
Equipment and Instrumentation	(8,618)	(8,618)	(8,618)	(8,618)	(8,618)
Structure	225	225	225	225	225
Environmental Control System-Equipment	304	304	304	304	304
Guidance System	63	63	63	63	63
Control System Electronics	67	67	67	67	67
Telemetering and Measuring Equipment	3,608	36,608	3,608	3,608	3,608
Electrical System	907	907	907	907	907
Range Safety Equipment	505	505	505	505	505
Separation System	2,492	2,492	2,492	2,492	2,492
Pneumatic System	447	447	447	447	447
Design Uncertainties (growth)	(431)	(431)	(431)	(431)	(431)
TOTAL DRY WEIGHT	289,409	289,409	289,409	289,409	289,483

TABLE 10.6-V S-IC/S-II INTERSTAGE WEIGHT (LBS)

DESCRIPTION	SA-511 BASELINE	LOR MISSION	SYNCH. MISSION	LEO MISSION	POLAR MISSION
Stage Structure	(1,481)	(1,481)	(1,481)	(1,481)	(1,481)
Fairings and Associated Structure	55	55	55	55	55
Interstage Structure	1,416	1,416	1,416	1,416	1,416
Paint and Sealer	10	10	10	10	10
Equipment and Instrumentation	(67)	(67)	(67)	(67)	(67)
Separation System	57	57	57	57	57
Systems for Total Vehicle	10	10	10	10	10
Short Interstage at Ground Lift-Off	1,548	1,548	1,548	1,548	1,548
Stage Structure	(8,170)	(8,127)	(8,127)	(8,127)	(8,127)
Fairings and Associated Structure	152	149	149	149	149
Interstage Structure	7,878	7,838	7,838	7,838	7,838
Paint and Sealer	140	140	140	140	140
Propulsion System and Accessories	(159)	(7)	(7)	(7)	(7)
Fuel System	159	7	7	7	7
Equipment and Instrumentation	(1,106)	(202)	(202)	(202)	(202)
Telemetry and Measuring Equipment	130	60	60	60	60
Separation System	85	85	85	85	85
Ullage System	834	-	-	-	-
Systems for Total Vehicle	57	57	57	57	57
Long Interstage at Ground Lift-Off	9,435	8,336	8,336	8,336	8,336

TABLE 10.6-VI S-II STAGE DRY WEIGHT (LBS)

DESCRIPTION	SA-511 BASELINE	LOR MISSION	SYNCH. MISSION	LEO MISSION	POLAR MISSION
Stage Structure	(47,064)	(49,013)	(49,013)	(49,203)	(49,013)
Structural Propellant Container	29,179	29,179	29,179	29,329	29,179
Structure Forward of Tanks	4,057	4,057	4,057	4,057	4,057
Structure Aft of Tanks	1,130	4,130	4,130	4,130	4,130
Thrust Structure	7,302	9,302	9,302	9,302	9,302
Fairings and Associated Structure	1,432	1,351	1,351	1,391	1,351
Base Heat Protection	630	660	660	660	660
Paint and Sealer	334	334	334	334	334
Propulsion System and Accessories	(26,439)	(26,472)	(26,472)	(27,943)	(26,472)
Liquid Rocket Engine and Accessories	18,030	19,248	19,248	19,248	19,248
Purge System for Upper Stage Childown	386	416	416	416	416
Fuel System	4,076	3,166	3,166	3,191	3,166
Oxidizer System	2,824	2,519	2,519	2,519	2,519
Stage Control System Hardware	1,123	1,123	1,123	2,569	1,123
Equipment and Instrumentation	(6,765)	(6,645)	6,645)	(7,085)	(6,645)
Environmental Control System Equipment	1,121	1,121	1,121	1,331	1,121
Telemetering and Measuring Equipment	2,734	2,734	2,734	2,934	2,734
Propellant Utilization System	634	634	634	634	634
Electrical System	949	829	829	859	829
Range Safety Equipment	319	319	319	319	319
Pneumatic System	410	410	410	410	410
Separation System	117	117	117	117	117
Systems for Total Vehicle	481	481	481	481	481
Design Uncertainties (Growth)	(418)	(418)	(418)	(418)	(418)
TOTAL DRY WEIGHT	80,686	82,548	82,548	84,649	82,548

TABLE 10.6-VII S-II/S-IVB INTERSTAGE WEIGHT (LBS)

DESCRIPTION	SA-511 BASELINE	LOR MISSION	SYNCH. MISSION
Stage Structure	(6,249)	(6,249)	(6,249)
Interstage Structure	5,678	5,678	5,678
Paint and Sealer	49	49	49
Heat and Flame Protection	522	522	522
Equipment and Instrumentation	(772)	(772)	(772)
Environmental Control System-Equipment	17	17	17
Telemetering and Measuring Equipment	15	15	15
Range Safety Equipment	3	3	3
Separation System	727	727	727
System for Total Vehicle	10	10	10
TOTAL DRY WEIGHT OF INTERSTAGE	7,021	7,021	7,021

TABLE 10.6-VIII S-IVB STAGE DRY WEIGHT (LBS)

DESCRIPTION	SA-511 BASELINE	LOR MISSION	SYNCH. MISSION
Stage Structure	(13,241)	(13,235)	(13,544)
Structural Propellant Container	8,925	8,923	9,232
Structure Forward of Tanks	1,242	1,242	1,242
Structure Aft of Tanks	1,816	1,801	1,801
Thrust Structure	774	809	809
Fairings and Associated Structure	196	174	174
Paint and Sealer	104	104	104
Ablative Insulation	184	182	182
Propulsion System and Accessories	(7,263)	(7,539)	(6,845)
Liquid Rocket Engine and Accessories	3,562	4,012	4,073
Purge System for Upper Stage Chilldown	276	0	0
Fuel System	1,567	1,616	1,067
Oxidizer System	1,261	1,317	1,111
Stage Control System Hardware	284	284	284
Cryogenic Repressurization System	313	310	310
Equipment and Instrumentation	(4,580)	(4,176)	(4,958)
Structure	428	427	431
Environmental Control System - Equipment	232	231	268
Control System Electronics	113	116	116
Telemetering and Measuring Equipment	1,156	1,126	1,533
Propellant Utilization System	175	175	175
Electrical System	830	726	1,060
Range Safety Equipment	69	69	69
Pneumatic System	302	269	269
Auxiliary Propulsion System	854	829	829
Separation System	119	117	117
Systems for Total Vehicle	91	91	91
Ullage System	211	0	0
TOTAL DRY WEIGHT	25,084	24,950	25,347

TABLE 10.6-IX INSTRUMENT UNIT DRY WEIGHT (LBS)

DESCRIPTION	SA-511 BASELINE	LOR MISSION	SYNCH. MISSION	LEO MISSION	POLAR MISSION
Stage Structure	(621)	(705)	(710)	(705)	(705)
Interstage Structure	613	682	687	682	682
Paint and Sealer	8	23	23	23	23
Equipment and Instrumentation	(3,266)	(3,299)	(3,623)	(3,299)	(3,299)
Structure	29	28	29	28	28
Environmental Control System	748	762	738	762	762
Guidance System	624	636	700	636	636
Control System Electronics	175	170	172	170	170
Tracking, Navigation and Observation Equipment	22	22	22	22	22
Telemetering and Measuring Equipment	294	294	293	294	294
Electrical System	1,307	1,316	1,522	1,316	1,316
Command System Equipment	67	69	147	69	69
Service Items	(296)	(299)	(342)	(299)	(299)
Pressurant Gases and Thermal Control Fluids	296	299	342	299	299
TOTAL DRY WEIGHT	4,183	4,301	4,675	4,301	4,301

TABLE 10.6-X J-2S VEHICLE DROP WEIGHTS - LOR MISSION

ITEM		WEIGHT (LBS)
TOTAL WEIGHT DROP AT S-IC STAGING		(370,266)
S-IC (Dry)		289,409
S-IC Residuals		67,585
S-IC Thrust Decay Propellant		9,069
Inboard Engine Thrust Decay Propellant	1,915	
Inboard Engine Expended Propellant	408	
Outboard Engine Thrust Decay Propellant	6,746	
S-IC/S-II Interstage (Small)		1,548
S-II Start Cartridge Propellant		67
S-II Thrust Build-up		2,550
S-II Bulkhead Purge Gas		38
TOTAL WEIGHT DROP AT S-IC/S-II INTERSTAGE DROP		(8,336)
TOTAL WEIGHT DROP AT LES JETTISON		(8,936)
TOTAL WEIGHT DROP AT S-II STAGING		(104,610)
S-II (Dry)		82,548
S-II Residuals		12,735
S-II Thrust Decay		655
S-II/S-IVB Interstage		7,021
S-II/S-IVB Interstage (Retro. Prop.)		1,062
S-IVB Separation Package		51
S-IVB Idle Mode Propellant		18
S-IVB Turbine Spin Propellant		10
S-IVB Thrust Build-Up		510
TOTAL S-IVB WEIGHT LOST IN ORBIT		(4,604)
S-IVB Thrust Decay		131
S-IVB Idle Mode		28
S-IVB Propellant Vented (3 orbits)		3,893
S-IVB APS* Propellant		30
S-IVB Thrust Build-Up		510
S-IVB Turbine Spin Propellant		10
S-IVB APS* Propellant		2

* Auxiliary Propulsion System

TABLE 10.6-X J-2S VEHICLE DROP WEIGHTS - LOR MISSION (Continued)

ITEM	WEIGHT (LBS)
TOTAL WEIGHT DROP AT S-IVB SECOND CUT-OFF	(27,955)
S-IVB (Dry)	24,950
S-IVB Residuals	2,846
S-IVB Thrust Decay	131
S-IVB Idle Mode Propellant	28
TOTAL INSTRUMENT UNIT DROP WEIGHT	(4,301)

TABLE 10.6-XI J-2S VEHICLE DROP WEIGHTS - SYNCHRONOUS MISSION

ITEM		WEIGHT (LBS)
TOTAL WEIGHT DROP AT S-IC STAGING		(370,266)
S-IC (Dry)		289,409
S-IC Residuals		67,585
S-IC Thrust Decay Propellant		9,069
Inboard Engine Thrust Decay Propellant	1,915	
Inboard Engine Expended Propellant	408	
Outboard Engine Thrust Decay Propellant	6,746	
S-IC/S-II Interstage (Small)		1,548
S-II Start Cartridge Propellant		67
S-II Thrust Build-up		2,550
S-II Bulkhead Purge Gas		38
TOTAL WEIGHT DROP AT S-IC/S-II INTERSTAGE DROP		(8,336)
TOTAL WEIGHT DROP AT LES JETTISON		(8,936)
TOTAL WEIGHT DROP AT S-II STAGING		(104,610)
S-II (Dry)		82,548
S-II Residuals		12,735
S-II Thrust Decay		655
S-II/S IVB Interstage		7,021
S-II/S-IVB Interstage (Retro. Prop.)		1,062
S-IVB Separation Package		51
S-IVB Idle Mode Propellant		18
S-IVB Turbine Spin Propellant		10
S-IVB Thrust Build-up		510
TOTAL S-IVB WEIGHT LOSS IN PARKING ORBIT		(7,760)
S-IVB Thrust Decay		131
S-IVB Idle Mode		28
S-IVB Propellant Vented (5 orbits)		7,006
S-IVB APS* Propellant		75
S-IVB Thrust Build-up		510
S-IVB Turbine Spin Propellant		9
S-IVB APS* Propellant		1

* Auxiliary Propulsion System

TABLE 10.6-XI J-2S VEHICLE DROP WEIGHTS - SYNCHRONOUS MISSION (Continued)

ITEM	WEIGHT (LBS)
TOTAL S-IVB WEIGHT LOSS IN TRANSFER ORBIT	(2,847)
S-IVB Thrust Decay	131
S-IVB Idle Mode	28
S-IVB Propellant Vented	2,100
S-IVB APS* Propellant	68
S-IVB Thrust Build-up	510
S-IVB Turbine Spin Propellant	9
S-IVB APS* Propellant	1
TOTAL WEIGHT DROP AT S-IVB THIRD CUT-OFF	(28,197)
S-IVB (Dry)	25,347
S-IVB Residuals	2,691
S-IVB Thrust Decay	131
S-IVB Idle Mode Propellant	28
TOTAL INSTRUMENT UNIT DROP WEIGHT	(4,675)

* Auxiliary Propulsion System

TABLE 10.6-XII J-2S VEHICLE DROP WEIGHTS - LEO MISSION

ITEM		WEIGHT (LBS)
TOTAL WEIGHT DROP AT S-IC STAGING		(370, 266)
S-IC (Dry)		289, 409
S-IC Residuals		67, 585
S-IC Thrust Decay Propellant		9, 069
Inboard Engine Thrust Decay Propellant	1, 915	
Inboard Engine Expended Propellant	408	
Outboard Engine Thrust Decay Propellant	6, 746	
S-IC/S-II Interstage (Small)		1, 548
S-II Start Cartridge Propellant		67
S-II Thrust Build-up		2, 550
S-II Bulkhead Purge Gas		38
TOTAL WEIGHT DROP AT S-IC/S-II INTERSTAGE DROP		(8, 336)
TOTAL WEIGHT DROP AT LES JETTISON		(8, 936)
TOTAL WEIGHT DROP DURING S-II COAST TRANSFER ORBIT		(1, 429)
S-II Thrust Decay		319
S-II Idle Mode		446
S-II Reaction Control Propellant		664
TOTAL WEIGHT DROP AT S-II SECOND IDLE MODE CUT-OFF		(94, 899)
S-II (Dry)		84, 649
S-II Residuals		10, 250
TOTAL INSTRUMENT UNIT DROP WEIGHT		(4, 301)

TABLE 10.6-XIII J-2S VEHICLE DROP WEIGHTS
TWO-STAGE POLAR MISSION

ITEM		WEIGHT (LBS)
TOTAL WEIGHT DROP AT S-IC STAGING		(370, 266)
S-IC (Dry)		289.409
S-IC Residuals		67, 585
S-IC Thrust Decay Propellant		9, 069
Inboard Engine Thrust Decay Propellant	1,915	
Inboard Engine Expended Propellant	408	
Outboard Engine Thrust Decay Propellant	6,746	
S-IC/S-II Interstage (Small)		1,548
S-II Start Cartridge Propellant		67
S-II Thrust Build-up		2, 550
S-II Bulkhead Purge Gas		38
TOTAL WEIGHT DROP AT S-IC/S-II INTERSTAGE DROP		(8,336)
TOTAL WEIGHT DROP AT LES JETTISON		(8,936)
TOTAL WEIGHT DROP AT S-II STAGING		(104,610)
S-II (Dry)		82,548
S-II Residuals		12,735
S-II Thrust Decay		655
S-II/S-IVB Interstage		7,021
S-II/S-IVB Interstage (Retro. Prop.)		1,062
S-IVB Separation Package		51
S-IVB Idle Mode Propellant		18
S-IVB Turbine Spin Propellant		10
S-IVB Thrust Build-up		510
TOTAL INSTRUMENT UNIT DROP WEIGHT		(4,301)

10.7 KSC LAUNCH OPERATIONS IMPACT

This study assesses the impact on KSC Saturn V launch operation of replacing the J-2 equipped S-II and S-IVB stages with J-2S equipped stages. This assessment includes:

a. Identifying the changes to KSC launch operations resulting from the change to the J-2S engine and associated mandatory stage changes.

b. Identifying the changes to KSC ground equipment and facilities required by the new engine and associated stage changes.

c. Providing a conceptual design for all new and modified equipment.

d. Providing an implementation schedule to reflect the time required to design, modify and activate the launch facility for the J-2S vehicles.

The study approach used the SA-503 configuration and processing flow as a baseline. Processing requirements for vehicles using the J-2S modified stages were identified and compared to SA-503 to establish a delta change in vehicle processing, launch rules, and interlock requirements.

Throughout this section, references to mission relate to the following stage combinations:

a. Lunar Orbit Rendezvous (LOR) Mission - This mission requires a vehicle configuration composed of an S-IVB with one restart and an S-II with no restarts. The vehicle changes will be limited to the addition of J-2S engines only.

b. Low Earth Orbit (LEO) Mission - This mission requires a vehicle configuration composed of an S-II with one restart and no S-IVB. This vehicle change will be a result of both J-2S addition and mission peculiar requirements.

c. Synchronous Orbit Mission - This mission requires a vehicle configuration composed of an S-IVB with two restarts and an S-II with no restarts. This configuration will be a result of the addition of J-2S engines and mission peculiar requirements.

10.7.1 Launch Operations Changes

10.7.1.1 Propulsion Related Processing Changes

The SA-503 vehicle was the baseline processing flow against which changes were measured. Each function, starting with Low Bay preparation to receive a stage through to vehicle launch and post launch refurbishment, was analyzed to determine whether it would be affected by the J-2S configured stage or vehicle. The results of this analysis are summarized and presented in Table 10.7-I.

There have been 133 propulsion related processing functions identified. For the LOR and Synchronous missions, 98 functions would be changed, 2 functions related to recirculation/chilldown would be deleted, and the balance would be unaffected. For the LEO mission, 60 functions would be changed, 55 functions related to the S-IVB stage would not apply, 1 function related to recirculation would be deleted, 8 functions would be added, and the remainder would be unaffected. Of the 8 new functions, 6 would be related to the S-II RCS, 1 to the IU/S-II mating and 1 to the IU/Space Station mating.

A review of procedures for accomplishing the processing functions indicated that the average procedure change resulted in an approximate 7 percent reduction in manhours. Since many LOR and Synchronous mission processing functions were not affected, the net impact on processing manhours for these vehicles is very small. For the LEO mission, while the S-II processing manhours were slightly increased due to the addition of new functions, the elimination of the S-IVB stage resulted in a large decrease of processing time.

Vehicle processing changes that are unique to the vehicle configurations are discussed in this section. Section 10.7.1.2 defines the impact of the J-2S engine on the Launch Mission Rules. This impact is primarily one of changing S-II and S-IVB stage redlines. Section 10.7.1.3 covers changes to the interlock system resulting from stage changes related to the J-2S engine.

a. Three Stage LOR Vehicle (Baseline)

The processing changes related to this vehicle will be as follows:

1. Installation and checkout of ullage rockets will be deleted for both the S-II and S-IVB stages.

2. Test and checkout of recirculation and engine chilldown will be deleted for both stages.

3. Checkout of the S-IVB APS ullage engine will be deleted.

10.7.1.1 (Continued)

4. Test and checkout of the engine startup provision and checkout of engine startup sequence will be modified for both stages. Start tank and gas generator functions will be deleted and Solid Propellant Turbine Starters (SPTS) and LOX dome purge functions will be added.

5. Test and checkout of the S-IVB prevalves and LOX depletion sensors will be deleted. In the case of the S-II stage, the processing related to prevalves and LOX depletion sensors will be modified to provide cutoff for all engines.

6. Installation and checkout of flight batteries will be modified to reflect deletion of two of the S-II batteries and the replacement of one of the S-IVB batteries with a smaller battery.

7. Test and checkout of the engine idle mode will be added and test and checkout of the hydraulic actuation will be modified for the S-IVB.

b. Three Stage Synchronous Orbit Vehicle

The additional processing changes will be as follows:

1. Test, checkout, and ordnance installation for the SPTS will be added to the S-IVB stage.

2. Telemetry test and checkout will be modified to account for measurements from the S-IVB and IU. The change to the S-IVB will include adding signal conditioning racks, multi-plexers and a new tape recorder to handle the additional measurements and adding a power amplifier to transmit them the additional distance. The IU has a modification to its power amplifiers.

3. Installation and checkout for increased battery capacity will be added to the S-IVB and IU. For the S-IVB, two electrical isolation switches will be added to prevent the two batteries from bucking each other during ground operations.

c. Two Stage LEO Vehicle

The processing changes related to this vehicle will be as follows:

1. All S-IVB processing will be deleted.

2. All processing changes for the S-II identified in Section 10.7.1.1, Paragraph a for the LOR vehicle will apply to the LEO mission.

3. Installation, test, checkout, servicing, and gas removal for an RCS system will be added to the S-II stage. The RCS system will be identical to the LOR APS system, excluding ullage motors, used on the S-IVB stage.

10.7.1.1 (Continued)

4. Test and checkout of a balanced propulsion venting of the LH2 tanks will be added to the S-II stage. This system will provide positive oxidizer and fuel settling for the S-II stage.

5. Test and checkout for an auxiliary hydraulic system including installation of hydraulic system batteries and servicing of hydraulic system air cooling supply will be added to the S-II stage.

10.7.1.2 Launch Rules

Launch Mission Rules are developed to provide guidance to the Launch Director and launch team organization by specifying preplanned decisions which are designed to minimize real time rationalization required when non-nominal situations occur during the launch countdown and applicable prelaunch tests. The SA-503 Launch Mission Rules were reviewed to determine if the replacement of J-2 engines by J-2S engines would change the Launch Mission Rules. Only mandatory redlines were considered in the review, and it was assumed that all SA-503 mandatory redlines would be available for stages equipped with J-2S engines except for systems which were deleted on J-2S engines. A detailed listing of the redline differences is included in Table 10.7-II. The following is a summary of redline changes.

a. Delete 15 of 42 mandatory redline measurements and add 4 new mandatory redline measurements for S-II with no restarts.

b. Delete 15 of 42 mandatory redline measurements and add 11 mandatory redline measurements for S-II with one restart.

c. Delete 5 of 32 mandatory redlines and add 1 mandatory redline for the S-IVB.

It appears that the probability of meeting preplanned launch windows will be increased due to the reduced number of redlines; however, since launch mission rules must be prepared for each missile to be fired and since the impact of J-2S engines on the launch mission rules is small, the impact on KSC costs or schedules is considered negligible.

10.7.1.3 Interlocks

The Saturn V Interlock System consists of relay logic circuits which react to timed commands from the Terminal Countdown Sequencer. The relay logic circuits issue the commands to the electromechanical components on each of the launch vehicle stages and ground support equipment and respond to the vehicle condition established by the command to set up an interlock condition which allows the next event in the countdown to be completed. Only launch critical functions and conditions are interlocked as prerequisites for a safe launch and the accomplishment of the primary mission objectives.

10.7.1.3 (Continued)

The present Saturn V Interlock System as defined in Boeing Document D5-16266-4, "Functional Analysis Document for Saturn V Interlock System," was reviewed to determine the impact resulting from the use of J-2S engines on the S-II and S-IVB stages.

The detailed results of this review are tabulated in Table 10.7-III. These results indicate that approximately 54 interlocks can be deleted from the system for a LOR or Synchronous mission. The interlock deletions are associated with Recirculation System, Ullage Systems, Gas Generator, Prevalves, and LOX Depletion Cutoff System. The impact of a LEO mission will result in the deletion of the interlocks associated with the use of J-2S engines on the S-II (approximately 34), the deletion of all S-IVB related interlocks (approximately 45 top level interlocks), and the addition of S-II interlocks (four top level) due to the addition of RCS Propellant Settling System and the Auxiliary Hydraulic System for thrust vector control during idle mode.

10.7.2 Facility and Equipment Modifications

10.7.2.1 General

This section includes the results of the facility and equipment requirement development and conceptual design. The major changes to KSC hardware resulting from this effort are in the GSE Pneumatic System, the Control and Monitor Panels in the LCC, and the addition of RCS servicing equipment for the S-II Stage.

10.7.2.2 Pneumatics

The specific requirements imposed on the GSE Pneumatic Systems by the replacement of J-2 engines with J-2S engines are based on the design concept that: 1) consoles which are no longer required will be removed rather than deactivated, and 2) S-II Pneumatic System consoles and associated plumbing will be modified to provide the capability to support either a LEO or LOR mission without further modification.

A summary of the Pneumatic System changes is as follows:

a. S7-41A "S-II Regulation and Distribution Console"

1. Add 32 new components plus interconnecting wiring and plumbing.
2. Modify or recalibrate 10 components.
3. Delete 17 components plus interconnecting wiring and plumbing.

10.7.2.2 (Continued)

b. S7-41B "S-II Pneumatic Control Console"

1. Add 20 new components plus interconnecting wiring and plumbing.

2. Modify or recalibrate one pressure transducer.

3. Delete 11 components plus interconnecting wiring and plumbing.

c. S7-41C "S-II Pneumatic Actuation, Purge, and Checkout Console"

1. Add 8 components plus interconnecting wiring and plumbing.

2. Delete 10 components plus interconnecting wiring and plumbing.

d. S7-41D "S-II GH_2 Servicing Console" and associated plumbing, is deleted.

e. A7-71 "S-II LH_2 Heat Exchanger"

1. Disconnect and cap two lines

f. DSV-4B-432-A-1414 "S-IVB Pneumatic Console"

1. Add one solenoid valve and associated wiring and plumbing.

g. DSV-4B-433A-1415 "S-IVB Pneumatic Console"

1. Add 25 new components and associated wiring and plumbing.

2. Delete 61 components and associated interconnecting wiring and plumbing.

h. Delete the S-IVB Heat Exchanger (DSV-4B-438-A-1416) and its associated plumbing and circuitry.

10.7.2.3 Electrical/Instrumentation

Detailed functional changes to Electrical and Instrumentation GSE were identified and collected by GSE system or item to identify the total change to each.

The GSE changes necessary to support the respective J-2S configured stages were:

a. For the S-IVB stage with one restart, one LCC ESE panel was deleted and eight were modified; three LCC and five ML patch distributors were modified, one measurement rack, two ML crossover distributors and one relay rack were modified; the 110A, DDAS, DEE and PTCS systems were modified; and S-IVB TCC panels and distributors were modified.

b. For the S-IVB stage with two restarts, the same changes as above were made with the only difference being that additional modifications were made to one LCC ESE panel to add provisions for the third SPTS.

c. For the S-II stage without restart, one LCC ESE panel was deleted and eleven were modified; three LCC and six ML patch distributors were modified; one measurement rack, two ML crossover distributors, three power distributors, a bus terminal assembly and a 5VDC module were modified; the 110A, DDAS, DEE and PTCS systems were modified; a 56VDC ground power supply was deleted and the S-II TCC panels and distributors were modified.

d. For the S-II stage with one restart, all the above S-II changes were made and in addition to these, two APS panels were added to the LCC ESE; four OIS boxes and two phones were added on MSS Platform #1 to support S-II RCS servicing; and two OTV cameras were relocated on the MSS to view hypergolic servicing of the RCS modules.

Table 7-2 of Reference 10.7-1 lists GSE systems or items that change to support processing of the J-2S configured stages. Each change is related to the particular stage and stage configuration it supports.

Conceptual design schematics were prepared for the changes that involve redesignating existing elements to new functions or that add new elements. (Figure 10.7-1 illustrates such a change.) Where the change is a simple deletion of an element (switch, patchcord, data channel) sufficient definition of the design concept change was made.

A design concept for adding RCS control and monitoring to the S-II stage ESE was prepared. While the S-II RCS provisions are identical to that of the S-IVB APS, it

10.7.2.3 (Continued)

is necessary to develop the conceptual design to a level to identify all related equipment changes. The GSE distributor has sufficient existing spare capacity to absorb the addition of patches without hardware changes. The only case where new computer channels will exceed channel deletions is in the case of the DDAS System for the S-II with one restart case. Since all channels allocated to the S-IVB stage will be available in this case, no changes to add extra DDAS capacity will be required.

In the case of the S-II stage with one restart where two new APS panels will be added in the LCC, particular attention is given to related cabling and electrical power capacities. In both cases existing cables and existing electric power service will have adequate capacity to support the new panels.

The general design concept related to the additions will be to use existing spare components and cabling wherever possible and where new components are needed will use the same type of components as already in use.

In the case of instrumentation changes, the increased number of telemetry measurements identified for all of the stage configurations are within the capacity of the existing telemetry systems. No new links nor transmission data rates were identified. Where data transmission range is increased, as in the case of the S-IVB with two restarts (synchronous mission), this results in stage changes to transmit more data a greater distance but these changes will not change ground equipment.

The total impact of the changes discussed above will be a small reduction in hardware and processing for each vehicle except for the case of the LEO mission where the elimination of the S-IVB stage will result in a major reduction in processing.

10.7.2.4 RCS Servicing System Modifications

The S-II with one restart will require RCS units (S-IVB APS less ullage engines) for attitude control of the stage during flight. The S-II RCS units will be identical to the existing S-IVB APS units (less ullage engines) and will be located on the aft end (Vehicle Station 1760) of the S-II stage. Onboard propellant storage and ullage pressurization systems will be identical to that utilized by the S-IVB APS. GSE units required for servicing the RCS units will be the same as are now provided for the S-IVB APS. The hypergolic servicing of either RCS/APS system occurs during the Hypergolic Load Test Sequence of the vehicle countdown preparations. Contractors will conduct RCS/APS loading for their respective stages.

GN_2 (750 psig) will be provided for the S-IVB APS servicing over Swing Arm No. 6 and GN_2 (500 psig) will be provided for the S-II RCS servicing to Platform No. 1 from

10.7.2.4 (Continued)

S-II Service Arm 4. GHe (3200 psig) for the onboard APS He storage and pressurization will be provided for the S-IVB APS over the umbilical on Service Arm 6. GHe (3200 psig) will be provided to the S-II onboard RCS He storage and pressurization over the S-II umbilical on Service Arm 4.

The method considered for S-II RCS loading is that of installing the S-IVB APS Loading System at the S-II level. This will require the installation of two additional isolation valve boxes on the 133-foot level of the MSS similar to APS servicing boxes on the 221-foot level and two control assemblies on Platform No. 1. This approach is shown in plan view on Figure 10.7-2. The RCS fuel and oxidizer systems are shown in Figure 10.7-3 and 10.7-4 respectively. The installation will be as follows: one fuel box for lines 23-0 (fill), 24-0 (return), and 25-0 (vent) and one oxidizer box for lines 27-0 (fill), 28-0 (return), and 29-0 (vent) (See Figure 10.7-3 and 10.7-4) on the 133-foot level of the MSS similar to APS servicing boxes on the 221-foot level. Lines (approximately 170 feet each) from the isolation valve boxes will be routed, oxidizer on one side of the MSS and fuel on the other side, along the 133-foot level to interface plates adjacent to MSS Platform No. 1 located at approximately the 135-foot level for propellant loading of the S-II RCS. The horizontal pipe chases to the RCS interface at Platform No. 1 will maintain a positive slope from the interface plates to insure proper draining of the hypergol lines following propellant transfer. This system is identical to the system that now services the S-IVB APS, and S-II RCS propellant loading will be procedurally the same. There will be no special drain or purge requirement as no additional points will be developed in the propellant delivery and return lines.

The addition of this system at the 133-foot MSS level and on Platform No. 1 will also require the following additions:

a. Hardline piping (insulated) from the isolation valve boxes along each side of the MSS to a new MMH Bulkhead (Interface Plate) on the fuel side and to a new N_2O_4 Bulkhead (Interface Plate) on the oxidizer side.

b. Pipe chases and a catwalk for the hardline piping mentioned above on each side of the MSS.

c. Flex hoses from the bulkheads to the control assemblies and from the control assemblies to the vehicle RCS/APS units.

d. Four additional OIS boxes and 2 OIS phones on Platform No. 1 to monitor and control hypergol loading.

e. Relocation of 2 OTV cameras from the S-IVB APS Loading function to the S-II Loading function for the LEO mission only.

10.7.2.4 (Continued)

f. GN_2 flex lines from Service Arm 4 to the control assemblies for purging and valve actuation. This system will also require (2) 0-50 PSIG regulators and (1) 0-500 PSIG regulator.

g. Additional safety equipment will have to be installed at the 133-foot level of the MSS. Included in this equipment will be a safety shower, eyewash and a first aid station.

h. A GN_2 purge capability for the new isolation valve boxes on the MSS. This capability already exists at the spacecraft and S-IVB MSS levels. Therefore, the change is minor and will only require the installation of valves in the GN_2 lines at the 133-foot level and pipe runs to the isolation valve boxes.

The addition of an RCS System on the S-II stage will require external vehicle access between station 1760 and 1900 at vehicle positions I and III. MSS Platform 1 positioned at station 1760.00 will satisfy this requirement. 65ICD9144 states that vehicle to MSS Platform 1 compatibility exists between station 1646 and 1920. The hinged clamshell on Platform 1 which closes in the vicinity of Service Arm 4 will have to remain partially open at this position; however, this will not limit the access to the RCS System.

10.7.2.5 Handling and Access Equipment

A7-84 Heat Shield Platform

Use of the J-2S engine will require that the present heat shield be reduced in size. Approximately three inches will be cut away from the shield in the engine nozzle area and the support struts will be relocated. These changes will require fabrication of a new heat shield platform. This platform is installed during vehicle checkout to provide a walkway for personnel to service the S-II stage. The platform consists of a one-inch thick, aluminum skin, honeycomb interior structure support by four-inch thick styrofoam blocks.

In order to conform to the reduced size of the heat shield, the platform will have to be cut to allow new struts to pass through the structure (there are four new strut points called for in the changed heat shield). In addition, a kickplate approximately one-inch high will be needed around the circumference of the platform.

SDD-259 LOX Tank Internal Access Platform Outer Stand

A slosh baffle will be installed in the aft end of the S-II LOX tank. This baffle consists of a 24-inch wide conical ring perpendicular to the tank wall. This baffle will interfere with the strut of the outer stand SDD-259 as shown in Figure 10.7-5. The strut will be cut and lengthened so as to clear the top of the baffle. It will be welded at the break point.

TABLE 10.7-I - VEHICLE PROCESSING OPERATIONS

PROCESSING FUNCTION	S-II with no restarts and S-IVB with 1 or 2 restarts (LOR and Sync Missions)		S-II with one restart ; No S-IVB (LEO Mission)	
	CHANGE	NO CHANGE	CHANGE	NO CHANGE
PHASE I - LOW BAY OPERATIONS				
1. Conduct low bay ESE verification - S-II	X		X	
2. Checkout and acceptance test the low bay pneumatics and fluid distribution system - S-II	X		X	
3. Conduct measurement station checkout - S-II		X		X
4. Conduct continuity checks between LCC and LUT	X		X	
5. Transfer S-II to VAB low bay area		X		X
6. Install S-II engine compartment set	X		X	
7. Install S-II engine compartment heat shield protection set	X		X	
8. Erect S-II in low bay cell		X		X
9. Conduct S-II receiving inspection		X		X
10. Install S-II access equipment		X		X
11. Conduct S-II pre-power tests	X		X	
12. Checkout S-II instrumentation	X		X	
13. Power up S-II stage	X		X	
14. Checkout S-II leak detection GSE	X		X	
15. Conduct S-II leak tests	X		X	
16. Checkout S-II telemetry	X		X	
17. Checkout S-II range safety		X		X
18. Checkout S-II thermal control system		X		X
19. Verify S-II leak detection system	X		X	
20. Checkout S-II engines	X		X	
21. Prepare for transfer to High Bay - S-II		X		X
22. Add S-II APS Checkout			NEW	
23. Add S-II APS Installation			NEW	
24. Conduct low bay ESE verification S-IVB	X		N/A	
25. Checkout low bay pneumatics - S-IVB	X		N/A	
26. Conduct instrumentation readiness check - S-IVB	X		N/A	
27. Transfer S-IVB to VAB low bay area		X	N/A	
28. Install S-IVB static dessicants	X		N/A	

TABLE 10.7-I - VEHICLE PROCESSING OPERATIONS (CONTINUED)

PROCESSING FUNCTION	S-II with no restarts and S-IVB with 1 or 2 restarts (LOR and Sync Missions)		S-II with one restart : No S-IVB (LEO Mission)	
	CHANGE	NO CHANGE	CHANGE	NO CHANGE
29. Install access equipment on S-IVB		X	N/A	
30. Erect S-IVB in low bay cell		X	N/A	
31 Install S-IVB access equipment		X	N/A	
32. Conduct receiving inspection on S-IVB		X	N/A	
33. Conduct umbilical impedance test - S-IVB	X		N/A	
34. Verify S-IVB instrumentation readiness	X		N/A	
35. Install umbilical - S-IVB	X		N/A	
36. Conduct stage logic reset	X		N/A	
37. Conduct power setups - S-IVB	X		N/A	
38. Conduct power distribution and control switchings S-IVB	X		N/A	
39. Verify chilldown load - S-IVB	DELETE		N/A	
40. Conduct propulsion system test Setups - S-IVB	X		N/A	
41. Conduct propulsion system tests - S-IVB	X		N/A	
42. Checkout APS module - S-IVB	X		N/A	
43. Install APS module - S-IVB	X		N/A	
44. Prepare for transfer to High Bay - S-IVB	X		N/A	
PHASE II - VEHICLE ERECTION				
45. Erect and checkout - S-IC		X		X
46. Erect S-II on S-IC		X		X
47. Hookup and verify S-II umbilical	X		X	
48. Conduct S-II ESE measurements compatability checks	X		X	
49. Conduct S-IC/S-II interface verification tests		X		X
50. Add S-II APS to vehicle interface checkout			NEW	
51. Checkout S-II thermal control systems		X	X	
52. Erect S-IVB on S-II		X	N/A	
53. Add erection of space station on S-II			NEW	
54. Checkout S-IVB/S-II electrical mating	X		N/A	

TABLE 10.7-I - VEHICLE PROCESSING OPERATIONS (CONTINUED)

PROCESSING FUNCTION	S-II with no restarts and S-IVB with 1 or 2 restarts (LOR and Sync Missions)		S-II with one restart : No S-IVB (LEO Mission)	
	CHANGE	NO CHANGE	CHANGE	NO CHANGE
55. Erect I. U.		X	X	
56. Checkout I. U. /S-IVB electrical mating		X	N/A	
57. Hookup and verify S-IVB umbilical	X		N/A	
58. Conduct S-IVB ESE Measurements compatibility checks	X		N/A	
59. Conduct S-IVB APS to vehicle interface checkout	X		N/A	
60. Conduct L/V electrical system preps	X		X	
61. Add electrical mating of I. U. to space station			NEW	
PHASE III - L/V SUBSYSTEM CHECKS				
62. Conduct L/V electrical systems verification test	X		X	
63. Conduct S-II LCC meter and recorder calibration	X		X	
64. Conduct S-IVB LCC meter and recorder calibration	X		N/A	
65. Conduct S-IVB power distribution and control switching test	X		N/A	
66. Conduct S-IVB DDAS subsystem test	X		N/A	
67. Conduct S-IVB engine ECS test	X		N/A	
68. Conduct S-IVB measurements verification	X		N/A	
69. Checkout S-IVB hydraulics	X		N/A	
70. Checkout S-IVB APS	X		N/A	
71. Checkout S-IVB telemetry system	X		N/A	
72. Calibration of S-IVB transducers	X		N/A	
73. Conduct range safety test - S-IVB		X	N/A	
74. Checkout S-IVB control system	X		N/A	
75. Checkout S-IVB SSB	X		N/A	
76. Checkout S-IVB propulsion	X		N/A	
77. Checkout S-IVB EBW	X		N/A	
78. Check S-IVB frequency response and actuator calibration	X		N/A	
79. Calibrate S-IVB liquid level sensor	X		N/A	

TABLE 10.7-I - VEHICLE PROCESSING OPERATIONS (CONTINUED)

PROCESSING FUNCTION	S-II with no restarts and S-IVB with 1 or 2 restarts (LOR and Sync Missions)		S-II with one restart : No S-IVB (LEO Mission)	
	CHANGE	NO CHANGE	CHANGE	NO CHANGE
80. Conduct ullage checks on S-IVB	X		N/A	
81. Conduct S-II engine ignition system checkout	X		X	
82. Conduct S-II recirculation system electrical checkout	DELETE		DELETE	
83. Verify S-II switch selector operation	X		X	
84. Conduct S-II umbilical disconnect leak check	X		X	
85. Conduct S-II range safety checks		X		X
86. Conduct S-II telemetry checks	X		X	
87. Checkout S-II temperature measurement	X		X	
88. Checkout S-II engine actuation system	X		X	
89. Checkout and leak check S-II propellant fill system	X		X	
90. Checkout S-II separation system		X	X	
91. Checkout S-II propellant dispersion	X		X	
92. Conduct S-II/DDAS test	X		X	
93. Checkout S-II engine safety circuits	X		X	
94. Perform combined G&C tests	X		X	
95. Conduct S-IVB preparations for malfunction OAT	X		N/A	
96. Conduct S-II preparations for malfunction OAT	X		X	
97. Add S-II APS checkout			NEW	
PHASE IV - VAB VEHICLE TESTS				
98. Conduct L/V malfunction OAT	X		X	
99. Conduct S-IVB differential pressure feedback test	X		N/A	
100 Verify vehicle/propellant networks interface	X		X	
101 Install S-IVB ordnance	X		N/A	
102 Conduct S-IVB APS checkout	X		N/A	
103 Install S-II ordnance	X		X	
104 Checkout S-II S&A device	X		X	
105 Conduct S-IVB live ordnance tests	X		N/A	
106 Conduct S-II live ordnance tests	X		X	
107 Checkout S-II measurements system	X		X	
108 Conduct preparations for swing arm OAT	X		X	

TABLE 10.7-I - VEHICLE PROCESSING OPERATIONS (CONTINUED)

	PROCESSING FUNCTION	S-II with no restarts and S-IVB with 1 or 2 restarts (LOR and Sync Missions)		S-II with one restart. No S-IVB (LEO Mission)	
		CHANGE	NO CHANGE	CHANGE	NO CHANGE
109	Conduct swing arm OAT		X		X
110	Erect Spacecraft		X		X
111	Conduct spacecraft testing		X		X
112	Conduct preparations of spacecraft electrical mating		X		X
113	Conduct preparations for S/V OAT #1	X		X	
114	Add S-II APS checkout			NEW	
	PHASE VB - PAD/ SV INTEGRATION				
115	Conduct SV OAT #1	X		X	
116	Install S-IVB ordnance	X		N/A	
117	Install S-II ordnance	X		X	
118	Transfer to Pad		X		X
119	Conduct pad ESE qualification	X		X	
120	Conduct LV/Pad electrical interface verification	X		X	
121	Conduct L/V hardwire tests	X		X	
122	Conduct S-IVB PC card calibration	X		N/A	
123	Conduct umbilical leak checks	X		X	
124	Prepare for S-II engine sequence test	X		X	
125	Prepare for S-IVB engine sequence test	X		N/A	
126	Conduct S-II engine sequence test	X		X	
127	Conduct S-IVB engine sequence test	X		N/A	
128	Conduct simulated cryogenic loading test		X	X	
129	Conduct facility power out test		X	X	
130	Conduct S-IVB PU calibration and checkout		X	N/A	
131	Conduct LV MCC-H interface command test	X		X	
132	Prepare for FRT	X		X	

TABLE 10.7-I - VEHICLE PROCESSING OPERATIONS (CONTINUED)

PROCESSING FUNCTION	S-II with no restarts and S-IVB with 1 or 2 restarts (LOR and Sync Missions)		S-II with one restart ; No S-IVB (LEO Mission)	
	CHANGE	NO CHANGE	CHANGE	NO CHANGE
PHASE VI - FINAL CHECKS, SERVICING & LAUNCH				
133 Conduct FRT	X		X	
134 Load S-IVB APS	X		N/A	
135 Add load S-II APS hypergolics			NEW	
136 Conduct CDDT preparations	X		X	
137 Conduct CDDT	X		X	
138 Securing L/V propellant systems		X	X	
139 Conduct countdown	X		X	
140 Launch	X	X	X	X
141 Refurbishment				

TABLE 10.7-II REDLINE COMPARISON

S-II AS-503 MANDATORY	S-II - NO RESTARTS		S-II - ONE RESTART	
	SAME	DELETE	SAME	DELETE
1. LOX Pump Discharge Temperature		X		X
2. Start Tank Gas Temperature		X		X
3. LH_2 Tank Precondition Temperature	X		X	
4. Thrust Chamber Jacket Temperature		X		X
5. Engine Inlet LH_2 Temperature		X		X
6. Reservoir Outlet Fluid Temperature	X		X	
7. He Tank Pressure	X		X	
8. Start Tank Pressure		X		X
9. LOX Tank Ullage Pressure	X		X	
10. LH_2 Tank Ullage Pressure	X		X	
11. Valve Actuation He Bottle Pressure	X		X	
12. Valve Actuation Regulator Outlet Pressure		X		X
13. Feedline Elbow Inlet Pressure	X		X	
14. Common Bulkhead Internal Pressure	X		X	
15. Engine Inlet LH_2 Pressure		X		X
16. Common Bulkhead Insulation Outlet Pressure	X		X	
17. Sidewall Insulation Outlet Pressure	X		X	
18. Hydraulic Gas Pressure	X		X	
19. He Injection System Pressure		X		X
20. Common Bulkhead Internal Pressure	X		X	
21. P. U. Valve Position		X		X
22. He Injection PRI Orifice Outlet Pressure		X		X
23. Hydraulic Reservoir Piston Position	X		X	
24. Yaw Actuator Piston Position	X		X	
25. Pitch Actuator Piston Position	X		X	
26. Engine Cut-Off	X		X	
27. LOX Depletion Sensors Open (Wet)	X		X	
28. Fuel Depletion Sensors Open (Wet)	X		X	
29. Main DC Bus Voltage	X		X	
30. Main Battery Voltage	X		X	
31. Instrumentation DC Bus Voltage	X		X	
32. Instrumentation Battery Voltage	X		X	
33. RSCR Signal Strength Low	X		X	

TABLE 10.7-II REDLINE COMPARISON (CONTINUED)

S-II AS-503 MANDATORY	S-II - NO RESTARTS		S-II - ONE RESTART	
	SAME	DELETE	SAME	DELETE
34. Recirculation DC Bus Voltage		X		X
35. Recirculation Battery Voltage		X		X
36. Ignition Battery Voltage		X		X
37. Ignition DC Bus Voltage		X		X
38. 115 VAC P. U. Package Voltage	X		X	
39. LOX Fill/Drain Line Umbilical Coupling Pressure		X		X
40. Common Bulkhead Circuit H_2 Concentration	X		X	
41 Common Bulkhead Circuit O_2 Concentration	X		X	
42. LH_2 Feedline Circuit Hydrogen Concentration	X		X	
	Engine He Tank Press Engine LOX Pump Inlet Temp LH_2 Pump Inlet Temp Fill & Drain Coupling Temp		Attitude Control Fuel Mod. Temp. Attitude Control Oxidizer Mod Temp. Attitude Control Tank Pressure Oxidizer Tank Ullage Volume Mod (APS) Pressure Fuel Tank Ullage Volume Mod (APS) Pressure TVC Battery Voltage TVC Bus Voltage Engine He Tank Press Engine LOX Pump Inlet Temp LH_2 Pump Inlet Fill & Drain Coupling Temp	

TABLE 10.7-II REDLINE COMPARISON (CONTINUED)

S-IVB MEASUREMENT (AS-503 MANDATORY)	S-IVB WITH J-2S REQUIRED			
	YES	NO		
1. Fuel Pump Inlet Temperature		X		
2. GH_2 Start Bottle Temperature		X		
3. Attitude Control Fuel Module Temperature	X			
4. Attitutde Control Oxidizer Module Temperature	X			
5. Hydraulic Pump Inlet Oil Temperature	X			
6. Reservoir Oil Temperature	X			
7. Control He Regulator Discharge Pressure	X			
8. Cold He Sphere Pressure	X			
9. GH_2 Start Bottle Pressure		X		
10. Engine Control He Sphere Pressure	X			
11. Fuel Tank He Bottle Repressurization Pressure	X			
12. Attitude Control He Pressurization Tank Pressure	X			
13. Hydraulic System (Auxiliary Pump On) Pressure	X			
14. Reservoir Oil (Auxiliary Pump OFF) Pressure	X			
15. Oxidizer Tank Ullage Volume Mod. (APS) Press.	X			
16. Fuel Tank Ullage Volume Mod. (APS) Pressure	X			
17. LOX Tank Repressurization He Spheres Pressure	X			
18. Auxiliary Hydraulic Pump Air Tank Pressure	X			
19. Ambient He Pneumatic Sphere Pressure	X			
20. Fuel Tank Ullage Umbilical Pressure	X			
21. Oxidizer Circulation Pump Flow Rate		X		
22. Fuel Circulation Pump Flow Rate		X		
23. Actuator Piston Pot Pitch Position	X			
24. Actuator Piston Pot Yaw Position	X			
25. Cut-Off Signal (Lock-In) Event	X			
26. Reservoir Oil Level	X			
27. Aft and Forward Battery Output Voltage	X			
28. Engine Control Bus Voltage	X			
29. RSCR Signal Strength Low Level	X			
30. Propellant Utilization Oven Stability Monitor	X			
31. Oxidizer Tank Umbilical Pressure	X			
32. PU System Ratio Valve (VSG 10-401)	X			
33. SPTS Case Temperature	X			

TABLE 10.7-III INTERLOCKS DELETED FROM THE S-II SYSTEM

RELAY	FUNCTION	REFERENCE FIGURE
K118	Recirculation Ready for Launch	2
K126	Any LH2 Depletion C/O Sensors Dry	2
K227	LOX Return Line Valves Open	5
K223	All LOX Prevalves Open	5
K206	All LOX Prevalves Open	5
K228	Pump Valves Open	5
K217	Pump Valves Open	5
K226	All LH2 Prevalves Closed	5
K213	All LH2 Prevalves Closed	5
K218	Return Line Valve Open	5
K216	Close Recirculation Pump Return Line Valves	5
K215	Close Recirculation Pump Valves and Return Line Valves	5
A24K1	LOX and LH2 Return Line and LH2 Pump Valve Control	5
K111	Recirculation Stop Reset	6
K50	Recirculation Reset	6
K1-1	Ullage Trigger	6
K823	All Gas Generator Valves Closed	7
K125	Recirculation Ready for LOX Load	8
K237	10% LOX Level	8
K223	All LOX Prevalves Open	8
K206	All LOX Prevalves Open	8
K823	All Gas Generator Valves Closed	8
K9	Recirculation Bus Supervision	4
K96	+2DS11 Power On	10
K40	Recirculation Power Transfer Isolation	10
K28	Recirculation Bus Power Transfer	10
K34	Recirculation Bus Internal	10
K103	OAT Recirculation Bus Supervision	10
K18	Recirculation Bus Voltage OK	4
K848	LH2 1 Dry	9
K850	LH2 2 Dry	9
K852	LH2 3 Dry	9
K854	LH2 4 Dry	9
K856	LH2 5 Dry	9

NOTE The prevalve interlocks deleted from the Recirculation System should be used in the Propellant Loading Chain to ensure that the prevalves are open prior to start of propellant flow.

TABLE 10.7-III INTERLOCKS DELETED FROM S-IVB SYSTEM (CONTINUED)

RELAY	FUNCTION	REFERENCE FIGURE
K54	Ullage Rocket Pilot Relays Reset	2, 11
K633	LH_2 Chilldown Inverter Power On	2, 11
A45A6K6	LH_2 Chilldown Inverter Power On	13
K623	LH_2 Chilldown Valve Open	2, 11
K631	LH_2 Prevalve Closed	2, 11
K622	LOX Chilldown Valve Open	2, 11
K632	LOX Chilldown Inverter Power On	2, 11
A45A6K5	LOX Chilldown Inverter Power On	13
K629	LOX Prevalve Closed	2, 11
K646	LOX & LH_2 Prevalve Emergency Close Command	2, 11
K639	LOX & LH_2 Prevalve Emergency Close Command	13
K424	LH_2 Bleed Valve Closed	2, 11
A45A9K19	Fuel Bleed Valve Closed	13
K425	LOX Bleed Valve Closed	2, 11
A45A9K18	LOX Bleed Valve Closed	13
K890	70 lb. Thrust Ullage Engine Relay Reset	12
A3A10K1	Ullage Rocket Relays	12
A3A10K2	Ullage Rocket Relays	12
A3A10K3	Ullage Rocket Relays	12
A3A10K4	Ullage Rocket Relays	12

NOTE These interlocks can be deleted from the interlock chain which activates K968 - S-IVB Stage Ready for Firing and K972 - S-IVB Stage Ready for Launch.

TABLE 10.7-III INTERLOCKS ADDED TO S-II SYSTEM FOR LEO MISSION (CONTINUED)

Interlock Condition	Reason
APS No. 1 Engine Valve Power On	To ensure that APS engine valve power is available prior to S-IC ignition.
APS No. 2 Engine Valve Power On	Same as above
LH_2 Vent Directional Control in Flight Position	To ensure that the LH_2 tank is venting through balanced vent system.
Auxiliary Hydraulic Pump Power On (Flight Mode)	To ensure that power is available for TVC during idle mode.

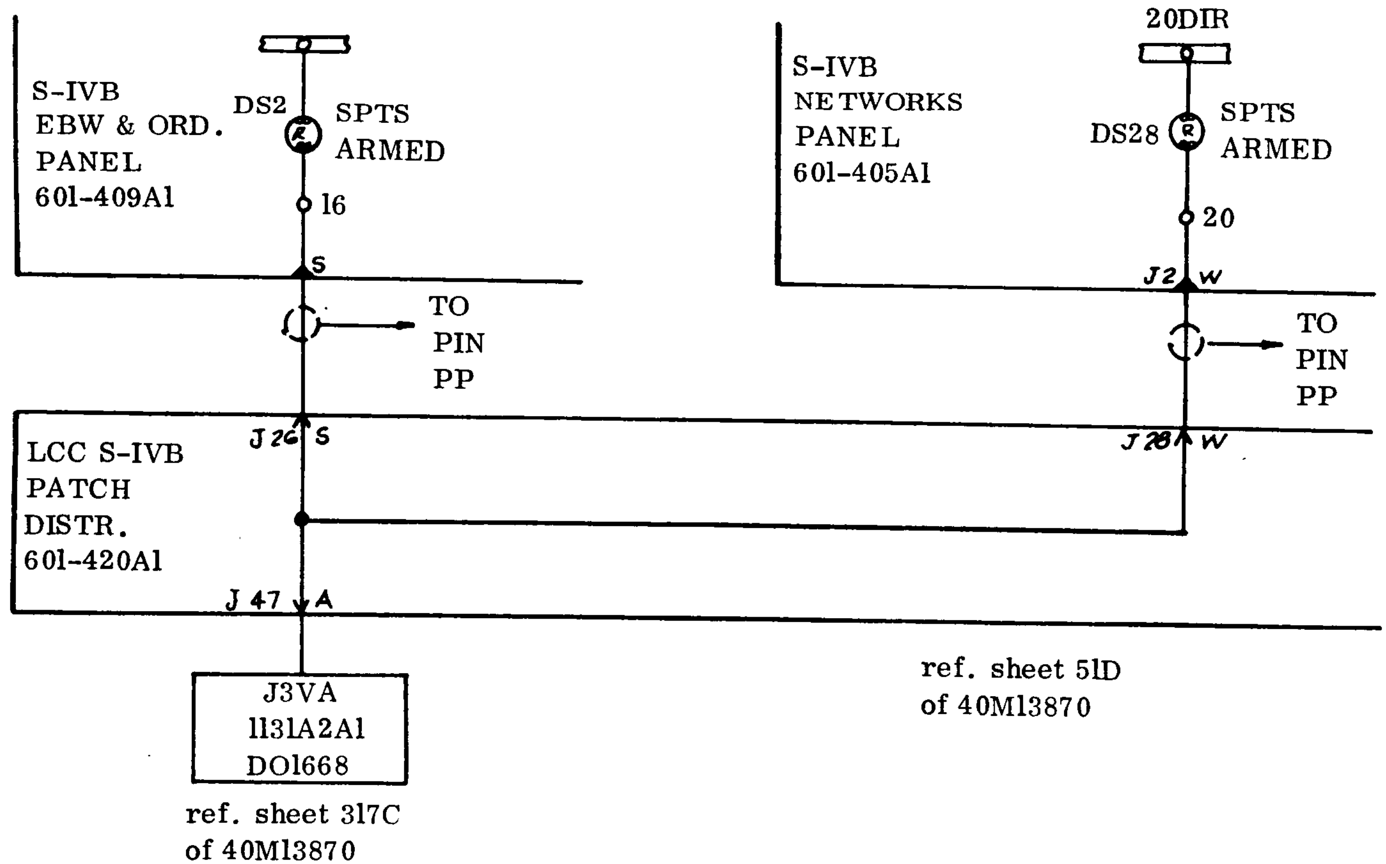


FIGURE 10.7-1 - CONCEPTUAL DESIGN SCHEMATIC
Reidentify "Ullage Rockets Pilot Relays Reset" Indicators to "SPTS Armed" Indicators

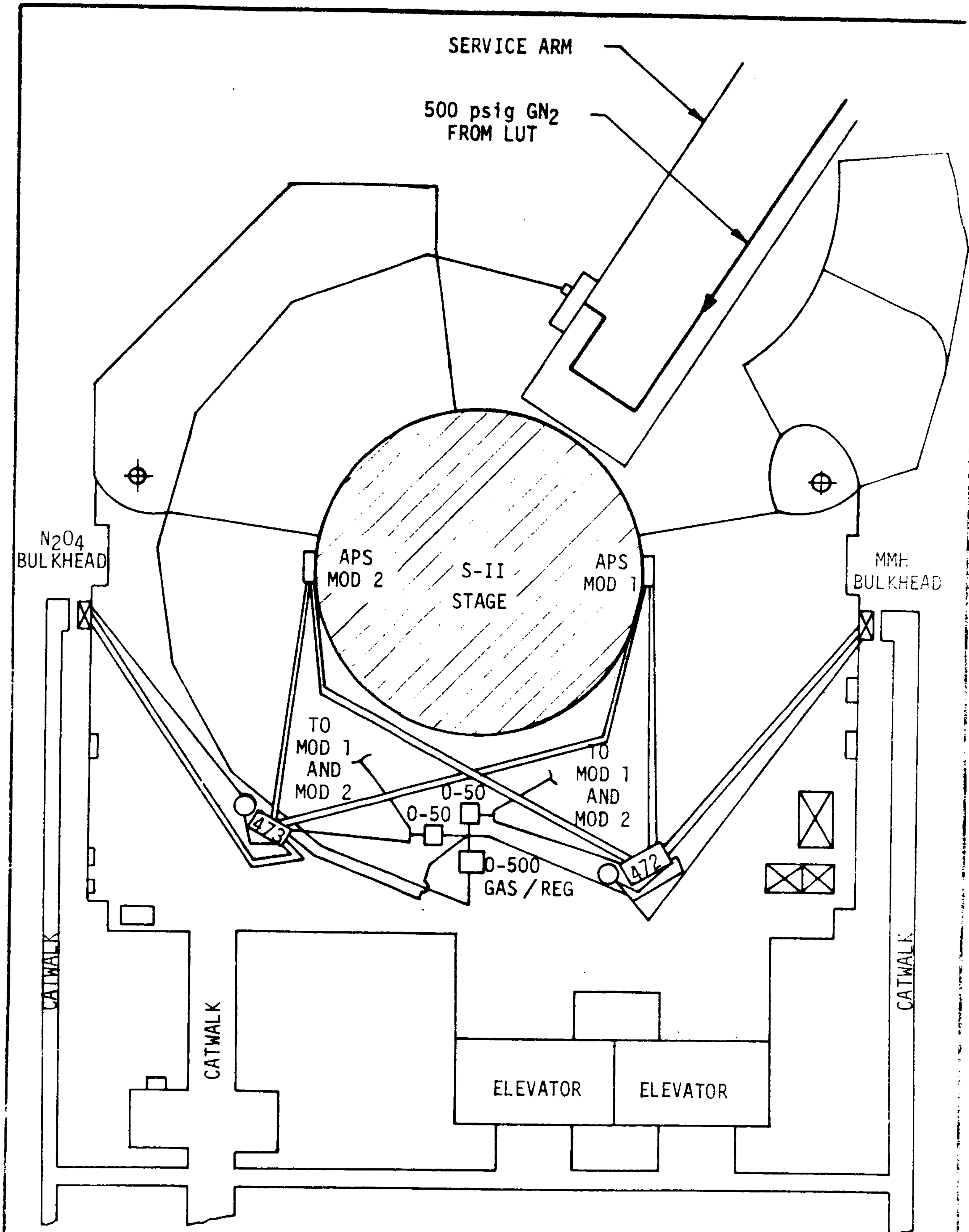


FIGURE 10.7-2 PLANVIEW OF MSS PLATFORM #1

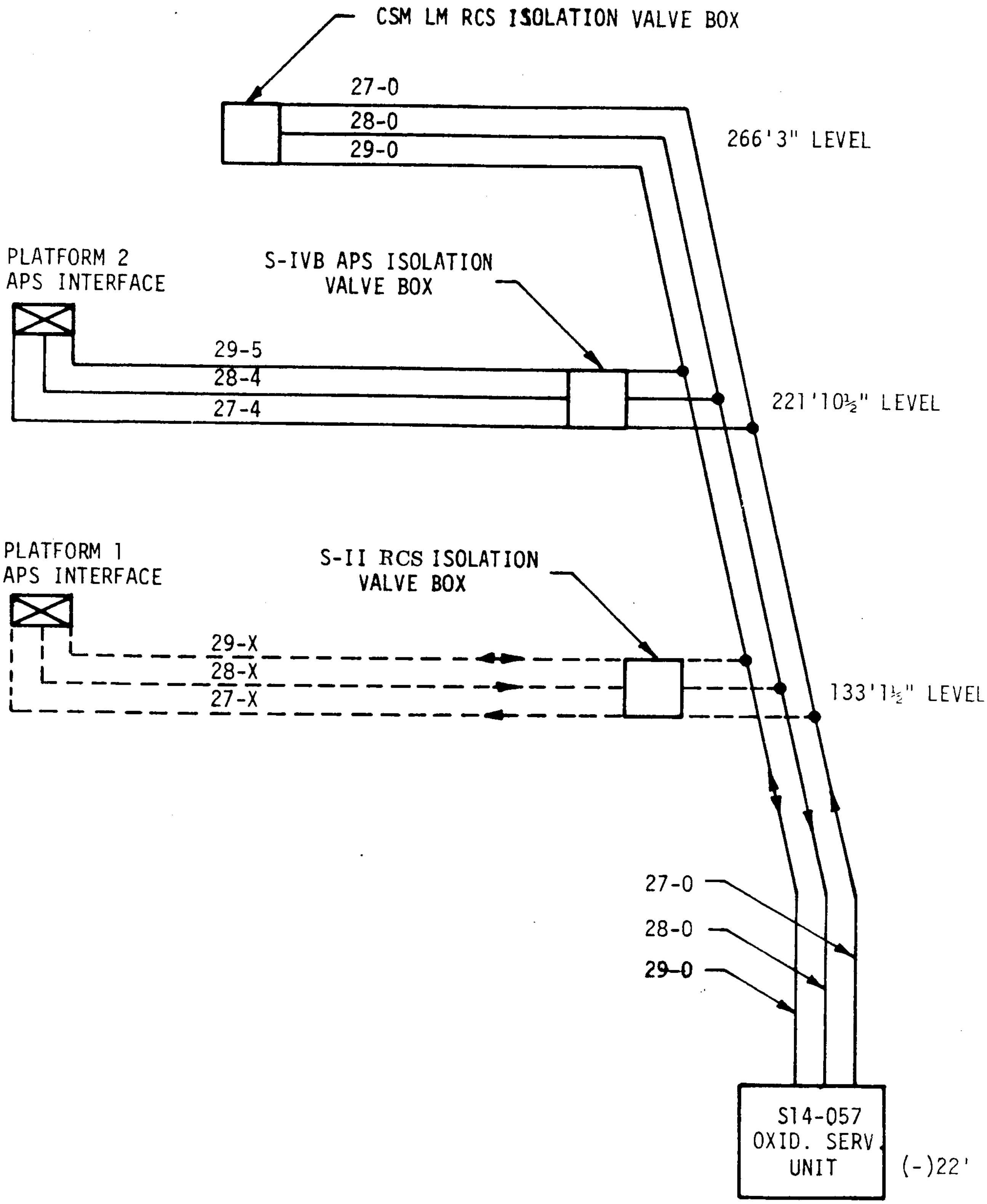


FIGURE 10.7-3 RCS OXIDIZER SYSTEM

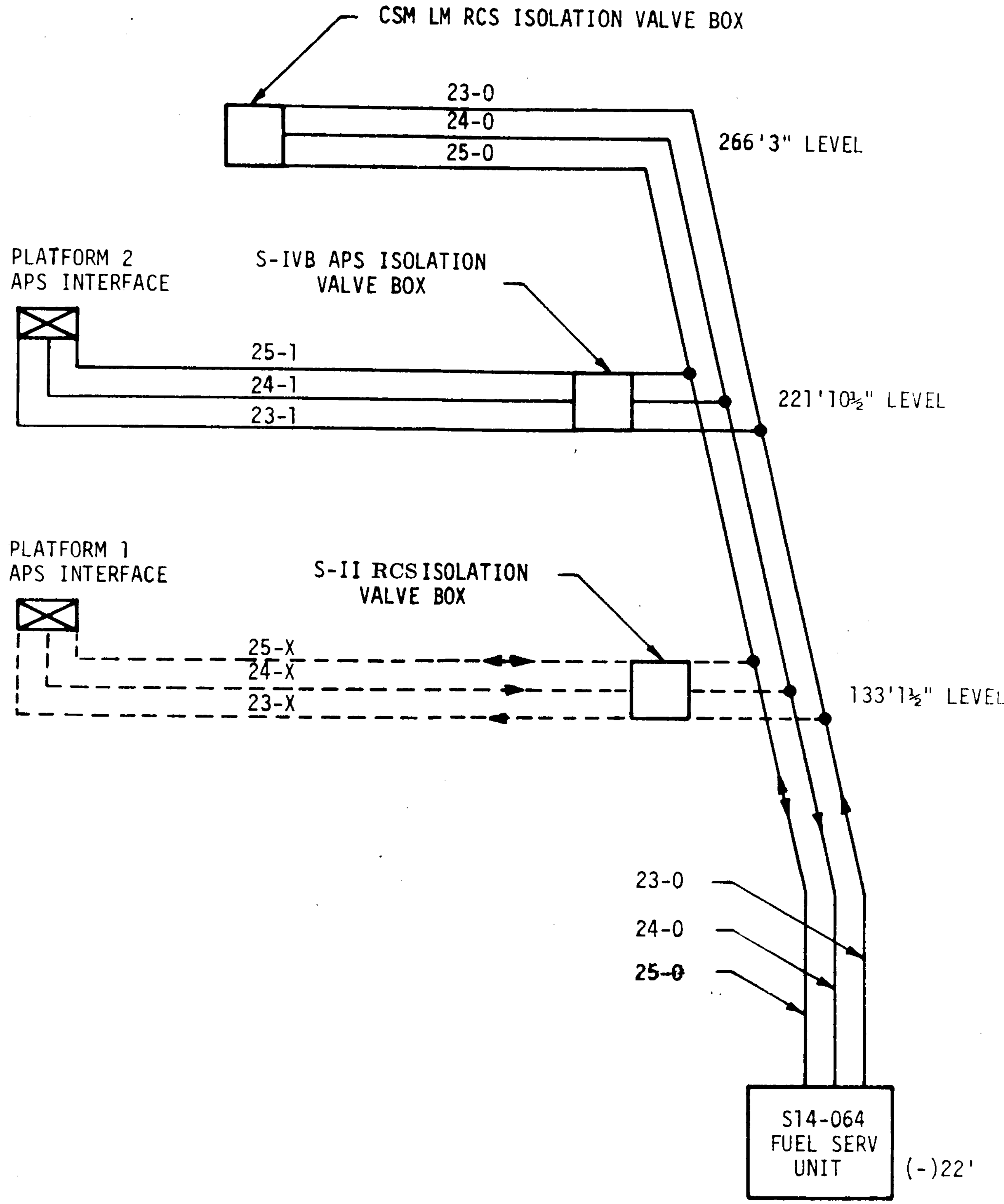


FIGURE 10.7-4 RCS FUEL SYSTEM

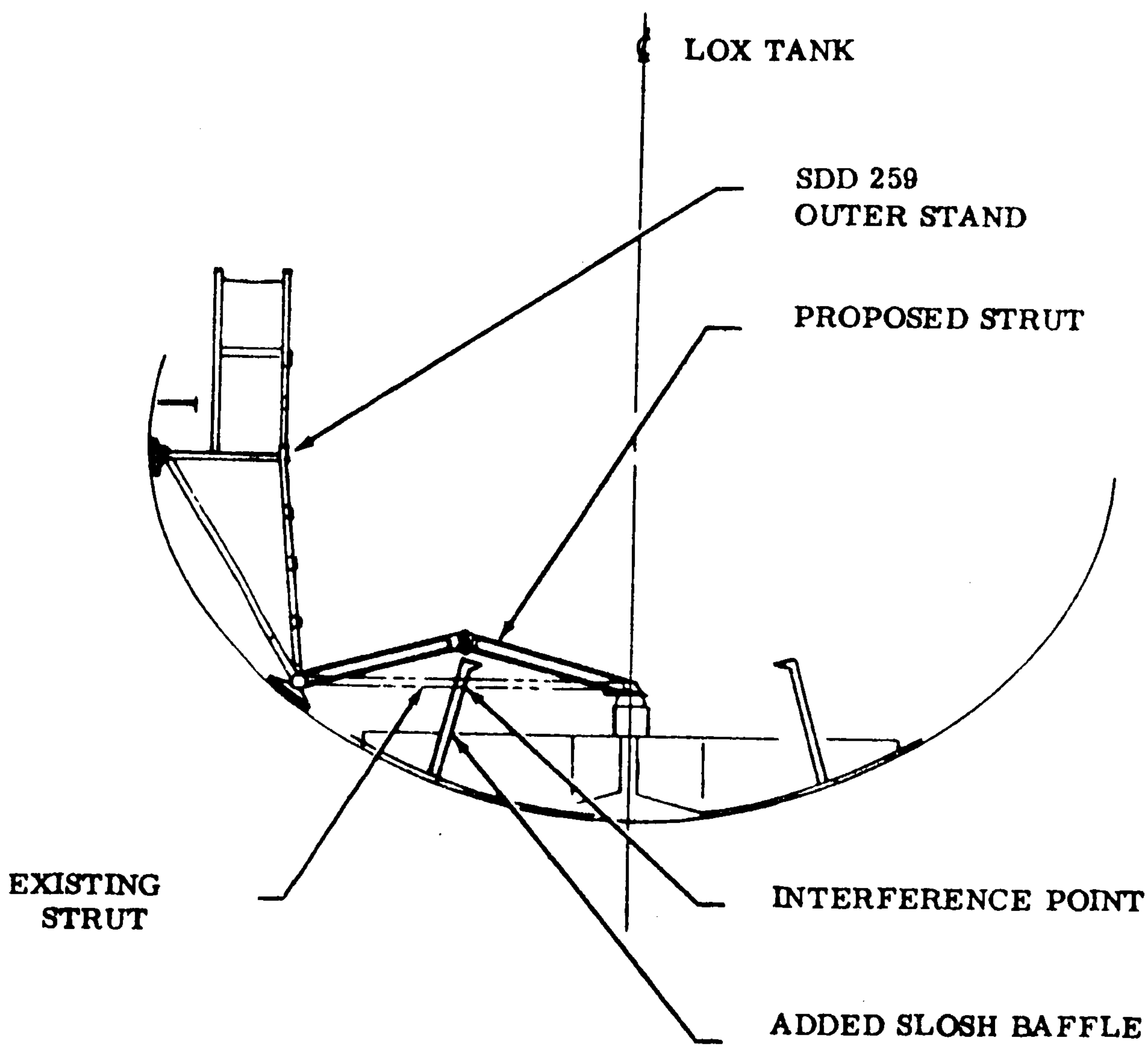


FIGURE 10.7-5 SDD 259 LOX TANK INTERNAL ACCESS KIT MOD FOR LEO MISSION

10.8 SYSTEMS ENGINEERING AND INTEGRATION (SE&I)

SE&I is presently performed by The Boeing Company for the Saturn V vehicle and is divided into four activity categories.

Activity Category	Tasks
System Integration	Program Control, Configuration Management, Test Program Integration and Launch Readiness Assessment, and Logistics
Systems Engineering	Interface Engineering, System Definition, Pre-Launch System Analysis, Design Certification Review, Reliability Analysis Mission Rules, and System Safety.
Technology	Flight Evaluation, Propulsion System Analysis, Structural System Analysis, and Instrumentation System Analysis
Launch Vehicle and Mechanical Ground Support Equipment (LVGSE)	LVGSE Logistics and System Development Facility (SDF)

The tasks, such as Program Control, Configuration Management, Test Program Integration, etc., investigate the overall launch vehicle and its systems including the engines, but are independent of type of engine used. These tasks are not impacted.

The tasks, Interface Engineering, Flight Evaluation, and Propulsion System Analysis required additional SE&I development and recurring efforts. The LOR Mission requires efforts to develop the necessary guidance system modifications and computer trajectory simulation modifications. The three-stage synchronous mission with three-start S-IVB requires additional analysis and development to assess J-2S synchronous orbit mission capabilities and to rework the assumed developed synchronous orbit guidance to incorporate J-2S capabilities. Additional work is also required in guidance and simulation modifications for the two-stage LEO mission.

10.8.1 SE&I Tasks Definition

10.8.1.1 System Integration

Systems Integration tasks consist of the following:

10.8.1.1 (Continued)

a. Program Control: Program Control responsibility is to prepare and maintain the Saturn V baseline schedule plans from schedules and technical data collected from other NASA/MSFC organizations and contractors to provide visibility to the program management on overall Saturn V schedules. The schedules are monitored and kept up-to-date. Status reports and contingency and long range planning with recommendations are prepared.

Impact: This task is independent of the type of engine and thus requires no increased effort.

b. Configuration Management: Configuration Management responsibility is to prepare and maintain MSFC configuration management manual and contractor requirements as directed by MSFC to assure compatibility of equipment, perform change integration activities subject to MSFC review and approval, and maintain MSFC specification index and centralized specification.

Impact: This task is independent of the type engine, and thus requires no increased effort.

c. Test Program Integration and Launch Readiness Assessment: This task responsibility is to develop program level plans and directives which establish requirements to integrate and control Saturn V test activities and mission activities. The task's other responsibilities are also to develop and maintain readiness review plans, procedures and schedules, and to develop and maintain launch vehicle test and checkout requirements, specifications, criteria and redline documentation for use at KSC.

Impact: This task is independent of the type of engine, and thus require no increased effort.

d. Logistics: Logistics responsibility is to provide and manage logistics products and services in support of the Saturn V system at KSC and test sites such as procurement and delivery of hardware kits, reporting spares status, identifying and assessing logistics problems.

Impact: This task is independent of the type of engine and thus requires no increased effort.

10.8.1.2 System Engineering

System Engineering tasks consists of the following:

10.8.1.2 (Continued)

a. Interface Engineering: Interface Engineering task responsibility is to maintain the Saturn V mechanical Interface Control Documents which control stage to stage and stage to engine interfaces as well as vehicle to MGSE and vehicle to launch facility interfaces. ECPs, ECRs and AVOs submitted by Contractors and NASA Labs are analyzed and evaluated for interface compatibility. Acceptance or rejection of the ECPs, ICRs and AVOs is recommended to MSFC; after government concurrence, IERs and IRNs are prepared for contract compliance.

Impact: The Interface Engineering task is impacted by the J-2S requirements. Development effort is needed for researching, defining and documenting new interface changes resulting from the S-II and S-IVB stages design modifications. Such changes involve relocating pneumatics, deletion or relocation of fluid lines and disconnects rerouting and clamping of electrical harnesses and deletion or de-activation of kits.

b. System Definition: The system definition task is to compile data furnished by the government and the contractors to assemble a vehicle system definition document for NASA approval. The document, when approved, provides consistent technical baseline for design operation, maintenance, logistics, reliability and safety analysis. Such documents are Saturn Flight Manual, Mechanical Schematic and Ordnance system documents.

Impact: This task is independent of the type of engine, and thus requires no increased effort.

c. Prelaunch System Analysis: This task responsibility is to analyze the Saturn V system, identify problems and provide data and recommendations for NASA management to evaluate and determine the effectiveness of the launch vehicle and LVGSE.

Impact: This task is independent of the type of engine, and thus requires no increased effort.

d. Design Certification Review (DCR): The DCR responsibility is to prepare directives and procedures which establish requirements and outline for reporting proof of design and development maturity of Saturn V vehicle. DCR reports are collected from stage and GSE contractors, then integrated, critiqued and distributed to cognizant government personnel for review and evaluation. DCR activities also include planning and managing DCR oral presentations, identifying problems and tracking action items.

Impact: This task is independent of the type of engine, and thus requires no increased effort.

10.8.1.2 (Continued)

e. Reliability Analysis Program: This task responsibility is to compile data from contractors to prepare and maintain Failure Effect Analysis (FEA), Criticality Determination (CD) and Reliability Analysis Model (RAM) reports for use by other NASA organizations and stage contractors to evaluate the reliability of the stage systems and overall vehicle performance.

   Impact: This task is independent of the type of engine, and thus requires no increased effort.

f. Mission Rules: This task responsibility is to review, integrate and update Saturn V launch and flight mission rule documents which provide operating ground rules for vehicle prelaunch and flight operation.

   Impact: This task is independent of the type of engine, and thus requires no increased effort.

g. Saturn System Safety: This task responsibility is to maintain system safety program standards, document performance analysis of safety criteria, development requirements and level of compliance, define manned flight awareness and schedule exhibits.

   Impact: This task is independent of the type of engine, and thus requires no increased effort.

10.8.1.3 Technology

Flight System Analysis consists of the following tasks:

a. Flight Evaluation: The SE&I Flight Evaluation responsibilities include trajectory and performance analysis. Input data in digital form are received, through the MSFC Computation Laboratory from the launch vehicle stage contractors and launch range facilities. These raw data are used to reconstruct the launch vehicle trajectory through use of digital and analog computers. The computers compare flight data with preflight predictions and plot all data which fall outside the predicted bands.

   Detailed technical system performance analysis is also performed utilizing a Unified Flight Analysis System concept. The output from these analyses and stage contractor's analyses are incorporated into the final evaluation report and submitted to NASA headquarters. The analysis results are used as a feedback for design changes and anomaly corrections.

10.8.1.3 (Continued)

Impact: The flight system analysis is impacted by the change in mission configuration. The three-stage synchronous mission with three-start S-IVB and two-stage LEO mission impose changes in analysis which are unique and partly independent of the use of J-2 or J-2S engines. The changes involve necessary guidance system modification and computer trajectory simulation modifications to both digital and hybrid simulators.

Additional analysis and development is required to assess J-2S synchronous orbit mission capabilities and rework the assumed developed synchronous orbit guidance to incorporate J-2S capabilities. Hybrid and digital simulations will also require additional update. Additional work is required in guidance and simulation modifications to implement the two-stage configuration capability.

b. Propulsion System Analysis: Propulsion System Analysis responsibility includes propulsion system performance prediction, flight evaluation, environmental control and structural heating study.

Impact: The impact on propulsion performance to incorporate J-2S engine into propulsion models involves checkout, modifications, revision and verifications of engine digital and hybrid programs. The following are descriptive summaries of the changes involved:

1. Revise J-2S Engine Steady State Digital Program (PAST), developed by Rocketdyne. A basic PAST engine program should be furnished as GFD.

   (a) Revise PAST Program
   (b) Modify Program (develop separate S-II and S-IVB Models)
   (c) Revise the engine models in other S-II and S-IVB analysis programs.
   (d) Verification of all programs utilizing the J-2S digital model

2. Revise J-2S Engine Transient Analysis Hybrid Program; sufficient J-2S test data should be available to allow development of an adequate model.

   (a) Analyze test data and develop performance maps
   (b) Revise existing J-2 engine program
   (c) Check out new program

10.8.1.3 (Continued)

c. Structural System Analysis: This task includes flight loads and mass analysis, ground winds, structural dynamic characteristics, vibration and acoustics data, structural design accuracy and flight evaluation.

Impact: The type of engine or mission variation imposes no impact on this task.

d. Instrumentation System Analysis: This task includes telemetry systems and RF systems analysis.

Impact: The type of engine or mission variation imposes no impact on this task.

10.8.1.4 Launch Vehicle and Mechanical Ground Support Equipment (LVGSE)

a. LVGSE Logistics: This task responsibilities are to provide and manage logistics resources for specific LVGSE. Specific responsibilities include planning, analysis, spares provisioning, procurement, repair and modification, supply system management and O&M manual preparation.

Impact: This task is independent of use of J-2S, and thus requires no increased effort.

b. Systems Development Facility (SDF)

The Systems Development Facility, known as the "Breadboard" is located in Building 4708 at MSFC. The Breadboard houses the LCC electrical support equipment, LUT GSE, LCC RCA 110A, LUT RCS 110A, Operational Display System, DDAS, S-IC electrical simulator, S-II electrical simulator, S-IC Mechanical Automation Breadboard, S-IVB Mock-up (500 ST stage) and the I. U. stage.

The Breadboard is a functional replica of the launch systems and associated GSE. It simulates electrically the functions of the vehicle systems and subsystems. The three main functions of the Breadboard are:

1. To verify LVGSE and ESE equipment Mod Kits (physical and/or electrical intent) prior to their incorporation at KSC.

2. To debug and validate all program tapes and procedures such as the LVDC flight program, Saturn V launch computer complex operating systems and vehicle test programs, and

10.8.1.4 (Continued)

3. To investigate and analyze vehicle systems and subsystem problems.

The modifications and changes to the SDF due to the J-2S engine are mainly electrical patch work to the ESE patch boards and wiring changes to the panels. Some plumbing changes to the S-IVB-500 ST stage, S-IVB Pneumatic Console model 433A and the auxiliary pneumatic console are required. Installation of new mod kits are required such as the Flight Control Computer (FCC) and the Control Distributor which are modified by IBM on the I.U. stage to adapt J-2S engines.

A J-2S engine will be required to replace the existing J-2 engine in S-IVB-500 ST stage. The J-2S engine will be acquired from the engine development program and will be available at no cost to the government.

A summarized list of SDF changes is as follows:

1. Replace existing S-IVB J-2 engine with J-2S engine. This would include the removal of existing S-IVB J-2 engine, installation of the J-2S engine and checkout of the J-2S engine and checkout of the J-2S engine integrated with the S-IVB 500 ST stage.

2. Incorporate modifications to existing equipment. This effort would require installation and checkout of modifications to the following equipment:

   (a) S-IVB-500 ST stage
   (b) S-IVB electrical support equipment
   (c) S-IVB mechanical support equipment simulator
   (d) S-IVB (DDAS (Digital Data Acquisition System)
   (e) S-IVB auxiliary pneumatic console
   (f) S-IVB pneumatic console, model 433A
   (g) S-II stage electrical simulator
   (h) S-II electrical support equipment
   (i) S-II DDAS
   (j) S-II mechanical support equipment simulator
   (k) I.U. DDAS

   Impact: There is no increased effort to the SDF task, since all changes and modifications will be routine operations, without interfering with the scheduled work at the time.

SECTION 11

RESOURCES

11.0 INTRODUCTION

This section describes the integrated vehicle resources.

Resources data has been collected from the stage data and grouped to portray an overall resource picture of the integrated vehicle.

The following integrated vehicle resources plans are presented:

Vehicle Facilities Plan
Vehicle Test Plan
Vehicle Master Schedule Plan

11.1 VEHICLE FACILITIES PLAN

The J-2S implementation requirements impose no change, modification or addition to the existing Saturn V facilities.

11.1.1 Contractors Facilities

These facilities are normally located at the Contractor's headquarters. Some of these facilities are Government funded. The contractors anticipate no additional requirement for the expansion of these facilities to support the J-2S efforts, however, the S-II stage Contractor anticipates a continued situation where government-owned equipment and facilities are inadequate for their intended use because of obsolescence or innovations. Special funds have been available for these minor facility replacements and modifications. The S-II Contractor anticipates a continued requirement of approximately $100, 000 per year, regardless of manufacturing schedule change or production rate.

11.1.2 Marshall Space Flight Center (MSFC)

No expansions or modifications are anticipated at MSFC. The DTV facility is not affected since no vehicle dynamic testing is required for the J-2S program (test plan is presented in section 11.2.1). Special structural tests such as the S-II stage thrust structure and aft skirt will be conducted on NASA R-P&VE facilities at MSFC. These facilities are adequate and require no expansion or modifications.

11.1.3 Mississippi Test Facility (MTF)

This is a government owned facility at which the Contractor provides management services during static firing of the 1st and 2nd stages (S-IC & S-II) of the Saturn V

11.1.3 (Continued)

vehicle. The test complex for each stage contains static test stands, a control center, and other related support facilities. The MTF is adequate to support the J-2S program; no problem is anticipated.

11.2 VEHICLE TEST PLAN

11.2.1 Dynamic Testing

The SA-500D Dynamic Test Vehicle (DTV) Program has been completed at MSFC. The objectives of the DTV were to measure the vehicle characteristics during dynamic excitation of the vehicle and to develop mathematical models to predict and verify these characteristics. Math models and prediction techniques derived during SA-500D DTV Program are available for accurately predicting structural dynamic characteristics of Saturn V type boost stages. The math models and prediction techniques are developed for the three configurations shown in Figure 11.2-1, (1) S-IC/S-II/S-IVB/IU/PL, (2) S-II/S-IVB/IU/PL and (3) S-IVB/IU/PL. The boost stages of all basic vehicles investigated in this study (LOR, Synchronous, and LEO missions) are similar to SA-500D and their structural dynamic characteristics can be predicted analytically. No dynamic testing of the boost stages is needed.

Apollo type payload dynamic responses are predictable with math models updated from Short Stack Dynamic Vehicle (SSDV) test data run at MSC. SSDV configuration (S-IVB forward skirt/IU/PL) is shown in Figure 11.2-2. Payload local dynamic responses are difficult to predict; therefore, payloads other than Apollo will probably require a SSDV test at the MSC Spacecraft Acoustic Lab.; however, these tests are part of the payload development and would be accountable as such and not charged against the J-2S/Saturn V vehicle program.

11.2.2 Wind Tunnel Testing

Wind Tunnel Model tests investigate aerodynamic characteristics and dynamic behavior of the Saturn V under laboratory conditions. Wind tunnel testing is not required because the flight vehicle configurations are geometrically identical to Apollo. Aerodynamic coefficients have been established. Force and stability data have been gathered from Vehicle-Force Model and Vehicle Ground Winds Elastic Model to establish the capability of the flight vehicle to withstand the dynamic loads.

11.2.3 Manrating Flight Qualification Testing

First J-2S Saturn V will be manrated. The Saturn V vehicle is considered far advanced in development and development and design maturity. The risk is minimized. Each Saturn stage affected (S-II, S-IVB and IU) has its own test plan to perform the

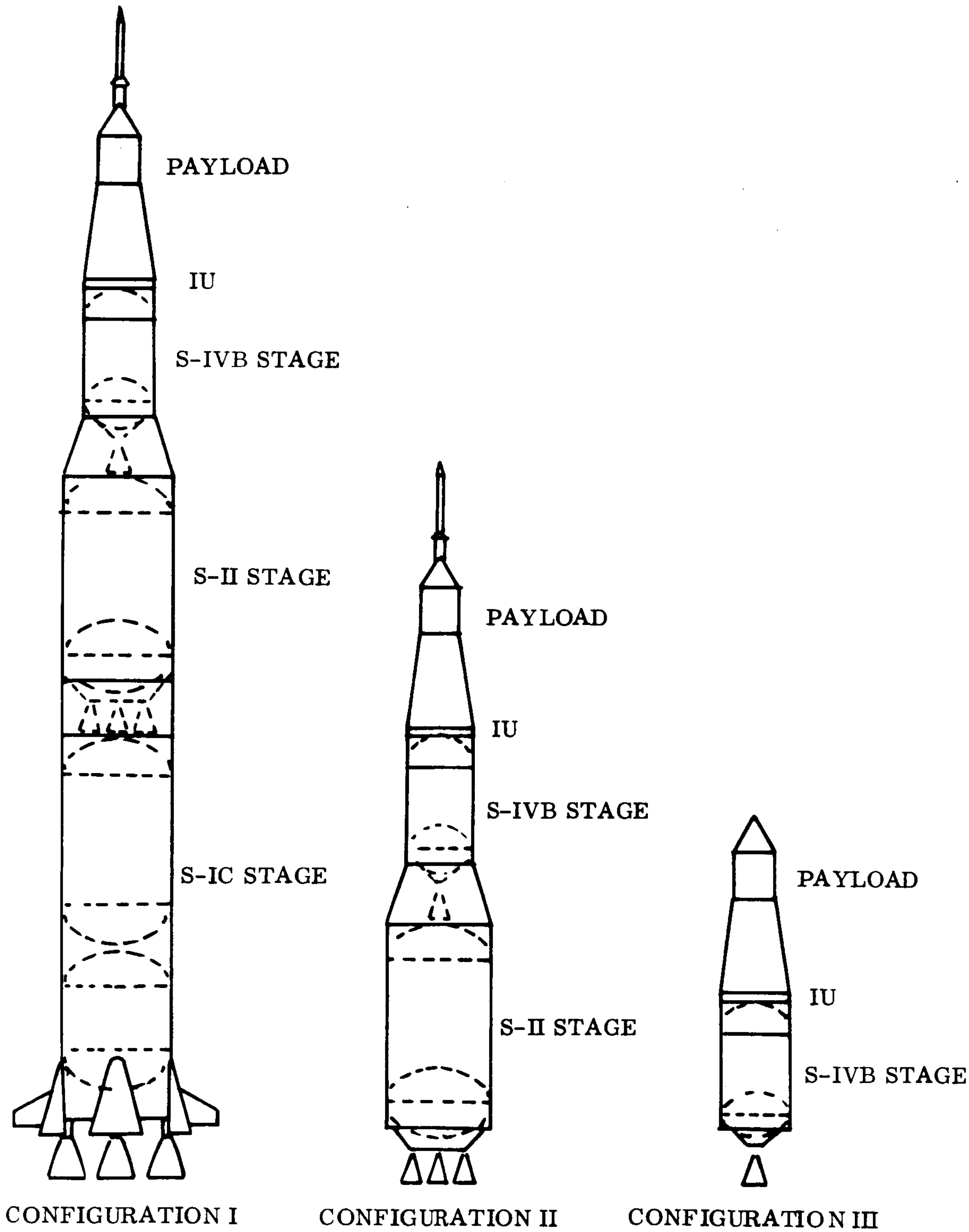


FIGURE 11.2-1 SA-500D DYNAMIC TEST CONFIGURATIONS

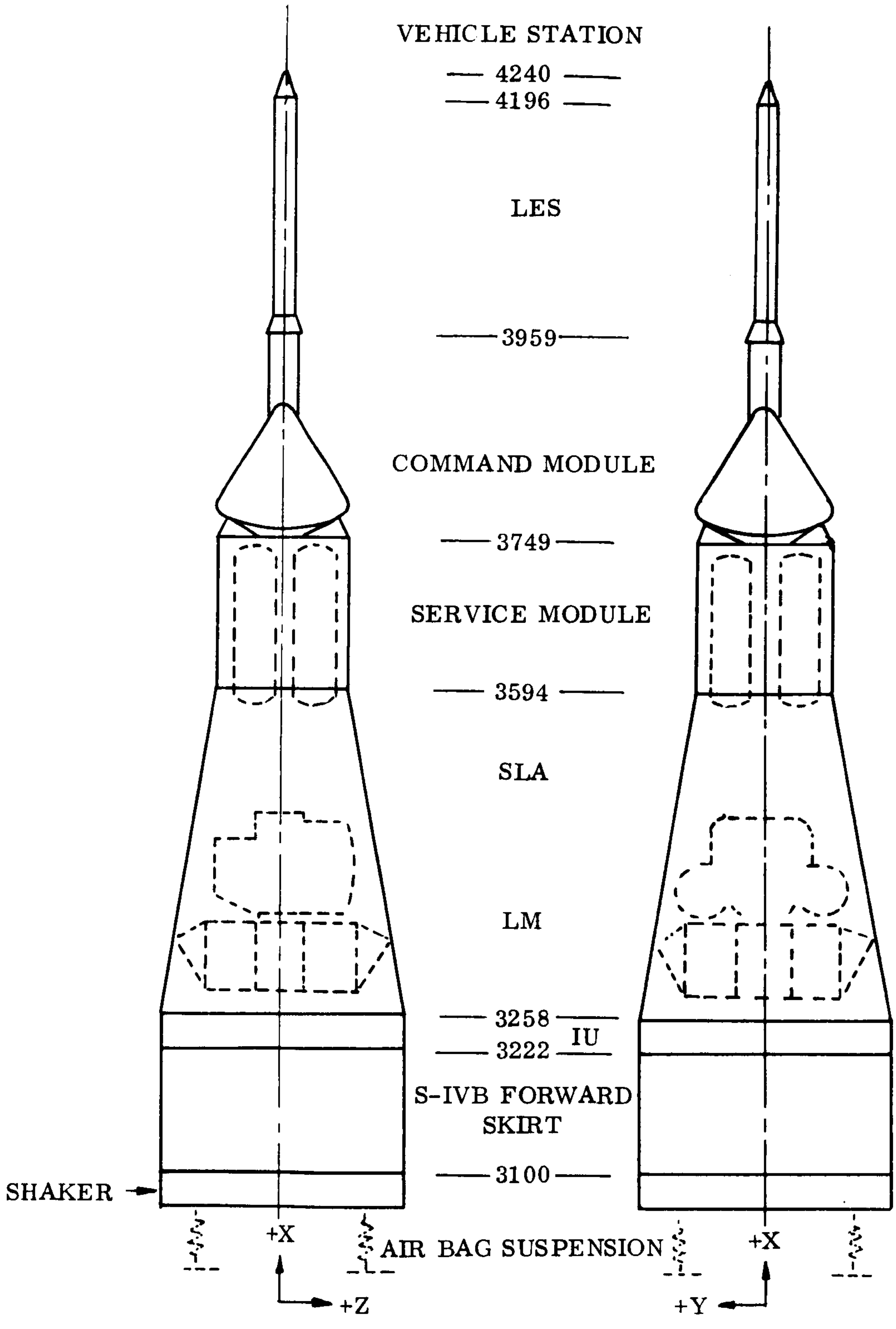


FIGURE 11.2-2 SHORT STACK CONFIGURATION

11.2.3 (Continued)

minimum development and/or verification tests considered necessary to provide assurance that the advanced stage as modified to incorporate J-2S engines and added mission capability as directed by NASA, has retained a man-rated status and will, therefore, require no development flight tests.

Design Certification Reviews (DCR's) as well as other reviews such as PFR and FRR may be conducted to evaluate the design maturity and determine the degree of risk involved to manrate the first flight. In the event the first flight is not certified as manrated, it will be utilized as an unmanned useful payload mission such as a deep space probe or a synchronous orbit mission. Unmanned missions are usually common and necessary to every program to gain knowledge and confidence in new systems introduced for new space probes. Therefore, no cost for manrating flight qualification testing is considered in the resource study.

11.3 VEHICLE SCHEDULE PLAN

The integrated vehicle master development and delivery schedule plan is presented in this section. The schedule data are developed and provided individually by the stage contractors. These data are integrated to form a master schedule plan. The master schedule demonstrates flow time and lead time relationships among milestones such as design and facility modifications, stage fabrication and stage delivery.

11.3.1 Schedule Ground Rules

The schedule is based on a low production rate of two vehicles per year with six months intervals between launches. The delivery plan requires that all the stages are delivered on dock KSC at the same time as indicated in Figure 11.3-1. On Dock KSC was taken as a commitment date from which the schedule is stretched backward to determine the authority to proceed (ATP).

Schedules as based on time from ATP to delivery on Dock KSC rather than on calendar year. The ATP varies from stage to stage. The ATP for the S-IC and the S-II stages is 42 months prior to delivery on Dock KSC while periods of 35 and 18 months, respectively, are required for the S-IVB and IU stages.

A minimum of six months effort is assigned to phase "C" which is a program definition phase. Phase "C" proceeds the ATP. During phase "C", the Contract End Item (CEI) specification is defined; equipment requirement specification functional plan and specification control drawings are revised and prepared. All efforts are complete to a level sufficient for contract definition and cost and schedule estimating.

11.3.1 (Continued)

The schedule is based on a one-shift, five-day week for manufacturing and engineering; it is also based on an assumed delivery of two Saturn V vehicles per year from the AS-515 and on; so the "on hand capabilities" is assumed to be maintained at the same level going with the J-2S program (AS-518 and on).

11.3.2 Schedule

Figure 11.3-1 shows the integrated vehicle master development and delivery schedule. The schedule is common for the LOR, synchronous and LEO mission except for the IU stage and SE&I where the synchronous mission requires extra lead time effort, as further discussed below.

Phase "C" is not required for the S-IC stage. All program definition efforts for the stage are complete. J-2S and/or mission profile impose no design changes or modifications to the S-IC; however, the two stage polar orbit mission profile loads cause an overload condition in the S-IC fuel tank. Insignification modifications to manufacturing operation (new mill tape) are required; this will not alter the S-IC schedule.

The S-II stage Phase "C" include the initiation of raw material for the "Mini-Thrust Structure" test article to permit earliest possible fabrication and testing. Although testing of the Mini-Thrust Structure will not be completed until after completion of the S-II-18 thrust structure, this is not considered to be a risk situation.

Mission peculiarity has no effect on the development and deliveries schedule except for the IU stage where the synchronous mission program required an extra four months for design and certification testing of the Hi-density Core Structure Segment and design of Hi-energy Cell Battery. For Systems Engineering and Integration (SE&I), an extra 6 months is required for development for the LOR and LEO missions, and an extra 12 months is required for development for the synchronous orbit mission. This extra effort for the IU stage and the SE&I applies only to the first launch, subsequent launches resume the normal schedule as illustrated.

11.3.3 Schedule Impact

The overall schedule, as illustrated, is straightforward. The stage schedules are integrated without any activity or event that poses critical phasing or schedule problem. The ATP varies from stage to stage (42 months prior to on Dock KSC for S-IC and S-II, 36 months for the S-IVB and 18 months for the IU stage). This variation is considered favorable for budgetary purposes where cost may spread out over a longer period.

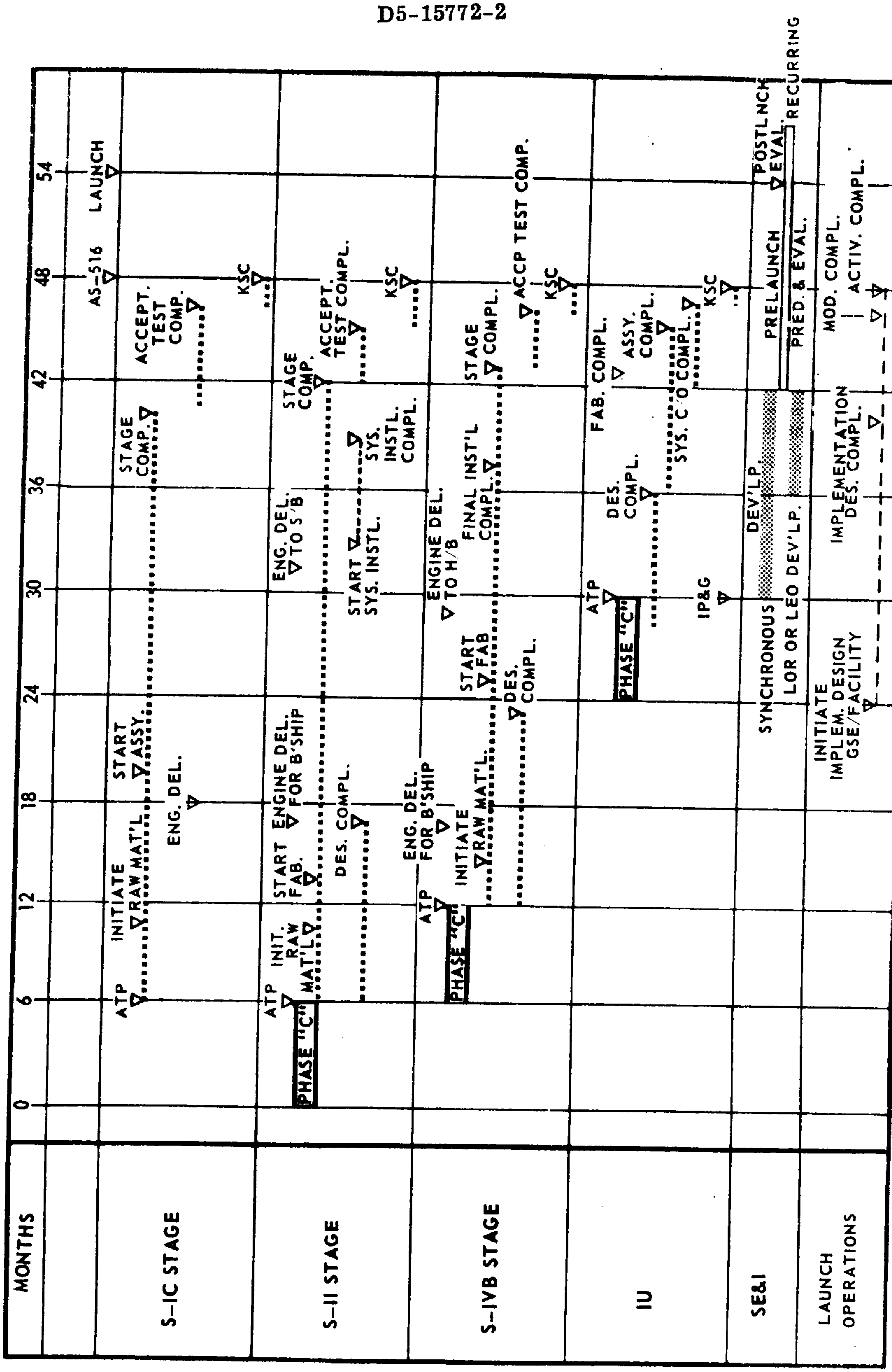


FIGURE 11.3-1 INTEGRATED VEHICLE MASTER DEVELOPMENT AND DELIVERY SCHEDULE

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SECTION 12

WIND SENSITIVITY STUDY

12.0 INTRODUCTION

The four J-2S/Saturn V design vehicles used the Saturn V external geometry with an Apollo Spacecraft payload shape. This shape provides adequate payload volume for the LOR, Synchronous Orbit, Low-Earth Orbit, and Polar Orbit missions. However, other missions, or future experiments, may have greater payload volume requirements.

A candidate payload shape to provide greater payload volume could be a 260-inch diameter cylinder with a MLV nose cone. To determine the effect of this shape on the Saturn V vehicle, a wind sensitivity study was conducted to define the relationship between design wind speed, payload length, payload weight, and design wind exceedence percentile. The wind exceedence percentile numbers (95 percent, 90 percent, 80 percent design winds, etc.) are the percentage of time during which a given wind speed magnitude will not be exceeded within a particular period. The relationship between exceedence percentile values and wind speed is shown in Table 12-I. If the given peak wind speed magnitude is determined to be the maximum allowable within the structural capability of the vehicle, then the exceedence percentile number may be taken to be launch availability. This means that the vehicle may be launched the same percentage of the time that the maximum allowable wind speed is not exceeded.

J-2S/Saturn V structural design is based upon wind criteria formulated for March, the month with the highest probable wind speed. For other months, the design wind criteria will be less severe and the structural loading will be reduced. The payload length can be increased by launching during a month with less severe design wind criteria, by reducing the specified structural factor of safety, and/or by making structural modifications to the critical stations. The critical station is that station most likely to fail with increased loads.

12.1 STUDY GROUND RULES

12.1.1 Payload Envelopes

The payload envelopes for the basic 260-inch diameter MLV payload shape were chosen to encompass the full range of allowable payload lengths expected for month-to-month peak wind speed variations, for changes in launch availability and for two factors of safety. Figure 12-1 shows the vehicle configuration and the four payload envelopes used in the study. Payload weights of 109,200 pounds and 65,600 pounds were considered. These weights are representative of LOR and synchronous orbit mission payloads. The payload weights were uniformly distributed throughout the payload envelope. The relationship between payload length, payload volume, and payload density is shown in Figure 12-2.

TABLE 12-I IDEALIZED MONTHLY WIND SPEED FOR EXCEEDENCE PERCENTILE VALUES; Cape Kennedy, Florida (Reference: R-AERO-Y-73-65)

10-14 km Altitude

Percentile Exceedence	50%	80%	90%	95%	99%
			WIND SPEED		
Month	m/s	m/s	m/s	m/s	m/s
Jan.	45.0	58.0	66.0	72.0	84.0
Feb.	41.0	58.0	68.0	75.0	93.0
Mar.	47.0	60.0	68.0	75.0	97.0
Apr.	40.0	54.0	61.0	66.0	78.0
May	23.0	36.0	44.0	50.0	58.0
June	13.0	22.0	28.0	34.0	42.0
July	10.0	15.5	19.0	23.0	31.0
Aug.	10.0	15.0	19.0	22.0	28.0
Sept.	11.0	19.0	23.0	26.0	30.0
Oct.	22.0	34.0	41.0	47.0	58.0
Nov.	30.0	40.0	47.0	53.0	79.0
Dec.	36.0	50.0	57.0	63.0	75.0

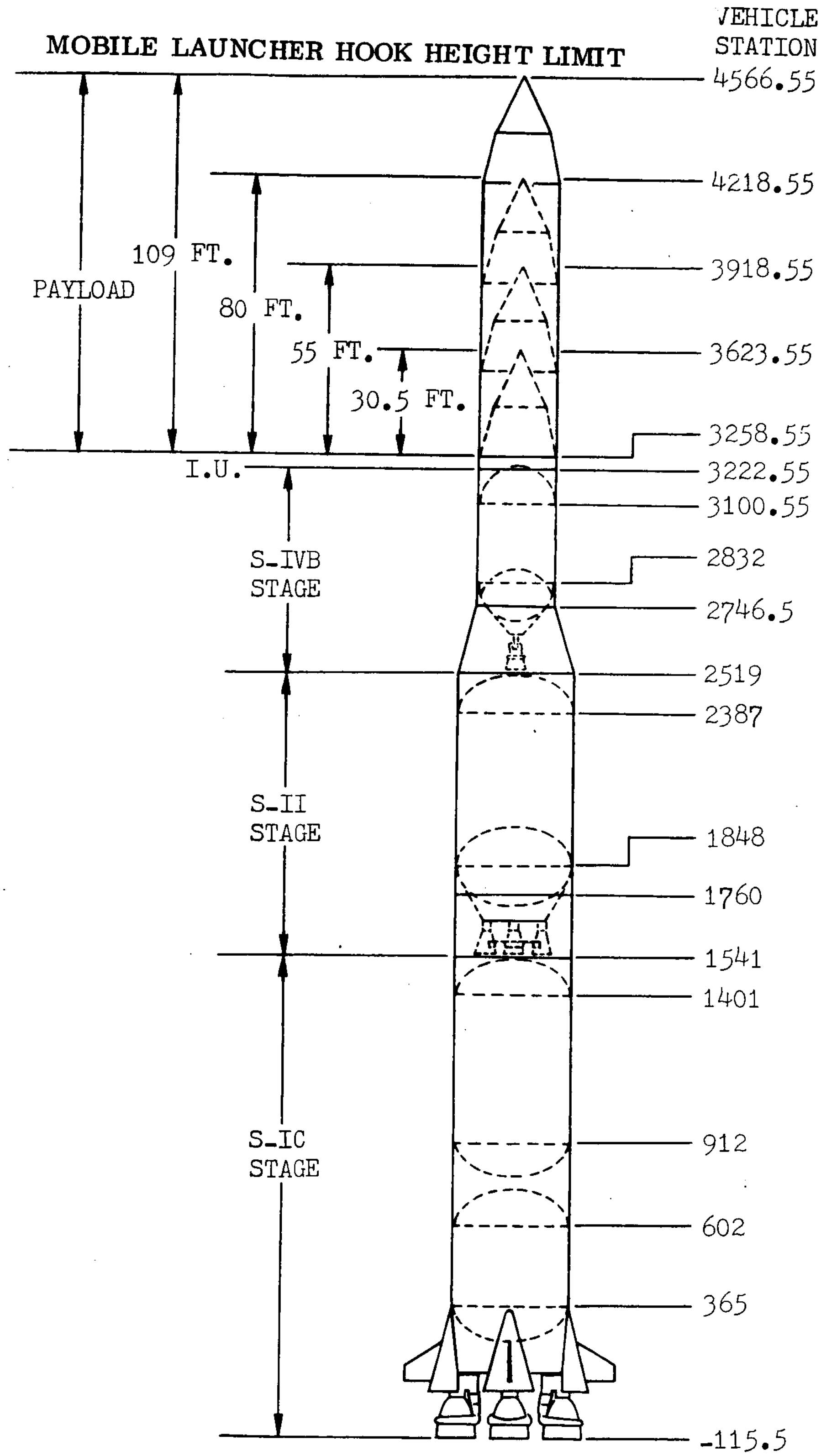

FIGURE 12-1 THREE STAGE J-2S/SATURN V VEHICLE WITH 260-INCH DIAMETER PAYLOAD ENVELOPES

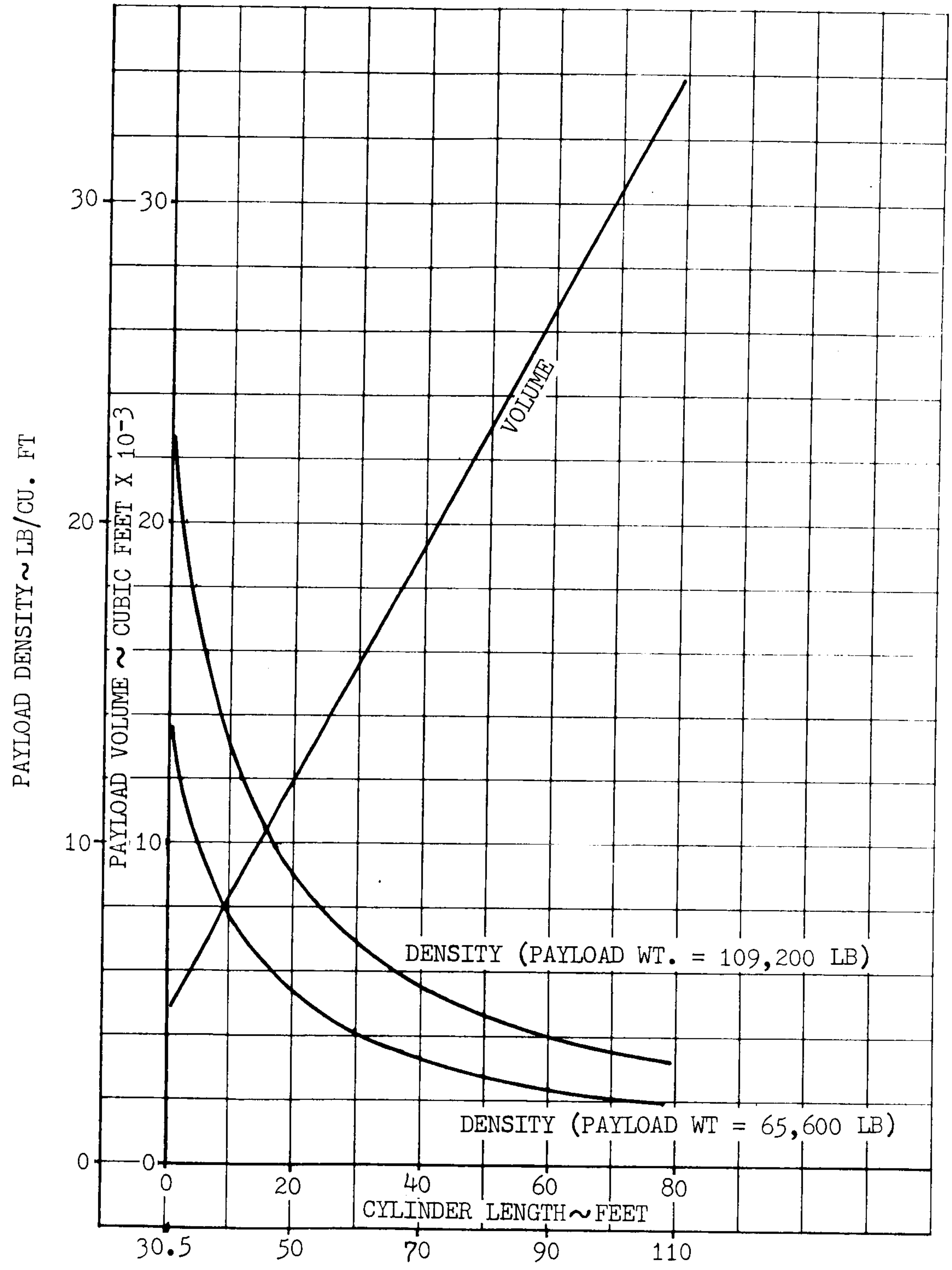

FIGURE 12-2 J-2S/SATURN V PAYLOAD DENSITY AND PAYLOAD VOLUME VERSUS PAYLOAD LENGTH

12.1.2 Structural Capability

The compressive structural capability for the S-IC stage, the S-II stage, the S-IVB stage, and the Instrument Unit were taken from References 12-1 through 12-5

12.1.3 Wind Criteria

The inflight wind profiles used for this study were taken from References 12-6. Superimposed upon each wind profile is an embedded jet gust.

12.1.4 Aerodynamics

The normal force coefficient distributions for the 260-inch diameter payload cylinder and Modified Launch Vehicle (MLV) nose cone were taken from MSFC Memorandum R-AERO-AD-68-44, "Static Aerodynamic Characteristics of the Saturn V + J-2S Vehicle," dated July 18, 1968.

12.2 TECHNICAL APPROACH

Flight simulations were performed for the two payload weights with the payload lengths shown in Figure 12-1. The flexible body responses during first stage boost were obtained. Flight simulation included rigid body translation and rotation in the yaw plane, one free-free bending mode, and two nozzle degrees of freedom. The wind profiles were applied in the yaw plane.

Bending moment values at the vehicle stations were obtained as functions of payload length and wind speed. Using factors of safety of 1.25 and 1.4, allowable bending moments for each vehicle station were determined using the following formula:

$$BM(X) = \frac{R(x)^2 \pi}{F.S.}\left[N_{CAP} + \frac{Pu_{min} \quad R(x)}{2}\right] - \left[\pi R(x)^2\right]\left[\frac{P(x)}{2\pi \ R(x)}\right]$$

Where: $BM(X)$ = allowable bending moment

$R(X)$ = distributed body radius

$F.S.$ = structural factor of safety

Pu_{min} = minimum ullage pressure (applicable to tank shells only)

$P(X)$ = distributed longitudinal force including aerodynamic forebody drag

N_{CAP} = compressive structural capability

Allowable wind speed as a function of payload length was determined for all vehicle stations and the most critical stations were determined. Then, relating wind speed to launch availability for each month, allowable payload lengths were determined for each month of the year for launch availabilities of 95 percent, 90 percent, 80 percent and 50 percent for both payload weights. In addition, allowable payload lengths for each month of the year were determined for several levels of structural modifications.

12.3 RESULTS AND DISCUSSION

Both manned and unmanned missions (factors of safety of 1.4 and 1.25, respectively) were considered. Results were obtained for two payload weights so that the effect of payload weight could be assessed. In addition to determining the maximum allowable payload envelopes without structural modifications, payload length gains resulting from successive structural modifications to critical stations were obtained.

Three levels of structural modifications were considered: 1) no structural modifications; 2) structural modifications to the Instrument Unit; and 3) structural modifications to the Instrument Unit, the S-IVB forward skirt, the S-II/S-IVB interstage, and the S-II forward skirt. Structural modifications to the S-IVB LH_2 tank shell were not considered since this modification would be significant from a cost standpoint and rather than modify the entire stage, other structural designs and concepts should be investigated.

The relationship between allowable payload length, launch availability, and month of year for an unmodified J-2S/Saturn V are shown in Figures 12-3 through 12-6. The effect of structural modifications on the payload length is shown in Figures 12-7 and 12-8. Modification to the S-II forward skirt is not required with the 65,600 pound payload with a safety factor of 1.25 because the aft Y-ring of the S-IVB LH_2 tank is more critical than either of the S-II forward skirt stations thereby eliminating them as critical design stations.

A summary of obtainable payload lengths for both payload weights and for manned and unmanned factors of safety is presented in Table 12-II. This table contains data for 95 percent launch availability in the month of March (75 meter per second design wind speed) and shows the gains in allowable payload length which can result from modifying the critical vehicle stations.

The detailed results of this wind sensitivity study are contained in Document D5-15772-6, "Structural Analysis - J-2S Improvement Study".

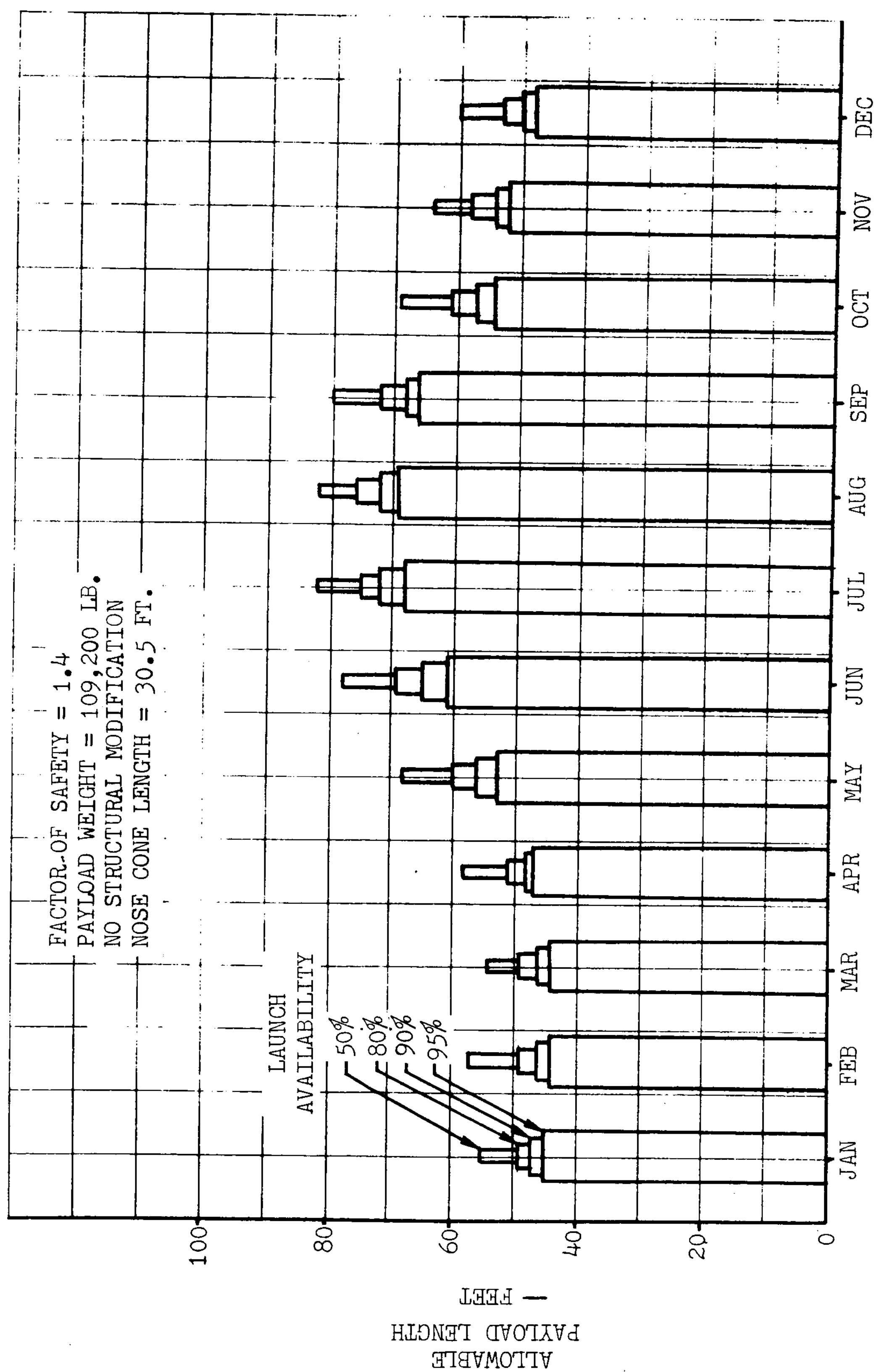

FIGURE 12-3 J-2S/SATURN V LOR ALLOWABLE PAYLOAD LENGTH VERSUS MONTH OF YEAR FOR VARIOUS LAUNCH AVAILABILITIES

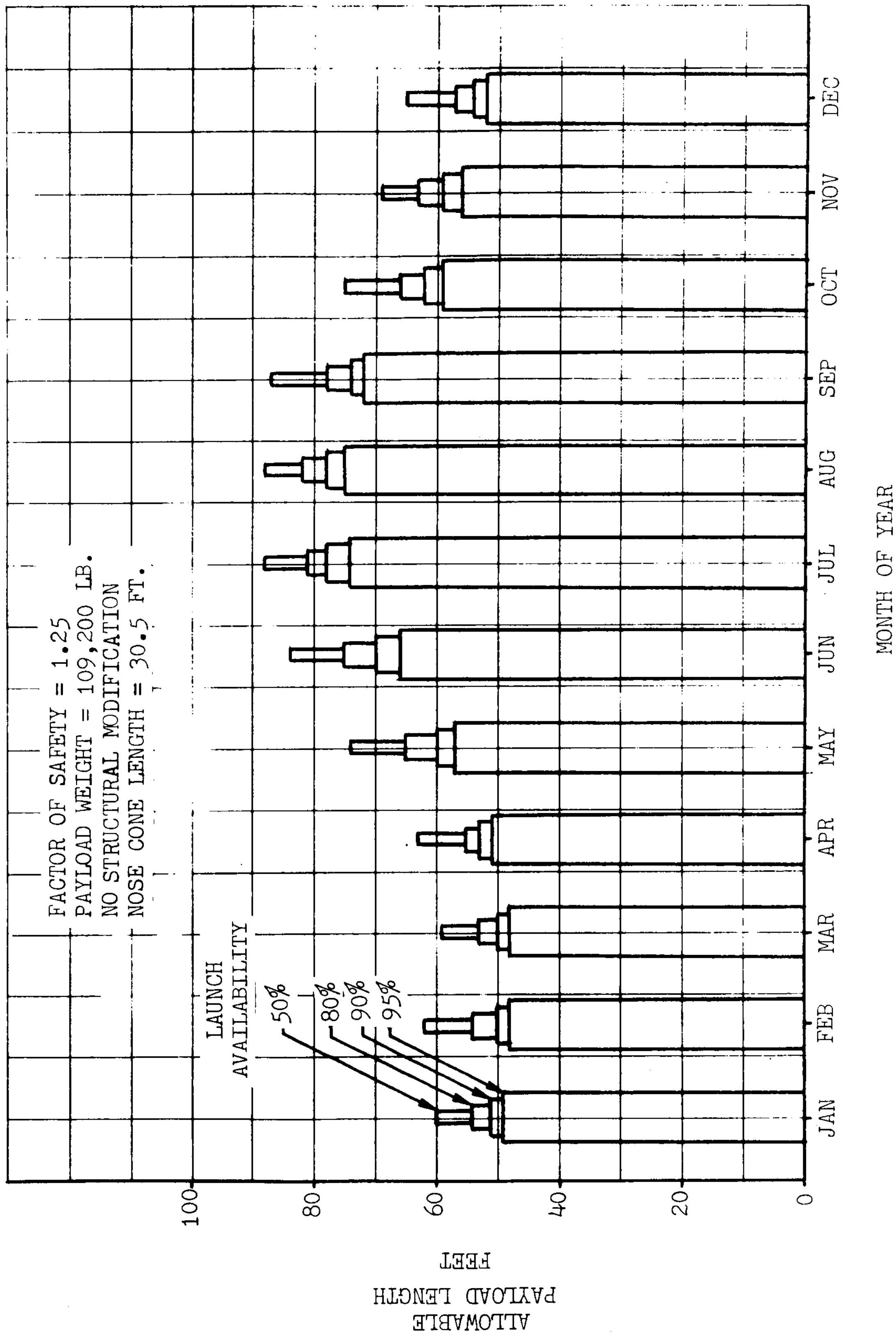

FIGURE 12-4 J-2S/SATURN V LOR ALLOWABLE PAYLOAD LENGTH VERSUS MONTH OF YEAR FOR VARIOUS LAUNCH AVAILABILITIES

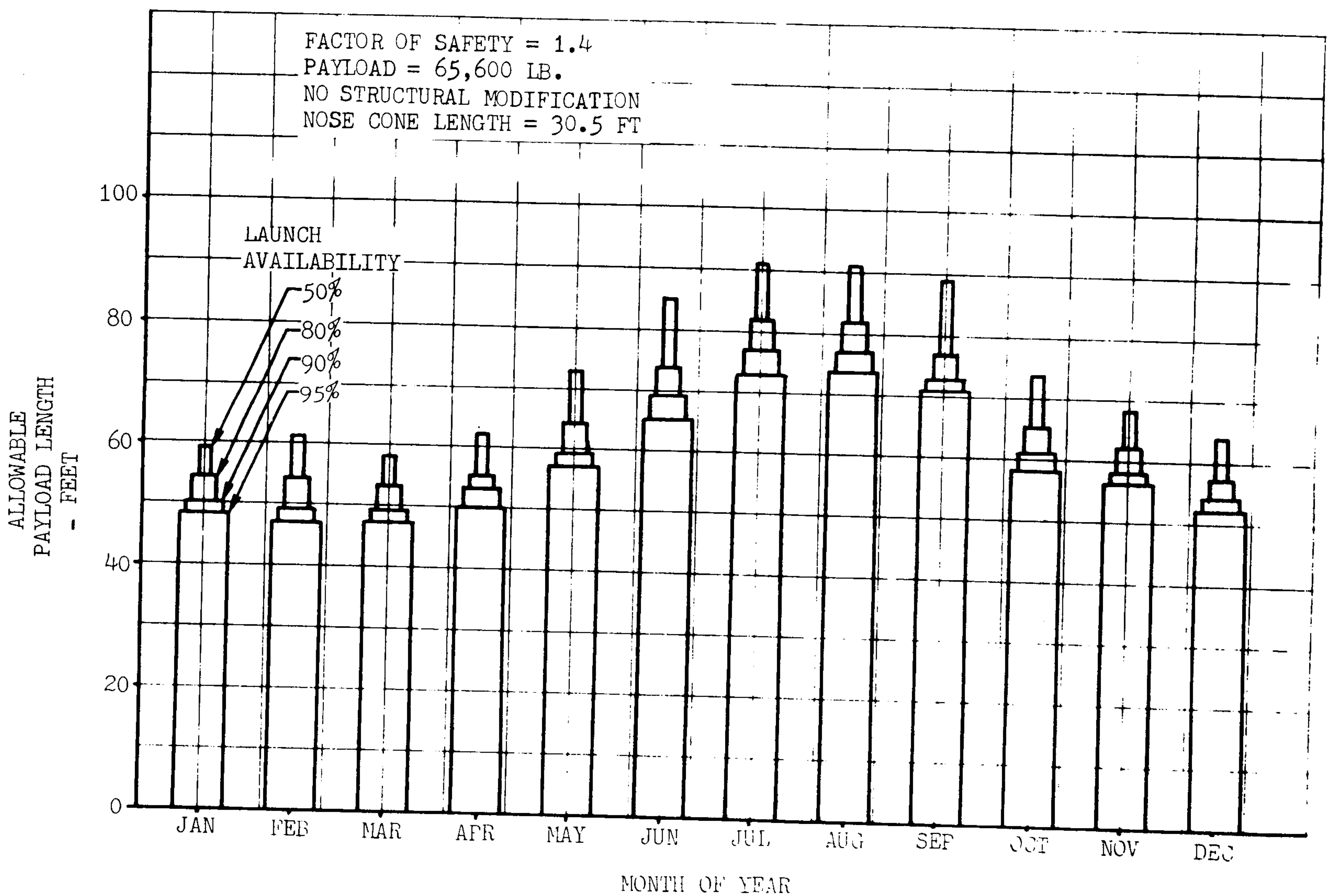

FIGURE 12-5 J-2S/SATURN V SYN ORBIT VEHICLE ALLOWABLE PAYLOAD LENGTH VERSUS MONTH OF YEAR FOR VARIOUS LAUNCH AVAILABILITIES

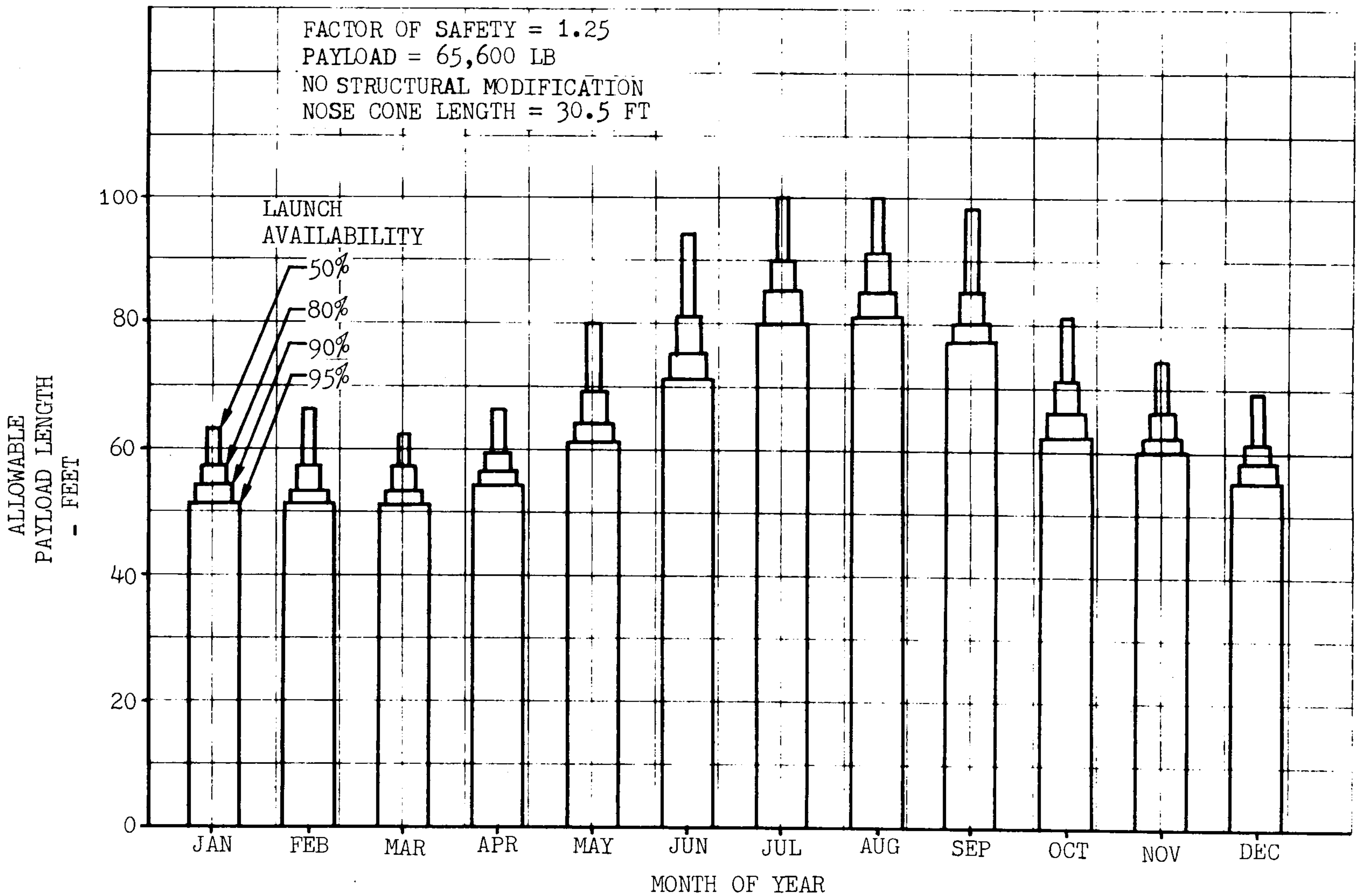

FIGURE 12-6 J-2S/SATURN V SYN ORBIT VEHICLE ALLOWABLE PAYLOAD LENGTH VERSUS MONTH OF YEAR FOR VARIOUS LAUNCH AVAILABILITIES

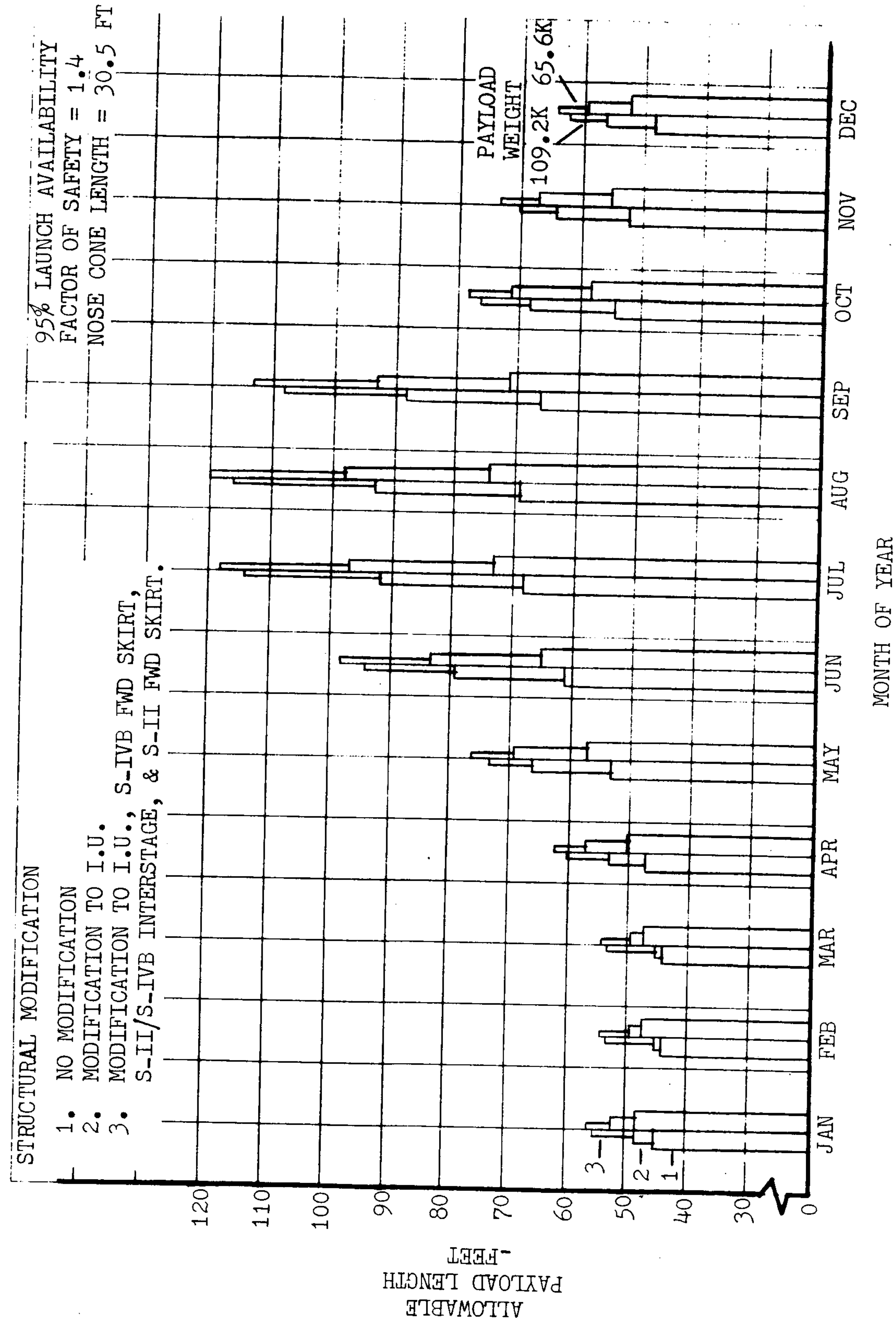

FIGURE 12-7

J-2S/SATURN V ALLOWABLE PAYLOAD LENGTH VERSUS MONTH OF YEAR WITH STRUCTURAL MODIFICATION

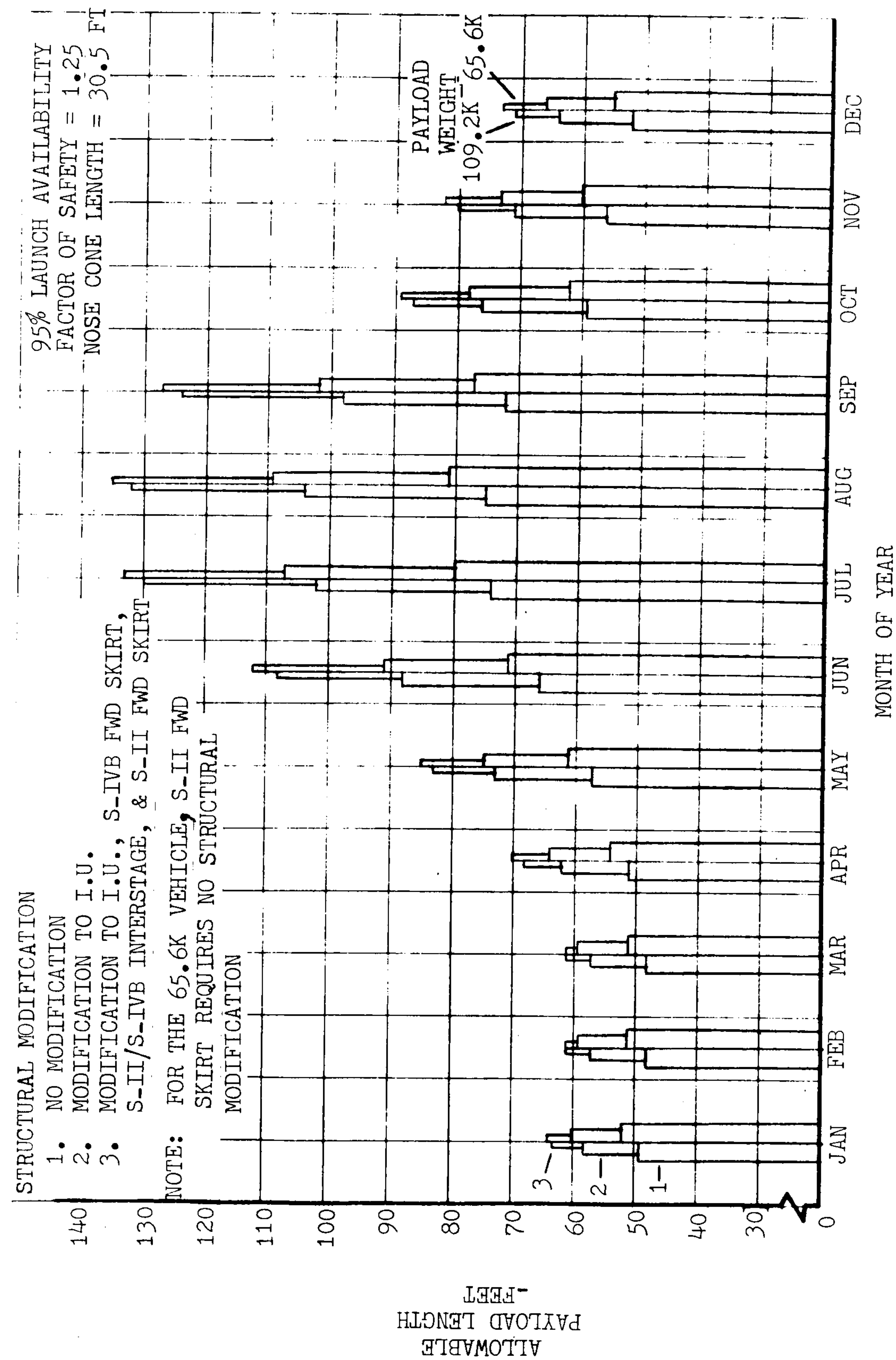

FIGURE 12-8

J-2S/SATURN V ALLOWABLE PAYLOAD LENGTH VERSUS MONTH OF YEAR WITH STRUCTURAL MODIFICATION

TABLE 12-II

SUMMARY OF OBTAINABLE PAYLOAD LENGTHS FOR J-2S/SATURN V
WITH 95% LAUNCH AVAILABILITY
FOR MONTH OF MARCH

PAYLOAD WEIGHT (LB.)	FACTOR OF SAFETY	ALLOWABLE PAYLOAD LENGTH FOR (1) (FT.)	ALLOWABLE PAYLOAD LENGTH FOR (2) (FT.)	ALLOWABLE PAYLOAD LENGTH FOR (3) (FT.)
109,200	1.4	44	45	53
109,200	1.25	48	57	61
65,600	1.4	47	49	54
65,600	1.25	51	59	61

PAYLOAD WEIGHT (LB.)	FACTOR OF SAFETY	GAIN IN PAYLOAD FOR (1) TO (2) (FT.)	GAIN IN PAYLOAD FOR (1) TO (3) (FT.)	GAIN IN PAYLOAD FOR (2) TO (3) (FT.)
109,200	1.4	1	9	8
109,200	1.25	9	13	4
65,600	1.4	2	7	5
65,600	1.25	8	10	2

(1) NO STRUCTURAL MODIFICATION

(2) STRUCTURAL MODIFICATION TO I.U.

(3) STRUCTURAL MODIFICATION TO I.U., S-IVB FWD SKIRT, S-II/S-IVB INTERSTAGE, AND S-II FWD SKIRT.

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APPENDIX A

BASELINE SA-511 VEHICLE DEFINITION

This appendix contains the Baseline SA-511 vehicle definition for a LOR mission as provided by MSFC. The data includes payload capability, vehicle environments, and stage descriptions to provide a base for necessary vehicle changes when J-2S engines are incorporated in the S-II and S-IVB stages.

J-2S IMPROVEMENT STUDY

BASELINE LAUNCH VEHICLE SA-511

SEPTEMBER 1968

PROPULSION DIVISION

PROPULSION AND VEHICLE ENGINEERING LABORATORY
GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

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SECTION I

INTRODUCTION

This document has been prepared to provide a baseline launch vehicle to which all changes resulting from the J-2S Improvement Study will be referenced. This baseline launch vehicle, designated SA-511, is an estimated projection of the current operational Saturn V vehicle. The baseline launch vehicle data as contained herein reflect currently available information. Where specific technical data have not yet been generated, reasonable values have been assigned. The baseline launch vehicle is to be used only for comparison purposes in conjunction with the J-2S Improvement Study and does not necessarily represent approved changes to the vehicle. Any redefinition or change to the baseline launch vehicle during the course of the J-2S Improvement Study will be provided by the Propulsion Division (R-P&VE -P) and will be published as an amendment to this document.

Section II contains the general description of the reference launch vehicle and its nominal flight trajectory. Also provided are weight summaries, mass characteristics and other pertinent design data.

Section III contains descriptions of the baseline S-IC, S-II, and S-IVB stages and the Instrument Unit.

SECTION II. BASELINE VEHICLE DESCRIPTION

1.0 CONFIGURATION DESCRIPTION

The Saturn V SA-511 launch vehicle consists of an S-IC stage utilizing five 1.522 million pound thrust (sea level) LOX/RP-1 F-1 engines, an S-II stage powered by five 230,000 pound-vacuum thrust (5.5 M.R.) LOX/LH_2 J-2 engines, an S-IVB stage utilizing one of the above J-2 engines, an Instrument Unit, and an Apollo-type payload as shown in Figure 1.

2.0 WEIGHT SUMMARY AND MASS CHARACTERISTICS

Assigned weights for the S-IC stage, the S-IC/S-II interstage, the S-II stage, the S-II/S-IVB interstage, the S-IVB stage, and the Instrument Unit are given in Tables I through VI. Residual weight summary for the first, second and third stage is given in Table VII.

Table VIII gives the vehicle mass characteristics summary for the flight. Mass properties versus time (mass, C. G. moment of inertia) during first, second, and third stage flight are given in Table IX.

3.0 TRAJECTORY

The baseline trajectory for vehicle SA-511 is applicable to the Saturn V Apollo geometry configuration flown to the standard LOR flight profile. All stages are filled to capacity and the first stage liftoff weight and "pitch" profile optimized to yield the maximum payload at translunar injection. The net payload forward of the instrument unit (I. U.) at TLI is 100,078 lb. A 72° launch azimuth measured from north toward east was used.

Table X is the baseline vehicle weight description of SA-511 as used in the trajectory calculations.

Table XI displays the baseline propulsion data.

Table XII is a time history of sequence of events from liftoff to final injection.

Figures 2 and 3 display the forebody axial force coefficient vs Mach number and the variation of average base pressure with altitude, respectively. Figure 4 displays the normal force ($C_{N\alpha}$) vs Mach number for the Apollo geometry configuration.

Table XIII contains the detailed LOR trajectory data and Table XIV gives the definitions and symbols used in the trajectory data of Table XIII.

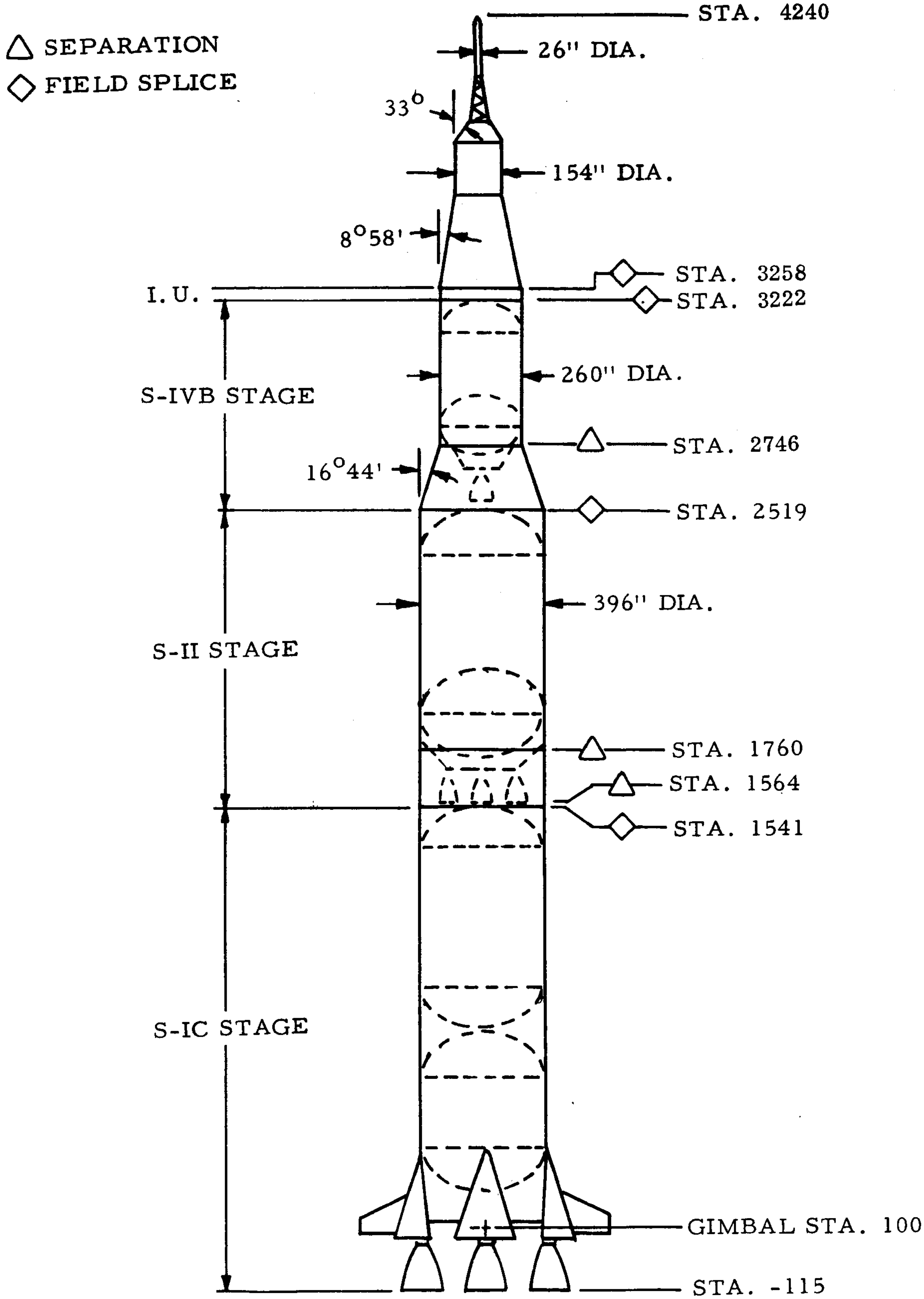

FIGURE 1. SATURN V SA-511 VEHICLE CONFIGURATION - THREE STAGE

R-P&VE-P

TABLE I
Assigned Weight, S-IC Stage (SA-511)

Description	Weight (lb)	
Stage Structure		140,835
Structural Fuel Container	22,334	
Structural Oxidizer Container	35,404	
Structure Forward of Tanks (Extension of Container, Sides, Transitions, etc.)	5,638	
Structure Between Tanks (Cluster of Tandem Configuration)	13,173	
Thrust Structure	47,503	
Fairings and Associated Structure (Includes Insulation)	9,024	
Non-Moveable Aerodynamic Control Surfaces	2,034	
Base Heat Protection	5,259	
Paint and Sealer	466	
Propulsion System and Accessories		139,525
Liquid Rocket Engine and Accessories	93,423	
Fuel System	13,508	
Oxidizer System	23,872	
Stage Control System	8,722	
Equipment and Instrumentation		8,618
Structure (for Equipment and Instrumentation)	225	
Environmental Control System - Equipment	304	
Guidance System	63	
Control System Electronics	67	
Telemetering and Measuring Equipment	3,608	
Electrical System	907	
Range Safety Equipment	505	
Separation System	2,492	
Pneumatic System	447	
Design Uncertainties (Growth)		431
TOTAL DRY WEIGHT		289,409

Table II

Assigned Weight, S-IC/S-II Interstage (SA-511)

Description	Weight (lb)	
Short Interstage		
Structure Stage		1,481
Fairings and Associated Structure (Includes Insulation)	55	
Interstage (Spacer) Vehicle Instrument Unit Structure	1,416	
Paint and Sealer	10	
Equipment and Instrumentation		67
Separation System	57	
Systems for Total Vehicle or Flight Stage	10	
SHORT INTERSTAGE AT GROUND LIFT-OFF		1,548
Long Interstage		
Structure Stage		8,170
Fairings and Associated Structure (Includes Insulation)	152	
Interstage (Spacer) Vehicle Instrument Unit Structure	7,878	
Paint and Sealer	140	
Propulsion System and Accessories		159
Fuel System	159	
Equipment and Instrumentation		1,106
Telemetry & Measuring Equipment	130	
Separation System	85	
Ullage System	834	
Systems for Total Vehicle or Flight Stage	57	
LONG INTERSTAGE AT GROUND LIFT-OFF		9,435
TOTAL DRY WEIGHT OF S-IC/S-II INTERSTAGE (SHORT & LONG)		10,983

TABLE III

Assigned Weight, S-II Stage (SA-511)

Description	Weight (lb)	
Structure Stage		47,064
Structural Propellant Container (Appl. with Common Bulkhead)	29,179	
Structure Forward of Tanks (Extension of Container Sides, Transitions, Etc.)	4,057	
Structure Aft of Tanks (From Container to Thrust Frame)	4,130	
Thrust Structure	7,302	
Fairings and Associated Structure	1,432	
Base Heat Protection	630	
Paint and Sealer	334	
Propulsion System and Accessories		26,439
Liquid Rocket Engine and Accessories	18,030	
Purge System for Upper Stage Chilldown	386	
Fuel System	4,076	
Oxidizer System	2,824	
Stage Control System Hardware	1,123	
Equipment and Instrumentation		6,765
Environmental Control System - Equipment	1,121	
Telemetering and Measuring Equipment	2,734	
Propellant Utilization System	634	
Electrical System	949	
Range Safety	319	
Pneumatic System	410	
Separation System	117	
Systems for Total Vehicle or Flight Stage	481	
Design Uncertainties (Growth)		418
TOTAL DRY WEIGHT		80,686

TABLE IV

Assigned Weight S-II/S-IVB Interstage (SA-511)

Description	Weight (lb)	
Structure Stage		6,249
Interstage (Spacer) Vehicle Instrument Unit Structure	5,678	
Paint and Sealer	49	
Heat & Flame Protection	522	
Equipment and Instrumentation		772
Environmental Control System - Equipment	17	
Telemetering and Measuring Equipment	15	
Range Safety Equipment	3	
Separation System	727	
System for Total Vehicle	10	
TOTAL DRY WEIGHT OF INTERSTAGE		7,021

TABLE V

Assigned Weight, S-IVB Stage (SA-511)

Description	Weight (lb)	
Structure Stage		13,241
Structural Propellant Container	8,925	
Structure Forward of Tanks (Extension of Container Sides, Transitions, Etc.)	1,242	
Structure Aft of Tanks. (From Container to Thrust Frame or Equivalent)	1,816	
Thrust Structure	774	
Fairings and Associated Structure (Includes Insulation)	196	
Paint and Sealer	104	
Ablative Insulation Thermolag	184	
Propulsion System and Accessories		7,263
Liquid Rocket Engine and Accessories	3,562	
Purge System for Upper Stage Childown	276	
Fuel System	1,567	
Oxidizer System	1,261	
Stage Control System Hardware	284	
Crogenic Repress System	313	
Equipment and Instrumentation		4,580
Structure (for Equipment and Instrumentation)	428	
Environmental Control System (Equipment)	232	
Control System Electronics	113	
Telemetering and Measuring Equipment	1,156	
Propellant Utilization System	175	
Electrical System	830	
Range Safety Equipment	69	
Pneumatic System	302	
Auxiliary Propulsion System	854	
Separation System	119	
Systems for Total Vehicle of Flight Stage	91	
Ullage System	211	
TOTAL DRY WEIGHT		25,084

TABLE VI

Assigned Weight, Instrument Unit (SA-511)

Description	Weight (lb)	
Structure Stage		621
Interstage (Spacer) Vehicle Instrument Unit Structure	613	
Paint and Sealer	8	
Equipment and Instrumentation		3,266
Structure (for Equipment and Instrumentation)	29	
Environmental Control System - Equipment	748	
Guidance System	624	
Control System Electronics	175	
Tracking, Navigation, and Observation Equipment	22	
Telemetering and Measuring Equipment	294	
Electrical System	1,307	
Command System Equipment	67	
Service Items		296
Press. Gases + Thermal Control Fluids	296	
TOTAL WEIGHT AT GROUND IGNITION		*4,183

* As of March, 1968

TABLE VII

Saturn V Launch Vehicle
Residual Summary

Description	Weight (Lbs.) S-IC Stage	S-II Stage	S-IVB Stage
Dry Weight	289,409	80,686	25,084
LOX in Tank	1,861	4,210	2,996
LOX Below Tank	25,737	1,764	369
LOX Pressurization Gas	5,451	3,294	347
Fuel in Tank	17,119	3,149	1,240
Fuel Below Tank	13,367	282	45
Fuel Pressurization Gas	512	1,330	553
Helium in Bottle	188		189
APS Propellant			123
Aft Frame and Detonation Package			-51
Ullage Rocket Cases			-130
Service Items	3,350	105	62
TOTAL AT SEPARATION	356,994	94,820	30,827

TABLE VIII
VEHICLE MASS CHARACTERISTICS SUMMARY

DESCRIPTION	WEIGHT LBS	X CG IN	Y CG IN	Z CG IN	INERTIA KG M SEC2 ROLL	PITCH	YAW
1ST FLT STG AT IGN	6506978.	1193.3	0.15	-0.12	378601.	93511439.	93502551.
S-IC THRUST BUILDUP	-100538.						
1ST FLT STG AT LO	6406440.	1190.8	0.15	-0.12	378873.	93347175.	93338299.
S-IC FROST	-650.	1157.0	0.00	0.00	380.	95.	95.
S-IC MAINSTAGE PROPELLANT	-4577113.						
S-IC N2 PURGE	-37.	259.0	-161.50	68.50	0.	0.	0.
S-IC INBD ENGINE T.D. PROP	-1915.						
S-II INSULATION PURGE GAS	-120.	2093.5	0.00	0.00	0.	1.	1.
S-II FROST	-450.	1903.7	0.00	0.00	263.	65.	65.
S-IVB FROST	-300.	2966.3	0.00	0.00	151.	129.	129.
S-IC INBD ENG EXPENDED PROP	-408.	55.0	16.60	18.20	20.	34.	36.
1ST FLT STAGE AT S-IC OECOS	1825447.	1831.9	0.47	-0.46	377065.	44572847.	44564213.
S-IC OTBD ENGINE T.D. PROP	-6746.						
S-IC/S-II ULLAGE RKT PROP	-180.	1619.5	0.00	0.00	226.	114.	114.
1ST FLT STAGE AT SIC/SII SEP	1818521.	1837.4	0.47	-0.46	376717.	44144031.	44135391.
S-IC STAGE AT SEPARATION	-356994.	372.9	-0.68	-2.05	273096.	1915116.	1905768.
S-IC/S-II INTERSTAGE SMALL	-1548.	1553.2	1.90	-1.40	1821.	915.	910.
S-IC/S-II ULLAGE RKT PROP	-450.	1619.5	0.00	0.00	566.	287.	287.
2ND FLT STAGE AT S-II IGN.	1459529.	2195.9	0.75	-0.07	101180.	13775361.	13776088.
S-II T.P. PROPELLANT	-1776.						
S-II START TANK	-25.	1652.4	0.00	0.00	1.	0.	0.
S-IC/S-II ULLAGE RKT PROP	-730.	1619.5	0.00	0.00	919.	466.	466.
2ND FLT STAGE 90 PC THRUST	1456998.	2196.4	0.75	-0.07	100259.	13764827.	13765555.
LAUNCH ESCAPE SYSTEM	-8936.	4058.1	-0.10	-0.40	102.	3700.	3700.
S-IC/S-II INTERSTAGE LARGE	-9435.	1655.0	5.20	-5.20	10993.	6522.	6582.
S-II MAINSTAGE & VENTING	-970442.						
2ND FLT STAGE AT S-II C.O.S.	468185.	2793.3	2.25	-0.11	89104.	4600257.	4600875.
S-II T.D. PROPELLANT	-342.						
S-IVB ULLAGE PROPELLANT	-5.	2772.3	0.00	0.00	0.	0.	0.
2ND FLT STG AT SII/SIVB SEP	467838.	2794.0	2.25	-0.11	89104.	4589481.	4590098.

TABLE VIII
VEHICLE MASS CHARACTERISTICS SUMMARY (Cont'd)

DESCRIPTION	WEIGHT LBS	X CG IN	Y CG IN	Z CG IN	INERTIA KG M SEC2 ROLL	PITCH	YAW
2ND FLT STG AT SII/SIVB SEP	467838.	2794.0	2.25	-0.11	89104.	4589481.	4590098.
S-II STAGE AT SEPARATION	-94820.	1881.2	5.83	-3.26	62372.	218649.	219416.
S-II/S-IVB INTERSTAGE-DRY	-7023.	2623.7	3.00	-0.60	5733.	4020.	3939.
S II/S IVB IS PROP	-1060.	2565.4	0.00	0.00	913.	478.	478.
S-IVB AFT FRAME SEP WITH IS	-48.	2746.6	0.00	0.00	23.	11.	11.
S-IVB ULLAGE ROCKET PROP	-90.	2772.3	0.00	0.00	5.	7.	3.
S-IVB DET PACKAGE	-3.	2764.5	0.00	0.00	0.	0.	0.
3RD FLT STG AT 1ST SIVB IGN	364793.	3035.2	1.31	0.72	19974.	1367647.	1367571.
S-IVB ULLAGE ROCKET PROP	-22.	2772.3	0.00	0.00	1.	1.	0.
S-IVB H2 IN START TANK	-4.	2634.3	-22.00	14.60	0.	0.	0.
S-IVB T.B. PROPELLANT	-436.						
3RD FLT STG AT 90 PC THRUST	364331.	3035.4	1.31	0.72	19973.	1367343.	1367267.
S-IVB ULLAGE ROCKET CASES	-130.	2771.7	0.00	0.00	57.	30.	29.
S-IVB MAINSTAGE PROP	-71871.						
S-IVB APS PROP POWER ROLL	-18.	2792.4	0.00	0.00	10.	10.	0.
3RD FLT STG AT 1ST SIVB COS	292312.	3075.6	1.62	0.90	19882.	1277291.	1277213.
S-IVB T.D. PROPELLANT	-107.						
3RD FLT STG AT ST OF COAST	292205.	3075.7	1.62	0.90	19882.	1277165.	1277087.
S-IVB ENG PROP EXPENDED	-40.	2616.9	0.00	0.00	0.	7.	7.
S-IVB H2 & HE VENTED IN ORB	-3281.						
S-IVB O2 VENTED	-138.						
S-IVB APS PROP LOSS IN ORBI	-438.	2792.4	0.00	0.00	257.	256.	2.
S-IVB H2 IN START TANK	-2.	2634.3	-22.00	14.60	0.	0.	0.
S-IVB O2/H2 BURNER	-16.						
3RD FLT STG AT 2ND SIVB IGN	288290.	3076.5	1.63	0.91	19611.	1276334.	1276491.
S-IVB H2 IN START TANK	-4.	2634.3	-22.00	14.60	0.	0.	0.
S-IVB T.B. PROPELLANT	-358.						
3RD FLT STG AT 90 PC THRUST	287918.	3076.7	1.64	0.92	19611.	1275790.	1275952.

TABLE VIII
VEHICLE MASS CHARACTERISTICS SUMMARY (Cont'd)

DESCRIPTION	WEIGHT LBS	X CG IN	Y CG IN	Z CG IN	INERTIA KG M SEC2 ROLL	PITCH	YAW
3RD FLT STG AT 90 PC THRUST	287918.	3076.7	1.64	0.92	19611.	1275780.	1275952.
S-IVB MS PROP 2ND BURN	-152648.						
S-IVB APS PROP POWER ROLL	-34.	2792.4	0.00	0.00	19.	19.	0.
3RD FLT STG AT 2ND SIVB COS	135236.	3389.9	3.40	1.98	19510.	506220.	506369.
S-IVB T.D. PROPELLANT	-93.						
3RD FLT STG AT END 2ND T.D.	135143.	3390.3	3.40	1.98	19510.	505078.	505227.
S-IVB ENG PROP EXPENDED	-40.	2616.9	0.00	0.00	0.	7.	7.
S-IVB APS PROPELLANT	-15.	2792.4	0.00	0.00	8.	8.	0.
SPACECRAFT	-100078.	3565.6	2.40	3.90	9066.	99031.	99357.
VEHICLE INSTRUMENT UNIT	-4183.	3245.0	-4.30	-9.60	1879.	1024.	891.
S-IVB STAGE AT SEP	-30827.	2842.6	7.69	-2.67	8480.	33729.	33714.

TABLE IX
MASS PROPERTIES VERSUS TIME

AS-511

FIRST FLIGHT STAGE

TIME	X STATION	TOTAL LOX	TOTAL FUEL			1
VEHICLE MASS	X STATION	Y STATION	Z STATION	IX (ROLL)	IY (PITCH)	2
VEHICLE MASS	X STATION	Y STATION	Z STATION	IX (ROLL)	IY (PITCH)	3
MASS-LOX TK	LEVEL-LOX	UL MASS-LOX	UL VOL-LOX	LOX DENSITY	IZ (YAW)	4
MASS-FUEL TK	LEVEL-FUEL	UL MASS-FUEL	UL VOL-FUEL	FUEL DENSITY	IZ (YAW)	5
SECONDS	*CALIBERS	LBM	LBM			1
LBM	INCHES	INCHES	INCHES	KG-M-SEC2	KG-M-SEC2	2
KILOGRAMS	METERS	METERS	METERS	KG-M2	KG-M2	3
LBM	STA-INCHES	LBM	CUBIC FEET	LBM/FT3	KG-M-SEC2	4
LBM	STA-INCHES	LBM	CUBIC FEET	LBM/FT3	KG-M2	5

FROM GIMBAL PLANE (STATION 100.000, 1 CALIBER = 1 DIA. = 396.0 IN.

-0.7000000E 01	0.2761186E 01	0.3308365E 07	0.1441397E 07	IGNITION		1
0.6506978E 07	0.1193429E 04	0.1530025E 00	-0.1236633E 00	0.3786013E 06	0.9351144E 08	2
0.2951515E 07	0.3031311E 02	0.3886263E-02	-0.3141047E-02	0.3712809E 07	0.9170337E 09	3
0.3261833E 07	0.1481994E 04	0.4920000E 03	0.1390916E 04	0.7096600E 02	0.9350256E 08	4
0.1431658E 07	0.7019699E 03	0.6400001E 02	0.6777731E 03	0.5010000E 02	0.9169465E 09	5
0.0000000E 00	0.2754778E 01	0.3228657E 07	0.1420567E 07	LIFTOFF		1
0.6406440E 07	0.1190892E 04	0.1581683E 00	-0.1225727E 00	0.3788738E 06	0.9334718E 08	2
0.2905912E 07	0.3024865E 02	0.4017476E-02	-0.3113347E-02	0.3715482E 07	0.9154228E 09	3
0.3180500E 07	0.1460052E 04	0.4920000E 03	0.2529620E 04	0.7095431E 02	0.9333830E 08	4
0.1407118E 07	0.6892017E 03	0.6400001E 02	0.1167592E 04	0.5010000E 02	0.9153358E 09	5
0.2000000E 02	0.2756134E 01	0.2827808E 07	0.1246196E 07			1
0.5831812E 07	0.1191429E 04	0.1736150E 00	-0.1330691E 00	0.3788141E 06	0.9118244E 08	2
0.2645265E 07	0.3026230E 02	0.4409820E-02	-0.3379954E-02	0.3714896E 07	0.8941940E 09	3
0.2779651E 07	0.1376289E 04	0.1109047E 04	0.8160602E 04	0.7092097E 02	0.9117363E 08	4
0.1232747E 07	0.6282573E 03	0.1192560E 03	0.4648046E 04	0.5010000E 02	0.8941076E 09	5
0.4000000E 02	0.2777645E 01	0.2426959E 07	0.1071825E 07			1
0.5257188E 07	0.1199947E 04	0.1924383E 00	-0.1458599E 00	0.3787543E 06	0.8903033E 08	2
0.2384620E 07	0.3047866E 02	0.4887931E-02	-0.3704842E-02	0.3714310E 07	0.8730891E 09	3
0.2378902E 07	0.1296487E 04	0.1726095E 04	0.1379687E 05	0.7088761E 02	0.8902150E 08	4
0.1059376E 07	0.5777882E 03	0.1745120E 03	0.8128500E 04	0.5010000E 02	0.8730024E 09	5
0.6000000E 02	0.2826414E 01	0.2026109E 07	0.8974548E 06			1
0.4682565E 07	0.1219260E 04	0.2158813E 00	-0.1617901E 00	0.3786943E 06	0.8662726E 08	2
0.2123975E 07	0.3096920E 02	0.5483385E-02	-0.4109468E-02	0.3713721E 07	0.8495230E 09	3
0.1977952E 07	0.1216588E 04	0.2343143E 04	0.1943844E 05	0.7085426E 02	0.8661849E 08	4
0.[illegible]4058E 06	0.5279544E 03	0.2297680E 03	0.1[illegible]60895E 05	0.5010000E 02	0.8494370E 09	5

TABLE IX
MASS PROPERTIES VERSUS TIME (Cont'd)

AS-511

0.8000001E 02	0.2913285E 01	0.1625260E 07	0.7230841E 06			1
0.4106540E 07	0.1253661E 04	0.2459668E 00	-0.1822389E 00	0.3778390E 06	0.8362364E 08	2
0.1862695E 07	0.3184299E 02	0.6247556E-02	-0.4628869E-02	0.3705334E 07	0.8200675E 09	3
0.1577103E 07	0.1136547E 04	0.2960191E 04	0.2508534E 05	0.7082090E 02	0.8361488E 08	4
0.7096351E 06	0.4777786E 03	0.2850241E 03	0.1508940E 05	0.5010000E 02	0.8199815E 09	5
0.1000000E 03	0.3058260E 01	0.1224411E 07	0.5487135E 06			1
0.3531915E 07	0.1311071E 04	0.2857561E 00	-0.2092776E 00	0.3777781E 06	0.7949537E 08	2
0.1602049E 07	0.3330120E 02	0.7258204E-02	-0.5315652E-02	0.3704737E 07	0.7795831E 09	3
0.1176254E 07	0.1055736E 04	0.3577239E 04	0.3073755E 05	0.7078755E 02	0.7948660E 08	4
0.5352645E 06	0.4278739E 03	0.3402800E 03	0.1856985E 05	0.5010000E 02	0.7794970E 09	5
0.1200000E 03	0.3294504E 01	0.8235621E 06	0.3743427E 06			1
0.2957291E 07	0.1404623E 04	0.3410083E 00	-0.2468239E 00	0.3777168E 06	0.7323620E 08	2
0.1341404E 07	0.3567743E 02	0.8661611E-02	-0.6269328E-02	0.3704135E 07	0.7182016E 09	3
0.7754051E 06	0.9747653E 03	0.4194287E 04	0.3639510E 05	0.7075419E 02	0.7322753E 08	4
0.3608937E 06	0.3779051E 03	0.3955361E 03	0.2205031E 05	0.5010000E 02	0.7181166E 09	5
0.1400000E 03	0.3688292E 01	0.4227128E 06	0.1999720E 06			1
0.2382667E 07	0.1560553E 04	0.4229106E 00	-0.3024801E 00	0.3776542E 06	0.6295063E 08	2
0.1080759E 07	0.3963831E 02	0.1074192E-01	-0.7682994E-02	0.3703522E 07	0.6173346E 09	3
0.3745558E 06	0.8942452E 03	0.4811334E 04	0.4205798E 05	0.7072084E 02	0.6294192E 08	4
0.1865230E 06	0.3266404E 03	0.4507921E 03	0.2553076E 05	0.5010000E 02	0.6172492E 09	5
0.1494860E 03	0.3976499E 01	0.2264420E 06	0.1172680E 06	I.E.C.O.S.		1
0.2103980E 07	0.1674693E 04	0.4283987E 00	-0.3906896E 00	0.3774548E 06	0.5533450E 08	2
0.9543493E 06	0.4253721E 02	0.1088132E-01	-0.9923514E-02	0.3701566E 07	0.5426460E 09	3
0.1844330E 06	0.8525532E 03	0.5104000E 04	0.4474575E 05	0.7070501E 02	0.5532583E 08	4
0.1038190E 06	0.2987298E 03	0.4770000E 03	0.2718154E 05	0.5010000E 02	0.5425608E 09	5
0.1614860E 03	0.4373786E 01	0.3195700E 05	0.3288100E 05	O.E.C.O.S.		1
0.1825447E 07	0.1832019E 04	0.4770990E 00	-0.4634395E 00	0.3770657E 06	0.4457284E 08	2
0.8280098E 06	0.4653329E 02	0.1211831E-01	-0.1177136E-01	0.3697751E 07	0.4371102E 09	3
0.1861000E 04	0.7792255E 03	0.5443000E 04	0.4732791E 05	0.7068501E 02	0.4456421E 08	4
0.1051500E 05	0.2584758E 03	0.5120001E 03	0.2886425E 05	0.5010000E 02	0.4370255E 09	5
0.1614870E 03	0.4387432E 01	0.2759900E 05	0.3045500E 05	SEPARATION		1
0.1318521E 07	0.1837423E 04	0.4789160E 00	-0.4652046E 00	0.3767171E 06	0.4414403E 08	2
0.8248672E 06	0.4667054E 02	0.1216446E-01	-0.1181619E-01	0.3694332E 07	0.4329049E 09	3
0.1361000E 04	0.7792255E 03	0.5451000E 04	0.4732791E 05	0.7068501E 02	0.4413539E 08	4
0.1711900E 05	0.2567144E 03	0.5120001E 03	0.2891208E 05	0.5010000E 02	0.4328202E 09	5

MASS PROPERTIES VERSUS TIME (Cont'd)


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SECOND FLIGHT STAGE

TIME            X STATION      TOTAL LOX      TOTAL FUEL                                 1
VEHICLE MASS    X STATION      Y STATION      Z STATION      IX (ROLL)      IY (PITCH)   2
VEHICLE MASS    X STATION      Y STATION      Z STATION      IX (ROLL)      IY (PITCH)   3
MASS-LOX TK     LEVEL-LOX      UL MASS-LOX    UL VOL-LOX     LOX DENSITY    IZ (YAW)     4
MASS-FUEL TK    LEVEL-FUEL     UL MASS-FUEL   UL VOL-FUEL    FUEL DENSITY   IZ (YAW)     5

SECONDS         *CALIBERS      LBM            LBM                                        1
LBM             INCHES         INCHES         INCHES         KG-M-SEC2      KG-M-SEC2    2
KILOGRAMS       METERS         METERS         METERS         KG-M2          KG-M2        3
LBM             STA-INCHES     LBM            CUBIC FEET     LBM/FT3        KG-M-SEC2    4
LBM             STA-INCHES     LBM            CUBIC FEET     LBM/FT3        KG-M2        5

*FROM GIMBAL PLANE (STATION 1664.000*, 1 CALIBER = 1 DIA. = 396.0 IN.

0.1652850E 03 0.1343487E 01 0.8259681E 06 0.1602390E 06 IGNITION                      1
0.1459529E 07 0.2196021E 04 0.7553565E 00-0.7319921E-01 0.1011809E 06 0.1377536E 08 2
0.6620312E 06 0.5577893E 02 0.1918605E-01-0.1859259E-02 0.9922455E 06 0.1350901E 09 3
0.8242041E 06 0.1931873E 04 0.2530000E 03 0.1161622E 04 0.7105281E 02 0.1377608E 08 4
0.1599570E 06 0.2470439E 04 0.1540000E 03 0.1163004E 04 0.4377100E 01 0.1350972E 09 5

0.1652860E 03 0.1344572E 01 0.8247031E 06 0.1597280E 06 90 PERCENT THRUST             1
0.1455998E 07 0.2196451E 04 0.7566686E 00-0.7332636E-01 0.1002599E 06 0.1376482E 08 2
0.6608832E 06 0.5578984E 02 0.1921938E-01-0.1862489E-02 0.9832138E 06 0.1349868E 09 3
0.8229391E 06 0.1931461E 04 0.2530000E 03 0.1179423E 04 0.7105279E 02 0.1376555E 08 4
0.1594460E 06 0.2467759E 04 0.1540000E 03 0.1279740E 04 0.4377100E 01 0.1349939E 09 5

0.1677860E 03 0.1346393E 01 0.8196721E 06 0.1586800E 06 THRUST EVENT                  1
0.1450981E 07 0.2197172E 04 0.7598065E 00-0.7363043E-01 0.1002616E 06 0.1375511E 08 2
0.6581540E 06 0.5580816E 02 0.1929908E-01-0.1870212E-02 0.9832310E 06 0.1348915E 09 3
0.8179081E 06 0.1929949E 04 0.2700000E 03 0.1249944E 04 0.7105104E 02 0.1375583E 08 4
0.1583980E 06 0.2462803E 04 0.1740000E 03 0.1514277E 04 0.4376508E 01 0.1348986E 09 5

0.1700000E 03 0.1348353E 01 0.8145628E 06 0.1577470E 06                               1
0.1444962E 07 0.2197948E 04 0.7629715E 00-0.7393714E-01 0.1002615E 06 0.1374484E 08 2
0.6554237E 06 0.5582787E 02 0.1937947E-01-0.1878003E-02 0.9832300E 06 0.1347908E 09 3
0.8127988E 06 0.1928449E 04 0.2864943E 03 0.1321604E 04 0.7104948E 02 0.1374557E 08 4
0.1574650E 06 0.2458503E 04 0.1810053E 03 0.1723143E 04 0.4375984E 01 0.1347980E 09 5

0.1900000E 03 0.1367774E 01 0.7684092E 06 0.1493192E 06                               1
0.1390593E 07 0.2205639E 04 0.7928019E 00-0.7682792E-01 0.1002605E 06 0.1364393E 08 2
0.6307623E 06 0.5602322E 02 0.2013716E-01-0.1951429E-02 0.9832202E 06 0.1338012E 09 3
0.7666452E 06 0.1915485E 04 0.4354928E 03 0.1969060E 04 0.7103538E 02 0.1364466E 08 4
0.1490372E 06 0.2426101E 04 0.2442872E 03 0.3612176E 04 0.4371250E 01 0.1338083E 09 5
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TABLE IX

MASS PROPERTIES VERSUS TIME (Cont'd)

AS-511

0.1939860E 03	0.1372022E 01	0.7592108E 06	0.1476396E 06			1
0.1379757E 07	0.2207321E 04	0.7990282E 00	-0.7743129E-01	0.1002603E 06	0.1362189E 08	2
0.6258472E 06	0.5606594E 02	0.2029531E-01	-0.1966754E-02	0.9832182E 06	0.1335851E 09	3
0.7574468E 06	0.1913105E 04	0.4651882E 03	0.2098129E 04	0.7103257E 02	0.1362261E 08	4
0.1473576E 06	0.2420335E 04	0.2568992E 03	0.3989152E 04	0.4370306E 01	0.1335922E 09	5

0.1939860E 03	0.1381625E 01	0.7592108E 06	0.1476396E 06	INTERSTAGE JETTISONED		1
0.1370322E 07	0.2211124E 04	0.7687264E 00	-0.4216115E-01	0.8925412E 05	0.1352888E 08	2
0.6215676E 06	0.5616253E 02	0.1952565E-01	-0.1070893E-02	0.3752837E 06	0.1326730E 09	3
0.7574468E 06	0.1913105E 04	0.4651882E 03	0.2098129E 04	0.7103257E 02	0.1352955E 08	4
0.1473576E 06	0.2420335E 04	0.2568992E 03	0.3989152E 04	0.4370306E 01	0.1326795E 09	5

0.1989860E 03	0.1387283E 01	0.7476725E 06	0.1455326E 06			1
0.1356730E 07	0.2213364E 04	0.7764277E 00	-0.4258354E-01	0.8925387E 05	0.1349957E 08	2
0.6154023E 06	0.5621945E 02	0.1972126E-01	-0.1081621E-02	0.8752812E 06	0.1323856E 09	3
0.7459085E 06	0.1910171E 04	0.5024379E 03	0.2260046E 04	0.7102905E 02	0.1350024E 08	4
0.1452506E 06	0.2413256E 04	0.2727197E 03	0.4462250E 04	0.4369123E 01	0.1323921E 09	5

0.1989860E 03	0.1356397E 01	0.7476725E 06	0.1455326E 06	L.E.S. JETTISONED		1
0.1347794E 07	0.2201133E 04	0.7822384E 00	-0.4021383E-01	0.8915120E 05	0.1258244E 08	2
0.6113491E 06	0.5590878E 02	0.1986885E-01	-0.1021431E-02	0.8742743E 06	0.1233916E 09	3
0.7459085E 06	0.1910171E 04	0.5024379E 03	0.2260046E 04	0.7102905E 02	0.1258311E 08	4
0.1452506E 06	0.2413256E 04	0.2727197E 03	0.4462250E 04	0.4369123E 01	0.1233981E 09	5

0.2000000E 03	0.1357520E 01	0.7453325E 06	0.1451053E 06			1
0.1345037E 07	0.2201578E 04	0.7838416E 00	-0.4029624E-01	0.8915114E 05	0.1257674E 08	2
0.6100987E 06	0.5592008E 02	0.1990957E-01	-0.1023524E-02	0.8742737E 06	0.1233356E 09	3
0.7435685E 06	0.1909589E 04	0.5099921E 03	0.2292884E 04	0.7102833E 02	0.1257740E 08	4
0.1448233E 06	0.2411830E 04	0.2759281E 03	0.4558232E 04	0.4358883E 01	0.1233421E 09	5

0.2200000E 03	0.1381139E 01	0.6991788E 06	0.1366775E 06			1
0.1290668E 07	0.2210931E 04	0.8168607E 00	-0.4199371E-01	0.8915010E 05	0.1245910E 08	2
0.5854373E 06	0.5615765E 02	0.2074826E-01	-0.1066640E-02	0.8742636E 06	0.1221820E 09	3
0.6974148E 06	0.1898544E 04	0.6589907E 03	0.2940727E 04	0.7101423E 02	0.1245976E 08	4
0.1363955E 06	0.2384614E 04	0.3392100E 03	0.6453417E 04	0.4364150E 01	0.1221885E 09	5

0.2400000E 03	0.1408322E 01	0.6530252E 06	0.1282498E 06			1
0.1236299E 07	0.2221696E 04	0.8527840E 00	-0.4384048E-01	0.8914896E 05	0.1232825E 08	2
0.5607760E 06	0.5643107E 02	0.2166071E-01	-0.1113548E-02	0.3742525E 06	0.1208988E 09	3
0.6512612E 06	0.1888211E 04	0.8079893E 03	0.3588826E 04	0.7100013E 02	0.1232891E 08	4
0.1279678E 06	0.2357862E 04	0.4024918E 03	0.8352714E 04	0.4359416E 01	0.1209052E 09	5

TABLE IX
MASS PROPERTIES VERSUS TIME (Cont'd)

AS-511

0.2600000E 03	0.1439476E 01	0.6068717E 06	0.1198220E 06			1
0.1191930E 07	0.2234033E 04	0.8920123E 00	-0.4585715E-01	0.8914773E 05	0.1217717E 08	2
0.5361146E 06	0.5674443E 02	0.2265711E-01	-0.1164771E-02	0.8742403E 06	0.1194172E 09	3
0.6051077E 06	0.1878349E 04	0.9569879E 03	0.4237184E 04	0.7098605E 02	0.1217783E 08	4
0.1195400E 06	0.2331032E 04	0.4657736E 03	0.1025614E 05	0.4354681E 01	0.1194237E 09	5
0.2800000E 03	0.1475123E 01	0.5607181E 06	0.1113942E 06			1
0.1127561E 07	0.2248149E 04	0.9350237E 00	-0.4806831E-01	0.8914637E 05	0.1200307E 08	2
0.5114530E 06	0.5710298E 02	0.2374960E-01	-0.1220935E-02	0.8742270E 06	0.1177099E 09	3
0.5589541E 06	0.1868800E 04	0.1105986E 04	0.4885798E 04	0.7097195E 02	0.1200373E 08	4
0.1111122E 06	0.2304147E 04	0.5290555E 03	0.1216370E 05	0.4349947E 01	0.1177163E 09	5
0.3000000E 03	0.1515869E 01	0.5145645E 06	0.1029664E 06			1
0.1073192E 07	0.2264284E 04	0.9823930E 00	-0.5050350E-01	0.8914487E 05	0.1180269E 08	2
0.4867917E 06	0.5751281E 02	0.2495278E-01	-0.1282789E-02	0.8742123E 06	0.1157448E 09	3
0.5128005E 06	0.1859450E 04	0.1254985E 04	0.5534670E 04	0.7095785E 02	0.1180335E 08	4
0.1026844E 06	0.2277193E 04	0.5923374E 03	0.1407543E 05	0.4345213E 01	0.1157513E 09	5
0.3200000E 03	0.1562622E 01	0.4684109E 06	0.9453868E 05			1
0.1018823E 07	0.2282798E 04	0.1034817E 01	-0.5319859E-01	0.8914323E 05	0.1157094E 08	2
0.4621303E 06	0.5798307E 02	0.2628437E-01	-0.1351244E-02	0.8741962E 06	0.1134721E 09	3
0.4666469E 06	0.1850204E 04	0.1403983E 04	0.6183799E 04	0.7094375E 02	0.1157159E 08	4
0.9425668E 05	0.2250186E 04	0.6556192E 03	0.1599133E 05	0.4340479E 01	0.1134785E 09	5
0.3400000E 03	0.1616366E 01	0.4222573E 06	0.8611092E 05			1
0.9644540E 06	0.2304081E 04	0.1093153E 01	-0.5619756E-01	0.8914139E 05	0.1130065E 08	2
0.4374689E 06	0.5852365E 02	0.2776609E-01	-0.1427417E-02	0.8741781E 06	0.1108215E 09	3
0.4204933E 06	0.1840940E 04	0.1552982E 04	0.6833188E 04	0.7092965E 02	0.1130130E 08	4
0.8582892E 05	0.2223125E 04	0.7189011E 03	0.1791141E 05	0.4335745E 01	0.1108279E 09	5
0.3600000E 03	0.1678300E 01	0.3761036E 06	0.7768314E 05			1
0.9100848E 06	0.2328607E 04	0.1158459E 01	-0.5955484E-01	0.8913932E 05	0.1098632E 08	2
0.4128075E 06	0.5914651E 02	0.2942486E-01	-0.1512692E-02	0.8741580E 06	0.1077390E 09	3
0.3743396E 06	0.1831632E 04	0.1701980E 04	0.7482335E 04	0.7091557E 02	0.1098697E 08	4
0.7740114E 05	0.2196000E 04	0.7821829E 03	0.1983569E 05	0.4331012E 01	0.1077454E 09	5
0.3800000E 03	0.1750056E 01	0.3299501E 06	0.6925535E 05			1
0.8557153E 06	0.2357022E 04	0.1232063E 01	-0.6333874E-01	0.8913700E 05	0.1061792E 08	2
0.3881461E 06	0.5986837E 02	0.3129441E-01	-0.1608803E-02	0.8741351E 06	0.1041262E 09	3
0.3281861E 06	0.1922194E 04	0.1850979E 04	0.8132739E 04	0.7090147E 02	0.1061857E 08	4
0.6397335E 05	0.2168808E 04	0.8454648E 03	0.2176417E 05	0.4326273E 01	0.1041325E 09	5

TABLE IX
MASS PROPERTIES VERSUS TIME (Cont'd)

AS-511

0.4000000E 03	0.1833645E 01	0.2837966E 06	0.6082757E 05			1
0.8013468E 06	0.2390124E 04	0.1315655E 01	-0.6763608E-01	0.8913437E 05	0.1018330E 08	2
0.3634848E 06	0.6070913E 02	0.3341764E-01	-0.1717956E-02	0.8741093E 06	0.9986404E 08	3
0.2820326E 06	0.1812502E 04	0.1999978E 04	0.8782902E 04	0.7083737E 02	0.1018394E 08	4
0.6054557E 05	0.2141563E 04	0.9087467E 03	0.2369689E 05	0.4321544E 01	0.9987038E 08	5
0.4200000E 03	0.1931702E 01	0.2376430E 06	0.5239978E 05			1
0.7469777E 06	0.2428954E 04	0.1411416E 01	-0.7255901E-01	0.8913134E 05	0.9666928E 07	2
0.3388233E 06	0.6169543E 02	0.3584996E-01	-0.1842998E-02	0.8740796E 06	0.9480014E 08	3
0.2358790E 06	0.1802439E 04	0.2148977E 04	0.9433324E 04	0.7087327E 02	0.9667572E 07	4
0.5211778E 05	0.2114265E 04	0.9720285E 03	0.2563384E 05	0.4316810E 01	0.9480646E 08	5
0.4263120F 03	0.1966190E 01	0.2230770E 06	0.4974000E 05	THRUST EVENT		1
0.7298191E 06	0.2442611E 04	0.1444599E 01	-0.7426492E-01	0.8913029E 05	0.9483790E 07	2
0.3310403E 06	0.6204232E 02	0.3669282E-01	-0.1886328E-02	0.8740693E 06	0.9300417E 08	3
0.2213130E 06	0.1799159E 04	0.2196000E 04	0.9638648E 04	0.7086883E 02	0.9484432E 07	4
0.4945800E 05	0.2105600E 04	0.9920001E 03	0.2624602E 05	0.4315316E 01	0.9301048E 08	5
0.4300000E 03	0.1983175E 01	0.2162723E 06	0.4828891E 05			1
0.7216083E 06	0.2449337E 04	0.1461036E 01	-0.7510994E-01	0.8912978E 05	0.9393818E 07	2
0.3273160E 06	0.6221316E 02	0.3711032E-01	-0.1907792E-02	0.8740643E 06	0.9212185E 08	3
0.2145083E 06	0.1797604E 04	0.2230458E 04	0.9734556E 04	0.7086622E 02	0.9394460E 07	4
0.4800691E 05	0.2100823E 04	0.1002607E 04	0.2658003E 05	0.4314443E 01	0.9212816E 08	5
0.4500000E 03	0.2083559E 01	0.1793712E 06	0.4041978E 05			1
0.6770825E 06	0.2489089E 04	0.1557116E 01	-0.8004927E-01	0.8912675E 05	0.8859080E 07	2
0.3071194E 06	0.6322286E 02	0.3955075E-01	-0.2033251E-02	0.8740346E 06	0.8687787E 08	3
0.1776072E 06	0.1788899E 04	0.2417319E 04	0.1025477E 05	0.7085214E 02	0.8859720E 07	4
0.4013778E 05	0.2074982E 04	0.1060129E 04	0.2839372E 05	0.4309709E 01	0.8688414E 08	5
0.4700000E 03	0.2200174E 01	0.1424700E 06	0.3255065E 05			1
0.6325565E 06	0.2535269E 04	0.1666723E 01	-0.8568400E-01	0.8912329E 05	0.8230954E 07	2
0.2869227E 06	0.6439582E 02	0.4233475E-01	-0.2176373E-02	0.8740007E 06	0.8071806E 08	3
0.1407060E 06	0.1779575E 04	0.2604181E 04	0.1077519E 05	0.7083804E 02	0.8231590E 07	4
0.3226865E 05	0.2049393E 04	0.1117651E 04	0.3021139E 05	0.4304975E 01	0.8072430E 08	5
0.4900000E 03	0.2336884E 01	0.1055689E 06	0.2468152E 05			1
0.5880306E 06	0.2589406E 04	0.1792927E 01	-0.9217201E-01	0.8911931E 05	0.7485511E 07	2
0.2667261E 06	0.6577092E 02	0.4554036E-01	-0.2341168E-02	0.8739616E 06	0.7340777E 08	3
0.1038049E 06	0.1769355E 04	0.2791043E 04	0.1129582E 05	0.7082394E 02	0.7486143E 07	4
0.2439952E 05	0.2023738E 04	0.1175173E 04	0.3203307E 05	0.4300241E 01	0.7341396E 08	5

TABLE IX
MASS PROPERTIES VERSUS TIME (Cont'd)

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0.5100000E 03	0.2498748E 01	0.6866782E 05	0.1681239E 05			1
0.5435047E 06	0.2653504E 04	0.1939811E 01	-0.9972308E-01	0.8911468E 05	0.6591723E 07	2
0.2465295E 06	0.6739901E 02	0.4927119E-01	-0.2532965E-02	0.8739162E 06	0.6464270E 08	3
0.6690382E 05	0.1757602E 04	0.2977905E 04	0.1181666E 05	0.7080984E 02	0.6592351E 07	4
0.1653039E 05	0.1998033E 04	0.1232695E 04	0.3385876E 05	0.4295508E 01	0.6464886E 08	5
0.5300001E 03	0.2692961E 01	0.3176668E 05	0.8943261E 04			1
0.4989789E 06	0.2730413E 04	0.2112907E 01	-0.1086217E 00	0.8910921E 05	0.5503993E 07	2
0.2263330E 06	0.6935247E 02	0.5366785E-01	-0.2758991E-02	0.8738626E 06	0.5397572E 08	3
0.3000268E 05	0.1742601E 04	0.3164767E 04	0.1233771E 05	0.7079574E 02	0.5504615E 07	4
0.8661261E 04	0.1970567E 04	0.1290218E 04	0.3568848E 05	0.4290774E 01	0.5398181E 08	5
0.5438320E 03	0.2851837E 01	0.6246000E 04	0.3501000E 04	CUTOFF SIGNAL		1
0.4681850E 06	0.2793328E 04	0.2251879E 01	-0.1157661E 00	0.8910484E 05	0.4600258E 07	2
0.2123651E 06	0.7095053E 02	0.5719772E-01	-0.2940458E-02	0.8738197E 06	0.4511310E 08	3
0.4482000E 04	0.1725312E 04	0.3294000E 04	0.1269818E 05	0.7078601E 02	0.4600876E 07	4
0.3219000E 04	0.1938989E 04	0.1330000E 04	0.3695627E 05	0.4287500E 01	0.4511916E 08	5
0.5438330E 03	0.2853726E 01	0.5974000E 04	0.3431000E 04	END THRUST DECAY		1
0.4678380E 06	0.2794076E 04	0.2253550E 01	-0.1158519E 00	0.8910448E 05	0.4589481E 07	2
0.2122077E 06	0.7096952E 02	0.5724015E-01	-0.2942639E-02	0.8738162E 06	0.4500742E 08	3
0.4210000E 04	0.1724928E 04	0.3294000E 04	0.1270202E 05	0.7078601E 02	0.4590099E 07	4
0.3149000E 04	0.1938390E 04	0.1330000E 04	0.3697260E 05	0.4287500E 01	0.4501348E 08	5

TABLE IX
MASS PROPERTIES VERSUS TIME

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THIRD FLIGHT STAGE

TIME	X STATION	TOTAL LOX	TOTAL FUEL			1
VEHICLE MASS	X STATION	Y STATION	Z STATION	IX (ROLL)	IY (PITCH)	2
VEHICLE MASS	X STATION	Y STATION	Z STATION	IX (ROLL)	IY (PITCH)	3
MASS-LOX TK	LEVEL-LOX	UL MASS-LOX	UL VOL-LOX	LOX DENSITY	IZ (YAW)	4
MASS-FUEL TK	LEVEL-FUEL	UL MASS-FUEL	UL VOL-FUEL	FUEL DENSITY	IZ (YAW)	5
SECONDS	*CALIBERS	LBM	LBM			1
LBM	INCHES	INCHES	INCHES	KG-M-SEC2	KG-M-SEC2	2
KILOGRAMS	METERS	METERS	METERS	KG-M2	KG-M2	3
LBM	STA-INCHES	LBM	CUBIC FEET	LBM/FT3	KG-M-SEC2	4
LBM	STA-INCHES	LBM	CUBIC FEET	LBM/FT3	KG-M2	5

FROM GIMBAL PLANE (STATION 2645.853, 1 CALIBER = 1 DIA. = 260.0 IN.

0.5438310E 03	0.1497891E 01	0.1924083E 06	0.4182800E 05	IGNITION		1
0.3647933E 06	0.3035304E 04	0.1314738E 01	0.7211365E 00	0.1997479E 05	0.1367647E 07	2
0.1654674E 06	0.7709672E 02	0.3339436E-01	0.1831686E-01	0.1958857E 06	0.1341203E 08	3
0.1920390E 06	0.2855367E 04	0.5000000E 02	0.1181973E 03	0.7090400E 02	0.1367571E 07	4
0.4178300E 05	0.3166061E 04	0.7500001E 02	0.7931806E 03	0.4319000E 01	0.1341129E 08	5
0.5438320E 03	0.1498420E 01	0.1920863E 06	0.4171400E 05	1ST 90 PERCENT THRUST		1
0.3643313E 06	0.3035442E 04	0.1316647E 01	0.7218906E 00	0.1997335E 05	0.1367343E 07	2
0.1652578E 06	0.7710021E 02	0.3344284E-01	0.1833602E-01	0.1958716E 06	0.1340905E 08	3
0.1917170E 06	0.2854912E 04	0.5000000E 02	0.1227384E 03	0.7090399E 02	0.1367267E 07	4
0.4166900E 05	0.3164930E 04	0.7500001E 02	0.8195740E 03	0.4318999E 01	0.1340831E 08	5
0.5502320E 03	0.1503470E 01	0.1891416E 06	0.4117559E 05			1
0.3608508E 06	0.3036755E 04	0.1328763E 01	0.7290167E 00	0.1997176E 05	0.1364378E 07	2
0.1636791E 06	0.7713357E 02	0.3375059E-01	0.1851702E-01	0.1958560E 06	0.1337997E 08	3
0.1887723E 06	0.2851087E 04	0.5242067E 02	0.1640202E 03	0.7089738E 02	0.1364302E 07	4
0.4113059E 05	0.3159604E 04	0.7814688E 02	0.9438538E 03	0.4318828E 01	0.1337923E 08	5
0.5502320E 03	0.1503836E 01	0.1891416E 06	0.4117559E 05	ULLAGE CASES JETTISONED		1
0.3607208E 06	0.3036850E 04	0.1329242E 01	0.7292795E 00	0.1991439E 05	0.1364096E 07	2
0.1636201E 06	0.7713598E 02	0.3376276E-01	0.1852369E-01	0.1952934E 06	0.1337720E 08	3
0.1887723E 06	0.2851087E 04	0.5242067E 02	0.1640202E 03	0.7089738E 02	0.1364020E 07	4
0.4113059E 05	0.3159604E 04	0.7814688E 02	0.9438588E 03	0.4318828E 01	0.1337646E 08	5
0.5600001E 03	0.1511949E 01	0.1846473E 06	0.4035385E 05			1
0.3554087E 06	0.3038959E 04	0.1348206E 01	0.7404328E 00	0.1991196E 05	0.1359429E 07	2
0.1612106E 06	0.7718957E 02	0.3424443E-01	0.1880699E-01	0.1952696E 06	0.1333144E 08	3
0.1842780E 06	0.2846005E 04	0.5611524E 02	0.2270422E 03	0.7088729E 02	0.1359353E 07	4
0.4030885E 05	0.3151968E 04	0.8294981E 02	0.1133566E 04	0.4318568E 01	0.1333069E 08	5

TABLE IX

MASS PROPERTIES VERSUS TIME (Cont'd)

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0.5800001E 03	0.1530273E 01	0.1754453E 06	0.3867133E 05			1
0.3445326E 06	0.3043724E 04	0.1388857E 01	0.7643414E 00	0.1990695E 05	0.1348821E 07	2
0.1562773E 06	0.7731057E 02	0.3527696E-01	0.1941427E-01	0.1952204E 06	0.1322741E 08	3
0.1750760E 06	0.2837308E 04	0.6367982E 02	0.3561342E 03	0.7086665E 02	0.1348745E 07	4
0.3862633E 05	0.3137439E 04	0.9278376E 02	0.1522060E 04	0.4318034E 01	0.1322666E 08	5
0.6000001E 03	0.1550944E 01	0.1662433E 06	0.3698882E 05			1
0.3336564E 06	0.3049098E 04	0.1432158E 01	0.7898091E 00	0.1990189E 05	0.1336734E 07	2
0.1513440E 06	0.7744709E 02	0.3637682E-01	0.2006115E-01	0.1951708E 06	0.1310688E 08	3
0.1658740E 06	0.2829888E 04	0.7124440E 02	0.4853012E 03	0.7084599E 02	0.1336673E 07	4
0.3694382E 05	0.3123946E 04	0.1026177E 03	0.1910649E 04	0.4317500E 01	0.1310828E 08	5
0.6200001E 03	0.1574016E 01	0.1570413E 06	0.3530631E 05			1
0.3227902E 06	0.3055097E 04	0.1478378E 01	0.8169927E 00	0.1989678E 05	0.1323223E 07	2
0.1464106E 06	0.7759945E 02	0.3755079E-01	0.2075161E-01	0.1951207E 06	0.1297638E 08	3
0.1566720E 06	0.2823259E 04	0.7880897E 02	0.6145440E 03	0.7082534E 02	0.1323146E 07	4
0.3526131E 05	0.3110938E 04	0.1124516E 03	0.2299333E 04	0.4316965E 01	0.1297563E 08	5
0.6400001E 03	0.1599705E 01	0.1473393E 06	0.3362379E 05			1
0.3119040E 06	0.3061776E 04	0.1527821E 01	0.8460723E 00	0.1989160E 05	0.1308082E 07	2
0.1414772E 06	0.7776910E 02	0.3880664E-01	0.2149023E-01	0.1950700E 06	0.1282790E 08	3
0.1474700E 06	0.2817160E 04	0.8637356E 02	0.7438616E 03	0.7080468E 02	0.1308020E 07	4
0.3357879E 05	0.3098327E 04	0.1222856E 03	0.2688116E 04	0.4316431E 01	0.1282729E 08	5
0.6600001E 03	0.1628106E 01	0.1386373E 06	0.3194128E 05			1
0.3010278E 06	0.3069160E 04	0.1580837E 01	0.8772535E 00	0.1988637E 05	0.1291791E 07	2
0.1365439E 06	0.7795666E 02	0.4015325E-01	0.2228223E-01	0.1950186E 06	0.1266814E 08	3
0.1382680E 06	0.2811450E 04	0.9393814E 02	0.8732558E 03	0.7078404E 02	0.1291698E 07	4
0.3189628E 05	0.3085546E 04	0.1321195E 03	0.3076994E 04	0.4315897E 01	0.1266722E 08	5
0.6760271E 03	0.1653102E 01	0.1312633E 06	0.3059300E 05	1ST CUTOFF SIGNAL		1
0.2922123E 06	0.3075659E 04	0.1626169E 01	0.9039146E 00	0.1988212E 05	0.1277292E 07	2
[illegible]	0.7812175E 02	0.4130467E-01	0.2295942E-01	0.1949769E 06	0.1252595E 08	3
[illegible]	[illegible]	[illegible]	0.9769995E 03	0.7076748E 02	0.1277213E 07	4
[illegible]	0.3075133E 04	[illegible]	0.3389691E 04	0.4315469E 01	0.1252518E 08	5
[illegible]	0.1653371E 01	0.1311873E 06	0.3056200E 05	1ST E.T.D.		1
[illegible]	0.3075729E 04	0.1626763E 01	0.9042456E 00	0.1988210E 05	0.1277166E 07	2
[illegible]	[illegible]	0.4131973E-01	0.2294783E-01	0.1949768E 06	0.1252471E 08	3
[illegible]	[illegible]	[illegible]	0.9790732E 03	0.7076748E 02	[illegible]	4
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	5

TABLE IX
MASS PROPERTIES VERSUS TIME (Cont'd)

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0.6760300E 03	0.1656420E 01	0.1308973E 06	0.2705000E 05	2ND IGNITION		1
0.2982902E 06	0.3076522E 04	0.1637725E 01	0.9196437E 00	0.1961168E 05	0.1276335E 07	2
0.1307462E 06	0.7814366E 02	0.4159823E-01	0.2335894E-01	0.1922248E 06	0.1251655E 08	3
0.1305280E 06	0.2806881E 04	0.2300000E 03	0.9821713E 03	0.7076749E 02	0.1276491E 07	4
0.2700500E 05	0.3047833E 04	0.3660000E 03	0.4209690E 04	0.4315468E 01	0.1251810E 08	5
0.6760310E 03	0.1657427E 01	0.1306363E 06	0.2694300E 05	2ND 90 PERCENT THRUST		1
0.2979183E 06	0.3076784E 04	0.1640147E 01	0.9206290E 00	0.1961155E 05	0.1275781E 07	2
0.1305975E 06	0.7815029E 02	0.4165974E-01	0.2333397E-01	0.1923235E 06	0.1251113E 08	3
0.1302670E 06	0.2806729E 04	0.2300000E 03	0.9858594E 03	0.7076748E 02	0.1275953E 07	4
0.2689000E 05	0.3047005E 04	0.3660000E 03	0.4234485E 04	0.4315468E 01	0.1251282E 08	5
0.6800001E 03	0.1663514E 01	0.1290442E 06	0.2662224E 05			1
0.2959737E 06	0.3079366E 04	0.1650774E 01	0.9270361E 00	0.1959077E 05	0.1272220E 07	2
0.1297154E 06	0.7819050E 02	0.4192966E-01	0.2354671E-01	0.1921198E 06	0.1247621E 08	3
0.1286749E 06	0.2805812E 04	0.2314643E 03	0.1008251E 04	0.7076338E 02	0.1272407E 07	4
0.2657724E 05	0.3044542E 04	0.3683404E 03	0.4308661E 04	0.4315362E 01	0.1247805E 08	5
0.7000001E 03	0.1695074E 01	0.1210217E 06	0.2500599E 05			1
0.2763468E 06	0.3086572E 04	0.1705541E 01	0.9600983E 00	0.1958670E 05	0.1254186E 07	2
0.1253488E 06	0.7839892E 02	0.4332075E-01	0.2438649E-01	0.1920799E 06	0.1229936E 08	3
0.1206524E 06	0.2801298E 04	0.2388430E 03	0.1121124E 04	0.7074273E 02	0.1254373E 07	4
0.2496099E 05	0.3031967E 04	0.3801337E 03	0.4682479E 04	0.4314828E 01	0.1230119E 08	5
0.7200001E 03	0.1729732E 01	0.1129992E 06	0.2338974E 05			1
0.2667200E 06	0.3095583E 04	0.1764261E 01	0.9955471E 00	0.1958256E 05	0.1234118E 07	2
0.1209821E 06	0.7862780E 02	0.4481223E-01	0.2528689E-01	0.1920393E 06	0.1210256E 08	3
0.1126299E 06	0.2796927E 04	0.2462217E 03	0.1234062E 04	0.7072207E 02	0.1234304E 07	4
0.2334474E 05	0.3019644E 04	0.3919271E 03	0.5056390E 04	0.4314294E 01	0.1210439E 08	5
0.7400001E 03	0.1767790E 01	0.1049767E 06	0.2177349E 05			1
0.2570931E 06	0.3105478E 04	0.1827378E 01	0.1033650E 01	0.1957834E 05	0.1211361E 07	2
0.1166154E 06	0.7887913E 02	0.4641541E-01	0.2625472E-01	0.1919979E 06	0.1188429E 08	3
0.1046074E 06	0.2792677E 04	0.2536004E 03	0.1347067E 04	0.7070143E 02	0.1212047E 07	4
0.2172949E 05	0.3007295E 04	0.4037203E 03	0.5430393E 04	0.4313760E 01	0.1188611E 08	5
0.7600001E 03	0.1809598E 01	0.9695429E 05	0.2015724E 05			1
0.2474663E 06	0.3116349E 04	0.1895407E 01	0.1074718E 01	0.1957403E 05	0.1187218E 07	2
0.1122498E 06	0.7915524E 02	0.4814334E-01	0.2729784E-01	0.1919556E 06	0.1164263E 08	3
0.9658500E 05	0.2788513E 04	0.2609791E 03	0.1460138E 04	0.7068077E 02	0.1187388E 07	4
0.2011224E 05	0.2994789E 04	0.4155137E 03	0.5804489E 04	0.4313226E 01	0.1164429E 08	5

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MASS PROPERTIES VERSUS TIME (Cont'd)

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0.7800001E 03	0.1855728E 01	0.8893182E 05	0.1854099E 05			1
0.2378394E 06	0.3128342E 04	0.1968943E 01	0.1119111E 01	0.1956962E 05	0.1159604E 07	2
0.1078821E 06	0.7945988E 02	0.5001115E-01	0.2842541E-01	0.1919123E 06	0.1137183E 08	3
0.8856253E 05	0.2734286E 04	0.2683578E 03	0.1573274E 04	0.7066011E 02	0.1159774E 07	4
0.1849599E 05	0.2982404E 04	0.4273070E 03	0.6178677E 04	0.4312692E 01	0.1137349E 08	5
0.8000001E 03	0.1906580E 01	0.8090934E 05	0.1692474E 05			1
0.2282125E 06	0.3141563E 04	0.2048683E 01	0.1167249E 01	0.1956509E 05	0.1128969E 07	2
0.1035154E 06	0.7979571E 02	0.5203653E-01	0.2964812E-01	0.1918680E 06	0.1107140E 08	3
0.8054004E 05	0.2779935E 04	0.2757365E 03	0.1686477E 04	0.7063946E 02	0.1129137E 07	4
0.1687974E 05	0.2970063E 04	0.4391003E 03	0.6552959E 04	0.4312158E 01	0.1107305E 08	5
0.8200001E 03	0.1962898E 01	0.7288687E 05	0.1530849E 05			1
0.2185856E 06	0.3156206E 04	0.2135446E 01	0.1219627E 01	0.1956044E 05	0.1094583E 07	2
0.9914879E 05	0.8016763E 02	0.5424031E-01	0.3097852E-01	0.1918223E 06	0.1073419E 08	3
0.7251757E 05	0.2775436E 04	0.2831152E 03	0.1799746E 04	0.7061380E 02	0.1094751E 07	4
0.1526349E 05	0.2957718E 04	0.4508936E 03	0.6927333E 04	0.4311624E 01	0.1073583E 08	5
0.8400001E 03	0.2025490E 01	0.6486439E 05	0.1369224E 05			1
0.2089588E 06	0.3172480E 04	0.2230204E 01	0.1276831E 01	0.1955564E 05	0.1055906E 07	2
0.9478210E 05	0.8058099E 02	0.5664718E-01	0.3243152E-01	0.1917753E 06	0.1035490E 08	3
0.6449510E 05	0.2770762E 04	0.2904939E 03	0.1913082E 04	0.7059815E 02	0.1056073E 07	4
0.1364724E 05	0.2945369E 04	0.4626869E 03	0.7301799E 04	0.4311090E 01	0.1035654E 08	5
0.8600001E 03	0.2095248E 01	0.5684192E 05	0.1207599E 05			1
0.1993319E 06	0.3190617E 04	0.2334114E 01	0.1339561E 01	0.1955067E 05	0.1012284E 07	2
0.9041545E 05	0.8104157E 02	0.5928649E-01	0.3402484E-01	0.1917265E 06	0.9927114E 07	3
0.5647262E 05	0.2765874E 04	0.2978726E 03	0.2026483E 04	0.7057749E 02	0.1012450E 07	4
0.1203099E 05	0.2933017E 04	0.4744802E 03	0.7676359E 04	0.4310556E 01	0.9928740E 07	5
0.8800001E 03	0.2173240E 01	0.4881944E 05	0.1045974E 05			1
0.1897050E 06	0.3210895E 04	0.2448571E 01	0.1408557E 01	0.1954551E 05	0.9629615E 06	2
0.8604978E 05	0.8155673E 02	0.6219370E-01	0.3577990E-01	0.1916759E 06	0.9443424E 07	3
0.4845014E 05	0.2760710E 04	0.3052513E 03	0.2139952E 04	0.7055685E 02	0.9631265E 06	4
0.1041474E 05	0.2920650E 04	0.4862735E 03	0.8051010E 04	0.4310021E 01	0.9445042E 07	5
0.9000001E 03	0.2260903E 01	0.4079696E 05	0.8843490E 04			1
0.1800781E 06	0.3233687E 04	0.2575266E 01	0.1485141E 01	0.1954012E 05	0.9068901E 06	2
0.8168207E 05	0.8213566E 02	0.6541176E-01	0.3772260E-01	0.1916231E 06	0.8893552E 07	3
0.4042767E 05	0.2755201E 04	0.3126300E 03	0.2253486E 04	0.7053619E 02	0.9070541E 06	4
0.8798490E 04	0.2908299E 04	0.4980668E 03	0.8425757E 04	0.4309487E 01	0.8895160E 07	5

TABLE IX
MASS PROPERTIES VERSUS TIME (Cont'd)

AS-511

0.9200001E 03	0.2360013E 01	0.3277449E 05	0.7227240E 04			1
0.1704513E 06	0.3259456E 04	0.2716271E 01	0.1570265E 01	0.1953448E 05	0.8426600E 06	2
0.7731542E 05	0.8279019E 02	0.6899330E-01	0.3988473E-01	0.1915677E 06	0.8263669E 07	3
0.3240519E 05	0.2749226E 04	0.3200087E 03	0.2367087E 04	0.7051554E 02	0.8428148E 06	4
0.7182240E 04	0.2895934E 04	0.5098601E 03	0.8800595E 04	0.4308953E 01	0.8265188E 07	5
0.9400001E 03	0.2472609E 01	0.2475202E 05	0.5610989E 04			1
0.1608245E 06	0.3288731E 04	0.2874157E 01	0.1665579E 01	0.1952852E 05	0.7687918E 06	2
0.7294876E 05	0.8353376E 02	0.7300360E-01	0.4230571E-01	0.1915093E 06	0.7539271E 07	3
0.2438273E 05	0.2742588E 04	0.3273874E 03	0.2480754E 04	0.7049488E 02	0.7689453E 06	4
0.5565989E 04	0.2883563E 04	0.5216534E 03	0.9175527E 04	0.4308419E 01	0.7540776E 07	5
0.9600001E 03	0.2601515E 01	0.1672955E 05	0.3994738E 04			1
0.1511976E 06	0.3322247E 04	0.3052150E 01	0.1773031E 01	0.1952219E 05	0.6833260E 06	2
0.6858207E 05	0.8438507E 02	0.7752460E-01	0.4503499E-01	0.1914473E 06	0.6701137E 07	3
0.1636025E 05	0.2734881E 04	0.3347661E 03	0.2594489E 04	0.7047424E 02	0.6834780E 06	4
0.3949738E 04	0.2870415E 04	0.5334467E 03	0.9550550E 04	0.4307885E 01	0.6702628E 07	5
0.9800001E 03	0.2750501E 01	0.8707076E 04	0.2378487E 04			1
0.1415707E 06	0.3360983E 04	0.3254349E 01	0.1895097E 01	0.1951542E 05	0.5826413E 06	2
0.6421539E 05	0.8536897E 02	0.8266046E-01	0.4813546E-01	0.1913809E 06	0.5713758E 07	3
0.8337777E 04	0.2725197E 04	0.3421448E 03	0.2708290E 04	0.7045358E 02	0.5827840E 06	4
0.2333487E 04	0.2853462E 04	0.5452401E 03	0.9925667E 04	0.4307351E 01	0.5715157E 07	5
0.9931600E 03	0.2861974E 01	0.3428300E 04	0.1315000E 04	2ND CUTOFF SIGNAL		1
0.1352363E 06	0.3389966E 04	0.3403098E 01	0.1984895E 01	0.1951069E 05	0.5062204E 06	2
0.6134214E 05	0.8610513E 02	0.8643868E-01	0.5041633E-01	0.1913344E 06	0.4964326E 07	3
0.3059000E 04	0.2715924E 04	0.3470000E 03	0.2783207E 04	0.7044000E 02	0.5063695E 06	4
0.1270000E 04	0.2838034E 04	0.5530001E 03	0.1017254E 05	0.4307000E 01	0.4965788E 07	5
0.9931610E 03	0.2863653E 01	0.3365300E 04	0.1285000E 04	2ND E.T.D.		1
0.1351433E 06	0.3390402E 04	0.3405440E 01	0.1986261E 01	0.1951064E 05	0.5050781E 06	2
0.6129996E 05	0.8611622E 02	0.8649817E-01	0.5045103E-01	0.1913340E 06	0.4953123E 07	3
0.2996000E 04	0.2715775E 04	0.3470000E 03	0.2784102E 04	0.7044000E 02	0.5052271E 06	4
0.1240000E 04	0.2837495E 04	0.5530001E 03	0.1017950E 05	0.4307000E 01	0.4954585E 07	5

TABLE X

BASELINE VEHICLE WEIGHT DESCRIPTION

I.	S-IC Stage			
	Total Mainstage Propellant		4, 577, 113	LBS.
	S-IC Stage (dry)		289, 409	
	S-IC Residuals		67, 585	
	S-IC Thrust Decay Propellant		9, 068	
	Inboard Eng T. D.	1914		
	Inboard Eng Exp. Prop.	408		
	Outboard Eng T. D.	6746		
	S-IC/S-II Interstage (small)		1, 548	
	S-II Ullage Rocket Propellant		1, 360	
	S-II Thrust Buildup		1, 801	
	S-IC/S-II Large Interstage		9, 435	
	Launch Escape System		8, 936	
II.	S-II Stage			
	Total Mainstage Propellant		970, 442*	
	S-II Stage (dry)		80, 686	
	S-II Residuals		14, 136	
	S-II Thrust Decay Propellant		342	
	S-II/S-IVB Interstage and Retro Propellant		8, 086	
	S-IVB Aft Frame & Detonation Package (Sep with Interstage)		51	
	S-IVB Ullage Rocket Propellant		117	
	S-IVB Thrust Buildup Propellant		440	
III.	S-IVB Stage			
	S-IVB Weight Lost In Parking Orbit (100 N.MI. for 3 Orbits)		4, 411	
	S-IVB Thrust Decay Propellant	107		
	LH_2 Vented	3, 281		
	GOX Vented	170		
	Auxiliary Prop. Losses and Ullage for Restart	438		
	First Burn Propellant	18		
	S-IVB Restart Thrust Buildup Propellant	397		
	S-IVB Mainstage Prop (Incl FPR and FGR)		227, 991**	
	S-IVB Stage (dry, Less Aft Frame & Detonation Pkg.)		25, 033	
	S-IVB Residuals		2, 508	
	S-IVB Thrust Decay Propellant		93	
	S-IVB Second Burn Roll Propellant		34	

TABLE X (Cont'd)

BASELINE VEHICLE WEIGHT DESCRIPTION

Instrument Unit	4,183	LBS.
Flight Geometry Reserves (=39 m/sec)	1,211	
Flight Performance Reserves (= 70 m/sec)	2,260	
Net Payload	100,078	

*S-II LOX Loading = 815,427
S-II LH_2 Loading = 155,014

**S-IVB LOX Loading = 190,914
S-IVB LH_2 Loading = 37,077

TABLE XI

PROPULSION DATA

S-IC Stage

Thrust/eng = 1, 522, 000 lb
$\dot{w}$/eng = 5754. 2533 lb/sec
Ae/eng = 9. 9313349 m^2/eng
Cross sectional area used for Aero computation = 79. 45976 m^2

S-II Stage

MR = 5.0:1
F = 205052 lb/eng
$\dot{w}$/eng = 481. 34272 lb/sec

MR = 5. 5:1
F = 229927 lb/eng
$\dot{w}$/eng = 543. 69117 lb/sec

MR = 4. 702:1
F = 190125 lb/eng
$\dot{w}$ /eng = 445. 25761 lb/sec

S-IVB Stage

MR = 5. 5:1
F = 231012 lb/eng
$\dot{w}$ = 543. 68557 lb/sec

MR = 5. 0:1
F = 206012 lb/eng
$\dot{w}$ = 481. 33644 lb/sec

The S-IVB stage burns at MR = 5. 5:1 from ignition to orbital insertion. It is restarted and burns to injection at MR = 5. 0:1

TABLE XII

EVENT HISTORY

Time (sec)	Event
0	Lift-off
12	Initial Tilt Program
35	Terminate Tilt Program, Begin $\alpha_p = 0$ flight
149.486	Inboard Engine Cutoff
153.0	Initiate 'chi freeze' Mode
161.486	Outboard Engine Cutoff (Separation)
165.286	Ignite S-II at MR = 5.0:1
167.786	Shift MR to 5.5:1
193.986	Jettison S-IC/S-II Interstage
198.986	Jettison LES, Initiate Guidance
426.312	MR Shift to 4.7:1
543.832	S-II Cutoff/Separation S-IVB Ignition @ MR = 5.5:1
676.027	S-IVB Cutoff in 100 N. M. Parking Orbit S-IVB Reignition @ MR = 5.0:1
993.160	S-IVB Final Cutoff

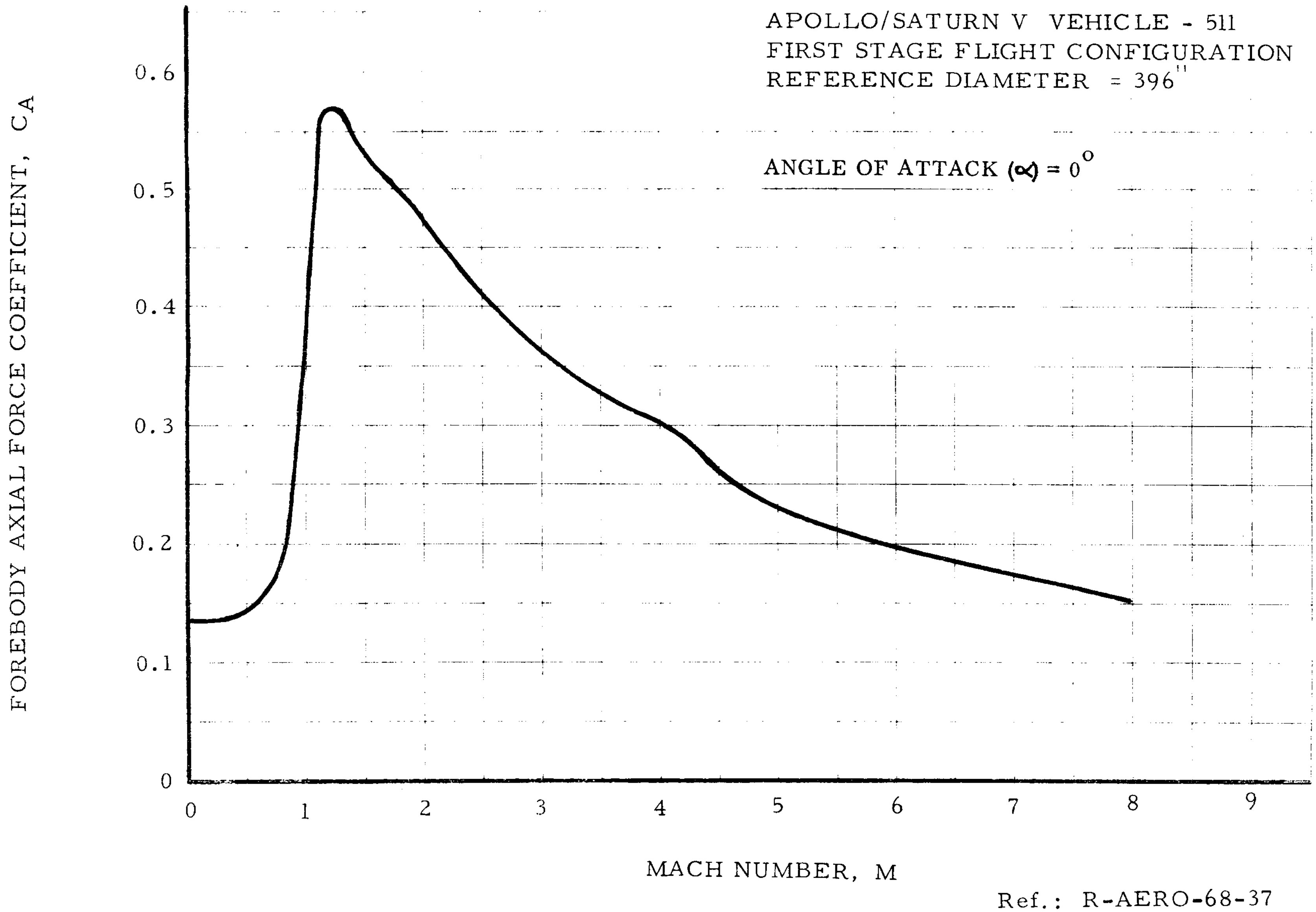


FIGURE 2 VARIATION OF FOREBODY AXIAL FORCE COEFFICIENT

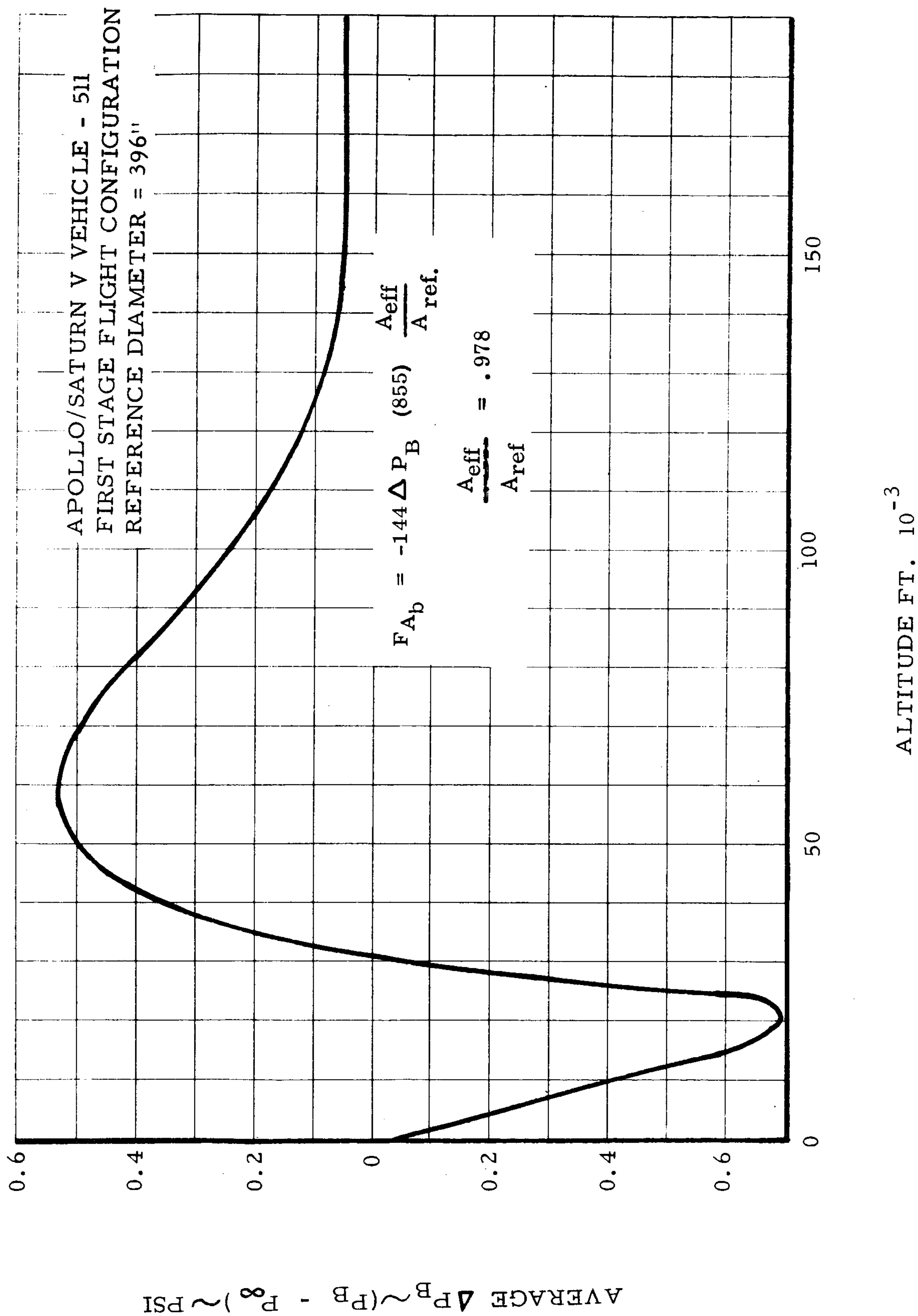
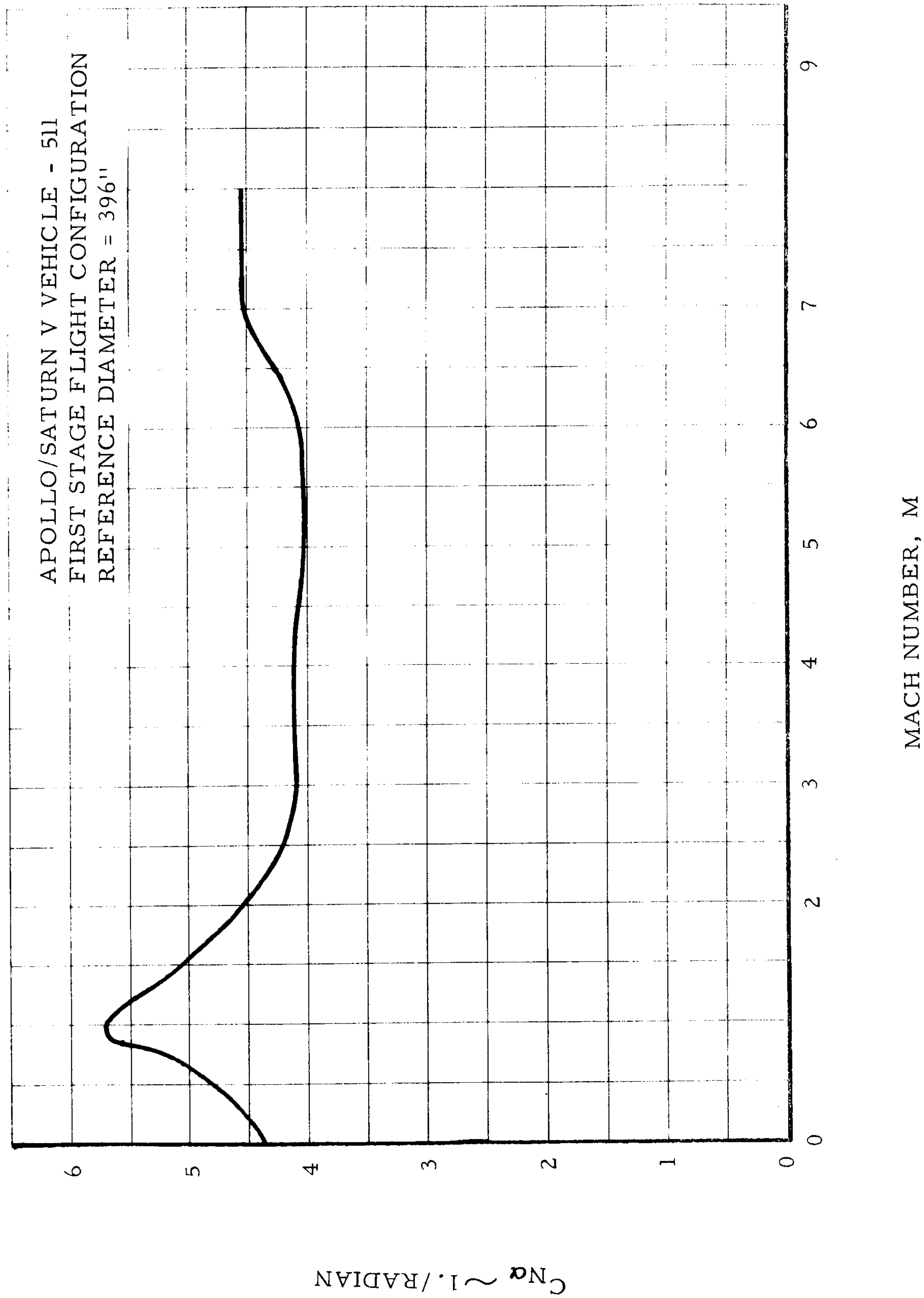


FIGURE 3 VARIATION OF AVERAGE BASE PRESSURE

Ref. R-AERO-AD-68-37



Ref.: NASA TMX 53517

FIGURE 4 NORMAL FORCE COEFFICIENT

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TABLE NO. 1

TABLE XIII
BASELINE LOR TRAJECTORY DATA

	TIME SEC	INERTIAL VELOCITY M/SEC	INERTIAL VELOCITY FT/SEC	RELATIVE VELOCITY M/SEC	RELATIVE VELOCITY FT/SEC	GAMMA S DEG	GAMMA R DEG
LIFT-OFF	.000	408.633	1340.661	.000	.000	-.000	-1.515
	4.000	408.707	1340.903	7.821	25.661	1.097	89.837
	5.000	408.752	1341.050	9.920	32.547	1.391	89.839
	10.000	409.181	1342.456	21.281	69.819	2.981	89.835
BEGIN TILT	12.000	409.466	1343.392	26.247	86.112	3.675	89.833
	15.000	410.103	1345.481	34.173	112.116	4.780	89.763
	20.000	412.229	1352.457	48.728	159.870	6.788	89.048
	25.000	415.874	1364.415	65.103	213.593	9.001	87.974
	30.000	421.275	1382.137	83.469	273.850	11.410	86.827
	35.000	428.698	1406.491	104.010	341.239	14.001	85.704
END TILT	35.000	428.698	1406.491	104.010	341.239	14.001	85.704
	40.000	440.674	1445.782	126.979	416.596	16.636	83.506
	45.000	457.270	1500.230	152.676	500.905	19.247	80.850
	50.000	479.208	1572.205	181.438	595.270	21.720	77.806
	55.000	507.044	1663.529	213.647	700.941	23.951	74.462
	60.000	541.122	1775.335	249.679	819.157	25.852	70.913
	65.000	581.582	1908.079	289.884	951.063	27.366	67.255
	70.000	628.675	2060.615	334.163	1096.335	28.452	63.568
	75.000	680.475	2232.529	382.824	1255.985	29.131	59.918
10 KMS.	75.107	681.670	2236.449	383.927	1259.604	29.142	59.840
	80.000	739.646	2426.660	437.323	1434.787	29.489	56.363
Q MAXIMUM	83.358	783.330	2569.962	477.526	1566.688	29.572	54.054
	85.000	805.828	2643.792	498.253	1634.688	29.571	52.953
14 KMS.	86.973	819.521	2688.716	510.880	1676.115	29.558	52.309
	90.000	878.986	2883.812	565.849	1856.458	29.414	49.721
	95.000	959.183	3146.926	640.397	2101.042	29.065	46.688
	100.000	1046.507	3433.422	722.159	2369.289	28.565	43.862
	105.000	1141.007	3743.462	811.300	2661.745	27.953	41.243
	110.000	1242.810	4077.459	908.024	2979.082	27.262	38.825
	115.000	1352.145	4436.170	1012.610	3322.210	26.521	36.601
	120.000	1469.345	4820.685	1125.411	3692.294	25.752	34.559
	125.000	1594.854	5232.462	1246.878	4090.808	24.975	32.688
	130.000	1729.318	5673.614	1377.649	4519.845	24.206	30.976
	135.000	1873.488	6146.616	1518.462	4981.831	23.455	29.412

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd) TABLE NO. 1

EVENT	TIME SEC	INERTIAL VELOCITY M/SEC	INERTIAL VELOCITY FT/SEC	RELATIVE VELOCITY M/SEC	RELATIVE VELOCITY FT/SEC	GAMMA S DEG	GAMMA R DEG
	140.000	2028.292	6654.500	1670.224	5479.737	22.731	27.986
	145.000	2194.855	7200.967	1834.039	6017.188	22.043	26.688
	149.486	2355.454	7727.867	1992.404	6536.758	21.458	25.625
IECO	149.486	2355.454	7727.867	1992.404	6536.758	21.458	25.625
	150.000	2370.375	7776.820	2007.065	6584.860	21.386	25.509
BEGIN CHIFRZ	153.000	2459.588	8069.514	2094.812	6872.741	20.973	24.850
	155.000	2521.094	8271.306	2155.402	7071.529	20.711	24.435
	160.000	2682.354	8800.375	2314.661	7594.031	20.138	23.514
OECO	161.486	2732.460	8964.829	2364.263	7756.769	19.989	23.271
SEPARATION	161.486	2732.460	8964.829	2364.263	7756.769	19.989	23.271
	165.000	2721.033	8927.272	2351.055	7713.435	19.402	22.611
	165.286	2720.116	8924.264	2349.996	7709.961	19.354	22.556
S-II IGNITION	165.286	2720.116	8924.264	2349.996	7709.961	19.354	22.556
	167.786	2729.400	8954.723	2358.142	7736.687	18.972	22.104
MR SHIFT	167.786	2729.400	8954.723	2358.142	7736.687	18.972	22.104
	170.000	2739.652	8988.361	2367.424	7767.140	18.642	21.710
	180.000	2788.164	9147.519	2411.857	7912.918	17.196	19.985
	190.000	2840.177	9318.168	2460.268	8071.746	15.827	18.352
	193.986	2861.860	9389.304	2480.639	8138.579	15.303	17.727
WEIGHT DROP	193.986	2861.860	9389.304	2480.639	8138.579	15.303	17.727
	198.986	2890.082	9481.896	2507.312	8226.090	14.664	16.966
WEIGHT DROP	198.986	2890.082	9481.896	2507.312	8226.090	14.664	16.966
INITIATE GUIDANCE	198.986	2890.082	9481.896	2507.312	8226.090	14.664	16.966
	200.000	2895.764	9500.537	2512.724	8243.845	14.553	16.832
	210.000	2953.599	9690.286	2568.051	8425.364	13.480	15.551
	220.000	3014.678	9890.676	2626.889	8618.402	12.457	14.332
	230.000	3079.001	10101.710	2689.219	8822.895	11.483	13.176
	240.000	3146.577	10323.414	2755.027	9038.803	10.559	12.081
	250.000	3217.420	10555.838	2824.311	9266.111	9.684	11.047
	260.000	3291.553	10799.060	2897.073	9504.832	8.857	10.075
	270.000	3369.012	11053.190	2973.330	9755.019	8.077	9.161
	280.000	3449.837	11318.363	3053.106	10016.750	7.345	8.306
	290.000	3534.080	11594.751	3136.436	10290.144	6.658	7.507
	300.000	3621.806	11882.566	3223.371	10575.363	6.016	6.763

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd) TABLE NO. 1

	TIME SEC	INERTIAL VELOCITY M/SEC	INERTIAL VELOCITY FT/SEC	RELATIVE VELOCITY M/SEC	RELATIVE VELOCITY FT/SEC	GAMMA S DEG	GAMMA R DEG
	310.000	3713.091	12182.057	3313.973	10872.614	5.419	6.073
	320.000	3808.022	12493.512	3408.317	11182.142	4.864	5.436
	330.000	3906.703	12817.268	3506.494	11504.246	4.350	4.848
	340.000	4009.252	13153.713	3608.613	11839.280	3.878	4.309
	350.000	4115.601	13503.285	3714.796	12187.650	3.445	3.818
	360.000	4226.504	13866.482	3825.187	12549.827	3.051	3.372
	370.000	4341.530	14243.866	3939.951	12926.348	2.695	2.970
	380.000	4461.075	14636.073	4059.273	13317.825	2.376	2.611
	390.000	4585.356	15043.817	4183.366	13724.954	2.092	2.293
	400.000	4714.614	15467.895	4312.466	14148.511	1.843	2.015
	410.000	4849.127	15909.210	4446.844	14589.385	1.629	1.776
	420.000	4989.204	16368.778	4586.806	15048.576	1.448	1.575
	426.312	5080.642	16668.773	4678.180	15348.361	1.351	1.467
MR SHIFT	426.312	5080.642	16668.773	4678.180	15348.361	1.351	1.467
	430.000	5125.556	16816.130	4723.053	15495.580	1.270	1.378
	440.000	5250.430	17225.819	4847.827	15904.943	1.071	1.160
	450.000	5379.988	17650.880	4977.303	16329.734	.898	.971
	460.000	5514.487	18092.150	5111.734	16770.782	.753	.812
	470.000	5654.211	18550.562	5251.401	17229.006	.634	.682
	480.000	5799.480	19027.165	5396.622	17705.453	.540	.581
	490.000	5950.452	19523.136	5547.753	18201.289	.472	.507
	500.000	6106.126	20039.783	5705.190	18717.816	.430	.460
	510.000	6272.357	20578.598	5869.388	19256.522	.412	.441
	520.000	6443.858	21141.267	6040.855	19819.079	.419	.447
	530.000	6623.211	21729.694	6220.175	20407.397	.451	.480
	540.000	6811.082	22346.071	6408.010	21023.656	.507	.539
S-II CUTOFF	543.832	6885.499	22590.219	6482.411	21267.754	.535	.568
S-IVB IGNITION	543.832	6885.499	22590.219	6482.411	21267.754	.535	.568
	550.000	6923.093	22713.558	6519.968	21390.970	.479	.509
	560.000	6984.985	22916.618	6581.806	21593.851	.396	.420
	570.000	7048.053	23123.533	6644.827	21800.613	.320	.340
	580.000	7112.311	23334.354	6709.045	22011.301	.253	.268
	590.000	7177.777	23549.137	6774.476	22225.972	.193	.204
	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 1

	TIME SEC	INERTIAL VELOCITY M/SEC	INERTIAL VELOCITY FT/SEC	RELATIVE VELOCITY M/SEC	RELATIVE VELOCITY FT/SEC	GAMMA S DEG	GAMMA R DEG
	610.000	7312.403	23990.823	6909.051	22667.491	.097	.102
	620.000	7381.602	24217.853	6978.231	22694.460	.060	.064
	630.000	7452.087	24449.104	7048.701	23125.661	.031	.033
	640.000	7523.882	24684.653	7120.484	23361.168	.010	.011
	650.000	7597.011	24924.576	7193.603	23601.060	-.003	-.004
	660.000	7671.500	25168.963	7268.084	23845.420	-.009	-.010
	670.000	7747.379	25417.909	7343.956	24094.343	-.008	-.008
	676.027	7793.793	25570.187	7390.367	24246.610	-.003	-.003
INJECTION	676.027	7793.793	25570.187	7390.367	24246.610	-.003	-.003
THRUST EVENT	676.027	7793.793	25570.187	7390.367	24246.610	-.003	-.003
	680.000	7821.704	25661.761	7418.276	24338.175	-.014	-.015
	690.000	7892.826	25895.100	7489.394	24571.503	-.033	-.035
	700.000	7965.211	26132.583	7561.776	24808.976	-.038	-.040
	710.000	8038.876	26374.266	7635.441	25050.658	-.028	-.029
	720.000	8113.844	26620.223	7710.407	25296.610	-.004	-.004
	730.000	8190.138	26870.532	7786.698	25546.909	.034	.036
	740.000	8267.785	27125.279	7864.340	25801.642	.087	.091
	750.000	8346.815	27384.564	7943.361	26060.896	.154	.161
	760.000	8427.260	27648.491	8023.792	26324.777	.235	.247
	770.000	8509.156	27917.177	8105.668	26593.400	.331	.347
	780.000	8592.541	28190.751	8189.028	26866.890	.442	.463
	790.000	8677.459	28469.355	8273.913	27145.384	.567	.595
	800.000	8763.958	28753.142	8360.370	27429.036	.707	.741
	810.000	8852.090	29042.289	8448.451	27718.015	.862	.903
	820.000	8941.909	29336.973	8538.210	28012.499	1.032	1.081
	830.000	9033.480	29637.403	8629.709	28312.695	1.217	1.274
	840.000	9126.871	29943.804	8723.017	28618.822	1.418	1.483
	850.000	9222.157	30256.420	8818.206	28931.123	1.633	1.708
	860.000	9319.418	30575.520	8915.358	29249.864	1.864	1.949
	870.000	9418.746	30901.398	9014.563	29575.339	2.111	2.206
	880.000	9520.240	31234.384	9115.922	29907.880	2.373	2.478
	890.000	9624.011	31574.839	9219.540	30247.834	2.651	2.768
	900.000	9730.172	31923.136	9325.535	30595.587	2.945	3.073
	910.000	9838.848	32279.686	9434.029	30951.537	3.255	3.395

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TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 1

	TIME SEC	INERTIAL VELOCITY M/SEC	INERTIAL VELOCITY FT/SEC	RELATIVE VELOCITY M/SEC	RELATIVE VELOCITY FT/SEC	GAMMA S DEG	GAMMA R DEG
	920.000	9950.192	32644.989	9545.174	31316.186	3.581	3.733
	930.000	10064.365	33019.569	9659.130	31690.057	3.923	4.088
	940.000	10181.534	33403.983	9776.066	32073.706	4.282	4.460
	950.000	10301.883	33798.830	9896.166	32467.735	4.657	4.849
	960.000	10425.641	34204.860	10019.655	32872.885	5.050	5.255
	970.000	10553.050	34622.867	10146.777	33289.953	5.459	5.678
	980.000	10684.361	35053.677	10277.784	33719.762	5.885	6.119
	990.000	10819.850	35498.197	10412.951	34163.225	6.329	6.577
INJECTION	993.160	10863.590	35641.700	10456.585	34306.382	6.473	6.726
	993.160	10963.590	35641.700	10456.585	34306.382	6.473	6.726

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CASE 1

TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 2

	TIME SEC	MASS KGS	WEIGHT LBS	THRUST N	THRUST LBS	LONGIT. ACCELERATION M/(SEC)SQ	LONGIT. ACCELERATION FT/(SEC)SQ
LIFT-OFF	.000	2905206.2	6404883.3	33850966.0	7609999.8	11.63	38.15
	4.000	2853009.5	6289798.2	33859759.0	7611976.6	11.86	38.92
	5.000	2839954.0	6261026.9	33864582.0	7613061.1	11.92	39.10
	10.000	2774701.9	6117170.6	33909011.0	7623049.0	12.21	40.07
BEGIN TILT	12.000	2748601.1	6059628.0	33936358.0	7629196.8	12.34	40.48
	15.000	2709449.8	5973314.2	33987589.0	7640714.0	12.53	41.11
	20.000	2644197.6	5829457.8	34102711.0	7666594.5	12.88	42.25
	25.000	2578945.5	5685601.5	34258062.0	7701518.6	13.25	43.48
	30.000	2513693.3	5541745.2	34454124.0	7745595.4	13.66	44.82
	35.000	2448441.2	5397888.8	34691548.0	7798970.3	14.10	46.27
END TILT	35.000	2448441.2	5397888.8	34691548.0	7798970.3	14.10	46.27
	40.000	2383189.1	5254032.4	34968963.0	7861335.4	14.58	47.83
	45.000	2317936.9	5110176.1	35282806.0	7931890.3	15.09	49.52
	50.000	2252684.8	4966319.8	35627250.0	8009324.5	15.65	51.33
	55.000	2187432.6	4822463.5	35994541.0	8091894.7	16.23	53.26
	60.000	2122180.5	4678607.2	36375497.0	8177537.1	16.86	55.30
	65.000	2056928.4	4534750.9	36759125.0	8263780.0	17.48	57.36
	70.000	1991676.2	4390894.5	37134700.0	8348212.6	17.97	58.95
	75.000	1926424.1	4247038.2	37489735.0	8428027.6	18.74	61.49
10 KMS.	75.107	1925027.1	4243958.3	37497081.0	8429679.1	18.76	61.55
	80.000	1861172.0	4103181.9	37814149.0	8500958.8	19.66	64.51
Q MAXIMUM	83.358	1817342.4	4006554.2	38009192.0	8544806.2	20.31	66.63
	85.000	1795919.8	3959325.6	38096908.0	8564525.5	20.63	67.69
14 KMS.	85.273	1783219.0	3931325.0	38146397.0	8575651.3	20.83	68.32
	90.000	1730667.7	3815469.2	38329862.0	8616895.8	21.64	71.01
	95.000	1665415.6	3671612.9	38508763.0	8657114.4	22.72	74.53
	100.000	1600163.4	3527756.6	38636746.0	8685885.7	23.83	78.19
	105.000	1534911.3	3383900.2	38723897.0	8705928.0	24.99	82.00
	110.000	1469659.2	3240043.9	38786624.0	8719579.9	26.22	86.02
	115.000	1404407.1	3096187.6	38827392.0	8728745.1	27.52	90.28
	120.000	1339154.9	2952331.2	38854309.0	8734796.1	28.91	94.86
	125.000	1273902.6	2808475.0	38871699.0	8738705.5	30.44	99.86
	130.000	1208650.7	2664618.6	38882785.0	8741197.6	32.12	105.37
	135.000	1143398.5	2520762.3	38889746.0	8742767.3	33.98	111.47

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd) TABLE NO. 2

	TIME SEC	MASS KGS	WEIGHT LBS	THRUST N	THRUST LBS	LONGIT. ACCELERATION M/(SEC)SQ	LONGIT. ACCELERATION FT/(SEC)SQ
	140.000	1078146.4	2376905.9	38894139.0	8743750.4	36.05	118.28
	145.000	1012894.3	2233049.6	38896886.0	8744367.6	38.39	125.96
	149.486	954346.8	2103974.5	38898487.0	8744727.6	40.76	133.73
IECO	149.486	954346.8	2103974.5	31118790.0	6995782.3	32.61	106.98
	150.000	948983.1	2092149.6	31118906.0	6995808.3	32.79	107.59
BEGIN CHIFRZ	153.000	917662.0	2023098.5	31119479.0	6995937.2	33.92	111.28
	155.000	896781.3	1977064.5	31119776.0	6996003.9	34.71	113.88
	160.000	844579.6	1861979.4	31120296.0	6996120.9	36.86	120.94
OECO	161.486	829062.7	1827770.3	31120403.0	6996145.0	37.56	123.21
SEPARATION	161.486	660883.8	1456999.3	.0	.0	-.02	-.05
	165.000	660883.8	1456999.3	.0	.0	-.01	-.03
	165.286	660883.8	1456999.3	.0	.0	-.01	-.03
S-II IGNITION	165.286	660883.8	1456999.3	4560583.6	1025260.0	6.90	22.64
	167.786	658154.6	1450982.5	4560583.6	1025260.0	6.93	22.73
MR SHIFT	167.786	658154.6	1450982.5	5113831.1	1149635.0	7.77	25.49
	170.000	655424.9	1444964.6	5113831.1	1149635.0	7.80	25.60
	180.000	643094.2	1417780.0	5113831.1	1149635.0	7.95	26.09
	190.000	630763.5	1390595.4	5113831.1	1149635.0	8.11	26.60
	193.986	625848.2	1379759.0	5113831.1	1149635.0	8.17	26.81
WEIGHT DROP	193.986	621568.5	1370324.0	5113831.1	1149635.0	8.23	26.99
	198.986	615403.2	1356731.7	5113831.1	1149635.0	8.31	27.26
WEIGHT DROP	198.986	611349.9	1347795.7	5113831.1	1149635.0	8.36	27.44
INITIATE	198.986	611349.9	1347795.7	5113831.1	1149635.0	8.36	27.44
GUIDANCE	200.000	610099.8	1345039.8	5113831.1	1149635.0	8.38	27.50
	210.000	597769.1	1317855.3	5113831.1	1149635.0	8.55	28.07
	220.000	585438.4	1290670.7	5113831.1	1149635.0	8.74	28.66
	230.000	573107.7	1263486.2	5113831.1	1149635.0	8.92	29.27
	240.000	560777.0	1236301.7	5113831.1	1149635.0	9.12	29.92
	250.000	548446.3	1209117.1	5113831.1	1149635.0	9.32	30.59
	260.000	536115.6	1181932.5	5113831.1	1149635.0	9.54	31.29
	270.000	523784.9	1154748.0	5113831.1	1149635.0	9.76	32.03
	280.000	511454.2	1127563.4	5113831.1	1149635.0	10.00	32.80
	290.000	499123.5	1100378.9	5113831.1	1149635.0	10.25	33.61
	300.000	486792.7	1073194.3	5113831.1	1149635.0	10.51	34.47

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 2

	TIME SEC	MASS KGS	WEIGHT LBS	THRUST N	THRUST LBS	LONGIT. ACCELERATION M/(SEC)SQ	LONGIT. ACCELERATION FT/(SEC)SQ
	310.000	474462.0	1046009.7	5113831.1	1149635.0	10.78	35.36
	320.000	462131.3	1018825.2	5113831.1	1149635.0	11.07	36.30
	330.000	449800.6	991640.6	5113831.1	1149635.0	11.37	37.30
	340.000	437469.9	964456.1	5113831.1	1149635.0	11.69	38.35
	350.000	425139.2	937271.5	5113831.1	1149635.0	12.03	39.46
	360.000	412808.5	910086.9	5113831.1	1149635.0	12.39	40.64
	370.000	400477.8	882902.4	5113831.1	1149635.0	12.77	41.89
	380.000	388147.1	855717.8	5113831.1	1149635.0	13.17	43.23
	390.000	375816.4	828533.3	5113831.1	1149635.0	13.61	44.64
	400.000	363485.7	801348.7	5113831.1	1149635.0	14.07	46.16
	410.000	351155.0	774164.2	5113831.1	1149635.0	14.56	47.78
	420.000	338824.2	746979.6	5113831.1	1149635.0	15.09	49.52
	426.312	331040.8	729820.0	5113831.1	1149635.0	15.45	50.68
MR SHIFT	426.312	331040.8	729820.0	4228590.6	950625.0	12.77	41.91
	430.000	327316.8	721610.0	4228590.6	950625.0	12.92	42.39
	440.000	317218.5	699347.1	4228590.6	950625.0	13.33	43.73
	450.000	307120.3	677084.3	4228590.6	950625.0	13.77	45.17
	460.000	297022.0	654821.4	4228590.6	950625.0	14.24	46.71
	470.000	286923.7	632558.5	4228590.6	950625.0	14.74	48.35
	480.000	276825.4	610295.6	4228590.6	950625.0	15.28	50.12
	490.000	266727.2	588032.7	4228590.6	950625.0	15.85	52.01
	500.000	256628.9	565769.9	4228590.6	950625.0	16.48	54.06
	510.000	246530.6	543507.0	4228590.6	950625.0	17.15	56.27
	520.000	236432.4	521244.1	4228590.6	950625.0	17.88	58.68
	530.000	226334.1	498981.2	4228590.6	950625.0	18.68	61.30
	540.000	216235.8	476718.4	4228590.6	950625.0	19.56	64.16
S-II CUTOFF	543.832	212365.8	468186.4	4228590.6	950625.0	19.91	65.33
S-IVB IGNITION	543.832	165257.9	364331.4	1027592.6	231012.0	6.22	20.40
	550.000	163736.9	360978.1	1027592.6	231012.0	6.28	20.59
	560.000	161270.8	355541.3	1027592.6	231012.0	6.37	20.91
	570.000	158804.7	350104.4	1027592.6	231012.0	6.47	21.23
	580.000	156338.6	344667.6	1027592.6	231012.0	6.57	21.56
	590.000	153872.5	339230.7	1027592.6	231012.0	6.68	21.91
	600.000	151406.3	333793.8	1027592.6	231012.0	6.79	22.27

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd) TABLE NO. 2

	TIME	MASS	WEIGHT	THRUST		LONGIT. ACCELERATION	
	SEC	KGS	LBS	N	LBS	M/(SEC)SQ	FT/(SEC)SQ
	610.000	148940.2	328357.0	1027592.6	231012.0	6.90	22.64
	620.000	146474.1	322920.1	1027592.6	231012.0	7.02	23.02
	630.000	144008.0	317483.3	1027592.6	231012.0	7.14	23.41
	640.000	141541.9	312046.4	1027592.6	231012.0	7.26	23.82
	650.000	139075.8	306609.6	1027592.6	231012.0	7.39	24.24
	660.000	136609.6	301172.7	1027592.6	231012.0	7.52	24.68
	670.000	134143.5	295735.9	1027592.6	231012.0	7.66	25.13
	676.027	132657.2	292459.1	1027592.6	231012.0	7.75	25.41
INJECTION	676.027	130656.4	288048.1	916387.0	206012.0	7.01	23.01
THRUST EVENT	676.027	130656.4	288048.1	916387.0	206012.0	7.01	23.01
	680.000	129789.0	286135.7	916387.0	206012.0	7.06	23.16
	690.000	127605.7	281322.4	916387.0	206012.0	7.18	23.56
	700.000	125422.4	276509.0	916387.0	206012.0	7.31	23.97
	710.000	123239.1	271695.6	916387.0	206012.0	7.44	24.40
	720.000	121055.8	266882.3	916387.0	206012.0	7.57	24.84
	730.000	118872.4	262068.9	916387.0	206012.0	7.71	25.29
	740.000	116689.1	257255.5	916387.0	206012.0	7.85	25.77
	750.000	114505.8	252442.2	916387.0	206012.0	8.00	26.26
	760.000	112322.5	247628.8	916387.0	206012.0	8.16	26.77
	770.000	110139.2	242815.4	916387.0	206012.0	8.32	27.30
	780.000	107955.9	238002.1	916387.0	206012.0	8.49	27.85
	790.000	105772.6	233188.7	916387.0	206012.0	8.66	28.42
	800.000	103589.3	228375.3	916387.0	206012.0	8.85	29.02
	810.000	101406.0	223562.0	916387.0	206012.0	9.04	29.65
	820.000	99222.7	218748.6	916387.0	206012.0	9.24	30.30
	830.000	97039.4	213935.2	916387.0	206012.0	9.44	30.98
	840.000	94856.1	209121.9	916387.0	206012.0	9.66	31.70
	850.000	92672.8	204308.5	916387.0	206012.0	9.89	32.44
	860.000	90489.5	199495.2	916387.0	206012.0	10.13	33.23
	870.000	88306.2	194681.8	916387.0	206012.0	10.38	34.05
	880.000	86122.9	189868.4	916387.0	206012.0	10.64	34.91
	890.000	83939.6	185055.1	916387.0	206012.0	10.92	35.82
	900.000	81756.3	180241.7	916387.0	206012.0	11.21	36.77
	910.000	79573.0	175428.3	916387.0	206012.0	11.52	37.78

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 2

	TIME	MASS	WEIGHT	THRUST		LONGIT. ACCELERATION	
	SEC	KGS	LBS	N	LBS	M/(SEC)SQ	FT/(SEC)SQ
	920.000	77389.6	170615.0	916387.0	206012.0	11.84	38.85
	930.000	75206.3	165801.6	916387.0	206012.0	12.18	39.98
	940.000	73023.0	160988.2	916387.0	206012.0	12.55	41.17
	950.000	70839.7	156174.9	916387.0	206012.0	12.94	42.44
	960.000	68656.4	151361.5	916387.0	206012.0	13.35	43.79
	970.000	66473.1	146548.1	916387.0	206012.0	13.79	45.23
	980.000	64289.8	141734.8	916387.0	206012.0	14.25	46.77
	990.000	62106.5	136921.4	916387.0	206012.0	14.76	48.41
INJECTION	993.160	61416.5	135400.3	916387.0	206012.0	14.92	48.95
	993.160	45394.8	100078.3	.0	.0	.00	.00

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd) TABLE NO. 3

	TIME SEC	ALTITUDE M	ALTITUDE FT	RADIUS M	RADIUS FT	INCL DEG
LIFT-OFF	.000	-.75	-2.5	6373293.70	20909756.0	28.4470
	4.000	15.25	50.0	6373309.70	20909809.0	28.4470
	5.000	23.62	77.5	6373318.10	20909836.0	28.4470
	10.000	101.06	331.6	6373395.50	20910090.0	28.4470
BEGIN TILT	12.000	149.00	488.8	6373443.50	20910247.0	28.4470
	15.000	239.38	785.4	6373533.80	20910544.0	28.4470
	20.000	445.25	1460.8	6373739.80	20911220.0	28.4471
	25.000	729.50	2393.4	6374023.90	20912152.0	28.4474
	30.000	1099.56	3607.5	6374394.00	20913366.0	28.4482
	35.000	1566.19	5138.4	6374860.60	20914897.0	28.4499
END TILT	35.000	1566.19	5138.4	6374860.60	20914897.0	28.4499
	40.000	2140.06	7021.2	6375434.30	20916779.0	28.4551
	45.000	2831.75	9290.5	6376126.00	20919049.0	28.4672
	50.000	3651.38	11979.6	6376945.50	20921737.0	28.4916
	55.000	4608.69	15120.4	6377902.60	20924877.0	28.5351
	60.000	5713.12	18743.8	6379006.60	20928499.0	28.6048
	65.000	6970.69	22869.7	6380263.70	20932624.0	28.7056
	70.000	8388.37	27520.9	6381680.90	20937273.0	28.8387
	75.000	9964.87	32693.2	6383256.60	20942443.0	29.0019
10 KMS.	75.107	10000.62	32810.4	6383292.40	20942561.0	29.0057
	80.000	11704.06	38399.2	6384994.80	20948146.0	29.1922
Q MAXIMUM	83.358	12964.69	42535.1	6386254.70	20952279.0	29.3321
	85.000	13609.13	44649.4	6386898.80	20954392.0	29.4031
14 KMS.	85.973	14000.06	45932.0	6387289.40	20955674.0	29.4458
	90.000	15684.19	51457.3	6388972.30	20961195.0	29.6265
	95.000	17929.38	58823.4	6391215.70	20968556.0	29.8549
	100.000	20347.81	66757.9	6393632.10	20976483.0	30.0817
	105.000	22938.44	75257.3	6396220.30	20984975.0	30.3017
	110.000	25701.38	84322.1	6398980.30	20994030.0	30.5115
	115.000	28637.56	93955.3	6401913.20	21003652.0	30.7089
	120.000	31746.63	104155.6	6405018.60	21013840.0	30.8931
	125.000	35029.56	114926.4	6408297.20	21024597.0	31.0638
	130.000	38490.50	126281.2	6411753.40	21035936.0	31.2216
	135.000	42132.13	138228.8	6415389.70	21047866.0	31.3671

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BASELINE SA 511 TO LOR CASE 1

TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 3

	TIME SEC	ALTITUDE M	ALTITUDE FT	RADIUS M	RADIUS FT	INCL DEG
	140.000	45960.88	150790.3	6419212.40	21060408.0	31.5012
	145.000	49985.44	163994.2	6423230.30	21073590.0	31.6248
	149.486	53771.75	176416.5	6427010.20	21085992.0	31.7276
IECO	149.486	53771.75	176416.5	6427010.20	21085992.0	31.7276
	150.000	54215.44	177872.2	6427453.10	21087445.0	31.7367
BEGIN CHIFRZ	153.000	56837.25	186473.9	6430070.30	21096031.0	31.7884
	155.000	58612.25	192297.4	6431841.90	21101844.0	31.8217
	160.000	63156.88	207207.6	6436378.00	21116725.0	31.9014
OECO	161.486	64539.25	211742.9	6437757.70	21121252.0	31.9241
SEPARATION	161.486	64539.25	211742.9	6437757.70	21121252.0	31.9241
	165.000	67775.00	222358.9	6440987.10	21131847.0	31.9237
	165.286	68033.56	223207.2	6441245.10	21132694.0	31.9237
S-II IGNITION	165.286	68033.56	223207.2	6441245.10	21132694.0	31.9237
	167.786	70274.50	230559.4	6443481.40	21140031.0	31.9303
MR SHIFT	167.786	70274.50	230559.4	6443481.40	21140031.0	31.9303
	170.000	72229.88	236974.7	6445432.80	21146433.0	31.9369
	180.000	80747.44	264919.4	6453931.90	21174317.0	31.9661
	190.000	88758.91	291203.5	6461924.40	21200539.0	31.9941
	193.986	91816.13	301234.0	6464974.00	21210545.0	32.0050
WEIGHT DROP	193.986	91816.13	301234.0	6464974.00	21210545.0	32.0050
	198.986	95542.38	313459.2	6468690.60	21222738.0	32.0186
WEIGHT DROP	198.986	95542.38	313459.2	6468690.60	21222738.0	32.0186
INITIATE GUIDANCE	198.986	95542.38	313459.2	6468690.60	21222738.0	32.0186
	200.000	96283.94	315892.2	6469430.10	21225164.0	32.0212
	210.000	103383.50	339184.7	6476509.90	21248392.0	32.0468
	220.000	110096.94	361210.4	6483203.20	21270352.0	32.0717
	230.000	116433.31	381999.1	6489519.00	21291073.0	32.0959
	240.000	122400.56	401576.6	6495465.40	21310582.0	32.1194
	250.000	128010.19	419980.9	6501053.70	21328916.0	32.1422
	260.000	133271.44	437242.3	6506293.30	21346106.0	32.1643
	270.000	138193.44	453391.5	6511193.10	21362182.0	32.1857
	280.000	142787.06	468461.5	6515764.10	21377179.0	32.2064
	290.000	147063.31	482491.2	6520017.40	21391133.0	32.2265
	300.000	151032.[illegible]	495513.2	6523963.10	2140[illegible]	[illegible]

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BASELINE SA 511 TO LOR CASE 1

TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd) TABLE NO. 3

	TIME SEC	ALTITUDE M	ALTITUDE FT	RADIUS M	RADIUS FT	INCL DEG
	310.000	154707.13	507569.3	6527613.90	21416056.0	32.2647
	320.000	158097.75	518693.4	6530980.30	21427101.0	32.2829
	330.000	161217.06	528927.4	6534075.00	21437254.0	32.3004
	340.000	164078.82	538316.3	6536911.70	21446560.0	32.3174
	350.000	166695.13	546900.0	6539502.50	21455061.0	32.3337
	360.000	169081.87	554730.6	6541863.40	21462806.0	32.3495
	370.000	171252.44	561851.8	6544007.60	21469841.0	32.3646
	380.000	173222.62	568315.7	6545951.10	21476217.0	32.3792
	390.000	175009.56	574178.4	6547711.00	21481992.0	32.3933
	400.000	176631.13	579498.4	6549305.10	21487222.0	32.4068
	410.000	178105.00	584334.0	6550751.20	21491966.0	32.4197
	420.000	179460.69	588749.0	6552068.70	21496288.0	32.4321
	426.312	180244.31	591352.7	6552844.30	21498833.0	32.4397
MR SHIFT	426.312	180244.31	591352.7	6552844.30	21498833.0	32.4397
	430.000	180685.25	592799.4	6553274.80	21500245.0	32.4432
	440.000	181771.00	596361.5	6554331.90	21503714.0	32.4526
	450.000	182710.13	599442.7	6555242.20	21506700.0	32.4616
	460.000	183521.19	602103.6	6556024.40	21509267.0	32.4701
	470.000	184223.94	604409.2	6556698.20	21511476.0	32.4783
	480.000	184836.81	606420.0	6557282.10	21513393.0	32.4861
	490.000	185382.57	608210.5	6557798.80	21515088.0	32.4936
	500.000	185884.44	609857.1	6558271.70	21516640.0	32.5006
	510.000	186365.94	611436.8	6558724.40	21518125.0	32.5073
	520.000	186853.69	613037.0	6559183.40	21519630.0	32.5136
	530.000	187375.50	614749.0	6559676.80	21521249.0	32.5196
	540.000	187963.75	616679.0	6560236.70	21523087.0	32.5252
S-II CUTOFF	543.832	188213.01	617496.7	6560475.20	21523869.0	32.5273
S-IVB IGNITION	543.832	188213.01	617496.7	6560475.20	21523869.0	32.5273
	550.000	188607.07	618789.6	6560852.20	21525106.0	32.5282
	560.000	189164.25	620617.6	6561382.60	21526846.0	32.5296
	570.000	189627.38	622137.1	6561819.80	21528280.0	32.5309
	580.000	190005.69	623378.2	6562173.40	21529440.0	32.5321
	590.000	190306.25	624364.3	6562450.30	21530349.0	32.5332
	600.000	190537.69	625123.6	6562659.10	21531034.0	32.5342

BASELINE SA 511 TO LOR CASE 1

TABLE NO. 3

TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

	TIME	ALTITUDE		RADIUS		INCL
	SEC	M	FT	M	FT	DEG
	610.000	190709.06	625685.9	6562809.30	21531527.0	32.5352
	620.000	190828.50	626077.8	6562908.70	21531853.0	32.5360
	630.000	190905.94	626331.8	6562967.40	21532045.0	32.5368
	640.000	190949.13	626473.5	6562993.20	21532130.0	32.5375
	650.000	190969.01	626538.7	6562997.10	21532143.0	32.5381
	660.000	190974.12	626555.5	6562987.70	21532112.0	32.5386
	670.000	190974.63	626557.2	6562975.20	21532071.0	32.5391
	676.027	190974.69	626557.4	6562968.20	21532048.0	32.5393
INJECTION	676.027	190974.69	626557.4	6562968.20	21532048.0	32.5393
THRUST EVENT	676.027	190974.69	626557.4	6562968.20	21532048.0	32.5393
	680.000	190974.38	626556.3	6562963.50	21532033.0	32.5394
	690.000	190950.57	626478.2	6562929.80	21531922.0	32.5397
	700.000	190907.81	626338.0	6562879.00	21531755.0	32.5399
	710.000	190867.31	626205.1	6562832.10	21531601.0	32.5400
	720.000	190847.69	626140.7	6562807.80	21531522.0	32.5401
	730.000	190870.00	626213.9	6562827.40	21531586.0	32.5402
	740.000	190956.13	626496.5	6562912.60	21531866.0	32.5402
	750.000	191127.19	627057.7	6563084.70	21532430.0	32.5402
	760.000	191406.56	627974.3	6563367.20	21533357.0	32.5401
	770.000	191818.38	629325.4	6563784.30	21534725.0	32.5400
	780.000	192385.50	631186.0	6564358.60	21536610.0	32.5399
	790.000	193134.37	633643.0	6565116.90	21539097.0	32.5397
	800.000	194090.07	636778.4	6566084.30	21542271.0	32.5395
	810.000	195280.81	640685.1	6567289.00	21546224.0	32.5392
	820.000	196733.38	645450.7	6568757.90	21551043.0	32.5389
	830.000	198476.75	651170.4	6570520.00	21556824.0	32.5386
	840.000	200541.69	657945.2	6572606.20	21563669.0	32.5382
	850.000	202958.26	665873.5	6575046.50	21571675.0	32.5378
	860.000	205759.75	675064.8	6577874.30	21580952.0	32.5374
	870.000	208979.63	685628.7	6581123.00	21591611.0	32.5369
	880.000	212650.44	697672.0	6584825.50	21603758.0	32.5364
	890.000	216809.32	711316.6	6589018.60	21617515.0	32.5359
	900.000	221494.56	726683.2	6593741.00	216[illegible]9.0	32.5354
	910.000	226745.26	743914.3	6599031.50	2165[illegible]365.0	[illegible].5348

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BASELINE SA 51[illegible] TO LOR CASE 1

TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 3

	TIME	ALTITUDE		RADIUS		INCL
	SEC	M	FT	M	FT	DEG
	920.000	232598.88	763119.7	6604927.90	21669711.0	32.5342
	930.000	239100.19	784449.4	6611474.90	21691190.0	32.5336
	940.000	246291.56	808043.2	6618714.90	21714944.0	32.5329
	950.000	254221.81	834061.1	6626696.60	21741131.0	32.5323
	960.000	262933.69	862643.3	6635463.10	21769892.0	32.5316
	970.000	272479.56	893961.8	6645066.50	21801399.0	32.5309
	980.000	282910.63	928184.5	6655558.20	21835821.0	32.5302
	990.000	294273.82	965465.3	6666985.10	21873311.0	32.5295
INJECTION	993.160	298073.00	977929.8	6670805.10	21885844.0	32.5292
	993.160	298073.00	977929.8	6670805.10	21885844.0	32.5292

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BASELINE SA 511 TO LOR CASE 1

TABLE NO. 4

TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

	TIME SEC	MACH	DYN. PRESSURE N/(M)SQ	DYN. PRESSURE LB/(FT)SQ	ALFP DEG
LIFT-OFF	.000	.0000	.000	.000	91.731
	4.000	.0226	36.153	.755	-.014
	5.000	.0286	58.117	1.214	-.017
	10.000	.0614	265.604	5.547	-.033
BEGIN TILT	12.000	.0758	402.299	8.402	-.039
	15.000	.0988	676.428	14.128	-.243
	20.000	.1413	1349.782	28.191	-.123
	25.000	.1893	2346.606	49.010	.283
	30.000	.2437	3724.376	77.785	.752
	35.000	.3052	5527.217	115.438	1.195
END TILT	35.000	.3052	5527.217	115.438	1.195
	40.000	.3747	7783.320	162.558	.000
	45.000	.4536	10495.442	219.202	.000
	50.000	.5435	13633.424	284.740	.000
	55.000	.6466	17130.138	357.770	.000
	60.000	.7656	20868.181	435.841	.000
	65.000	.9039	24670.186	515.248	.000
	70.000	1.0641	28195.830	588.882	.000
	75.000	1.2500	31087.586	649.278	.000
10 KMS.	75.107	1.2543	31140.097	650.374	.000
	80.000	1.4682	33045.722	690.174	-.000
Q MAXIMUM	83.358	1.6318	33488.028	699.412	-.000
	85.000	1.7157	33371.489	696.978	.000
14 KMS.	85.973	1.7664	33189.071	693.168	.000
	90.000	1.9790	31529.053	658.498	.000
	95.000	2.2304	27490.262	574.146	.000
	100.000	2.4785	22881.754	477.895	.000
	105.000	2.7438	18616.442	388.812	.000
	110.000	3.0335	14865.712	310.477	.000
	115.000	3.3466	11640.641	243.120	.000
	120.000	3.6642	8860.342	185.053	.000
	125.000	3.9933	6614.342	138.143	.000
	130.000	4.3340	4855.648	101.412	.000
	135.000	4.6921	3524.347	73.608	.000

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BASELINE SA 511 TO LOR

CASE 1

TABLE NO. 4

TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

EVENT	TIME SEC	MACH	DYN. PRESSURE N/(M)SQ	DYN. PRESSURE LB/(FT)SQ	ALFP DEG
	140.000	5.0844	2544.849	53.150	.000
	145.000	5.5613	1847.425	38.584	.000
	149.486	6.1210	1391.970	29.072	.000
IECO	149.486	6.1210	1391.970	29.072	.000
	150.000	6.1779	1339.946	27.985	.000
BEGIN CHIFRZ	153.000	6.5303	1066.525	22.275	.000
	155.000	6.7847	910.392	19.014	.457
	160.000	7.4909	595.423	12.436	1.486
OECO	161.486	7.7205	519.792	10.856	1.762
SEPARATION	161.486	7.7205	519.792	10.856	1.762
	165.000	7.8445	334.393	6.984	2.503
	165.286	7.8547	322.532	6.736	2.564
S-II IGNITION	165.286	7.8547	322.532	6.736	2.564
	167.786	8.0018	238.087	4.973	3.074
MR SHIFT	167.786	8.0018	238.087	4.973	3.074
	170.000	8.1400	181.479	3.790	3.519
	180.000	8.7959	50.700	1.059	5.477
	190.000	9.1310	12.625	.264	7.350
	193.986	9.0708	7.144	.149	8.071
WEIGHT DROP	193.986	9.0708	7.144	.149	8.071
	198.986	8.9048	3.615	.076	8.956
WEIGHT DROP	198.986	8.9048	3.615	.076	8.956
INITIATE GUIDANCE	198.986	8.9048	3.615	.076	16.727
	200.000	.0000	.000	.000	16.774
	210.000	.0000	.000	.000	17.214
	220.000	.0000	.000	.000	17.603
	230.000	.0000	.000	.000	17.941
	240.000	.0000	.000	.000	18.228
	250.000	.0000	.000	.000	18.461
	260.000	.0000	.000	.000	18.645
	270.000	.0000	.000	.000	18.782
	280.000	.0000	.000	.000	18.867
	290.000	.0000	.000	.000	18.904
	300.000	.0000	.000	.000	18.897

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BASELINE SA 511 TO LOR CASE 1

TABLE NO. 4

TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

	TIME SEC	MACH	DYN. PRESSURE N/(M)SQ	DYN. PRESSURE LB/(FT)SQ	ALFP DEG
	310.000	.0000	.000	.000	18.844
	320.000	.0000	.000	.000	18.744
	330.000	.0000	.000	.000	18.603
	340.000	.0000	.000	.000	18.422
	350.000	.0000	.000	.000	18.196
	360.000	.0000	.000	.000	17.932
	370.000	.0000	.000	.000	17.632
	380.000	.0000	.000	.000	17.294
	390.000	.0000	.000	.000	16.918
	400.000	.0000	.000	.000	16.510
	410.000	.0000	.000	.000	16.070
	420.000	.0000	.000	.000	15.595
	426.312	.0000	.000	.000	15.280
MR SHIFT	426.312	.0000	.000	.000	15.280
	430.000	.0000	.000	.000	15.119
	440.000	.0000	.000	.000	14.667
	450.000	.0000	.000	.000	14.188
	460.000	.0000	.000	.000	13.681
	470.000	.0000	.000	.000	13.152
	480.000	.0000	.000	.000	12.600
	490.000	.0000	.000	.000	12.023
	500.000	.0000	.000	.000	11.425
	510.000	.0000	.000	.000	10.808
	520.000	.0000	.000	.000	10.171
	530.000	.0000	.000	.000	9.514
	540.000	.0000	.000	.000	8.841
S-II CUTOFF	543.332	.0000	.000	.000	8.579
S-IVB IGNITION	543.332	.0000	.000	.000	8.579
	550.000	.0000	.000	.000	8.263
	560.000	.0000	.000	.000	7.730
	570.000	.0000	.000	.000	7.190
	580.000	.0000	.000	.000	6.642
	590.000	.0000	.000	.000	6.085
	600.000	.0000	.000	.000	5.519

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BASELINE SA 511 TO LOR CASE 1

TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 4

	TIME SEC	MACH	DYN. PRESSURE N/(M)SQ	DYN. PRESSURE LB/(FT)SQ	ALFP DEG
	610.000	.0000	.000	.000	4.946
	620.000	.0000	.000	.000	4.367
	630.000	.0000	.000	.000	3.779
	640.000	.0000	.000	.000	3.187
	650.000	.0000	.000	.000	2.591
	660.000	.0000	.000	.000	1.991
	670.000	.0000	.000	.000	1.387
	676.027	.0000	.000	.000	1.023
INJECTION	676.027	.0000	.000	.000	1.023
THRUST EVENT	676.027	.0000	.000	.000	-3.464
	680.000	.0000	.000	.000	-3.374
	690.000	.0000	.000	.000	-3.153
	700.000	.0000	.000	.000	-2.940
	710.000	.0000	.000	.000	-2.735
	720.000	.0000	.000	.000	-2.538
	730.000	.0000	.000	.000	-2.349
	740.000	.0000	.000	.000	-2.167
	750.000	.0000	.000	.000	-1.994
	760.000	.0000	.000	.000	-1.828
	770.000	.0000	.000	.000	-1.670
	780.000	.0000	.000	.000	-1.519
	790.000	.0000	.000	.000	-1.376
	800.000	.0000	.000	.000	-1.240
	810.000	.0000	.000	.000	-1.111
	820.000	.0000	.000	.000	-.990
	830.000	.0000	.000	.000	-.876
	840.000	.0000	.000	.000	-.769
	850.000	.0000	.000	.000	-.669
	860.000	.0000	.000	.000	-.576
	870.000	.0000	.000	.000	-.490
	880.000	.0000	.000	.000	-.411
	890.000	.0000	.000	.000	-.339
	900.000	.0000	.000	.000	-.274
	910.000	.0000	.000	.000	-.216

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BASELINE SA 511 TO LOR CASE 1

TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 4

	TIME SEC	MACH	DYN. PRESSURE N/(M)SQ	DYN. PRESSURE LB/(FT)SQ	ALFP DEG
	920.000	.0000	.000	.000	-.164
	930.000	.0000	.000	.000	-.119
	940.000	.0000	.000	.000	-.081
	950.000	.0000	.000	.000	-.049
	960.000	.0000	.000	.000	-.024
	970.000	.0000	.000	.000	-.006
	980.000	.0000	.000	.000	.006
	990.000	.0000	.000	.000	.011
INJECTION	993.160	.0000	.000	.000	.011
	993.160	.0000	.000	.000	.011

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TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 5

	TIME SEC	AZ.S DEG	LATITUDE DEG	LONGITUDE DEG	CHIP DEG	CHIY DEG	CHIR DEG
LIFT-OFF	.000	90.000	28.447	-80.604	-.059	.019	88.743
	4.000	89.997	28.447	-80.604	-.073	.024	88.743
	5.000	89.996	28.447	-80.604	-.077	.025	88.743
	10.000	89.992	28.447	-80.604	-.094	.031	88.743
BEGIN TILT	12.000	89.990	28.447	-80.604	-.101	.000	86.743
	15.000	89.981	28.447	-80.604	-.420	.000	83.743
	20.000	89.938	28.447	-80.604	-1.121	.000	78.743
	25.000	89.860	28.447	-80.604	-1.821	.000	73.743
	30.000	89.746	28.447	-80.604	-2.521	.000	68.743
	35.000	89.596	28.447	-80.604	-3.222	.000	63.743
END TILT	35.000	89.596	28.447	-80.604	-3.222	.000	63.743
	40.000	89.309	28.447	-80.603	-6.636	.000	58.743
	45.000	88.900	28.448	-80.602	-9.312	.000	53.743
	50.000	88.358	28.448	-80.601	-12.377	.000	48.743
	55.000	87.684	28.449	-80.598	-15.742	.000	43.743
	60.000	86.895	28.450	-80.595	-19.311	.000	38.743
	65.000	86.017	28.451	-80.590	-22.992	.000	33.743
	70.000	85.090	28.453	-80.584	-26.703	.000	28.743
	75.000	84.147	28.456	-80.576	-30.378	.000	23.743
10 KMS.	75.107	84.127	28.456	-80.576	-30.456	.000	23.636
	80.000	83.206	28.459	-80.565	-33.961	.000	18.743
Q MAXIMUM	83.358	82.588	28.461	-80.557	-36.289	.000	15.385
	85.000	82.292	28.462	-80.552	-37.401	.000	13.743
14 KMS.	85.973	82.120	28.463	-80.549	-38.050	.000	12.770
	90.000	81.426	28.467	-80.536	-40.664	.000	8.743
	95.000	80.619	28.473	-80.517	-43.733	.000	3.743
	100.000	79.879	28.479	-80.493	-46.599	.000	.000
	105.000	79.209	28.487	-80.466	-49.261	.000	.000
	110.000	78.605	28.496	-80.434	-51.725	.000	.000
	115.000	78.068	28.507	-80.397	-54.001	.000	.000
	120.000	77.589	28.519	-80.355	-56.099	.000	.000
	125.000	77.164	28.532	-80.307	-58.032	.000	.000
	130.000	76.787	28.547	-80.253	-59.812	.000	.000
	135.000	76.455	28.565	-80.193	-61.449	.000	.000

BASELINE SA 511 TO LOR

CASE 1

TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 5

	TIME SEC	AZ.S DEG	LATITUDE DEG	LONGITUDE DEG	CHIP DEG	CHIY DEG	CHIR DEG
	140.000	76.161	28.583	-80.125	-62.955	.000	.000
	145.000	75.901	28.605	-80.050	-64.340	.000	.000
	149.486	75.695	28.625	-79.975	-65.487	.000	.000
IECO	149.486	75.695	28.625	-79.975	-65.487	.000	.000
	150.000	75.679	28.628	-79.966	-65.613	.000	.000
BEGIN CHIFRZ	153.000	75.586	28.643	-79.912	-66.332	.000	.000
	155.000	75.527	28.653	-79.875	-66.332	.000	.000
	160.000	75.394	28.680	-79.776	-66.332	.000	.000
OECO	161.486	75.357	28.689	-79.745	-66.332	.000	.000
SEPARATION	161.486	75.357	28.689	-79.745	-66.332	.000	.000
	165.000	75.401	28.709	-79.671	-66.332	.000	.000
	165.286	75.405	28.711	-79.665	-66.332	.000	.000
S-II IGNITION	165.286	75.405	28.711	-79.665	-66.332	.000	.000
	167.786	75.419	28.725	-79.613	-66.332	.000	.000
MR SHIFT	167.786	75.419	28.725	-79.613	-66.332	.000	.000
	170.000	75.430	28.738	-79.566	-66.332	.000	.000
	180.000	75.484	28.797	-79.350	-66.332	.000	.000
	190.000	75.544	28.856	-79.127	-66.332	.000	.000
	193.986	75.569	28.881	-79.036	-66.332	.000	.000
WEIGHT DROP	193.986	75.569	28.881	-79.036	-66.332	.000	.000
	198.986	75.602	28.911	-78.921	-66.332	.000	.000
WEIGHT DROP	198.986	75.602	28.911	-78.921	-66.332	.000	.000
INITIATE	198.986	75.602	28.911	-78.921	-60.866	.000	.000
GUIDANCE	200.000	75.609	28.918	-78.897	-60.956	.000	.000
	210.000	75.681	28.980	-78.661	-61.841	.000	.000
	220.000	75.758	29.043	-78.418	-62.733	.000	.000
	230.000	75.840	29.108	-78.168	-63.633	.000	.000
	240.000	75.927	29.174	-77.910	-64.541	.000	.000
	250.000	76.020	29.240	-77.645	-65.460	.000	.000
	260.000	76.118	29.309	-77.372	-66.386	.000	.000
	270.000	76.221	29.378	-77.091	-67.319	.000	.000
	280.000	76.331	29.448	-76.801	-68.264	.000	.000
	290.000	76.446	29.520	-76.503	-69.218	.000	.000
	300.000	76.567	29.593	-76.195	-70.180	.000	.000

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BASELINE SA 511 TO LOR CASE 1

TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd) TABLE NO. 5

	TIME SEC	AZ.S DEG	LATITUDE DEG	LONGITUDE DEG	CHIP DEG	CHIY DEG	CHIR DEG
	310.000	76.694	29.667	-75.878	-71.152	.000	.000
	320.000	76.828	29.742	-75.552	-72.135	.000	.000
	330.000	76.968	29.819	-75.215	-73.127	.000	.000
	340.000	77.115	29.896	-74.863	-74.127	.000	.000
	350.000	77.269	29.975	-74.510	-75.140	.000	.000
	360.000	77.431	30.055	-74.141	-76.162	.000	.000
	370.000	77.599	30.136	-73.760	-77.194	.000	.000
	380.000	77.775	30.218	-73.367	-78.236	.000	.000
	390.000	77.960	30.301	-72.961	-79.290	.000	.000
	400.000	78.152	30.385	-72.542	-80.354	.000	.000
	410.000	78.353	30.471	-72.109	-81.426	.000	.000
	420.000	78.563	30.557	-71.662	-82.512	.000	.000
	426.312	78.700	30.611	-71.372	-83.202	.000	.000
MR SHIFT	426.312	78.700	30.611	-71.372	-83.202	.000	.000
	430.000	78.784	30.643	-71.200	-83.608	.000	.000
	440.000	79.018	30.731	-70.724	-84.712	.000	.000
	450.000	79.260	30.818	-70.235	-85.828	.000	.000
	460.000	79.510	30.906	-69.732	-86.955	.000	.000
	470.000	79.769	30.994	-69.214	-88.092	.000	.000
	480.000	80.038	31.081	-68.680	-89.237	.000	.000
	490.000	80.316	31.169	-68.131	-90.395	.000	.000
	500.000	80.604	31.256	-67.566	-91.562	.000	.000
	510.000	80.902	31.343	-66.983	-92.737	.000	.000
	520.000	81.211	31.429	-66.382	-93.922	.000	.000
	530.000	81.532	31.515	-65.763	-95.118	.000	.000
	540.000	81.864	31.600	-65.123	-96.321	.000	.000
S-II CUTOFF	543.332	81.994	31.632	-64.873	-96.784	.000	.000
S-IVB IGNITION	543.832	81.994	31.632	-64.873	-96.784	.000	.000
	550.000	82.217	31.683	-64.466	-97.531	.000	.000
	560.000	82.582	31.763	-63.800	-98.752	.000	.000
	570.000	82.953	31.841	-63.127	-99.979	.000	.000
	580.000	83.328	31.914	-62.446	-101.213	.000	.000
	590.000	83.708	31.985	-61.757	-102.454	.000	.000

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BASELINE SA 511 TO LOR CASE 1

TABLE XIII

TABLE NO. 5

BASELINE LOR TRAJECTORY DATA (Cont'd)

	TIME SEC	AZ.S DEG	LATITUDE DEG	LONGITUDE DEG	CHIP DEG	CHIY DEG	CHIR DEG
	610.000	84.484	32.115	-60.356	-104.954	.000	.000
	620.000	84.879	32.174	-59.643	-106.211	.000	.000
	630.000	85.280	32.230	-58.923	-107.475	.000	.000
	640.000	85.686	32.281	-58.194	-108.741	.000	.000
	650.000	86.097	32.329	-57.457	-110.011	.000	.000
	660.000	86.513	32.372	-56.711	-111.284	.000	.000
	670.000	86.934	32.410	-55.957	-112.558	.000	.000
	676.027	87.190	32.431	-55.499	-113.327	.000	.000
INJECTION	676.027	87.190	32.431	-55.499	-113.327	.000	.000
THRUST EVENT	676.027	87.190	32.431	-55.499	-117.814	.000	.000
	680.000	87.360	32.444	-55.195	-118.006	.000	.000
	690.000	87.793	32.473	-54.424	-118.489	.000	.000
	700.000	88.229	32.497	-53.646	-118.973	.000	.000
	710.000	88.671	32.516	-52.860	-119.457	.000	.000
	720.000	89.117	32.529	-52.066	-119.941	.000	.000
	730.000	89.567	32.538	-51.263	-120.425	.000	.000
	740.000	90.022	32.540	-50.453	-120.910	.000	.000
	750.000	90.481	32.537	-49.634	-121.394	.000	.000
	760.000	90.945	32.528	-48.808	-121.879	.000	.000
	770.000	91.413	32.513	-47.973	-122.364	.000	.000
	780.000	91.885	32.491	-47.130	-122.849	.000	.000
	790.000	92.361	32.463	-46.279	-123.334	.000	.000
	800.000	92.841	32.429	-45.420	-123.819	.000	.000
	810.000	93.325	32.387	-44.553	-124.303	.000	.000
	820.000	93.813	32.339	-43.678	-124.788	.000	.000
	830.000	94.304	32.284	-42.795	-125.272	.000	.000
	840.000	94.799	32.221	-41.904	-125.756	.000	.000
	850.000	95.298	32.151	-41.005	-126.240	.000	.000
	860.000	95.799	32.072	-40.098	-126.723	.000	.000
	870.000	96.304	31.986	-39.183	-127.206	.000	.000
	880.000	96.811	31.892	-38.261	-127.688	.000	.000
	890.000	97.321	31.790	-37.331	-128.170	.000	.000
	900.000	97.833	31.679	-36.393	-128.651	.000	.000
	910.000	98.348	31.560	-35.449	-129.131	.000	.000

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BASELINE SA 511 TO LOR CASE 1

TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd) TABLE NO. 5

	TIME SEC	AZ.S DEG	LATITUDE DEG	LONGITUDE DEG	CHIP DEG	CHIY DEG	CHIR DEG
	920.000	98.865	31.432	-34.496	-129.611	.000	.000
	930.000	99.383	31.294	-33.537	-130.089	.000	.000
	940.000	99.904	31.148	-32.571	-130.566	.000	.000
	950.000	100.425	30.992	-31.597	-131.043	.000	.000
	960.000	100.948	30.826	-30.617	-131.518	.000	.000
	970.000	101.471	30.651	-29.631	-131.991	.000	.000
	980.000	101.995	30.466	-28.637	-132.463	.000	.000
	990.000	102.520	30.271	-27.637	-132.934	.000	.000
INJECTION	993.160	102.686	30.207	-27.320	-133.082	.000	.000
	993.160	102.686	30.207	-27.320	-133.082	.000	.000

BASELINE SA 511 TO LOR CASE 1

TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 6

	TIME SEC	AERO HEATING INDIC. N-M/(M)SQ	AERO HEATING INDIC. LB-FT/(FT)SQ	ALPHA Q N-RAD	ALPHA Q LB-DEG
LIFT-OFF	.000	.0	.0	.000	.000
	4.000	173.7	11.9	.000	.000
	5.000	439.6	30.1	.022	.289
	10.000	8321.2	570.2	.181	2.325
BEGIN TILT	12.000	18406.9	1261.3	.303	3.899
	15.000	49453.4	3388.6	2.870	36.966
	20.000	182364.6	12495.9	3.178	40.934
	25.000	515826.8	35345.4	12.031	154.966
	30.000	1235071.9	84629.3	49.240	634.237
	35.000	2627008.8	180007.3	115.744	1490.849
END TILT	35.000	2627008.8	180007.3	115.744	1490.849
	40.000	5071674.0	347520.1	14.655	188.760
	45.000	9139253.3	626237.8	20.158	259.644
	50.000	15558449.0	1066092.4	26.090	336.059
	55.000	25240949.0	1729554.4	32.018	412.407
	60.000	39271502.0	2690952.6	37.410	481.865
	65.000	58865217.0	4033548.6	41.714	537.300
	70.000	85205314.0	5838418.5	44.517	573.412
	75.000	119125831.0	8162712.3	45.750	589.288
10 KMS.	75.107	119939625.0	8218474.8	45.759	589.405
	80.000	161129660.0	11040888.4	45.419	585.027
Q MAXIMUM	83.358	193747450.0	13275916.7	44.013	566.916
	85.000	210813320.0	14445301.1	42.926	552.918
14 KMS.	85.973	221227430.0	15158893.9	42.154	542.966
	90.000	266043310.0	18229757.0	38.023	489.756
	95.000	322982470.0	22131329.0	31.134	401.025
	100.000	377444970.0	25863196.0	24.359	313.759
	105.000	427883290.0	29319318.0	18.603	239.616
	110.000	473455200.0	32442671.0	13.894	178.961
	115.000	513767490.0	35204255.0	10.121	130.361
	120.000	548406260.0	37577765.0	7.115	91.651
	125.000	577381610.0	39563207.0	4.862	62.630
	130.000	601119980.0	41189186.0	3.231	41.615
	135.000	620220480.0	42498602.0	2.093	26.958

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BASELINE SA 511 TO LOR CASE 1

TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd) TABLE NO. 6

	TIME SEC	AERO HEATING INDIC. N-M/(M)SQ	AERO HEATING INDIC. LB-FT/(FT)SQ	ALPHA Q N•RAD	ALPHA Q LB•DEG
	140.000	635445760.0	43541865.0	1.325	17.065
	145.000	647540170.0	44373595.0	.824	10.610
	149.486	656317050.0	44972003.0	.526	6.774
IECO	149.486	656317050.0	44972003.0	.526	6.774
	150.000	657210520.0	45033225.0	.497	6.399
BEGIN CHIFRZ	153.000	661903460.0	45354793.0	.351	4.522
	155.000	664579170.0	45538138.0	7.261	93.527
	160.000	669917930.0	45903960.0	15.439	198.865
OECO	161.486	671173080.0	45989965.0	15.989	205.946
SEPARATION	161.486	671173080.0	45989965.0	15.989	205.946
	165.000	673443960.0	46145569.0	14.610	188.184
	165.286	673588710.0	46155487.0	14.434	185.913
S-II IGNITION	165.286	673588710.0	46155487.0	14.434	185.913
	167.786	674664790.0	46229222.0	12.773	164.521
MR SHIFT	167.786	674664790.0	46229222.0	12.773	164.521
	170.000	675395610.0	46278615.0	11.145	143.560
	180.000	677030600.0	46391332.0	4.847	62.430
	190.000	677489740.0	46422793.0	1.620	20.861
	193.986	677555520.0	46427300.0	1.006	12.962
WEIGHT DROP	193.986	677555520.0	46427300.0	1.006	12.962
	198.986	677600640.0	46430393.0	.565	7.278
WEIGHT DROP	198.986	677600640.0	46430393.0	.565	7.278
INITIATE	198.986	677600640.0	46430393.0	1.066	13.726
GUIDANCE	200.000	677600640.0	46430393.0	.000	.000
	210.000	677600640.0	46430393.0	.000	.000
	220.000	677600640.0	46430393.0	.000	.000
	230.000	677600640.0	46430393.0	.000	.000
	240.000	677600640.0	46430393.0	.000	.000
	250.000	677600640.0	46430393.0	.000	.000
	260.000	677600640.0	46430393.0	.000	.000
	270.000	677600640.0	46430393.0	.000	.000
	280.000	677600640.0	46430393.0	.000	.000
	290.000	677600640.0	46430393.0	.000	.000
	300.000	677600640.0	46430393.0	.000	.000

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CASE 1

TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 7

APOLLO 13 POSITION COORDINATES

	TIME SEC	X M	X FT	Y M	Y FT	Z M	Z FT
LIFT-OFF	.000	6373265.2	20909463.0	19225.5	63075.8	1057.5	3469.5
	4.000	6373278.6	20909707.0	19729.6	64729.6	2612.4	8570.9
	5.000	6373286.8	20909733.0	19855.6	65142.9	3001.1	9846.3
	10.000	6373361.0	20909977.0	20485.3	67209.0	4944.9	16223.6
BEGIN TILT	12.000	6373407.0	20910128.0	20737.1	68035.1	5722.5	18774.7
	15.000	6373495.1	20910417.0	21114.7	69274.0	6889.0	22601.7
	20.000	6373697.1	20911080.0	21743.6	71337.3	8835.1	28986.5
	25.000	6373975.6	20911993.0	22372.0	73399.0	10787.0	35390.5
	30.000	6374339.9	20913189.0	22999.9	75459.0	12749.0	41827.4
	35.000	6374800.1	20914696.0	23627.2	77517.2	14725.2	48311.0
END TILT	35.000	6374800.1	20914698.0	23627.2	77517.2	14725.2	48311.0
	40.000	6375366.6	20916557.0	24254.0	79573.5	16725.6	54874.0
	45.000	6376049.9	20918799.0	24880.2	81628.0	18767.7	61573.8
	50.000	6376860.2	20921458.0	25505.9	83680.7	20870.3	68472.2
	55.000	6377807.4	20924565.0	26131.0	85731.6	23056.2	75643.7
	60.000	6378900.0	20928150.0	26755.6	87780.7	25351.5	83174.1
	65.000	6380145.0	20932234.0	27379.6	89828.1	27785.2	91158.7
	70.000	6381547.0	20936834.0	28003.1	91873.8	30388.4	99699.4
	75.000	6383106.3	20941950.0	28626.1	93917.8	33192.7	108900.1
10 KMS.	75.107	6383141.4	20942065.0	28639.5	93961.6	33255.2	109105.1
	80.000	6384824.9	20947588.0	29248.6	95960.1	36232.8	118874.2
Q MAXIMUM	83.358	6386070.7	20951676.0	29666.5	97330.9	38425.9	126069.1
	85.000	6386706.7	20953762.0	29870.6	98000.5	39546.1	129744.4
14 KMS.	85.973	6387092.1	20955027.0	29991.5	98397.5	40226.0	131975.2
	90.000	6388753.6	20960478.0	30491.9	100039.0	43171.1	141637.4
	95.000	6390966.7	20967736.0	31112.5	102075.2	47147.0	154681.6
	100.000	6393345.9	20975544.0	31732.5	104109.1	51513.5	169007.6
	105.000	6395890.6	20983893.0	32351.6	106140.5	56310.9	184747.2
	110.000	6398599.4	20992780.0	32970.0	108169.1	61579.6	202032.8
	115.000	6401470.8	21002200.0	33587.4	110195.0	67360.4	220998.8
	120.000	6404503.2	21012150.0	34204.0	112217.9	73695.3	241782.5
	125.000	6407696.0	21022625.0	34819.6	114237.7	80627.4	264525.7
	130.000	6411048.9	21033625.0	35434.3	116254.2	88202.2	289377.3
	135.000	6414563.2	21045155.0	36047.9	118267.5	96467.9	316495.7

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BASELINE SA 511 TO LOR CASE 1

TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 7

APOLLO 13 POSITION COORDINATES

EVENT	TIME SEC	X M	X FT	Y M	Y FT	Z M	Z FT
	140.000	6418241.5	21057223.0	36660.6	120277.4	105476.3	346051.0
	145.000	6422087.8	21069842.0	37272.1	122283.9	115284.0	378228.2
	149.486	6425686.9	21081650.0	37819.9	124081.1	124815.0	409498.0
IECO	149.486	6425686.9	21081650.0	37819.9	124081.1	124815.0	409498.0
	150.000	6426107.8	21083031.0	37882.6	124286.7	125951.9	413228.0
BEGIN CHIFRZ	153.000	6428586.3	21091162.0	38248.3	125486.7	132748.9	435527.8
	155.000	6430258.7	21096649.0	38492.0	126285.9	137433.8	450898.2
	160.000	6434517.8	21110623.0	39100.2	128281.5	149705.4	491159.4
OECO	161.486	6435807.8	21114855.0	39280.8	128873.9	153512.5	503649.9
SEPARATION	161.486	6435807.8	21114855.0	39280.8	128873.9	153512.5	503649.9
	165.000	6438812.2	21124712.0	39707.3	130273.2	162600.2	533465.3
	165.286	6439051.7	21125498.0	39742.0	130387.1	163340.4	535893.8
S-II IGNITION	165.286	6439051.7	21125498.0	39742.0	130387.1	163340.4	535893.8
	167.786	6441118.8	21132280.0	40045.1	131381.4	169824.0	557165.2
MR SHIFT	167.786	6441118.8	21132280.0	40045.1	131381.4	169824.0	557165.2
	170.000	6442914.5	21138171.0	40313.2	132261.1	175598.8	576111.5
	180.000	6450632.8	21163493.0	41521.4	136225.1	202106.7	663079.6
	190.000	6457713.3	21186724.0	42724.7	140172.9	229312.5	752337.7
	193.986	6460360.1	21195407.0	43202.9	141741.9	240354.3	788564.1
WEIGHT DROP	193.986	6460360.1	21195407.0	43202.9	141741.9	240354.3	788564.1
	198.986	6463539.6	21205839.0	43801.7	143706.2	254365.0	834530.8
WEIGHT DROP	198.986	6463539.6	21205839.0	43801.7	143706.2	254365.0	834530.8
INITIATE GUIDANCE	198.986	6463539.6	21205839.0	43801.7	143706.2	254365.0	834530.8
	200.000	6464165.7	21207893.0	43922.9	144103.9	257227.5	843922.3
	210.000	6470041.9	21227171.0	45115.8	148017.8	285849.6	937826.9
	220.000	6475372.1	21244659.0	46303.4	151913.9	315183.8	1034067.4
	230.000	6480154.8	21260350.0	47485.3	155791.8	345247.9	1132703.0
	240.000	6484388.1	21274239.0	48661.6	159651.0	376060.8	1233795.2
	250.000	6488069.9	21286318.0	49832.0	163470.8	407642.1	1337408.4
	260.000	6491197.3	21296579.0	50996.4	167310.9	440012.1	1443609.3
	270.000	6493766.8	21305010.0	52154.6	171110.8	473192.2	1552467.8
	280.000	6495774.8	21311597.0	53306.4	174889.9	507204.5	1664056.6
	290.000	6497216.5	21316327.0	54451.8	176647.7	542072.2	1778451.9
	300.000	6498087.1	21319184.0	55590.6	182383.8	577819.5	1895733.1

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BASELINE SA 511 TO LOR CASE 1

TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd) TABLE NO. 7

APOLLO 13 POSITION COORDINATES

	TIME	X		Y		Z	
	SEC	M	FT	M	FT	M	FT
	310.000	6498380.5	21320146.0	56722.6	186097.7	614472.0	2015984.1
	320.000	6498090.5	21319195.0	57847.6	189788.8	652056.1	2139291.7
	330.000	6497209.7	21316305.0	58965.6	193456.7	690599.9	2265747.9
	340.000	6495730.1	21311451.0	60076.4	197101.0	730132.8	2395448.9
	350.000	6493643.0	21304603.0	61179.8	200721.0	770685.7	2528496.5
	360.000	6490938.3	21295729.0	62275.7	204316.5	812291.2	2664997.4
	370.000	6487605.3	21284795.0	63363.9	207886.7	854983.7	2805064.5
	380.000	6483632.3	21271759.0	64444.3	211431.4	898799.6	2948817.5
	390.000	6479006.0	21256582.0	65516.8	214950.0	943777.4	3096382.5
	400.000	6473712.2	21239213.0	66581.1	218442.1	989958.1	3247894.1
	410.000	6467735.2	21219604.0	67637.3	221907.1	1037385.3	3403495.0
	420.000	6461057.7	21197696.0	68685.1	225344.7	1086105.3	3563337.5
	426.312	6456473.7	21182656.0	69342.1	227500.2	1117546.5	3666491.0
MR SHIFT	426.312	6456473.7	21182656.0	69342.1	227500.2	1117546.5	3666491.0
	430.000	6453658.7	21173421.0	69724.3	228754.3	1136149.6	3727525.0
	440.000	6445495.6	21146639.0	70754.9	232135.5	1187368.8	3895566.8
	450.000	6436542.5	21117266.0	71776.7	235487.9	1239747.1	4067411.6
	460.000	6426778.3	21085230.0	72789.6	238811.0	1293323.0	4243185.8
	470.000	6416180.2	21050460.0	73793.4	242104.3	1348137.6	4423023.7
	480.000	6404723.5	21012872.0	74788.0	245367.5	1404234.5	4607068.5
	490.000	6392381.3	20972379.0	75773.3	248600.0	1461660.2	4795473.0
	500.000	6379124.1	20928884.0	76749.1	251801.4	1520464.6	4988401.0
	510.000	6364920.1	20882284.0	77715.2	254971.3	1580701.4	5186028.3
	520.000	6349734.4	20832462.0	78671.7	258109.2	1642428.3	5388544.2
	530.000	6333529.1	20779295.0	79618.3	261214.8	1705707.6	5596153.5
	540.000	6316262.8	20722647.0	80554.9	264287.6	1770606.3	5809075.5
S-II CUTOFF	543.832	6309354.5	20699982.0	80911.1	265456.5	1795923.1	5892136.1
S-IVB IGNITION	543.832	6309354.5	20699982.0	80911.1	265456.5	1795923.1	5892136.1
	550.000	6297920.3	20662469.0	81481.3	267327.2	1836938.0	6026699.6
	560.000	6278595.1	20599066.0	82397.6	270333.2	1903731.5	6245838.3
	570.000	6258286.8	20532437.0	83303.4	273305.1	1970885.9	6466160.9
	580.000	6236983.1	20462543.0	84198.8	276242.6	2038399.2	6687661.3
	590.000	6214671.7	20389343.0	85083.5	279145.3	2106269.5	6910332.9
	600.000	6191339.5	20312794.0	85957.5	282012.8	2174494.6	7134168.5

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TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 7

APOLLO 13 POSITION COORDINATES

	TIME SEC	X M	X FT	Y M	Y FT	Z M	Z FT
	610.000	6166973.2	20232852.0	86820.6	284844.6	2243072.2	7359160.7
	620.000	6141559.0	20149472.0	87672.8	287640.5	2311999.9	7585301.3
	630.000	6115082.7	20062608.0	88513.9	290399.9	2381274.9	7812581.7
	640.000	6087529.7	19972210.0	89343.8	293122.6	2450894.6	8040992.8
	650.000	6058884.9	19878231.0	90162.3	295808.1	2520856.0	8270525.1
	660.000	6029132.7	19780619.0	90969.4	298456.1	2591156.1	8501168.3
	670.000	5998257.1	19679321.0	91765.0	301066.2	2661791.5	8732911.7
	676.027	5979097.7	19616462.0	92238.8	302620.9	2704522.2	8873104.4
INJECTION THRUST EVENT	676.027	5979097.7	19616462.0	92238.8	302620.9	2704522.2	8873104.4
	676.027	5979097.7	19616462.0	92238.8	302620.9	2704522.2	8873104.4
	680.000	5966238.5	19574273.0	92548.9	303638.1	2732751.0	8965718.4
	690.000	5933052.8	19465397.0	93321.0	306171.3	2803965.7	9199362.4
	700.000	5896687.8	19352651.0	94081.3	308665.6	2875416.1	9433779.5
	710.000	5863136.9	19236013.0	94829.5	311120.6	2947100.0	9668963.3
	720.000	5826393.4	19115464.0	95565.7	313535.9	3019015.8	9904907.3
	730.000	5788450.2	18990979.0	96289.7	315911.2	3091161.7	10141606.3
	740.000	5749300.1	18862533.0	97001.4	318246.1	3163536.3	10379056.0
	750.000	5708935.8	18730105.0	97700.7	320540.4	3236138.6	10617252.3
	760.000	5667349.4	18593666.0	98387.5	322793.8	3308967.6	10856192.7
	770.000	5624533.0	18453192.0	99061.8	325005.8	3382022.8	11095875.2
	780.000	5580478.1	18308655.0	99723.3	327176.2	3455304.1	11336299.5
	790.000	5535176.2	18160027.0	100372.1	329304.7	3528811.5	11577465.4
	800.000	5488618.3	18007278.0	101008.0	331391.1	3602545.6	11819375.3
	810.000	5440794.9	17850377.0	101631.0	333434.9	3676507.5	12062032.2
	820.000	5391696.3	17689293.0	102240.9	335436.1	3750698.3	12305440.6
	830.000	5341312.3	17523991.0	102837.8	337394.2	3825120.3	12549607.0
	840.000	5289632.1	17354436.0	103421.4	339309.1	3899775.7	12794539.3
	850.000	5236644.5	17180593.0	103991.8	341180.5	3974667.5	13040247.6
	860.000	5182337.7	17002420.0	104548.9	343008.3	4049799.6	13286743.9
	870.000	5126702.0	16819888.0	105092.6	344792.0	4125174.0	13534035.1
	880.000	5069718.5	16632934.7	105622.9	346531.7	4200800.5	13782153.7
	890.000	5011376.0	16441522.2	106139.6	348227.1	4276683.0	14031112.0
	900.000	4951660.1	16245604.0	106642.8	349878.1	4352828.0	14280931.5
	910.000	4890560.2	16045144.8	107132.4	351484.4	[illegible]	[illegible]

BASELINE SA 511 TO LOR

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 7

APOLLO 13 POSITION COORDINATES

	TIME SEC	X M	X FT	Y M	Y FT	Z M	Z FT
	920.000	4828049.8	15840058.3	107608.4	353046.0	4505933.3	14783245.8
	930.000	4764115.6	15630300.3	108070.7	354562.8	4582916.8	15035816.3
	940.000	4698739.1	15415810.5	108519.3	356034.5	4660200.6	15289371.9
	950.000	4631908.2	15196549.1	108954.2	357461.3	4737790.9	15543933.2
	960.000	4563584.8	14972391.0	109375.3	358843.0	4815714.1	15799586.8
	970.000	4493752.6	14743282.6	109782.7	360179.5	4893980.3	16056365.7
	980.000	4422386.9	14509143.2	110176.3	361471.0	4972606.7	16314326.1
	990.000	4349455.2	14269866.2	110556.2	362717.3	5051604.9	16573506.9
INJECTION	993.160	4326080.4	14193176.9	110673.4	363101.7	5076654.6	16655691.0
	993.160	4326080.4	14193176.9	110673.4	363101.7	5076654.6	16655691.0

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BASELINE SA 511 TO LOR CASE 1

TABLE XIII

BASELINE LOR TRAJECTORY DATA (CONT'D) TABLE NO. 8

APOLLO 13 VELOCITY COMPONENTS

	TIME SEC	XDOT M/SEC	XDOT FT/SEC	YDOT M/SEC	YDOT FT/SEC	ZDOT M/SEC	ZDOT FT/SEC
LIFT-OFF	.000	-.445	-1.459	126.044	413.530	388.708	1275.289
	4.000	7.272	23.859	125.992	413.361	388.735	1275.376
	5.000	9.345	30.659	125.980	413.319	388.742	1275.399
	10.000	20.575	67.502	125.917	413.114	388.781	1275.527
BEGIN TILT	12.000	25.488	83.623	125.893	413.034	388.798	1275.583
	15.000	33.336	109.369	125.834	412.842	388.894	1275.899
	20.000	47.753	156.669	125.733	412.511	389.671	1278.448
	25.000	63.960	209.842	125.629	412.168	391.251	1283.632
	30.000	82.102	269.363	125.521	411.814	393.671	1291.571
	35.000	102.336	335.750	125.411	411.453	396.966	1302.381
END TILT	35.000	102.336	335.750	125.411	411.453	396.966	1302.381
	40.000	124.629	408.889	125.299	411.087	403.685	1324.426
	45.000	149.029	488.939	125.187	410.720	413.781	1357.548
	50.000	175.446	575.610	125.076	410.354	428.037	1404.319
	55.000	203.713	668.348	124.966	409.994	447.189	1467.155
	60.000	233.558	766.265	124.859	409.644	471.884	1548.174
	65.000	264.621	868.179	124.757	409.306	502.643	1649.091
	70.000	296.120	971.522	124.656	408.976	539.678	1770.598
	75.000	327.677	1075.054	124.553	408.639	583.233	1913.494
10 KMS.	75.107	328.360	1077.296	124.551	408.631	584.243	1916.809
	80.000	359.940	1180.904	124.443	408.279	634.061	2080.254
Q MAXIMUM	83.356	381.978	1253.208	124.365	408.021	672.483	2206.308
	85.000	392.817	1288.771	124.325	407.890	692.529	2272.076
14 KMS.	85.973	399.259	1309.905	124.301	407.811	704.809	2312.365
	90.000	425.986	1397.592	124.197	407.472	758.767	2489.393
	95.000	459.243	1506.702	124.060	407.019	832.909	2732.641
	100.000	492.434	1615.598	123.911	406.531	915.058	3002.160
	105.000	525.401	1723.756	123.752	406.009	1005.255	3298.079
	110.000	558.062	1830.913	123.583	405.457	1103.571	3620.640
	115.000	590.420	1937.073	123.406	404.877	1210.153	3970.319
	120.000	622.541	2042.457	123.222	404.271	1325.230	4347.867
	125.000	654.551	2147.477	123.030	403.642	1449.133	4754.375
	130.000	686.670	2252.853	122.832	402.992	1582.383	5191.546
	135.000	719.147	2359.405	122.628	402.322	1725.616	5661.469

BASELINE SA 511 TO LOR CASE 1

TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 8

APOLLO 13 VELOCITY COMPONENTS

	TIME SEC	XDOT M/SEC	XDOT FT/SEC	YDOT M/SEC	YDOT FT/SEC	ZDOT M/SEC	ZDOT FT/SEC
	140.000	752.284	2468.124	122.418	401.635	1879.641	6166.800
	145.000	786.440	2580.195	122.204	400.930	2045.474	6710.874
	149.486	818.307	2684.735	122.007	400.286	2205.369	7235.461
IECO	149.486	818.307	2684.735	122.007	400.286	2205.369	7235.461
	150.000	820.302	2691.279	121.984	400.211	2220.563	7285.312
BEGIN CHIFRZ	153.000	832.091	2729.957	121.951	399.772	2311.352	7583.177
	155.000	840.361	2757.091	121.760	399.477	2373.791	7788.028
	160.000	864.021	2834.714	121.532	398.728	2536.478	8321.780
OECO	161.486	871.929	2860.660	121.464	398.503	2586.780	8486.812
SEPARATION	161.486	871.929	2860.660	121.464	398.503	2586.780	8486.812
	165.000	838.155	2749.853	121.300	397.965	2585.885	8483.875
	165.286	835.406	2740.832	121.286	397.921	2585.811	8483.631
S-II IGNITION	165.286	835.406	2740.832	121.286	397.921	2585.811	8483.631
	167.786	818.333	2684.818	121.169	397.534	2601.014	8533.509
MR SHIFT	167.786	818.333	2684.818	121.169	397.534	2601.014	8533.509
	170.000	804.003	2637.804	121.063	397.189	2616.222	8583.406
	180.000	739.799	2427.163	120.578	395.597	2685.520	8810.760
	190.000	676.466	2219.376	120.077	393.953	2755.827	9041.428
	193.986	651.464	2137.350	119.873	393.283	2784.146	9134.336
WEIGHT DROP	193.986	651.464	2137.350	119.873	393.283	2784.146	9134.336
	198.986	620.417	2035.490	119.613	392.431	2820.168	9252.520
WEIGHT DROP	198.986	620.417	2035.490	119.613	392.431	2820.168	9252.520
INITIATE	198.986	620.417	2035.490	119.613	392.431	2820.168	9252.520
GUIDANCE	200.000	614.892	2017.361	119.560	392.256	2827.200	9275.591
	210.000	560.337	1838.375	119.027	390.509	2897.517	9506.290
	220.000	505.667	1659.013	118.479	388.709	2969.604	9742.795
	230.000	450.837	1479.125	117.915	386.859	3043.533	9985.343
	240.000	395.800	1298.557	117.335	384.959	3119.378	10234.180
	250.000	340.508	1117.152	116.741	383.008	3197.220	10489.566
	260.000	284.909	934.740	116.131	381.007	3277.143	10751.781
	270.000	228.946	751.134	115.506	378.956	3359.239	11021.125
	280.000	172.564	566.154	114.865	376.855	3443.603	11297.911
	290.000	115.705	379.609	114.210	374.705	3530.339	11582.476
	300.000	58.301	191.276	113.540	372.506	3619.557	11875.186

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 8

APOLLO 13 VELOCITY COMPONENTS

	TIME SEC	XDOT M/SEC	XDOT FT/SEC	YDOT M/SEC	YDOT FT/SEC	ZDOT M/SEC	ZDOT FT/SEC
	310.000	.285	.934	112.855	370.258	3711.376	12176.429
	320.000	-58.415	-191.651	112.155	367.961	3805.922	12486.622
	330.000	-117.876	-386.733	111.440	365.616	3903.334	12806.215
	340.000	-178.183	-534.589	110.710	363.223	4003.760	13135.696
	350.000	-239.423	-785.508	109.966	360.781	4107.360	13475.590
	360.000	-301.692	-989.802	109.207	358.292	4214.308	13826.469
	370.000	-365.093	-1197.812	108.434	355.756	4324.793	14188.955
	380.000	-429.740	-1409.908	107.647	353.171	4439.024	14563.726
	390.000	-495.755	-1626.494	106.845	350.540	4557.225	14951.526
	400.000	-563.266	-1847.987	106.028	347.862	4679.646	15353.168
	410.000	-632.422	-2074.876	105.198	345.138	4806.560	15769.552
	420.000	-703.384	-2307.691	104.354	342.367	4938.270	16201.674
	426.312	-749.184	-2457.952	103.813	340.595	5024.029	16483.035
MR SHIFT	426.312	-749.184	-2457.952	103.813	340.595	5024.029	16483.035
	430.000	-777.472	-2550.760	103.495	339.551	5065.190	16618.079
	440.000	-855.475	-2806.678	102.623	336.689	5179.251	16992.294
	450.000	-935.501	-3069.229	101.736	333.781	5297.052	17378.781
	460.000	-1017.717	-3338.966	100.836	330.828	5418.824	17778.295
	470.000	-1102.308	-3616.497	99.923	327.831	5544.821	18191.670
	480.000	-1189.482	-3902.500	98.996	324.789	5675.324	18619.830
	490.000	-1279.465	-4197.721	98.055	321.703	5810.646	19063.801
	500.000	-1372.503	-4502.964	97.101	318.573	5951.136	19524.723
	510.000	-1468.672	-4819.134	96.134	315.400	6097.183	20003.881
	520.000	-1568.887	-5147.266	95.154	312.184	6249.228	20502.716
	530.000	-1672.893	-5488.493	94.161	308.926	6407.768	21022.861
	540.000	-1781.278	-5844.089	93.155	305.626	6573.372	21566.180
S-II CUTOFF	543.832	-1824.070	-5984.483	92.766	304.350	6638.845	21780.986
S-IVB IGNITION	543.832	-1824.070	-5984.483	92.766	304.350	6638.845	21780.986
	550.000	-1883.762	-6180.323	92.136	302.285	6661.243	21854.473
	560.000	-1981.483	-6500.927	91.105	298.902	6697.421	21973.166
	570.000	-2080.396	-6825.444	90.062	295.480	6733.417	22091.261
	580.000	-2180.542	-7154.009	89.007	292.017	6769.216	22208.715
	590.000	-2281.963	-7486.754	87.939	288.514	6804.807	22325.483

BASELINE SA 511 TO LOR

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 8

APOLLO 13 VELOCITY COMPONENTS

	TIME SEC	XDOT M/SEC	XDOT FT/SEC	YDOT M/SEC	YDOT FT/SEC	ZDOT M/SEC	ZDOT FT/SEC
	610.000	-2488.790	-8165.321	85.768	281.391	6875.304	22556.772
	620.000	-2594.281	-8511.421	84.665	277.771	6910.180	22671.194
	630.000	-2701.215	-8862.254	83.550	274.113	6944.787	22784.735
	640.000	-2809.636	-9217.946	82.423	270.416	6979.109	22897.337
	650.000	-2919.588	-9578.702	81.285	266.683	7013.128	23008.952
	660.000	-3031.119	-9944.613	80.136	262.912	7046.830	23119.522
	670.000	-3144.272	-10315.853	78.975	259.104	7080.197	23228.994
	676.027	-3213.260	-10542.193	78.270	256.792	7100.144	23294.436
INJECTION	676.027	-3213.260	-10542.193	78.270	256.792	7100.144	23294.436
THRUST EVENT	676.027	-3213.260	-10542.193	78.270	256.792	7100.144	23294.436
	680.000	-3259.815	-10694.932	77.804	255.261	7109.614	23325.507
	690.000	-3377.428	-11080.801	76.621	251.382	7133.290	23403.183
	700.000	-3495.686	-11468.785	75.428	247.468	7156.751	23480.154
	710.000	-3614.608	-11858.951	74.224	243.519	7180.016	23556.484
	720.000	-3734.218	-12251.371	73.010	239.535	7203.107	23632.242
	730.000	-3854.537	-12646.120	71.786	235.519	7226.047	23707.504
	740.000	-3975.591	-13043.277	70.552	231.469	7248.861	23782.352
	750.000	-4097.405	-13442.928	69.307	227.386	7271.575	23856.874
	760.000	-4220.006	-13845.163	68.053	223.273	7294.220	23931.167
	770.000	-4343.424	-14250.077	66.790	219.128	7316.826	24005.335
	780.000	-4467.690	-14657.775	65.518	214.953	7339.429	24079.492
	790.000	-4592.839	-15068.368	64.236	210.748	7362.066	24153.759
	800.000	-4718.906	-15481.974	62.946	206.515	7384.777	24228.272
	810.000	-4845.931	-15898.723	61.647	202.255	7407.607	24303.173
	820.000	-4973.956	-16318.755	60.341	197.968	7430.603	24378.620
	830.000	-5103.029	-16742.220	59.026	193.656	7453.818	24454.782
	840.000	-5233.198	-17169.285	57.705	189.320	7477.306	24531.845
	850.000	-5364.519	-17600.127	56.376	184.960	7501.130	24610.006
	860.000	-5497.051	-18034.944	55.041	180.579	7525.355	24689.485
	870.000	-5630.860	-18473.952	53.699	176.178	7550.054	24770.518
	880.000	-5766.021	-18917.391	52.352	171.757	7575.305	24853.364
	890.000	-5902.612	-19365.524	50.999	167.320	7601.195	24938.304
	900.000	-6040.716	-17819.623	49.642	162.866	7627.813	25025.634
	910.000	-6180.426	-20276.990	48.280	158.398	7655.256	25115.670

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 8

APOLLO 13 VELOCITY COMPONENTS

	TIME SEC	XDOT M/SEC	XDOT FT/SEC	YDOT M/SEC	YDOT FT/SEC	ZDOT M/SEC	ZDOT FT/SEC
	920.000	-6321.854	-20740.991	46.914	153.918	7683.638	25208.787
	930.000	-6465.118	-21211.018	45.545	149.427	7713.081	25305.383
	940.000	-6610.344	-21687.481	44.174	144.928	7743.709	25405.871
	950.000	-6757.668	-22170.825	42.801	140.422	7775.661	25510.701
	960.000	-6907.261	-22661.617	41.426	135.912	7809.100	25620.408
	970.000	-7059.306	-23160.453	40.050	131.399	7844.199	25735.562
	980.000	-7213.995	-23667.961	38.675	126.886	7881.139	25856.757
	990.000	-7371.516	-24184.764	37.300	122.376	7920.134	25984.693
INJECTION	993.160	-7421.885	-24350.015	36.866	120.951	7932.961	26026.773
	993.160	-7421.885	-24350.015	36.866	120.951	7932.961	26026.773

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TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 9

APOLLO 13 ACCELERATION COMPONENTS

	TIME	XDDOT		YDDOT		ZDDOT	
	SEC	M/SEC**2	FT/SEC**2	M/SEC**2	FT/SEC**2	M/SEC**2	FT/SEC**2
LIFT-OFF	.000	1.810	5.937	-.013	-.043	.006	.021
	4.000	2.045	6.709	-.013	-.042	.007	.023
	5.000	2.101	6.892	-.013	-.042	.007	.024
	10.000	2.395	7.857	-.012	-.040	.008	.027
BEGIN TILT	12.000	2.519	8.266	-.012	-.040	.009	.029
	15.000	2.714	8.903	-.020	-.065	.078	.254
	20.000	3.058	10.032	-.021	-.068	.234	.769
	25.000	3.430	11.253	-.021	-.070	.399	1.308
	30.000	3.832	12.573	-.022	-.072	.570	1.871
	35.000	4.267	14.000	-.022	-.073	.749	2.457
END TILT	35.000	4.267	14.000	-.022	-.073	.749	2.457
	40.000	4.671	15.324	-.022	-.073	1.655	5.430
	45.000	5.086	16.686	-.022	-.073	2.410	7.905
	50.000	5.476	17.965	-.022	-.073	3.318	10.884
	55.000	5.820	19.095	-.022	-.071	4.365	14.319
	60.000	6.108	20.039	-.021	-.069	5.532	18.148
	65.000	6.298	20.663	-.020	-.066	6.783	22.252
	70.000	6.260	20.539	-.020	-.066	8.024	26.325
	75.000	6.392	20.938	-.021	-.070	9.423	30.916
10 KMS.	75.107	6.384	20.946	-.021	-.070	9.454	31.018
	80.000	6.526	21.411	-.023	-.075	10.924	35.841
Q MAXIMUM	83.358	6.592	21.628	-.024	-.079	11.958	39.231
	85.000	6.614	21.698	-.025	-.081	12.467	40.901
14 KMS.	85.973	6.624	21.733	-.025	-.082	12.770	41.897
	90.000	6.648	21.812	-.027	-.087	14.034	46.043
	95.000	6.651	21.821	-.029	-.094	15.628	51.272
	100.000	6.620	21.717	-.031	-.101	17.233	56.539
	105.000	6.564	21.536	-.033	-.108	18.848	61.836
	110.000	6.501	21.329	-.035	-.113	20.484	67.205
	115.000	6.444	21.143	-.036	-.117	22.156	72.691
	120.000	6.407	21.019	-.038	-.124	23.883	78.356
	125.000	6.405	21.013	-.039	-.128	25.696	84.304
	130.000	6.451	21.163	-.040	-.132	27.625	90.632
	135.000	6.550	21.491	-.041	-.136	29.695	97.425

BASELINE SA 511 TO LOR

CASE 1

TABLE NO. 9

TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

APOLLO 13 ACCELERATION COMPONENTS

	TIME SEC	XDDOT M/SEC**2	XDDOT FT/SEC**2	YDDOT M/SEC**2	YDDOT FT/SEC**2	ZDDOT M/SEC**2	ZDDOT FT/SEC**2
	140.000	6.716	22.039	-.042	-.139	31.947	104.814
	145.000	6.961	22.837	-.043	-.142	34.429	112.955
	149.486	7.259	23.815	-.044	-.145	36.895	121.045
IECO	149.486	3.874	12.718	-.044	-.145	29.477	96.711
	150.000	3.889	12.760	-.044	-.145	29.674	97.357
BEGIN CHIFRZ	153.000	3.973	13.035	-.045	-.147	30.863	101.256
	155.000	4.299	14.106	-.045	-.148	31.581	103.613
	160.000	5.181	16.998	-.046	-.151	33.533	110.016
OECO	161.486	5.463	17.923	-.046	-.152	34.160	112.075
SEPARATION	161.486	-9.618	-31.555	-.046	-.152	-.251	-.824
	165.000	-9.606	-31.517	-.047	-.154	-.259	-.849
	165.286	-9.605	-31.514	-.047	-.154	-.260	-.851
S-II IGNITION	165.286	-6.838	-22.436	-.047	-.154	6.073	19.924
	167.786	-6.820	-22.375	-.047	-.155	6.090	19.979
MR SHIFT	167.786	-6.483	-21.268	-.047	-.155	6.859	22.505
	170.000	-6.464	-21.206	-.048	-.157	6.881	22.575
	180.000	-6.377	-20.922	-.049	-.162	6.979	22.899
	190.000	-6.290	-20.635	-.051	-.167	7.083	23.238
	193.986	-6.254	-20.520	-.052	-.169	7.125	23.377
WEIGHT DROP	193.986	-6.232	-20.445	-.052	-.169	7.177	23.546
	198.986	-6.187	-20.298	-.052	-.172	7.232	23.728
WEIGHT DROP	198.986	-6.165	-20.226	-.052	-.172	7.283	23.894
INITIATE GUIDANCE	198.986	-5.450	-17.881	-.052	-.172	6.928	22.730
	200.000	-5.451	-17.884	-.052	-.172	6.945	22.787
	210.000	-5.460	-17.915	-.054	-.177	7.119	23.356
	220.000	-5.474	-17.960	-.056	-.182	7.300	23.949
	230.000	-5.493	-18.020	-.057	-.188	7.487	24.565
	240.000	-5.516	-18.096	-.059	-.193	7.683	25.206
	250.000	-5.544	-18.189	-.060	-.198	7.887	25.875
	260.000	-5.577	-18.298	-.062	-.203	8.099	26.573
	270.000	-5.616	-18.425	-.063	-.208	8.321	27.301
	280.000	-5.661	-18.573	-.065	-.213	8.553	28.062
	290.000	-5.712	-18.741	-.066	-.217	8.796	28.858
	300.000	-5.770	-18.930	-.068	-.222	9.050	[illegible]

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BASELINE SA 511 TO LOR CASE 1

TABLE XIII

BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 9

APOLLO 13 ACCELERATION COMPONENTS

	TIME	XDDOT		YDDOT		ZDDOT	
	SEC	M/SEC**2	FT/SEC**2	M/SEC**2	FT/SEC**2	M/SEC**2	FT/SEC**2
	310.000	-5.834	-19.142	-.069	-.227	9.316	30.564
	320.000	-5.907	-19.379	-.071	-.232	9.596	31.482
	330.000	-5.987	-19.643	-.072	-.237	9.889	32.445
	340.000	-6.076	-19.933	-.074	-.242	10.198	33.459
	350.000	-6.174	-20.255	-.075	-.247	10.524	34.529
	360.000	-6.282	-20.610	-.077	-.251	10.868	35.658
	370.000	-6.400	-20.999	-.078	-.256	11.232	36.851
	380.000	-6.531	-21.426	-.079	-.261	11.618	38.115
	390.000	-6.674	-21.897	-.081	-.265	12.027	39.458
	400.000	-6.831	-22.412	-.082	-.270	12.462	40.886
	410.000	-7.003	-22.975	-.084	-.275	12.926	42.408
	420.000	-7.192	-23.596	-.085	-.279	13.422	44.035
	426.312	-7.320	-24.017	-.086	-.282	13.753	45.121
MR SHIFT	426.312	-7.637	-25.056	-.086	-.282	11.097	36.409
	430.000	-7.705	-25.278	-.087	-.284	11.226	36.832
	440.000	-7.899	-25.914	-.088	-.288	11.589	38.023
	450.000	-8.109	-26.604	-.089	-.293	11.975	39.287
	460.000	-8.337	-27.354	-.091	-.298	12.384	40.630
	470.000	-8.585	-28.166	-.092	-.302	12.820	42.061
	480.000	-8.853	-29.047	-.093	-.306	13.286	43.588
	490.000	-9.147	-30.009	-.095	-.311	13.784	45.224
	500.000	-9.466	-31.056	-.096	-.315	14.320	46.981
	510.000	-9.814	-32.197	-.097	-.319	14.897	48.874
	520.000	-10.195	-33.447	-.099	-.324	15.520	50.920
	530.000	-10.613	-34.819	-.100	-.328	16.197	53.140
	540.000	-11.072	-36.327	-.101	-.332	16.934	55.558
S-II CUTOFF	543.832	-11.261	-36.945	-.102	-.334	17.235	56.544
S-IVB IGNITION	543.832	-9.643	-31.638	-.102	-.334	3.637	11.932
	550.000	-9.714	-31.868	-.102	-.336	3.627	11.898
	560.000	-9.831	-32.254	-.104	-.340	3.609	11.840
	570.000	-9.952	-32.652	-.105	-.344	3.590	11.778
	580.000	-10.078	-33.063	-.106	-.348	3.570	11.712
	590.000	-10.207	-33.488	-.107	-.352	3.548	11.641
	600.000	-10.341	-33.926	-.109	-.356	3.525	11.565

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BASELINE SA 511 TO LOR CASE 1

TABLE NO. 9

TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

APOLLO 13 ACCELERATION COMPONENTS

	TIME	XDDOT		YDDOT		ZDDOT	
	SEC	M/SEC**2	FT/SEC**2	M/SEC**2	FT/SEC**2	M/SEC**2	FT/SEC**2
	610.000	-10.478	-34.378	-.110	-.360	3.501	11.485
	620.000	-10.620	-34.844	-.111	-.364	3.474	11.399
	630.000	-10.767	-35.325	-.112	-.368	3.447	11.308
	640.000	-10.918	-35.820	-.113	-.372	3.417	11.212
	650.000	-11.073	-36.330	-.114	-.375	3.386	11.110
	660.000	-11.233	-36.855	-.115	-.379	3.354	11.003
	670.000	-11.398	-37.396	-.117	-.383	3.319	10.890
	676.027	-11.500	-37.729	-.117	-.385	3.298	10.820
INJECTION	676.027	-11.500	-37.729	-.117	-.385	3.298	10.820
THRUST EVENT	676.027	-11.705	-38.403	-.117	-.385	2.388	7.836
	680.000	-11.730	-38.433	-.118	-.386	2.379	7.805
	690.000	-11.793	-38.692	-.119	-.390	2.357	7.731
	700.000	-11.859	-38.906	-.120	-.393	2.336	7.664
	710.000	-11.926	-39.128	-.121	-.397	2.317	7.603
	720.000	-11.996	-39.357	-.122	-.400	2.301	7.550
	730.000	-12.068	-39.594	-.123	-.403	2.287	7.504
	740.000	-12.143	-39.839	-.124	-.407	2.276	7.467
	750.000	-12.220	-40.093	-.125	-.410	2.267	7.439
	760.000	-12.300	-40.356	-.126	-.413	2.262	7.421
	770.000	-12.384	-40.629	-.127	-.416	2.260	7.414
	780.000	-12.470	-40.913	-.128	-.419	2.261	7.419
	790.000	-12.560	-41.208	-.129	-.422	2.267	7.437
	800.000	-12.654	-41.515	-.129	-.425	2.276	7.468
	810.000	-12.752	-41.837	-.130	-.427	2.290	7.515
	820.000	-12.854	-42.172	-.131	-.430	2.310	7.577
	830.000	-12.961	-42.524	-.132	-.432	2.334	7.658
	840.000	-13.074	-42.892	-.133	-.435	2.365	7.758
	850.000	-13.192	-43.279	-.133	-.437	2.401	7.878
	860.000	-13.316	-43.687	-.134	-.439	2.445	8.021
	870.000	-13.447	-44.118	-.134	-.441	2.496	8.189
	880.000	-13.586	-44.573	-.135	-.443	2.555	8.383
	890.000	-13.733	-45.055	-.136	-.445	2.623	8.607
	900.000	-13.889	-45.566	-.136	-.446	2.701	8.861
	910.000	-14.055	-46.111	-.136	-.447	2.789	9.151

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TABLE XIII
BASELINE LOR TRAJECTORY DATA (Cont'd)

TABLE NO. 9

APOLLO 13 ACCELERATION COMPONENTS

	TIME	XDDOT		YDDOT		ZDDOT	
	SEC	M/SEC**2	FT/SEC**2	M/SEC**2	FT/SEC**2	M/SEC**2	FT/SEC**2
	920.000	-14.232	-46.692	-.137	-.449	2.889	9.477
	930.000	-14.421	-47.313	-.137	-.450	3.001	9.844
	940.000	-14.624	-47.980	-.137	-.450	3.126	10.256
	950.000	-14.842	-48.695	-.137	-.451	3.266	10.716
	960.000	-15.077	-49.466	-.138	-.451	3.423	11.230
	970.000	-15.331	-50.300	-.138	-.451	3.597	11.802
	980.000	-15.606	-51.202	-.138	-.451	3.791	12.439
	990.000	-15.905	-52.182	-.137	-.451	4.007	13.146
INJECTION	993.160	-16.005	-52.509	-.137	-.451	4.080	13.386
	993.160	-5.813	-19.072	-.137	-.451	-6.818	-22.368

TABLE XIV

DEFINITION AND SYMBOLS FOR TRAJECTORY TABLES

Symbol	Table	Units	Definition
Time	all	seconds	Instantaneous time from liftoff
Inertial Velocity	1	m/sec ft/sec	Vehicle velocity in the space fixed coordinate system
Relative Velocity	1	m/sec ft/sec	Vehicle velocity in the earth fixed coordinate system
Gamma S	1	degrees	Flight path angle, measured from local horizontal to the inertial velocity vector
Gamma R	1	degrees	Flight path angle, measured from local horizontal to the relative velocity vector
Mass	2	kilograms	Vehicle mass at time t
Weight	2	pounds	Vehicle weight at time t
Thrust	2	Newtons pounds	Vehicle thrust at time t
Longit. Acceleration	2	m/sec^2 ft/sec^2	Acceleration along the longitudinal vehicle axis
Altitude	3	meters feet	Distance above the spheriod at time t
Radius	3	meters feet	Distance from center of earth at time t
Incl	3	degrees	Instantaneous inclination
Mach	4	-----	Mach number, based on relative velocity
Dyn Pressure	4	N/m^2 lb/ft^2	Dynamic pressure due to relative velocity
ALFP	4	degrees	Relative pitch attitude angle measured from relative velocity
AZ.S	5	degrees	Azimuth angle, measured in space fixed system from north to south over east

TABLE XIV (Cont'd)

DEFINITIONS AND SYMBOLS FOR TRAJECTORY TABLES

Symbol	Table	Units	Definition
Latitude	5	degrees	Geocentric latitude of vehicle position at time t
Longitude	5	degrees	Earth relative longitude at time t (+east)
CHIP	5	degrees	Apollo 13 pitch attitude command
CHIY	5	degrees	Apollo 13 yaw attitude command
AERO Heating Indicator	6	$N\text{-}m/m^2$ $lb\text{-}ft/ft^2$	$\int_o^t \frac{qV_R}{\frac{\pi}{2} - \alpha} dt$
Alpha*Q	6	N-radian lb-degree	Product of angle of attack and dynamic pressure
X	7	m ft	Apollo 13 position coordinates
Y	7	m ft	Apollo 13 position coordinates
Z	7	m ft	Apollo 13 position coordinates
X DOT	8	m/sec ft/sec	Apollo 13 velocity components
Y DOT	8	m/sec ft/sec	Apollo 13 velocity components
Z DOT	8	m/sec ft/sec	Apollo 13 velocity components
XD DOT	9	m/sec^2 ft/sec^2	Apollo 13 acceleration components
YD DOT	9	m/sec^2 ft/sec^2	Apollo 13 acceleration components
ZD DOT	9	m/sec^2 ft/sec^2	Apollo 13 acceleration components

4.0 AERODYNAMICS

Assigned aerodynamics characteristics for the SA-511 baseline are assumed to be the same as for the Apollo-Saturn V vehicle. These are given in NASA TM X-53517; Subject, Static Aerodynamics Characteristics of the Apollo - Saturn V Vehicle; dated September 16, 1966. The data includes normal force characteristics and local force distributions necessary for performance, control, and basic structural analyses of the vehicle. The axial force characteristics given in TM X-53517 have been updated based on flight results. These updated axial force characteristics have been assigned to SA-511 and are given in MSFC Memorandum R-AERO-AD-68-37, Subject, Saturn V Axial Force Characteristics; dated June 18, 1968, and Figures 2, 3, and 4 of this memorandum.

5.0 FLIGHT CONTROL

5.1 CONTROL SYSTEM DESIGN OBJECTIVES

General control system design objectives for each of the three flight stages are given below.

a. Provide adequate control to ensure stable flight

b. Provide well damped responses in order to ensure vehicle structural integrity

c. Compensate for component and structural tolerances

Vehicle flight may be classified into the following categories.

a. Nominal — System or parameters within the range having a 3σ probability of occurrence.

  1. Reference — System or parameters at the specified design values.

  2. Three Sigma — System or parameters at the boundary of $\pm 3\sigma$ (0.9973 probability of occurrence) range.

b. Off Nominal — System or parameters outside the $\pm 3\sigma$ range.

Specific stability objectives to satisfy the above stated general objectives are listed below. These objectives should be used as desired objectives and should not be construed as specifications. Definitions used in the stability objectives are as follows:

a. Phase Margin — The amount of phase lead or lag required to cause the characteristic roots of a mode of motion to just depart from the left-half s-plane, i.e., become unstable.

b. Gain Margin — The amount of gain increase required to cause the characteristic roots of a mode of motion to just depart from the left-half s-plane, i.e., become unstable.

c. Control System Gain Margin (S-IC only)

The smallest gain increase in the rigid body and slosh frequency range required to cause the characteristic root (s) to just depart from the left-half plane, i. e., become unstable, with the exception of those gain margins related to hydrogen-tank slosh. These are excluded because their contribution to vehicle response is negligible due to their small modal masses.

d. Peak Gain Margin

The minimum increase in gain required to cause the characteristic root (s) of a mode of motion to just depart from the left-half s-plane in the presence of the most adverse phase lead or lag.

e. Aerodynamic Gain Margin

The decrease in gain required to cause the characteristic root (s) introduced by aerodynamics to just depart from the left-half s-plane.

5.2 NOMINAL FLIGHT

Control system design objectives for nominal flight are given in the following subparagraphs.

To meet the design objectives, it is expedient to define certain stability margin objectives for the linear system open at each of the three locations:

a. Engine summation point

b. Attitude error channel

c. Attitude rate channel

Structural mode peak gain margin objectives depend on the structural damping values used. Where the damping value obtained from structural tests is used in analysis, a smaller margin objective is used. Prior to tests, all margin objectives were large because damping values used were not considered conservative.

5.2.1 S-IC Reference Flight

The function of the S-IC control system is to provide adequate vehicle control such that the desired trajectory is followed with sufficient accuracy that any dispersions at S-IC burnout can be compensated by the upper flight stages. Stability objectives for the linear pitch (yaw) system, using reference parameter values, are as follows:

a. A minimum aerodynamic gain margin of 6 db

b. A minimum control system gain margin of 6 db

c. A minimum control system phase margin of 30 degrees

d. A minimum phase margin for phase stabilized bending modes of 45 degrees

e. A minimum gain margin at the peak of gain stabilized bending modes of 6 db

f. A minimum gain margin at the peak of LOX and RP-1 slosh modes of 0 db

No objective is specified for hydrogen slosh modes since their contribution to vehicle response is negligible due to their small modal masses.

Because of vehicle symmetry in the pitch and yaw planes, objectives for the pitch and yaw control systems are identical.

Stability objectives for the linear roll control system, using reference parameter values, are as follows:

a. A minimum control system gain margin of 6 db

b. A minimum control system phase margin of 30 degrees

c. A minimum gain margin for all torsional mode peaks of 12 db

5.2.2 S-II Reference Flight

The functions of the S-II control system are to provide adequate recovery capability following S-IC/S-II separation and to provide adequate control for the remainder of the flight such that the desired trajectory is followed with sufficient accuracy that any dispersions at S-II burnout can be compensated by the S-IVB flight stage.

Stability objectives for the linear pitch (yaw) system, using reference parameter values, are as follows:

a. A minimum control system gain margin of 6 db

b. A minimum control system phase margin of 35 degrees

c. A minimum phase margin for the first bending mode of 45 degrees

d. A minimum gain margin for the second bending mode peak prior to LET jettison of 6 db, and after LET jettison of 12 db

e. A minimum gain margin for the third and fourth bending mode peak of 12 db

f. A minimum phase margin for the LOX slosh modes of 35 degrees

Because of vehicle symmetry in the pitch and yaw planes, objectives for the pitch and yaw control systems are identical.

Stability objectives for the linear roll control system, using reference parameter values are as follows:

a. A minimum control system gain margin of 6 db

b. A minimum control system phase margin of 35 degrees

c. A minimum gain margin for all torsional mode peaks of 12 db

5.2.3 S-IVB REFERENCE FLIGHT

The S-IVB control system is designed to provide adequate recovery capability following S-II/S-IVB separation and to provide adequate control for the remainder of flight such that the design trajectory is followed with sufficent accuracy that any dispersions at burnout do not significantly affect the vehicle mission. Linear pitch (yaw) system stability objectives for powered flight, using reference parameter values, are as follows:

a. A minimum control system gain margin of 6 db

b. A minimum control system phase margin of 35 degrees

c. Minimum bending mode peak gain margins of 12 db

d. A minimum phase margin for LOX slosh of 35 degrees

5.2.4 Three Sigma Flight

The stability objectives considering 3 σ parameter variations for the S-IC, S-II, and S-IVB stages are:

a. a minimum aerodynamic gain margin of 3 db

b. a minimum control system gain margin of 3 db

c. a minimum control system phase margin of 15 degrees

d. a minimum slosh peak gain margin of 0 db (S-IC only)

e. a minimum slosh phase margin of 20 degrees

f. a minimum bending mode phase margin of 20 degrees

g. a minimum bending mode gain margin of 3 db

5.3 OFF NOMINAL FLIGHT

Control system design objectives for off-nominal flight are to recover and successfully stage following S-IC or S-II engine malfunction or actuator hardover conditions. Recovery requires that the vehicle remain controllable and maintain vehicle dynamics such that structural failure does not occur.

5.4 ACCOMPLISHMENT OF OBJECTIVES

5.4.1 Stability

AS-504 stability data are the currently available data most representative of the expected configuration of the 511 vehicle. These data are found in the Boeing Company Document, D5-15508-4, "Launch Vehicle Flight Control System Stability Analysis, SA-504 (Initial)," July 15, 1967. The degree to which the control system designed for this vehicle meets the stability design objectives is summarized in the following data and conclusions as presented in the referenced document.

Results of the analysis are:

a. S-IC pitch control system reference gain and phase margins of 3.4 db and 24.7° are below the design objectives of 6 db and 30° respectively as shown in Table XV. Margins with 3σ tolerances (including nonlinear effects) are reduced to 1.6 db and 17.5°. Control system phase margin is not considered a problem since the 3σ objective of 15° is met. Control system gain margin is below the 3 σ objective of 3 db; however, this condition results from the use of slosh damping

TABLE XV
S-IC PITCH (YAW) MINIMUM STABILITY MARGINS

STABILITY MARGIN	FLIGHT TIME (SEC)	REFERENCE		THREE SIGMA	
		DESIGN OBJECTIVE	MINIMUM VALUE	OBJECTIVE	MINIMUM VALUE
Aerodynamic Gain Margin	83	6 db	6.7 db	3 db	5.4 db
Control System Phase Margin	110^-	30^o	[24.7^o]	15^o	17.5^o
Control System Gain Margin	110^-	6 db	[3.4 db]	3 db	[1.6 db]
First Bending Mode Phase Margin	0	45^o	60.0^o	20^o	39.5^o
Second Bending Mode Peak Gain Margin	0	6 db	6.8 db	3 db	4.0 db
Third Bending Mode Peak Gain Margin	110^-	6 db	9.6 db	3 db	6.2 db
Fourth Bending Mode Peak Gain Margin	100	6 db	20.7 db	3 db	17.0 db
Fifth Bending Mode Peak Gain Margin	159	6 db	15.3 db	3 db	10.1 db
Slosh Peak Gain Margin (LOX and RP-1)	110^-	0 db	3.0 db	0 db	1.3 db

[ ] Margin is below objective value.

corresponding to low tank wave heights (0. 05 m). With larger wave heights, damping increases sufficiently to raise the control gain margin above its 3σ objective value.

b. For the SA-504 vehicle, S-IC TVC nonlinearities cause a 0. 3 db reduction in minimum gain margin. Previous analysis indicated a 2. 0 db reduction. This change in effect of TVC nonlinearities is caused by the SA-504 vehicle gain margin being relatively insensitive to the phase shift resulting from nonlinearities.

c. A trade study to evaluate the S-IC control gain switch time indicates that the nominal 110 second switch time provides early engine-out capability with 95 percentile winds. Engine failures only after 6 seconds were considered. A change to 100 second switching will give an improvement of 1. 1 db in the minimum control system gain margin, but at the expense of more marginal early engine-out capability due to poorer transient response. It is recommended that the 110 second gain switch be retained. The second gain switch time is satisfactory at 130 seconds.

d. S-IC roll control system gain and phase margins are 5. 4 db and 29.2° as shown in Table XVI. These are below the reference objectives of 6 db and 30°; however, the design is considered satisfactory as the 3σ margins meet the design objectives.

e. S-IC roll first torsional peak gain margin is 8. 7 db; under 3σ conditions, the margin reduces to 0. 8 db. However, the first torsional mode is phase stabilized with a peak phase margin of 152° for reference conditions. Torsional mode gain margins are reduced from the SA-501 analysis primarily as a result of lower torsional frequencies corresponding to the heavier vehicle, and tolerance increases. It is recommended that an alternate objective of phase stabilization be employed on the first torsional mode because tolerances that affect phase shift are defined sufficiently to establish 3σ phase conditions. At the frequency in question (6 Hz) a 3σ objective of 45° phase margin and 75° phase margin measured at peak modal amplitude are considered adequate.

f. The second torsional mode 3σ gain margin is also below the 3 db objective of 2. 6 db. This mode is phase stabilized with a peak phase margin of 124°. Because of the conservative tolerances placed on torsional data, the second torsional mode 3σ gain margin is considered adequate at 0. 4 db below the objective. It is recommended that the objective be relieved sufficiently to satisfy the calculated margin.

g. S-II stage second bending mode peak gain margin prior to LET jettison is below the objective with a value of 3. 0 db as shown in Table XVII. Under 3σ conditions, this reduces to -1. 7 db. This mode is adequately phase stabilized with a peak phase margin of 174° during the period that gain margin objectives are not met An alternate objective of phase stabilization is recommended.

TABLE XVI
S-IC ROLL STABILITY MARGINS AT 159 SECONDS

STABILITY MARGIN	REFERENCE		THREE SIGMA	
	OBJECTIVE	VALUE	OBJECTIVE	VALUE
Control System Gain Margin	6 db	[5.4 db]	3 db	4.4 db
Control System Phase Margin	35^{o}	[29.2^{o}]	15^{o}	20.1^{o}
First Torsion Peak Phase Margin	*	152^{o}	*	92^{o}
First Torsion Peak Gain Margin	12 db	[8.7 db]	3 db	[0.8 db]
Second Torsion Peak Gain Margin	12 db	14.0 db	3 db	[2.6 db]
Second Torsion Peak Phase Margin	*	124^{o}	*	
Third Torsion Peak Gain Margin	12 db	27.2 db	3 db	15.4 db
Fourth Torsion Peak Gain Margin	12 db	17.7 db	3 db	9.8 db

* Objective values are not given because design is based on gain stabilization.

[ ] Margin is below objective value.

TABLE XVII
S-II PITCH (YAW) MINIMUM STABILITY MARGINS

STABILITY MARGIN	FLIGHT TIME (SEC)	REFERENCE		THREE SIGMA	
		DESIGN OBJECTIVE	MINIMUM VALUE	OBJECTIVE	MINIMUM VALUE
Control System Phase Margin	190^{+}	35^{o}	39.5^{o}	15^{o}	30^{o}
Control System Gain Margin	190^{-}	6 db	11.0 db	3 db	9.5 db
LOX Slosh Phase Margin	190^{-}	35^{o}	30.5^{o}	20^{o}	23.5^{o}
LOX Slosh Peak Gain Margin	60^{-}	* *	-31.0 db	* *	-33.5 db
First Bending Mode Phase Margin At -6 db	31^{-}	45^{o}	[43.5^{o}]	20^{o}	28.0^{o}
Second Bending Mode Peak Gain Margin	31^{-}	12 db	[3.0 db]	3 db	[- 1.7 db]
Second Bending Mode Peak Phase Margin	31^{-}	*	174^{o}	*	
Third Bending Mode Peak Gain Margin	60^{-}	12 db	33.0 db	3 db	25.5 db
Fourth Bending Mode Peak Gain Margin	190^{-}	12 db	23.5 db	3 db	16.5 db

[ ] Margin is below objective value.

* Objective value is not given because design is based on gain stabilization.

** Objective value is not given because design is based on phase stabilization.

h. Table XVIII and XIX show that the S-II roll control system and the S-IVB pitch control system meet all stability objectives.

5.4.2 Responses

The rigid body responses of various control variable of AS-504 vehicle (considered representative of 511 vehicle responses) to a 95 percentile nondirectional design wind applied as a cross wind are shown in Figures 5 through 10. Both the nominal condition and three sigma condition are included for the wind peaking at the 10 kilometer altitude which represents the approximate condition of maximum dynamic pressure-angle of attack product. Control is adequate and well damped.

TABLE XVIII

S-II ROLL STABILITY MARGINS AT T = 31^-

STABILITY MARGIN	REFERENCE		THREE SIGMA	
	OBJECTIVE	VALUE	OBJECTIVE	VALUE
Control System Phase Margin	35^o	43.9^o	15^o	35.3^o
Control System Gain Margin	6 db	15.2 db	3 db	14.4 db
First Torsion Peak Gain Margin	12 db	25.4 db	3 db	18.3 db
Second Torsion Peak Gain Margin	12 db	20.6 db	3 db	12.6 db
Third Torsion Peak Gain Margin	12 db	28.0 db	3 db	17.3 db
Fourth Torsion Peak Gain Margin	12 db	51.8 db	3 db	37.1 db
Fifth Torsion Peak Gain Margin	12 db	39.7 db	3 db	29.2 db

TABLE XIX
S-IVB PITCH (YAW) MINIMUM STABILITY MARGINS

STABILITY MARGIN	FLIGHT TIME (SEC.)	REFERENCE		THREE SIGMA	
		OBJECTIVE	MINIMUM VALUE	OBJECTIVE	MINIMUM VALUE
Control System Phase Margin	381^+	35^o	[33.5^o]	15^o	22^o
Control System Gain Margin	381^-	6 db	8.7 db	3 db	7.5 db
LOX Slosh Phase Margin	121^-	35^o	35.5^o	20^o	26.6^o
LOX Slosh Peak Gain Margin	381^-	*	-54.3 db	*	-56.5 db
First Bending Mode Peak Gain Margin	80	12 db	26.8 db	3 db	20.5 db
Second Bending Mode Peak Gain Margin	0	12 db	26.5 db	3 db	17.5 db
Third Bending Mode Peak Gain Margin	121^-	12 db	34.5 db	3 db	25.5 db
Fourth Bending Mode Peak Gain Margin	121^-	12 db	22.5 db	3 db	13.5 db

[ ] Margin is below the objective.

* Objective value is not given because design is based on phase stabilization.

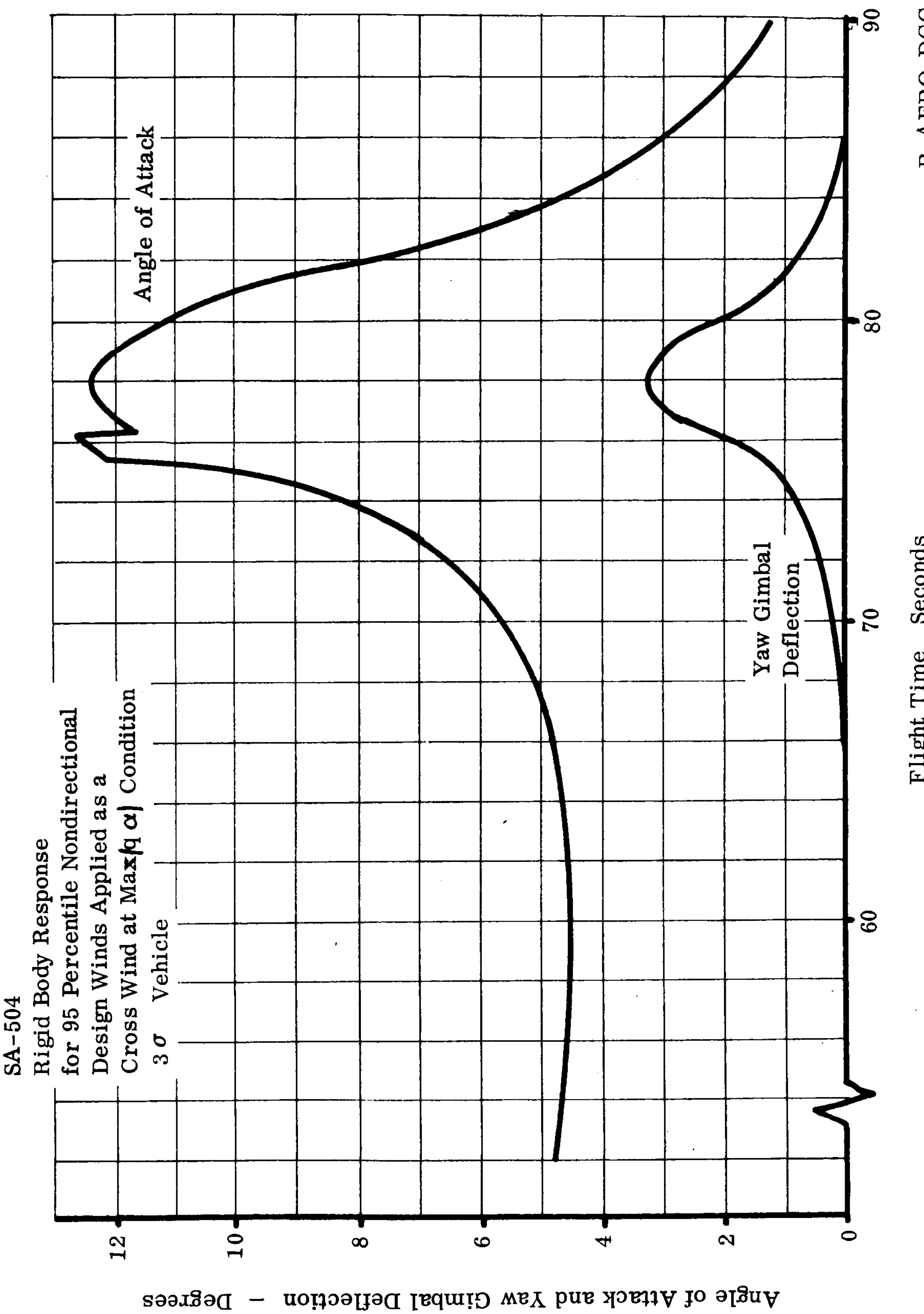


FIGURE 5 ANGLE OF ATTACK & YAW GIMBAL DEFLECTION FOR 3 σ VEHICLE

R-AERO-DCC

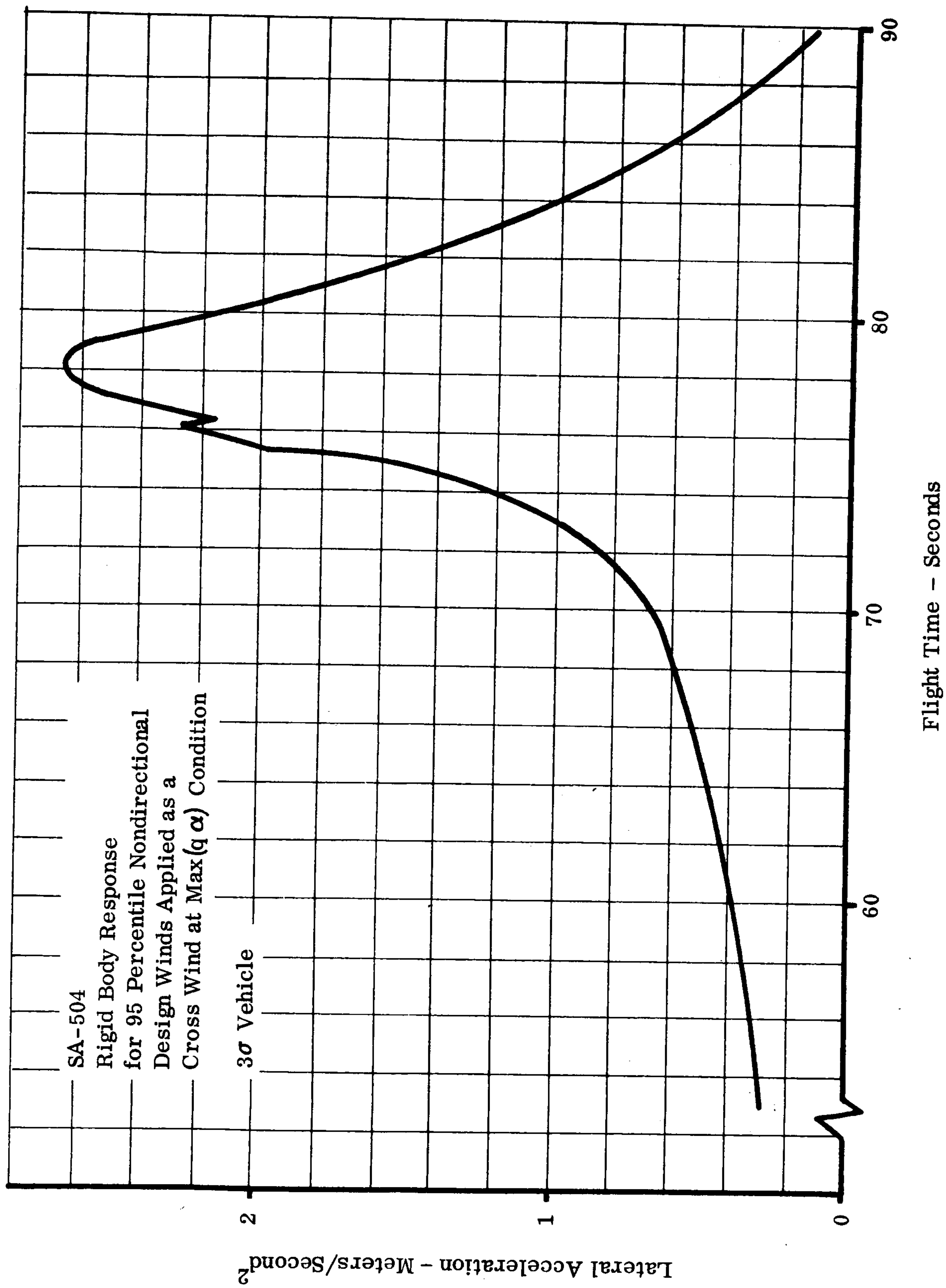
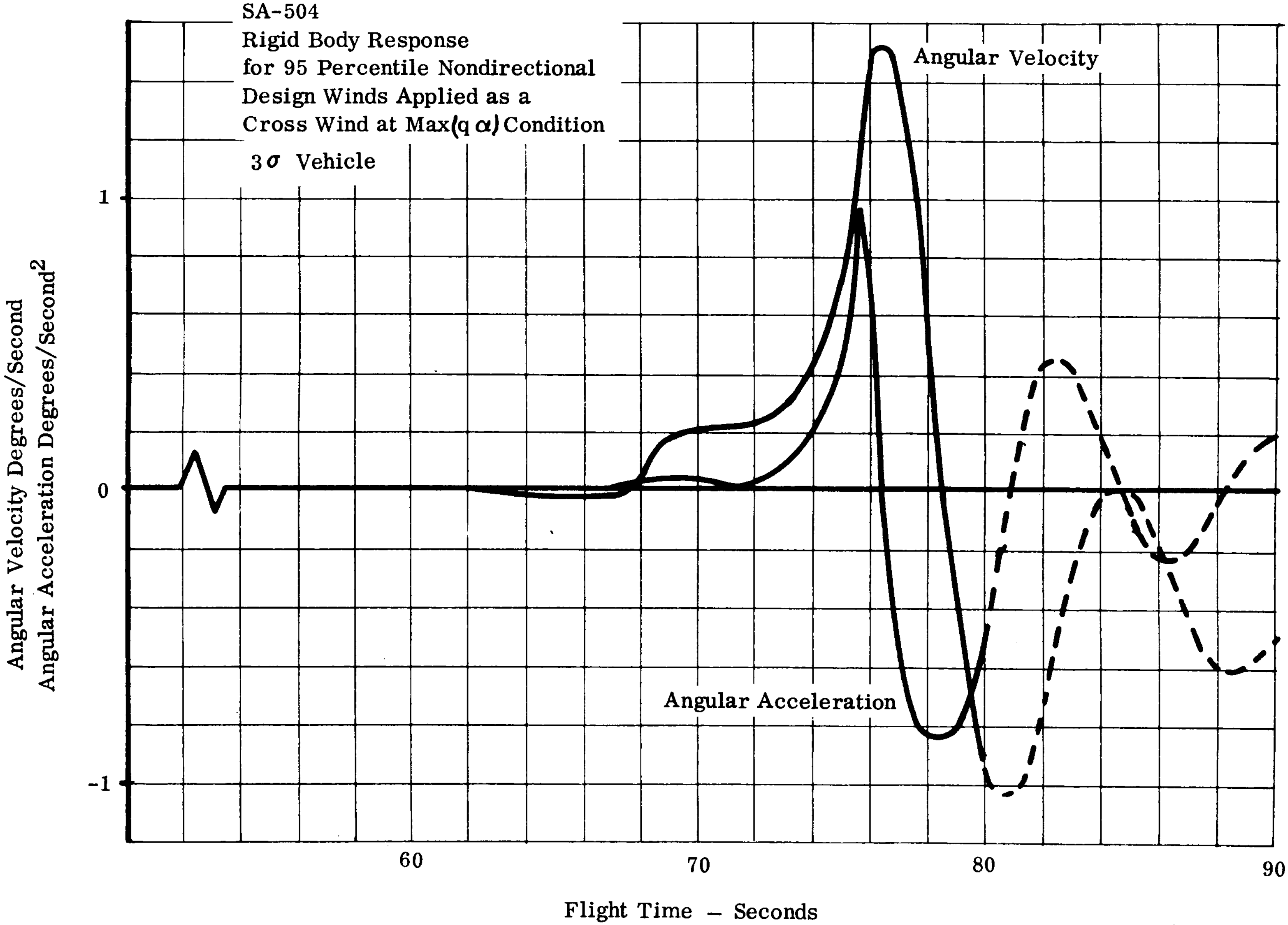


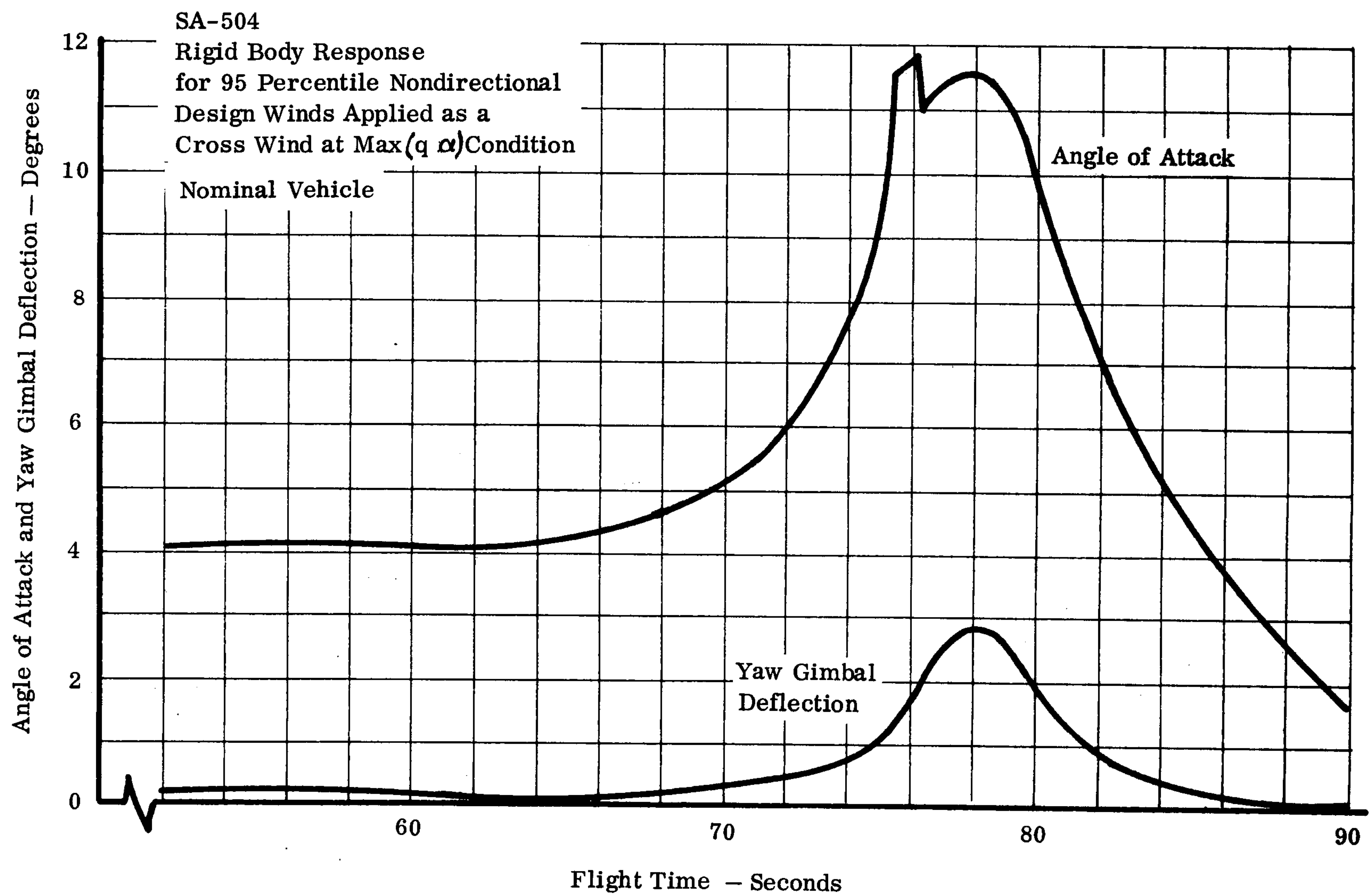
FIGURE 6 LATERAL ACCELERATION FOR 3σ VEHICLE

R-AERO-DCC



R-AERO-DCC

FIGURE 7 ANGULAR VELOCITY & ANGULAR ACCELERATION FOR 3 σ VEHICLE



R-AERO-DCC

FIGURE 8 ANGLE OF ATTACK & YAW GIMBAL DEFLECTION FOR NOMINAL VEHICLE

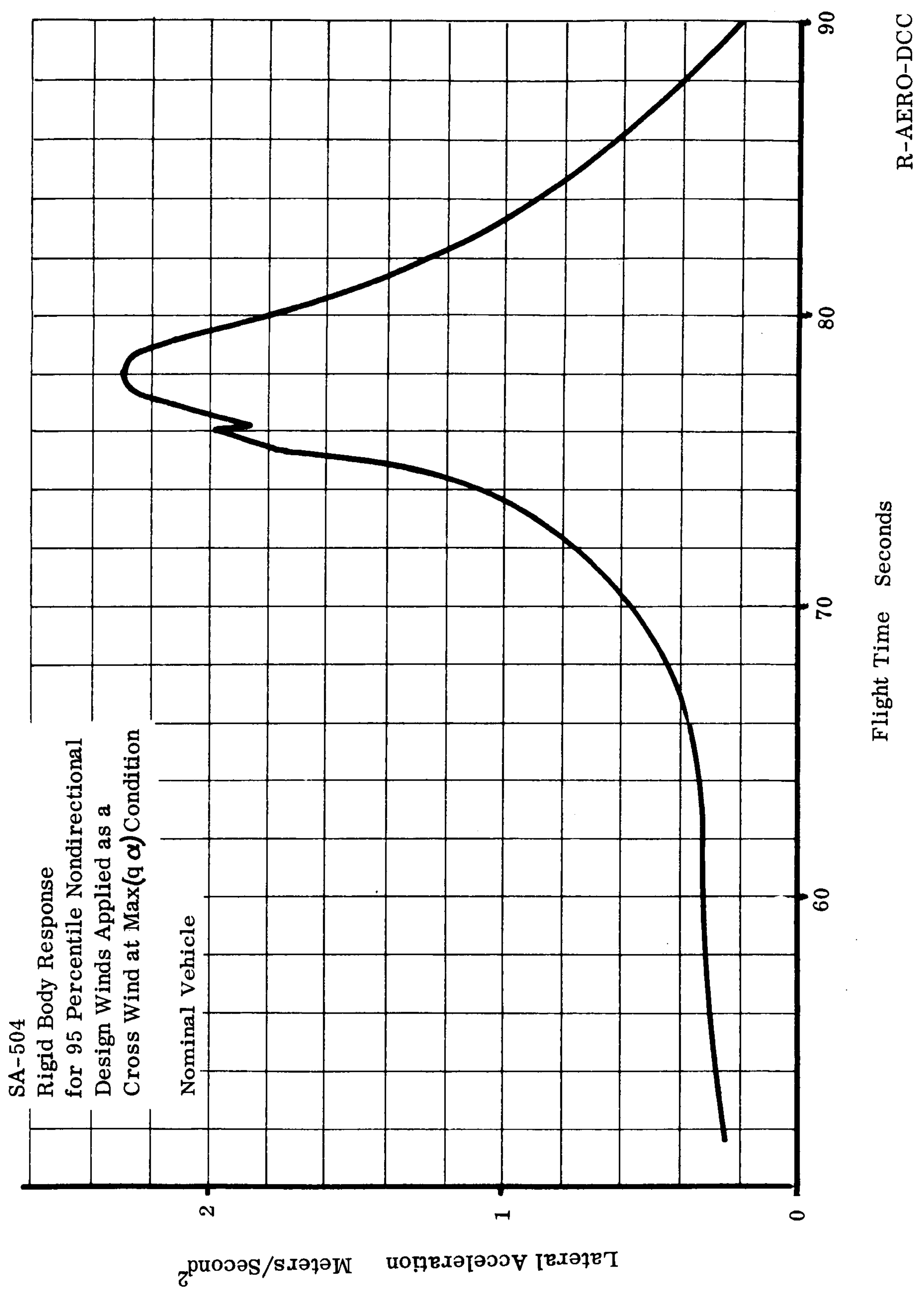


FIGURE 9 LATERAL ACCELERATION FOR NOMINAL VEHICLE

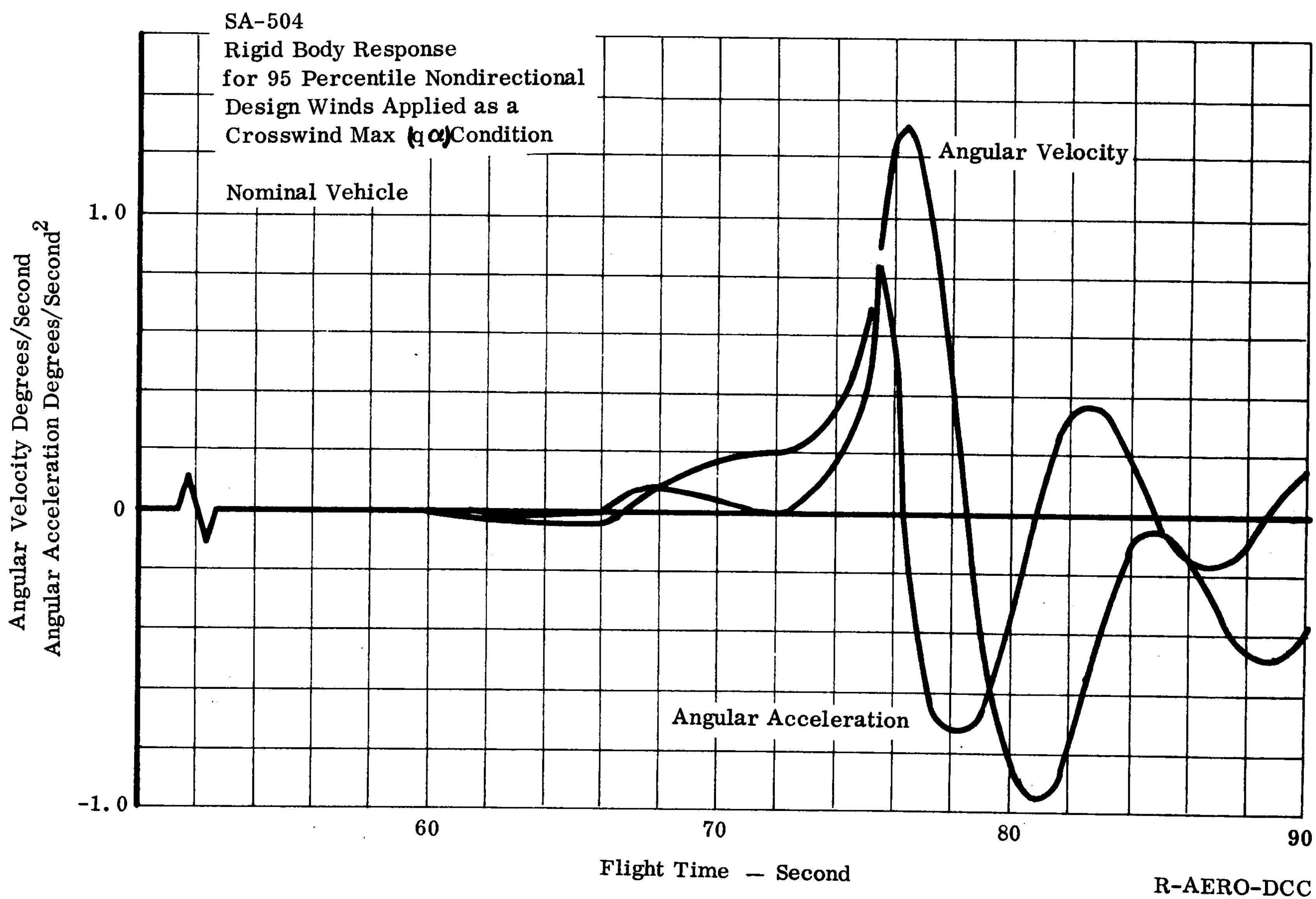


FIGURE 10 ANGULAR VELOCITY & ANGULAR ACCELERATION FOR NOMINAL VEHICLE

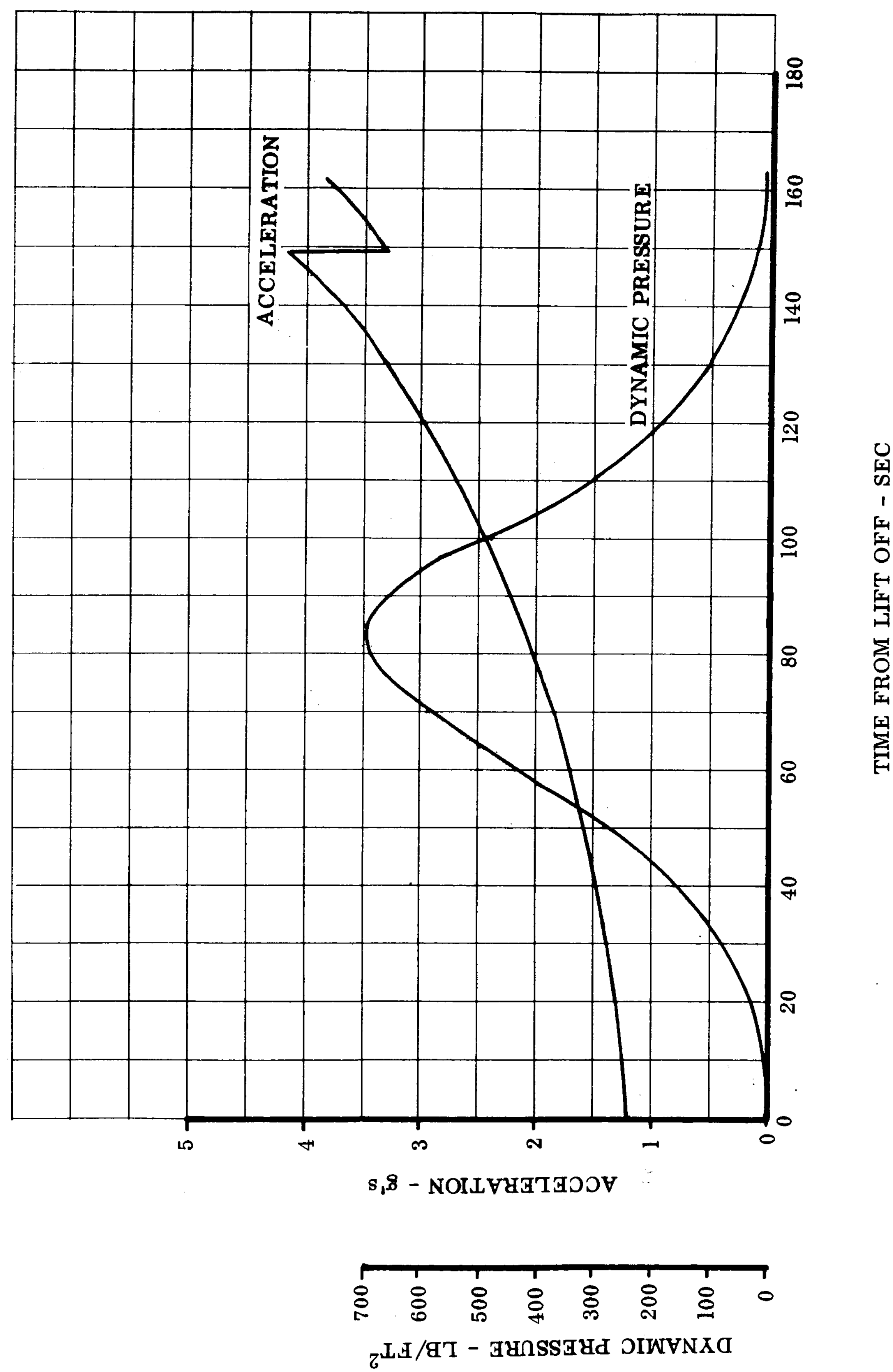
6.0 STRUCTURAL LOADS

Longitudinal acceleration and dynamic pressure during first stage boost is given in Figure 11. The maximum acceleration is 4.16 g's and the maximum dynamic pressure is 700 lb/ft^2.

Ground wind shear distribution for a 99.9 percent prelaunch wind (unfueled vehicle) and a 99 percent launch wind (fueled vehicle) are given in Figures 12 and 13. Inflight shear distribution for the maximum (q_α) time of flight is given in Figure 14.

Bending moment distribution for a 99.9 percent prelaunch wind (damper attached) and 99 percent launch wind (damper detached) are given in Figures 15 and 16. Inflight bending moment distribution for maximum (q_α) time of flight is given in Figure 17.

On-pad longitudinal force distribution (fueled vehicle) with a load factor of 1.0 g's is given in Figure 18. Longitudinal force distributions during the S-IC center engine cutoff (CECO) and maximum (q_α) times of flight are shown in Figures 19 and 20.



R-P&VE-SLL

FIGURE 11 VEHICLE LONGITUDINAL ACCELERATION & DYNAMIC PRESSURE PROFILES

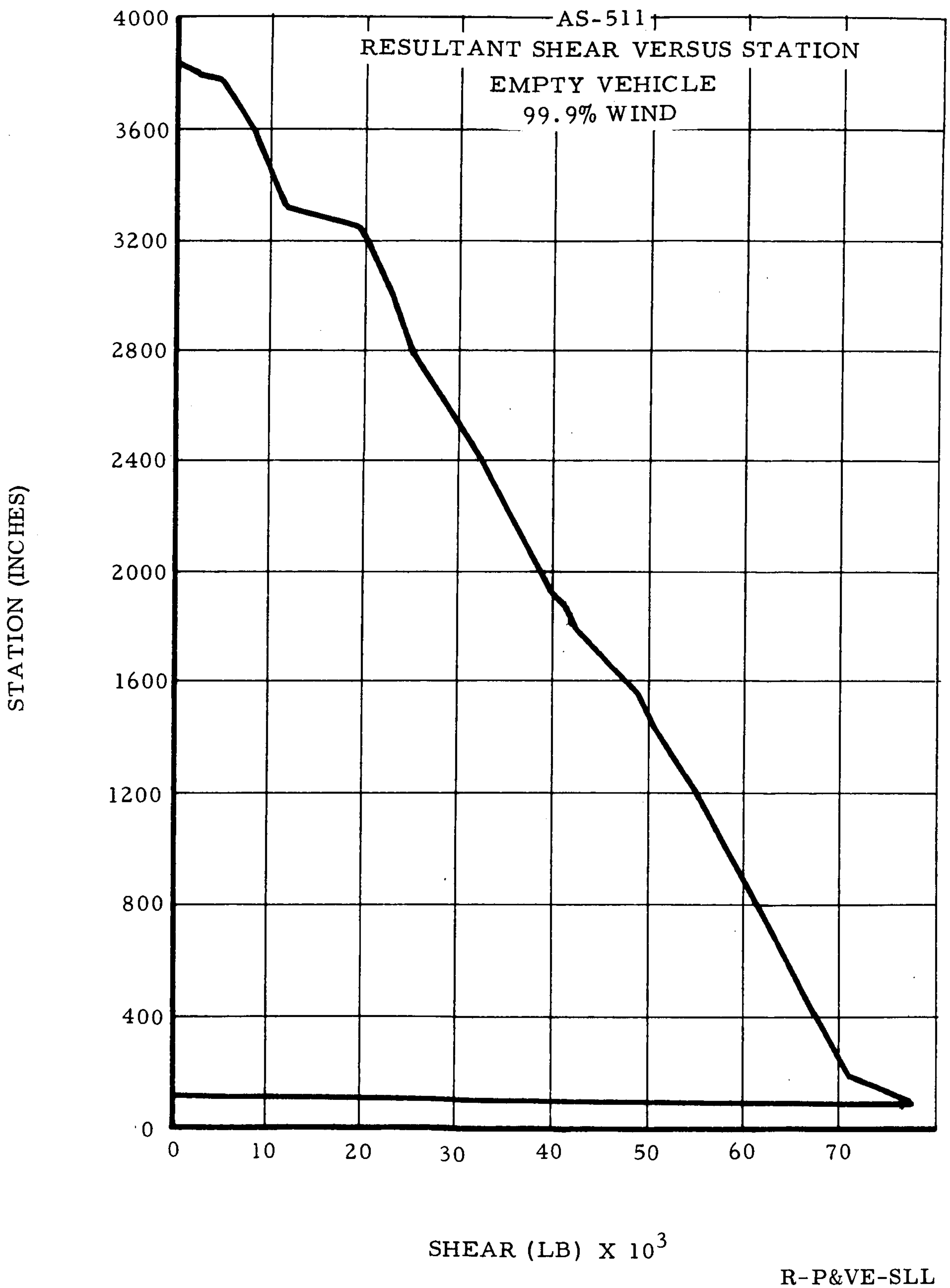
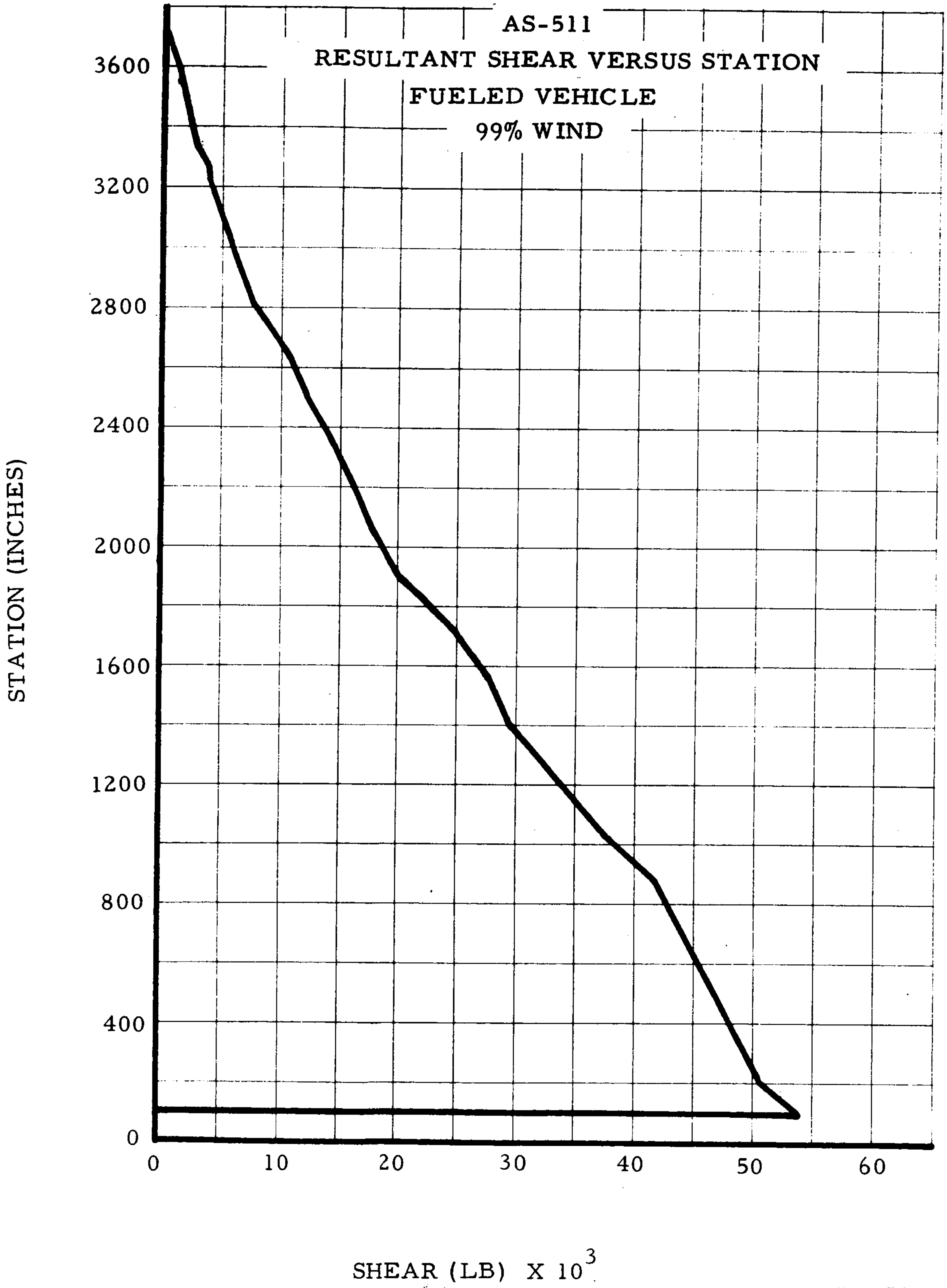


FIGURE 12 ON-PAD SHEAR FORCE DISTRIBUTION FOR 99.9 PERCENT WIND



R-P&VE-SLL

FIGURE 13 ON-PAD SHEAR FORCE DISTRIBUTION FOR 99 PERCENT WIND

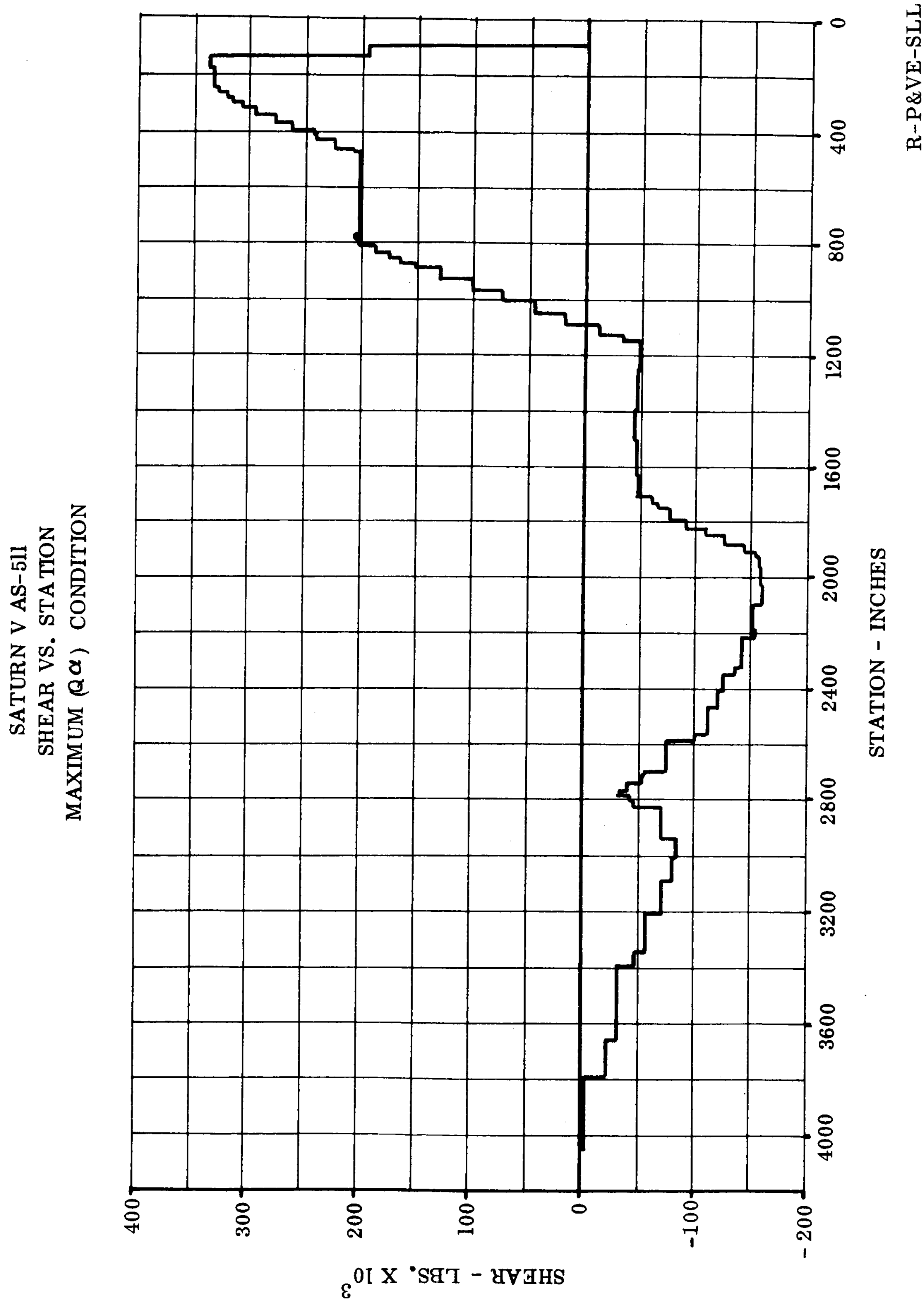
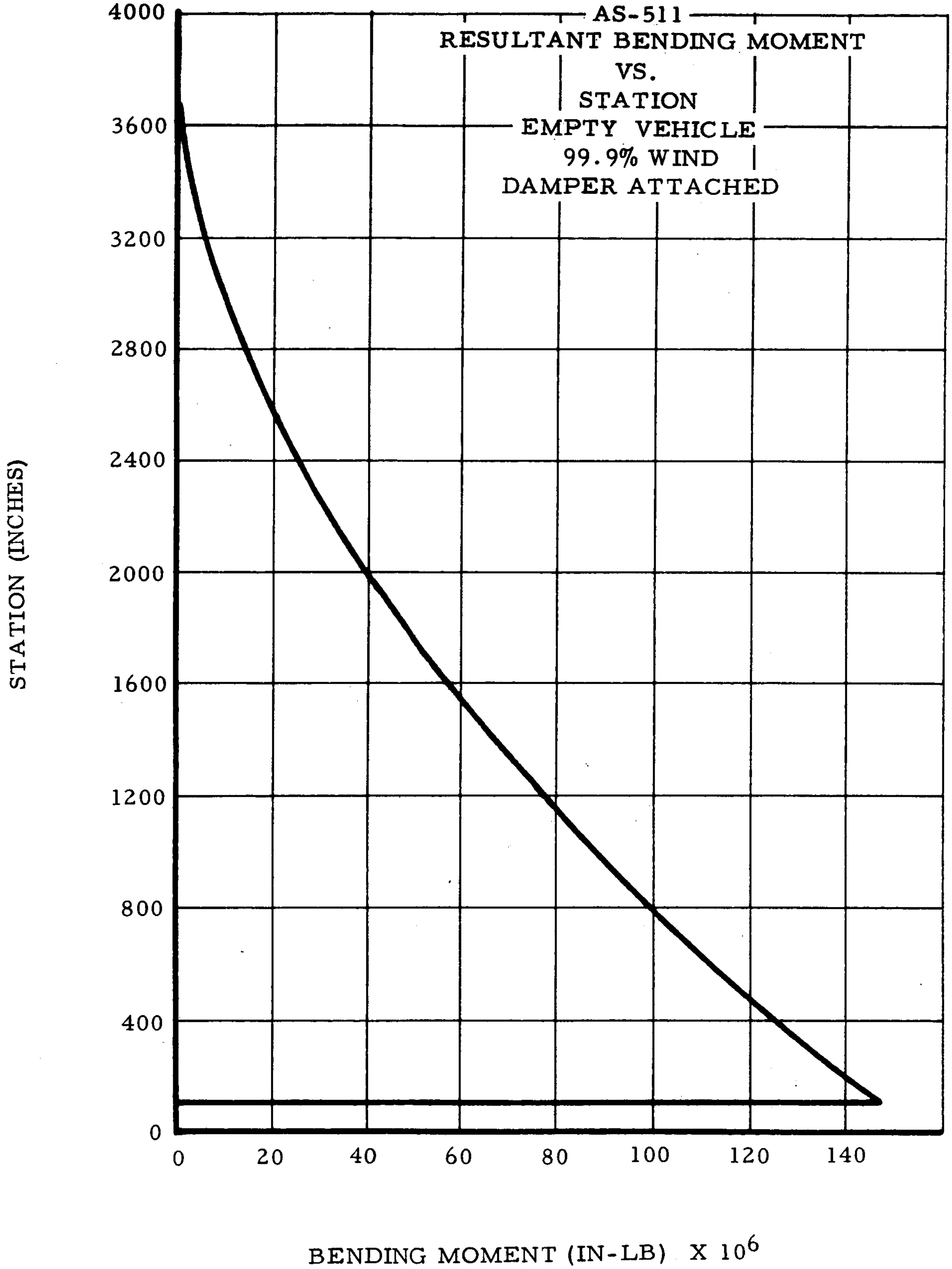
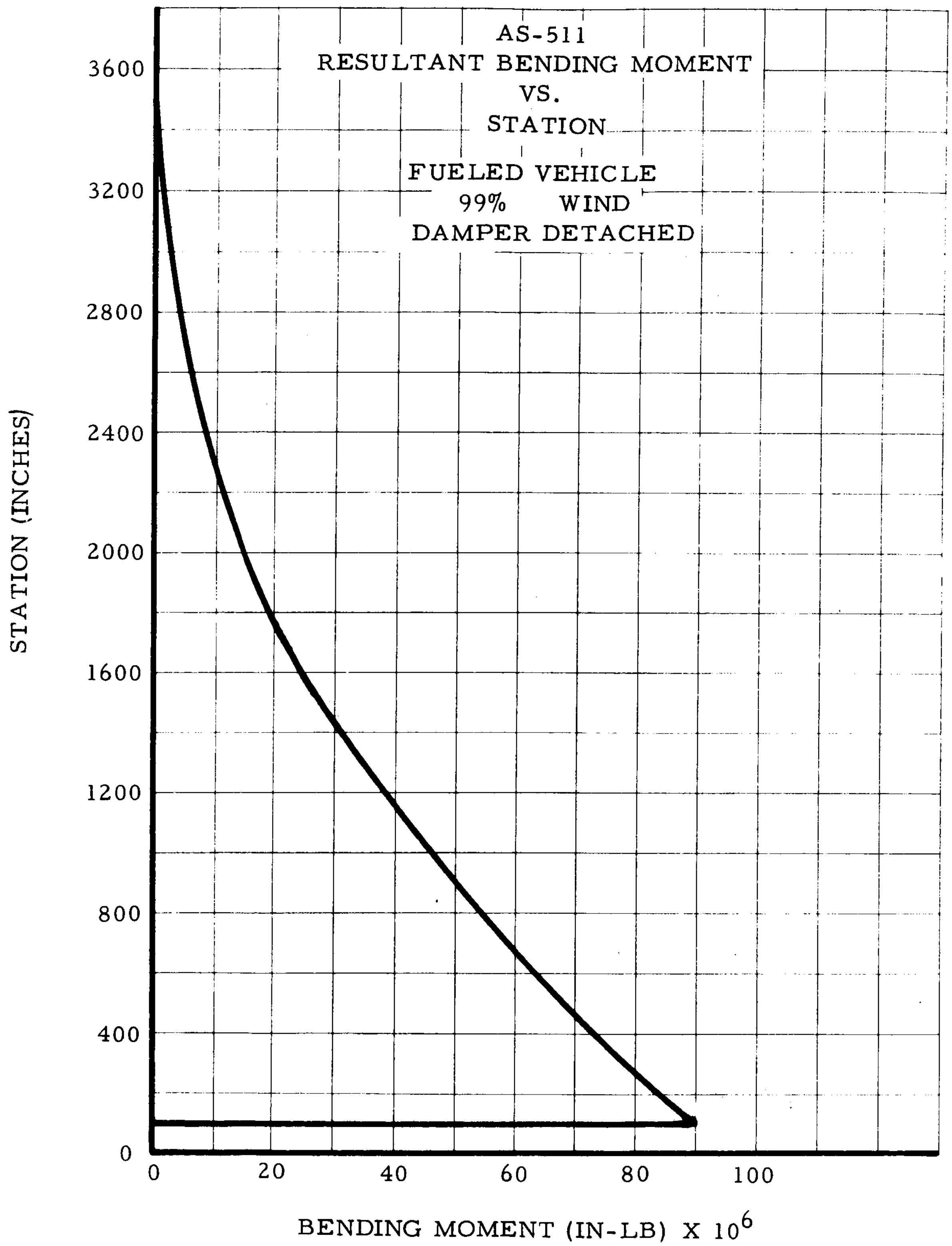


FIGURE 14 SHEAR DISTRIBUTION AT MAXIMUM (Qα) CONDITION



R-P&VE-SLL

FIGURE 15 ON-PAD BENDING MOMENT DISTRIBUTION FOR 99.9 PERCENT WIND



R-P&VE-SLL

FIGURE 16 ON-PAD BENDING MOMENT DISTRIBUTION FOR 99 PERCENT WIND

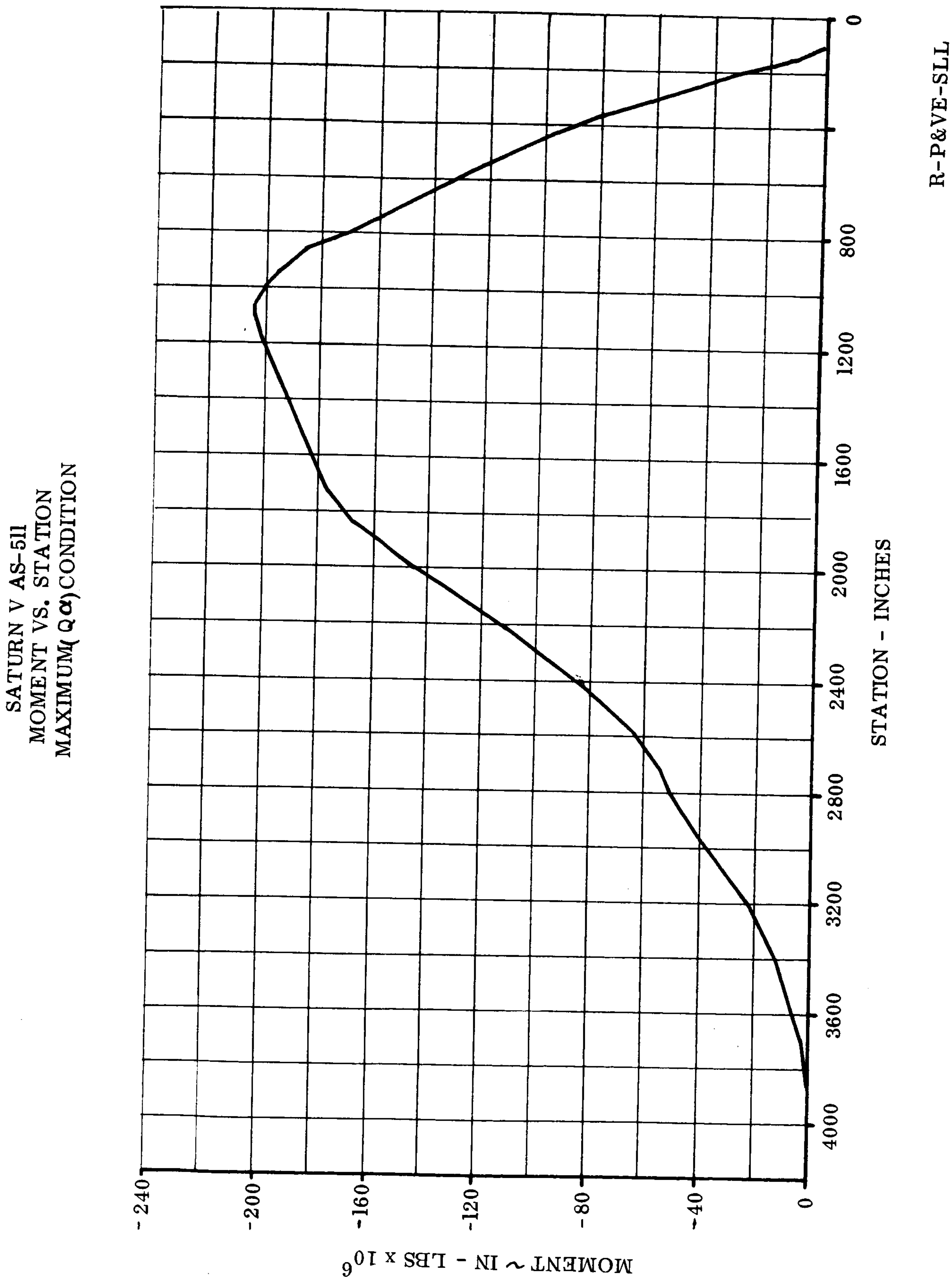
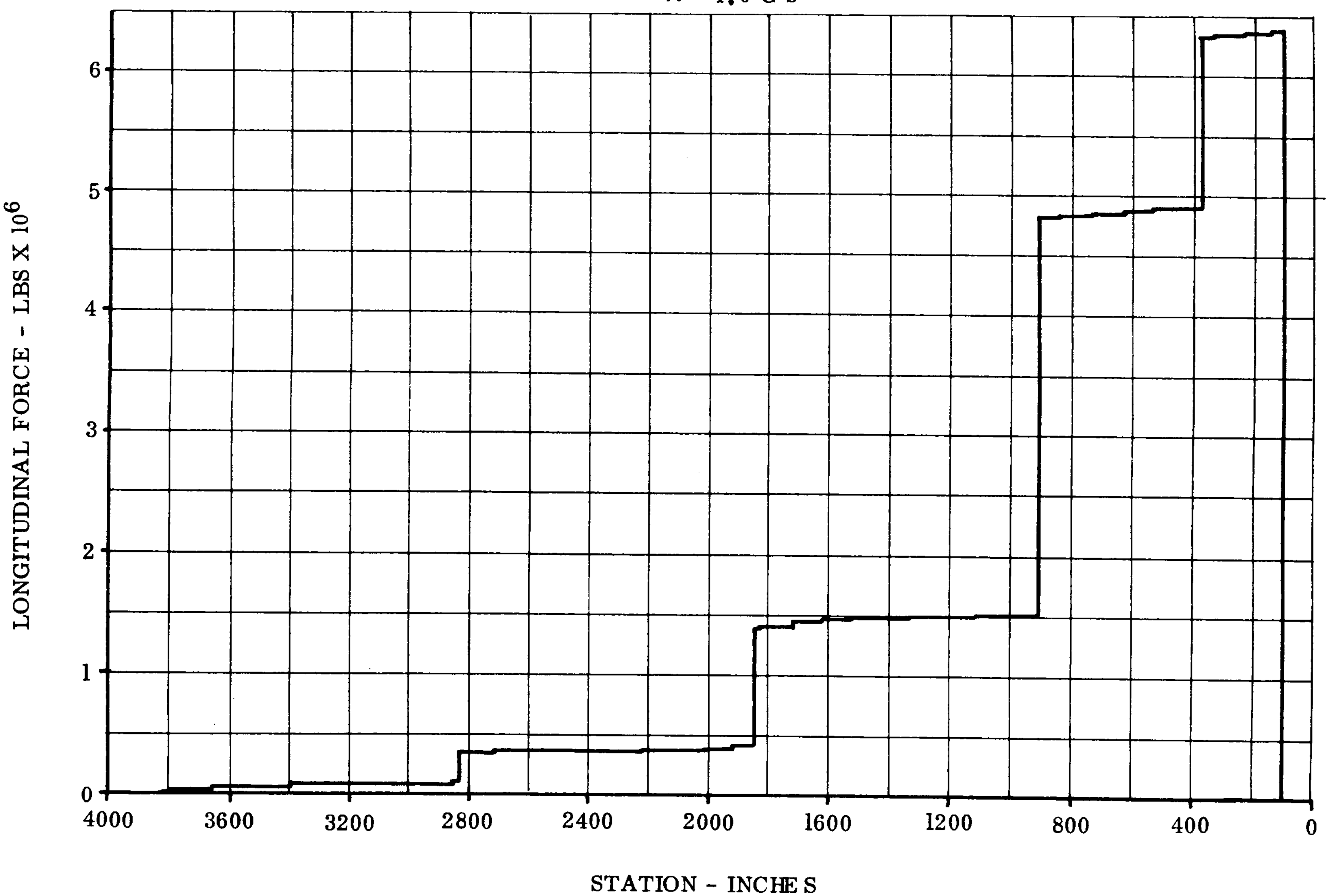


FIGURE 17 BENDING MOMENT DISTRIBUTION AT MAXIMUM (Qα) CONDITION

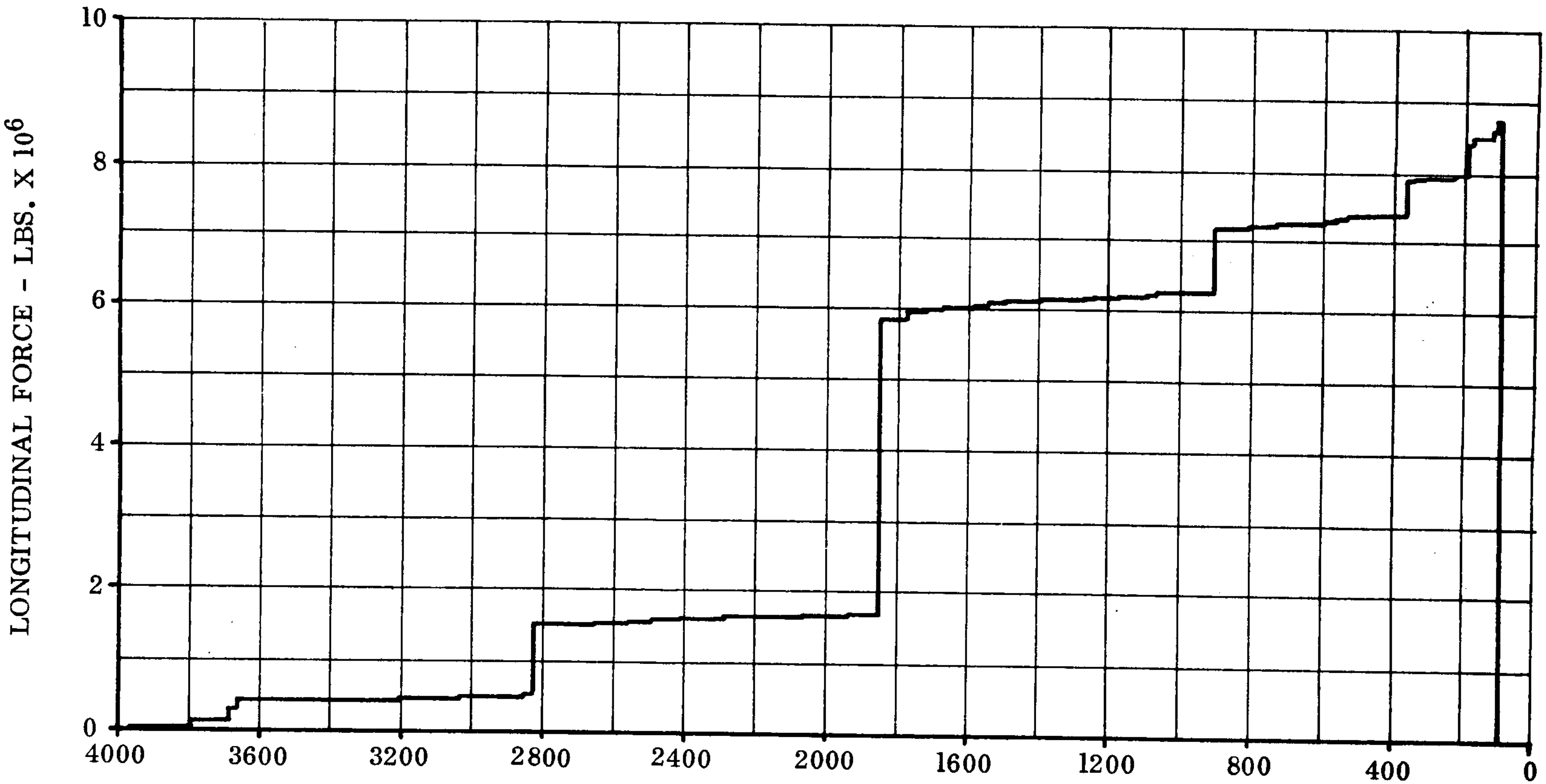
SATURN V AS-511
LONGITUDINAL FORCE VS. STATION
FUELED VEHICLE
$\lambda = 1.0$ G's



R-P&VE-SLL

FIGURE 18 ON-PAD LONGITUDINAL FORCE DISTRIBUTION

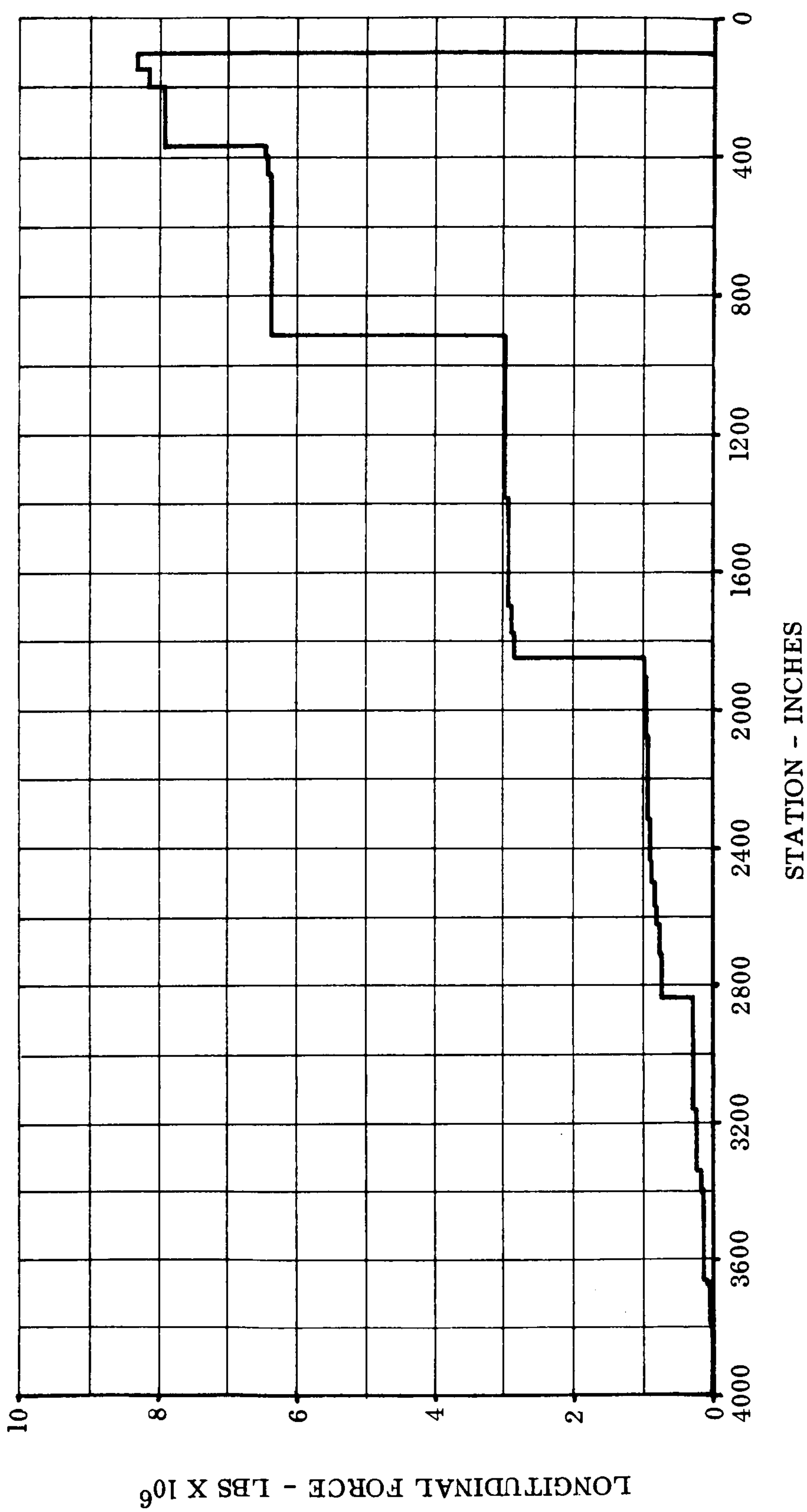
SATURN V AS-511
LONGITUDINAL FORCE VS. STATION
CENTER ENGINE CUTOFF CONDITION



R-P&VE-SLL

FIGURE 19 LONGITUDINAL FORCE DISTRIBUTION AT MAXIMUM g CONDITION

SATURN V AS-511
LONGITUDINAL FORCE VS. STATION
MAXIMUM (Qα) CONDITION
TIME = 75 SEC.



R-P&VE-SLL

FIGURE 20 LONGITUDINAL FORCE DISTRIBUTION AT MAXIMUM (Qα) CONDITION

SECTION III

BASELINE STAGE DESCRIPTIONS

1.0 S-IC STAGE

1.1 CONFIGURATION DESCRIPTION

The S-IC stage consists of two cylindrical propellant tanks separated by an inter-tank structural assembly, a thrust structure and four aerodynamic fins. This stage is powered by five F-1 engines. The thrust structure transmits stage thrust loads to a 396-inch diameter shroud which supports both of the cylindrical propellant tanks. The fins are mounted to the engine skirts which protect each of the four engines from excessive aerodynamic loading. The fins provide vehicle stability during flight. Four vehicle holddown points are located angularly between the outer engines.

1.2 STRUCTURE

1.2.1 Thrust Structure Assembly

The thrust structure distributes the thrust loads from the F-1 engines to the fuel tank during flight and provides holddown points for the vehicle during tests and launch. The thrust structure is made of aluminum and composed of four holddown posts, four engine thrust posts, a cross beam structure between holddown posts to support the center F-1 engine, stiffened skin, and frames.

1.2.2 Tail Assembly Shroud and Fairing

The tail assembly shroud continuously surrounds the base of the S-IC stage except at locations of the engine skirts. The shroud and engine skirts are composed of an aluminum skin with external longitudinal stiffeners and internal lateral stiffeners. The longitudinal shroud stiffeners are channel sections which are chamferred at both ends.

1.2.3 Fins

Each of the four fins are attached to aerodynamic fairings for the engines and provide vehicle stability during flight.

1.2.4 Propellant Containers

Both of the cylindrical propellant tanks are made of 2219-T87 aluminum tanks. Ellipsoidal bulkheads are used at the forward and aft end of both tanks. The cylindrical skin sections of the fuel tank have internal integral milled stringers.

1.2.5 Intertank Structure

The intertank section provides structural continuity between the oxidizer and fuel tanks and is composed of corrugated aluminum skin panels stiffened by I-rings.

1.2.6 Forward Skirt Assembly

The aft end of the forward skirt is attached to the LOX tank and the forward end interfaces with the S-II stage. The skin panels, fabricated from 7075-T6 aluminum, are stiffened and strengthened by ring frames and stringers.

1.3 PROPULSION SYSTEMS

1.3.1 F-1 Engine

This engine is a single-start, fixed-thrust, gimbaled bipropellant system which uses liquid oxygen as the oxidizer and RP-1 as the fuel. The fuel also serves as the turbopump lubricant and the control actuator hydraulic fluid. Basic engine components include a regeneratively cooled tubular-walled thrust chamber with a turbine exhaust gas cooled extension skirt, a gas generator with a dual gas heat exchanger, a direct-drive turbopump, an engine gimbal system, and an engine control system.

Each outboard engine is attached to the thrust structure by a gimbal assembly. The center engine is mounted to the cross beams of the stage thrust structure and cannot be gimbaled. Outboard engines gimbal in a five-degree square pattern for vehicle thrust vector control.

1.3.2 Propellant Container Pressurization

Before engine ignition, the LOX and RP-1 tanks are prepressurized with helium from launch support equipment. During flight, the RP-1 tank is pressurized with heated helium supplied from on-board storage cylinders while the LOX is pressurized by GOX obtained by heating LOX bled from the LOX turbopump outlet. Both the LOX and RP-1 pressurization systems use the F-1 engine heat exchangers as the heat source.

1.4 Thermal Protection

The base heat shield consists of numerous honeycomb panels with a hot side layer of M-31 thermal barrier trowelled into the open face honeycomb.

2.0 S-II STAGE

2.1 CONFIGURATION DESCRIPTION

This stage consists of a cylindrical liquid hydrogen tank located forward of the liquid oxygen tank. Both tanks are separated by a common insulated bulkhead. Principal structural components include the forward skirt, the forward bulkhead, the liquid hydrogen tank walls, the common intertank bulkhead, the intermediate skirt, the thrust structure and the aft skirt and interstage.

2.2 STRUCTURE

2.2.1 Aft Skirt and Interstage

This structure transmits first stage thrust loads to the S-II stage and is built of 7075 aluminum alloy material. The structure is stiffened by external hat-section stringers and stabilized internally by circumferential ring frames.

2.2.2 Thrust Structure

An inverted conical frustrum thrust structure transfers S-II stage engine thrust loads to the body shell. This aluminum structure is stiffened by circumferential ring frames and hat-section stringers. Four pairs of thrust longerons and a center engine support beam cruciform assembly accept and distribute the thrust loads of the J-2 engines.

2.2.3 Propellant Containers

The LOX tank is composed of the common bulkhead and an ellipsoidal aft bulkhead which are welded together. The common bulkhead is an ellipsoidal aluminum sandwich structure with a fiberglas/phenolic core. The LH_2 tank has an aluminum cylindrical section which is stiffened by internal rings. The cylindrical section separates the elliptically-shaped bulkhead from the common bulkhead. A layer of foam-filled honeycomb insulation is attached externally to the LH_2 tank to prevent excessive boil-off.

2.2.4 Forward Skirt

This structure transmits lower stage thrust loads to the S-IVB stage and is similar to the aft skirt and interstage.

2.3 Propulsion Systems

2.3.1 J-2 Engine

Primary propulsion for the S-II stage is provided by five J-2 rocket engines which are high-performance engines utilizing liquid oxygen and liquid hydrogen as propellants. The only fluids used in the engine are the propellants and helium gas. The extremely low operating temperature of the engine prevents the use of lubricants or other fluids. Programmed mixture ratio shift has been utilized for the S-II stage.

Propellant utilization is accomplished by passing liquid oxygen from the discharge side of the oxidizer turbopump to the inlet side through a valve driven by a servomotor.

The J-2 features a single tubular-wall, bell-shaped thrust chamber, and two independently driven, direct-driven turbopumps for liquid oxygen and liquid hydrogen. Both turbopumps are powered in series by a single gas generator, which uses the same propellants as the thrust chamber. An electrical control system which contains solid-state logic elements is used to sequence the start and shutdown operations of the engine.

The four outer J-2 engines are mounted to the S-II stage thrust structure aft section. The other engine is attached to cross beams in the thrust structure. The outer four engines are capable of being gimbaled $\pm 7^{\circ}$ square pattern to provide thrust vector control.

2.3.2 Propellant Container Pressurization

Prepressurization of the LOX and LH_2 tanks prior to launch is accomplished with helium supplied from launch support equipment. Inflight pressurization for the LOX tank is accomplished with GOX obtained by heating LOX bled from the LOX turbopump outlet. The LOX is converted to GOX by a heat exchanger located in the engine turbine exhaust duct. Inflight pressurization for the LH_2 tank is accomplished by gaseous hydrogen bled from the thrust chamber hydrogen injector manifold of each of the four outer J-2 engines.

3.0 S-IVB STAGE

3.1 CONFIGURATION DESCRIPTION

The S-IVB stage is basically a 260-inch diameter semimonocoque structure consisting of dual propellant tanks separated by a common bulkhead, a thrust structure assembly, a forward skirt assembly, an aft skirt, and an S-II/S-IVB interstage. The LOX tank is located aft of the LH_2 tank.

3.2 STRUCTURE

3.2.1 Forward Skirt

The forward skirt is an aluminum alloy, cylindrical skin and stringer structure. It houses the various antennae, the forward umbilical plate, panels with telemetry and electrical equipment, and joins the S-IVB to the Instrument Unit.

3.2.2 Liquid Hydrogen Tank

The fuel tank consists of a cylindrical section 268.5 inches long and 260 inches in diameter with hemispherical shaped bulkheads. It is fabricated of an aluminum alloy except for the aft bulkhead (common bulkhead) which is made of aluminum sheets and fiberglas-phenolic honeycomb core which serves as an insulator between the LOX and LH_2 tanks. The tank cylinder is formed by longitudinally welded aluminum sheets. The internal surface is machine milled in a waffle pattern to obtain tank stiffness with minimum weight.

Polyurethane insulation blocks, covered with a fiberglas sheet and coated with a sealant, are bonded into the waffle patterns to minimize LH_2 boil-off.

3.2.3 Liquid Oxygen Tank

The oxidizer tank is formed by the intersection of the aft LH_2 tank bulkhead (common bulkhead) with the aft LOX bulkhead. The radius of both bulkheads is 130 inches but each intersects the other so that there is no cylindrical element in the LOX tank.

3.2.4 Thrust Structure

The thrust structure consists of a truncated cone made of an aluminum alloy, skin and stringer type consturction. It provides the attach point for the J-2 engine and distributes the engine thrust over the entire tank circumference. It is attached to the LOX tank bulkhead by means of a bolted joint.

3.2.5 Aft Skirt

The cylindrical shaped aft skirt is the load bearing structure between the LH_2 tank and aft interstage. It is made of aluminum alloy skin and stringer panels mechanically joined. The skirt contains the aft umbilical plate and houses panels with telemetry and electrical equipment. The skirt also supports the auxiliary propulsion system.

3.2.6 Aft Interstage

The aft interstage is a truncated cone that provides the load supporting structure between the S-IVB stage and the S-II stage. Interstage diameter varies from 396 inches to 260 inches. It is formed from aluminum alloy skin and stringer constructed panels. The aft interstage contains the S-II retrorocket motors and separates with the S-II stage.

3.3 PROPULSION SYSTEMS

3.3.1 J-2 Engine

The J-2 engine used in the S-IVB stage is virtually identical to the engines which provide thrust to the S-II stage with the exception of an added restart capability. The engine has the capability of being gimbaled $\pm 7^{o}$ (square pattern) to provide thrust vector control in pitch and yaw planes.

A rechargeable, engine-mounted, hydrogen start tank and a spark ignition system featuring automatic reset provide the basic elements for the multiple restart capability of the engine.

3.3.2 Auxiliary Propulsion System

The S-IVB auxiliary propulsion system provides stage three axis attitude control and stage propellant control during coast flight. The system is made up of two modules located 180^{o} apart on the aft skirt. Engines for the attitude control system use storable hypergolic propellants (MMH and N_2O_4). This system provides for pitch and yaw control during unpowered flight and roll control during powered and unpowered flight. The engines are nongimbaled and are pressure-fed.

The ullage thrust engines are used to maintain a settled propellant condition during times of LH_2 non-propulsive venting. These engines share the propellant and pressurization systems of the attitude control engines.

3.3.3 Propellant Container Pressurization

Prepressurization of the LOX and LH_2 tanks prior to launch is accomplished with helium supplied from launch support equipment.

Inflight pressurization for the LOX tank is accomplished with heated helium. The helium is stored in cold helium spheres located in the LH_2 tank and is heated by the engine heat exchanger located in the turbopump turbine exhaust duct. Inflight pressurization for the LH_2 tank is accomplished by gaseous hydrogen bled from the J-2 thrust chamber hydrogen injector manifold.

3.3.3 (continued)

Repressurization for the LOX and LH_2 tanks is also accomplished with heated helium, but in this case, it is heated by a heat exchanger located in a hydrogen/oxygen burner ("helium heater"). In the event of a failure of the burner, ambient helium is provided by a secondary system.

4.0 INSTRUMENT UNIT

4.1 CONFIGURATION DESCRIPTION

The launch vehicle guidance, navigation, and control equipment, and most of the instrumentation is packaged in the Instrument Unit which is located immediately forward of the S-IVB stage. The Instrument Unit forms a part of the vehicle load-bearing structure with interface to the S-IVB stage and spacecraft.

4.2 STRUCTURE

The basic IU structure is a short cylinder fabricated of an aluminum alloy honeycomb sandwich material. The structure is fabricated from three honeycomb sandwich segments of equal length.

Attached to the inner surface of the cylinder are cold plates which serve both as mounting structure and thermal conditioning units for the electrical/electronic equipment.