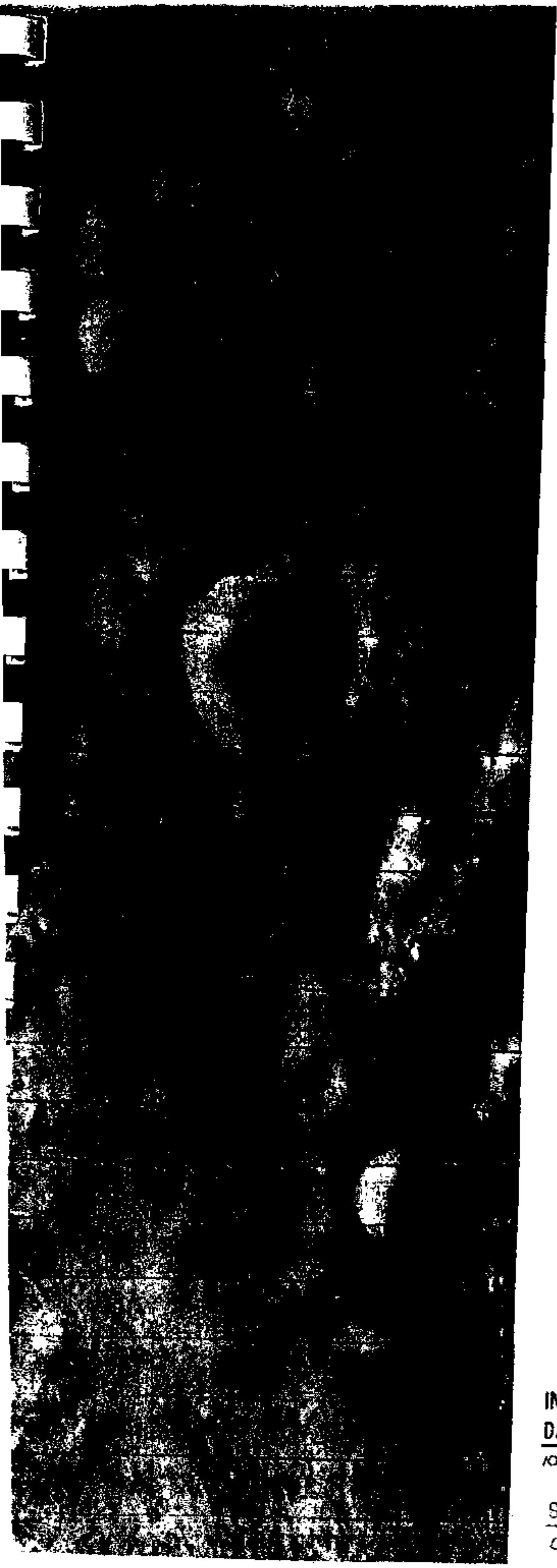

D5-13183

Final Report - Studies of Improved Saturn V Vehicles and Intermediate Payload Vehicles (P-115)

SUMMARY

Prepared for

NASA - George C. Marshall Space Flight Center under Contract NAS8-20266

October 7, 1966

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THE BOEING COMPANY — SPACE DIVISION

STUDIES OF IMPROVED SATURN V VEHICLES AND INTERMEDIATE PAYLOAD SATURN VEHICLES (P-115)

SUMMARY DOCUMENT
D5-13183

FINAL REPORT
PREPARED UNDER CONTRACT NUMBER NAS8-20266

SUBMITTED TO
GEORGE C. MARSHALL SPACE FLIGHT CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
OCTOBER 7, 1966

SUBMITTED BY
SYSTEMS ANALYSIS CONTRACTOR
THE BOEING COMPANY
SPACE DIVISION
LAUNCH SYSTEMS BRANCH
HUNTSVILLE, ALABAMA

FOREWORD

This volume summarizes five technical Vehicle Description Documents reporting a ten-month study to prepare technical and resource data on uprated payload Saturn V and intermediate payload Saturn vehicles. This study is part of a continuing effort by the National Aeronautics and Space Administration (NASA) to investigate the capability and flexibility of the Saturn V launch vehicle and to identify practical methods for diversified utilization of its payload capability. NASA Contract NAS8-20266 authorizes the work reported herein and was supervised and administered by the Marshall Space Flight Center (MSFC). S-II data was supplied by the Space and Information Division of North American Aviation. S-IVB data was supplied by the Missile and Space Systems Division of Douglas Aircraft Company. Launch system data was supplied by the Denver Division of The Martin Company. Solid motor data were supplied by United Technology Corporation. The Launch Systems Branch, Aerospace Group, Space Division of The Boeing Company was the Systems Analysis contractor for this study.

Program documentation includes a summary volume (this document), five volumes covering vehicle descriptions, research and technology implications report, and a cost document. Individual designations are as follows:

D5-13183	Summary Document
D5-13183-1	Vehicle Description MLV-SAT-INT-20, -21
D5-13183-2	Vehicle Description MLV-SAT-V-3B
D5-13183-3	Vehicle Description MLV-SAT-V-25(S)
D5-13183-4	Vehicle Description MLV-SAT-V-4(S)B
D5-13183-5	Vehicle Description MLV-SAT-V-23(L)
D5-13183-6	Research and Technology Implications Report
D5-13183-7	First Stage Cost Plan

ABSTRACT

This document summarizes a study conducted under NASA/MSFC Contract NAS8-20266, "Studies of Improved Saturn V Vehicles and Intermediate Payload Saturn Vehicles (P-115)", from December 6, 1965 to October 7, 1966. The details of this study are contained in five "Vehicle Description Documents" (D5-13183-1, -2, -3, -4, and -5). Phase I of the study was a parametric performance and resources analysis to select one baseline configuration for each of the six vehicles. Phase II of the study included a fluid and flight mechanics study, design impact on systems, and a resources analysis for each baseline vehicle. The uprated vehicles are feasible configurations and logical candidates for payloads in excess of the current Saturn V capability. No major problem areas were identified for either development or production. The intermediate payload vehicle derivatives of Saturn V are a logical means of providing orbital payload capability between that of the Saturn IB and the two-stage Saturn V.

KEY WORDS

Contract NAS8-20266
D5-13183
Vehicle Description Document
Saturn V
NASA/MSFC
Uprating
Trade Studies
Payload to 72 Hour Lunar Injection
Fluid and Flight Mechanics
Impact
Resources
Cost
Payload to 100 NM orbit
MLV-INT
Baseline Configuration
MLV-SAT-V

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1.0 INTRODUCTION

This study is part of a continuing effort by NASA to identify a spectrum of practical launch vehicles to meet future payload and mission requirements as they become defined. The launch vehicles studied under Contract NAS8-20266 cover a payload range between the existing Saturn IB and the Saturn V (intermediate payload vehicles) and a payload range beyond the existing Saturn V capabilities (uprated Saturn V vehicles).

The vehicles studied were combinations of existing or modified Saturn V stages; some vehicles also included boost-assist components. A primary study requirement was to make maximum use of existing Saturn technology and support equipment.

In general, the NAS8-20266 study program objectives were to:

a. Select feasible and cost effective baseline vehicles from each of several categories.

b. Prepare sufficient technical data to define vehicle environments, design, capabilities, and characteristics.

c. Define support system requirements.

d. Determine the date that the first flight article could be available within study ground rules.

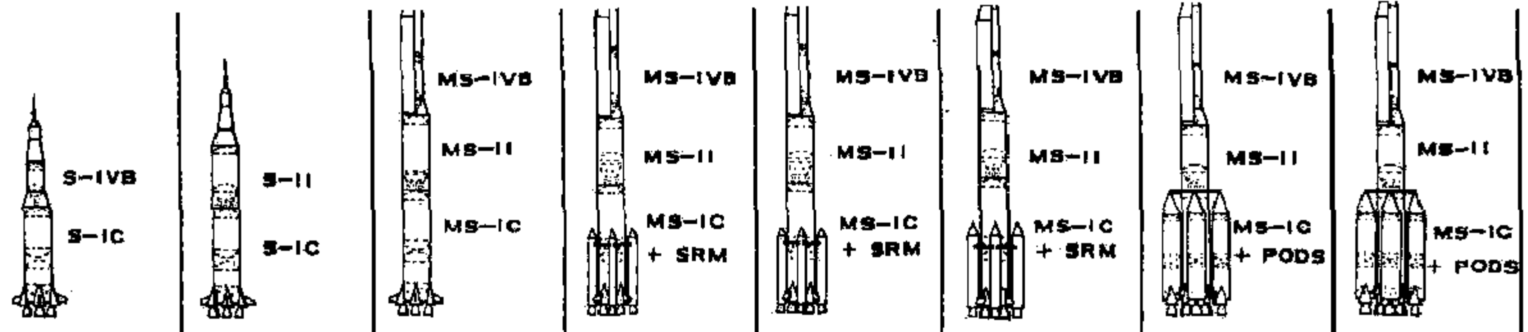
e. Estimate cost required for implementation of the system plus production of thirty flight articles in five years.

There were two phases of study work. Phase I was a twelve-week effort in which candidate vehicle performance and preliminary cost trade studies were conducted to select a feasible and cost effective baseline vehicle from each of five categories (shown in Figure 1-1). An additional baseline vehicle was later added from Category 4.

For each of the six baseline vehicles selected (see Figure 1-2), Phase II directed the effort to defining ground and flight environments, defining system design and resource impact for each stage and the total vehicle, and determining vehicle mission capabilities and characteristics.

The launch vehicles in Categories 1 and 2 are Saturn V stage combinations for missions in the payload range between the current Saturn IB and Saturn V payload capability. The launch vehicles in

SIZE-410 FT
MAX MS-II
SIZE 1160 IN
MAX MS-IVB
SIZE 350K
PROPELLANT
AT MR 5:1
MAX PAYLOAD
5 LB/FT^3
(2 STG)
11 LB/FT^3
(3 STG)
STD: STANDARD
ΔL: CHANGE IN
STAGE LENGTH



LAUNCH VEHICLE	INT-20	INT-21	SAT-V-3B	SAT-V-4(S)B	SAT-V-22(S)	SAT-V-25(S)	SAT-V-23(L)	SAT-V-24(L)
CATEGORY	1	2	3	4			5	
THIRD STAGE			ADVANCED ENGINE ΔL VARIABLE	STD J-2 ΔL VARIABLE	ADVANCED ENGINE ΔL VARIABLE	STD J-2 ΔL VARIABLE	STD J-2 ΔL VARIABLE	ADVANCED ENGINE ΔL VARIABLE
SECOND STAGE	STD J-2's ΔL - 0	STD J-2's ΔL - 0	ADVANCED ENGINES ΔL VARIABLE	STD J-2's ΔL VARIABLE	ADVANCED ENGINES ΔL VARIABLE	STD J-2's ΔL VARIABLE	STD J-2's ΔL VARIABLE	ADVANCED ENGINES ΔL VARIABLE
FIRST STAGE	STD F-1's ΔL - 0	STD F-1's ΔL - 0	5 X 1.8M F-1 ENGINES ΔL VARIABLE	STD F-1's ΔL VARIABLE	STD F-1's ΔL VARIABLE	STD F-1's ΔL VARIABLE	STD F-1's ΔL VARIABLE	5 X 1.8M F-1 ENGINES ΔL VARIABLE
STRAP-ON COMPONENTS				4 X 120 IN DIA SOLID MOTORS	4 X 120 IN DIA SOLID MOTORS	4 X 156 IN DIA SOLID MOTORS	4 LIQUID PODS 2 X STD F-1 ENGINES	4 LIQUID PODS 2 X 1.8M F-1 ENGINES

FIGURE 1-1 PHASE I LAUNCH VEHICLE CANDIDATES

Categories 3, 4, and 5 are advanced Saturn V configurations with payload capabilities beyond that of the existing Saturn V.

The five categories of vehicles are:

Category 1 (MLV-SAT-INT-20) during Phase I was a family of two-stage launch vehicle candidates with standard size S-IC and S-IVB stages using standard F-1 engines (three, four, and five) and a standard J-2 engine. A single baseline launch vehicle (Figure 1-2) was selected for the Phase II study effort.

Category 2 (MLV-SAT-INT-21) during Phase I was a family of two-stage launch vehicle candidates with standard size S-IC and S-II stages using standard F-1 engines (three, four, and five) and J-2 engines (three, four, and five). A single baseline launch vehicle (Figure 1-2) was selected for the Phase II study effort.

Category 3 (MLV-SAT-V-3B) during Phase I was a family of two- and three-stage launch vehicle candidates with modified uprated Saturn V stages using various types, numbers, and thrust levels of advanced engines in the upper stages and uprated F-1 engines in the modified S-IC stage. A single baseline launch vehicle (Figure 1-2) was selected for the Phase II study effort.

Category 4 included modified Saturn V launch vehicles with strap-on solid boost-assist components. Three families of vehicles were studied as follows:

a. MLV-SAT-V-4(S)B during Phase I was a family of two- and three-stage launch vehicles with modified Saturn V stages, standard F-1 and J-2 engines with strap-on 120-inch diameter (five, six, and seven segment) solid motors. A single baseline launch vehicle (Figure 1-2) was selected for the Phase II study effort.

b. MLV-SAT-V-22(S) during Phase I was a family of two- and three-stage launch vehicles with modified Saturn V stages using various types, numbers, and thrust levels of advanced engines in the upper stages, a modified S-IC stage with standard F-1 engines in the first stage, and strap-on 120-inch diameter (five, six, and seven segment) solid motors. No launch vehicle in this family was studied beyond Phase I.

c. MLV-SAT-V-25(S) during Phase I was a family of two- and three-stage launch vehicles with modified Saturn V stages, standard F-1 and J-2 engines, and strap-on 156-inch diameter (two and three segment) solid motors. A single baseline launch vehicle (Figure 1-2) was selected for the Phase II study effort.

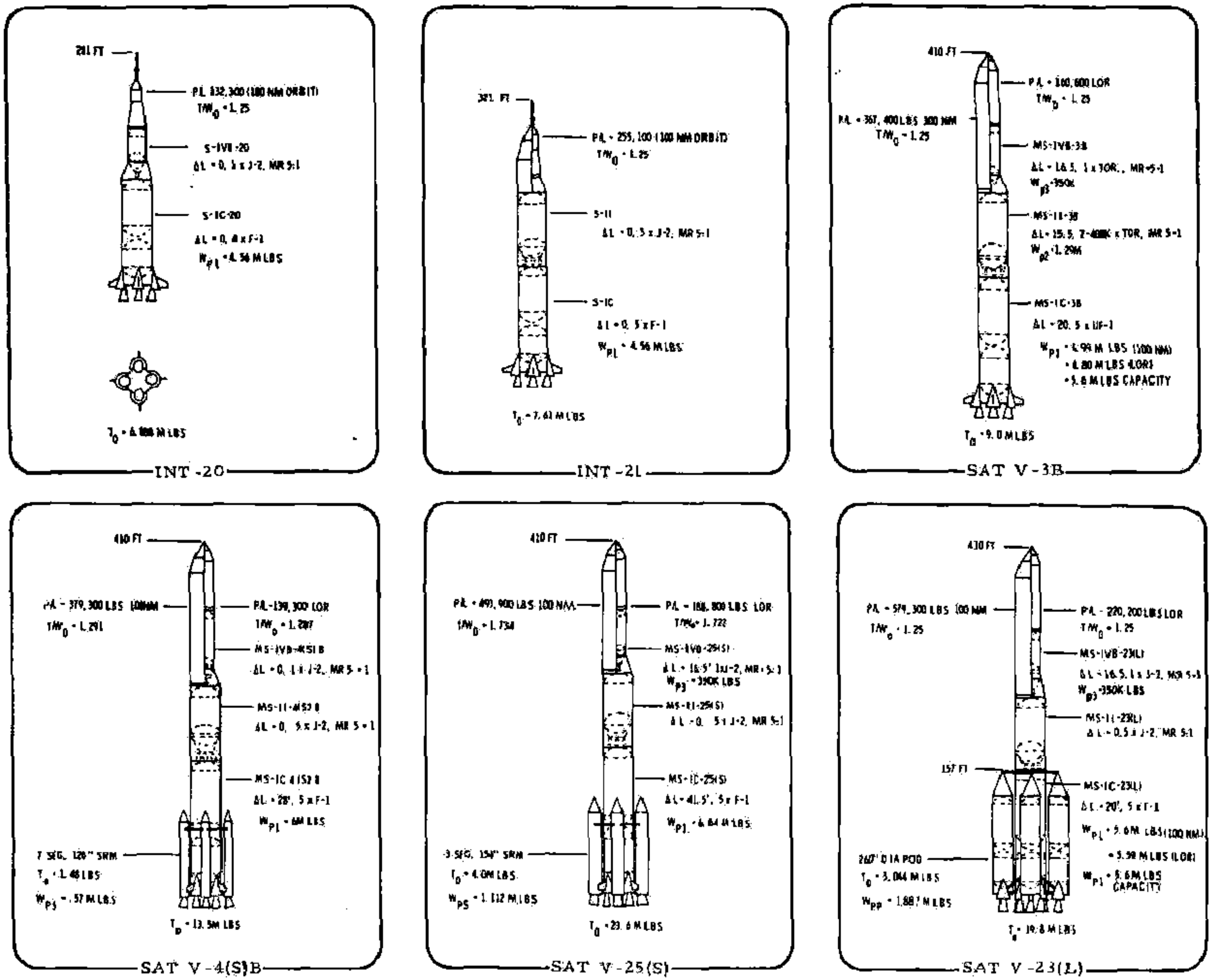


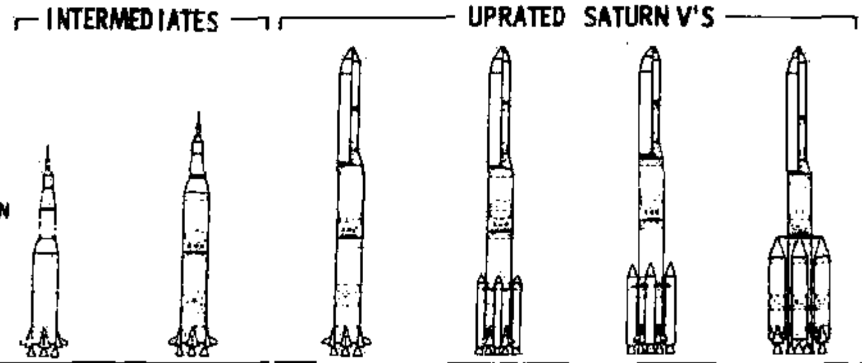
FIGURE 1-2 SELECTED BASELINE LAUNCH VEHICLES FOR PHASE II STUDY EFFORT

Category 5 included modified Saturn V launch vehicles with strap-on boost-assist liquid propellant pods. Two families of vehicles were studied as follows:

a. MLV-SAT-V-23(L) during Phase I was a family of two- and three-stage launch vehicles with modified Saturn V stages, standard F-1 and J-2 engines and four strap-on liquid propellant pods each using two standard F-1 engines. A single baseline launch vehicle (Figure 1-2) was selected for the Phase II study effort.

b. MLV-SAT-V-24(L) during Phase I was a family of two- and three-stage launch vehicles with modified Saturn V stages using various types, numbers, and thrust levels of advanced engines in the upper stages, a modified S-IC stage with 1,800,000 pound F-1 engines, and four liquid propellant pods each containing two 1,800,000 pound F-1 engines. No launch vehicles in this family were studied beyond Phase I.

2.0 SUMMARY



* $176.7M 12/YR. PRODUCTION
45M 8/YR. PRODUCTION
14.6M 6/YR. PRODUCTION
SINGLE COST FOR SYMULTANEOUS IMPLEMENTATION OF ALL 8-20/-21 ENGINE/STAGE COMBINATIONS

	INT-20 (2 STG) F-1	INT-21 (2 STG) F-1/J-2	SAT V-3B 2 STG	SAT V-3B 3 STG	SAT V-4(S)B 2 STG	SAT V-4(S)B 3 STG	SAT V-25(S) 2 STG	SAT V-25(S) 3 STG	SAT V-23(L) 2 STG	SAT V-23(L) 3 STG
PAYLOAD - KLB	3 4 78 132	43 44 45 53 54 55 161 186 196 222 246 255	367	161	380	139	494	189	579	220
DDT&E - $M	*	*		1097.6		431.6		603.1		944.9
R&D FLIGHT VEHICLES - $M	(1) 60.8	0		325.6		279.1		324.2		413.0
AVG. OPERATIONAL UNIT COST - $M	60.5	74.6	116.7	139.0	105.1	124.8	109.6	131.0	142.5	163.8
OPERATIONAL COST EFFICIENCY - $/LB	458	292	318	864	276	896	222	694	246	745
FIRST DELIVERY (ATP - JAN 68)	AS-516 FEB '70	AS-516 FEB '70	AS-537 SEPT '73		AS-524 JUNE '71		AS-524 JULY '71		AS-535 MAY '73	

FIGURE 2-1 VEHICLE COMPARISON

The Phase I study effort resulted in selection of six baseline launch vehicles. The Phase II study effort included detailed technical and resource analysis on these six baseline launch vehicles. Payload capabilities, costs, and availability data are compared on Figure 2-1. Operational costs shown are the averages for thirty launch vehicles. It should be noted that the total of $176.7 million for the SAT-INT-20 and SAT-INT-21 is proposed as a single R&D expenditure to implement all eight stage/engine combinations listed. This would allow NASA the flexibility of selecting the vehicle matching each of many different payloads expected in the range between present Saturn IB and Saturn V capabilities. The data required is very sensitive to launch rate as indicated by the reductions noted for eight per year and six per year launches.

Figure 2-2 illustrates the delta payload increase (from Saturn V) and compares the investment costs for developing the uprated Saturn V launch vehicles. The more favorable vehicles from an investment standpoint fall to the left, i.e., least cost for a given payload improvement. Figure 2-3 summarizes the total program cost efficiency for the six two-stage baseline launch vehicles and Figure 2-4 summarizes cost efficiency for the three-stage uprated Saturn V launch vehicles. All of these comparisons favor the solid strap-on method of uprating, usually by a relatively small margin.

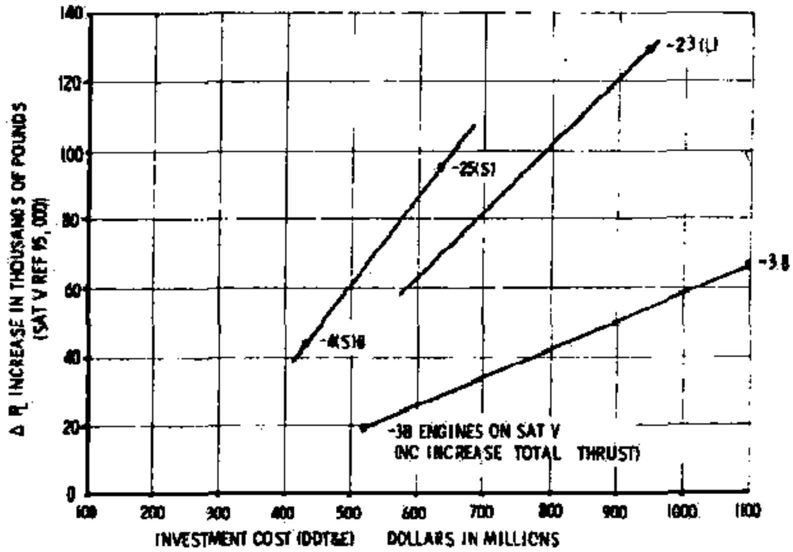


FIGURE 2-2 UPRATED VEHICLE INVESTMENT COST COMPARISON

Availability of the SAT-INT-20 and SAT-INT-21 intermediate vehicles exceeds the normal Saturn V procurement time by only one month. For uprating, the solid strap-on method requires the least lead time (3-1/2 years) which is comparable to the liquid pod strap-on (SAT-V-23(L)) method except a two-year delay has been included in the SAT-V-23(L) lead time to build an assembly facility. The five-year, nine-month lead time for the increased thrust liquid engine - larger tank uprating method (SAT-V-3B) is due to the new toroidal aerospike engine development for upper stage applications.

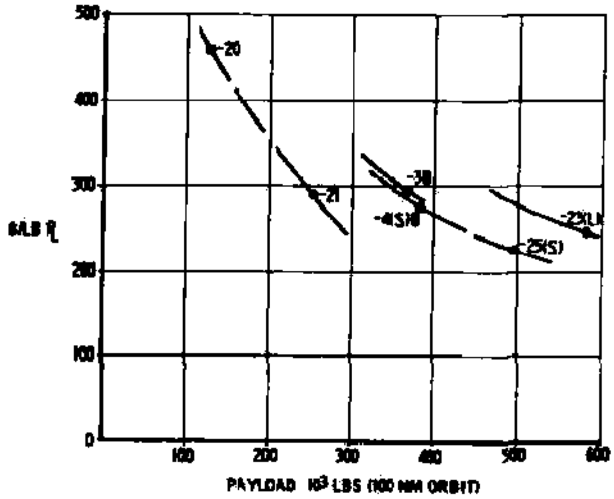


FIGURE 2-3 TWO-STAGE VEHICLE COST EFFICIENCY

All the baseline launch vehicles were feasible and logical configurations for their respective payload capabilities. Each was configured within restrictive existing facility limitation ground rules, limiting the maximum payload achieved to 579,000 pounds to 100 nautical mile Earth orbit (SAT-V-23(L)). The liquid pod strap-on concept, with uprated F-1s and advanced engines in the second stage (SAT-V-24(L)), achieved payloads to 960,000 pounds to 100 nautical mile Earth orbit when stage and total vehicle length restrictions were relaxed.

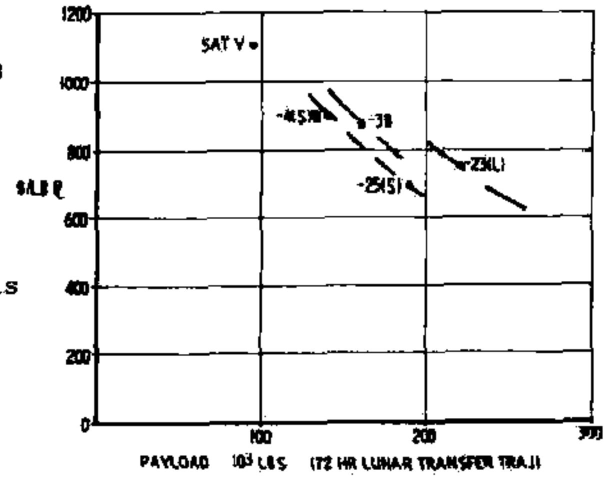


FIGURE 2-4 THREE-STAGE VEHICLE COST EFFICIENCY

3.0 GUIDE LINES AND ASSUMPTIONS

The following guide lines, grcund rules, and assumptions were used in the study.

3.1 GENERAL

a. Applicable data from previous studies were utilized to the greatest extent possible.

b. The baseline vehicles were the AS-516 and the AS-213 as defined by MSFC. Apollo design criteria was used except where otherwise specified or approved by MSFC. Memorandum R-P&VE-DIR-65-143, "Saturn V Improvement Studies", dated November 5, 1965, was issued by MSFC to serve as the reference document for MSFC and contractor personnel directly involved in the Saturn V Improvement Studies. Memo R-P&VE-DIR-65-143 contains a description and definition of the projected launch vehicle AS-516 to be used as the baseline reference for the Saturn V Improvement Studies. One minor deviation to the AS-516 S-IC stage definition was made with MSFC concurrence. The redesign of the center engine crossbeam support was eliminated as a basic change because of a lack of definitive design data.

c. All propulsion data used by the stage contractors were approved by MSFC.

d. Both launch vehicle and launch facility modifications were considered. Exchange of information between the launch facility and launch vehicle study contractors was coordinated with MSFC and KSC.

e. Trajectory, propellant distribution, and stage size optimization procedures used were comparable to MSFC methods.

f. The nominal mission profiles used to size and establish the baseline vehicle design, to establish trajectories for heating and control analysis, and as a basis for performance comparison were:

1. Two-stage, direct ascent to 100 nautical mile circular orbit altitude.

2. Three-stage, with pre-orbital ignition of the third stage to 100 nautical mile circular parking orbit followed by a second burn out of orbit to 72-hour lunar injection. This is the planned Saturn V method.

Some vehicles used two-stage, direct ascent to a 100 nautical mile circular parking orbit followed by ignition of a third stage and boost to a 72-hour lunar transfer trajectory.

g. Launch azimuth from AMR was 70 degrees measured from north to south over east and flight profiles were optimized in the pitch plane.

h. Vehicle height, for both two- and three-stage vehicles, was limited to 410 feet.

i. Payload density was held at five pounds per cubic foot maximum for two stage operation and 11 pounds per cubic foot maximum for the three-stage vehicle.

3.2 FIRST STAGE

Thirty-three foot diameter and 2.29 propellant mixture ratio of the existing S-IC stage were to be maintained.

3.3 SECOND STAGE

a. Propellant mixture ratio of 5:1 and 33-foot diameter were to be maintained.

b. Maximum stage length for baseline selection was limited to 1,160 inches.

3.4 THIRD STAGE

a. Propellant mixture ratio of 5:1 and 260-inch diameter were to be maintained.

b. Maximum stage length equivalent to 350,000 pounds propellant capacity at a mixture ratio of 5:1 (about 16.5 foot increase) was to be maintained.

3.5 RESOURCES

a. Where two- and three-stage configurations of the same basic vehicle were exercised, the three-stage configuration was analyzed.

b. Uprated Saturn V stages were to be fabricated by the present contractors and cost data for the stages were obtained from the contractors.

c. The impact of study vehicles on test facilities at MSFC, test facilities at MTF, and launch facilities at KSC were considered with the assistance of those agencies or their designated contractors.

d. Two flight tests were specified to qualify uprated vehicles.

e. A production program of thirty operational uprated vehicles to be produced in five years was specified.

f. Uprated vehicles will be considered to be produced at six per year with Saturn IB a companion program at six per year.

g. Intermediate payload vehicles will be considered to be produced at six per year with Saturn V a companion program at six per year.

h. A dynamic test vehicle was required.

3.6 SCHEDULE

A program schedule was required subject to the following restrictions:

a. The uprated vehicle development program was to be parallel with the existing Saturn V program and not interfere with the existing Saturn V delivery schedule.

b. Vehicle development time to be a minimum, consistent with completion of a thorough test program.

c. A program definition phase (PDP) was required prior to beginning uprating vehicle design and development. Earliest allowed PDP start was January 1967.

d. Earliest allowed authority to proceed for hardware design and development was Janaury 1968.

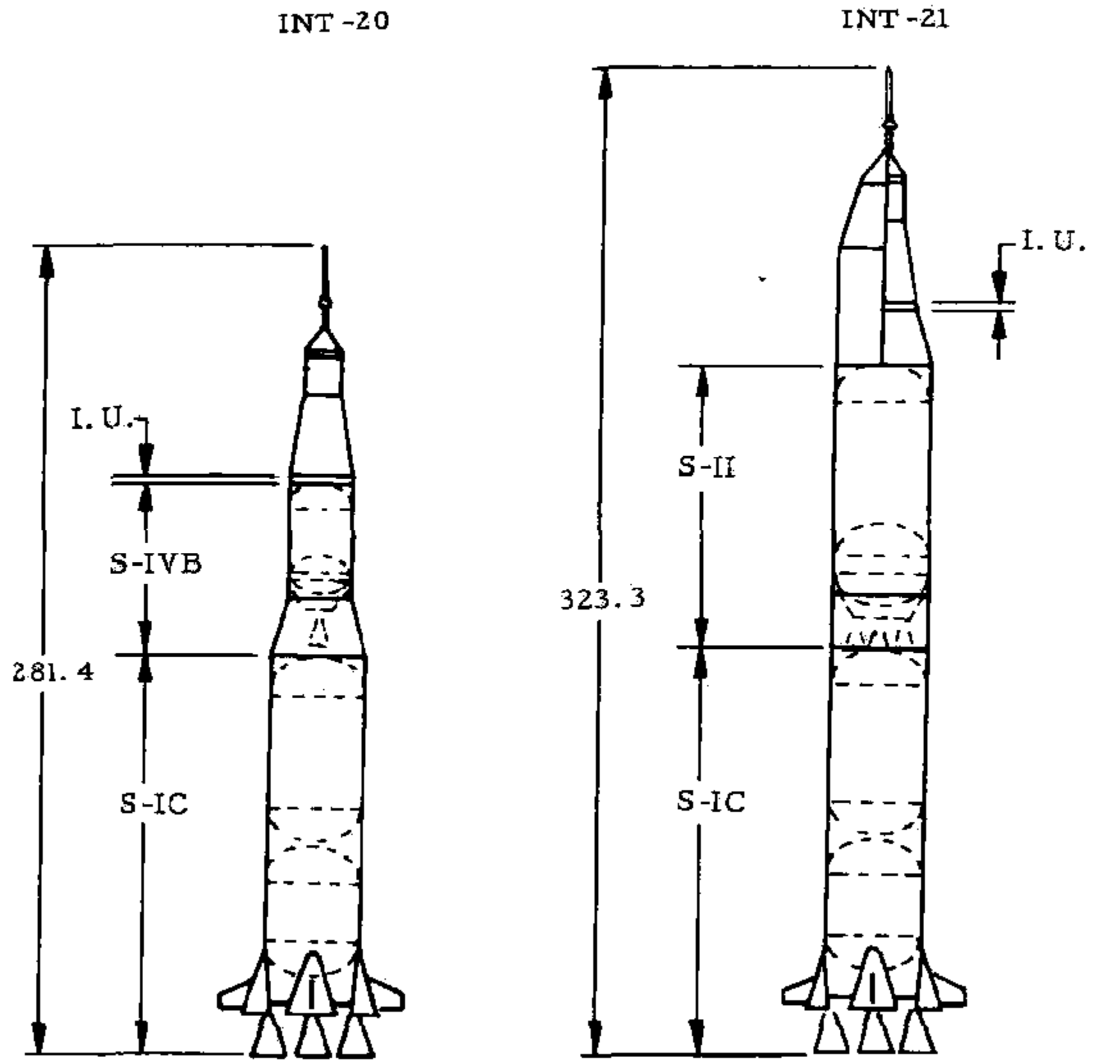
3.7 PRICING

It was also required in performing these resources analyses that the following pricing criteria be met:

a. Necessary funds are available as required.

b. All costs were quoted in 1966 dollars with no inflationary factor or mid-point estimate.

c. All costs were based on two-shift, five-day week for manufacturing and one-shift, five-day week for engineering.



J-2/F-1	100 N Mi Orbit P_L - 10^3 LBS
3/1	78
4/1	132

F-1/J-2	100 N Mi P_L - 10^3 LBS
4/3	167
4/4	186
4/5	196
5/3	222
5/4	246
5/5	255

FIGURE 4-1 INT VEHICLES

4.0 MLV-SAT-INT-20/-21 LAUNCH VEHICLES

The MLV-SAT-INT-20 is a combination of the Saturn V S-IC and S-IVB stages. The MLV-SAT-INT-21 combines the Saturn V S-IC and S-II stages. All arrangements (see Figure 4-1) were found to be feasible. Each of eight stage/engine configurations could be used efficiently to launch a payload in increments between 78,000 pounds and 255,000 pounds to a 100 nautical mile Earth orbit.

Since future requirements will likely vary over a wide range of payloads and configurations, all INT-20/-21 vehicles should be implemented simultaneously. This would allow NASA planners to select the vehicle matching a specific payload requirement.

If an uprated vehicle is chosen for development, there are similar logical intermediate derivatives to be considered.

4.1 CONFIGURATION SELECTION (PHASE I)

Combinations of the three Saturn V stages and numbers of engines were studied during Phase I to establish the most promising configurations for detailed investigation.

4.1.1 Candidate Configurations

Three configurations were studied for INT-20, each having an S-IVB with a three-, four-, or five-engine S-IC. INT-21 arrangements included a three-, four-, or five-engine S-II combined with a four- or five-engine S-IC. This resulted in six INT-21 vehicles.

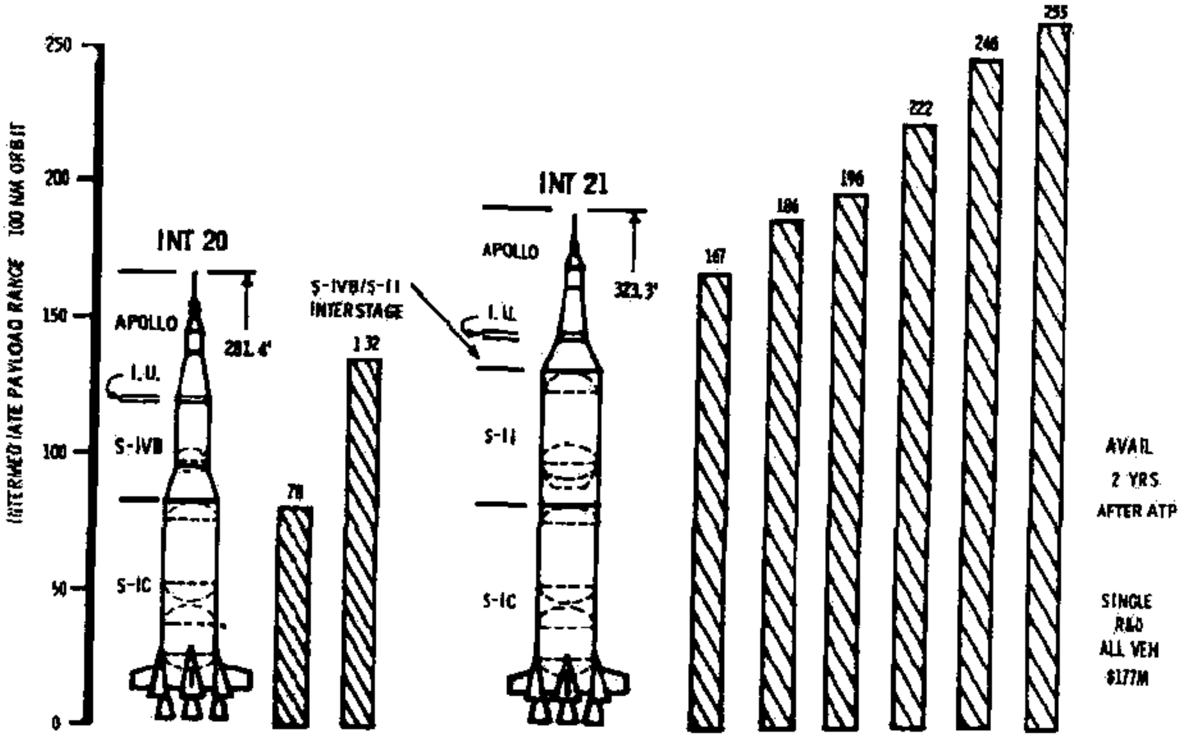
4.1.2 Trade Studies

Parametric data were generated for the candidate INT vehicles covering the following: (1) weight and mass characteristics, (2) trajectories and performance, (3) aerodynamics and heating, (4) vehicle control, (5) design loads, and (6) separation. A summary of INT-20 and INT-21 launch, propellant, and payload weights is shown in Table 4-I. The five-engine INT-20 vehicle, even though launched at a thrust-to-weight ratio of 1.25, depletes first-stage propellant rapidly. It therefore reaches a structural load limit at about 88 seconds after launch and three engines must be shut down. The resulting payload is not significantly better than the four-engine case (see Table 4-I) and therefore the five F-1 engine INT-20 was not considered further.

TABLE 4-I INTERMEDIATE VEHICLE PERFORMANCE SUMMARY

VEHICLE	STAGE ARRANGEMENT	NUMBER OF ENGINES	LAUNCH WEIGHT 10^6 LB	W_{P1} 10^6 LB	W_{P2} 10^6 LB	100 NM PAYLOAD 10^3 LB
SAT-INT-20	S-IC/S-IVB	3/1	3.65	3.0	0.23	78
		4/1	4.87	4.1	0.23	132
		5/1	5.07	4.3	0.23	133
SAT-INT-21	S-IC/S-II	4/3	4.87	3.56	0.71	167
		4/4	4.87	3.40	0.85	186
		4/5	4.87	3.30	0.93	196
		5/3	6.09	4.56	0.84	222
		5/4	6.09	4.47	0.91	246
		5/5	6.09	4.42	0.93	255

W_{P1} = First stage mainstage propellant
W_{P2} = Second stage mainstage propellant
Initial launch azimuth - 70 degrees



F1/J-2	3/1	4/1	4/3	4/4	4/5	5/3	5/4	5/5	SINGLE R&D
M$R&D 8/YR	42.3	42.3	37.8	37.8	35.8	35.5	35.5	33.5	NO PROD INC. 14.3M
M$R&D 12/YR	170.5	170.5	168.2	168.2	166.2	165.9	165.9	163.9	
$/LB	806	496	431	402	395	337	315	314	
S-IC BURNTIME	146	212	154	148	143	161	155	150	
	(3-146)	(2-146)	(2-146)	(2-146)	(4-143)	(1-146)	(1-146)	(1-146)	

FIGURE 4-2 INT-20/-21 VEHICLE COMPARISON

For the remaining baseline candidates (shown on Figure 4-2), it was necessary to shut down one or more F-1 engines during first stage flight to avoid the present 4.68 g longitudinal acceleration limit. The time of shutdown and number of engines shut down are shown with the first-stage burntime on Figure 4-2.

INT-20 and INT-21 vehicle costs were derived for six launches per year for five years. It was assumed that Saturn V was also launched at the same rate during this period. Bulk of the non-recurring cost (see Figure 4-2) is due to the increase in production and launch rates. Approximately 124 million dollars are required at KSC to build and equip for the new rate. The remainder covers mostly facilities, tools, and equipment. *Note the marked reduction in R&D cost for eight per year and* six per year INT production rates. The non-recurring cost for implementing all configurations simultaneously is estimated to be 13 million dollars (eight percent) more than the lowest-cost single arrangement at 12 per year.

Payload-cost-efficiencies (dollars per pound of payload, see Figure 4-2) vary uniformly, following the natural trend which is: smallest payload - largest cost per pound for delivery, and largest payload - least cost per pound for delivery.

Since all vehicles are shown to be cost efficient and NASA future requirements are more likely to be a number of different weight payloads, rather than one, it was recommended that all INT configurations be studied further. But, because funds, time, and manpower were limited, design analyses were restricted to the four-engine S-IC/S-IVB INT-20 and the five-engine S-IC/five-engine S-II INT-21. However, resources were prepared for all configurations.

4.2 DESIGN STUDY VEHICLE (PHASE II)

The baseline vehicles chosen during the Phase I activity were defined in detail, their complete characteristics determined, and their resource requirements established.

4.2.1 Vehicle Description

INT configurations chosen for further design study are shown on Figure 4-1. The INT-20 has a standard Saturn V S-IC first stage with the center F-1 engine and associated systems removed. The second stage is a standard S-IVB with its aft interface adapted to S-IC requirements. The INT-21 uses standard S-IC and S-II stages. An S-IVB/S-II interstage is used to adapt to the instrument unit and payload.

The manner in which a four-engine S-IC is achieved is illustrated on Figure 4-3. The baseline S-IC stage provided by NASA incorporates an insulated LOX duct rather than a duct-in-tunnel arrangement. With the duct removed, it is necessary to support the center duct spool to retain cross-feed capability. Cover plates and seals close the LOX and fuel bulkheads where lines are removed. Heat shield panels and supports from other locations replace those used where the engine is mounted. This installation can be made on any S-IC stage with insulated LOX ducts. Conversely, an S-IC INT-20 could readily be returned to the Saturn V configuration.

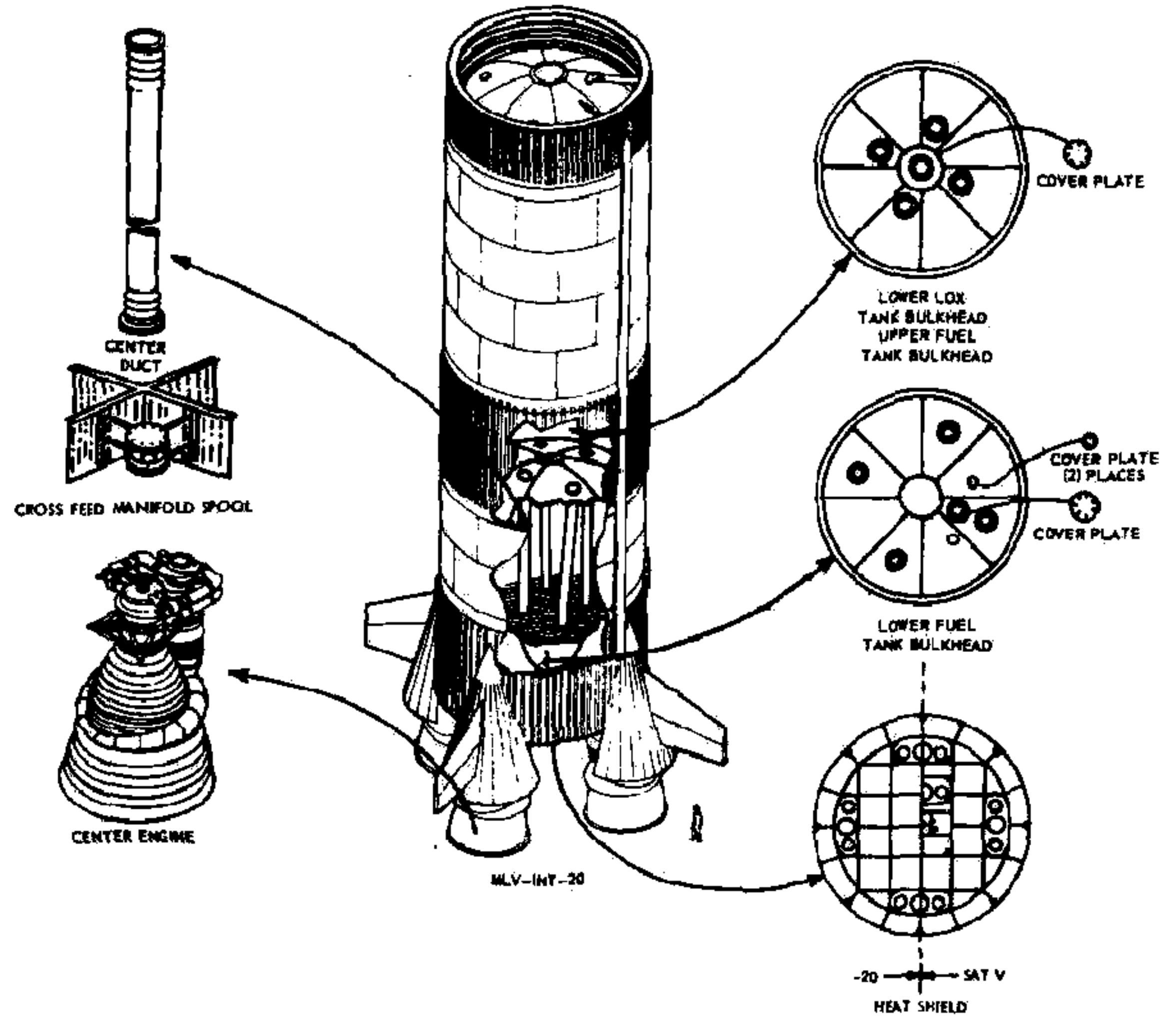


FIGURE 4-3 FOUR ENGINE S-IC-20 STAGE

4.2.2 Design Study Results

Significant load criteria and other data pertinent to vehicle design are shown on Table 4-II with comparative Saturn V values. As shown, load criteria is less than that for the existing Saturn V for both intermediate baseline configurations. Therefore, no structural modifications are required to the existing Saturn V stages.

Control requirements, as shown in Table 4-II, are below that of the existing Saturn V. Aerodynamic heating has increased slightly for both the INT-20 and INT-21 but is still within existing design criteria requiring no additional protection. Base heating on INT-20 is reduced compared with Saturn V and is identical to Saturn V on INT-21.

Detailed studies indicate that structural loads are less severe than for AS-516. No control or separation problems exist.

The reliability of the INT-20 is 0.999 and the INT-21 is the same as the baseline AS-516 reliability of 0.990.

Performance data were developed for the INTs for numerous missions. The nominal mission was direct ascent to a 100 nautical mile circular Earth orbit with a liftoff thrust-to-weight of 1.25 and a launch azimuth of 70 degrees. Alternate missions considered a range of orbit altitudes and launch azimuths.

	INT-20	INT-21	SAT V *
LOAD CRITERIA			
MAX q (LBS/FT^2)	758	760	766
g's AT MAX. q α	1.92	1.95	1.954
HEIGHT (FT)	281	323	363
CONTROL			
MODE	GIMBALED F-1'S	GIMBALED F-1'S	GIMBALED F-1'S
MAX. DEFLECTION ANGLE IN FLIGHT	2.3 DEG.	2.3 DEG.	3.5 DEG.
HEATING			
TYP AERODYNAMIC S-IC FWD SKT MAX. TEMP	197°F	176°F	167°F
BASE MAX. TEMP.	1720°F	1900°F	1900°F

* BASELINE 516 WITH T_0/W_0 = 1.25

TABLE 4-II SIGNIFICANT LOAD CRITERIA

Figure 4-4 summarizes the orbit/altitude capability for INT-20. Similar data are shown on Figure 4-5 for INT-21. Net payloads for the nominal mission are 132,000 pounds for INT-20 and 255,000 pounds for INT-21. Payloads for polar and sun synchronous orbits are shown on Figures 4-6 and 4-7. A boost turn is required to obtain these orbits from Cape Kennedy. This manuever requires energy expenditure which is reflected in less payload capability.

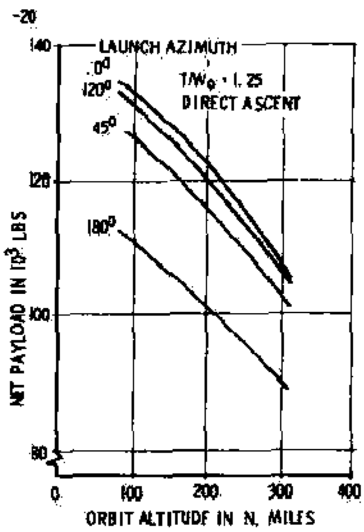


FIGURE 4-4 INT-20 ORBIT ALTITUDE - AZIMUTH PAYLOAD CAPABILITY

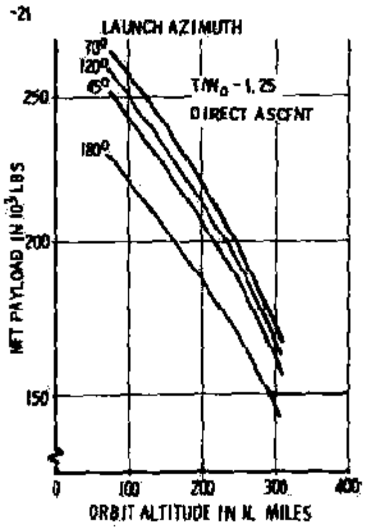


FIGURE 4-5 INT-21 ORBIT ALTITUDE AZIMUTH PAYLOAD CAPABILITY

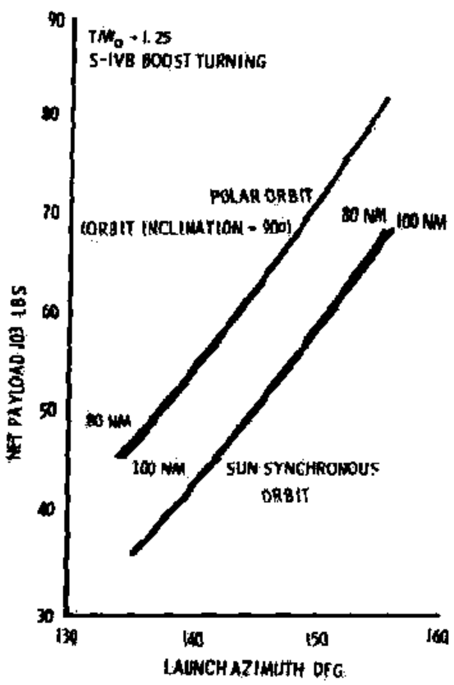


FIGURE 4-6 INT-20 POLAR & SUN SYNCHRONOUS PAYLOAD (DIRECT ASCENT)

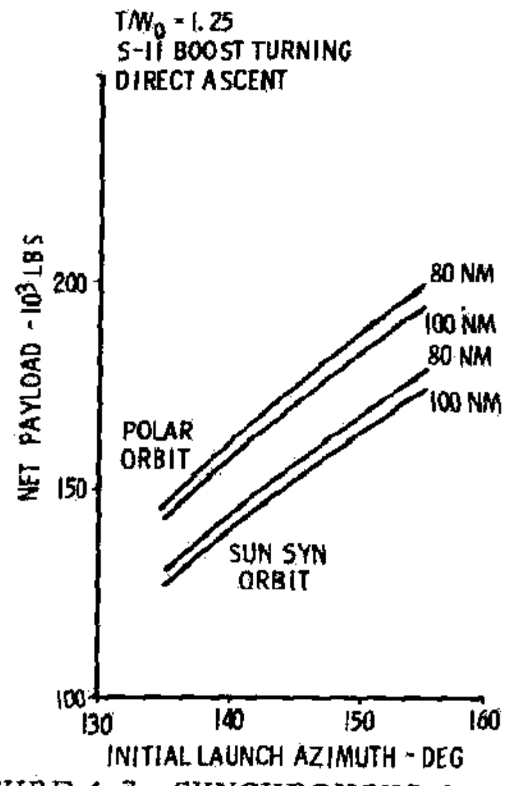
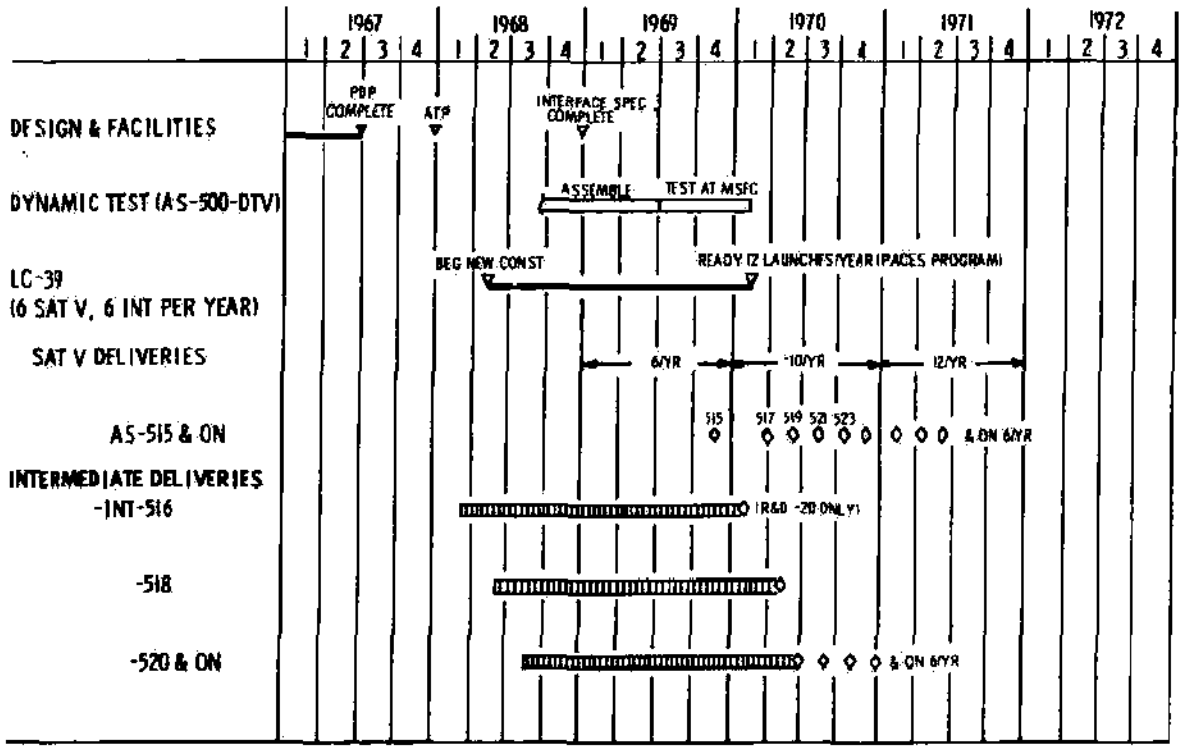


FIGURE 4-7 SYNCHRONOUS ORBIT PAYLOAD CAPABILITY

4.2.3 Resources

Study ground rules requiring six intermediate and six Saturn V vehicles per year have the strongest influence on resources. The greatest impact on facilities occurs at MILA. Here, 100 million dollars are required for an additional mobile launcher, a mobile service structure, and firing room equipment. Other launch system modifications and equipment are about 23 million dollars.



NOTE: MATERIAL PROCUREMENT FOR 12/YR PROD. AND LC-39 Mod
AND LC-39 MOD'S PACE PROGRAM
8/YR PROGRAM PACED BY PROCUREMENT AND MSS MOD
*AS REQUESTED BY KSC, PRODUCTION CAN REACH 12/YR IN 1970

FIGURE 4-8 INT-20/-21 VEHICLE DEVELOPMENT AND DELIVERY PLAN

A schedule for INT-20 or INT-21 development and delivery is shown on Figure 4-8. Study ground rules require a program definition phase (PDP) which may start as early as January 1967. Earliest authority to proceed (ATP) was ground-ruled to January 1968. With these controls, first intermediate vehicle delivery is February 1970. Since the INT-21 is a two-stage Saturn V vehicle, it will not require man-rating flights. The INT-20, even with previously man-rated stages, requires an R&D flight by MSFC ground rules.

Tables 4-III and 4-IV summarize the costs that would be incurred for implementing INT-20 and INT-21, respectively, plus the cost for 30 vehicles including launch.

COST DOLLARS IN MILLIONS	DEVELOPMENT STAGE	DEVELOPMENT ENGINE	OPERATIONAL STAGE	OPERATIONAL ENGINE	TOTAL
	$	$	$	$	$
LAUNCH VEHICLE					
S-IC STAGE	6.2		494.5	207.0	707.7
S-IVB STAGE	7.1		285.1	36.3	328.5
INSTRUMENT UNIT			128.2		128.2
LAUNCH VEHICLE TOTAL	13.3		907.8	243.3	1164.4
GROUND SUPPORT EQUIPMENT					
S-IC STAGE	9.1		21.5		30.6
S-IVB STAGE	.2		38.7		38.9
GSE TOTAL	9.3		60.2		69.5
FACILITIES					
S-IC STAGE	13.2				13.2
S-IVB STAGE			4.9		4.9
LAUNCH VEHICLE - KSC	121.7		364.2		485.9
FACILITIES TOTAL	134.9		369.1		504.0
SYSTEMS ENGINEERING AND INTEGRATION			234.0		234.0
LAUNCH SYSTEMS TOTAL	$157.5		$1571.1	$243.3	
	$157.5		$1814.4		$1979.7

R&D FLIGHTS (1) 60.4

TABLE 4-III INT-20 COST SUMMARY

COST DOLLARS IN MILLIONS	DEVELOPMENT STAGE	DEVELOPMENT ENGINE	OPERATIONAL STAGE	OPERATIONAL ENGINE	TOTAL
LAUNCH VEHICLE					
S-IC STAGE	3.9		508.3	256.9	769.1
S-II STAGE	13.6		449.4	176.8	639.8
S-IVB STAGE (INTERSTAGE)*			43.3		43.3
INSTRUMENT UNIT			128.2		128.2
LAUNCH VEHICLE TOTAL	17.5		1129.2	433.7	1580.4
GROUND SUPPORT EQUIPMENT					
S-IC STAGE	9.1		27.0		36.1
S-II STAGE			21.1		21.1
GSE TOTAL	9.1		48.1		57.2
FACILITIES					
S-IC STAGE	13.2				13.2
S-II STAGE	.6				.6
LAUNCH VEHICLE - KSC	123.5		393.2		516.7
FACILITIES TOTAL	137.3		393.2		530.5
SYSTEMS ENGINEERING & INTEGRATION			234.0		234.0
LAUNCH SYSTEMS TOTAL	163.9		1804.5	433.7	
	163.9		2238.2		2402.1

*ADAPT S-II TO IU

TABLE 4-IV INT-21 COST SUMMARY

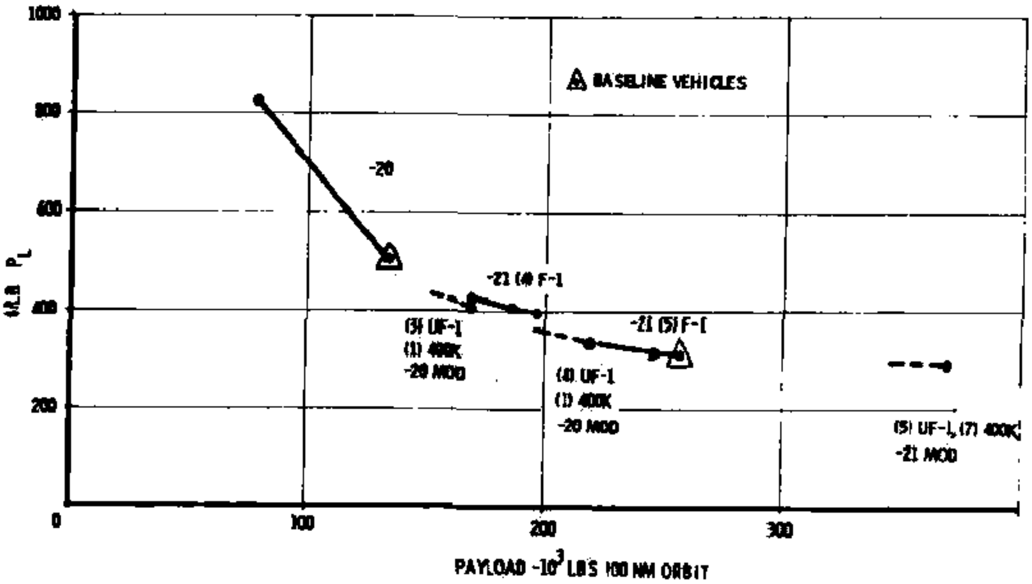


FIGURE 4-9   INT VEHICLE COST EFFICIENCY
COMPARISON

Figure 4-9 summarizes the payload cost efficiency of the eight INT vehicles (solid lines) not including R&D flights. The dotted lines are vehicles similar to INT-20 and INT-21 but using the MLV-SAT-V-3B stages (see Section 5.0 of this document). These data demonstrate that INT vehicles could be derived from any of the uprated configurations studied. The strap-on systems, in addition to first and third or first and second stage combinations, can also be assembled with zero, two, or four boost-assist units.

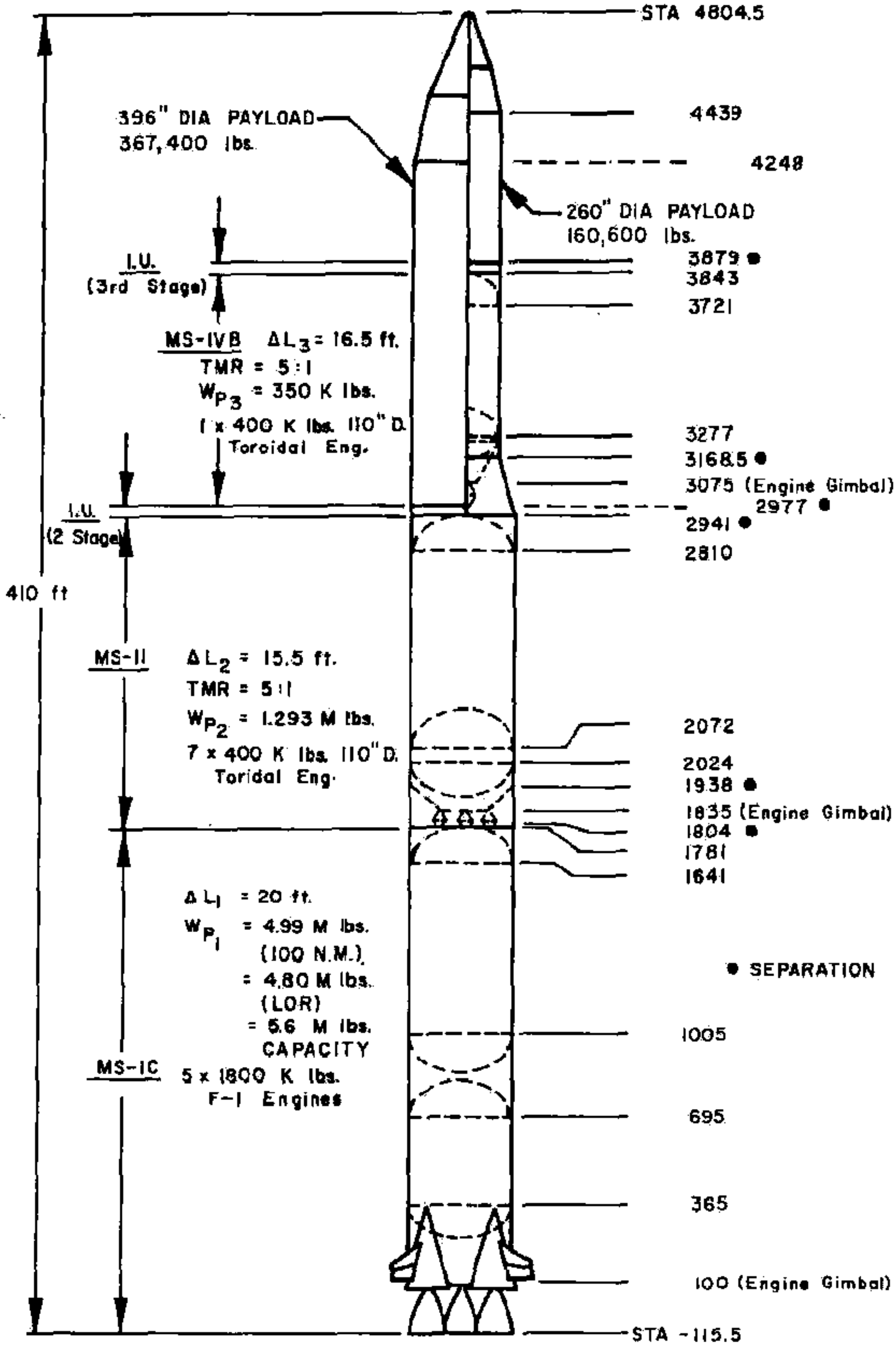


FIGURE 5-1 MLV-SAT-V-3B BASELINE LAUNCH VEHICLE

5.0 MLV-SAT-V-3B LAUNCH VEHICLES

The SAT-V-3B (see Figure 5-1) is a Saturn V with all stages lengthened and the thrust of each stage increased.

The vehicle, as defined in the trade study activity and studied in detail in the Phase II activity, is a feasible configuration and a logical candidate to provide payloads in excess of those currently available with the Saturn V vehicle.

5.1 CONFIGURATION SELECTION (PHASE I)

Trade studies were directed towards deriving sufficient data to allow MSFC to determine the best compromise MS-II-3B stage thrust level which satisfied requirements for both the SAT-V-3B three-stage vehicle and the MLV-SAT-INT-17 two-stage vehicle. The INT-17 uses the SAT-V-3B upper stages (MS-II/MS-IVB) as a ground launch vehicle, and was studied concurrently by North American Aviation under separate contract.

5.1.1 Candidate Configurations

MS-IC-3B thrust was fixed at five 1.8 million-pound F-1 engines for all configurations. The second stage is an MS-II-3B using from four to seven advanced engines with 300,000 to 700,000 pounds of thrust. The three-stage vehicles have an MS-IVB-3B as the third stage using a single engine of the same type and thrust level as for the second stage. Maximum vehicle length was 410 feet.

Upper stage propulsion considered two advanced engine concepts, a toriodal aerospike engine and an advanced bell engine (see Figure 5-2).

The LOX/LH_2 aerospike engine has a toroidal combustor and truncated aerodynamic spike annular nozzle. This design results in a 64-inch reduction in engine length. The other engine considered was a high-pressure LOX/LH_2 concept with a bell nozzle. Bell nozzle engine length, from gimbal point to nozzle exit plane, was maintained at 116 inches because of upper stage interstage clearance requirements.

5.1.2 Trade Studies

Parametric data developed for the SAT-V-3B two- and three-stage vehicles included: (1) weight and mass characteristics, (2) trajectories and performance, (3) aerodynamics and heating, (4) design loads, and (5) separation. Trades were also made for the two types of advanced engines operating over a range of thrust levels.

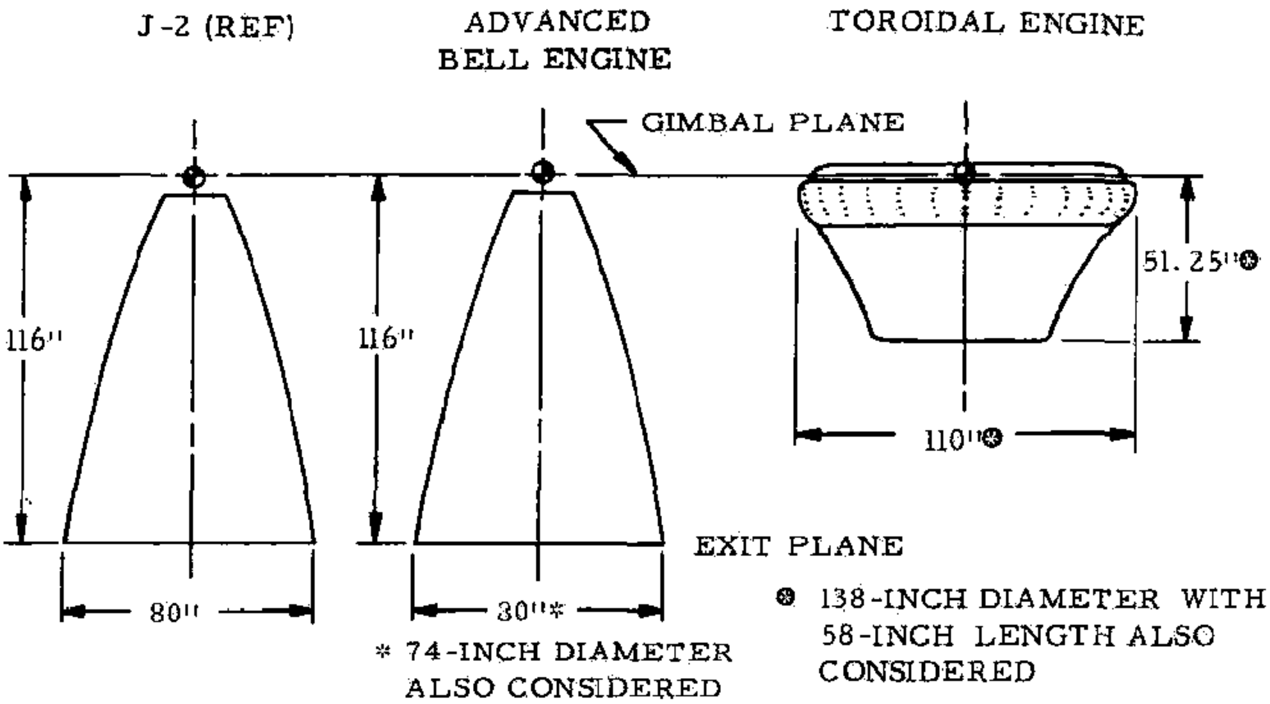


FIGURE 5-2 TRADE STUDY UPPER STAGE ENGINES

Data was prepared for vehicles whose stage lengths were optimized to yield maximum payload capability. Payload performance was also determined for vehicles whose upper stage length increases were fixed at 15.5 feet for the second stage and 16.5 feet for the third stage.

Net payloads for the three-stage vehicle with bell nozzles are shown on Figure 5-3. The data are typical of those prepared for the trade study. Dashed lines on Figure 5-3 refer to the number of engines on the MS-II-3B stage while the solid lines give thrust per engine. The lower group of curves shows payload with the upper stage sizes fixed. The upper set of curves covers the propellant-optimized stages. With optimized stages, the payload increases with increasing thrust in the upper stages. However, the vehicle height limit of 410 feet is quickly exceeded. The same type of data for the two-stage bell-nozzle configuration exhibit the same trends. With fixed upper stages, maximum payload is achieved with MS-II-3B thrust of around two million pounds for both the two- and three-stage vehicles.

Figure 5-4 summarizes performance for fixed upper stage configurations and propellant-optimized versions within the 410-foot limit. These data cover two- and three-stage vehicles with bell and toroidal upper stage engines. Use of toroidal engines gives a two to five percent increase in payload. Performance results for the SAT-V-3B favor a total MS-II

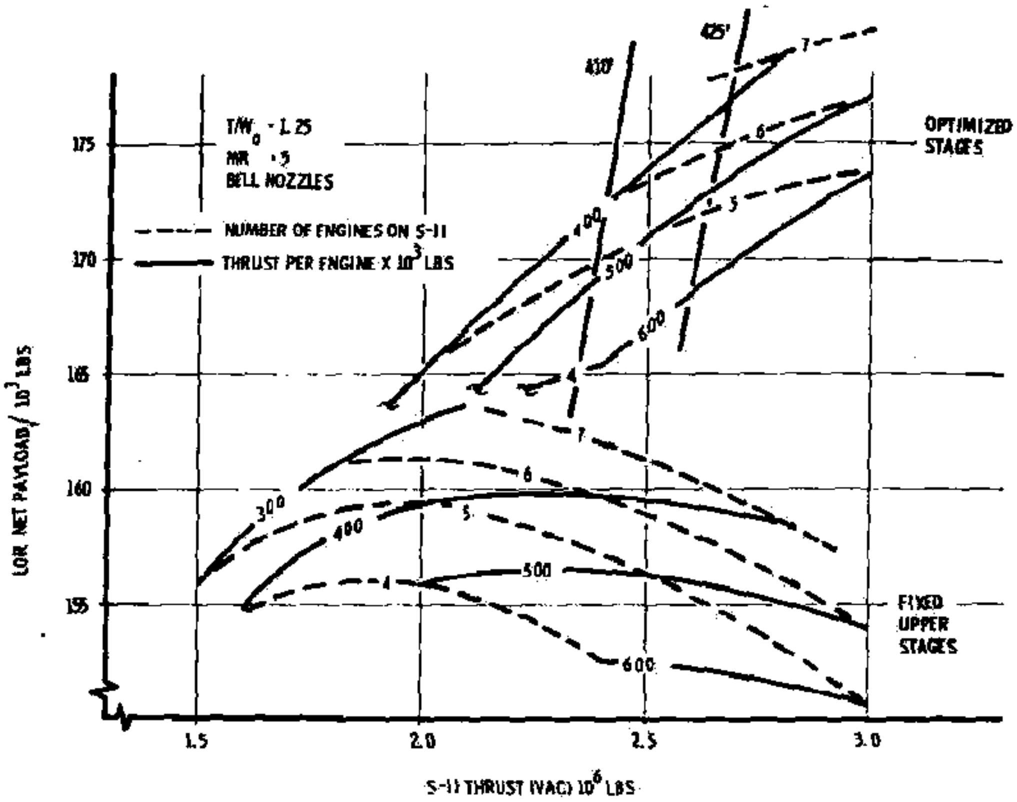


FIGURE 5-3 TRADE STUDY PERFORMANCE DATA

thrust of around two million pounds using 400, 000 to 500, 000 pounds of thrust per engine. Seven 300, 000 pound thrust second stage engines show a 2.6 percent increase over five 400, 000 pound thrust engines. The lower thrust engines exhibit better performance because engine length was held constant. To decrease engine thrust, the nozzle throat area was decreased thereby increasing engine area ratio and thus specific impulse. Higher mixture ratio (6:1) in upper stages showed a small payload improvement as indicated.

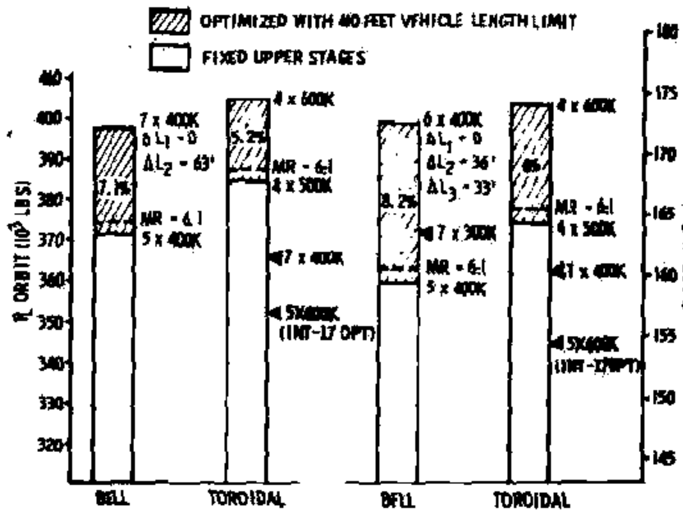


FIGURE 5-4 TRADE STUDY VEHICLE PAYLOAD COMPARISON

These trade study data, those prepared on MLV-SAT-INT-17 and their respective trade study resource analysis were compared by NASA/MSFC in order to reach a compromise on engine quantity, thrust, and type. It was found that the INT-17 vehicle would operate most efficiently at approximately 3.0 million pounds of thrust. On the other hand, SAT-V-3B needs a second stage thrust closer to 2.0 million pounds for most efficient operation. Further, the SAT-V-3B third stage requires not more than 180,000 pounds of thrust for most efficient operation. NASA/MSFC selected a compromise second stage thrust of 2.8 million pounds, seven 400 thousand pound engines, for further study. The compromised performance by this choice is indicated on Figure 5-5. Also indicated is the degradation to SAT-V-3B which would have resulted had NASA chosen a still higher thrust level (3.0 million pounds). MSFC selected the toroidal aerospike engine rather than the bell nozzle engine since the bell was examined in detail in last year's studies.

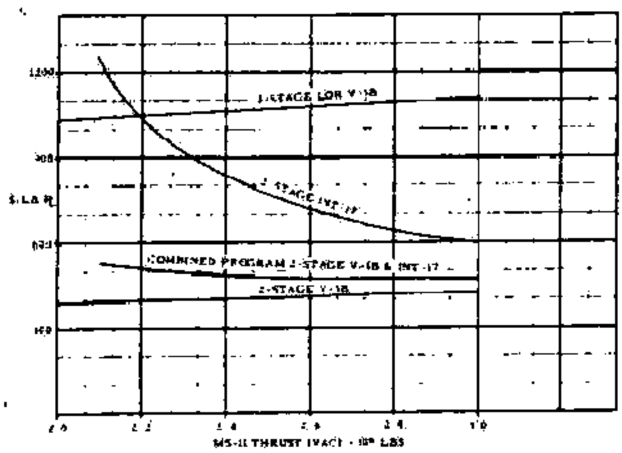


FIGURE 5-5 SAT-V-3B & INT-17 COST EFFICIENCY TRADE

	-3B	SAT V *
LOAD CRITERIA		
MAX. q (LBS/FT^2)	735	766
g's AT MAX q α	1.92	1.954
HEIGHT (FT)	410	363
CONTROL		
MODE	GIMBALED F-1'S	GIMBALED F-1'S
MAX. DEFLECTION ANGLE IN FLIGHT	4.3 DEG.	3.5 DEG.
HEATING		
TYP. AERODYNAMIC (S-IC FWD SKT)	167°F	167°F
BASE MAX. TEMP.	1940°F	1900°F

* BASELINE 516 WITH T_0/W_0 = 1.25

TABLE 5-I SIGNIFICANT LOAD CRITERIA

5.2 DESIGN STUDY VEHICLE (PHASE II)

Design and analyses were conducted for two-and three-stage SAT-V-3B vehicles using the propulsion systems chosen by MSFC. Complete resources were prepared for the three stage vehicle.

5.2.1 Vehicle Description

The MLV-SAT-V-3B baseline vehicle is shown in Figure 5-1. The MS-IC-3B stage uses five 1.8 million pound thrust uprated F-1 engines. First stage length increase is 20 feet for a propellant capacity of 5.6 million pounds with a propellant loading (T/Wo = 1.25) of 4.99 million pounds and 4.8 million pounds for the two and three stage vehicles, respectively. The second stage has seven 400,000 pound thrust toroidal aerospike engines. It has a length increase of 15.5 feet for a propellant capacity of 1.29 million pounds. The shorter toroidal engines allows a 62-inch reduction in interstage length thereby permitting a commensurate tankage increase. The third stage (for three stage application) uses a single 400,000 pound thrust toroidal aerospike engine, a 16.5 foot length increase for a propellant capacity of 350,000 pounds of propellant.

5.2.2 Design Study Results

Payload performance data for the SAT-V-3B for both nominal and alternate missions were determined. The nominal mission for the two stage version is direct ascent to a 100 nautical mile circular Earth orbit. Nominal mission for the three-stage vehicle is direct ascent to a 100 nautical mile circular parking orbit, followed by reignition of the third stage and boost into a 72 hour lunar transfer trajectory. Alternate missions considering a range of altitudes and launch azimuths were also considered.

Figure 5-6 summarizes the orbit/altitude capability for the two stage SAT-V-3B. Net payload for the nominal mission is 367,400 pounds. However, with the high thrust (2.8 million pounds) and short burn time of the MS-II-3B stage, a sizable performance loss occurs at the higher orbit altitudes. For example, more payload is obtained at a 300 nautical mile orbit with existing two stage Saturn V (INT-21) than is obtained with a SAT-V-3B. If engine throttling is used in the MS-II-3B second stage, the payload losses to the higher orbits are reduced considerably as shown in Figure 5-6.

High energy mission (C_3) performance of the three stage vehicle is illustrated on Figure 5-7. Net payload for the nominal 72 hour lunar injection mission is 160,000 pounds. Payloads for polar and sun synchronous orbits are shown on Figures 5-8 and 5-9. A boost turn is required to obtain these orbits from Cape Kennedy. This maneuver requires energy expenditure which is reflected in less payload capability.

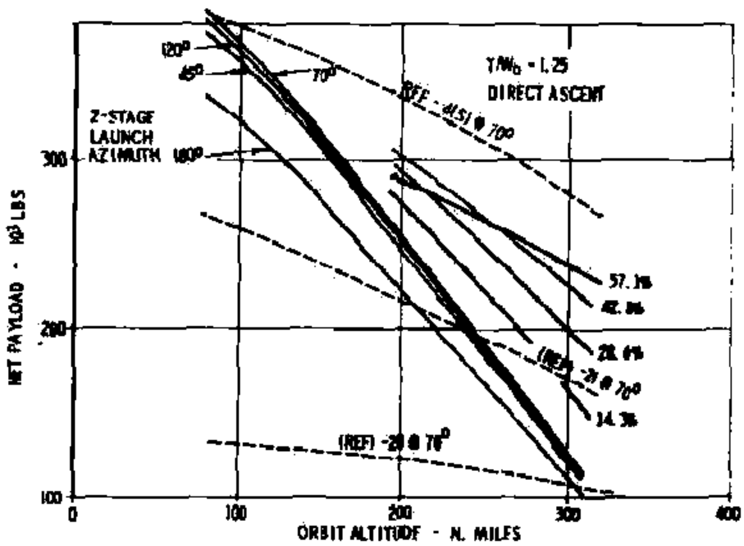


FIGURE 5-6 ORBIT ALTITUDE - AZIMUTH PAYLOAD CAPABILITY

Significant load criteria, and other data pertinent to vehicle design, are shown on Table 5-1, with comparative Saturn V values. Although maximum dynamic pressure (q) and acceleration are slightly less for the

SAT-V-3B the increase in vehicle height, the 33-foot diameter two-stage payload, and increased engines thrusts have significantly increased structural loads over the existing Saturn V.

Control maximum deflection angle in flight has increased from 3.5 degrees (on Saturn V) to 4.3 degrees using the current attitude, attitude rate control system. This is well within the current 5.15 degree gimbaling capability of F-1 engines. Alternate control mode studies showed that approximately 12 percent reduction in maximum bending response could be expected if an angle of attack feedback loop were added to the attitude, attitude rate control mode.

Both aerodynamic and base heating temperatures on the SAT-V-3B vehicle are comparable to Saturn V.

The reliability of the two and three stage configuration of SAT-V-3B vehicle is 0.975 and 0.965, respectively, as compared to 0.990 and 0.980 for the baseline AS-516.

The additional propellant in all stages increases the 0.4 psi on-pad explosive overpressure range 600/feet compared to Saturn V but not enough to impact the adjacent pad at MILA.

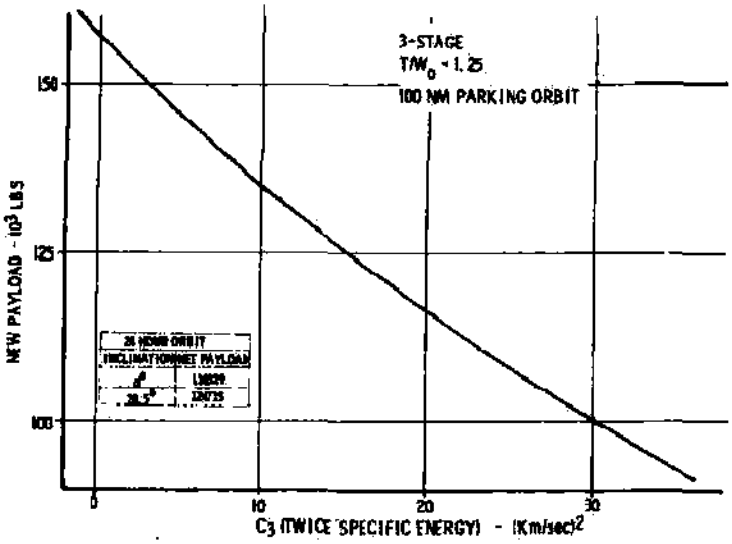


FIGURE 5-7 HIGH ENERGY PAYLOAD CAPABILITY

Saturn V flight and crew provisions are satisfactory for use with this vehicle. No problems were found with structural dynamics, RF attenuation or antenna look angle.

Structural loads and acoustic environments are illustrated on Figure 5-10. The acoustic loads have increased slightly in the first stage but are within the existing specification limit. Some selective requalification of components may be required. The second stage acoustic level has increased sufficiently from the static firing of seven 4,000,000 pound thrust toroidal engines and inflight conditions to require a new spec limit for this stage. No problems exist in the third stage.

As shown in Figure 5-10, the combined loading condition has increased significantly compared to Saturn V, requiring a major structural beefup as

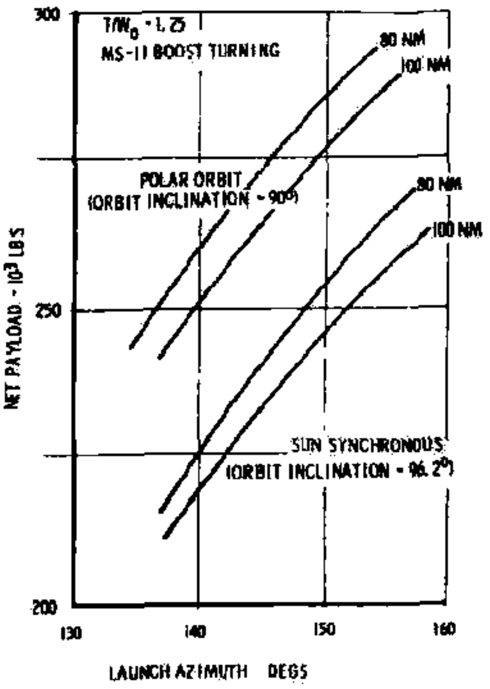


FIGURE 5-8 TWO-STAGE POLAR & SUN SYNCHRONOUS ORBIT PAYLOAD CAPABILITY

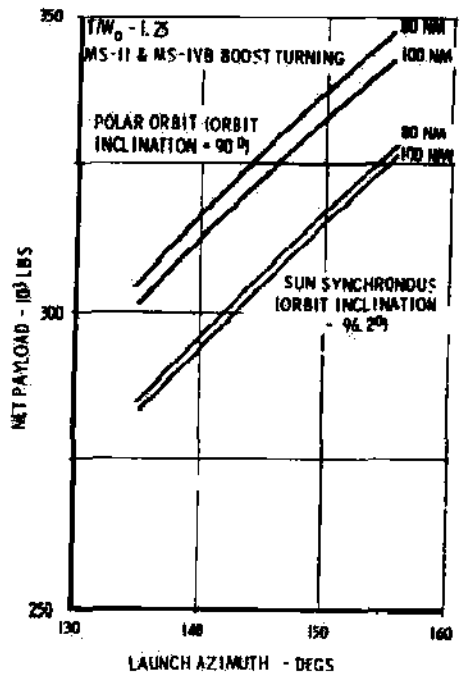


FIGURE 5-9 THREE-STAGE POLAR & SUN SYNCHRONOUS ORBIT PAYLOAD CAPABILITY

as shown in Figure 5-11. Percentage increases in dry weights of each SAT-V-3B stage are also tabulated in Figure 5-11.

5.3 RESOURCES

Increases in the length and thrust of the SAT-V-3B - stages compared to Saturn V impact production, test, transportation and launch facilities. Uprated F-1 engine and new toroidal upper stage engine developments are the most costly items required. Existing facilities will be employed to manufacture and test the MLV-SAT-V-3B. These facilities are to be used on a non-interference basis with normal Saturn V production schedules. The present stage and I. U. vendors were assumed to be contractors for the modified vehicle components.

A dynamic test vehicle, structural test components, and two man-rating R&D flights are required. Relocation of work platforms and increase in height is required at the MSFC Dynamic Test Stand to handle the new configuration. The first stage of the dynamic test vehicle will be refurbished after test and used as a flight stage. The second stage of the dynamic test vehicle will have undergone structural static test prior to -D testing.

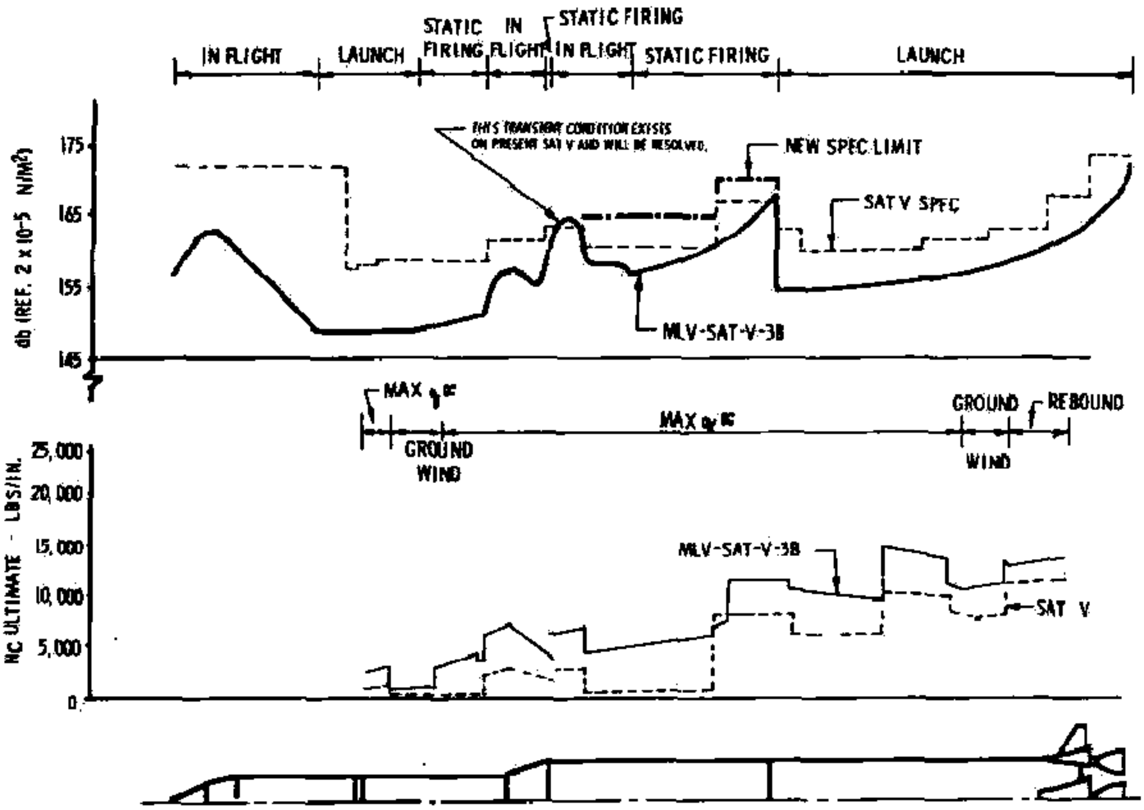


FIGURE 5-10 ACOUSTIC ENVIRONMENT AND STRUCTURAL LOADS

A production rate of six vehicles per year for a period of five years was used to assess production impact. A companion intermediate payload vehicle, also produced at six per year for five years, was the Saturn IB, making use of the MS-IVB-3B.

MS-IC-3B

The thrust increase from 7.5 to 9.0 million pounds necessitates increases in skin stiffener and ring frame gage throughout the MS-IC stage. These changes plus the increase in stage length require requalification of the entire structure.

Revisions are made to S-IC tooling to account for the new material thickness and stiffener locations or for length increases. An additional tank assembly position and new tank clean position are added to avoid long downtime while tooling up for the new configuration. The MTF and MSFC test stands require modification to accept the longer stage. Both stands, however, are capable of handling the greater thrust but more holes are needed in the flame deflectors.

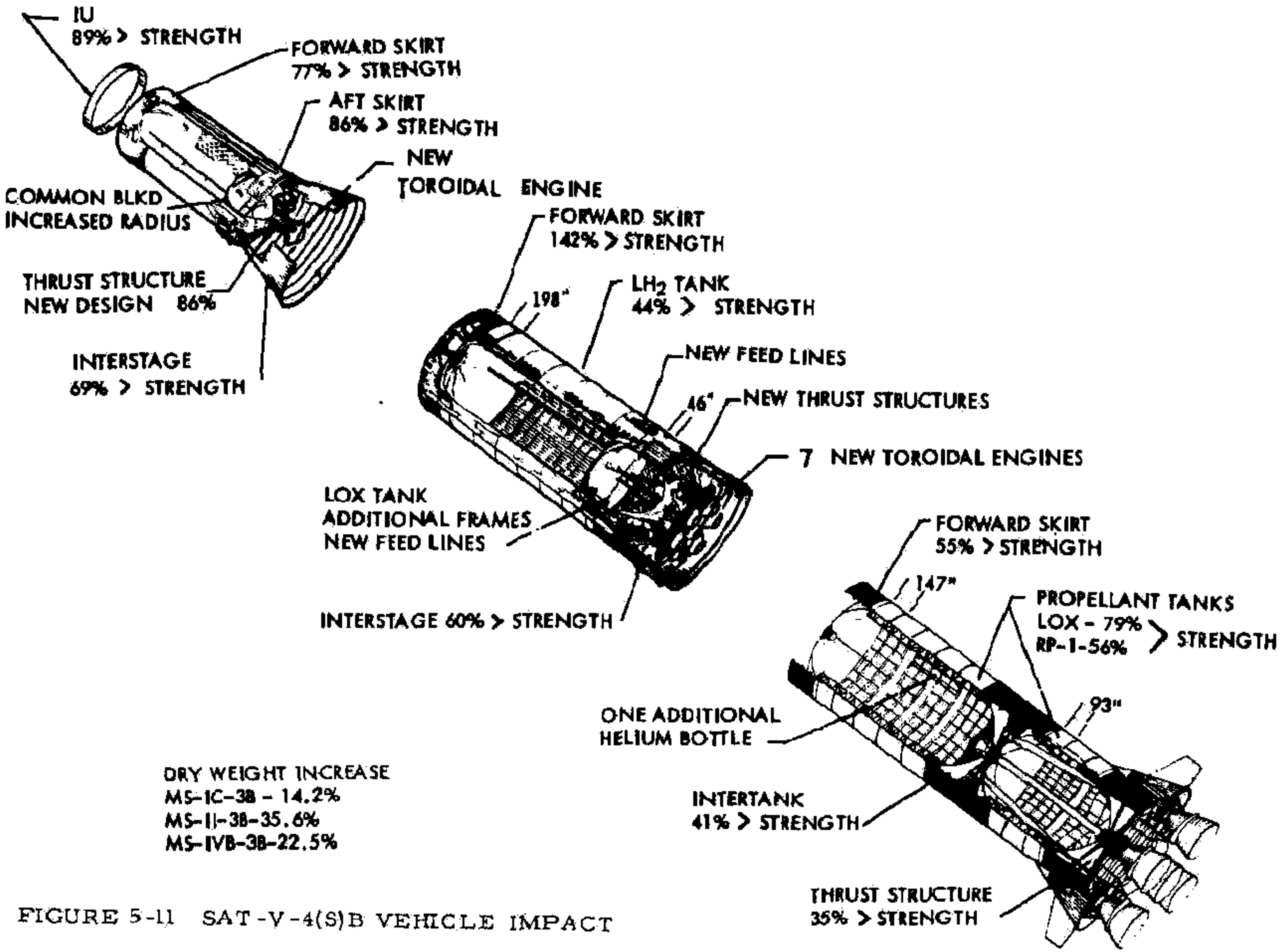


FIGURE 5-11 SAT-V-4(S)B VEHICLE IMPACT

Timing of the MS-IC-3B production is paced by upper stage requirements; therefore, Authority to Proceed is not required until 26 months after start of the upper stage programs. With this timing, the first flight article will be delivered to MILA 3.5 years after first stage ATP.

MS-II-3B

The structure and propulsion system changes to this stage will necessitate static and dynamic structural tests and a "battleship" test development program. To meet the required delivery date of the dynamic test stage, production of the standard S-II would be accelerated, starting with S-II-18. Two flight test stages are included in the development phase.

Standard transport equipment is generally compatible with the MS-II-3B, although lengthening of stage transporters would be necessary, as well as minor modifications such as relocation of tie downs on the Point Barrow. Both Type I and Type II transporters can handle the added weight, except that tire loads may be excessive on the Type II.

Delivery of the dynamic test stage, nine months prior to the first flight stage, determines the point at which production of the standard S-II would be phased out to allow phase-in of new tooling for the MS-II-3B. The structure and propulsion system revisions require modification of standard tools or design of new tools. Tooling mainly affected is that for manufacture of the LH_2 tank, forward and aft skirts, interstage, LOX tank extension and new thrust structure.

The Seal Beach facility requires some modifications, including minor building extensions and revised assembly sequences. New facilities are not necessary, but stage assembly and buildup tools will be modified. Handling equipment at all facilities will be modified for the increased stage weight. Modification and checkout of the Static Test Tower would be completed in December 1971, the dynamic test stage shipped to MSFC in December 1971, and the first operational flight stage completed in September 1973.

MS-IVB-3B

Engineering design or redesign effort is needed mainly in the structure and propulsion system areas because of the increased tank volume, higher flight loads and the installation of a new type higher thrust engine. Development and qualification effort involves a moderate amount of work associated with the same vehicle changes. A dynamic test stage will be furnished for test at NASA/MSFC to verify predicted vibration modes and frequencies.

To allow for modification to tooling and facilities without interfering with standard stage deliveries, a temporary speed-up of the standard stage assembly is planned. No new fabrication technology is proposed nor any new or unique testing procedures. Some expansion and/or modification of facilities is required, at the Santa Monica and Huntington Beach plants, and at the Sacramento Test Center.

The increased stage size precludes transportation by the Super Guppy and will mean dependence on ocean shipment to the Sacramento test site and to Kennedy Space Center. Major modification of the stage transporter is needed because of the added length, and lesser modifications to other items of handling equipment.

Launch Facility and Operation Impact

Changes at MILA for the SAT-V-3B vehicle are primarily due to increased vehicle length. Mobile launcher swing arms as well as VAB high and low bays access platforms are relocated. A new (taller) mobile service structure is needed since insufficient time is available between last Saturn V and first MLV-SAT-V-3B to rework the existing MSS.

No changes are required in the Saturn V operational plan for SAT-V-3B.

Schedule

An MLV-SAT-V3B vehicle development and delivery schedule is shown in Figure 5-12. The vehicle timing is based on almost four years required for upper stage engine development. Upper stage design is paced so that battleship stages are available when PFRT engines are ready. Completion of all critical stage ground development testing is completed before the first flight article reaches MILA in September of 1973.

Cost

A cost summary for the SAT-V-3B is shown in Table 5-II.

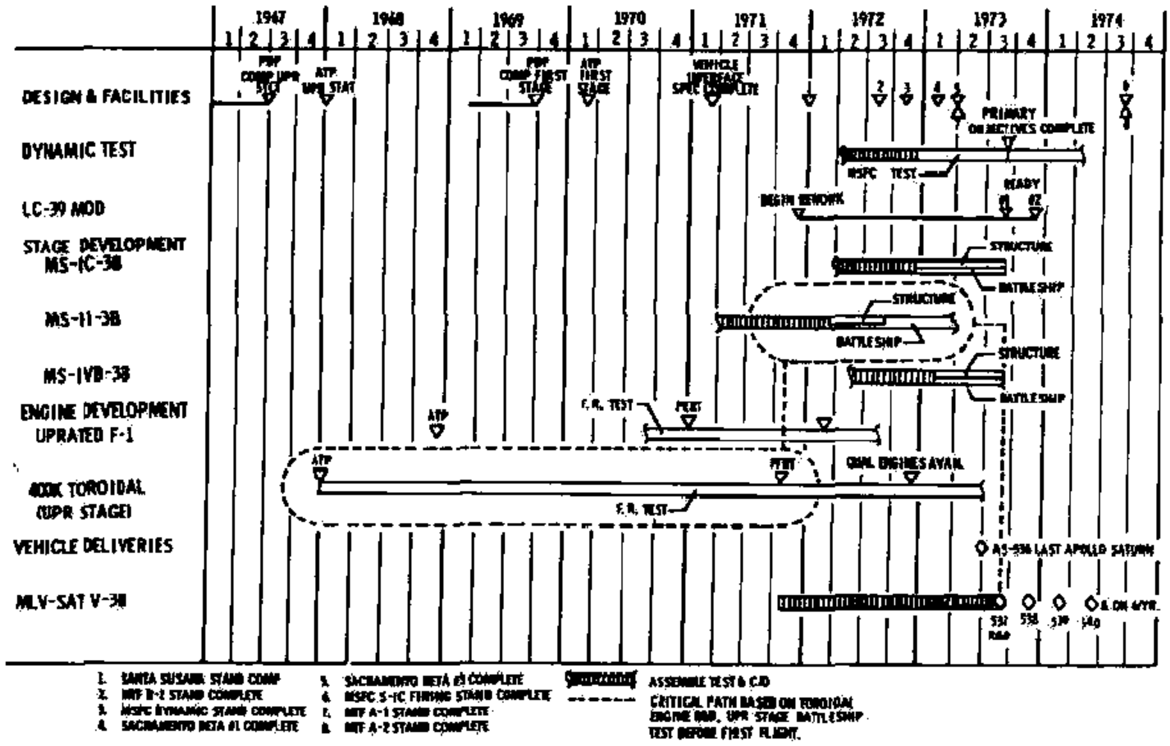


FIGURE 5-12 SAT-V-3B VEHICLE DEVELOPMENT AND DELIVERY PLAN

COST - DOLLARS IN MILLIONS	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE					
S-IC Stage	70.0	123.0	596.3	439.7	1214.0
S-II Stage	165.7	336.2	551.1	686.9	1774.2
S-IVB Stage	97.8	62.9	362.4	114.4	642.4
Instrument Unit			130.7		130.7
LAUNCH VEHICLE TOTAL	333.5	522.1	1640.5	1241.0	3761.3
GROUND SUPPORT EQUIPMENT					
S-IC Stage	10.0		21.9		31.9
S-II Stage	28.1		64.8		92.9
S-IVB Stage	33.7		48.5		82.2
GSE TOTAL	71.8		135.2		207.0
FACILITIES					
Toroidal Engines		39.2			39.2
S-IC Stage	13.3				13.3
S-II Stage	21.7				21.7
S-IVB Stage	7.2		5.4		12.6
Launch Vehicle - KSC	81.7		721.8		803.5
Launch Vehicle - Other	4.8				4.8
FACILITIES TOTAL	128.7	39.2	727.2		895.1
SYSTEMS ENGINEERING AND INTEGRATION	2.3		425.1		425.1
LAUNCH SYSTEMS TOTAL	536.3	561.3	2928.0	1241.0	
	1097.6		4169.0		5266.6
			R&D FLIGHTS (2)		325.6

TABLE 5-II SAT-V-3B COST SUMMARY

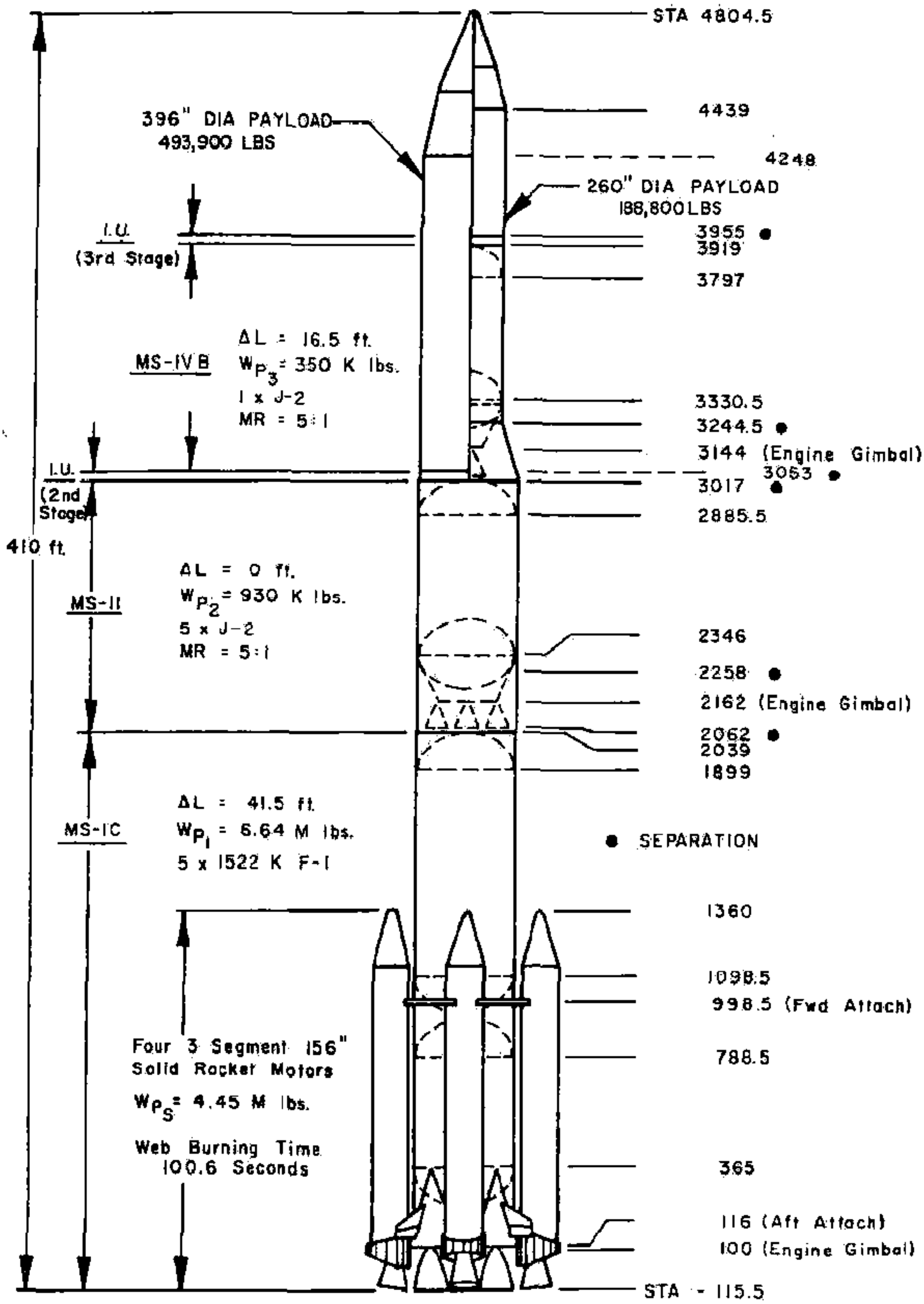


FIGURE 6-1 MLV-SAT-V-25(S) BASELINE VEHICLE

6.0 MLV-SAT-V-25(S) LAUNCH VEHICLE

The Saturn V-25(S) vehicle (see Figure 6-1) is a Saturn V with lengthened first and third stages, adapted for attachment of four 156-inch diameter solid propellant motors.

The vehicle as defined in the Phase I trade study activity and studied in detail in the Phase II activity is a feasible configuration and a logical candidate to provide payloads in excess of those currently available with the Saturn V vehicle.

6.1 CONFIGURATION SELECTION (PHASE I)

By varying the weight and thrust of the 156-inch solid rocket motors and the weight of propellant in the core stages, a number of related SAT-V-25(S) vehicles were evolved. Payload capability and vehicle costs were established for these vehicles in order to choose one arrangement for more detailed analysis.

6.1.1 Candidate Configurations

For the trade study, both two- and three-stage operation was considered. Vehicle height was fixed at 410 feet for both two- and three- stage configurations. Propulsion and engine type for all stages was fixed to correspond to the baseline AS-516 vehicle. Varying weights of propellant and corresponding stage lengths were studied for all stages. Four 156-inch solid propellant rocket motors were attached to the vehicle for thrust augmentation. The number of segments (and thus solid propellant weight) in the solid motors was varied between two and four. Solid motor thrust/time restraints were specified by MSFC. Burntimes and thrust levels of the various sized solid motors were varied, also, within the restraints, to optimize vehicle liftoff thrust-to-weight.

6.1.2 Trade Studies

Figure 6-2 is typical of the parametric performance data prepared for the trade study. Figure 6-2 illustrates the net payload for the various number of segments in the 156-inch motors as a function of liftoff thrust-to-weight for the three-stage vehicle. This chart shows two conditions, i.e., (1) optimized first stage propellant weight with standard second stage propellant weight,

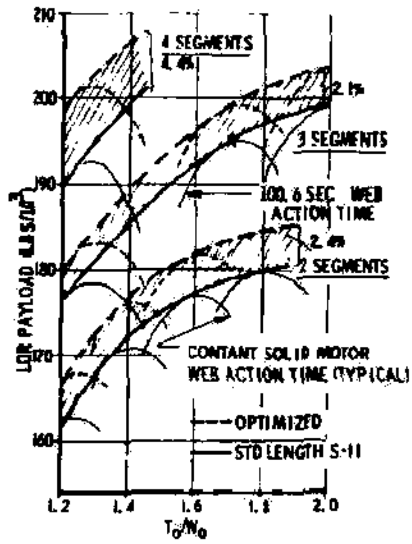


FIGURE 6-2 PERFORMANCE TRADE DATA

and (2) optimized propellant weights for the first and second stages. The MS-IVB in all cases was sized to maximize payload out of 100 nautical mile Earth orbit to lunar injection. Note that the curves are the loci of maximum payload resulting from considering different constant solid motor web action times combined with various core vehicle launch weights.

The data demonstrate that:

a. Payload increases approximately eight percent with each additional solid motor segment.

b. A payload increase of more than two percent accrues by optimizing second stage length. Typical optimized S-II stage length increases are on the order of an additional ten feet.

c. Significant payload increases are attributable to the shorter burn time solid rocket motors and the resulting higher values for liftoff thrust-to-weight ratio.

In developing these data, no structural penalties were assessed to the candidate vehicles for the liftoff thrust-to-weight variation. When structural weight penalties are considered, as they were in other similar studies, it is found that beyond 1.6 to 1.8 thrust-to-weight, payload increases are not as large as indicated on Figure 6-2.

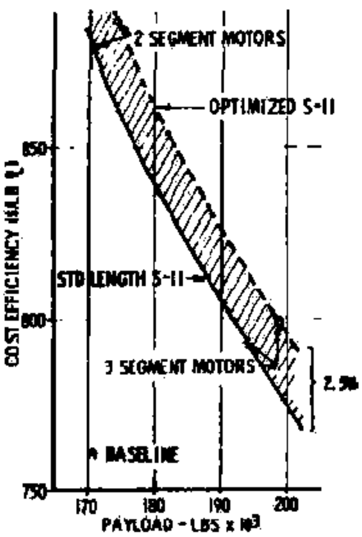


FIGURE 6-3 COST EFFICIENCY TRADE DATA

Figure 6-3 compares the payload cost efficiency of different solid motor weights for the optimized S-II stage and standard length S-II stage. The chart indicates that S-II stage optimization is not worthwhile but that larger solid motor weights can significantly improve vehicle cost efficiency. From these data, and because minimum solid motor burntime was ground ruled at 100.6 seconds, it was recommended to MSFC that the vehicle indicated as "baseline" on Figures 6-2 and 6-3 be chosen for the next phase of study. MSFC approved this selection.

6.2 DESIGN STUDY VEHICLE (PHASE II)

The single SAT-V-25(S) vehicle chosen during the Phase I activity was defined in detail, its capabilities and characteristics were determined, and its resource requirements were established.

6.2.1 Vehicle Description

The MLV-SAT-V-25(S) baseline vehicle is shown in Figure 6-1. It utilizes four three-segment 156-inch strap-on solid rocket motors each with 1.1 million pounds of propellant for thrust augmentation. Each solid motor has a launch thrust of 4.0 million pounds. They burn regressively so that at burnout thrust is reduced to about 65 percent of the liftoff value. Each of the solid motors has a liquid injection (N_2O_4) thrust vector control system to augment the capability of the gimbaled F-1 engines during flight through the max q regime. The liquid core stages of SAT V-25(S) are equipped with standard F-1 and J-2 engines. First stage propellant capacity has been increased to 6.64 million pounds by lengthening the stage 41.5 feet. The second stage is standard S-II length with a propellant capacity of 930,000 pounds. The third stage (for three-stage applications) is increased in length by 16-1/2 feet and has a propellant capacity of 350,000 pounds.

6.2.2 Design Study Results

As noted on Figure 6-1, the SAT-V-25(S) two stage payload capability to 100 nautical mile orbit is almost 494,000 pounds and its 72-hour lunar injection three-stage capability is almost 189,000 pounds.

Use of this vehicle was also considered for applications where the baseline core vehicle (liquid stages without solids) could be flown by itself or with only two strap-on solid motors. The payloads identified for these alternates are shown on Table 6-I.

To further improve payload, a special study was conducted to determine the effect of tailoring the solid motor thrust time trace. For this study, the thrust was made regressive until the vehicle has passed through the maximum dynamic pressure regime. The thrust level was then made progressive until solid motor burnout. The payload improvement for the optimum regressive-progressive thrust time trace over that available with the baseline vehicle was approximately one percent. This improvement was not considered significant enough to warrant complicating solid motor design for regressive-progressive burning.

Significant load criteria, and other data pertinent to vehicle design, are

TABLE 6-I PAYLOAD CAPABILITY

	NET PAYLOAD (LB) TWO-STAGE 100 NM Orbit	NET PAYLOAD (LB) THREE-STAGE 72-hour Lunar Traj.
Core Vehicle Without Solid Motors		
$T_o/W_o = 1.25$, $W_{Pl} = 4,343,423$ lb	231,466	88,475
$T_o/W_o = 1.18$, $W_{Pl} = 4,696,492$ lb	239,558	91,568
Core Vehicle With Two Solid Motors		
$T_o/W_o = 1.40$, $W_{Pl} = 6,000,000$	387,073	147,954
Core Vehicle With Four Solid Motors		
$T_o/W_o = 1.734$, $W_{Pl} = 6,640,000$ lb	493,900	188,800

shown on Table 6-II with comparative Saturn V values. Although max dynamic pressure (q) and acceleration are increased slightly, the 410 foot vehicle height coupled with the 33-foot diameter two-stage payload cause the largest impact on structural design requirements.

Control requirements necessitate additional capability beyond the present fin and F-1 engine gimbaling of the first stage. The use of the liquid injection thrust vector control on the solid motor is required for 26 seconds near max q time of flight. Also, because the first stage is rotated 45 degrees compared to Saturn V, the flight control signal must be modified to compensate for the rotation.

Aerodynamic heating is significantly lower than the Saturn V. The shock wave from the solid motor nose cap may impinge on the first stage LOX tank and local insulation may be required. No problems are anticipated as a result of aerodynamic heating.

	-25(S)	SAT V
LOAD CRITERIA		
MAX q (LB'S/FT^2)	859	766
g'S AT MAX q α	1.99	1.954
HEIGHT (FT)	410	363
CONTROL		
MODE	GIMBALED F-1'S PLUS N_2O_4 LITVC ON SOLIDS	GIMBALED F-1'S
SOLID MAX. DEFLECTION ANGLE	1.8° PER MOTOR	N/A
SOLID TVC OPERATING	46-71 SEC	N/A
HEATING		
TYP AERODYNAMIC (S-IC FWD SKT) MAX TEMP	132°F	167°F
(MAX TEMP BASE)	2080°F	1900°F
OTHER	MS-IC-25 (S) ROTATED 45°	

BASELINE 516 WITH T_0/W_0 = 1.25

TABLE 6-II SIGNIFICANT LOAD CRITERIA

The base heating environment is more severe for the MLV-SAT-V-25(S) than for the Saturn V due to the solid motor exhaust plumes. Heat shield materials can withstand the anticipated 2080°F temperatures successfully. The aft solid motor attachment skirt will reach 1950°F. Insulation protection here will be required.

The reliability of the two- and three-stage configurations of Saturn V-25(S) are 0.986 and 0.964, respectively, as compared to 0.990 and 0.980 for the baseline AS-516. The lower values for reliability can be attributed to the addition of the strap-on solid motors and to longer first and third stage burntimes.

Separation of the 156-inch SRM's from the vehicle can be accomplished satisfactorily using explosive separation devices and small solid rockets for separation force.

The addition of more fuel in the first stage and the four solid motors increases the 0.4 psi over-pressure distance to a value greater than the distance between Pad A and Pad B on Launch Complex 39. Waivers for this distance will be required for joint usage of these pads when either pad contains a fueled core vehicle with the solid motors attached.

Saturn V flight and crew safety provisions are satisfactory for use with this vehicle. Communications for some stations will be "blacked out" due to the exhaust plume interference. Other stations, however, will have clear antenna access during these periods and continuous communications can be maintained

Structural loads and acoustic environment are illustrated in Figure 6-4. The design loads are higher than those for the standard Saturn V requiring an increase in vehicle structural weight. The present acoustic specification limits are exceeded at several locations on the first stage. Requalification of acoustically sensitive components on this stage will be required.

Major core vehicle changes including the impact of structural load increases is summarized in Figure 6-5. Dry weight increases are also tabulated.

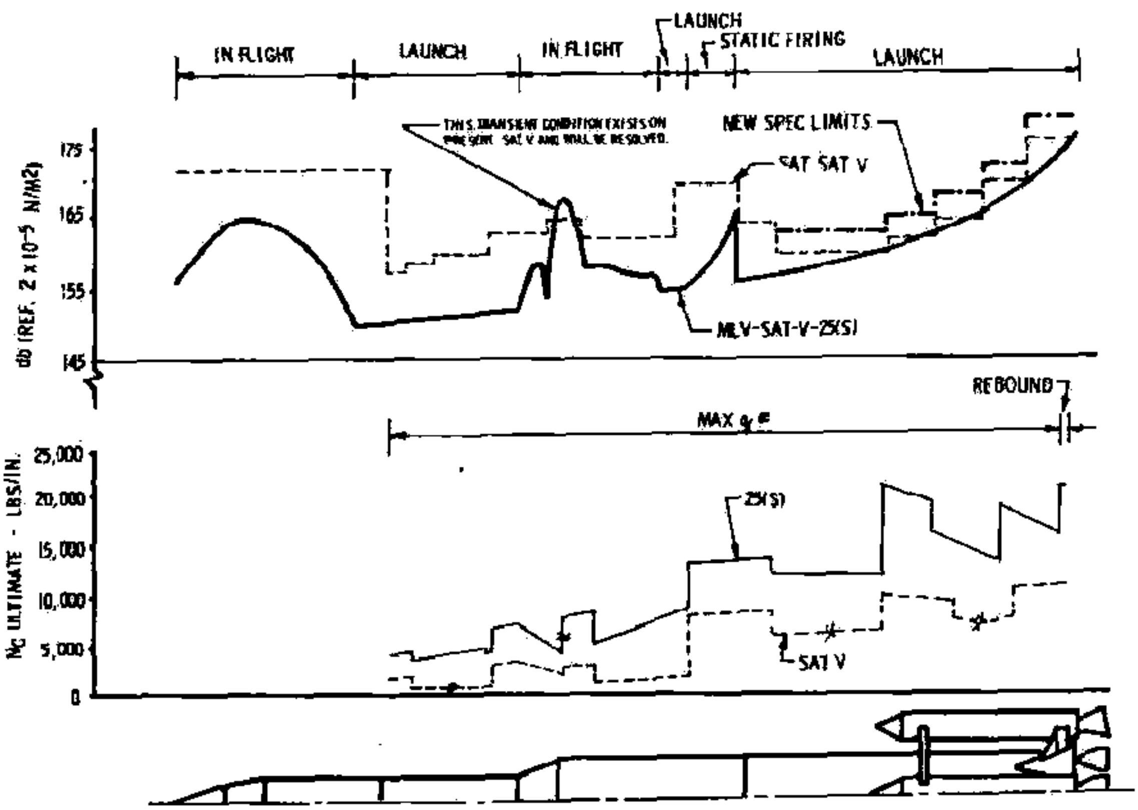
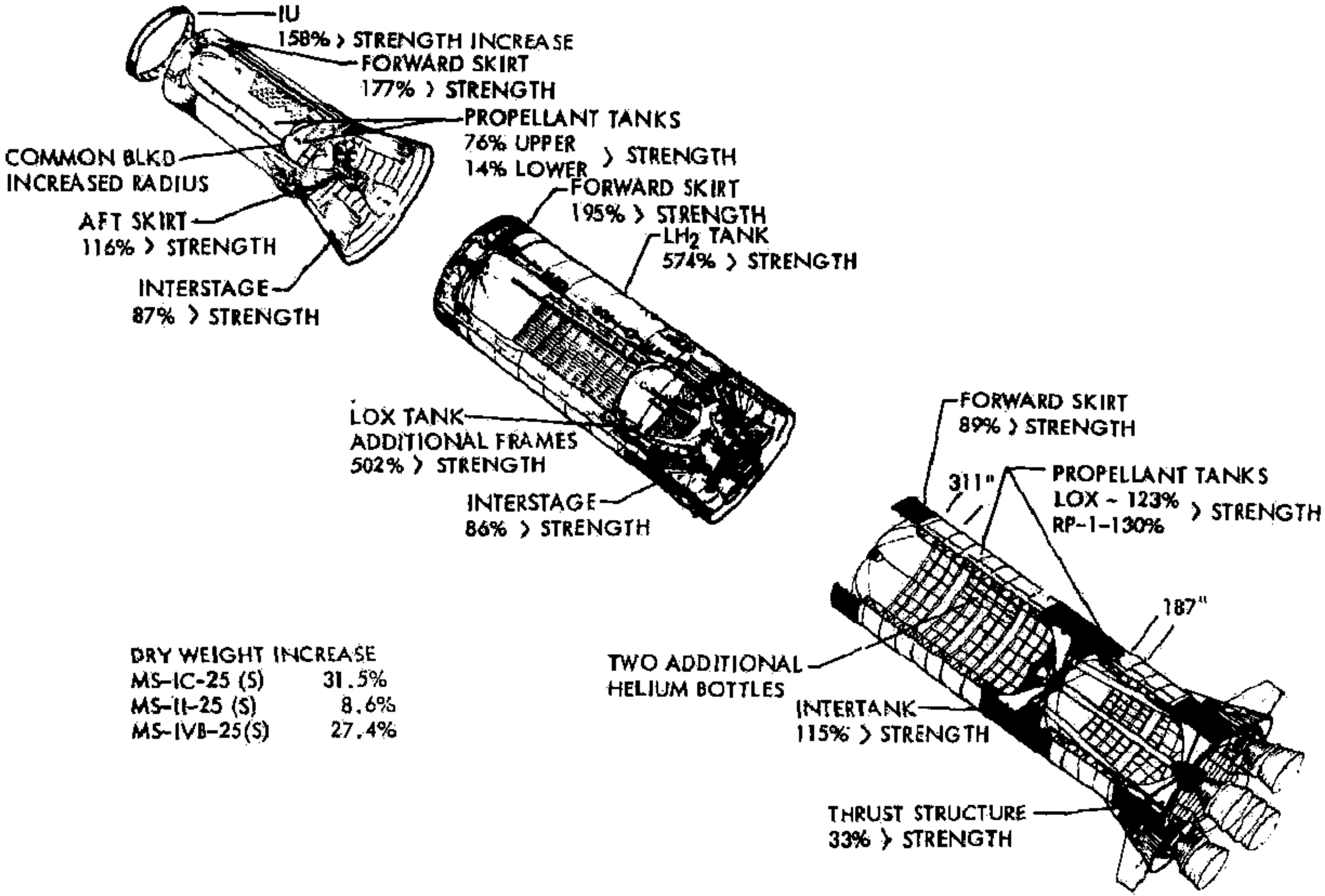


FIGURE 6-4 ACOUSTIC ENVIRONMENT AND STRUCTURAL LOADS



DRY WEIGHT INCREASE	
MS-IC-25 (S)	31.5%
MS-II-25 (S)	8.6%
MS-IVB-25(S)	27.4%

FIGURE 6-5 SAT-V-25(S) VEHICLE IMPACT

6.3 RESOURCES

The present stage, engine, and I.U. manufacturers were assumed to be contractors for the modified vehicle. A dynamic test vehicle, structural test components, and two R&D flights for man-rating are required. A new dynamic test stand will be required for test of this vehicle since its launch weight exceeds present Saturn V stand capability. The S-IC stage of the dynamic test vehicle will be refurbished after test and used as a flight article. The 156-inch solid rocket motors with their thrust vector control system must be developed and qualified for this application

A production rate of six vehicles per year for a period of five years was utilized to assess the production impact.

MS-IC-25(S)

First stage length increase coupled with the more severe structural loads require changes in skin gage, stiffener spacing, and frame chord area. These changes necessitate revisions in manufacturing tools. New tools are required for solid motor attachment structure manufacture. The addition of solid motor functions (ignition, separation, instrumentation) necessitates changes and additions to first stage electrical cable manufacturing boards.

The longer, heavier tanks of the MS-IC-25(S) cannot be hydrostatic tested in the present Michoud VAB position. A new stand will be needed and can be located in a presently unused position in that building. The final assembly and tank assembly stations also must be modified for increased stage length

To introduce the new configuration, without factory modification downtime, an additional tank assembly station and more storage space must be added in the Michoud factory.

The two presently unused stage test positions must be modified to accept the longer S-IC stages. Cables will be installed to the present test control stations and computers. Some new test equipment (solid motor simulators) and minor modification of existing equipment is required.

Modification of the S-IC test firing stands at MTF and MSFC are required only because of the increased stage length and associated propellant capacity.

A minor modification to the stage transporter cables and steering potentiometer will adapt it for the new stage length. However, the additional stage weight exceeds the forward (thrust structure) carriage capability. Two new wheel dollies must be added to the forward carriage.

Both shuttle and sea going barges must have their decks strengthened and supports and tie downs relocated for the longer heavier stage.

For the solid rocket motors, a development program of ten firings is assumed. Minor facilities are required but new tools, jigs and fixtures are a major item. New test and checkout and transportation equipment is also included. Rail transportation from manufacturer to KSC is assumed.

MS-II-25(S)

Manufacturing requirements for the MS-II-25(S) stage are defined by the schedule delivery dates and the stage structural modifications. A separate stage will be manufactured to be utilized for both static structures test and for stage dynamic test. Delivery of this static/dynamic (S/D) stage requires that the standard S-II production be accelerated to accumulate sufficient stages to maintain a constant delivery rate at one stage every two months.

The revised structural design will require modification of the fabrication and assembly tools for the forward and aft skirts, LH_2 tank walls, interstage and aero-fairings. The Seal Beach facilities require a minimum of modifications; the major work required is modification to the structural test tower for the increased test loads. Some handling equipment at Tulsa and Seal Beach will require modification as a result of the increased stage weight.

The current S-II program transport equipment and vehicles are compatible with the MS-II-25(S) stage design; no modifications would be required to handle the additional stage weight.

MS-IVB-25(S)

The elongated tank of the MS-IVB required by this vehicle has a significant impact on resources. The standard S-IVB facilities for manufacturing, assembly, test and checkout will need modification to accommodate this larger, heavier stage. Additional machine tools and space are required for the detail part manufacturing. The most significant area is the skin mills for machining the tanks. The 24 to 36 month delivery time for these machines make early delivery of modified stages difficult. The assembly and checkout towers must be reworked to increase their vertical capacity. Welding torches, platforms, and stage interfaces must be relocated and adapted to the new vehicle. A complicated scheduling problem exists to provide time to modify the facilities with minimum interference with delivery of standard stages. This requires accelerating the production rate of the standard stages and storing them for delivery on their normal shipping date. This could overload some of the checkout towers and require either extensive overtime or additional facilities and GSE. The static firing at Sacramento requires modifying the test stand. A streamlining of the test procedure is recommended as a means of significantly reducing the cost of these modifications. The streamlining would permit making the post-firing checkout in the test stand and eliminate modifying the vertical checkout laboratory cells.

The transporting equipment will require some redesign and all shipments will be by water since the present Super Guppy aircraft cannot carry the elongated stage.

Launch Facility and Launch Operations Impact

The impact of this vehicle on the launch facility and operations was studied by The Martin Company under separate contract to Kennedy Space Center. This study activity described the minimum impact launch sequences for this vehicle as shown below.

The modified core vehicle will be assembled according to standard procedures in the VAB on a modified mobile launcher (ML) and will subsequently be transported to the pad for attachment of the solid rocket motors. Concurrent with the core vehicle assembly and checkout, the solid rocket motors (SRM) segments and closure assemblies will undergo receiving inspection, component installation and individual checkout in a new mobile erection and processing structure (MEPS) at a remote site. After the liquid core vehicle on the mobile launcher has been secured to the launch pad, the MEPS with inspected segments and pre-assembly closures for all four of the solid rocket motors will move to the launch pad and will be mated with the mobile launcher and ground structure for transfer operations of the solid rocket motor segments (see Figure 6-6). Two cranes mounted on the MEPS will be used to lift and attach the aft solid rocket motor closure (with the pre-assembled aft attachment skirt) to the liquid core. Assembly of two SRMs will be accomplished concurrently. The three center segments and the forward closure will then be stacked on top of each of the aft closures. This procedure will be duplicated for assembly and mating of the remaining two solid rocket motors. After assembly is made and alignment of all four SRMs is completed, the MEPS will then be transported back to its parking position. From this point on, the launch operations proceed in a manner similar to those for the Saturn V vehicle with the exception of the added operations for integrated solid rocket motor checkout and for solid rocket motor arming.

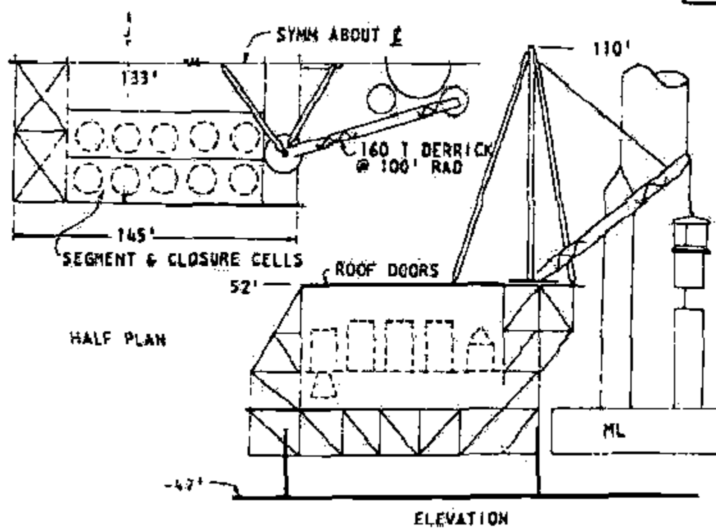


FIGURE 6-6 MOBILE ERECTION AND PROCESSING STRUCTURE

Introduction of the 156-inch solid rocket motors increases pad occupancy time from 58 to 70 days. However, emergency take-down can be accomplished in 39 hours, well within the 72-hour hurricane warning time.

The existing vertical assembly building with the work platform locations altered can be utilized.

Modifications at the launch pad include reinforcement of the mobile launcher support piers and pad structure and the provision of heat shields for pad mounted equipment and structure, new flame deflectors and improved flame deflector anchorage, flame protection for flame trench walls, auxiliary exhaust deflector shields and increased high pressure gas and propellant storage capabilities. Additional quantity and flow rates of industrial water will be required with increased pumping capacity and upgrading of the hydromatic systems. The water mains serving the pad area are adequate without modification. Existing electrical power and communications are satisfactory.

A solid rocket motor inert components building must be provided. A mobile erection and processing structure (MEPS) must be provided with parking position and additional crawler transporter road way for access.

The mobile service structure (MSS) will require a height extension to a level of 396 feet to permit work platforms to be raised to the required service levels. This will require increased structural reinforcement and increased elevator runs. The cantilever framing which supports the platforms in the vicinity of the solids must be reworked to increase the lateral clearance. It is not possible to accomplish these required modifications in the five-month time span between the last Saturn V launch and the first MLV-SAT-V-25(S) R&D launch; therefore, a new MSS incorporating the above changes must be built.

One new and one modified mobile launcher (ML) are required to satisfy the launch rate and program phase-in requirements for this vehicle. The principal modifications involve relocation to higher levels of all umbilical arms, shielding of the front umbilical face, increased elevator runs, an enlargement of the aspirator hole, from 45 feet to 55 feet, strengthening of the ML platform structure, replacement of the existing vehicle support arms and relocation of equipment in the umbilical tower and mobile launcher platform. Protection from exhaust impingement on the bottom of the ML will be required because of the exhaust plume spillover from the flame trench.

The crawler transporter which will be used to transport the mobile launcher and MEPS will require uprating by approximately 11 percent to handle the increased loads caused by the heavier MSS. These modifications will include structural beefup and a new, more powerful steering system.

Schedules

Within the study groundrules and after an analysis of the required design and development plans and manufacturing impact, a schedule for development and production of this vehicle was prepared. See Figure 6-7. This schedule shows that the MLV-SAT-V-25(S) first flight vehicle can be available 42 months after hardware authority to proceed (ATP).

Costs

A vehicle cost summary is shown in Table 6-III.

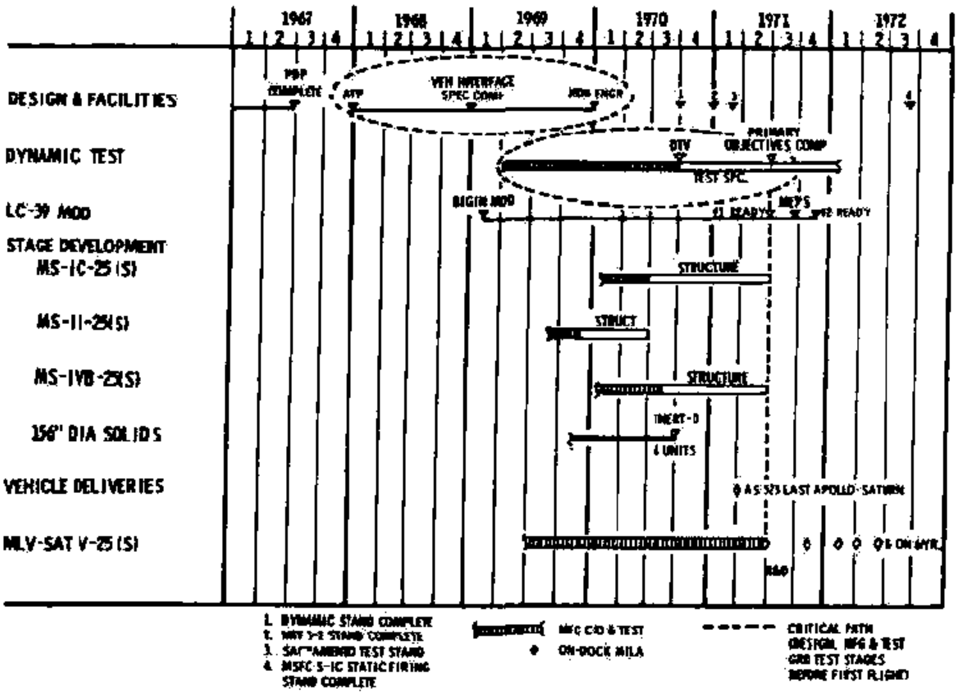


FIGURE 6-7 SAT-V-25(S) LAUNCH VEHICLE DEVELOPMENT AND DELIVERY PLAN

COST - DOLLARS IN MILLIONS	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE					
Boost Assist		96.7		440.5	537.2
S-IC Stage	77.1		640.1	273.8	991.0
S-II Stage	58.7		513.8	188.5	761.0
S-IVB Stage	72.0		359.3	37.7	469.0
Instrument Unit			136.1		136.1
LAUNCH VEHICLE TOTAL	207.8	96.7	1649.3	940.5	2894.3
GROUND SUPPORT EQUIPMENT					
Boost Assist		5.5			5.5
S-IC Stage	17.0		26.4		43.4
S-II Stage	11.5		57.6		69.1
S-IVB Stage	33.7		48.5		82.2
GSE TOTAL	62.2	5.5	132.5		200.2
FACILITIES					
S-IC Stage	19.6				19.6
S-II Stage	.7				.7
S-IVB Stage	5.4		5.7		11.1
Launch Vehicle - KSC	192.5		727.2		919.7
Launch Vehicle - Other	10.0				10.0
FACILITIES TOTAL	228.2		732.9		961.1
SYSTEMS ENGINEERING & INTEGRATION	2.7		475.8		478.5
LAUNCH SYSTEMS TOTAL	500.9	102.2	2990.5	940.5	
	603.1		3931.0		4534.1
			R&D FLIGHTS (2)		324.2

TABLE 6-III SAT-V-25(S) COST SUMMARY

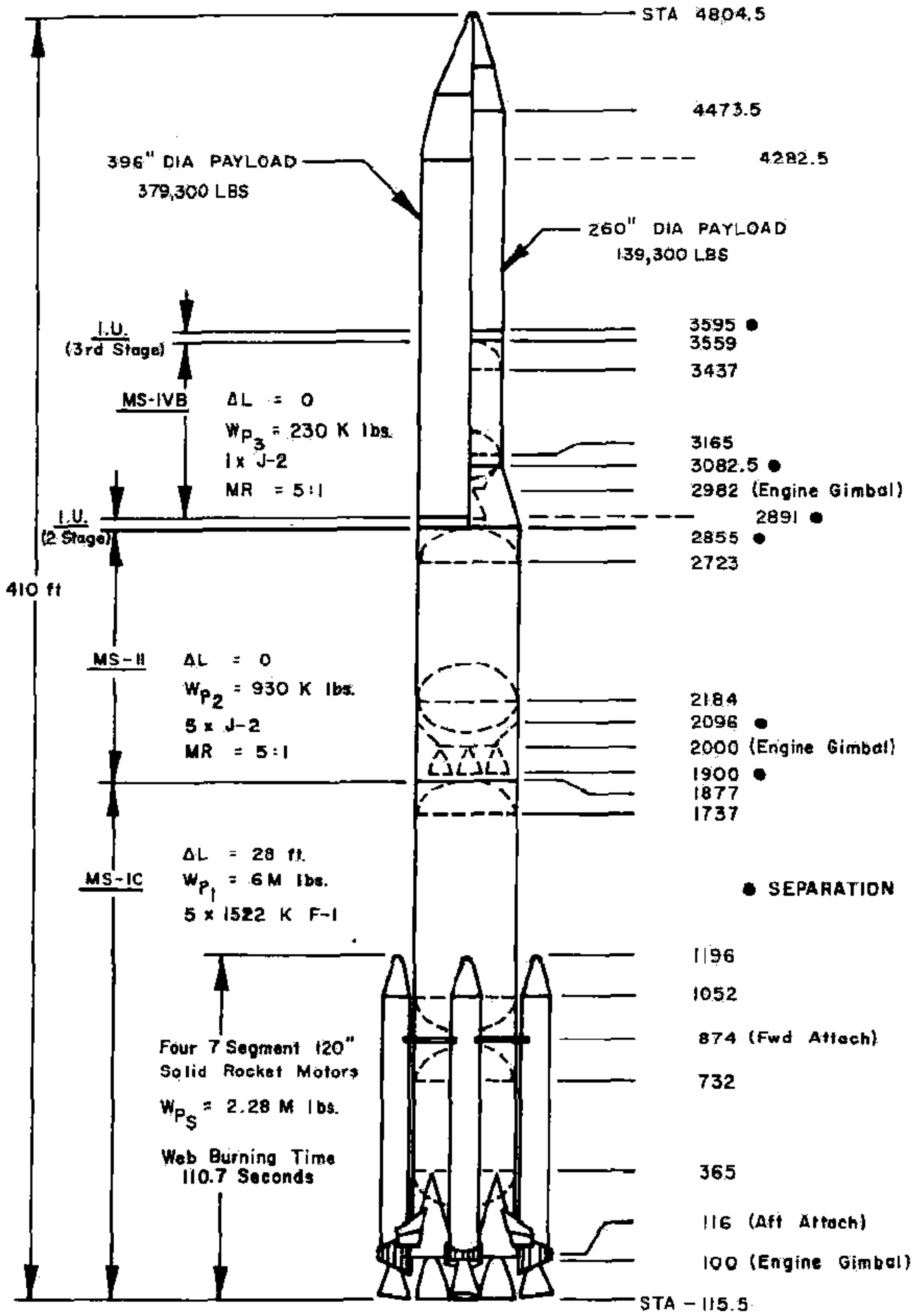


FIGURE 7-1 MLV-SAT-V-4(S)B BASELINE VEHICLE

7.0 MLV-SAT-V-4(S)B

The SAT-V-4(S)B (See Figure 7-1) is a Saturn V vehicle with lengthened first stage, adapted to accept attachment of four 120-inch diameter solid motors.

The vehicle as defined in the trade study activity and studied in detail in the Phase II activity is a feasible and cost effective configuration and is, therefore, a logical candidate to provide payloads in excess of those currently available with the Saturn V vehicle. No major problem areas were identified for either development or production of this vehicle.

7.1 CONFIGURATION SELECTION (PHASE I)

By varying the number of solid rocket motor segments (and thus solid propellant weight), and considering both optimized length and fixed length core stages, a number of related SAT-V-4(S)B vehicles were evolved. Payload capability and costs were established for these vehicles in order to choose one arrangement for more detailed analysis.

7.1.1 Candidate Configurations

For the trade study both two and three stage operation was considered. The vehicle height was fixed at 410 feet for both the two and three stage configurations. Propulsion and engine type for all stages was fixed to correspond to the baseline AS-516 vehicle. Varying weights of propellant and corresponding stage lengths were studied for all stages. Four 120-inch solid propellant rocket motors were attached to the vehicle for thrust augmentation. The number of segments in the solid motors was varied between five and seven. The characteristics of each solid motor were specified by MSFC. Significant solid motor parameters are shown in Table 7-I. The vehicle liftoff weight was varied to maintain a liftoff thrust-to-weight of approximately 1.25.

7.1.2 Trade Studies

Figure 7-2 is typical of the parametric performance data prepared for the trade study. Figure 7-2 illustrates the net payload versus the number of segments in the 120-inch motors for the three-stage vehicle. This curve shows two conditions, optimized first-stage propellant weight with the upper stage propellant

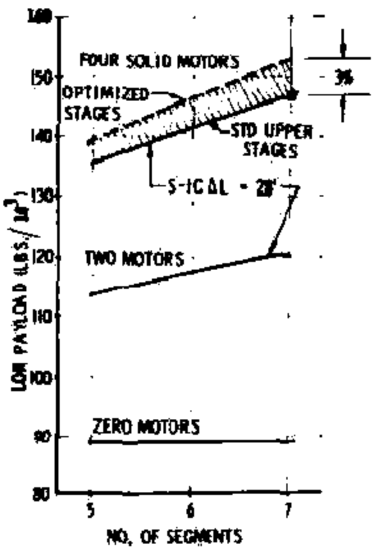


FIGURE 7-2 TRADE STUDY PERFORMANCE DATA

TABLE 7-I SOLID MOTOR CHARACTERISTICS

	UA-1205 (5-Segment)	UA-1206 (6-Segment)	UA-1207 (7-Segment)
Motor diameter	120 in	120 in	120 in
Average sea level I_{sp} (approximately)	230 sec	234 sec	232 sec
Sea level action time total impulse	97,132,000 lb-sec	116,080,000 lb-sec	116,080,000 lb-sec
Total propellant weight	421,480 lb	495,018 lb	571,324 lb
Total motor weight	489,519 lb	570,690 lb	655,108 lb
Characteristic velocity	5,170 ft/sec	5,170 ft/sec	5,170 ft/sec
Overall motor length	1,015.8 in	1,127 in	1,305 in
Maximum chamber pressure	690 psia	680 psia	760 psia
Initial nozzle throat area	1,116.3 sq in	1,301 sq in	1,301 sq in
Initial nozzle exit area	8,930 sq in	8,922 sq in	12,490 sq in
Initial expansion ratio	8.0	6.86	9.6
Nozzle length	114.0 in	111.6 in	175.4 in
Nozzle weight	7,705 lb	7,705 lb	9,574 lb
Burn action time	112.0 sec	107.8 sec	119.6 sec

weights fixed, and propellant weights for all stages optimized. For the optimized vehicles, the S-IVB stage would have to be lengthened aproximately 14 feet while the S-II stage remains at its standard length and the S-IC stage is increased in length by about 28 feet. A similar study of two-stage vehicles shows the optimum core vehicle to be basically a standard S-II stage and a 28-foot longer MS-IC stage.

The Figure 7-2 data demonstrates that:

a. Payload increases approximately 4.5 percent with each additional solid motor segment (increased solid propellant weight).

b. Payload gains of approximately 3 percent accrue by optimizing propellant weights in all stages.

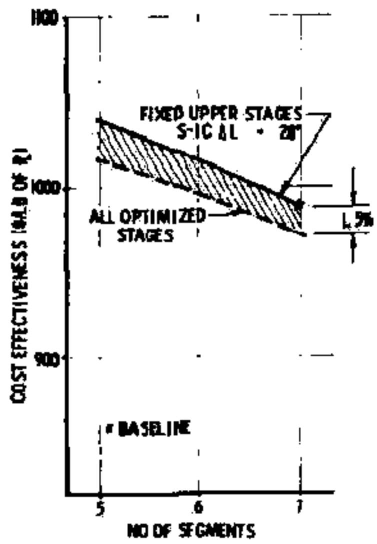


FIGURE 7-3 TRADE STUDY COST DATA

Figure 7-3 compares the payload cost efficiency of using various solid motor weights (numbers of segments) with either fixed or optimized core stages. It should be noted that the major difference between optimized vehicles and fixed upper stage vehicles is the transfer of 100,000 pounds of propellant from the S-IC to the S-IVB stage. The resultant increase in S-IVB cost only allows a 1.5 percent improvement in cost efficiency even though payload is improved 3 percent.

Since study ground rules specified that upper stage modifications be minimized, a fixed length upper stage vehicle was chosen for the next phase of study. The selected vehicle is indicated on Figures 7-2 and 7-3 as "Baseline."

7.2 DESIGN STUDY VEHICLE (PHASE II)

The single SAT-V-4(S)B selected during the Phase I activity was defined in detail, its capabilities and characteristics were determined and its resource requirements established.

7.2.1 Vehicle Description

The baseline SAT-V-4(S)B vehicle is shown in Figure 7-1. It incorporates standard length upper stages and a 28-foot longer first-stage augmented by four seven-segment 120-inch rocket motors. The solid motors, as designated by MSFC, conform to preliminary designs developed by United Technology Center for Titan III-C applications. Each motor has an initial sea level thrust of 1.4 million pounds and a propellant weight of 579,000 pounds. Each motor has a liquid injection (N_2O_4) thrust vector control system to augment the control capabilities of the gimbaled F-1 engines during flight through the max q regime. The liquid core stages of SAT-V-4(S)B are equipped with standard F-1 and J-2 engines. The first stage of the vehicle is rotated 45 degrees from its position in the standard Saturn V configuration to minimize the impact on launch facilities and operations. The second stage is standard S-II length with a propellant capacity of 930,000 pounds. The third stage (for three-stage applications) is standard S-IVB length with 230,000 pounds propellant capacity. Since study funds and timing were limited, the desirable increased length S-IVB was not studied and the S-II stage for MLV-SAT-V-25(S) was used directly on SAT-V-4(S)B.

7.2.2 Design Study Results

The SAT-V-4(S)B two-stage payload capability to 100 nautical miles orbit is 379 thousand pounds and its 72 hour lunar injection three-stage capability is 139 thousand pounds.

Use of this vehicle was also considered for application where the core vehicle (liquid stages without solids) could be flown by itself or with only two strap-on solid motors. The payloads identified for these alternates are as shown in Table 7-II.

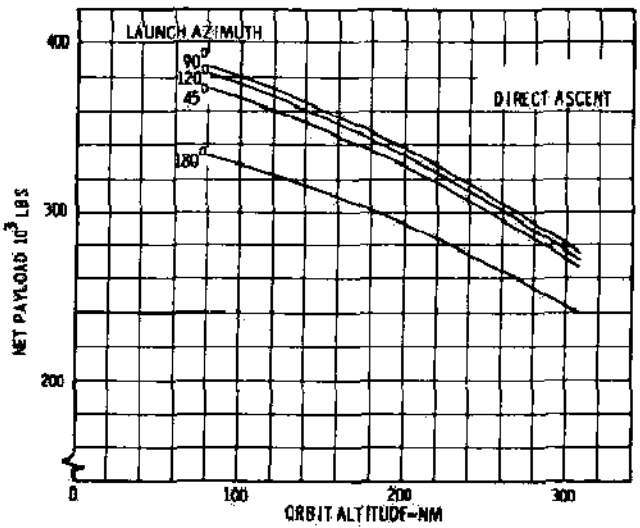


FIGURE 7-4 ORBIT ALTITUDE - AZIMUTH PAYLOAD CAPABILITY

Additional studies identified information useful for mission planning. Payloads available for various orbital altitudes between 80 and 300 nautical miles and launch azimuths between 45 degrees and 180 degrees are shown in Figure 7-4. Polar and near polar orbit payloads are shown

TABLE 7-II PAYLOAD CAPABILITY

	NET PAYLOAD (LB) TWO-STAGE 100 NM Orbit	NET PAYLOAD (LB) THREE-STAGE 72-hour Lunar Injection
Core Vehicle Without Solid Motors		
$T_o/W_o = 1.25$ $W_{P1} = 7,387,368$ lb	243,512	89,444
$T_o/W_o = 1.18$ $W_{P1} = 4,740,350$ lb	251,683	92,445
Core Vehicle With Two Solid Motors		
$T_o/W_o = 1.25$ $W_{P1} = 5,358,842$ lb	320,725	117,805
$T_o/W_o = 1.18$ $W_{P1} = 5,855,789$ lb	330,920	121,549
Core Vehicle with Four Solid Motors		
$T_o/W_o = 1.25$ $W_{P1} = 6,000,000$ lb	379,300	139,300

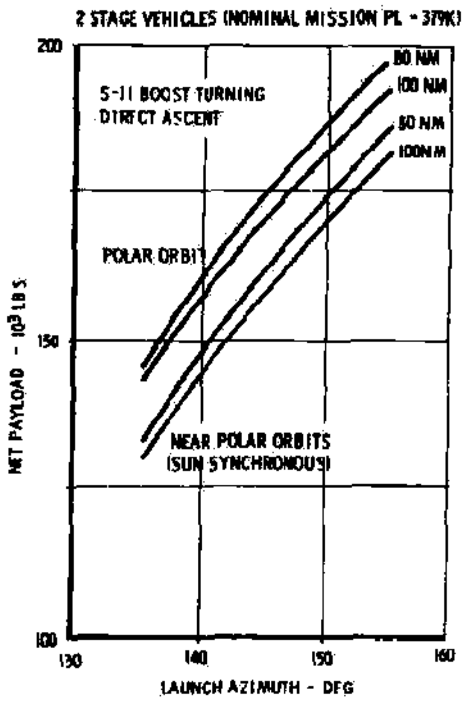


FIGURE 7-5 POLAR & SUN SYNCHRONOUS ORBIT PAYLOAD CAPABILITY

in Figure 7-5. Three-stage mission payload capability for a 24-hour sun synchronous orbit, and more generally, payload as a function of the specific energy parameter are shown on Figure 7-6.

Significant load criteria and other data pertinent to vehicle design are shown on Table 7-III with comparative Saturn V values. Although max dynamic pressure (q) and acceleration are reduced, compared to SAT V, the 410-foot vehicle height coupled with the 33-foot diameter two-stage payload shape has a large impact on structural design requirements.

The first stage control requirements of the SAT-V-4(S)B necessitate additional control beyond the present fins and gimbal capability of the F-1's using the current attitude, attitude rate control system. Alternate control mode studies showed that approximately a 5 percent reduction in maximum bending response could be expected if an angle of attack feedback control mode were employed. The use of the liquid injection thrust vector control on the solid motor is required for 30 seconds near maximum q time of flight, as shown in Table 7-III. Since the solid motor TVC requirements are less than the Titan III-C, the liquid injectant tanks will be off-loaded to carry only the required fluid. The use of enlarged fins in lieu of solid motor TVC was also considered. This analysis indicated that the fin size for the baseline SAT-V-4(S)B vehicle would have to be double that required for AS-516 (150 square feet

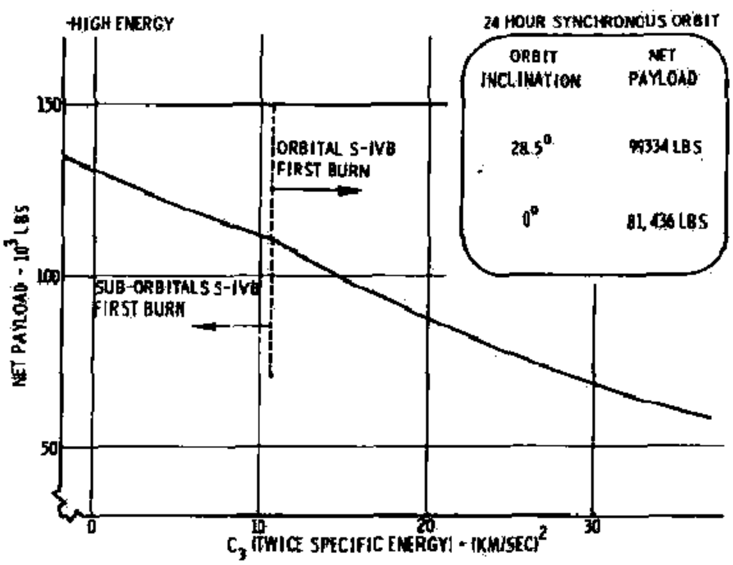


FIGURE 7-6 THREE-STAGE HIGH ENERGY MISSION CAPABILITY

per fin versus 75 square feet per fin). Wind tunnel tests will be required to substantiate this analysis. Because of the rotation of the first stage 45 degrees, the flight control signal must be modified to compensate for the rotation.

	-4(S)B	SAT V
LOAD CRITERIA		
MAX q (LBS/FT^2)	604	766
g's AT MAX q_α	1.83	1.954
HEIGHT (FT)	410	363
CONTROL		
MODE	GIMBALED F-1'S PLUS N_2O_4 LITVC ON SOLIDS	GIMBALED F-1'S
SOLID MAX. DEFLECTION ANGLE	2.3° PER MOTOR	N/A
SOLID TVC OPERATING	70-100 SEC	N/A
HEATING		
AERODYNAMIC (AHI) FT-LB/FT^2	653,000	792,000
BASE MAX. TEMP.	2200°F	1900°F
OTHERS	MS-IC-4(S)B ROTATED 45°	

BASELINE 516 WITH T_0/W_0 - 1.25

TABLE 7-III SIGNIFICANT LOAD CRITERIA

Control studies of vehicles defined in the special study on payload sensitivities below indicated that for reduced payload lengths for the two-stage vehicle (payload densities of 5 pounds/feet^3 and greater) and for nominal payload lengths for the three-stage vehicle, that no additional control capability beyond nominal for AS-516 is required for the SAT-V-4(S)B.

Studies were conducted to determine the payload envelope and corresponding wind limitations for the vehicle with: (1) no or minimum upper stage modifications; and (2) the full structural modifications as indicated for the MLV-SAT-V-4(S)A vehicle in the previous fiscal year 1964 contracted study effort. Typical results are summarized on Figures 7-7 and 7-8. This data shows that for the nominal payload lengths (i.e., 159 feet for the two-stage payload and 101 feet for the three-stage payload) that there is basically no possibility of flying either the two- or three-stage vehicle from December through March unless modifications are made to the upper stages. With minimal modifications, the availability for launch during these months is approximately 20 percent. This data further shows that, with no modifications to the upper stages, a 95 percent or better launch availability can be obtained for every month of the year by reducing the two-stage payload length to approximately 70 feet (payload density equals 12.5 pounds/foot^3). The data shows, however, that three-stage applications will not have 95 percent availability with any length payload during February and March unless the upper stages are modified. A 50 percent or better availability for a three-stage payload length of 50 feet is possible for every month of the year with no upper stage modifications.

The addition of more fuel in the first stage and the four solid motors increases the 0.4 psi overpressure distance to a value greater than the distances between Pad A and Pad B on Launch Complex 39. Waivers for this distance will be required for joint usage of these pads when either

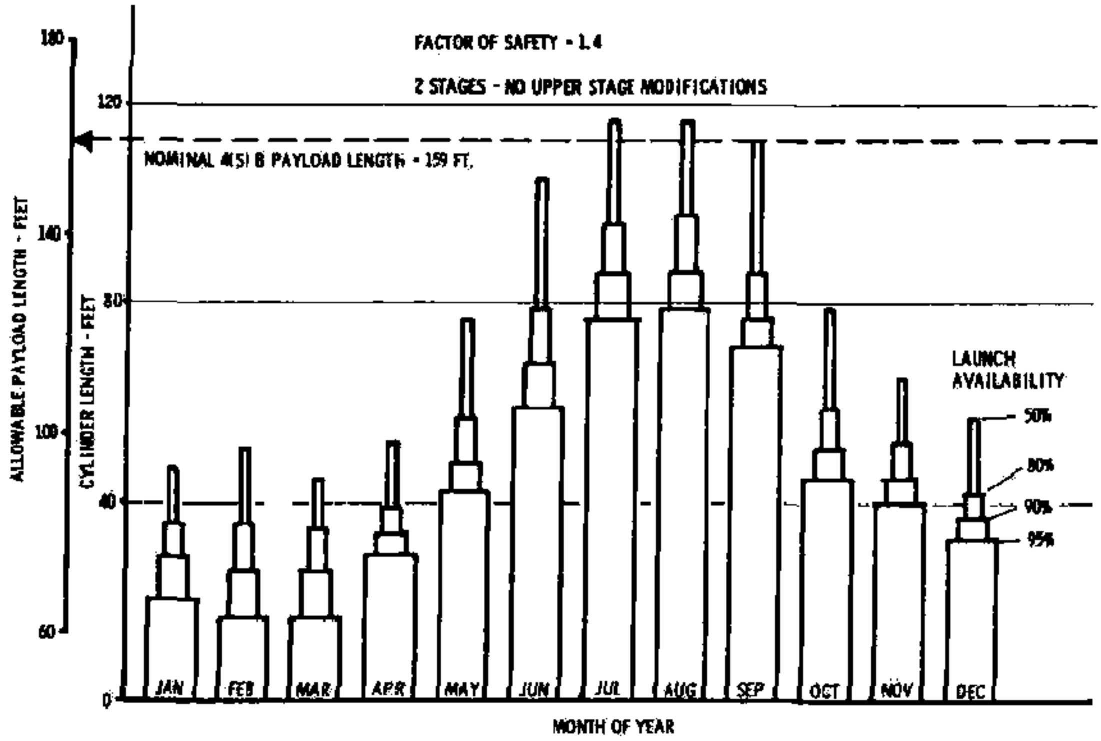


FIGURE 7-7 TWO-STAGE WIND/PAYLOAD SENSITIVITY

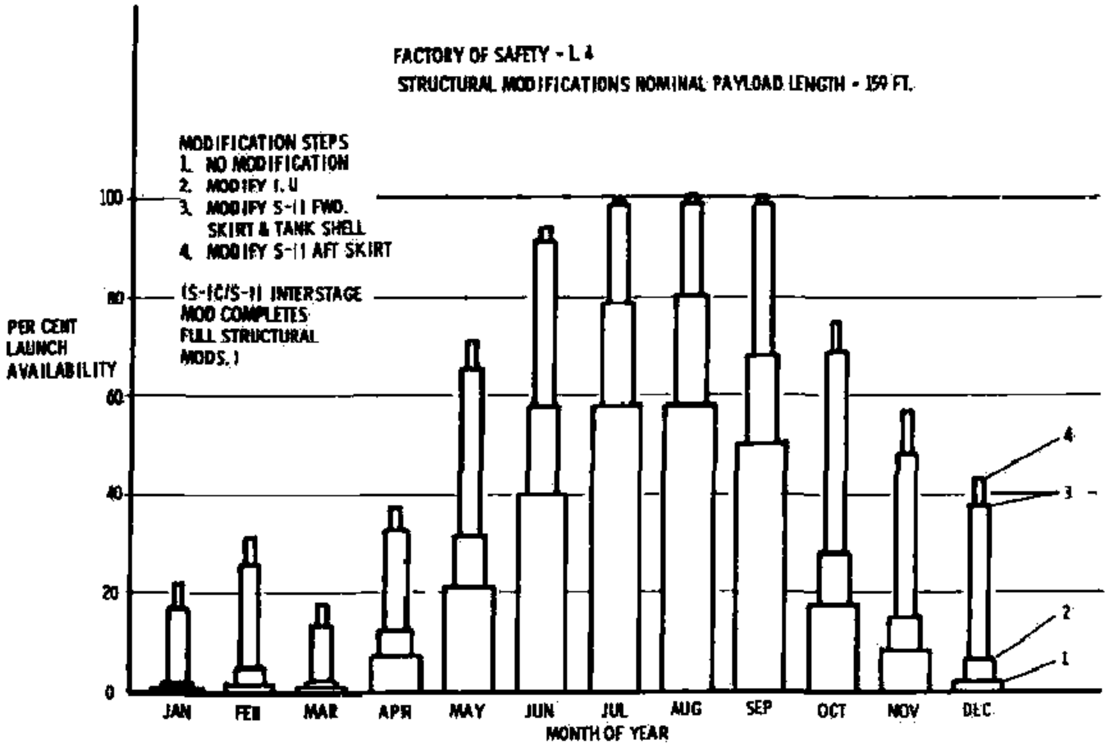


FIGURE 7-8 THREE-STAGE WIND/PAYLOAD SENSITIVITY

pad contains a fueled core vehicle with the solid motors attached. The acoustic level will require protection of personnel during launch.

Flight and crew safety are satisfactory for this vehicle. Communications for some stations may possibly be "blacked out" due to the exhaust plume interference. Other stations, however, will have clear communications during these periods and can provide continuous communications.

Assuming an Apollo payload and launch escape system, the abort lead time for this vehicle was developed for comparison with that of the Saturn V. Using a solid motor TNT equivalency of 100 percent, an abort lead time of 2.76 seconds is required for this vehicle versus 2.19 seconds for the Saturn V. Abort lead time is that time prior to explosion of the vehicle that the launch escape system firing pulse must be initiated. Other studies showed that no problems for abort are anticipated because of separation dynamics.

Aerodynamic heating for the MLV-SAT-V-4(S)B is significantly lower than the Saturn V. The aerodynamic heating indicator (AHI) at 653,000 foot-pounds per square foot for the SAT-V-4(S)B is 18 percent lower than the nominal SAT V AHI value of 792,000 foot-pounds per square foot and 30 percent below the maximum AHI value for the Saturn V of 924,000 foot-pounds per square foot. The shock wave from the solid motor nose cap may impinge on the first stage near the intertank and local insulation may be required.

The base heating environment is more severe for the MLV-SAT-V-4(S)B than for the Saturn V due to the solid motor exhaust plumes. However, heat shield materials can withstand the 2200 degree F temperatures anticipated successfully. The aft solid motor attachment skirt will reach 2480 degrees F. Insulation protection on the aft skirt will be required. A base heat shield will also be required for each of the solid motors.

The reliability of the two- and three-stage MLV-SAT-V-4(S)B vehicle is 0.984 and 0.967, respectively, as compared to .990 and .980 for the baseline AS-516. The lower reliability can be attributed to the modifications to the stages and the addition of the strap-on solid motors.

Separation of the solid motors from the core vehicle can be accomplished satisfactorily using explosive separation devices and the present Titan IIIC separation rocket motors.

Vehicle combined loads and acoustics are illustrated in Figure 7-9. The design loads are approximately 60 percent higher than those for the standard Saturn V. The acoustic specification limits are exceeded at several locations on the first stage. Acoustic requalification of approximately 70 percent of the acoustically sensitive components on this stage will be required.

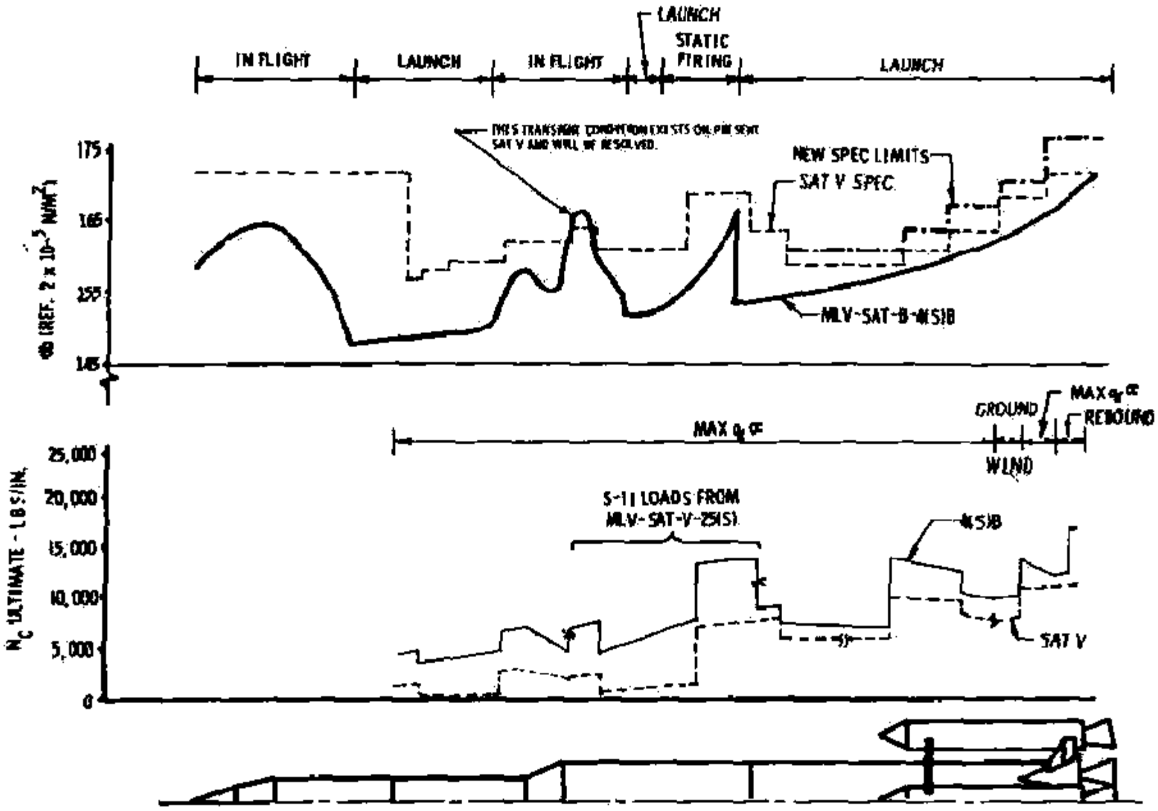


FIGURE 7-9 ACOUSTIC ENVIRONMENT AND STRUCTURAL LOADS

Major vehicle changes including the impact of structural load increases are summarized on Figure 7-10. Dry weight increases are also tabulated.

4.3 RESOURCES

The present stage and I. U. vendors were assumed to be contractors for the modified vehicle components. A dynamic test vehicle, structural test components and two R&D man-rating flight vehicles are required.

The existing Dynamic Test facility will be employed to test the MLV-SAT-V-4(S)B-D. Necessary modifications include relocation of *platforms for increased vehicle length and an additional hydrodynamic* support.

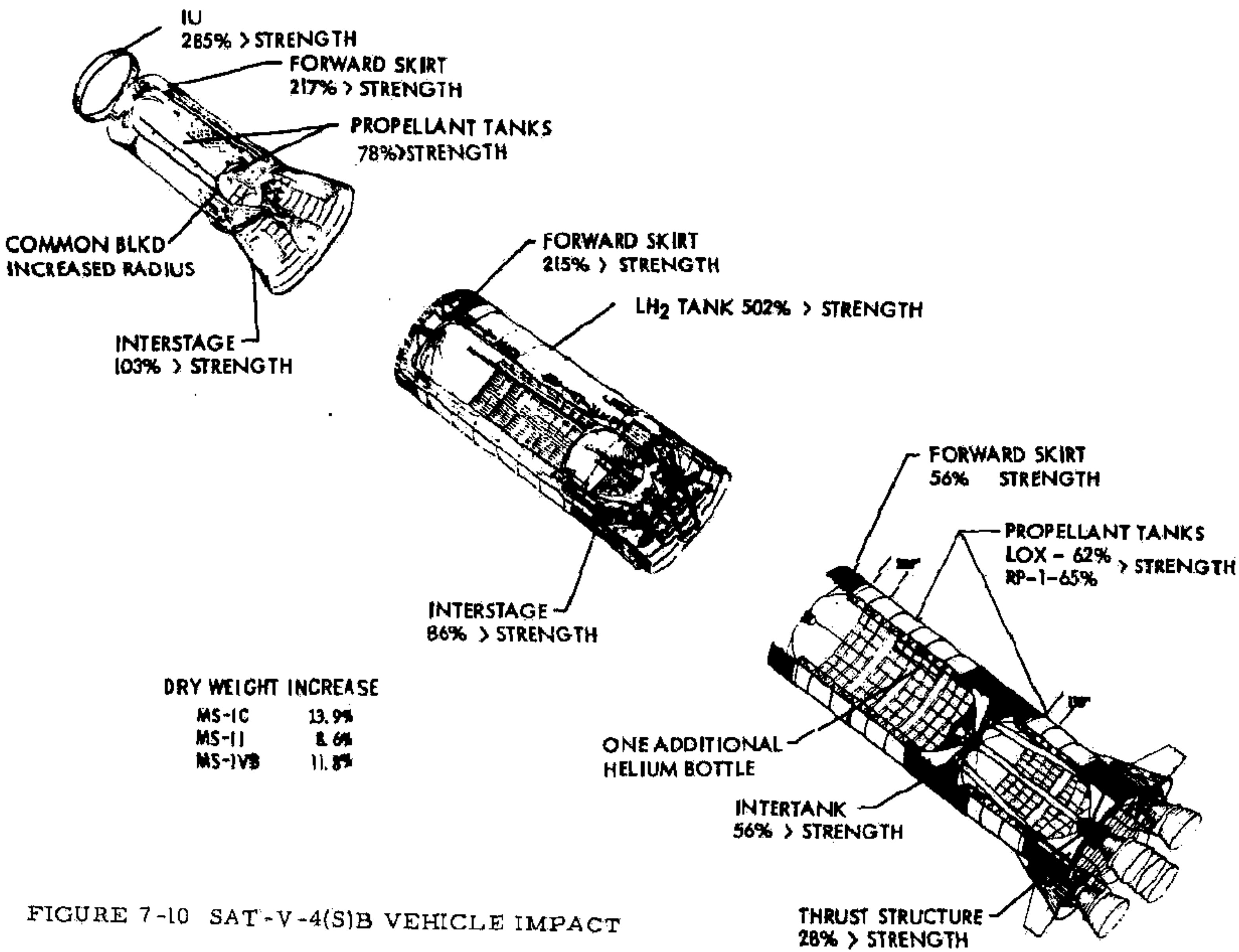


FIGURE 7-10 SAT-V-4(S)B VEHICLE IMPACT

The S-IC stage of the dynamic test vehicle will be refurbished after test and used as a flight article.

The 120-inch solid rocket motors with their thrust vector control system are assumed to be flight qualified in the Titan III-C vehicle systems.

A production rate of six vehicles per year for a period of five years was utilized to assess production impact.

MS-IC-4(S)B

The major impact of the first stage changes on Michoud facilities will be due to the added SRM functions and manufacture of the SRM aft skirt structure. Additional assembly equipment, checkout and handling, and transportation equipment will be required. The aft attachment structure is a maraging steel structure requiring boring machines, welding fixtures, and additional welding facility area. The heavier and longer first stage will require rework of much of the existing equipment.

Major tooling and assembly requirements at Michoud include an additional tank assembly station, an additional tank cleaning position, and some additional and modified tooling. Additional warehousing, quality assurance, and receiving inspection areas will be required. These additions provide the capability of introducing the new configuration with minimum downtime.

Modification of the S-IC test firing stands at MTF and MSFC are required only due to increased stage length and propellant capacity. Solid motors will not be fired in conjunction with the stage static test.

The stage transporter and the barges must be modified to accommodate the increased stage length. Additional solid motor handling and transportation equipment will be required.

MS-II-4(S)B

Manufacturing requirements for the MS-II-4(S)B stage are defined by the schedule delivery dates and the stage structural modifications. A separate stage will be manufactured to be utilized for both static structures test and for stage dynamic test. Delivery of this static/dynamic (S/D) stage requires that the standard S-II production be accelerated to accumulate sufficient stages to maintain a constant delivery rate at one stage every two months.

The revised structural design will require modification of the fabrication and assembly tools for the forward and aft skirts, LH_2 tank walls, interstage and aero-fairings. The Seal Beach facilities require a minimum of modification; the major work required is modification to the structural test tower for the increased test loads. Some handling equipment at Tulsa and Seal Beach will require modification as a result of the increased stage weight.

The current S-II program transport equipment and vehicles are compatible with the MS-II-4(S)B stage design; no modifications would be required to handle the additional stage weight.

Due to study funding limitations, a separate MS-II-4(S)B resources study was not made. The MS-II-25(S) stage resource data was used without modification for SAT-V-4(S)B.

MS-IVB-4(S)B

The engineering redesign of the standard S-IVB to convert it to the MS-IVB-4(S)B presents no schedule or technical problems. Strengthening the basic structure of the stage will require a few development and qualification tests. The techniques and procedures required for these tests are similar to many tests conducted on the S-IVB stage. Tools and facilities for performing the tests are readily available and will not present any problem. The fabrication, assembly, checkout, and firing test facilities used for the standard S-IVB can be adapted to the MS-IVB-4(S)B. The machine tool capacity required to produce the standard S-IVB is adequate to produce the same rate of MS-IVB-4(S)B stages. The detail tooling will require numerous minor changes but these present no schedule problems. The assembly and checkout towers can be modified to accommodate the MS-IVB-4(S)B without any schedule complications. However, the 12 per year production rate (six for MLV-SAT-V-4(S)B and six for Saturn IB) taxes the capability of Towers 5 and 6. The Sacramento Test Center facilities are adequate and can be adapted to the MS-IVB-4(S)B without interference with the standard stage delivery rates. The present transportation equipment is adequate for the modified stage though some strengthening will be required on selected pieces of equipment.

Launch Facility and Operations Impact

The modified core vehicle will be assembled according to standard procedures in the VAB on the Mobile Launcher and will subsequently be transported to the pad where the solid rocket motors will be attached. The solid motor segments will be assembled in a Mobile Assembly and Handling Structure (MAHS), at a site to be provided, and transported by this MAHS to the launch pad for subsequent assembly of the solids to the

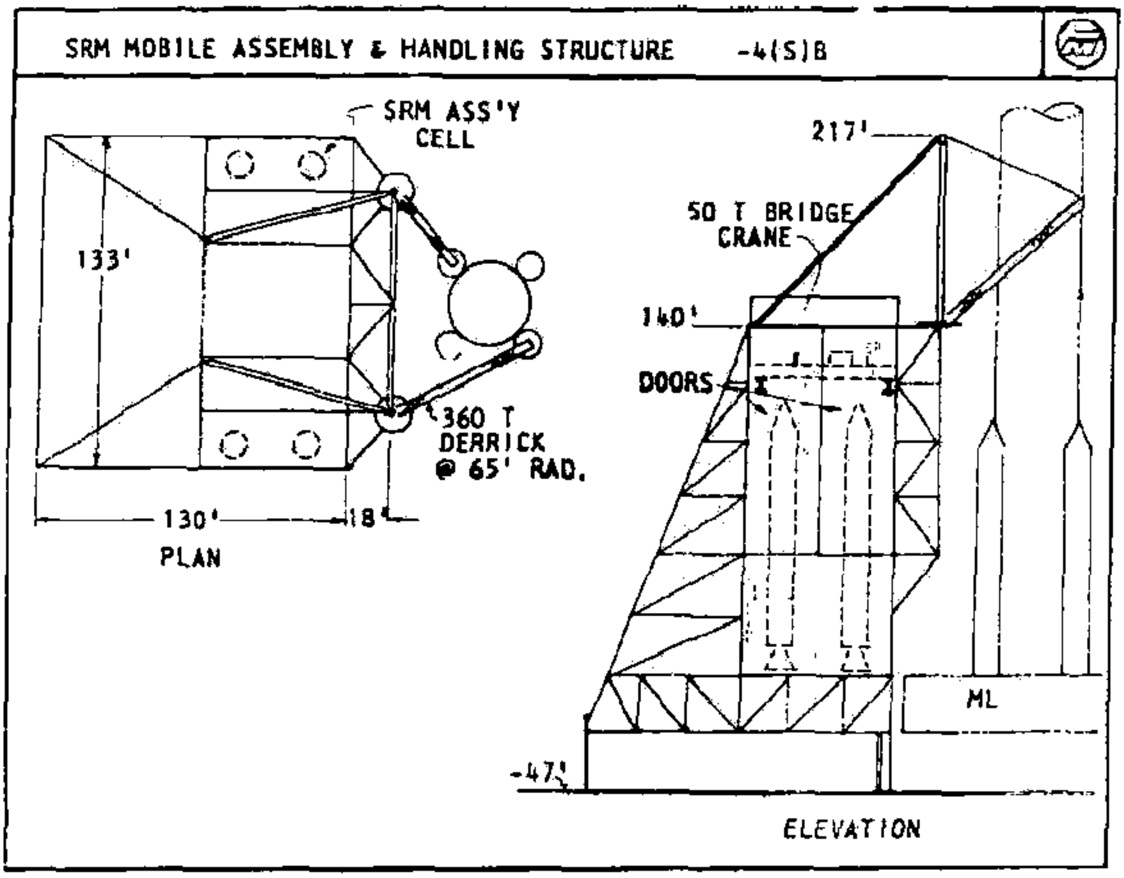


FIGURE 7-11 MOBILE ASSEMBLY AND HANDLING STRUCTURE

core vehicle. The MAHS will mate with the mobile launcher for this assembly operation and handling equipment within the MAHS will be utilized for placement of the solid motors against the core vehicle (see Figure 7-11). After assembly of the solid motors, the MAHS will be removed and replaced by the service tower. Normal operations for the core vehicle will be resumed and additional operations as required for solid motor final checkout and arming will be accomplished.

The existing VAB with work platform locations altered and the existing launch pad and its existing flame tranch can be utilized. The crawler transporter roadways are sufficient for this vehicle with the exception of the requirement for some additional crawler transporter roadways as required for access to the solid motor assembly site. Major impact areas include the development and construction of the MAHS and modifications to the mobile launcher (ML) to increase its deck load capacity, to relocate the swing arms, to relocate the tail service masts and holddown structure, and to enlarge the aspirator hole to allow additional space for the solid rocket motor nozzles. Insulation in selected areas will be required to protect the ML during launch.

The total cost for the modification and additions described is 177.3 million dollars. Of significance is the fact that a major portion of this cost is due to the requirements for a new as well as a modified mobile launcher (ML) and a new as well as a modified mobile service structure (MSS). These new items are forced because of the ground rule which limits the time between the last standard SAT V launch and the first MLV-SAT-V-4(S)B launch to five months. If this time could be extended to preclude these new items, the above cost could be reduced by approximately 80 million dollars.

Schedule

Within the study ground rules and after an analysis of the required design and development plans and manufacturing impact, a schedule for development and production of this vehicle was prepared (see Figure 7-12). This schedule shows that the MLV-SAT-V-4(S)B first flight vehicle for mission applications can be available 41 months after hardware Authority to Proceed (ATP).

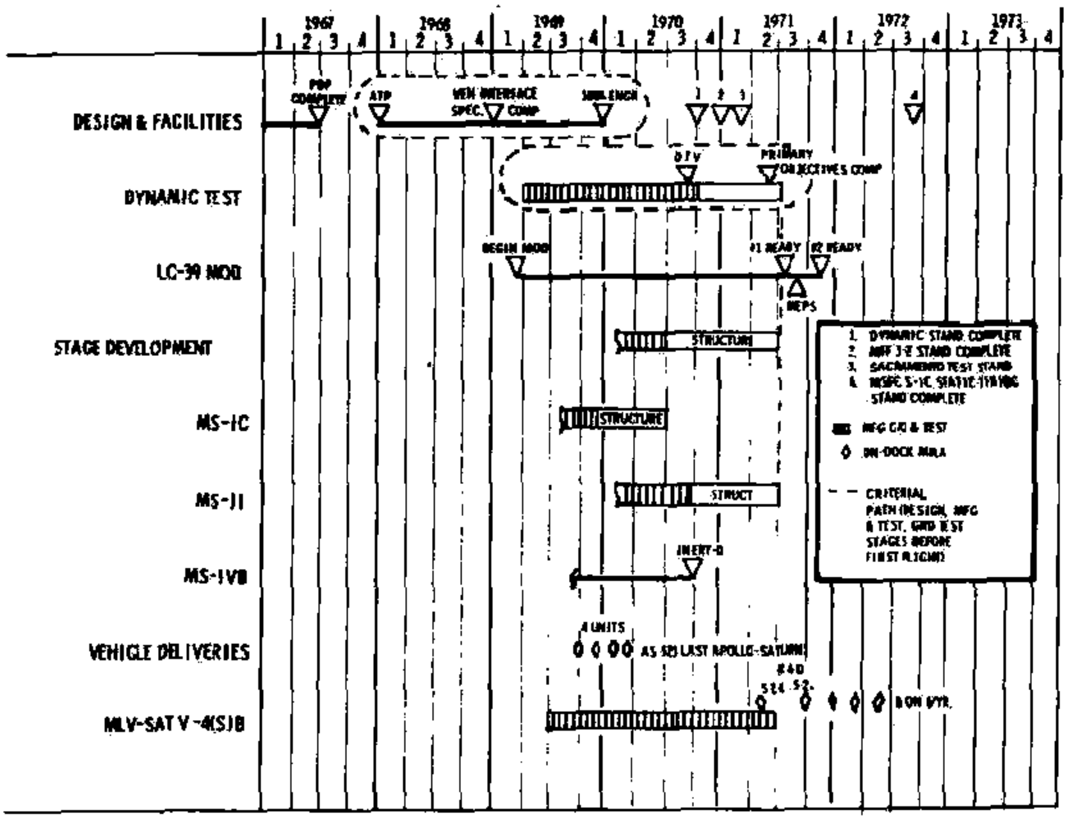


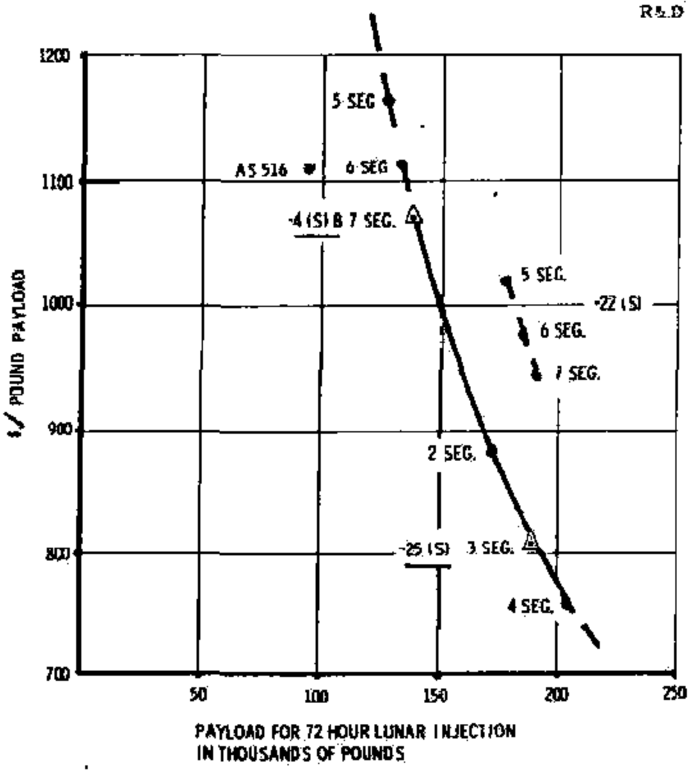
FIGURE 7-12 SAT-V-4(S)B VEHICLE DEVELOPMENT AND DELIVERY PLAN

Costs

A vehicle cost summary is shown in Table 7-IV.

TABLE 7-IV SAT-V-4(S)B COST SUMMARY

COST - DOLLARS IN MILLIONS	DEVELOPMENT STAGE	DEVELOPMENT ENGINE	OPERATIONAL STAGE	OPERATIONAL ENGINE	TOTAL
LAUNCH VEHICLE					
Boost Assist	3.0			319.4	322.4
S-IC Stage	74.0		634.5	273.4	[illegible]
S-II Stage	58.7		513.8	138.5	761.0
S-IVB Stage	47.[illegible]		314.2	37.8	[illegible]
Instrument Unit			136.1		[illegible]
LAUNCH VEHICLE TOTAL	1?3.5		[illegible]	[illegible]	2601.[illegible]
GROUND SUPPORT EQUIPMENT					
Boost Assist		5.5			[illegible]
S-IC Stage	15.[illegible]		26.4		[illegible]
S-II Stage	11.5		57.6		6[illegible].1
S-IVB Stage	16.8		43.5		60.3
GSE TOTAL	43.3	5.[illegible]	127.5		176.[illegible]
FACILITIES					
S-IC Stage	18.0				18.0
S-II Stage	.7				.7
S-IVB Stage	2.0		4.9		6.[illegible]
Launch Vehicle - KSC	177.3		718.0		[illegible]
Launch Vehicle - Other	1.1				1.1
FACILITIES TOTAL	199.1		722.9		922.0
SYSTEMS ENGINEERING AND INTEGRATION	2.7		475.8		478.[illegible]
LAUNCH SYSTEMS TOTAL	429.1	5.5	29[illegible]4.8	819.5	
	434.6		[illegible]744.7		4178.[illegible]
R&D FLIGHT [illegible] (2)					307.8



A comparison of the relative values of cost effectiveness for the MLV-SAT-V-4 (S)B; -22(S) and -25(S), shows that the most cost effective method for further improvement of performance is through the use of larger solid motors rather than through the use of uprated upper stage engines. This comparison is shown in Figure 7-13.

FIGURE 7-13 COST EFFICIENCY COMPARISON

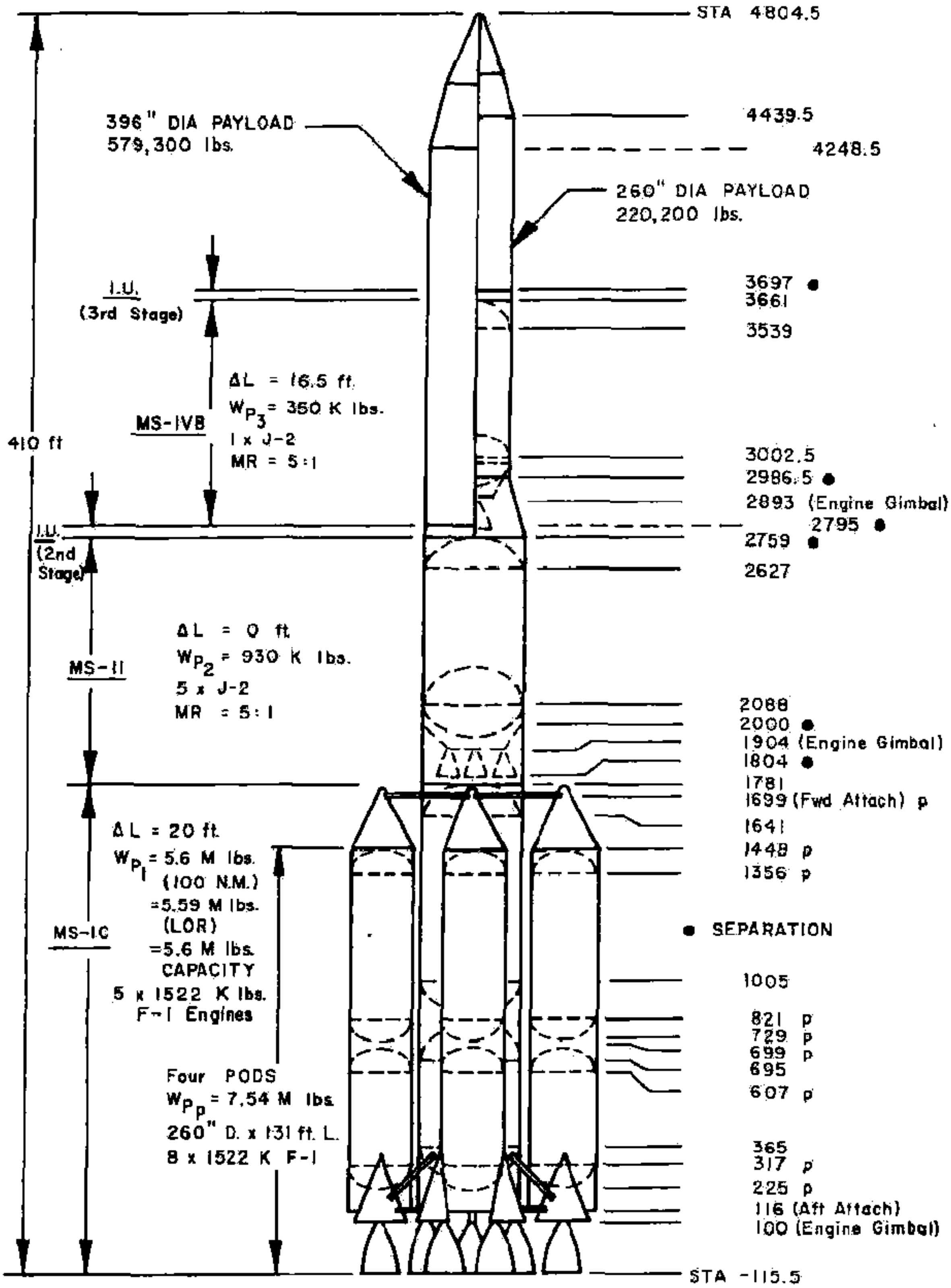


FIGURE 8-1 SAT-V-23(L) BASELINE VEHICLE

8.0 MLV-SAT-V-23(L) LAUNCH VEHICLE

The Saturn V-23(L) vehicle (see Figure 8-1) is a modified Saturn V with lengthened first and third stages, and adapted for attachment of four 260-inch diameter liquid propellant pods. The modified Saturn V core and liquid propellant pods uses standard F-1 and J-2 engines.

The SAT-V-23(L) vehicle as defined in the trade study activity and studied in detail in Phase II activity is a feasible configuration and a logical candidate to provide payloads in excess of those currently available with the Saturn V vehicle.

The Saturn V-24(L) vehicle (see Figure 1-1) is similar to SAT-V-23(L) except that first stage and liquid propellant pods used uprated 1.8 million pound thrust F-1 engines and the upper stages used various numbers and thrust levels of uprated engines as defined for the SAT-V-3B vehicle.

The SAT-V-24(L) vehicle was only studied during the Phase I study activity. A vehicle configuration could not be defined that fell within the 410 foot study ground rule height limitation as defined in Section 3.0.

8.1 CONFIGURATION SELECTION (PHASE I)

By varying the propellant weight between the core stages and pods, a number of related SAT-V-23(L) and SAT-V-24(L) vehicles resulted. Payload capability and vehicle costs were established for these two families of vehicles in order to choose a single baseline vehicle for more detailed analysis.

8.1.1 Candidate Configurations

During the trade study both two and three stage vehicles were considered. Propulsion and engine type for all stages and the liquid propellant pods for the SAT-V-23(L) vehicle was fixed to correspond to standard Saturn V engines as defined for the baseline AS-516 vehicle. Each of the four liquid propellant pods used two standard F-1 engines.

On the SAT-V-24(L) vehicle, five 1.8 million pound thrust uprated F-1 engines were used in the MS-IC-24(L) stage and two in each of the liquid propellant pods. MS-II-24(L) stage used various numbers and thrust levels of uprated engines as defined in the MLV-SAT-V-3B section. The MS-IVB-24(L) stage used a single uprated engine of the same type and thrust level as defined in the MS-II-24(L) stage.

Pod propellant capacity was determined by trading propellant between liquid core stages and the pods. Pod diameters ranging between 156 and 396 inches were considered.

8.1.2 Trade Studies

Figure 8-2 represents the propellant trade between the MS-IC-23(L) core stage and the liquid propellant pods for the SAT-V-23(L) vehicle. The lower set of curves is for the three stage 72-hour lunar injection mission and the upper set of curves for the two-stage 100 nautical mile earth orbit mission. As shown, the variation of performance as a function of pod-to-MS-IC burn time and propellant loading is not extremely sensitive for either two or three stage vehicles. The baseline SAT-V-23(L) was selected, therefore, to satisfy the 410 foot vehicle height limit.

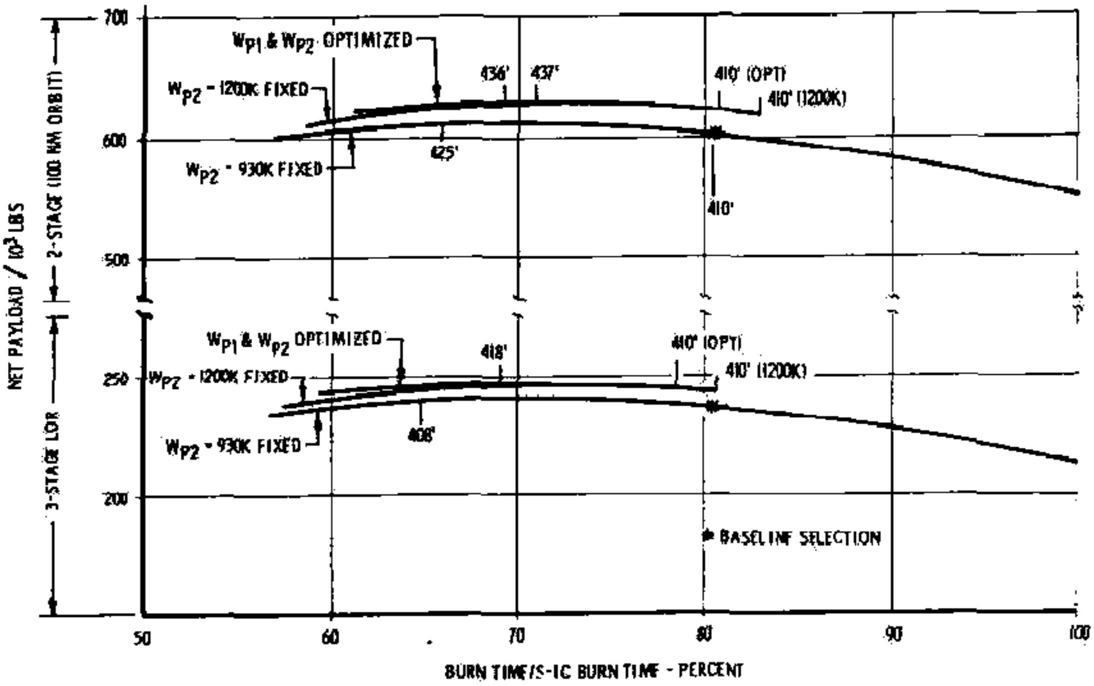


FIGURE 8-2 SAT-V-23(L) TRADE STUDY PERFORMANCE DATA

The SAT-V-23(L) first stage was lengthened 20 feet and the third stage lengthened 16.5 feet which was equivalent to the length and propellant capacity of the MLV-SAT-V-3B baseline vehicle.

Figure 8-3 shows net payload versus the number of engines (dashed lines) and thrust per engine (solid lines) on the MS-II-24(L) stage for vehicles with propellant optimized stages and vehicles with second stage limited to 15.5 feet length increase and third stage limited to 16.5 feet length increase. As shown on Figure 8-3, with fixed upper stages and a MS-II-24(L) thrust of 1.6 million pounds (lower family of curves), the minimum vehicle height for the SAT-V-24(L) was 507 feet at a payload of 348,000 pounds. Optimizing the propellant loading in the upper stages at a MS-II-24(L) thrust level of 1.6 million pounds would result in a payload of 369,000 pounds to 72-hour lunar injection and 860,000 pounds to 100 nautical mile orbit with a vehicle height of 536 feet. Also, at MS-II-24(L) thrust level of between 2.8 and 3.0 million pounds a payload of 410,000 pounds to LOR and 960,000 pounds to 100 nautical mile orbit can be obtained with a vehicle height of 600 feet. Since no vehicle fell within the 410 foot height limit, SAT-V-24(L) was not considered further.

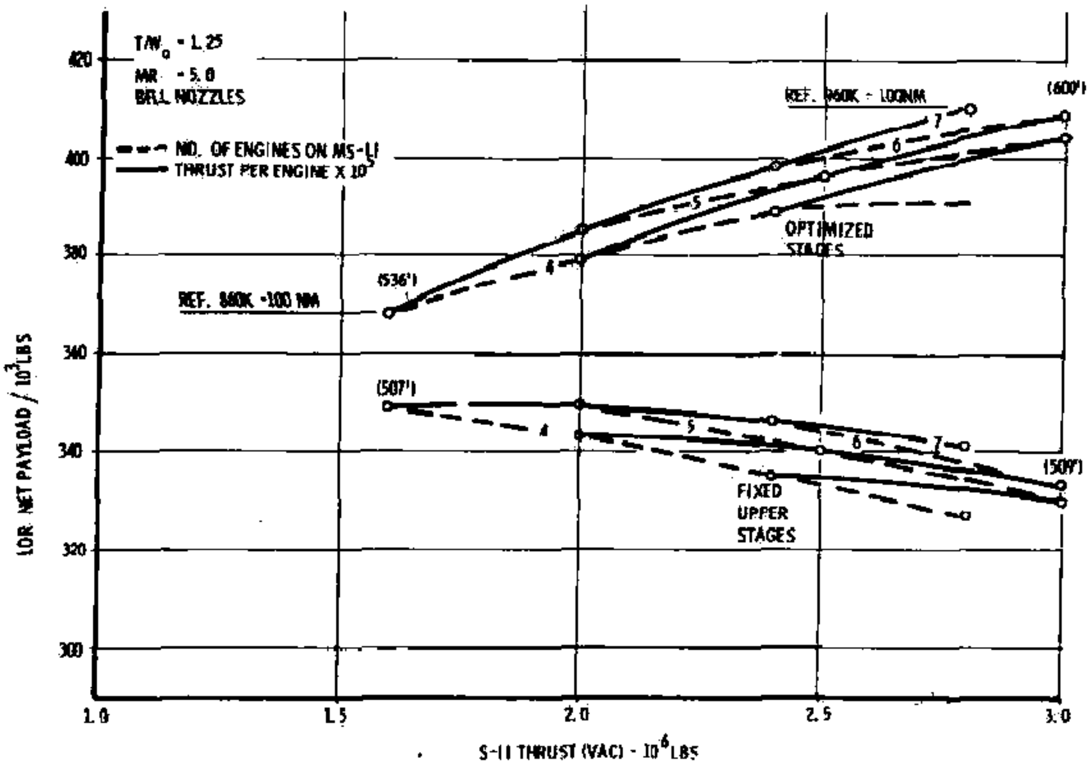


FIGURE 8-3 SAT-V-24(L) TRADE STUDY PERFORMANCE DATA

Trade studies of liquid propellant pod diameter size ranging between 156 to 396 inches show that vehicle and program cost are relatively insensitive to pod diameters between 200 to 300 inches. A 260-inch diameter pod was selected for best overall vehicle geometry.

8.2 DESIGN STUDY VEHICLE (PHASE II)

The baseline SAT-V-23(L) vehicle chosen during the Phase I activity was defined in detail, its capabilities and characteristics were determined and its resource requirements established.

8.2.1 Vehicle Description

The selected baseline vehicle incorporates a 16.5-foot longer third stage, standard length second stage, and 20-foot longer first stage thrust augmented by four 260-inch diameter liquid propellant pods.

The 131-foot long pods (as shown in Figure 8-4) attach to the MS-IC-23(L) stage at the outboard engine locations, use S-IC technology, structural concepts, and systems. Each pod has two standard F-1 engines which gimbal to supplement the control capabilities of the core vehicle Each pod is an independent stage which can be checked out and test fired as a unit.

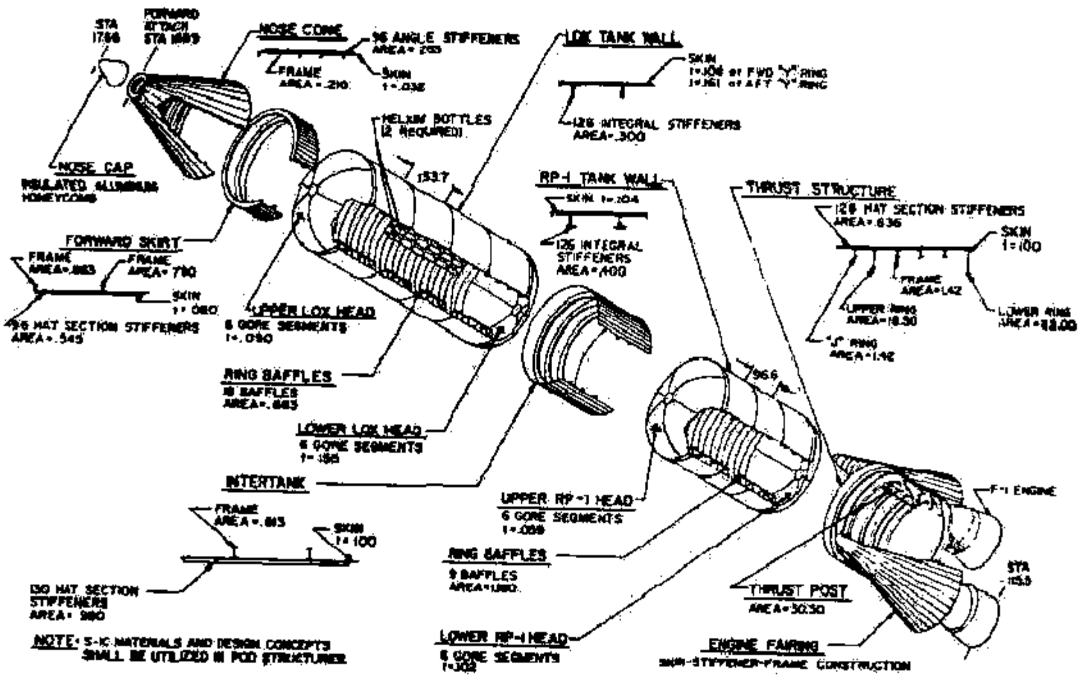


FIGURE 8-4 LIQUID PROPELLANT POD DESIGN

8.2.2 Design Study Results

The SAT-V-23(L) two-stage payload capability to 100 nautical mile orbit is 579,300 pounds and its 72-hour lunar injection payload is 220,200 pounds.

Use of this vehicle was also considered for application where the baseline core vehicle (without the liquid pods) could fly alone or with two strap-on liquid pods. The payloads identified for these alternates are shown on Table 8-1.

TABLE 8-I THREE-STAGE 72-HOUR LUNAR INJECTION PAYLOAD CAPABILITY

No. of Pods	P_L - 10^3 Lbs.
0	80
2	155
4	220

Significant load criteria and other data pertinent to vehicle design are shown on Table 8-II with comparative Saturn V values.

The control requirement of the SAT-V-23(L) requires that the four outboard engines on the core vehicle and all eight engines on the pods gimbal. The maximum gimbal requirements during max q flight is 4.1 degrees of the 5.15 degrees maximum available. Aerodynamic fins are not used.

The aerodynamic heating of the MS-IC-23(L) forward skirt has increased from 167°F (Saturn V) to 215°F due to the shock patterns from the pod nose cones. This increased temperature is not a problem.

Base heating of the MS-IC-23(L) increases from 1900°F (Saturn V) to 2200°F. The heat shield can withstand the increased base temperature.

	-23(L)	SAT V*
LOAD CRITERIA		
MAX q (LBS/FT^2)	792	766
g's AT MAX q, IC	1.90	1.954
HEIGHT (FT)	410	363
CONTROL		
MODE	12 GIMBALED F-1'S	GIMBALED F-1'S
MAX. DEFLECTION ANGLE IN FLIGHT	4.1 DEG	3.5 DEG.
HEATING		
TYP. AERODYNAMIC (S-IC FWD. SKT.) MAX TEMP	215°F	167°F
BASE (MAX TEMP.)	2200°F	1900°F

* BASELINE 516 WITH T_0/W_0 - 1.25

TABLE 8-II SIGNIFICANT LOAD CRITERIA

Communications for some stations will be "blacked out" due to the exhaust plume interference. Other stations, however, will have clear communications during these periods and can provide continuous communications. Crew safety studies show the abort lead time to be 15 to 20 percent greater than for the Saturn V. These abort lead times can be reduced by increasing the escape system rocket motor capability.

Digital simulation of separation dynamics for the expended pods demonstrates that a positive core/pod separation clearance is obtained and that axial clearance occurs at 1.83 seconds after separation.

The additional propellant in the first and third stages and the four liquid propellant pods increases the 0.4 psi on-pad over-pressure distance to a value 28 percent greater than the distance between Pad A and Pad B on Launch Complex 39. Waivers for this distance will be required for joint usage of these pads when either pad contains a fueled core with fueled pods. The acoustic level will require protection of personnel during launch operations.

Combined structural loads and acoustic environments are illustrated in Figure 8-5. The design loads have increased substantially and result in vehicle dry weights which are approximately 25 percent higher than those for the AS-516. The acoustical environment is higher than Saturn V specifications for some vehicle areas on the first stage. Acoustic re-qualification of approximately 70 percent of the acoustically sensitive components on this stage will be required.

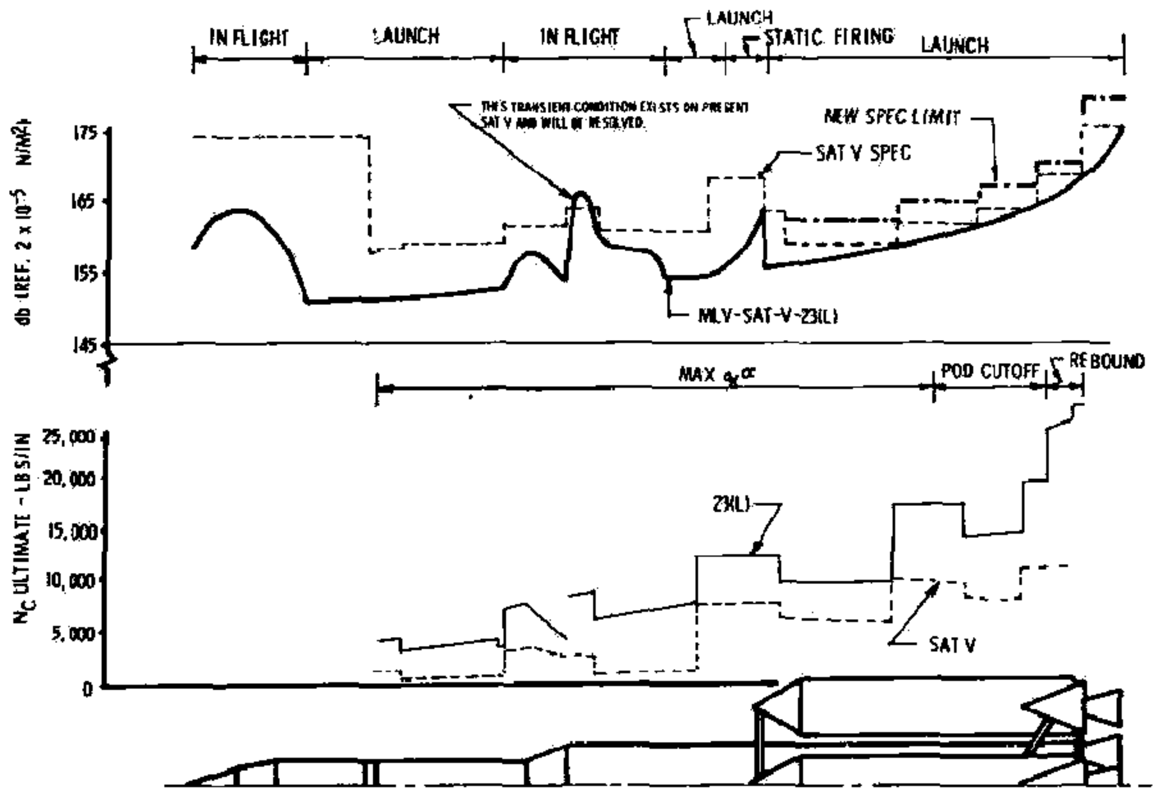


FIGURE 8-5 ACOUSTIC ENVIRONMENT AND STRUCTURAL LOADS

- The impact of structural load increase on the core vehicle is shown in Figure 8-6.

8.3 Resources

Existing and new facilities will be employed to manufacture and test the MLV-SAT-V-23(L). These facilities are to be used on a non-interference basis with the Saturn V production schedules. The present stage and I U manufacturers were assumed to be contractors for the MLV-SAT-V-23(L) components.

A dynamic test vehicle, structural test components, and two R&D vehicles are required. The MS-IC-23(L) stage and the four liquid pods of the dynamic test vehicle will be refurbished after test and used as flight articles.

Thirty MLV-SAT-V-23(L) operational vehicles at a rate of six per year for five years form the basis for the production cost

A new dynamic test stand is required because the SAT-V-23(L) launch weight exceeds Saturn V dynamic test stand foundation capability by 30 percent.

MS-IC-23(L)

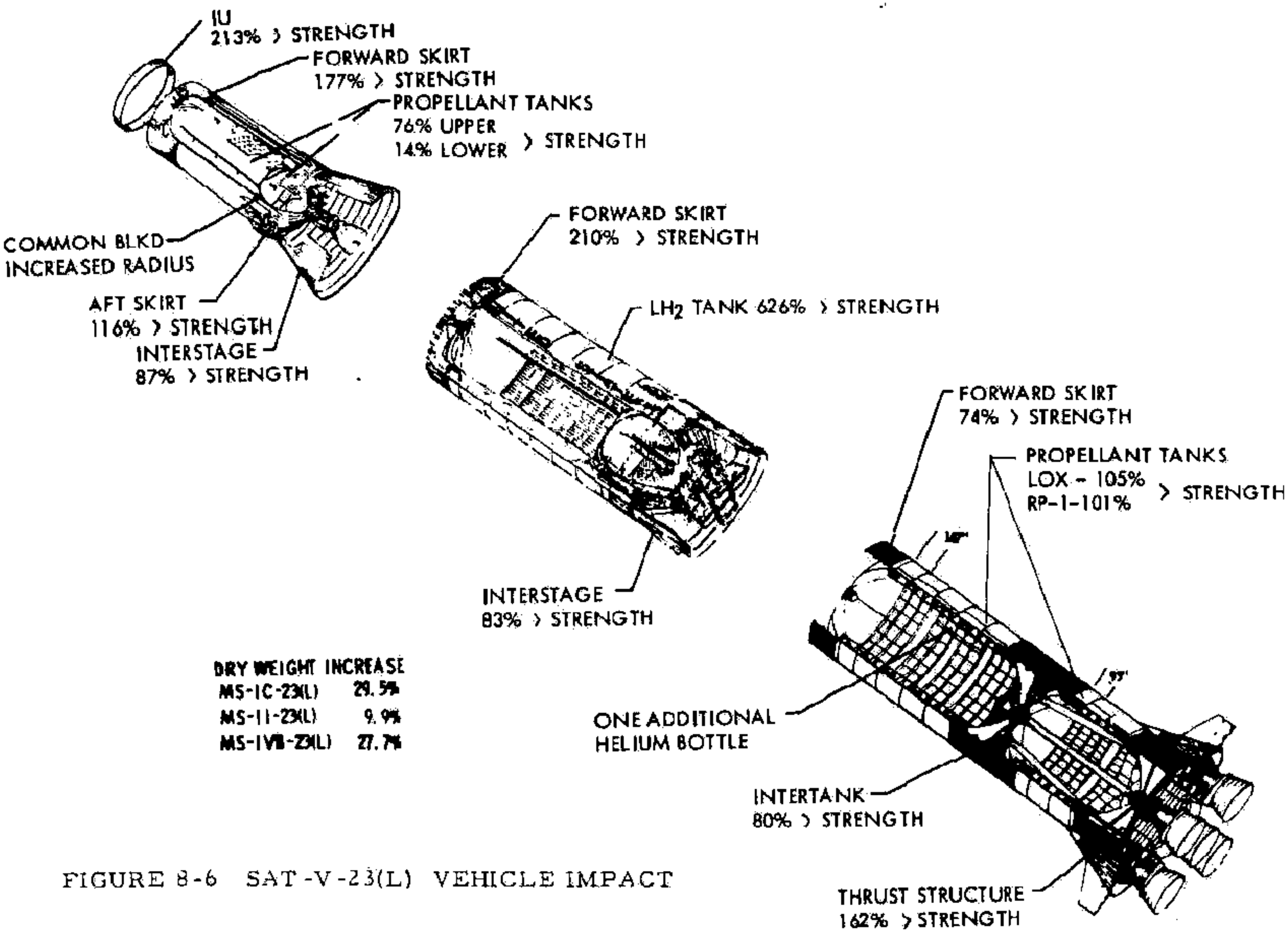
The impact of the first stage changes on manufacturing and test facilities is created primarily by the increased stage length and material thickness The revision to tooling for MS-IC-23(L) is the same as made for the MS-IC-3B since the stage lengths are identical and material thickness is similar.

Liquid Propellant Pods - MS-IC-23(L)

Pod requirements are similar to those of any new stage. New structure will be qualified and its ultimate load carrying capability determined. A full pod structure is constructed for this purpose. Operating components (propulsion, mechanical, electrical/electronic) presently used on S-IC and in an unchanged or less severe environment on the pod do not require further testing. The bulk of pod components fall in this category.

Pod post-manufacturing testing can be accomplished in the existing S-IC test cells at Michoud.

A static firing test stage (battleship weight) is provided to qualify the two engine cluster.



DRY WEIGHT INCREASE	
MS-IC-23(L)	29.5%
MS-II-23(L)	9.9%
MS-IVB-23(L)	27.7%

FIGURE 8-6 SAT-V-23(L) VEHICLE IMPACT

The manufacturing plan for the SAT-V-23(L) liquid pods is similar to the S-IC except for welding skins longitudinally rather than cylindrically and reducing bulkhead gores from sixteen to six.

A new manufacturing facility with approximately two million square feet of area is needed to produce 24 pods per year. The facility could be located at Michoud.

A scaled-down S-IC dual position test stand and storage facilities must be provided at MTF for pod acceptance firing.

MS-II-23(L)

The MS-II-23(L) stage RDT&E program was influenced by two factors - stage design modifications and schedule delivery dates for test hardware. The stage design includes a strengthened structure; therefore, a new structural static test will be required. Delivery of the dynamic test stage nine months before the flight test of the new stage requires acceleration of production to twelve stages per year for the standard S-II stage program. Storage has been assumed available for these stages at no additional cost. Two flight test stages are also included in the development program.

The current S-II program transport equipment and vehicles are compatible with the MS-II-23(L). No modification will be required to handle the additional stage weight.

The manufacturing requirements are also defined by schedule delivery dates and stage structural design. Scheduling of the MS-II-23(L) static/dynamic (S/D) test stage necessitates acceleration of the standard S-II production to accumulate sufficient stages to maintain delivery to KSC at two-month intervals. Manufacture of the first flight test stage is started four months after the S/D stage. The revised structural design requires modification of tools for fabrication and assembly of the forward and aft skirts, LH_2 tank walls, interstage, aft LOX bulkhead and aerodynamic fairings.

The Seal Beach facility requires only minor changes, principally modification to the structural test tower to take the increased test loads. Some handling equipment at Tulsa and Seal Beach will require modification. The Static Test Tower modification must be completed by September 1971. for test of the MS-II-23(L). After static test, the stage is modified for the dynamic test and shipped to MSFC in August 1972. Bi-monthly delivery of the 30 flight stages occurs after completion of the second flight test stage MS-II-36 (October 1973).

MS-IVB-23(L)

The greater length and weight necessitate major modification to the Stage Transporter and certain items of handling equipment, and minor changes to other items. Minor modifications of propulsion and electrical GSE are also required.

Complex arrangements are required to provide time for modification of tooling and facilities without affecting delivery of standard stages. The time for modification and expansion of the tooling and fabrication facilities is provided by a temporary acceleration of the assembly of the standard stage and then storage prior to delivery.

There are no major problems in the other resource areas. The design is within present fabrication technology. No new or unique testing procedure is required. Modifications will be required to transportation and handling equipment. The increased stage size precludes use of the Super Guppy and necessitates ocean shipment.

Launch Facility and Operational Impact

The modified core vehicle will be assembled according to standard procedures in the VAB on the Mobile Launcher. The pods assembled at Michoud will be shipped to MILA where they will be attached to the core vehicle in the VAB. After test and checkout, the vehicle will be moved to the launch pad.

Normal operations for the vehicle will be resumed and additional operations as required for the pod final checkout and arming will be accomplished.

The existing VAB with work platforms relocated and modified can be used. The launch pad and flame trench need modification to adapt to the MLV-SAT-V-23(L) configuration. The existing crawler transports will be replaced. One Saturn V mobile launcher will be modified to handle SAT-V-23(L). A new mobile launcher and mobile service structure are provided because of program timing.

Schedules

Within the study groundrules and after an analysis of the required design and development plans and manufacturing impact, a schedule for development and production of this vehicle was prepared. See Figure 8-7.

The vehicle timing is based on new manufacturing and test facilities for the pods, ground test pod manufacture and test before the first flight article reaches MILA in the second quarter of 1973.

Costs

The vehicle cost summary is shown on Table 8-III.

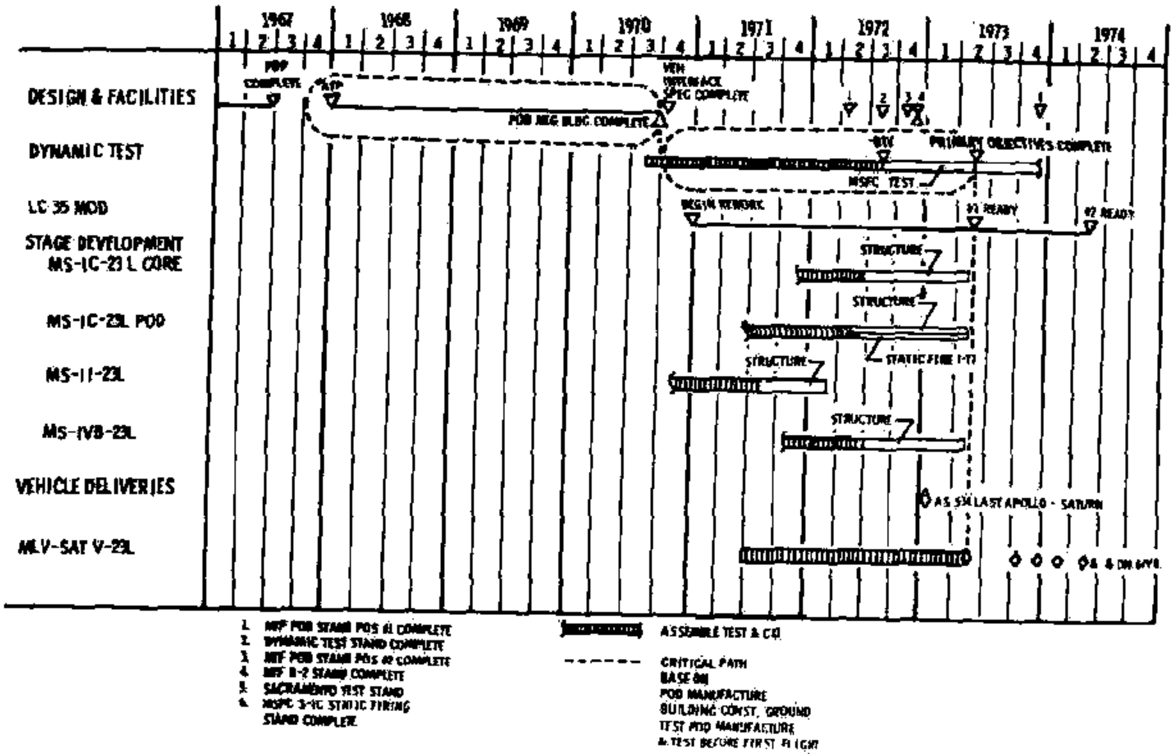


FIGURE 8-7 SAT-V-23(L) VEHICLE DEVELOPMENT AND DELIVERY PLAN

COST - DOLLARS IN MILLIONS	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE					
Boost Assist - Pod	132.7		1023.8	398.4	1554.9
S-IC Stage	67.6		595.7	249.0	912.3
S-II Stage	52.7		517.6	181.3	757.6
S-IVB Stage	72.1		559.3	36.4	467.8
Instrument Unit			131.5		131.5
LAUNCH VEHICLE TOTAL	331.1		2627.9	865.1	3824.1
GROUND SUPPORT EQUIPMENT					
Boost Assist - Pod	33.4		43.2		76.6
S-IC Stage	16.2		27.0		43.2
S-II Stage	11.5		57.6		69.1
S-IVB Stage	53.7		48.5		82.2
GSE TOTAL	94.8		176.3		371.1
FACILITIES					
Boost Assist - Pod	229.9				229.9
S-IC Stage	56.5				56.5
S-II Stage	.7				.7
S-IVB Stage	6.0		5.4		11.4
Instrument Unit					
Launch Vehicle - KSC	213.2		764.6		977.8
Launch Vehicle - Other	10.0				10.0
FACILITIES TOTAL	516.3		770.0		1276.3
SYSTEMS ENGINEERING & INTEGRATION	2.7		475.8		507.5
LAUNCH SYSTEMS TOTAL	944.9		4050.0	865.1	
	944.9		4915.1		5860.0
R&D FLIGHTS (2)					413.0

TABLE 8-III SAT-V-23(L) COST SUMMARY

9.0 CONCLUSIONS AND RECOMMENDATIONS

All the launch vehicles studied under the NAS8-20266 study contract are feasible configurations and candidates for payloads ranging from 78,000 pounds to 960,000 pounds to a 100 nautical mile Earth orbit.

Specific baseline vehicles studied within facility limitation ground rules were limited to a maximum payload of 579,000 pounds to a 100 nautical mile Earth orbit.

9.1 INTERMEDIATE PAYLOAD LAUNCH VEHICLES

From these studies, it was concluded that all the intermediate launch vehicles were feasible using existing Saturn V stages with minimum modifications. These eight launch vehicles will cover an incremental payload range from 78,000 pounds to 255,000 pounds in low Earth orbit. No major structural or other system changes are required to the existing Saturn V stages to obtain this family of flexible launch vehicles.

Therefore, it is recommended that in the selection of the intermediate payload vehicles that both cost effectiveness and flexibility be considered. In the utilization of existing Saturn V stages for intermediate payload application, all possible arrangements should be implemented simultaneously to obtain this flexibility. It is further concluded that all uprated Saturn V vehicles also have intermediate derivatives. In other words, when an uprated vehicle is chosen for development, consideration should be given to developing simultaneously its intermediate payload capabilities.

9.2 UPRATED SATURN V LAUNCH VEHICLES

The uprated vehicles defined in the trade study and studied in detail in Phase II are feasible configurations and logical candidates for payloads in excess of the current Saturn V capability. No major problem areas were identified for either development or production. No significant adverse flight environmental characteristics were identified.

Major vehicle modifications to adapt the Saturn V vehicle to these new uprated configurations were defined as follows.

a. Structural modification of all stages to enable the vehicle to withstand the higher loads imposed due to the larger payload capability and increased payload envelope (increased payload diameter and 410 foot vehicle height).

b. Structural modification of some stages for increased tank lengths and increased engine thrust.

c. Selective requalification of the acoustically sensitive components to qualify these components for the increased acoustical environment.

General comparative conclusions arrived at from the NAS8-20266 studies are as follows.

The SAT-V-3B launch vehicle has the best payload to launch weight of all the uprated vehicles studied. The SAT-V-3B has the minimum launch impact of all the uprated launch vehicles studied. On the other hand, this vehicle requires the most research and development cost per pound of payload, and requires the most lead time of all uprated launch vehicles studied.

The SAT-V-4(S)B launch vehicle has the best payload per research and development dollar with a nominal launch impact. However, as shown on Figures 2-3 and 2-4, when operation costs are included, the SAT-V-4(S)B does not become the most cost effective launch vehicle. It requires the least lead time and development cost of all the uprated launch vehicles.

Of all the uprated launch vehicles studied under the NAS8-20266 contract, the SAT-V-25(S) launch vehicle is the most cost effective (slightly ahead of SAT-V-23(L)). The SAT-V-25(S) vehicle, when compared to the SAT-V-4(S)B vehicle, does cost more to develop and has a greater impact at the launch facility.

The SAT-V-23(L) launch vehicle has the greatest payload capability of all the launch vehicles studied and is almost as cost effective as the SAT-V-25(S). It also has the advantage of using existing standard Saturn V engines, propellants, and systems. It does, however, have the greatest impact on the launch facility.

As shown in Figures 2-3 and 2-4, the cost efficiency of the SAT-V-23(L) launch vehicle is directly competitive with the modified Saturn Vs with strap-on solid motors (SAT-V-4(S)B and SAT-V-25(S)). The development of a low cost liquid stage for strap-on boost-assist purposes would further reduce the cost efficiency of the SAT-V-23(L) launch vehicle. Another factor restraining the potential payload capability of the SAT-V-23(L) vehicle is the 410-foot height limitation established as a ground rule for the NAS8-20266 study. Further work should be done to consider overcoming the 410-foot height limitation such as installing the payload outside VAB, modification to VAB, etc.

Further studies should be directed toward future refinements of the vehicle designs and specifically toward possible future applications. The increased payload capability and improved cost effectiveness over that of the existing Saturn V could be used to reduce significantly overall mission costs by allowing the payloads for these missions to increase in weight to provide payload design simplification. For example, a direct lunar shot for Apollo-type missions could greatly reduce the cost of the payload package from those required for the current LOR hardware.

Assuming manned interplanetary exploration is an ultimate NASA goal, it is further recommended that:

a. A study be conducted on a modified uprated Saturn V core with 260-inch diameter solid motor strap-ons as an "ultimate" Saturn V uprating step, which, e.g., could eliminate rendezvous as a requirement for manned fly-by missions and minimize rendezvous for more ambitious missions. This method of uprating should be compared both with the SAT-V-23(L) and -24(L) vehicles described in this study and "low-cost" pressure-fed storable liquid pod strap-on alternatives; and

b. The feasibility of a stepped uprating program utilizing a common core be explored which would minimize facilities impact. In other words, size a modified Saturn V liquid core that could efficiently accept as strap-on boost-assist components several different sizes of solid rocket motors and/or several different sizes of liquid pods. Also develop a complementary flexible launch support equipment concept.

D5-13183-6

Final Report - Studies of Improved Saturn V Vehicles and Intermediate Payload Vehicles (P-115)

RESEARCH & TECHNOLOGY IMPLICATIONS REPORT

Prepared for
NASA - George C. Marshall Space Flight Center under Contract NAS8-20266

October 7, 1966

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THE BOEING COMPANY — SPACE DIVISION

STUDIES OF IMPROVED SATURN VEHICLES AND INTERMEDIATE PAYLOAD SATURN VEHICLES (P-115)

RESEARCH & TECHNOLOGY
IMPLICATIONS REPORT
D5-13183-6

FINAL REPORT
PREPARED UNDER CONTRACT NUMBER NAS8-20266

SUBMITTED TO
GEORGE C. MARSHALL SPACE FLIGHT CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
OCTOBER 7, 1966

SUBMITTED BY
SYSTEMS ANALYSIS CONTRACTOR
THE BOEING COMPANY
SPACE DIVISION
LAUNCH SYSTEMS BRANCH
HUNTSVILLE, ALABAMA

ABSTRACT

This document reports the research and technology implications involving the Saturn vehicles studied under NASA/MSFC Contract NAS8-20266, "Studies of Improved Saturn V Vehicles and Intermediate Payload Saturn Vehicles (P-115)," from December 6, 1965, to October 7, 1966. Phase I of the study was a parametric performance and resources analysis to select one baseline configuration for each of the six vehicles. Phase II of the study included a fluid and flight mechanics study, design impact on systems, and a resources analysis for each baseline vehicle. The intermediate payload vehicle derivatives of Saturn V are a logical means of providing orbital payload capability between that of the Saturn IB and the two-stage Saturn V. The uprated vehicles are feasible configurations and logical candidates for payloads in excess of the current Saturn V capability. No major problem areas were identified for either development or production.

KEY WORDS

Contract NAS8-20266
D5-13183-6
Intermediate Saturn Vehicles
Saturn V
Research Implications
Technology Implications
Problem Areas
Uprated Saturn Vehicles

FOREWORD

This volume contains the research and technology implications resulting from a ten-month study to prepare technical and resource data on uprated payload Saturn V and intermediate payload Saturn vehicles. This study was part of a continuing effort by the National Aeronautics and Space Administration (NASA) to investigate the capability and flexibility of the Saturn V launch vehicle and to identify practical methods for diversified utilization of its payload capability. NASA Contract NAS8-20266 authorizes the work reported herein and was supervised and administered by the Marshall Space Flight Center (MSFC). S-II data were supplied by the Space & Information Division of North American Aviation. S-IVB data were supplied by the Missile & Space Systems Division of Douglas Aircraft Company. Launch system data were supplied by the Denver Division of The Martin Company. Solid motor data were supplied by United Technology Corporation. The Launch Systems Branch, Aerospace Group, Space Division of The Boeing Company, was the Systems Analysis contractor for this study.

Program documentation includes a summary volume, five volumes covering vehicle descriptions, research and technology implications report (this document), and a cost document. Individual designations are as follows:

D5-13183	Summary Document
D5-13183-1	Vehicle Description MLV-SAT-INT-20, -21
D5-13183-2	Vehicle Description MLV-SAT-V-3B
D5-13183-3	Vehicle Description MLV-SAT-V-25(S)
D5-13183-4	Vehicle Description MLV-SAT-V-4(S)B
D5-13183-5	Vehicle Description MLV-SAT-V-23(L)
D5-13183-6	Research & Technology Implications Report
D5-13183-7	First Stage Cost Plan

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ILLUSTRATIONS

1.0 INTRODUCTION

This report summarizes the technological problems involved when uprating the Saturn V payload capability or when using Saturn V stages for missions in the payload range between the current Saturn IB and Saturn V capability.

The study is part of a continuing effort by NASA to identify a spectrum of practical launch vehicles to meet future payload and mission requirements as they become defined.

The vehicles studied were combinations of existing or modified Saturn V stages; some vehicles also included boost-assist components. A primary study requirement was to make maximum use of existing Saturn technology and support equipment.

In general, the NAS8-20266 study program objectives were to:

a. Select feasible and cost effective baseline vehicles from each of several categories;

b. Prepare sufficient technical data to define vehicle environments, design, capabilities, and characteristics;

c. Define support system requirements;

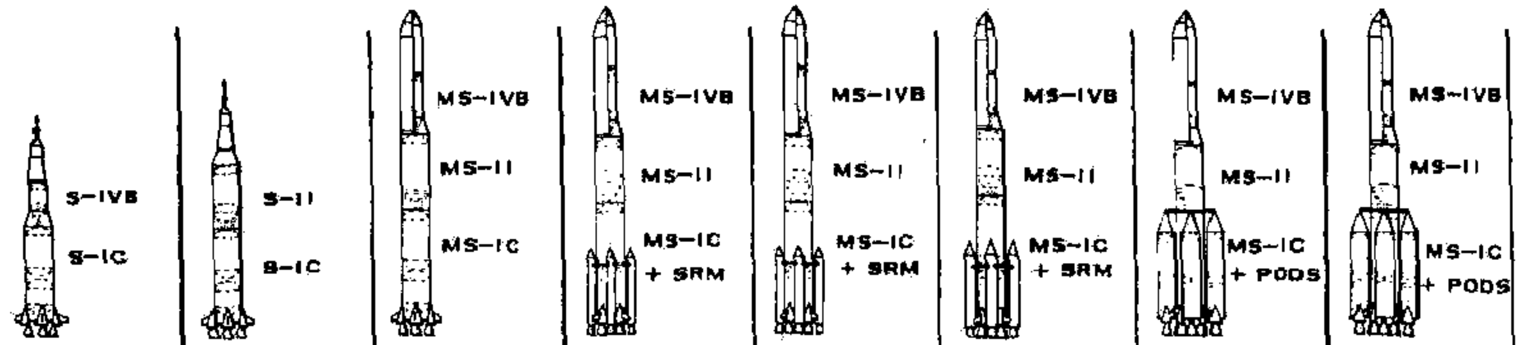
d. Determine the date that the first flight article could be available within study groundrules; and

e. Estimate cost required for implementation of the system plus production of thirty flight articles in five years.

There were two phases of study work. Phase I was a twelve-week effort in which vehicle performance and preliminary cost trade studies were conducted to select a feasible and cost effective baseline vehicle from each of five categories (shown in Figure 1-1). An additional baseline vehicle was later added from Category 4.

For each of the six baseline vehicles selected (see Figure 1-2), Phase II directed the effort to defining ground and flight environments, defining system design and resource impact for each stage and the total vehicle, and determining vehicle mission capabilities and characteristics.

GROUND RULES
MAX VEHICLE SIZE - 410 FT
MAX MS-II SIZE 1160 IN
MAX MS-IVB SIZE 350K PROPELLANT AT MR 5 :1
MAX PAYLOAD 5 LB/FT^3 (2 STG)
11 LB/FT^3 (3 STG)
STD = STANDARD
ΔL = CHANGE IN STAGE LENGTH



LAUNCH VEHICLE	INT-20	INT-21	SAT-V-3B	SAT-V-4(S)B	SAT-V-22(S)	SAT-V-25(S)	SAT-V-23(L)	SAT-V-24(L)
CATEGORY	1	2	3	4			5	
THIRD STAGE			ADVANCED ENGINE ΔL VARIABLE	STD J-2 ΔL VARIABLE	ADVANCED ENGINE ΔL VARIABLE	STD J-2 ΔL VARIABLE	STD J-2 ΔL VARIABLE	ADVANCED ENGINE ΔL VARIABLE
SECOND STAGE	STD J-2's ΔL - 0	STD J-2's ΔL - 0	ADVANCED ENGINES ΔL VARIABLE	STD J-2's ΔL VARIABLE	ADVANCED ENGINES ΔL VARIABLE	STD J-2's ΔL VARIABLE	STD J-2's ΔL VARIABLE	ADVANCED ENGINES ΔL VARIABLE
FIRST STAGE	STD F-1's ΔL - 0	STD F-1's ΔL - 0	5 X 1.8M F-1 ENGINES ΔL VARIABLE	STD F-1's ΔL VARIABLE	STD F-1's ΔL VARIABLE	STD F-1's ΔL VARIABLE	STD F-1's ΔL VARIABLE	5 X 1.8M F-1 ENGINES ΔL VARIABLE
STRAP-ON COMPONENTS				4 X 120 IN DIA SOLID MOTORS	4 X 120 IN DIA SOLID MOTORS	4 X 156 IN DIA SOLID MOTORS	4 LIQUID PODS 2 X STD F-1 ENGINES	4 LIQUID PODS 2 X 1.8M F-1 ENGINES

FIGURE 1-1 PHASE I LAUNCH VEHICLE CANDIDATES

BOEING

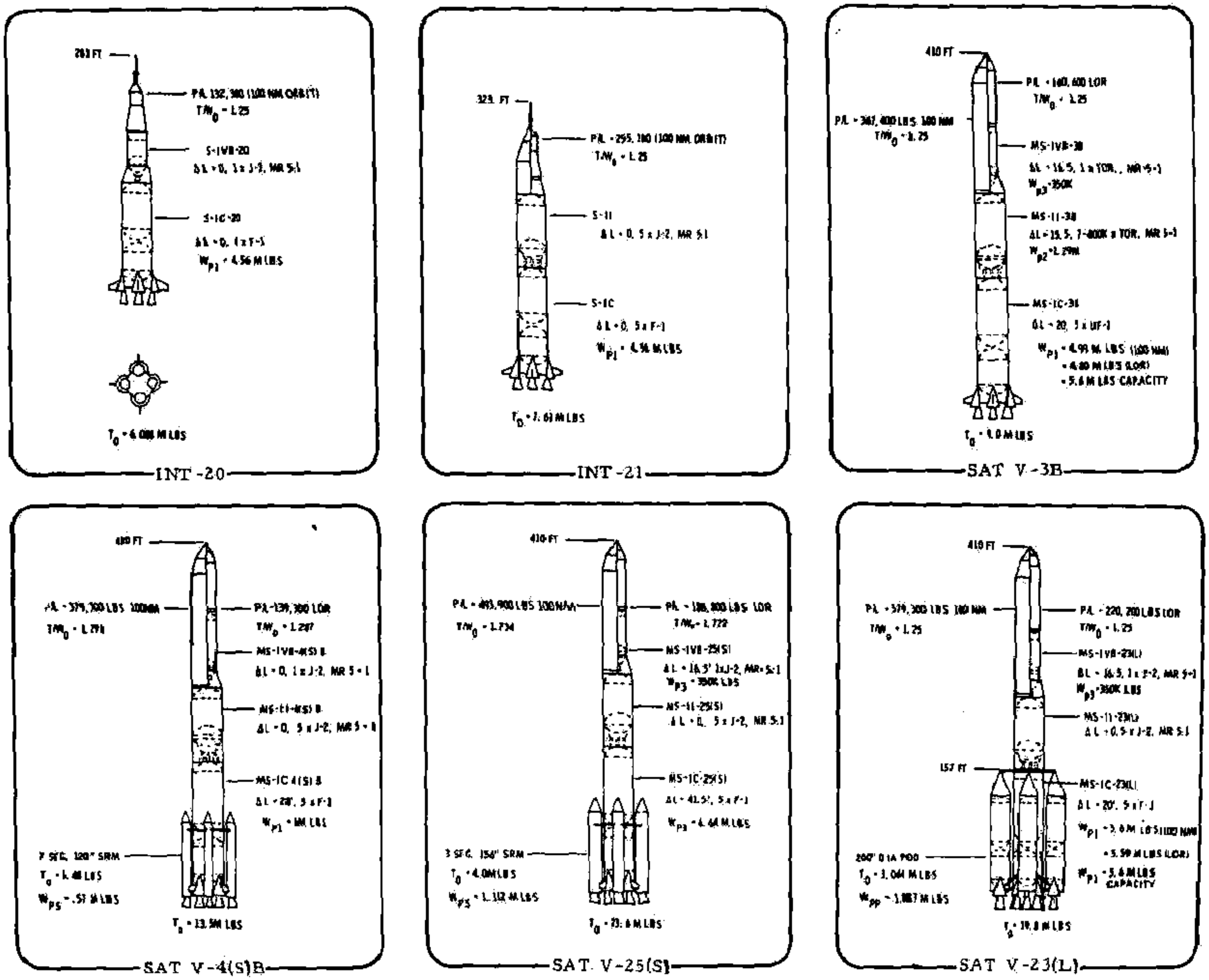


FIGURE 1-2 SELECTED BASELINE LAUNCH VEHICLES FOR PHASE II STUDY EFFORT

The launch vehicles in Categories 1 and 2 are Saturn V stage combinations for missions in the payload range between the current Saturn IB and Saturn V payload capability. The launch vehicles in Categories 3, 4, and 5 are advanced Saturn V configurations with payload capabilities beyond that of the existing Saturn V.

The five categories of vehicles are as follows:

Category 1 (MLV-SAT-INT-20) during Phase I was a family of two-stage launch vehicle candidates with standard size S-IC and S-IVB stages using standard F-1 engines (three, four, and five) and a standard J-2 engine. A single baseline launch vehicle (Figure 1-2) was selected for the Phase II study effort.

Category 2 (MLV-SAT-INT-21) during Phase I was a family of two-stage launch vehicle candidates with standard size S-IC and S-II stages using standard F-1 engines (three, four, and five) and J-2 engines (three, four, and five). A single baseline launch vehicle (Figure 1-2) was selected for the Phase II study effort.

Category 3 (MLV-SAT-V-3B) during Phase I was a family of two- and three-stage launch vehicle candidates with modified Saturn V stages using various types, numbers, and thrust levels of advanced engines in the upper stages and uprated F-1 engines in the modified S-IC stage. A single baseline launch vehicle (Figure 1-2) was selected for the Phase II study effort.

Category 4 included modified Saturn V launch vehicles with strap-on solid motor boost-assist components. Three families of vehicles were studies as follows:

a. MLV-SAT-V-4(S)B during Phase I was a family of two- and three-stage launch vehicles with modified Saturn V stages, standard F-1 and J-2 engines with strap-on 120-inch diameter (five, six, and seven segmented) solid motors. A single baseline launch vehicle (Figure 1-2) was selected for the Phase II study effort.

b. MLV-SAT-V-22(S) during Phase I was a family of two- and three-stage launch vehicles with modified Saturn V stages using various types, numbers, and thrust levels of advanced engines in the upper stages, a modified S-IC stage with standard F-1 engines in the first-stage, and strap-on 120-inch diameter (five, six, and seven segmented) solid motors. No launch vehicle in this family was studied beyond Phase I.

c. MLV-SAT-V-25(S) during Phase I was a family of two- and three-stage launch vehicles with modified Saturn V stages, standard F-1 and J-2 engines, and strap-on 156-inch diameter (two, three, and four segmented) solid motors. A single baseline launch vehicle (Figure 1-2) was selected for the Phase II study effort.

Category 5 included modified Saturn V launch vehicles with strap-on boost-assist liquid propellant pods. Two families of vehicles were studied as follows:

a. MLV-SAT-V-23(L) during Phase I was a family of two- and three-stage launch vehicles with modified Saturn V stages, standard F-1 and J-2 engines, and four strap-on liquid propellant pods, each using two standard F-1 engines. A single baseline launch vehicle (Figure 1-2) was selected for the Phase II study effort..

b. MLV-SAT-V-24(L) during Phase I was a family of two- and three-stage launch vehicles with modified Saturn V stages using various types, numbers, and thrust levels of advanced engines in the upper stages, a modified S-IC stage with uprated F-1 engines, and four liquid propellant pods each containing two uprated F-1 engines. No launch vehicles in this family were studied beyond Phase I.

2.0 STUDY RESULTS

The six baseline vehicles studied under the contract are logical configurations and candidates for payloads ranging from 78,000 pounds to 579,000 pounds to a 100 N.M. Earth orbit.

All eight configurations of the intermediate launch vehicles studied during the trade study phase are feasible using existing Saturn V stages with minimum modification. These vehicles have payload capabilities ranging from 78,000 pounds to 255,000 pounds and provide a family of flexible and cost effective launch vehicles which are available within the normal lead time required for the stage elements (2 years).

The four uprated baseline vehicles are feasible configurations and logical candidates for payloads in excess of the current Saturn V capability. No major problem areas for development, production, or flight environmental characteristics were identified. The increased payload capability and improved cost effectiveness over that of the existing Saturn V could be used to reduce overall mission costs. For some missions, increased payload capability can be used to simplify the mission mode and thus result in less complex payload designs and fewer components. For example, a direct lunar shot for Apollo-type missions could greatly reduce the costs of the payload package from those costs required for the current LOR hardware by the elimination of the lunar rendezvous.

Explanation of the technical and resource data on this contract is summarized and reported in detail in the following documentation:

D5-13183	Summary Document
D5-13183-1	Vehicle Description MLV-SAT-INT-20, -21
D5-13183-2	Vehicle Description MLV-SAT-V-3B
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D5-13183-5	Vehicle Description MLV-SAT-V-23(L)
D5-13183-6	Research & Technology Implications Report
D5-13183-7	First Stage Cost Plan

3.0 AERONAUTICS TECHNOLOGICAL IMPLICATIONS

3.1 AERODYNAMIC STATIC STABILITY

3.1.1 Problem Definition

The uprated Saturn V launch vehicles are aerodynamically unstable during boost flight. The amount of aerodynamic instability is one of the major factors that determines the thrust vector control requirement. The uprated launch vehicles that use boost assist components are difficult to analyze because of the complex flow fields about the core vehicle and the strap-on solid motors or liquid propellant pods. Estimates of the total normal force contribution to the core vehicle are difficult to make because of the carryover from the strap-on to the core vehicle and the carryover from the core vehicle to the strap-ons. The effect of the gap between the core vehicle and the strap-on and the effect of the strap-on on fin or engine shroud effectiveness are also difficult to determine. The lack of experimental data and the inability of the various theoretical methods to analyze these complex vehicles add to the difficulties.

3.1.2 Solution Approach

It is recommended that wind tunnel tests be conducted on a Saturn V uprated configuration that uses solid strap-on motors for boost assist to determine the effect of the strap-on motors on the vehicle aerodynamic coefficients. The tests should determine the increment in the aerodynamic coefficients so that the results can be applied to other vehicles with a minimum of effort.

Typical wind tunnel data would have a variety of uses if it followed the form shown on Figure 3-1.

3.2 WIND SPEED ENVELOPE FOR LAUNCH TOWER CLEARANCE STUDIES

3.2.1 Problem Definition

In the definition of launch tower clearance, the 99 percentile ground wind speed envelope is used. This wind speed envelope is the result of a statistical analysis of ground winds measured over a number of years in a clear field at Patrick Air Force Base at various altitudes and correlated with the power law $U = U_1 (Z/Z_1)^P$ A wind speed envelope determine in this manner could be different from what a Saturn V type launch vehicle experiences because of large protuberances around the launch vehicle. The schematic in Figure 3-2 illustrates the problem.

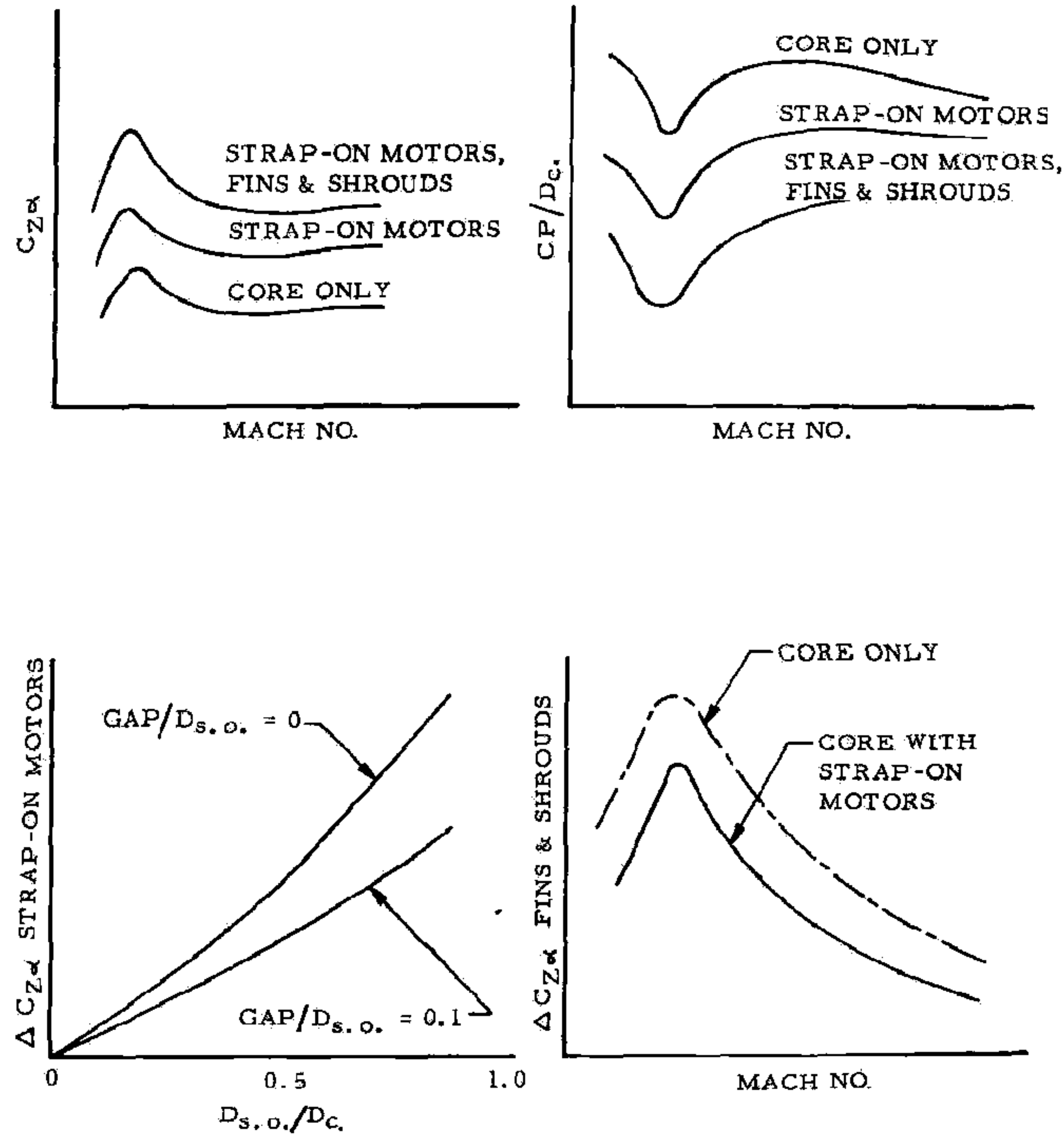

FIGURE 3-1 TYPICAL AERODYNAMIC DATA

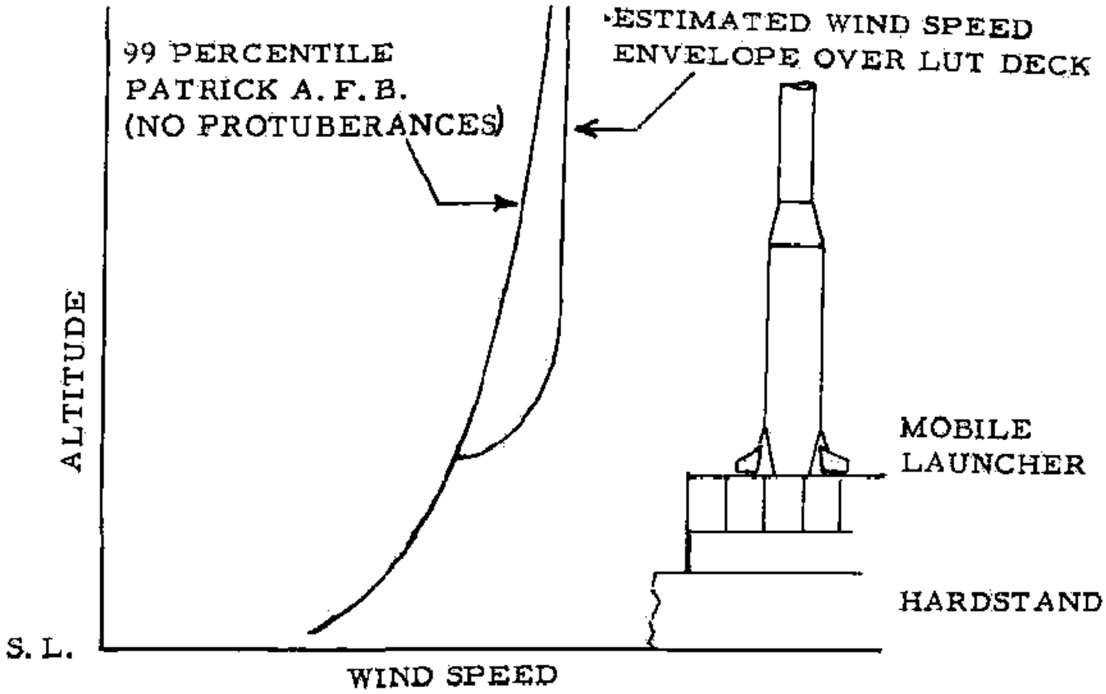

FIGURE 3-2 TYPICAL GROUND WIND SPEED ENVELOPE

Differences between a wind speed envelope determined in a clear field and that experienced by a Saturn V launch vehicle will result in differences in the aerodynamic disturbing moment on the vehicle which would affect the liftoff trajectory .

3 2 2 Solution Approach

Theoretical estimates of the induced wind velocity should be made using the Schwarz Christoffel theorm or other complex transformations to update the wind speed envelope. These theoretical estimates of the induced velocity over the hardstand and the mobile launcher can be used to determine, more accurately, the wind speed envelope that the launch vehicle will experience and to correlate the experimental wind speed data that is now being obtained on the mobile launcher.

4.0 ELECTRONICS AND CONTROL TECHNOLOGICAL IMPLICATIONS

4.1 *LAUNCH TOWER CLEARANCE*

4.1.1 Problem Definition

Large launch vehicles of the nature of those considered in this study experience high horizontal translations during lift-off when subjected to design winds and biased with off nominal design and construction parameters. Large drift distances and the resulting probability of tower collision can require extensive modification to existing launch pad facilities.

4.1.2 Solution Approach

Investigate the effect of distributed aerodynamics on vehicle lift-off trajectory and extend the study of lift-off dynamics to include representation of restrained release and malfunction conditions (engine loss). Parameterize rigid body control system feedback loops to minimize drift distance while remaining within attitude (guidance) constraints. A six degree of freedom digital simulation should be used to conduct these studies.

4.2 EFFECT OF CONTROL MODES ON VEHICLE LOADS

4.2.1 Problem Definition

The effect of alternate control modes on gimbal requirements was examined during the course of these studies. These results showed that the effect of varying control modes on gimbal displacement was negligible, and that one control system could not be recommended over another on the basis of reduced TVC requirements. This result does not preclude, however, the possible advantage of one control system over another in the area of reducing bending loads. Limited studies showed differences in maximum bending responses of 12 percent and 5 percent respectively for the MLV-Saturn V-3B and the MLV-Saturn V-4(S)B vehicles when comparing an attitude-attitude rate control system to attitude-attitude rate with angle of attack feedback. These results were obtained without optimizing control gains for load relief and using synthetic wind representations.

The amount of load reduction available by control systems optimization is a function of control mode, vehicle configuration, and the nature of the wind forcing input. These effects should be assessed to provide information upon which selections of optimum control systems for the uprated Saturn family of vehicles may be based.

4.2.2 Solution Approach

A comprehensive study is indicated using candidate control systems with gains optimized for load relief. In order to determine the relative effect of varying vehicle configuration, it is suggested that the family of uprated and intermediate Saturn V vehicles in its entirety be examined. Selected vehicles among these should be examined using both synthetic and real wind representations which are consistent with each other. A number of measured (Jimsphere) profiles of varying severity would be selected and consistent synthetic profiles having the same peak wind magnitude at the same altitude would be constructed. Wind shear of the synthetic profiles would be adjusted to give the same peak bending response using nominal attitude-attitude rate control before control systems optimizations for load relief are begun. Stability analyses using the elastic properties of the vehicles and varying sensor placement to assure practicability of the control gains selections and optimizations would be required.

The feasibility of employing multiple sensors and weighted control systems feedbacks to limit vehicle elastic bending response should be examined.

4.3 RANGE/VEHICLE COMMUNICATION INTERFERENCE

4.3.1 RF Interference From Added Strap-On Structure

4.3.1.1 Problem Definition

The addition of strap-on boosters, either liquid or solid, may place the structure of the booster in a position that interferes with antenna look angles from the vehicle to the ground station. Each antenna on the Saturn V stages should be considered to determine if structure obscures line of sight to any ground station.

The structure of the strap-on booster also may modify the antenna radiation patterns. The booster structure acts as reflectors which will modify the radiation characteristics of each antenna.

4.3.1.2 Solution Approach

Antenna radiation pattern tests should be conducted on an antenna range using models of the vehicles with strap-on boosters. These tests should be performed for all of the antenna systems on every stage. A complete description of an antenna's radiation field as it exists in space about the vehicle should be plotted. The tests and plotting of data should be conducted to the same groundrules as the data presented in Boeing Document D5-15526-1, May 16, 1966, title: "Saturn V Antenna Systems AS-501," conducted for contract NAS8-5608.

4.3.1.2 Solution Approach (Continued)

It may be necessary to relocate the antennas by moving them outward, forward, or to a different angular position to improve the radiation pattern and/or look angle. It may also be necessary to change the design of the antenna systems to improve the radiation pattern.

4.3.2 RF Interference From Modified Exhaust Plume

4.3.2.1 Problem Definition

The uprated vehicles have enlarged exhaust plumes due to increased mass flow. The solid strap-on motors also add different exhaust products, including metal particles, which increase the degradation of the RF system.

4.3.2.2 Solution Approach

The communication data recorded from Titan III flights should be evaluated to determine RF interference through the exhaust plume of combined liquid engines and solid rocket motors. This data should be correlated to data from Saturn I flights and other sources of RF interference data through exhaust plumes. This evaluation should consider the RF interference as a function of antenna look angle, using the look angle curves in Section 5.2.7 of documents D5-13183-3, -4, and -5.

4.3.3 Modified Antenna System

4.3.3.1 Problem Definition

The improvement study indicated a requirement to place a redundant set of MS-IC antennas on the strap-on boosters which would be used prior to staging of the strap-on boosters. The strap-on boosters have RF systems independent of the Saturn V, but require combination of the four booster RF circuits on the MS-IC and redistribution to antennas on the strap-ons. The longer circuit paths and added switching from the transmitters to the antennas may produce increased RF degradation as well as presenting hardware problems for the proposed system.

4.3.3.2 Solution Approach

The modified antenna systems proposed (in Sections 6.1.4 of documents D5-13183-3, -4, and -5) for use with the strap-on boosters should be evaluated analytically, and then tested with a systems breadboard using S-IC components.

4.3.3.2 Solution Approach (Continued)

The results of this investigation would be to determine areas of RF systems adequacy, recommend changes to the Saturn V systems, and define further testing required. It is necessary to investigate the RF systems for the upper stages as well as the MS-IC to assure adequare communication for the vehicle.

5.0 MATERIALS AND STRUCTURES TECHNOLOGICAL IMPLICATIONS

5.1 STABILITY CRITICAL STIFFENED CYLINDRICAL SHELL DESIGN

5.1.1 Problem Definition

Since the original design of the Saturn V vehicle, several methods of analyses have been developed to more accurately predict the load-carrying capability of stability-critical axially compressed stiffened cylinders. However, none of these methods have been proved reliable for all stiffened cylinder configurations applicable to the Saturn V vehicle. One of these methods, developed by Boeing, (Ref. D5-13272) considers all three possible modes of instability failures and has been partially verified with results from the S-IC corrugated intertank test program and from various test data in the literature. This method is currently being extended to account for the increase in load-carrying capability due to pressure stabilization.

5.1.2 Solution Approach

1. Verify by test and analyses the validity and range of application of the available analytical procedures including the method developed by Boeing.

2. Compare results of the various methods.

3. Use the best of these methods for each type of stiffened cylinder applicable to the Saturn V vehicle and re-evaluate the various uprated Saturn V vehicles for possible weight savings and increased reliability.

5.2 PAYLOAD ENVELOPE WIND SENSITIVITY

5.2.1 Problem Definition

The intermediate vehicles INT-20 and INT-21 were studied using the Apollo payload shape and were shown to have adequate strength without modification using current Saturn wind criteria. Due to the limited volume of the Apollo envelope, and larger payload weights afforded by these vehicles for orbital missions, payload densities are relatively high. Actually these vehicles have considerably more versatility for carrying payloads of higher volume and lower density during extensive periods of the year. This versatility is available from two sources: (1) using reduced factors of safety for un-manned missions, and (2) from reduced wind magnitudes which exist during periods of the year for which launches may be planned.

5.2.2 Solution Approach

In order to define the versatility which exists in this respect, it is suggested that comprehensive analyses be performed using a family of payload shapes and sizes and winds of varying magnitude as defined by MSFC criteria for each month of the year. Payload length limits would be established for each month of the year as a function of peak wind speed for manned and un-manned factors of safety within the structural capability of these vehicles. From these data estimates of launch availability versus month of the year may be obtained for payloads of varying size on a probabilistic basis. Such information is very desirable for planning purposes.

5.3 FLIGHT LOADS PROBABILITY

5.3.1 Problem Definition

Generally speaking synthetic wind representations which have been developed from statistical analysis of measured wind data do not produce loads responses with the same probability, i.e., there is no direct correlation between the probabilities of peak wind speed, wind shear, altitude, and direction, and load exceedance probability. Currently there are insufficient data upon which to base computation of the combined probability of all wind parameters so that even synthetic criteria must still be selected through exercise of sound engineering judgments. Obviously conservatism is a desirable consideration in this case. There are sufficient measured winds, however, to approach the problem from the direct load exceedance standpoint. Improved wind measurement systems are producing higher resolution profiles. It should be possible to use these data coupled with other measured wind information to compute load exceedance probability from statistical samples of measured profiles. This approach would reduce the need for conservatism associated with engineering judgments which have been required in the past in the development of synthetic profiles.

5.3.2 Solution Approach

Up to this time such solutions have not been practicable due to the limited wind information available, extensive computer times required to obtain solutions for adequate statistical samples of measured profiles, limited accuracy of the profiles themselves, and absence of correlations between mean wind and random turbulence responses. As a result, no methods have been tailored to this purpose which encompass the complete solution.

In order to obtain loads of known exceedance probability the following is required:

5.3.2 Solution Approach (Continued)

1. Development of high speed digital or analog simulations to obtain probability distributions of loads responses to statistical samples of mean winds.

2. Utilization of Jimsphere data to obtain power spectral densities of random turbulence and associated mean wind profiles.

3. Statistical ordering of vehicle responses to mean winds and random turbulence and correlation of these responses using vehicle responses to total wind profiles (Jimsphere).

The result would be bending load probability distributions for desired launch periods as functions of launch azimuth for critical vehicle stations for all times during boost. Such information would be very valuable to assist in the selection of design criteria for the advanced Saturn family of vehicles. In addition, correlations of these results could be made with synthetic profiles developed through statistical analysis of the winds themselves. This would provide consistent synthetic criteria for use in design studies and the relationships between primary wind statistical parameters and the statistical properties of vehicle responses.

5.4 GROUND WIND LOADS

5.4.1 Problem Definition

Computation of the ground winds loads response for large launch vehicles is one of the most problematic areas faced in the course of vehicle design. Limited knowledge of wind forcing functions and the coupled elastic response to random vortex shedding has historically required wind tunnel verification or total reliance on wind tunnel results. Vehicle size has progressed to the point where limited knowledge of scaling relationships seriously hinders application of wind tunnel results. Considerable time and effort is being spent on the current Saturn V program to learn more about this problem area including full scale ground winds testing using the AS-500F vehicle. An effort should be made to assemble all available knowledge from this and other sources and to extrapolate the results to the larger uprated configurations. In addition, wind tunnel tests of uprated Saturn configurations will be required.

5.4.2 Solution Approach

Apply these results to the larger uprated Saturn configurations and obtain revised ground wind loadings for design purposes.

5.4.2 Solution Approach (Continued)

1. Plan and conduct wind tunnel tests to establish ground wind load responses of uprated configurations.

2. Utilize this information and current Saturn V analysis and test results to extend analyses methods to include the larger uprated vehicles.

3. Revise ground wind loads for design purposes based on these results.

6.0 LOW COST LIQUID PROPELLANT POD

Large pressure fed liquid engine systems for boost-assist and for booster application are feasible, and approach the simple systems and structures of solid rocket motors.

6.1 PROBLEM DEFINITION

The use of liquid propellant pods for boost-assist on the MLV-SAT V-23(L) configuration provided a flexible method for vehicle uprating. However, the pod design was based on S-IC technology and concepts which result in higher costs than for solid motors.

The use of a pressure fed engine(s) and monocoque propellant tank structure can make the pods competitive with solid motors on a cost basis. The liquid propellants are cheaper than solid propellants, and transportation and handling of the dry stage is easier. The specific impulse of the liquid pod is better than the solid motors which provides a performance advantage.

6.2 SOLUTION APPROACH

The feasibility of a simplified liquid pod design which will be cost-competitive with solid motors should be determined. The design should be optimized from a cost and reliability standpoint, with lesser emphasis on performance.

Evaluation of pod components and systems to define performance characteristics and cost would lend not only to understanding the specific reasons for cost differences, but also to identify means of optimizing cost effectiveness. The potential of incorporating the following into the pod design should receive attention:

a. Simplified engine - eliminate turbopump (pressure-fed) - minimum valves and controls.

b. Simplified pressurization system - minimum complexity and components, and simple controls.

c. Simplified propellant tankage (monocoque structure) and thrust structure, made of inexpensive materials and fabrication techniques, requiring minimum inspection and testing.

d. Storable propellants and pressurization gases, packaged, and sealed.

These analyses of a low-cost pod should be compared in size and cost-effectiveness to the cryogenic pump-fed pod and solid motors used in the studies under NAS8-20266.