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STUDY OF ADVANCED MULTIPURPOSE LARGE LAUNCH VEHICLE - SUMMARY REPORT

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THE BOEING COMPANY
Huntsville, Alabama

for

AMES RESEARCH CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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ABSTRACT

Study results are summarized for a conceptual design analysis of "Advanced Multipurpose Large Launch Vehicles." This vehicle concept incorporates a single-stage-to-orbit main stage that orbits a million pound payload; and, with the addition of building block elements that include strap-on motors and a small injection stage, provides payload flexibility to nearly four million pounds. The objective of the study was to develop practical, representative vehicle configurations through a series of design and performance trade studies, and to assess this vehicle system in terms of technology needs and implications. A vehicle family is provided with its estimated performance for each of its possible flight configurations. The design and performance data developed in trade studies of design variables investigated in reaching the representative configuration are summarized.

Both multichamber/plug and toroidal/aerospike systems were considered for main stage propulsion. Strap-on boost assist elements investigated included solid motors and N_2O_4/UDMH pressure-fed pods. The structural impact of strap-ons to the main stage is described. The performance of strap-on configurations is given for both zero-stage and parallel-burn operation. The performance of configurations with a small orbital injection stage are presented. The implications of this possible future vehicle system on technology and resource requirements are assessed to provide data for technology planning, resource estimating, and mission analysis studies.

LIST OF KEY WORDS

Multipurpose Large Launch Vehicle
Toroidal/Aerospike
Multichamber/Plug
N_2O_4/UDMH Storable Pressure-Fed Pod
Solid Motors
Strap-ons
Vehicle Loads
Parallel Burn
Mass Fraction
Orbital Injection Stage
Performance
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Throttling
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FOREWORD

This report summarizes the findings of a "Study of Advanced Multipurpose Large Launch Vehicles" performed under Contract NAS2-4079. The study was conducted by the Launch Systems Branch, Space Division, of The Boeing Company from January 1, 1967, to September 30, 1967. The work was administered under the direction of Edward W. Gomersall of the Mission Analysis Division, Office of Advanced Research and Technology (OART).

The record of the study is presented in two reports. This report serves as a summary report that provides a concise account of the objectives, method of investigation, and significant results. The full account of the study is found in CR -73155, which provides a comprehensive document of the analyses conducted with their detail results.

All the propulsion systems data used in the performance of the study and shown in the reports were provided, at no cost, by:

Rocketdyne
Division of North American Rockwell Corporation
Canoga Park, California

Pratt and Whitney Aircraft
Division of United Aircraft Corporation
West Palm Beach, Florida

Aerojet-General Corporation
Space Booster Division
Sacramento, California

Lockheed Propulsion Company
Division of Lockheed Aircraft Corporation
Redlands, California

Their cooperation in contributing timely engine or motor data significantly enlarged the scope and value of the study results.

This report is also published as Boeing Document D5-13421-1.

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1.0 INTRODUCTION

It is an objective of the NASA's Office of Advanced Research and Technology (OART) to explore the practicality and potential of attractive new concepts to assure that the necessary technologies required to achieve the systems or maximize their effectiveness are developed. The purpose of these exploratory efforts is to provide data, and trade-off relationships for the concepts; not to select designs or propulsion systems, per se.

The "Advanced Multipurpose Large Launch Vehicle" (AMLLV) concept is an attractive launch vehicle design alternative suggested to accomplish future manned interplanetary explorations, extended lunar explorations, and large space station missions. This future vehicle system would take full advantage of technology advances and large vehicle design experience that have occurred since the early 1960's, especially the advent of altitude compensating aerospike or plug engines.

This report is the summary document that gives the key findings of a study, performed by The Boeing Company, under the sponsorship of NASA's OART. The study satisfied the need for detailed information, trade-offs and implications on the AMLLV system design.

The objectives of this study were threefold. First, working from the AMLLV concept, develop a practical, representative configuration through a series of trade-off and performance studies. Second, perform a more rigorous analysis on the selected configuration to substantiate its design and performance as well as provide detailed information on various system requirements. Finally, assess the implications and sensitivities of the vehicle to aid mission analyst, resource estimators, and technology planners.

The results of the study provide a set of considerations to be used in evaluating the objectives and achievements of continuing or future technology development programs. Equating the potential gains against the required effort provides OART with an effective tool for formulating long-range plans for technology research.

1.1 DEFINITION OF VEHICLE CONCEPT

The "Advanced Multipurpose Large Launch Vehicle" (AMLLV) is a concept for delivering one to four million pounds to low Earth orbit. This concept, as shown in Figure 1-1, features an LH_2/LOX main stage with single-stage-to-orbit capability of one million pounds. This main stage is used as the core stage of a building block system that incorporates both strap-on stages

1.1 (Continued)

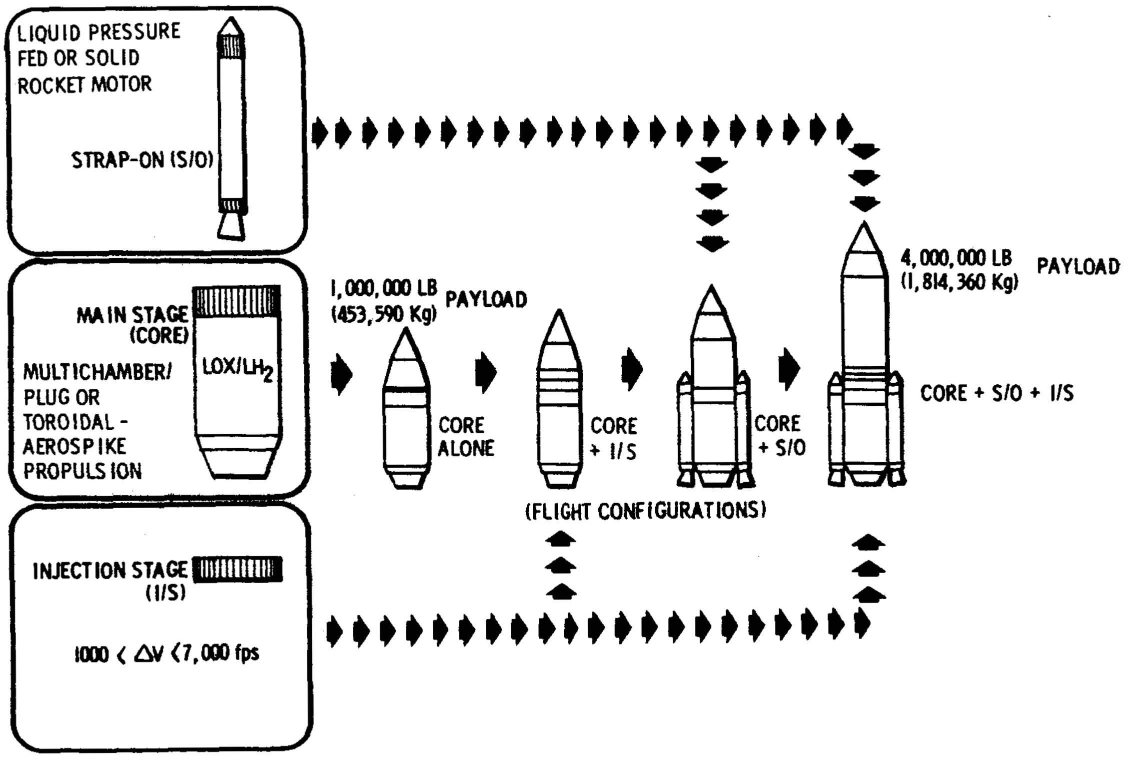


FIGURE 1-1 ADVANCED MULTIPURPOSE LARGE LAUNCH VEHICLE CONCEPT

and a small upper stage for orbital injection to achieve a broad range of payload capability. Main stage design includes an altitude compensating, toroidal/aerospike or multichamber/plug propulsion system. The strap-on options include solid motors or pressure-fed liquid motors for either parallel burn or zero stage operation to achieve basic payload versatility.

The orbital injection stage further enhances the payload versatility, reduces the main stage sensitivity to its inert weight, and provides orbital transfer and rendezvous capabilities. The total impulse provided by this stage would be varied by modularizing its propellant tanks to provide velocity increments of 1,000 to 7,000 feet-per-second over the payload range of one to four million pounds.

This basic concept formed the starting point of this study.

1.2 STUDY APPROACH

The initial trade study phase of the study investigated, in a preliminary manner, the options, alternatives, and parametric variations offered by the vehicle concept. These analyses formed the basis for selecting a logical configuration. This configuration, not necessarily optimum but a reasonable, representative definition of the concept, was then worked to a more detailed level of design in the second phase of the study to provide a more rigorous demonstration of the concept's validity and a better definition of its major systems. Finally, the implication of this configuration was further elaborated by considering aspects of its producibility and its sensitivity to technological improvements. Additional analyses were conducted to aid mission planners to assess the sensitivity of the vehicle to payload size variations and to outline a recoverable version of the design.

The logic employed in the trade study phase to evolve the representative configuration was a step-by-step process whereby first the main stage flight mode was established, and then, through a series of design trade studies of the major independent variables, the main stage design was chosen. The evaluation of the strap-ons and injection stage was then approached by defining representative designs that matched the main stage to establish their mass fractions and performance values. Performance trades were conducted for each of the logical configuration options, to determine the payload gains possible and select basic flight modes. Loads, stress, and weight analyses were performed for each major configuration option to define the structural changes required to the main LOX/LH_2 stage design.

The second phase of the study used this basic performance and structural requirements data to size and configure a main stage for a million pounds to orbit payload. A modularized injection stage and 260-inch solid rocket motors were sized to match the main stage and to provide nearly four-to-one payload versatility. Trajectories were finalized and detail load, stress, control, heating, and pressurization studies were completed to define the conceptual design and establish the final weight statement.

In the final phase, an assessment of the resource implications was conducted by studying the requirements for producing, developing, and lauching the selected vehicle stages. The sensitivity of the design to specific impulse and structural technology levels was determined by developing exchange ratios for each stage. The weight reduction possible with representative advanced structural materials was evaluated. The vehicle's sensitivity to the payload center of gravity locations and densities was determined. A recoverable version of the main stage design was outlined.

1.3 RELATIONSHIP TO OTHER STUDIES

The relationship of this study to other continuing and planned efforts within NASA was a guiding factor in determining the intent and format of the results of the study.

The study did not attempt to develop a highly optimized or sophisticated design in order to achieve superior performance or minimize cost, but rather, attempted to accomplish a system design and analysis of a reasonable approach, reflecting realizable technology, to a large launch vehicle. With this approach, actual data and projections from continuing and anticipated technology tasks were input into the study. Study output, then, includes system interactions, trade-off data, design point verifications, and technology implications directly and currently applicable to these technology activities. For example, this study:

a. Provides applicable data to on-going programs including the current advanced cryogenic propulsion technology program and the 260-inch solid rocket program. Propulsion parametric trade-off data, system interaction and interface data, and design point information contained herein should provide assistance in the guidance and future planning of these efforts.

b. Provides detail design data on load regimes and sizing applicable to the guidance and planning of advanced structures technology.

c. Provides insight into the manufacturing, transportation, test, and operational improvements required if systems of this size are to be developed. These requirements appear to be worthy of consideration in planning of future manufacturing or launch sites. The requirements suggested herein may provide upper limits in planning projections.

1.4 ASSUMPTIONS AND GUIDELINES

A due east launch from Atlantic Missile Range into a 100 nautical mile circular orbit was the primary flight mode.

The vehicle design applied man-rating design and control criteria. A payload density of five pounds per cubic foot was used in developing the trade study stage drawings and the final design configurations.

All propulsion data used in this study were obtained from or developed with the concurrence of the appropriate contractors.

2.0 SUMMARY

2.1 GENERAL

The major flight vehicle options of this study's Advanced Multipurpose Large Launch Vehicle (AMLLV) configurations are shown in Figure 2-1 together with their basic dimensions and weights.

The main stage single-stage-to-orbit capability is approximately one million pounds to 100 nautical mile orbit with either the toroidal/aerospike or multichamber/plug engine system. The Rocketdyne toroidal/aerospike system yields a gross payload of 1.019 million pounds. The Pratt and Whitney high pressure multichamber/plug system yields a payload of 0.994 million pounds. The high values of average engine specific impulse and stage mass fraction for the stage resulted in a payload to launch weight ratio of approximately 0.08 which is nearly double the value for the present Saturn V two stage (S-IC/S-II) vehicle.

The vehicle's LOX/LH_2 main stage has the structural capability to accommodate twelve 260-inch solid motors and the injection stage. The LOX and LH_2 tank skins and bulkheads are designed to meet the maximum loads encountered in all flight modes. By using forward skirt support and holddown provisions to react the strap-on thrust load, the mass fraction penalty incurred in accommodating the large strap-on impulse is minimized. The structural penalty is further reduced by using a series of forward skirt assemblies, one for each major configuration. These design innovations minimized the main stage mass fraction penalties to provide its multi-purpose flexibility to less than two percent (increased stage inert weight by 13 percent).

The basic LOX/LH_2 injection stage module is sized to 450,000 pounds of propellant as limited by the minimum liftoff thrust-to-weight of 1.18. The weight of the injection stage and payload when added to the fixed thrust main stage reduces the vehicle's liftoff thrust-to-weight ratio; 1.18 was set as the reasonable lower limit of this ratio based on launch drift considerations. This injection stage would increase the payload by 18 percent.

Twelve 260-inch solid rocket motors can be accommodated around the core stage. The motors, sized to attach to main stage frames at vehicle Stations 400 and 1955, have a propellant weight of 3.81 million pounds each. Their sea level thrust of 9.0 million pounds each and total burn time of approximately 130 seconds was determined to maximize payload without exceeding a maximum dynamic pressure of 1000 pounds per square foot.

GROSS PAYLOAD ≈ 1,000,000 LBS.
(454,000 KG.)
LIFTOFF WEIGHT = 12,800,000 LBS.
(5,811,200 KG)
LIFTOFF THRUST/WEIGHT = 1.25

CORE
LENGTH/DIAMETER = 2.20
MASS FRACTION ≈ 0.94
SEA LEVEL THRUST = 16,000,000 LBS.
(71,168,000 NEWTONS)
USEABLE PROPELLANT = 11,100,160 LB.
(5,039,473 KG)

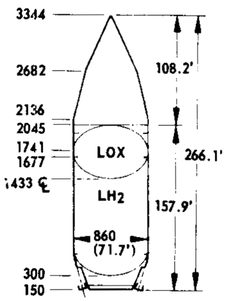
LOX = 9,514,460 LBS.
LH_2 = 1,585,700 LBS.

RESIDUAL PROPELLANT = 1% LH_2
= 0.1% LOX

ULLAGE VOLUME = 3%

MULTICHAMBER/PLUG
PRATT & WHITNEY

TOROIDAL 'AEROSPIKE
ROCKETDYNE



CORE VEHICLE

GROSS PAYLOAD ≈ 1,180,000 LBS.
(535,720 KG)
LIFTOFF WEIGHT = 13,559,300 LBS.
(6,155,922 KG)
LIFTOFF THRUST/WEIGHT = 1.18

CORE:

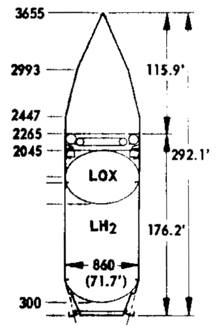
MASS FRACTION ≈ .94

INJECTION STAGE:

TOTAL WEIGHT = 548,780 LBS.
(249,146 KG)
MASS FRACTION = .82
VACUUM THRUST = 500,000 LBS.
(2,224,000 NEWTONS)
USEABLE PROPELLANT = 450,000 LBS.
(204,300 KG)

LOX = 385,700 LBS.
LH_2 = 64,300 LBS.

RESIDUAL PROPELLANT = 3%
ULLAGE VOLUME = 3%



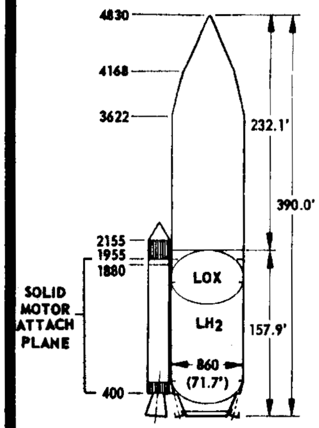
CORE + INJECTION STAGE

GROSS PAYLOADS ≈ 3,472,000 LBS.
(1,576,300 KG)
LIFT-OFF WEIGHT = 66,257,000 LBS.
(30,080,680 KG.)
LIFT-OFF THRUST/WEIGHT = 1.63

CORE:

MASS FRACTION ≈ 0.94

STRAP-ON MOTORS:
DIAMETER = 260 INCHES
NUMBER = 12
MASS FRACTION = 0.90
USEABLE PROPELLANT = 3,810,000 LBS/MOTOR
SEA LEVEL THRUST = 9,000,000 LBS/MOTOR
(40,032,000 NEWTONS/
MOTOR)



CORE + STRAP-ON'S

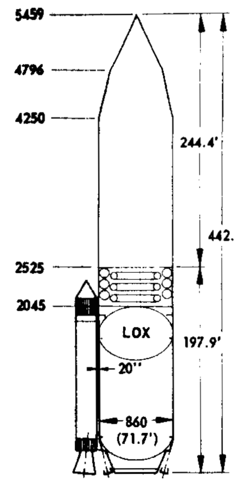
GROSS PAYLOAD = 3,740,000 LBS.
(1,698,000 KG.)
LIFT-OFF WEIGHT = 67,924,000 LBS.
(30,837,500 KG.)
LIFT-OFF THRUST WEIGHT = 1.59
CORE:
MASS FRACTION ≈ 0.940
STRAP-ON MOTORS:

NUMBER = 12

INJECTION STAGE:

TOTAL WEIGHT = 1,614,000 LBS.
(732,756 KG.)
VACUUM THRUST = 1,500,000 LBS.
(6,672,000 NEWTONS)
USEABLE PROPELLANT = 1,404,000

LOX = 1,203,000 LBS.
LH_2 = 201,000 LBS.
MASS FRACTION = .87



CORE + STRAP-ON'S + INJECTION STAGE

*ALL PAYLOADS SHOWN FOR ρ = 5#/FT^3 100 NAUTICAL MILE CIRCULAR ORBIT

FIGURE 2-1: ADVANCED MULTIPURPOSE LARGE LAUNCH VEHICLE FAMILY

2.1 (Continued)

The injection stage, with a full complement of solid motors, is made by stacking three modules of the basic injection stage sized for the core alone configuration. The three-wafer configuration gives a payload increase of 6.25 percent as compared to the core plus twelve solids configuration.

Figure 2-2 shows the payload as a function of launch weight for each of the major configurations and for strap-on configurations with intermediate numbers of the 260-inch solid motors. The launch weight varies from 12.8 to 67.5 million pounds; the payload varies from .994 to 3.75 million pounds. A greater maximum payload could be obtained with larger diameter solid motors.

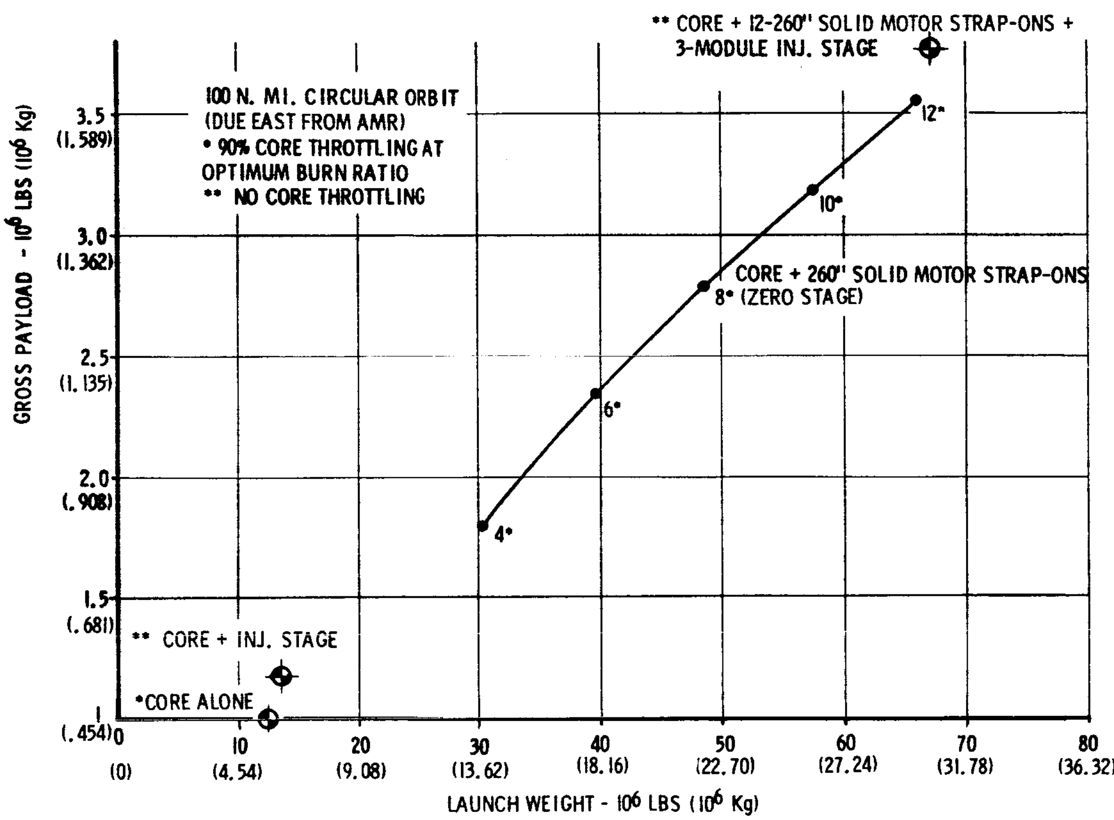


FIGURE 2-2 PAYLOAD SUMMARY

2.2 MAIN STAGE

The main stage, sized to orbit one million pounds to low Earth orbit, has 16.0 million pounds of sea level thrust with 11.1 million pounds of propellant.

2.2 (Continued)

Its inert weight (stage drop weight) of only 634,000 pounds results in a stage mass fraction of approximately 0.946 (numbers quoted are for the toroidal/aerospike main stage). Comparing this to an S-IC/S-II vehicle, the AMLLV has approximately four times the payload capability with only a 40 percent increase in inert weight. Physically, the AMLLV has over twice the diameter of the S-IC/S-II but is approximately 60 feet shorter.

Figure 2-3 is an isometric sketch of the main stage. The structure is principally conventional skin-stringer-frame construction using 2219 aluminum for the propellant tanks and 7075 aluminum for the forward skirt and thrust structure. The design has a forward LOX tank to minimize control requirements. Positioning the LOX tank aft would have resulted in a maximum thrust vector deflection of over 20 degrees as compared to less than four degrees with the tank forward. Both the aerospike and plug engine systems favor a low length/diameter (L/D) stage design which allows efficient structural design of the propellant tanks.

The common bulkhead is a sandwich structure designed to take buckling loads that occur near propellant depletion. This construction was determined to be more efficient than increasing the LOX tank pressure to maintain the bulkhead in tension. The bulkheads and tank skins are designed for loads encountered during the zero stage operation with twelve solid motors. Flight conditions for the main stage result in the maximum compressive loads for the LH_2 tank shell and thrust structure. Since the forward skirt is subjected to a wide range of loads from 4,000 pounds to 16,000 pounds per inch, the design and use of individual skirts was suggested to minimize the weight penalties for each major configuration.

The use of the forward skirt for vehicle support and solid motor thrust take-out minimizes ground wind and emergency rebound main stage loads and in-flight bending moments for the core plus solid configurations. The forward skirt reaction point provides a short load path between the support or thrust take-out connections and the large inertia payload and LOX tank elements. Although this system does impose a new type of launch stand design, it is significant in minimizing the structural weight of the flight stage.

Designing the main stage's structure to accommodate twelve 260-inch solid rocket motors reduces the stage's mass ratio by approximately two percent. This reduction in structural efficiency requires the core stage to be approximately 14 percent larger than if it were designed without the capability to accept the strap-on stages.

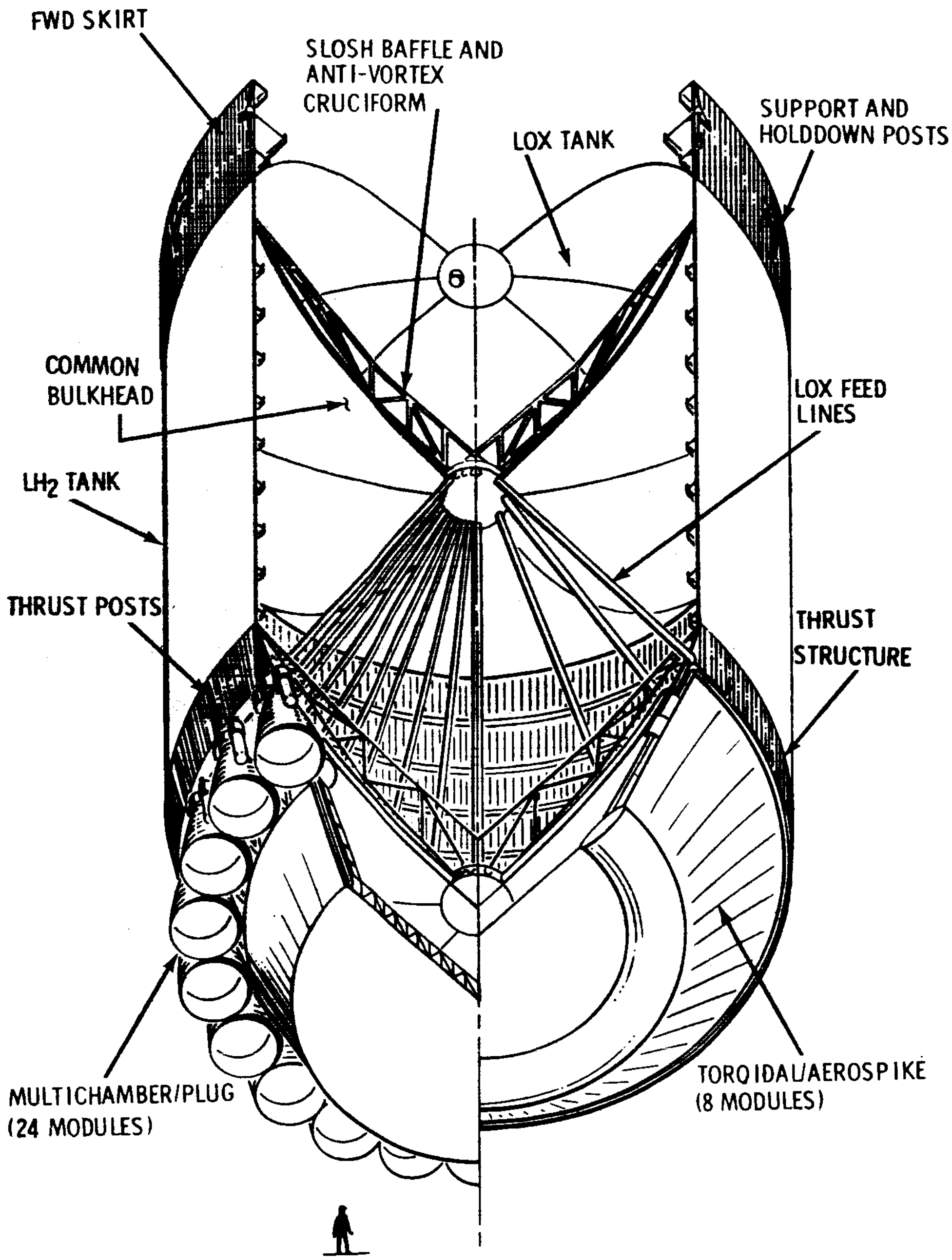


FIGURE 2-3 LOX/LH_2 MAIN STAGE

2.2 (Continued)

The thrust structure was the only element of the stage whose design was influenced by the engine systems. For the multichamber/plug engine system, a thrust post is required for each engine module to react the concentrated thrust load. When compared to the thrust structure for the same thrust level toroidal engine system, the multichamber's thrust structure is 6,000 pounds heavier. A representative thrust structure skin panel was analyzed to determine its reaction to the acoustical loading encountered in the twelve 260-inch solid motor configuration. The three sigma peak static pressure was estimated at 3.5 psi which results in maximum cyclic stress of 3,000 pounds per inch which is well within the fatigue life of the 7075 aluminum plate. The overall sound pressure level was estimated to vary from 179 dB at the base of the plug to 160 dB at the forward flange of the forward skirt.

Core control requirements of 3.4 degrees total thrust vector deflection were determined based on design wind and accounting for center of pressure, center of gravity, and thrust vector off-sets. This requirement is within the hinging capability of the multichamber module and either the LOX or hot gas injection system for the toroidal/aerospike engine system.

2.2.1 Trade Studies - Performance

Single-stage-to-orbit trajectory studies showed that with continuous burning with deep throttling (to ten percent thrust) of the engines, payload capabilities near that possible with burn-coast-burn modes could be achieved as shown in Figure 2-4.

Improved mission reliabilities would be expected with this mode over a burn-coast-burn because it would eliminate the restart operation. For the design and trajectory mode defined, a gross payload to launch weight function of approximately 0.08 is estimated with either propulsion system.

Throttling during a direct ascent trajectory to orbit significantly reduces the large thrust vectoring loss encountered when the flight path is turned to meet the required orbital cutoff conditions. Throttling also reduces the maximum longitudinal acceleration from 14.0 to approximately 6.5 g's for the selected 90 percent throttling flight mode.

The sensitivity of the main stage's launch weight to engine specific impulse and stage mass fraction (λ') for a constant million pound payload is shown in Figure 2-5. As the vehicle mass fraction or specific impulse is decreased, the vehicle becomes more sensitive to the other parameters.

2.2.1 (Continued)

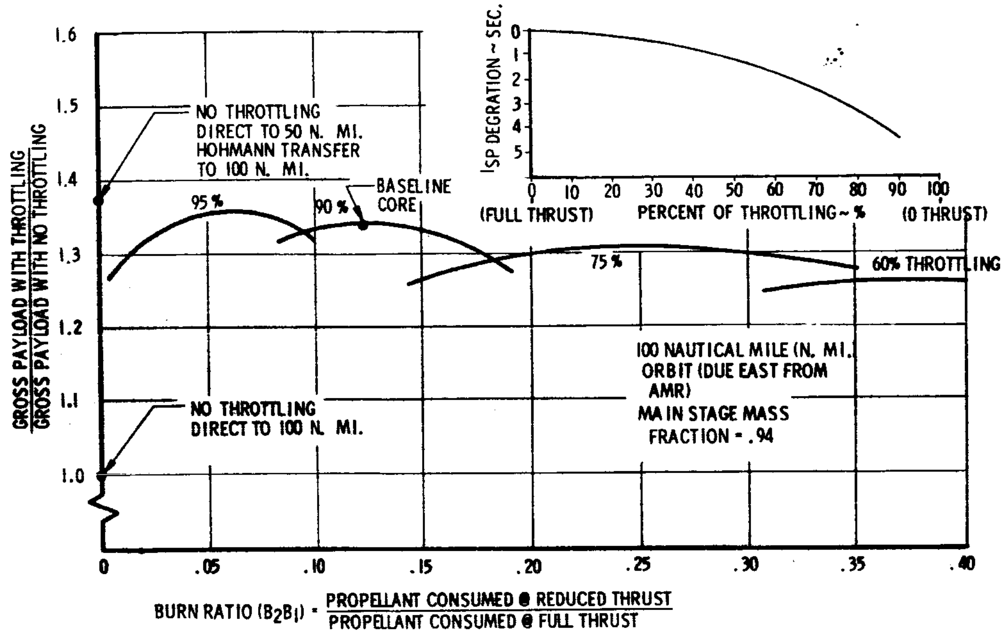


FIGURE 2-4 SINGLE STAGE TO ORBIT TRAJECTORY OPTIMIZATION

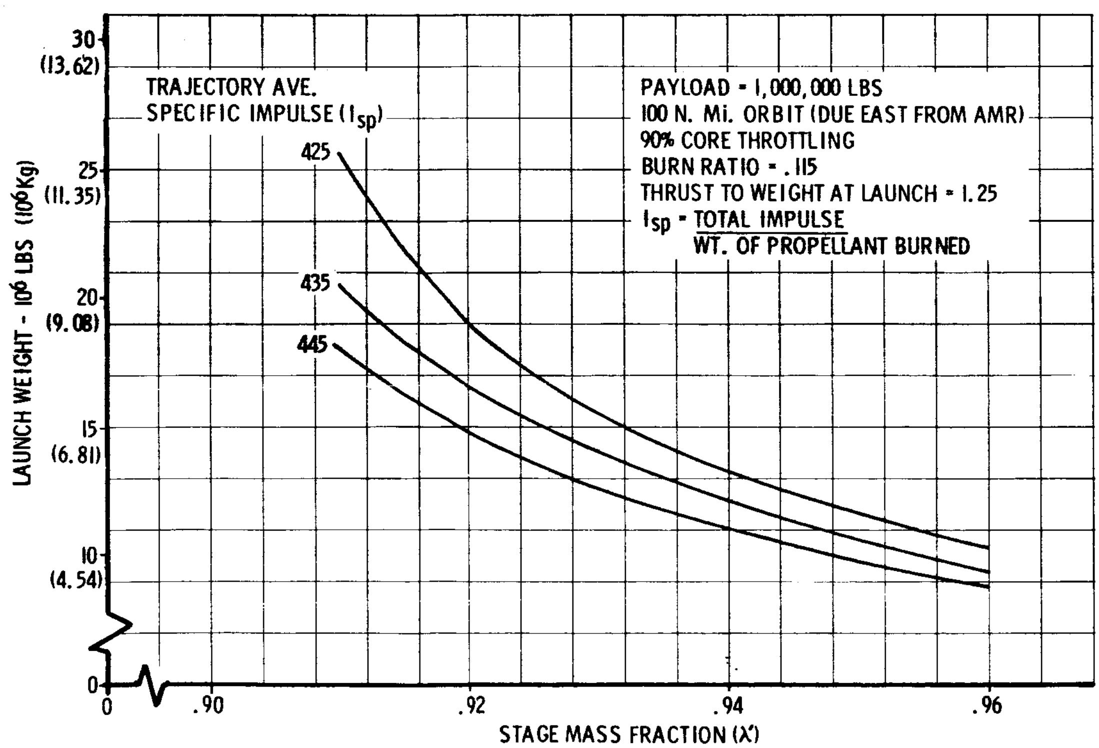


FIGURE 2-5 MAIN STAGE MASS FRACTION AND SPECIFIC IMPULSE EFFECTS ON LAUNCH WEIGHT

2.2.2 Trade Studies - Design

The stage design was evolved through a series of trade studies, where the influence of the major independent variables of mixture ratio, stage length to diameter ratio, engine chamber pressure, number of modules and tank pressures were investigated. Each trade considered the total consequence of the perturbed parameter as it affected performance and weights. A complete set of vehicle loads was developed for each point studied. Stress analysis was repeated for each case and a new stage weight was estimated. Aerodynamic characteristics were adjusted for each case where vehicle size was varied.

The mixture ratio investigation considered both engine performance effects and stage weight and size changes. A mixture ratio of 5:1 resulted in the best effective engine specific impulses while higher mixture ratios gave the lighter stage weights. An optimum value of 6:1 was determined for either propulsion system.

Stage length to diameter (L/D) influence revealed that both the engine system and stage design favored low L/D values until the LOX tank side wall was eliminated. Then the weight penalties associated with the flatter bulkheads negated any further improvement in engine specific impulse obtained from the larger diameters.

The engine system chamber pressure studies investigated the effects of variations in the engine weight and performance for Rocketdyne's regeneratively cooled multichamber and toroidal systems. Improvements in overall performance were noted until a chamber pressure of 2000 psia was reached; then the payload benefit leveled off. Rocketdyne indicated that 2000 psia was the upper limit for regeneratively cooled systems. A check point was run for the Pratt and Whitney high pressure (3000 psia) multichamber system with hinged modules that are transpiration cooled. The payload performance was found to be essentially equivalent to the 2000 psia Rocketdyne regeneratively cooled multichamber system.

The number of module trade studies, performed with both the Rocketdyne and Pratt and Whitney multichamber engine systems data, showed that payload performance was independent of the number of modules used. Engine performance slightly favored fewer modules but the performance gain was offset by the accompanying engine and stage weight increase.

The trade study of LH_2 tank pressures showed that 28.0 psia ullage pressure gave the lightest stage weight. Stage structure and pressurization system and gas weight effects led to the determination of this optimum value which is sufficient to meet the Net Positive Suction Head requirements as specified by the engine contractors. Table 2-I lists the range of each

2.2 (Continued)

parameter that could be accepted without penalizing the payload by more than one percent from the maximum value determined.

TABLE 2-I
PARAMETRIC DESIGN STUDY RESULTS

Variable	Range of Variable Within 1% of Max Gross Payload	
	Multichamber	Toroidal
Mixture Ratio	5.2:1 - 6.4:1	5.65:1 - 6.6:1
Stage Length/Dia (L/D)	2.24 - 2.80	2.20 - 2.78
Chamber Pressure (psia) (regeneratively cooled)	2000 - 3000	1950 - 2800
No. Modules Rocketdyne (P_c = 2000 psia) Pratt & Whitney (P_c = 3000 psia)	 8 - 16 12 - 24	
LH_2 Ullage Pressure (psia)	18.2 - 35.0	18.2 - 35.0

2.3 STRAP-ON CONFIGURATIONS

The 260-inch solid motor having 9,000,000 pounds of thrust and 3,810,000 pounds of propellant with a fifty percent regressive thrust-time trace was determined to be most suited for the strap-on configuration. The motor requirements are within the projected technology capabilities of solid motor design and maintain the vehicle's maximum dynamic pressure near 1,000 pounds per square foot. Twelve motors are the maximum number of 260-inch solids that can be accommodated around the core periphery. If future technology programs show that larger diameter motors can be designed and handled, further payload gains are possible. For example, as shown in Figure 2-6, ten 372-inch motors could provide payloads of over four million pounds within the maximum dynamic pressure limit of 1,000 pounds per square foot.

2. 3 (Continued)

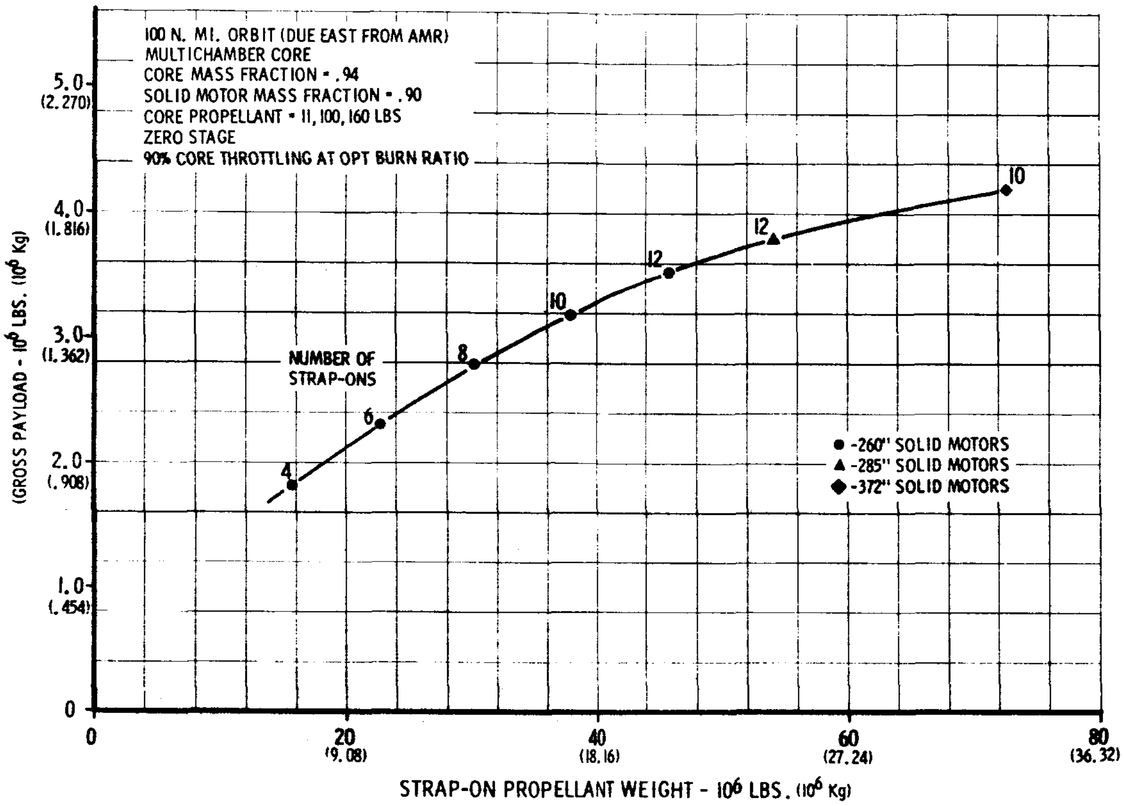


FIGURE 2-6 STRAP-ON PERFORMANCE

The zero stage mode of operation, i.e., the solid motors burn as a first stage with the main stage igniting during motor tailoff, gives the maximum payload with the minimum flight load conditions as compared to parallel burn operation where the core and solids operate together. Figure 2-7 shows these advantages for 260-inch solids. However, it was determined that using a flight mode where the core is ignited at launch and operates at a very reduced thrust level (ten percent thrust) during solid motor burn, a payload capability within one percent of that with zero staging is obtained. Since the flight ignition requirement of the core engines is eliminated with this mode, an improvement in mission reliability would be expected.

2.3 (Continued)

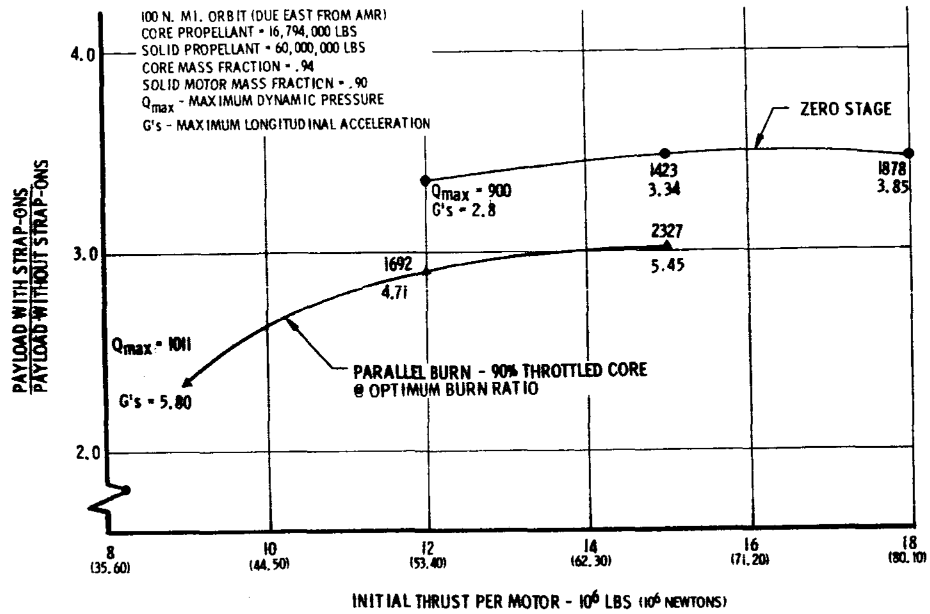


FIGURE 2-7 STRAP-ON PERFORMANCE OPTIMIZATION WITH 260-INCH SOLID MOTOR (TRADE STUDY DATA)

The parametric investigations of strap-on configurations included both pressure-fed UDMH/N_2O_4 liquid pods and solid motors over a range of diameters from 156 to 330 inches. Both systems offer approximately the same payload capabilities as measured by payload to launch weight ratios. The pressure-fed system considered in the study, shown in Figure 2-8, incorporated a single engine with liquid injection thrust vector control and a hot gas pressurization system. Final selection between the liquid and solid propellant strap-ons would require cost studies as well as technology confidence appraisals which at this time favor solid motors because of available data. However, both these considerations were beyond the scope of this study.

The addition of over 40 million pounds of solid motors generating over 100 million pounds of thrust requires main stage structural capabilities greater than necessary for single-stage-to-orbit operation. Increased load conditions on the cylindrical side walls are a consequence of higher bending moments created by the longer payload and higher dynamic pressures, and higher longitudinal forces created by the strap-on thrust. Increased loads in the tank bulkheads are a consequence of hydrostatic pressure created when the full tanks are subjected to the longitudinal acceleration of 3.1 g's at the solid motor cutoff flight conditions.

2.3 (Continued)

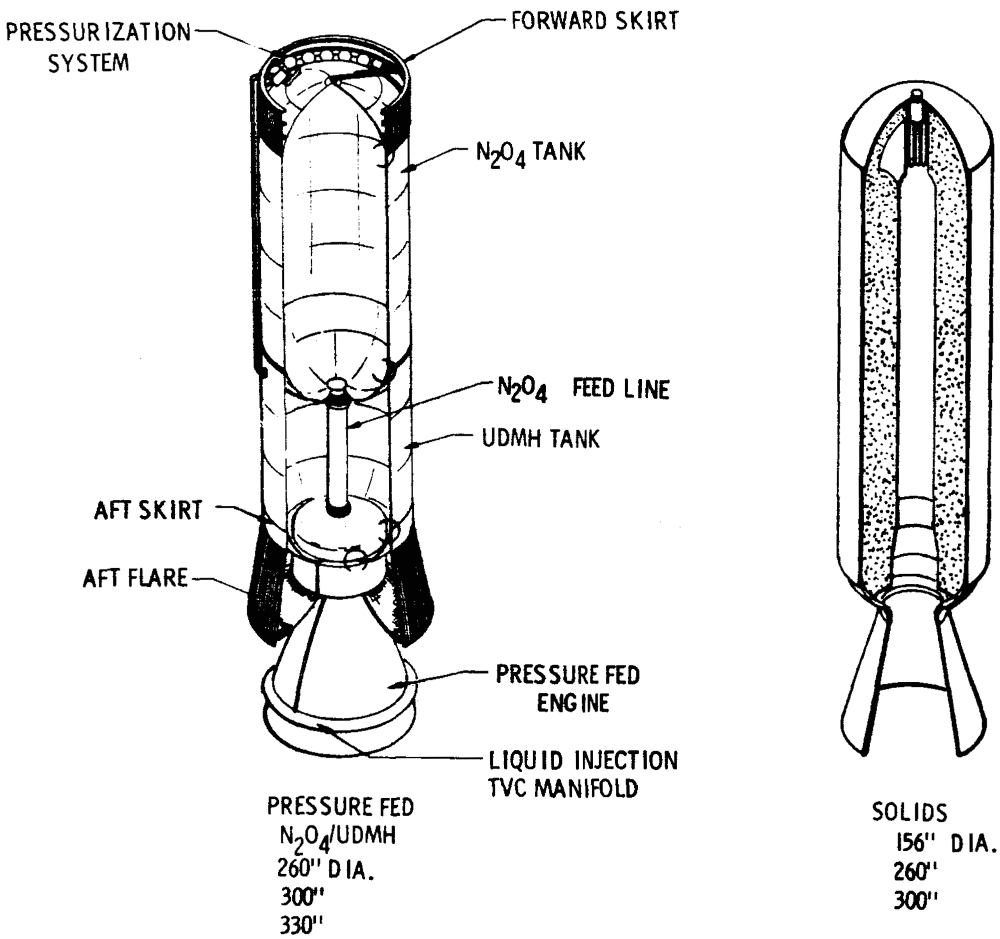


FIGURE 2-8 STRAP-ON STAGES

An effective means of minimizing the side wall structural increase is to react the solid motor thrust in the forward skirt rather than in the thrust structure. Figure 2-9 plots the designing compressive load for both the aft and forward thrust take-out with the loads encountered for core alone, single-stage-to-orbit operation for reference. It is seen that the loads with forward thrust take-out are less than with core alone operation for all stations aft of 2173. With aft attachment, the loads over this part of the vehicle are more than double the values for core alone operation. Forward of Station 2173, the loads are essentially the same with either strap-on thrust take-out arrangement.

2.3 (Continued)

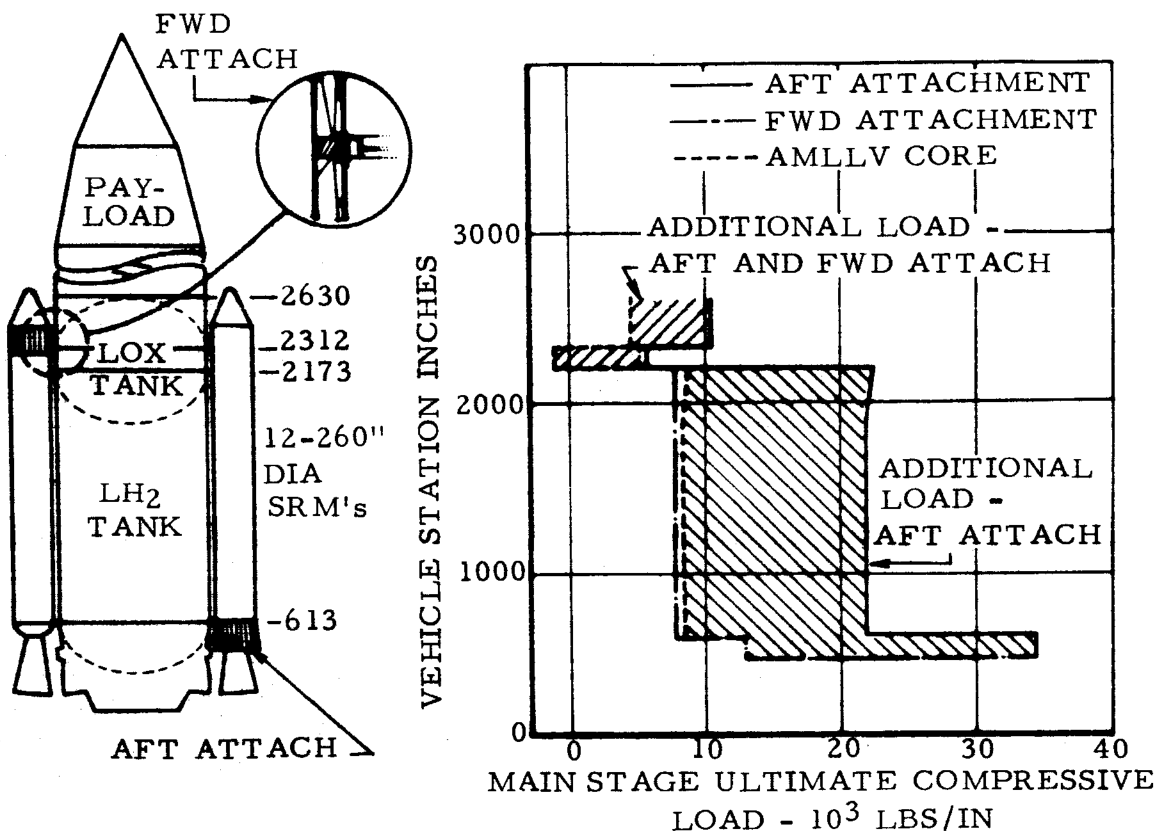


FIGURE 2-9 STRAP-ON ATTACHMENT LOADS

The effect of structural beef-up required in the forward skirt, Stations 2312 to 2630, can be avoided by using two forward skirt assemblies: one light-weight assembly for core alone operation and one heavier assembly for use with the strap-on configurations.

The structural weight increases due to the thicker bulkhead skins needed to contain the higher fluid pressures are independent of the attachment concept used. Table 2-II lists the percent changes in the bulkhead weights required with the zero stage flight modes.

2.3 (Continued)

TABLE 2-II
MAIN STAGE WEIGHT PENALTIES
FOR STRAP-ON OPERATION (1)

	Weight Increase Required (%)
LOX Tank Upper Bulkhead	2.75
Common Bulkhead	6.10
LH_2 Tank Lower Bulkhead	.79
TOTAL	9.64

(1) Strap-On Configuration: 12-260-inch Diameter Solid Motors - Zero Stage

2.4 INJECTION STAGE CONFIGURATIONS

The use of an orbital injection stage to increase payload versatility and reduce configuration sensitivities was considered for both core and core-plus-strap-on configurations. A LOX/LH_2 stage with toroidal propellant tanks and extendable nozzle high-pressure engine system was selected as a representative design solution (see Figure 2-10) for matching the core stage diameter. This design also lends itself to a modularizing flexibility where a series of propellant tank wafers are stacked and additional engines moun ted to a common thrust beam. A stage mass fraction of .82 was obtained for the single wafer configuration shown and improved to .88 when four wafers were stacked. Although technology problems are noted in the fabrication of the toroidal tank, the design can be considered representative for an advanced vehicle parametric study.

2.4 (Continued)

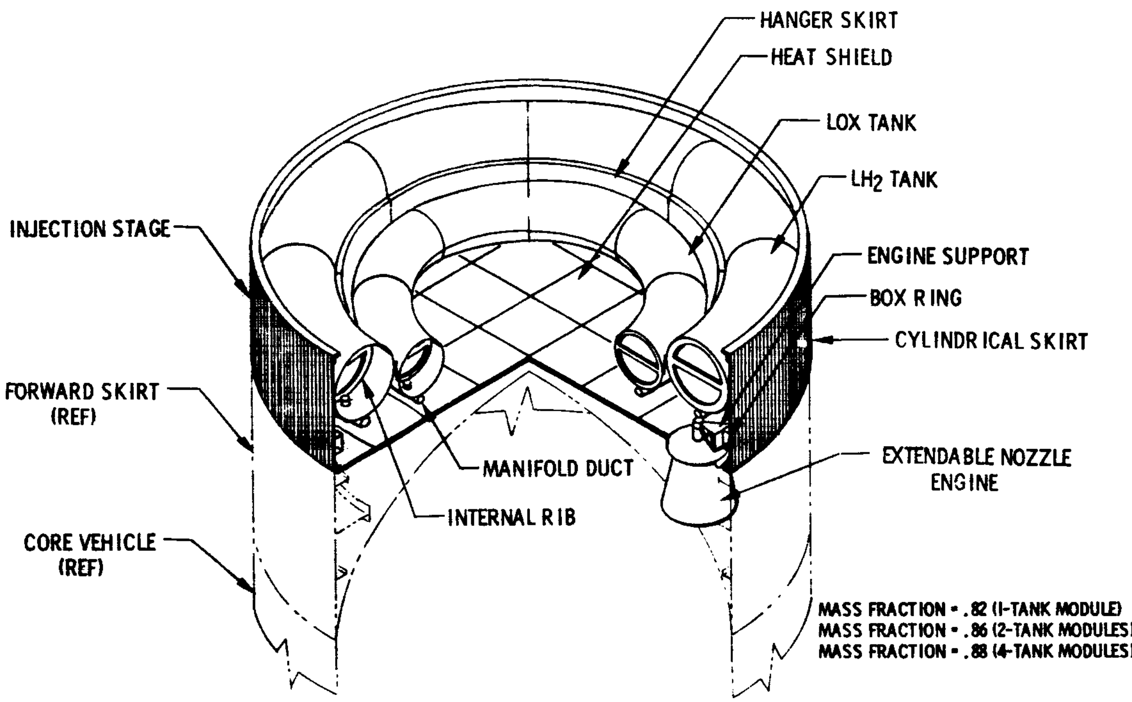


FIGURE 2-10 ORBITAL INJECTION STAGE

Performance studies using a range of thrust-to-weights and core throttling modes were conducted to determine possible payload capabilities. For configurations where the injection stage is added to only the main stage (no strap-ons), the payload improvement is constrained by the practical low limit of vehicle liftoff thrust-to-weight (T/W_O). The main stage was sized for a liftoff thrust-to-weight of 1.25. When an injection stage is added to that stage, its weight plus the additional payload weight reduces the liftoff thrust-to-weight. For this study, the limit was set at T/W_O = 1.18. At this value, the payload increase offered by the injection stage was limited to 18 percent for 100 nautical mile orbit missions as shown in Figure 2-11. Maximum performance was determined with a trajectory mode without main stage throttling. If the main stage had been found to have a lower mass fraction, the injection stage would display a better payload performance benefit.

2.4 (Continued)

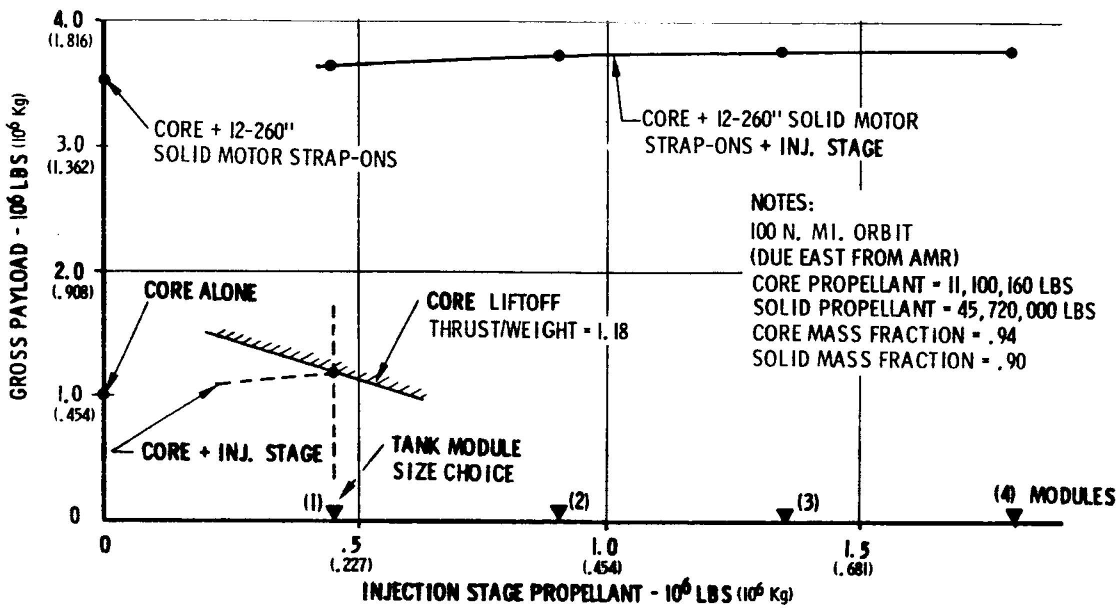


FIGURE 2-11 VEHICLE PERFORMANCE WITH INJECTION STAGE

For missions to higher Earth orbits, e.g., 300 nautical miles, the injection stage becomes more desirable since it is a practical approach for performing a Hohmann transfer type trajectory and provides a short coupling, high-response control system for accomplishing the final orbit injection maneuver. With the addition of the injection stage, the basic main stage can orbit one million pounds in the 300 nautical mile orbit. Whereas, without the stage, a direct injection of only 780, 000 pounds would be possible.

When the injection stage is added to the main stage plus solid motor configuration, the low T/W_o limit is not an influencing factor. The basic configuration has a T/W_o greater than 1.6 and the addition of an optimum injection stage (third stage) weight is reached with a T/W_o of 1.59. Stacking three wafers of the injection stage for the strap-on configuration provides nearly all the payload gain possible with the additional stage. This is six percent as compared with 6-1/2 percent (see Figure 2-11). Again, it was determined that the injection stage offers only a minimal payload performance gain for the 100 nautical mile orbit. Its main advantage is maintenance of payload capability to the higher Earth orbit.

2.5 VEHICLE SENSITIVITIES AND IMPLICATIONS

2.5.1 Advanced Structures

The AMLLV configuration performance and weights were developed using currently available and accepted structural materials and design approaches. However, with continued technology development, several alternate materials and construction methods could be used with confidence. They would offer further inert weight improvements in the core stage. With the use of sandwich construction with 6Al-4V titanium face sheets and approximately eight-inch thick aluminum core, a 50 percent reduction in thrust structure and forward skirt weight and a 34 percent reduction in LH_2 tank side wall weight is possible. An alternate advanced structure for the forward skirt assembly for the main stage application is the use of beryllium sandwich and a special jettisonable vehicle support fitting. This approach would yield an 83 percent reduction in skirt weight. No material substitution for 2219-T87 aluminum was identified for the LOX tank elements. Aluminum is the only recognized reliable, LOX compatible material available.

The application of advanced structure has the potential of reducing stage inert weight by 15 percent (an increase of stage mass fraction of 0.01).

2.5.2 Design Parameters

Payload sensitivities were determined for each stage by arbitrarily setting an off-design value of a major design factor and then optimizing the trajectory to establish the associated payload change. Figure 2-12 shows the sensitivities determined for the main stage. The partial derivative of most general interest is the trade-off of main stage inert weight to specific impulse (I_{sp}) while holding payload constant. The value was determined to be 7,656 pounds for each 1.0 second I_{sp}.

2.5.3 Payload Density

The final main stage structure is insensitive to payload densities over the range of 5 to 50 pounds per cubic foot. In-flight bending moments do not affect design loads over this range. Reducing densities to 2 pounds per cubic foot does affect the structure, but only reduces the mass fraction by 0.0006. The main influence of the low payload density is that it doubles the main stage control requirement as shown in Figure 2-13.

2.5.3 (Continued)

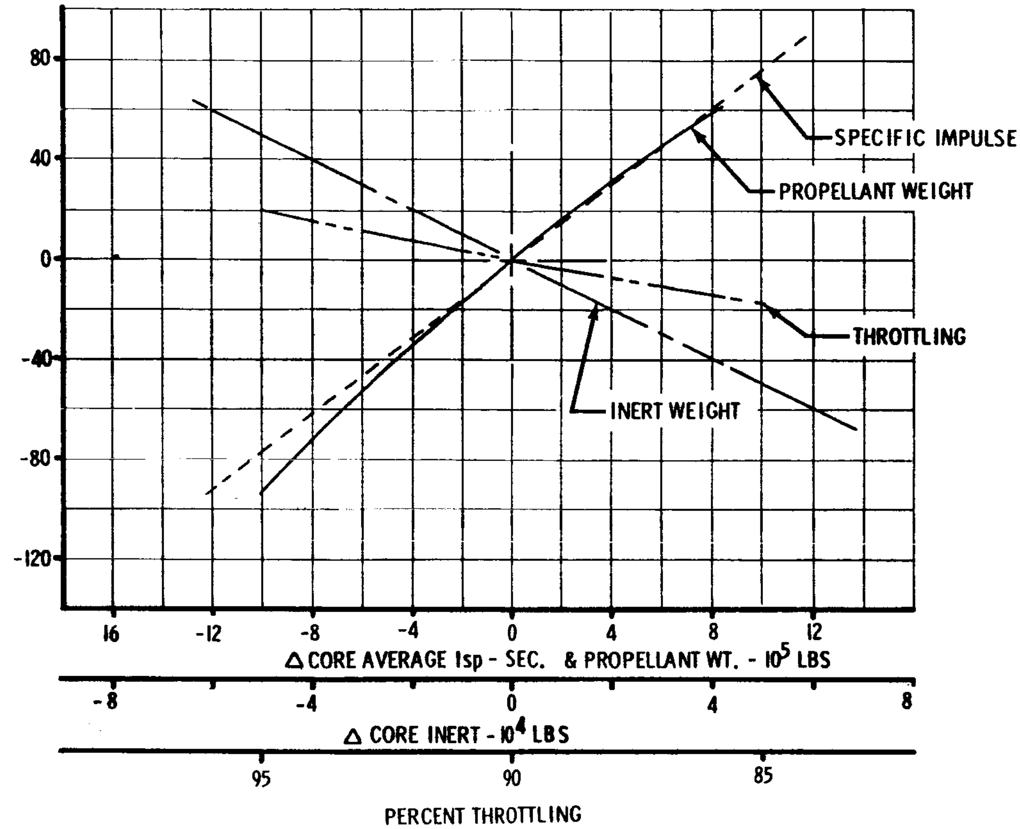


FIGURE 2-12 EFFECTS OF DESIGN PARAMETERS ON MAIN STAGE PAYLOAD PERFORMANCE

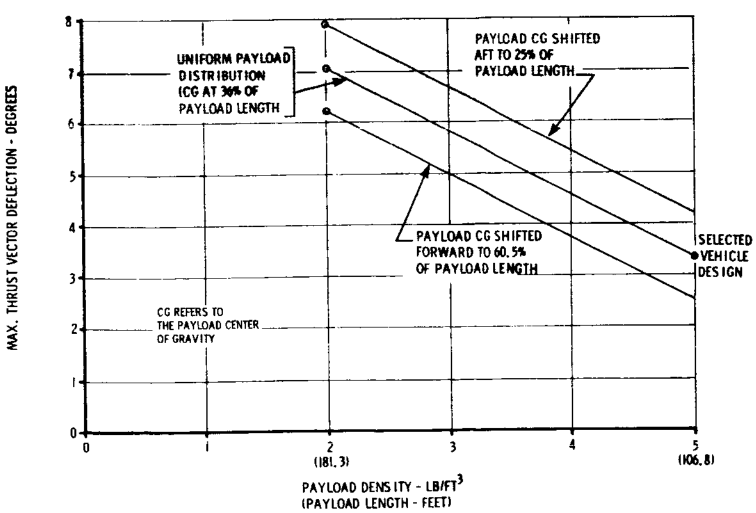


FIGURE 2-13 CONTROL REQUIREMENTS VERSUS PAYLOAD DENSITY

2.5.4 Resource Implications

A survey of the AMLLV's development, production, and launch requirements determined that the vehicle implementation would be possible with contemporary manufacturing and facility technology and could be accomplished as follows:

a. Main stage fabrication at the NASA Michoud site (or its equivalent located on a navigable waterway) in a new factory building;

b. Development testing of the main stage and injection stage in new dynamic and structural test facilities constructed adjacent to the factory building;

c. Injection stage fabrication in the existing factory building at Michoud;

d. Transportation of all vehicle elements from factory to launch sites by ocean-going towed barges;

e. Launch at Cape Kennedy from new launch facilities. An off-shore launch area at the Atlantic Test Range may be required based on the acoustic siting criteria. Specially designed hoisting devices are required to handle the 2,000 ton solid motors and 400-ton main stage.

2.5.5 Recoverable Design

The recovery of the AMLLV's stages may be possible with the addition of deceleration and stabilization devices to the basic designs. The core stage launch weight would be increased by 16 percent to off-set the payload capability lost with the addition of the necessary stabilization and let-down systems in the stage drop weight. This amount of increased weight could be reduced by adopting a conical shape stage design as shown in Figure 2-14 that is inherently stable in lieu of the present cylindrical shape. This design approach, derived from previous NASA studies, may be desirable if studies show that cost savings with reuse are beneficial.

2.5.5 (Continued)

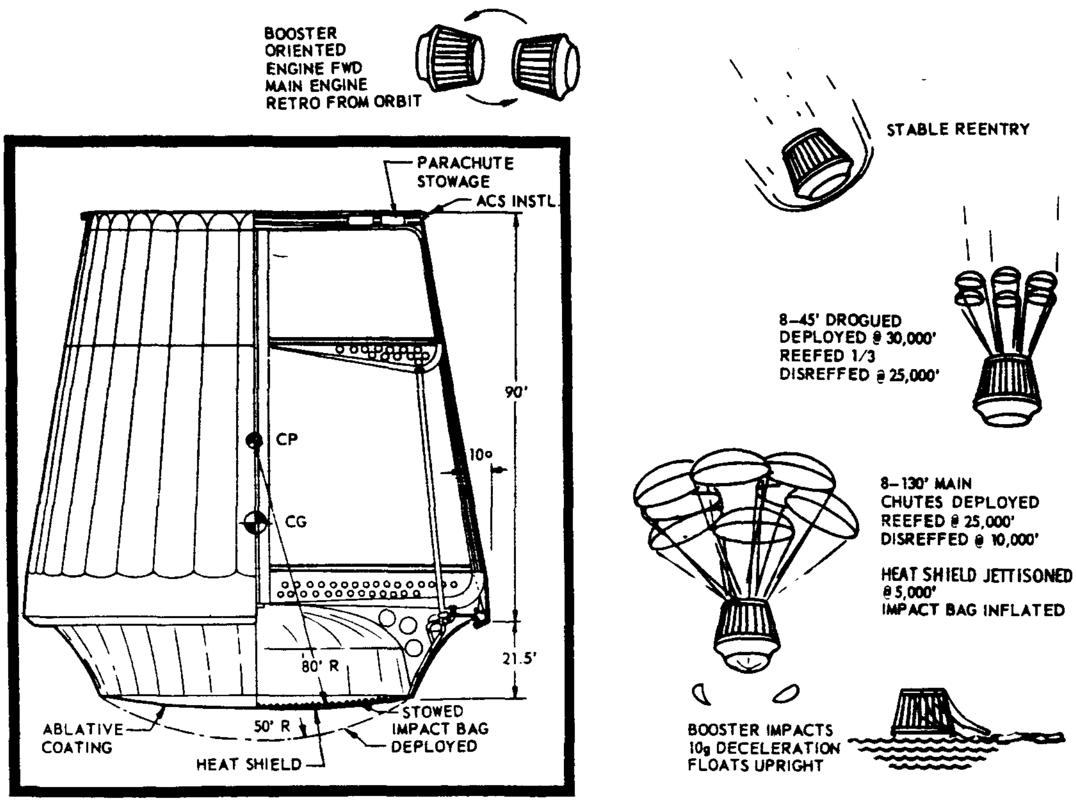


FIGURE 2-14 RECOVERABLE CONFIGURATION

3.0 CONCLUSIONS AND RECOMMENDATIONS

The results of the study establish that the AMLLV concept is a potentially attractive and practical design approach. Based on present knowledge of toroidal/aerospike and multichamber/plug engine performance, a single-stage-to-orbit vehicle with the structural capability to accommodate strap-on systems can have a gross payload to launch weight ratio which is nearly double the value of the present S-IC/S-II vehicle. The main stage with 260-inch solid motor strap-on configuration options can increase the capability by more than three hundred percent.

The payload capability of the main stage with either of the engine concepts considered was determined to be equal within the accuracy of the study. The study scope did not include influence of costs, development risk, or use of the basic engine systems for other vehicle applications which are considerations that would finally influence the eventual engine selection.

The LOX/LH_2 main stage design size and inert weight was determined to

3.0 (Continued)

be essentially independent of the engine selection. However, engine throttling capability to the level of ten percent thrust was required to maximize payload and limit longitudinal accelerations.

The use of an injection stage was determined to be ineffective for 100 nautical mile circular orbit missions. The maximum payload increased by only six percent over configurations without this stage option. For missions to higher Earth orbits, the use of the injection stage with Hohmann transfer flight modes was effective.

For strap-on configurations, the basic zero stage mode was preferred from both payload and main stage structural weight considerations. Igniting the core vehicle at a low thrust level (ten percent) simultaneously with the solids at liftoff may be a means of improving reliability and reducing main stage engine development cost since this mode eliminates altitude start requirements. The payload penalty associated with this mode is less than one percent as compared to the full zero stage mode.

Designing the main stage for reacting vehicle support and strap-on thrust loads through the forward skirt was a new design innovation identified for minimizing stage structural weight.

Recommendations

This study provides only design, performance, and systems analysis of a reasonable large launch vehicle approach. Costs and cost sensitivities were not treated in this analysis. Clearly, cost is a criterion in launch vehicle planning. For this reason, it is recommended that a study activity to define the costs and cost sensitivities of this vehicle concept be initiated. With both technological and economic aspects evaluated, NASA will be better able to guide existing technology programs and plan for future efforts.