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HEAVY LIFT LAUNCH VEHICLES FOR 1995 AND BEYOND

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Program Development

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16. ABSTRACT

A Heavy Lift Launch Vehicle (HLLV) designed to deliver 300,000 lb to a 540 n.mi. circular polar orbit may be required to meet national needs for 1995 and beyond. The vehicle described herein can accommodate payload envelopes up to 50 ft diameter by 200 ft in length. Design requirements include reusability for the more expensive components such as avionics and propulsion systems, rapid launch turnaround time, minimum hardware inventory, stage and component flexibility and commonality, and low operational costs. All ascent propulsion systems utilize liquid propellants, and overall launch vehicle stack height is minimized while maintaining a reasonable vehicle diameter. The ascent propulsion systems are based on the development of a new liquid oxygen/hydrocarbon booster engine and liquid oxygen/liquid hydrogen upper stage engine derived from today's SSME technology. Wherever possible, propulsion and avionics systems are contained in reusable propulsion/avionics modules that are recovered after each launch.

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DEFINITION OF SYMBOLS

α	Alpha, angle of attack in pitch
A_{Ref}	Reference area, $\pi D^2_{Ref}/4$
β	Beta, angle of attack in yaw
B_1	171 in. diameter booster
B_2	246 in. diameter booster
C_{A_F}	Forebody axial force coefficient
C_{A_T}	Total axial force coefficient
C_D	Drag coefficient
CP	Center of pressure
C_N	Normal force coefficient, $N/q_\infty A_{Ref}$
C_{N_α}	Normal force coefficient slope, ($\alpha \leq \pm 6$ deg)
$C_{N_{\alpha B}}$	Booster normal force coefficient slope
$C_{N_{\alpha F}}$	Fin normal force coefficient slope
C_Y	Side force coefficient
C_{Y_β}	Side force coefficient slope, ($\alpha \leq \pm 6$ deg)
$\frac{dC_{N_\alpha}}{d(X/D)}$	Local normal force coeffient slope, ($\alpha \leq \pm 6$ deg)
ε	Nozzle area ratio
ΔV	Velocity increment
D_{REF}	Reference diameter
λ'	Mass fraction
L_{RFE}	Reference length
M	Mach number
F	Thrust
F/W	Thrust-to-weight ratio

g	Acceleration of gravity
Max q	Maximum dynamic pressure
q	Dynamic pressure
q_{∞}	Free stream dynamic pressure

LIST OF ACRONYMS AND ABBREVIATIONS

APS	Auxiliary Propulsion System
BSM	Booster Separation Motor
CAD	Computer Aided Design
CAE	Computer Aided Engineering
CAM	Computer Aided Machining
CG	Center of Gravity
CP	Center of Pressure
DRM	Design Reference Mission
EAFB	Edwards Air Force Base
ET	External Tank
GLOW	Gross Lift Off Weight
GSE	Government Supplied Equipment
GSE	Ground Support Equipment
HC	Hydrocarbon
HLLV	Heavy Lift Launch Vehicle
IOC	Initial Operational Capability
I_{sp}	Specific Impulse
KSC	Kennedy Space Center
LCC	Launch Control Center
L/D	Lift-to-Drag Ratio
LH_2	Liquid Hydrogen
LO	Lift Off
LOX	Liquid Oxygen
LRB	Liquid Rocket Booster
MECO	Main Engine Cut Off
MLP	Mobile Launcher Platform

LIST OF ACRONYMS AND ABBREVIATIONS (Concluded)

MMH	Monomethyl Hydrazine
MSFC	Marshall Space Flight Center
N_2O_4	Nitrogen Tetroxide
n.mi.	Nautical Mile
OMS	Orbit Maneuvering System
OTV	Orbit Transfer Vehicle
P/A	Propulsion/Avionics
PLF	Payload Fairing
PSF	Pounds (force) per Square Foot
RCS	Reaction Control System
RF	Radio Frequency
SD/HLV	Shuttle Derived Heavy Lift Vehicle
SDV	Shuttle Derived Vehicle
SRB	Solid Rocket Booster
SSME	Space Shuttle Main Engine
STAR	Shuttle Turnaround Analysis Report
STS	Space Transportation System
STBE	Space Transportation Booster Engine
STME	Space Transportation Main Engine
TPS	Thermal Protection System
VAFB	Vandenberg Air Force Base
WTR	Western Test Range

TECHNICAL MEMORANDUM

HEAVY LIFT LAUNCH VEHICLES FOR 1995 AND BEYOND

I. INTRODUCTION

A Heavy Lift Launch Vehicle (HLLV) designed to deliver 300,000 lb payloads to a 540 n.mi. circular polar orbit may be required to meet national needs for 1995 and beyond. The vehicle described herein can accommodate payload envelopes up to 50 ft diameter by 200 ft in length. Payloads utilizing this capability may be Space Station elements, commercial space facilities, or advanced military systems.

Design requirements include reusability of the more expensive components such as avionics and propulsion systems, rapid launch turnaround time, minimum hardware inventory, stage and component flexibility and commonality, and low operational costs. All ascent propulsion systems utilize liquid propellants and overall launch vehicle stack height is minimized while maintaining a reasonable vehicle diameter.

The ascent propulsion systems are based on the development of a new liquid oxygen/hydrocarbon booster engine and a liquid oxygen/liquid hydrogen upper stage engine derived from today's SSME technology. The upper stage engine will have more thrust than the SSME, be more reliable with less maintenance, and have a two-position nozzle. The requirements placed on the avionics system are more stringent than on present launch vehicles because of the rapid turnaround of reusable components and the necessity to maintain continuous launch readiness after stackup.

Wherever possible, propulsion and avionics systems are contained in reusable Propulsion/Avionics (P/A) Modules that are recovered after each launch. The P/A Module has an ablative non-reusable Thermal Protection System (TPS) and a crushable honeycomb nose cone to absorb landing loads. P/A Module recovery is baselined as a terrene landing to avoid the complexities associated with water landing and recovery. The storable propellant Reaction Control System (RCS) and Orbit Maneuvering System (OMS) are also located in the P/A Module.

The HLLV structural design is based on current Space Transportation System (STS) technology to meet the Initial Operational Capability (IOC) date of 1995. Technology advancements in structural design and materials may increase the payload delivery capability, but at the cost of a longer development schedule.

Two development approaches are considered. The first approach is the direct development of a mature HLLV without intermediate steps. The second approach has an extended schedule where the booster systems are initially developed for application to the STS and Shuttle Derived Vehicle (SDV) programs, or to a new intermediate vehicle prior to HLLV application. This approach will reduce front end development costs but extend program development time and delay IOC. The first approach will compress overall development time leading to an earlier IOC but will increase initial costs accordingly.

II. CONFIGURATION SUMMARY

Three configuration concepts were investigated during this study, all satisfying the previously described requirements. Figure 1 shows the mold line and base view of each. The aerodynamic fairing for the 50 ft by 200 ft payload is jettisoned at 350,000-ft altitude for all concepts. The booster stages of all three configurations use new LOX/JP4 gas generator cycle booster engines, designated Space Transportation Booster Engines (STBEs), having a sea level thrust of 1.5 to 2.0 million lbf depending on specific booster application. All upper stages use staged combustion cycle LOX/LH_2 engines derived from the SSME and designated Space Transportation Main Engines (STMEs). The STME has a two-position nozzle for altitude compensation with thrust varying from 397,000 lbf at sea level to 481,000 lbf in vacuum. The operating characteristics and performance parameters for the STBE and STME are shown in Figures 2 and 3, respectively. Very brief descriptions of the three configuration concepts follow; however, Configuration II shows the greatest potential and is discussed in more detail in the remainder of this report.

A. Configuration I

Configuration I is a series/parallel burn three-stage vehicle designed for commonality of propellant tanks. All tanks have the same diameter to reduce the costs of development, design, tooiing, production, and qualification. The LOX/JP4 first stage, or booster, consists of four tank sets of sub-stages, each with two 1.75 million lbf sea level thrust STBEs. The second and third stages use LOX/LH_2 propellants and have four and two STMEs, respectively. The second stage consists of two tank sets. The third stage consists of a single tank set centered within the first and second stages. This configuration minimized the vehicle stackup height. The aft ends of the booster sub-stages are connected by box beams to distribute atmospheric flight bending moments and minimize structural weight of the upper stages. Longitudinal thrust loads during booster flight are distributed to the vehicle at a forward payload attach ring. The upper stages are carried in tension during booster burn with the STME nozzles in the retracted or stowed position for more efficient packaging and better thermal control.

After first stage separation, the second and third stages are ignited simultaneously and burn in parallel. The second stage, which consists of two tank sets or sub-stages, crossfeeds propellants to the core third stage. At second stage propellant depletion, the two sub-stages are separated and expended. The high staging velocity (Mach 16) results in a down range distance too great for practical recovery, even if the hardware could survive reentry.

After second stage separation, the third stage continues to burn into the perigee of a 100 x 540 n.mi. elliptical orbit. Following a coast to apogee, the third stage reignites, placing the payload into the operational 540 n.mi. circular orbit. After payload deployment, correct separation distance and orbit phasing, the third stage reignites to deboost the stage for disposal.

First stage hardware is recovered after water landing; however, designs for second and third stage recovery systems were not pursued for this configuration. Additional design details for Configuration I are available in Reference 1.

B. Configuration II

Configuration II is a parallel burn two-stage vehicle designed without the hardware commonality constraints of the previous configuration. The flight profile uses direct insertion into a 100 x 540 n.mi. orbit by the first and second stages with circularization at apogee achieved by either a kick stage or payload-supplied propulsion system.

The LOX/JP4 first stage, or booster, consists of four tank-sets or sub-stages, two are 246 in. in diameter and two are 171 in. in diameter. The larger diameter sub-stages have two 1.616 million lbf thrust STBEs each and the smaller tank sets have one STBE each, for a total of six booster engines. Each booster tank set contains three propellants: liquid hydrogen, liquid oxygen, and a hydrocarbon fuel (JP4). The second stage is 396 in. in diameter and has five two-position nozzle STMEs. All first and second stage engines are ground ignited and flown in parallel burn until booster staging. During booster burn liquid oxygen and liquid hydrogen are crossfed from the first stage tanks to the STMEs. This procedure shortens the second stage hydrogen tank by 30 ft and reduces weight.

It is feasible to recover the booster sub-stages and the second stage propulsion and avionics hardware (housed in a P/A Module).

C. Configuration III

Configuration III is a two-stage inline series burn vehicle that has been defined only to the depth necessary for comparison with the other options. The LOX/JP4 first stage is 50 ft in diameter and has eight STBEs. The LOX/LH_2 second stage, also 50 ft in diameter, has five STMEs. All LOX is carried in the second stage (i.e., the first and second stages share a common LOX tank). The booster stage is recovered and the second stage propulsion and avionics hardware is assumed to be recoverable by the use of an appropriately designed P/A Module. This option is viewed as having very little growth potential, essentially no capability for stage elements to be used as intermediate class vehicles, and would pose difficulties in ground transportation and handling.

III. SELECTED CONFIGURATION RESULTS

A. Two-Stage Parallel Burn Configuration (Configuration II)

The configuration resulting from detailed analyses is shown in Figure 4. The LOX/JP4 recoverable first stage consists of four tank sets or sub-stages. Two of the sub-stages are 246 in. in diameter and two are 171 in. in diameter. The large diameter sub-stages have two STBEs and the small diameter sub-stages have a single STBE. Each sub-stage of the booster contains three propellants: liquid oxygen, liquid hydrogen, and JP4. The LOX/LH_2 second stage is 396 in. in diameter, has five STMEs contained in a recoverable P/A Module, and a forward structural adapter/payload attach ring. A vehicle weight summary is displayed on Table 1. The resulting gross liftoff weight of this configuration is 8.6 million lb. The sub-stage and second stage weight statements are detailed on Table 2. Structural details are contained in Appendix H.

B. Ascent Flight Profile

The ascent flight profile is shown in Figure 5. All engines are ground ignited on the launch pad and burn in parallel until booster cutoff and staging. The second stage STME nozzles are retracted (ε = 55:1) until booster staging, then extended (ε = 150:1) and the second stage continues into the perigee of the 100 x 540 n.mi. orbit. All propellants for the parallel burn portion of flight are carried in the booster tanks with LOX and LH_2 being crossfed into the second stage. This procedure allows the second stage tanks to be full at booster burnout, reducing the tank volume and weight required to be carried to orbit.

C. Design Reference Mission and Deorbit Events

The design reference mission profile is shown in Figure 6. The flight events are depicted and times of occurrence covered from liftoff to P/A Module landing. The second stage propellant tanks are expendable and require controlled deboost from orbit. Following payload separation, the second stage coasts in orbit for approximately one revolution. It is then oriented for the deboost burn attitude by the P/A Module and a slow roll started for attitude stabilization during the retro burn. The P/A Module separates and performs an evasive maneuver away from the tankage. An onboard timer provides the deboost ignition signal to six solid rocket motors mounted in the forward conical adapter (see Appendix F). At the appropriate time, the P/A Module performs an orbital adjustment burn to allow correct phasing to align the orbital plane with the landing site (Edwards Air Force Base for this study).

The P/A Module performs a deboost burn using the aft firing OMS storable propellant engines. The reentry environment of the P/A Module is described in Appendix G and the resulting thermal protection system requirements in Appendix J. When the P/A Module has slowed to approximately Mach 1 after reentry, drogue parachutes are deployed for added drag and stabilization. Parawing type steerable devices were selected for the terminal landing event (Fig. 7). Terrene landing on a crushable nose cone has been baselined to reduce the refurbishment requirements and turnaround time. The honeycomb nose cone structure and the TPS are considered sacrificial and are replaced after every launch.

D. Booster Recovery

At booster burnout, the sub-stages are separated and fall into the ocean for partial retrieval. Figure 8 shows a recovery flight profile of a booster stage from drogue release to water impact. The retrieval technique is called the hydropneumatic option. Following separation the boosters coast through apogee and reenter the atmosphere. The induced environment during reentry is detailed in Appendix G. More detailed studies are required to define when the aerodynamic fins are to be separated from the sub-stages. After the boosters have slowed to approximately Mach 1, drogue parachutes are deployed for stablization. After the main chutes are deployed, a linear shaped charge severs the forward tank dome from the barrel section just aft of the "Y-ring." Shaped charges are used to provide "vent" holes in the tank barrel section just forward of the aft dome "Y-ring" to allow the trapped air to escape at water impact, providing the pneumatic cushion that yields a soft landing. Vent hole size will be traded between deceleration and rebound. The JP4 tank will provide the flotation and a flotation ring will be deployed to provide stability until recovery. A protective spray bag will be deployed before impact to enclose the aft end of the engine compartment and thus keep the booster engines dry. The booster propulsion and avionic subsystems are recovered for reuse and the JP4 tank has a high probability of reusability. The boosters will float in the inverted position until removed from the water. Cleaning for refurbishment can begin on the way back to port.

E. Fin Size Selection

Fin size effects on ascent aerodynamics were traded early in the study. The detailed aerodynamics generated for this launch vehicle are contained in Appendix A. The static stability of a launch vehicle is determined by the distance between the center of gravity and the center of pressure. Figure 9 displays the estimated pitch plane center of pressure (CP) variation versus the Mach number for no fins, four 300 ft^2 fins, and four 675 ft^2 fins. The estimated longitudinal center of gravity is also shown. The center of pressure is always forward of the center of gravity, resulting in an unstable vehicle. Based on these data, four 675 ft^2 fins were baselined until more detailed analyses could be performed. A cursory control study was performed after the design was frozen.

F. First Stage Control

Flight simulations with six-degree-of-freedom rigid-body dynamics and three-axis control were performed for the booster phase of flight. The preliminary results show that, with only the booster engines controlling, sufficient control authority is available with a square 6 deg gimbal pattern to withstand the effects of one booster engine out at 45 sec and the MSFC 95 percentile synthetic wind profile with the embedded gust [2]. Figure 10 displays the crosswind profile versus altitude.

Simulations were made for the 675 ft^2 fins, 300 ft^2 fins, and for no fins with the engine failure at 45 sec of flight and the wind speed of 270 ft/sec peaking at 43,000 ft altitude (near maximum dynamic pressure). One engine of the right side two-engine booster was assumed to fail. This causes an immediate yaw moment of 50 million ft-lb. The available control torque is 90 million ft-lb. A left crosswind is assumed, thus creating positive yaw moment, additive with the engine failure moment.

Figures 11 and 12 show that with the four 675 ft^2 fins, the required engine gimbal command does not reach the 6 deg limit due to the engine failure alone, and just momentarily touches it when the crosswind peaks. The pitch angle of attack (ALPHA) is smooth, but the yaw angle of attack (BETA) has a maximum value of 6 deg during the period of maximum dynamic pressure.

Results from the analysis using the 300 ft^2 fins are displayed in Figures 13 and 14. The yaw gimbal command hits the limit for both the engine out and crosswind disturbances. The resulting BETA angle is larger and lasts longer than for the larger fins.

The no-fin case can only be controlled through the wind disturbance; if a booster engine fails, vehicle control is lost. Since performance effects of these control studies remain to be determined, the 675 ft^2 fins were retained as the baseline.

G. Engine Out Operation

Propellant plumbing has been added to the launch vehicle to provide propellant crossfeed in case of a non-planned engine out during ascent. The operational philosophy is to set the flight power level at a given percentage less than the nominal 100 percent power level. The STBE flight power level will be 83 percent, and the STME 80 percent. If a STBE or STME fails, the functioning engines will be advanced to 100 percent for the remaining burn time. Control gains and guidance presettings will be automatically switched by the onboard computers. Additional studies, historically called "Abort and Alternate Mission," will be required for premission planning because, once launched, the vehicle is committed to space or destroyed.

H. Avionics

The HLLV avionics is composed of several major subsystems: communications and tracking, data processing, guidance and navigation, flight control, propulsion control, auxiliary flight control, electrical power, and range safety control. Design of the avionic subsystems will utilize current and evolving technology to meet the objective of improved performance with minimum risk, reduced turnaround time, and reduced cost. Advanced technology in data processing will enable much higher levels of automation to support design, analysis and mission planning, reconfiguration, checkout, and launch. Advanced distributed fault tolerant processing architectures and methodologies will be utilized to provide very reliable flight systems that can be partitioned for vehicle modularity, contractual separability, and interface simplification. Advanced electronic technology will also make possible a high level of vehicle autonomy. Many functions that have previously been performed by ground support equipment or the launch control center will be performed on-board the vehicle to minimize checkout and launch support personnel, vehicle turnaround time, and vehicle/GSE interface complexity.

Much of the vehicle avionics is distributed and physically dispersed to achieve modularity and partitioning objectives. Other vehicle avionic functions such as RF communications tend to be centralized in nature and relatively independent of the vehicle configuration. Subsystems with these centralized functions are integrated into a "central avionics package" located in the P/A Module. This central avionics package can be treated as a major vehicle module, completely integrated within itself and having "clean" data bus and power bus interfaces.

The elements of both the distributed and the centralized avionic systems on the vehicle are connected with a fault tolerant network. It is envisioned that the network will consists of a number of physically dispersed processing sites connected by a fault tolerant communication network. The general purpose processors at each site can have varying levels of throughput, memory size, fault tolerance, and local input/output.

A more complete description of the requirements, objectives, and characteristics for the avionic systems is given in Appendix B.

I. Vehicle Description

Figure 15 is a computer-generated drawing emphasizing the propellant feedlines of the launch vehicle. The second stage propellant tanks, feedlines, and component weights are detailed in Figure 16. The LOX feedlines are routed externally to avoid penetration of the LH_2 tank. Two feedlines are used instead of one large line to make the stage more symmetric in roll for the deboost stabilization. The aft plumbing in the core stage is the LOX crossfeed connection for the P/A Module and the JP4 crossfeed to ensure booster propellant depletion in case of a booster engine out. The booster engine out LOX crossfeed is available as a result of crossfeeding propellants to Stage 2, therefore, only additional JP4 lines are required. Various layout combinations were investigated for the JP4 crossfeed and the one presented here is the simplest. The LH_2 feedline is a short sump feeding in to the P/A Module. The LH_2 crossfeed inlets are located at the top of the tank at a level which will allow adequate ullage pressure volume at booster staging.

The one- and two-engine boosters are detailed in Figures 17 and 18, respectively, including a weight summary for each booster size. The moldline dimensions are displayed and primary stage and crossfeed propellant lines detailed. The 675 ft^2 fins are displayed. The upper feedlines are for LH_2 crossfeed into the core stage. The LOX feedlines run down the side of each stage and split, one part routed to the second stage and the other to the LOX/JP4 engines of the respective stages. The JP4 plumbing splits to the engine and to the crossfeed manifold in the second stage. The recovery parachute system is housed in the engine skirt region, and the post landing flotation collar is housed within the aft JP4 tank area.

The P/A Module is the most expensive component of the launch vehicle. Structural and plumbing drawings and a weight summary are displayed in Figures 19 and 20. The four STMEs are clocked 45 deg to the flight plane to reduce mechanical and heating interference with the boosters. The propellant distribution feedlines are on different planes to eliminate plumbing interferences. The individual lines for each engine straddle the thrust structure crossbeam. The gimbal points of the outboard engines (the inboard engine is fixed) will be 45 deg out of plane, i.e., rock and tilt as on the Space Shuttle Solid Rocket Boosters. These commands will be transformed by software within the control system. The Orbital Maneuvering System (OMS) and on-orbit Reaction Control System (RCS) will be housed in four replaceable modules, plumbed to the storable tanks. Three axis RCS and longitudinal translation is provided by this system. Thruster details are in Appendix F. The P/A Module houses most of the avionics for prelaunch checkout, flight, and reentry. Details and specific requirements assigned to each launch vehicle component are presented in Appendix B. Structural design details are in Appendix H.

The Payload Fairing (PLF) is used to protect the payload through aerodynamic flight. A moldline drawing is presented in Figure 21, and structural weights in Table 3. The PLF has a 50 ft outside diameter and houses a 200 ft long payload, not including the length available within upper nose cone. A double angle nose cone was selected based on studies performed for the Saturn launch vehicle [3]. This geometry provides an efficient trade between aerodynamic drag and internal volume. The PLF is designed in 24 ft long cylindrical sections for adaptability to various payload lengths.

The PLF will be separated into four 90 deg longitudinal sections which will be jettisoned in flight and expended. The longitude separation system is a non-contaminating expanding tube device as flown on the Skylab Mission (Fig. 22). Separation from the PLF adapter ring can be by explosive bolts or some other suitable device.

IV. LAUNCH FACILITIES AND GROUND OPERATIONS

The baseline launch scenario for the HLLV is a due south polar orbit launch. Considerations in selecting the launch site include the large size of the vehicle, rapid buildup and payload changeout requirements, non-interference with STS flights, launch azimuth and overflight restrictions, and other practical factors. A new launch site is recommended and possible locations include Hawaii, southern Alaska, the Vandenberg Air Force Base area, and certain other regions of the continental United States. The facility size will be a function of launch rate, however, there should be at least two launch pads for parallel launch capability, served by a single Launch Control Center. The vehicle is to be built up in an assembly building and transported to the launch pad by a mobile launcher. This approach provides more efficient use of the launch pads, allows parallel vehicle processing, isolates the launch pads from the buildup area, and facilitates launch vehicle changeout. The overall operational sequences given in Appendix D are similar to the STS processing flow at KSC, and some timelines (such as rollout, pad refurbishment, and mobile launch refurbishment) are derived from the STS processing assessment, STAR-027.

V. TEST PROGRAM

The major elements of the test programs for the HLLV are depicted in Figure 23 and detailed in Appendix I. The protoflight approach given in MIL-STD-1540B is to be used and is best exemplified by the large structural test items, which will be tested to levels exceeding flight loads but lower than yield values. Following rework, the test items can be used as flight hardware, thus avoiding the significant cost of dedicated hardware for testing only.

The test program is unique in the number and size of the tests to be conducted. An evolutionary or derivative approach would require more testing than the direct development approach. Every stage will have to be tested to the loads expected for each flight application.

It is important that all vehicle elements be designed to facilitate testing. Design personnel should be involved not only in the test planning process but should be engaged in all subsequent phases of the test program.

VI. DEVELOPMENT SCHEDULES

The scope of the HLLV project is similar in size to the development of the Saturn V and STS launch vehicle systems. New manufacturing and launch facilities will be required to avoid interference with the operational STS program.

The development of this launch vehicle system may be approached either directly or in an evolutionary fashion. The direct approach, which can include derivatives, leads to development of a mature flight system without intermediate steps. The evolutionary or derivative approach takes longer than the direct approach; however, the former may be more cost effective than the direct approach. A family of vehicles with diverse capabilities, all built with the same tooling, avionics, and propulsion, will be produced. Additionally, other vehicles can be derived including some that are sized to fit the STS cargo bay diameter.

The summary schedule is shown in Figure 24 and the supporting details contained in Appendix C. The development of derivative launch vehicles may be concurrent with HLLV development or be delayed until the HLLV is operational. The direct approach has the earliest IOC but requires extensive funding early in the program. The evolutionary development is characterized by a building block approach where the STBE and booster sub-stages are developed initially for application with other vehicles, to be later integrated with the second stage, P/A Module, and payload fairing to comprise the HLLV. This approach reduces initial funding requirements at the penalty of a later IOC for the HLLV. The STBE is the pacing item for either development approach and this engine's design, performance, and operating characteristics must be established early.

VII. EVOLUTIONARY CONCEPTS

The modular design allows the development of alternate launch vehicles from the basic HLLV stage elements or the use of these elements in other space applications.

It is not intended that elements or stages be directly exchanged, rather that the design and manufacturing data base, tooling, and assembly processes be effectively used for other applications (Fig. 25). The reference HLLV configuration has excellent growth capability by interchanging booster stages and increasing the power level setting of the stage two engines. Replacing the single engine boosters with two engine boosters for a total of 4 two engine boosters, results in an increase of thrust and available propellant. The crossfeed propellant capacity is increased, and to consume this requires increasing the stage two thrust during booster burn. This larger booster configuration will launch approximately 600 Klb of payload into a reference 100 x 100 n.mi., 28.5 deg inclination orbit.

A small payload two stage vehicle was derived from the two engine booster hardware. The first stage has two STBE's and the upper stage a single STME. The performance of this configuration is approxiamtely 123K lb to the 100 n.mi., 28.5 deg orbit and is detailed in Appendix E. Shuttle configurations in which the SRBs have been replaced with modified HLLV liquid rocket boosters (LRB) were studied. Initially, 2 two-engine LRBs with a full propellant load were investigated for the Shuttle. This results in a payload capability of approximately 167K lb which exceeds the current Shuttle load carrying capability; however, this capability could possibly be used for added mission flexibility (e.g., higher orbits, plane changes, etc.). A second option was investigated in which three single engine HLLV-derived LRBs were used with the Shuttle. This results in a payload capability of 95K lb to the above referenced orbit.

An intermediate class heavy lift vehicle, the Shuttle Derived/Heavy Lift Vehicle (SD/HLV), was studied which consists of a modified Shuttle ET with a reusable propulsion/avionics module as the second stage, boosted by a pair of the two engine HLLV boosters. This configuration has a payload capability in excess of 300K lb to the reference orbit.

In addition to the above, the single engine booster diameter (176 in.) was selected for potential application within the Shuttle cargo bay. The cryogenic tanks have potential application as orbital tankers, or with the addition of a propulsion system, an OTV. These particular applications could utilize tankage with reduced skin gauge.

The 33 ft diameter second stage HLLV tanks could be configured into a large propellant storage facility, launched by the HLLV. Propellant loading requirements of this nature have been defined by the interplanetary mission studies.

The next section details some of the thought processes required during the definition phase of this or the next generation of vehicles.

VIII. DESIGN AND MANUFACTURING

The design and manufacturing philosophy for the HLLV emphasizes versatility and flexibility of operation rather than unique application. The more expensive structures are the tank domes, thrust structure, interfaces, and attach points. The length of the cylindrical barrel sections can be changed easily if the original design process has foreseen this requirement and the required strength may be attained by reprogramming the numerical milling machines that are used for panel cutting. This process can also be applied to the tank domes but trade studies should verify a reasonable payback for the change in metal thickness versus reduction of payload or increased propellant load required to deliver a given payload.

This adaptability can be achieved by addressing the multiple use aspects of components early in the design process. Computer aided design and computer aided engineering (CAD/CAE) may be effectively applied. Before designing the manufacturing facility, CAD layouts should be performed to eliminate interference of different operations. Skin thicknesses and tolerances should be traded against manufacturing costs and other processes.

Manufacturing facilities should be designed to fabricate multiple stage sizes varying in diameter and length. Machining will be performed by high speed computer aided machines (CAM) and changes in skin thicknesses can be controlled from the design computer.

Procedural changes to vertical buildup and assembly of the tank structures is required to accommodate different cylindrical lengths without major floor modifications or extensive foundation restructuring. This must be addressed during plant design and before construction. Vertical buildup requires fewer internal sectional hoop jigs to maintain roundness during welding. Trades will determine whether the welding turret will be movable or the assembly table move vertically within either a high bay or a pit. Correct design will allow welding of different diameters by reprogramming the drive computer.

Plasma arc welding technology should be investigated as a method of reducing production time, weld inspection and ultimately reducing costs. Extrusion forming of short interstage panel sections may reduce costs for high production rates. This procedure would have structural weight penalties due to the constant skin thickness but may be more cost effective than tapered panels. Upper stages may benefit from the use of composite materials, however, high production rate techniques will be required to be cost effective.

Trade studies must determine the cost differences between unique and general designs and if payload penalties are justified by easier manufacturing operations. Reducing machining tolerances or allowing thicker panel sections could allow contracting to lower overhead rated machine shops.

The guideline of all these trade studies is to define a system that is affordable. The mass production state-of-the-art techniques must be implemented.

IX. CONCLUSIONS

The pacing development item for the HLLV is the new LOX/JP4 engine. To meet the projected capabilities of post-1995, the development of the STBE should be started very soon. New recovery methods and hardware for the boosters and P/A Module are required. A terrene landing instead of water touchdown could allow complete reuse of the boosters while turnaround times and refurbishment costs are reduced. A sub-scale reentry flight test of the P/A Module will be required to verify stability, attitude control, and TPS during reentry, and the steerable parachute system for the final landing must be demonstrated.

Updated design, engineering, and manufacturing processes must be applied to this next generation of launch vehicles. The building block use of stage components for evolution or derivation of other vehicles must be recognized at the beginning of design and carried through the systems' lifetime. Tooling and factory layout must reflect operational flexibility with a goal of no down time for conversion from one size component to another. High speed machinery and minimum inspection welding processes will make large contributions to cost reduction.

High technology materials may be implemented as product improvements within the life of this system. Goals must be defined before implementation considering possible payload capability increases versus cost increased incurred by implementing a technology.

New methods of efficient configuration management are required. One concept is to make vendors responsible for designated end items, rather than assuming general liability. This will reassign the warranty to the vendor and create incentives to reduce nonproductive costs.

The data presented above is to be considered as a point of departure for following launch vehicle studies in the areas of design, materials, propulsion, control, all avionics functions, manufacturing, and operations. Product improvement studies in all areas are welcomed, especially those leading to a reduction in program costs.

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2. Garner, Doyle: Control Theory Handbook. NASA/George C. Marshall Space Flight Center, TM X-53036, April 22, 1964.

3. Geissler, E. D.: Memorandum on Nose Shape on Saturn Vehicles. George C. Marshall Sapce Flight Center, April 9, 1963.

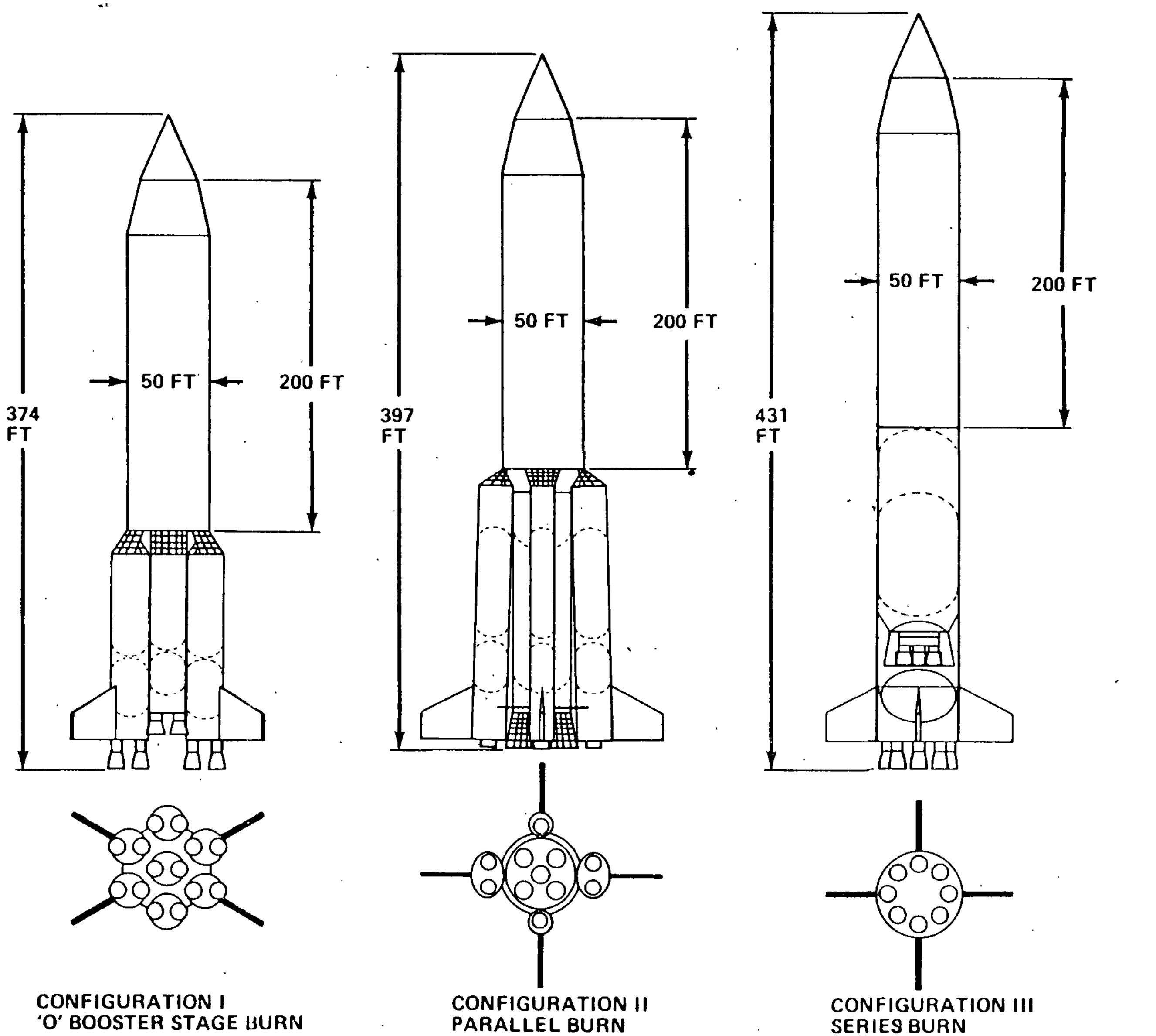


Figure 1. HLLV options comparison.

• PROPELLANTS	LOX/JP4
• NOZZLE AREA RATIO	25
• THRUST (SEA LEVEL) – KLBF	1500 TO 2000
• DELIVERED SEA LEVEL ISP SEC	289
• CHAMBER PRESSURE PSIA	2000
• MIXTURE RATIO (O/F)	2.8
• LENGTH IN	199 TO 226
• NOZZLE EXIT DIAMETER IN	116 TO 131
• ENGINE INSTALLED WT LBM	16340 TO 24160

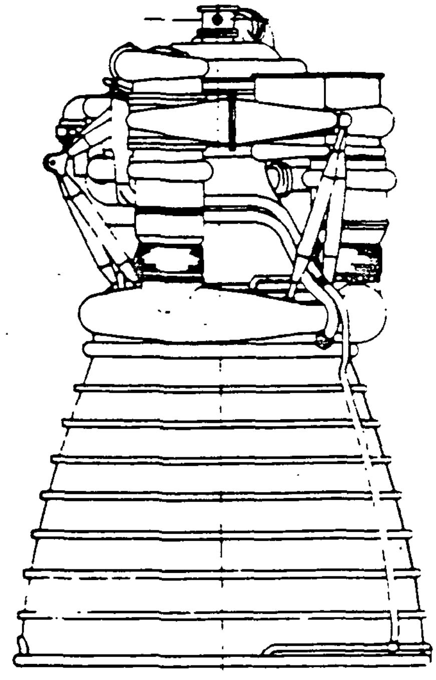


Figure 2. Space Transportation Booster Engine (STBE).

• PROPELLANTS	LOX/LH_2
• NOZZLE AREA RATIO (STOWED/EXTENDED)	55/150
• VACUUM THRUST KLBF	468/481
• VACUUM ISP SEC	449/461
• CHAMBER PRESSURE PSIA	3006
• MIXTURE RATIO (O/F)	6.0
• LENGTH IN	139/219
• NOZZLE EXIT DIAMETER IN	76.2/126.3
• ENGINE INSTALLED WT LBM	7142
• SEA LEVEL THRUST-KLBF (STOWED)	397
• SEA LEVEL ISP SEC	380.4
• FLOWRATE LB/SEC	1043.4

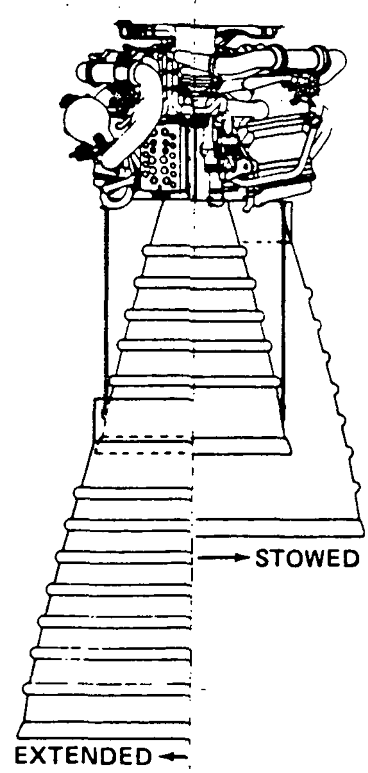


Figure 3. Space Transportation Main Engine (STME 481).

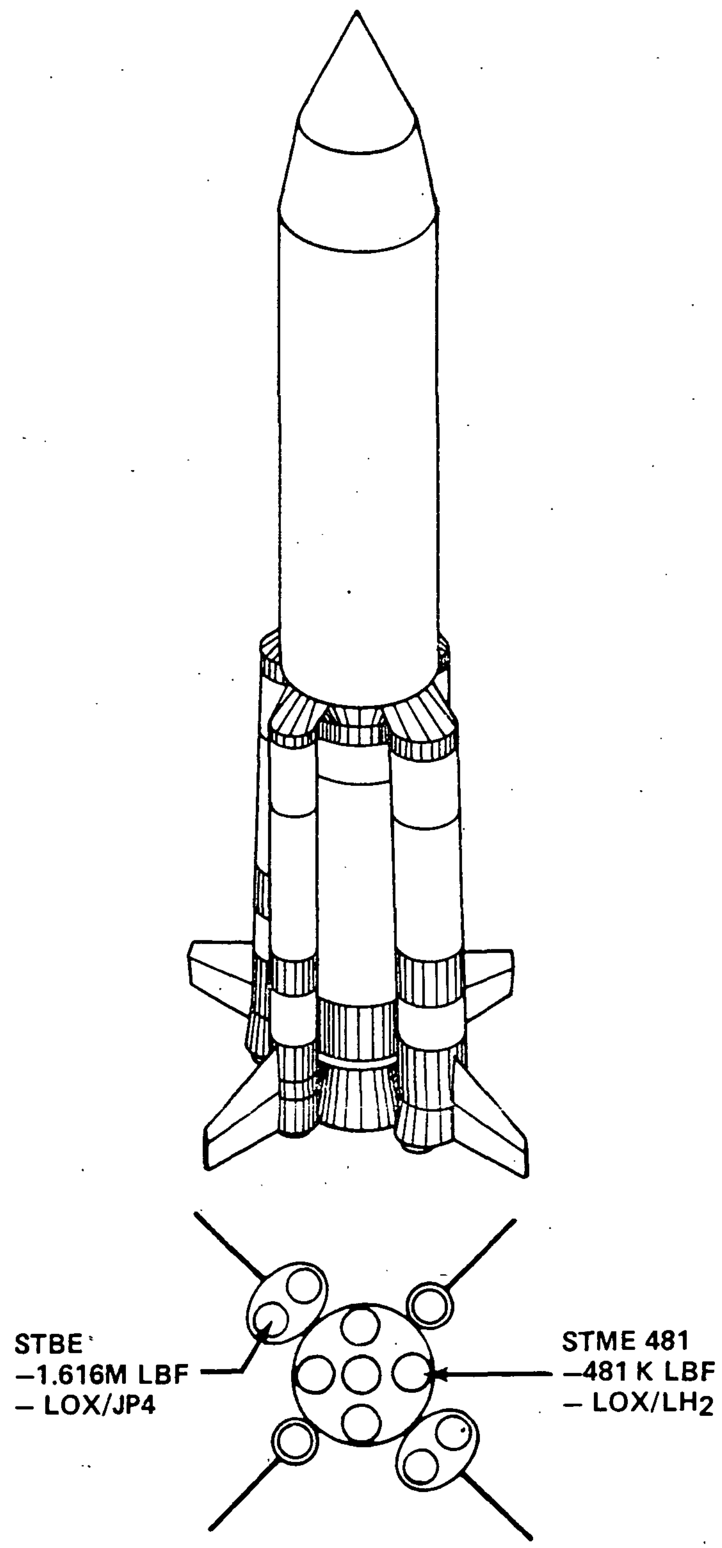


Figure 4. Heavy Lift Launch Vehicle (HLLV).

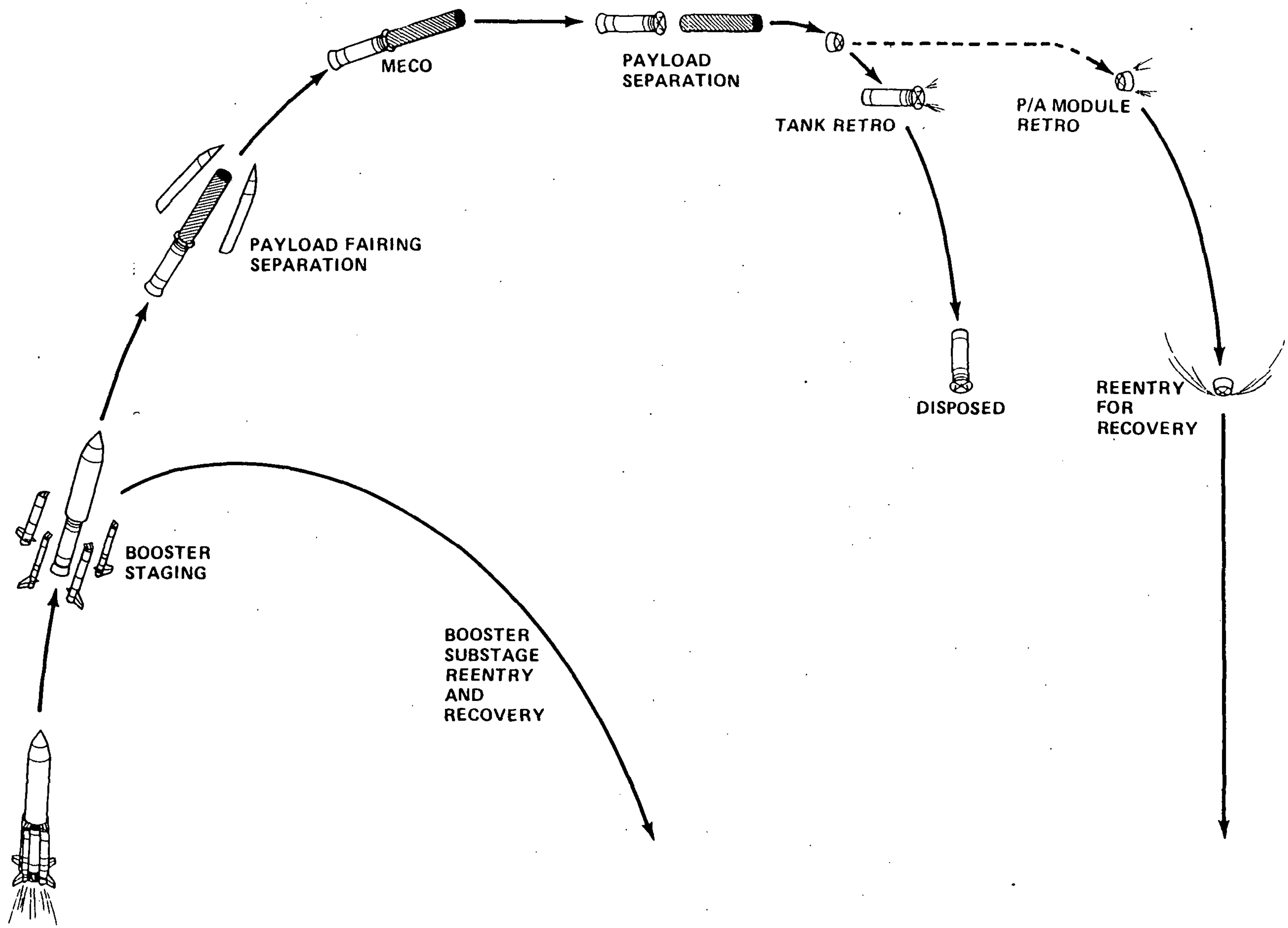
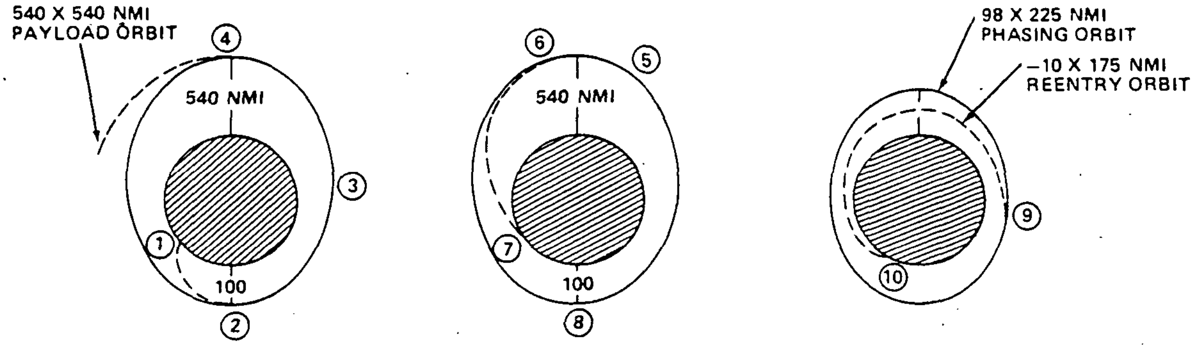


Figure 5. Ascent flight profile.



EVENT	TIME ~ HRS
1 – LIFTOFF	0.000
2 – INJECT @ 100 X 540 NMI	0.146
3 – PAYLOAD SEPARATION	0.600*
4 – KICK STAGE CIRCULARIZES PAYLOAD @ 1st APOGEE	0.951
5 – SEPARATE P/A MODULE FROM STAGE 2	2.4*
6 – STAGE 2 DEORBIT NEAR 2ND APOGEE	2.7*
7 – STAGE 2 SPLASHDOWN	3.2*
8 – P/A MODULE PHASING BURN @ 2nd PERIGEE AFTER INSERTION	3.364
9 – P/A MODULE DEORBIT BURN	11.090
10 – P/A MODULE LANDING	11.93*

*APPROXIMATE

Figure 6. Design reference mission profile.

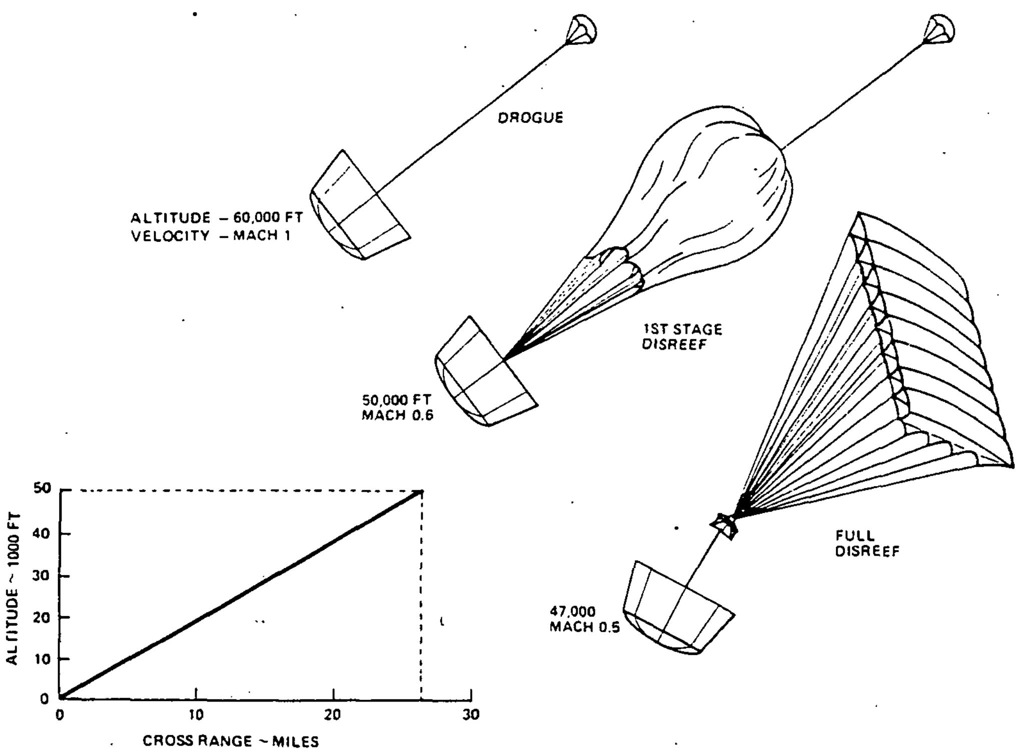


Figure 7. P/A Module recovery.

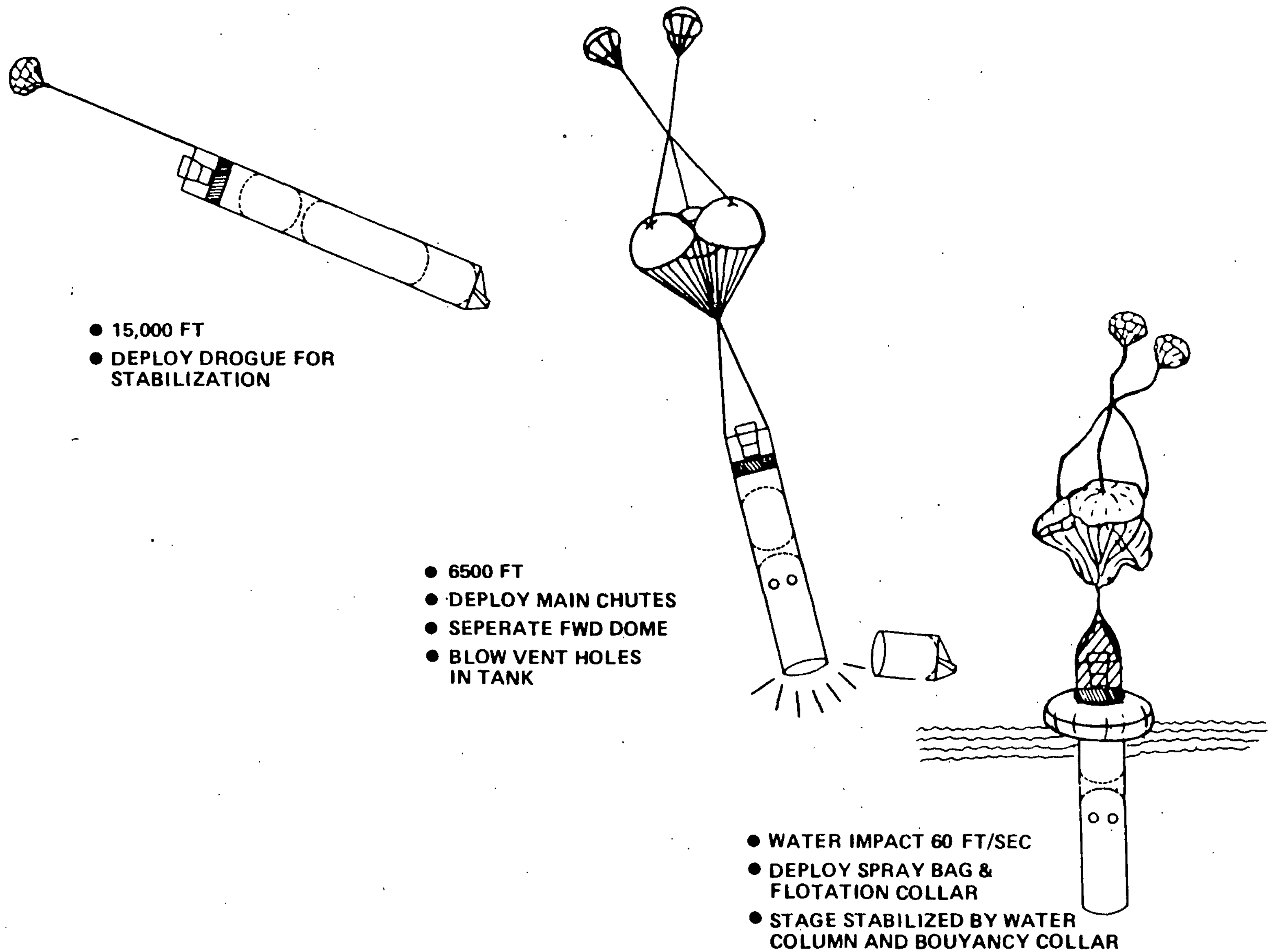


Figure 8. Booster recovery scenario.

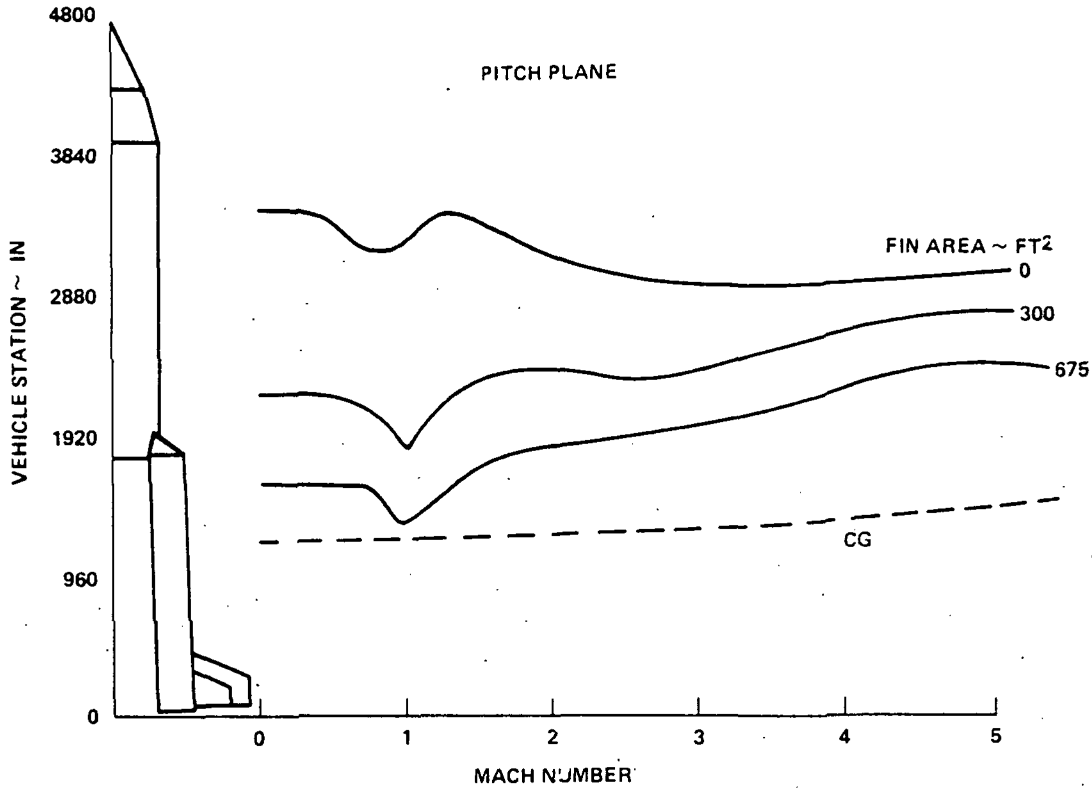


Figure 9. Center of pressure variation.

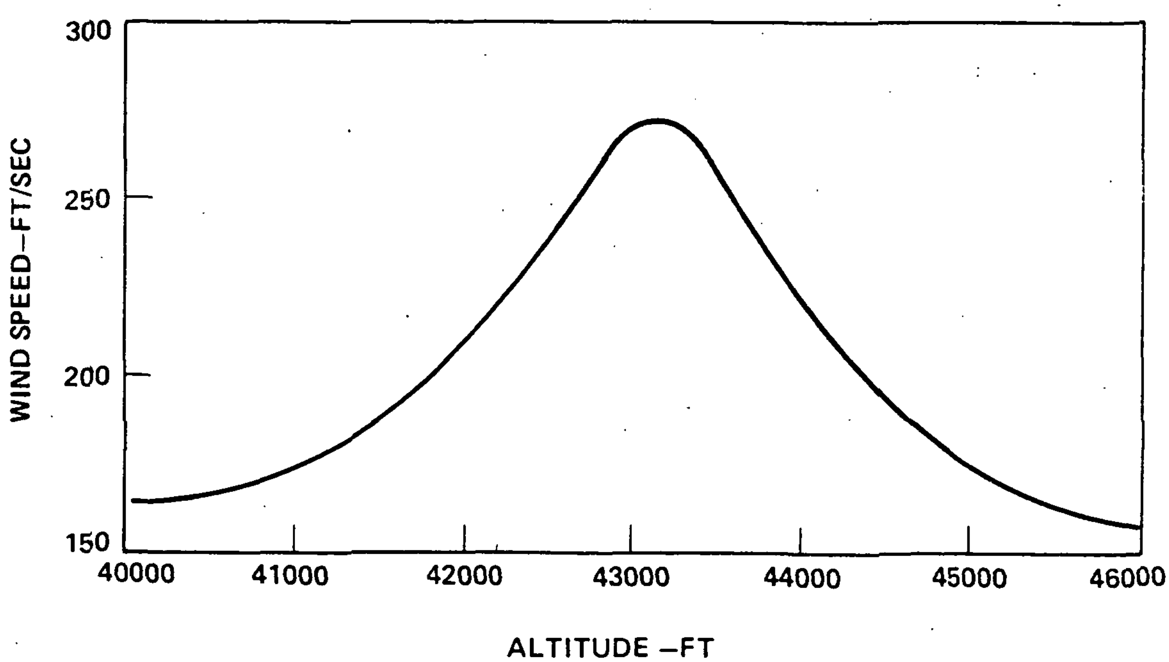


Figure 10. Crosswind profile versus altitude.

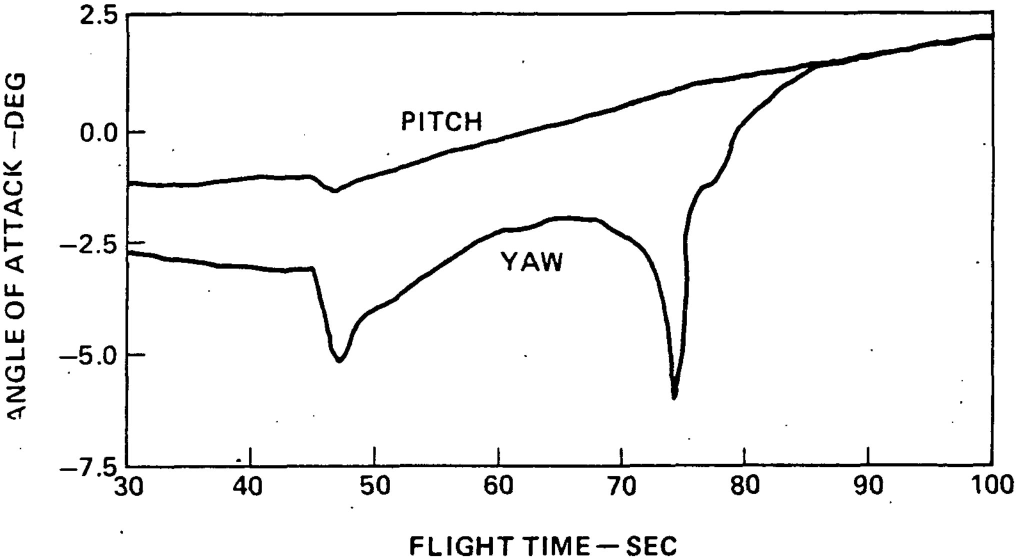


Figure 11. Pitch and yaw angle of attack history with four 675-ft^2 fins.

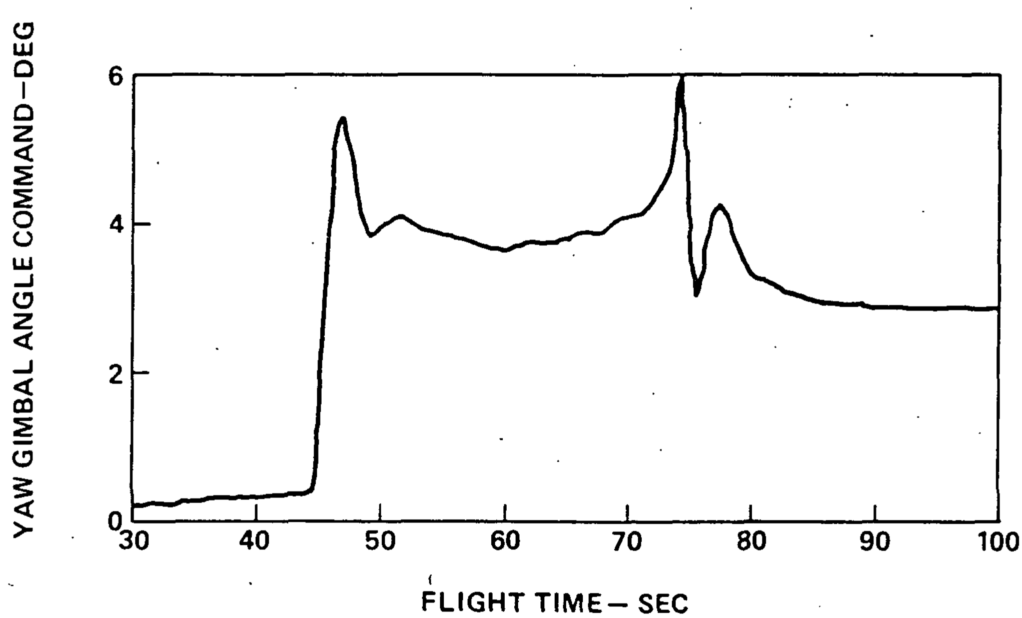


Figure 12. Yaw gimbal angle command history with four 675-ft^2 fins.

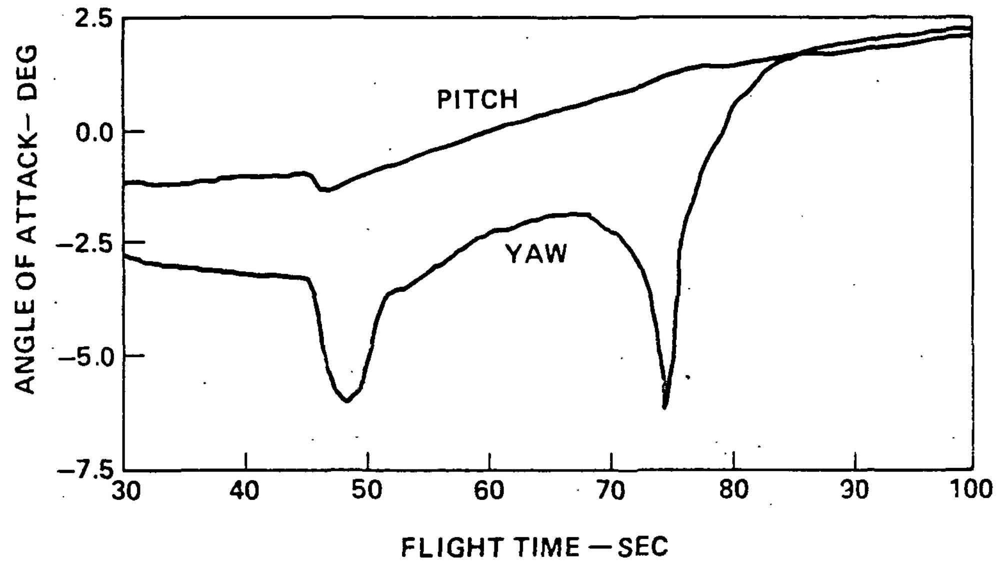


Figure 13. Pitch and yaw angle of attack history with four 300-ft^2 fins.

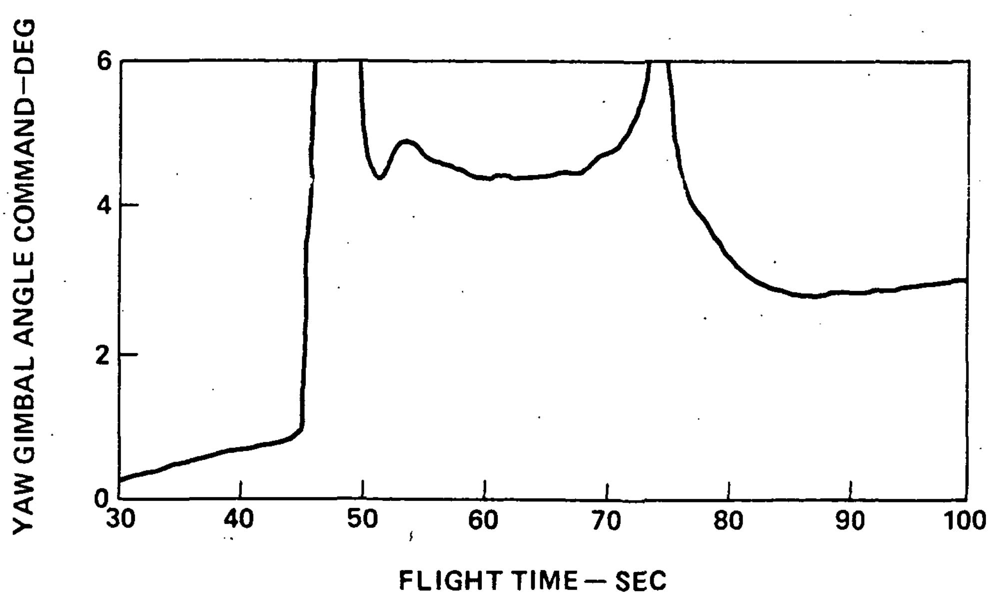


Figure 14. Yaw gimbal angle command history with four 300-ft^2 fins.

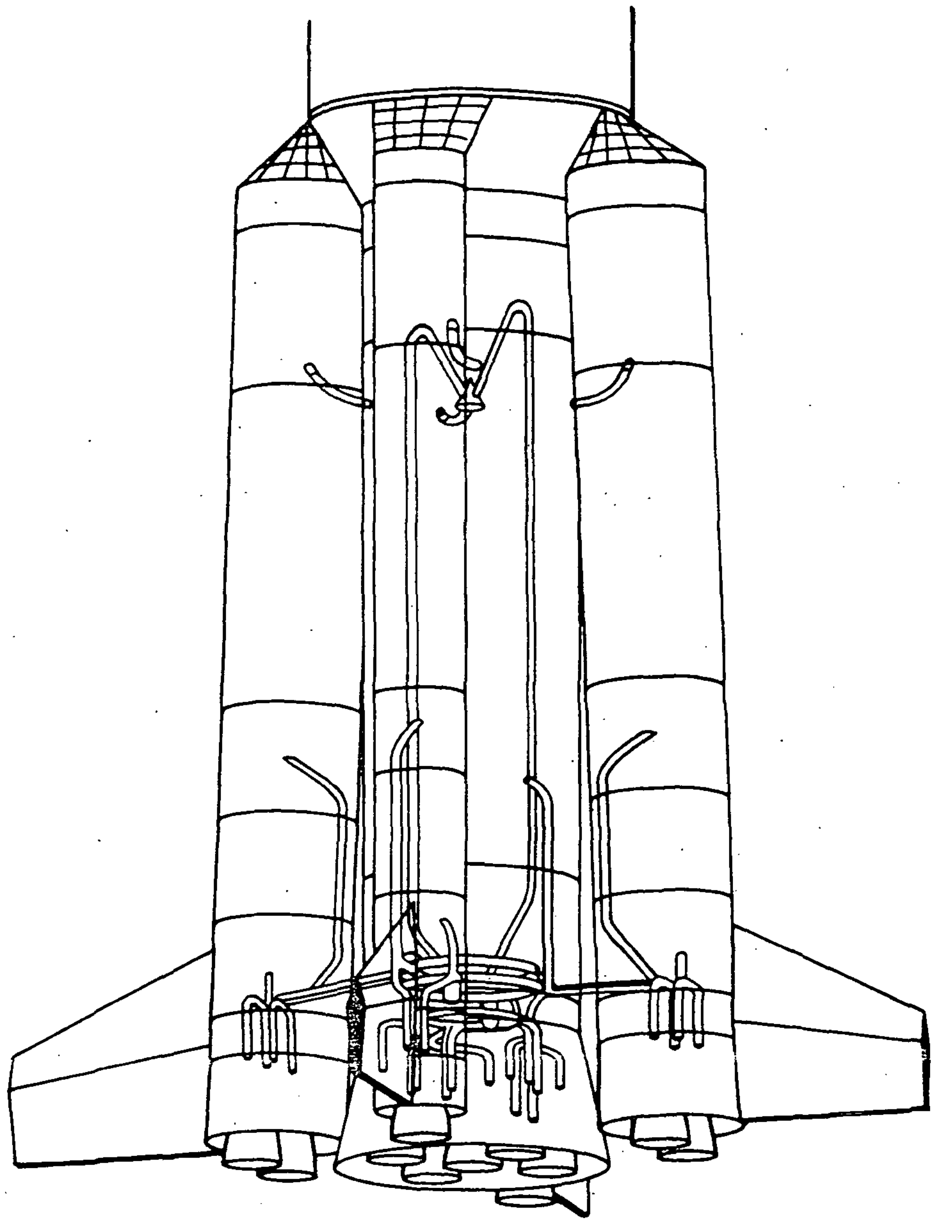


Figure 15. HLLV propellant feed line layout.

	WEIGHT ~ LBS
FWD STRUCTURES	29,332
DEORBIT PROPULSION	3,165
LOX CAPACITY	1,763,876
TANK	15,747
INSULATION	2,369
LH_2 CAPACITY	293,979
TANK	39,696
PROPELLANT FEED SYS.	9,074
AVIONICS	1,294
AFT SKIRT – THRUST STRUCTURE	17,813
CONTINGENCY (15%)	16,262
TOTAL LAUNCH WEIGHT	2,192,606

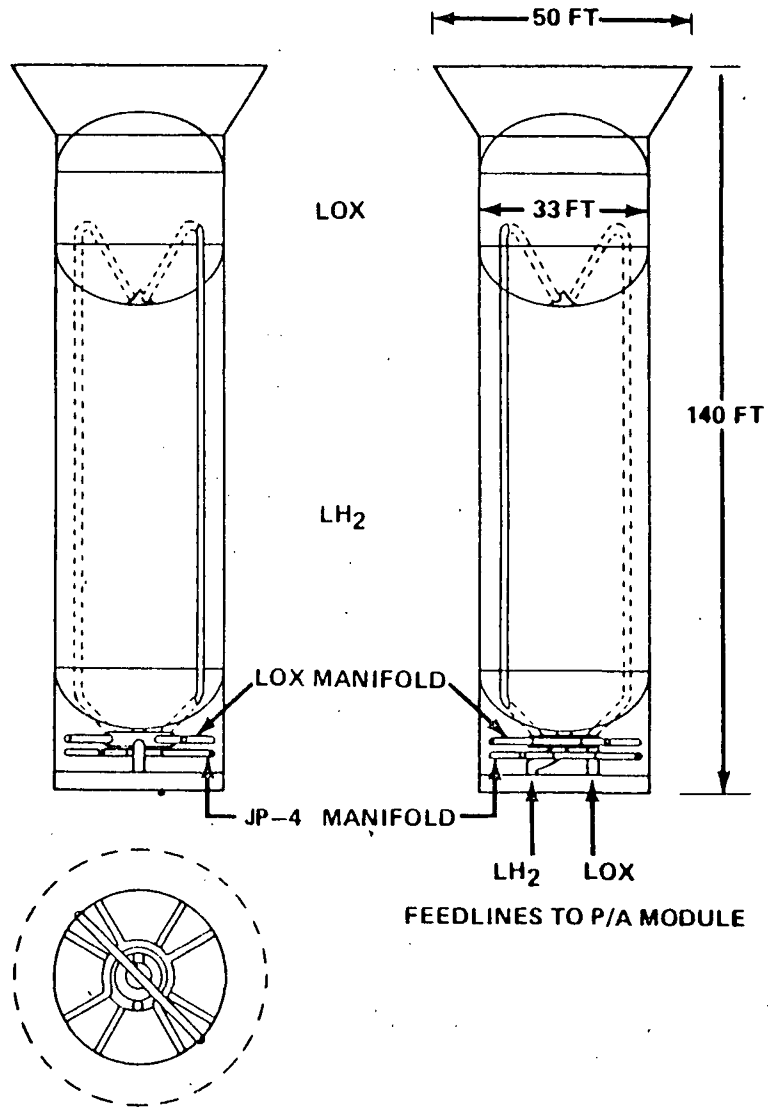


Figure 16. Center core stage description.

	WEIGHT ~ LBS
FWD STRUCTURES	7,300
SEPARATION	1,344
LH_2 CAPACITY	16,769
TANK	8,250
INSULATION	394
LOX CAPACITY	661,368
TANK	17,635
INTERTANK	5,060
PROPELLANT FEED SYS.	2,401
JP4 CAPACITY	200,233
TANK	5,860
AVIONICS	999
RECOVERY	2,600
THRUST AND CONNECTING STRUCTURES	12,940
PROPULSION SYSTEM	19,360
AFT SKIRT AND HEAT SHIELD	2,998
FIN	10,390
CONTINGENCY (15%)	14,328
TOTAL LAUNCH WEIGHT	990,229

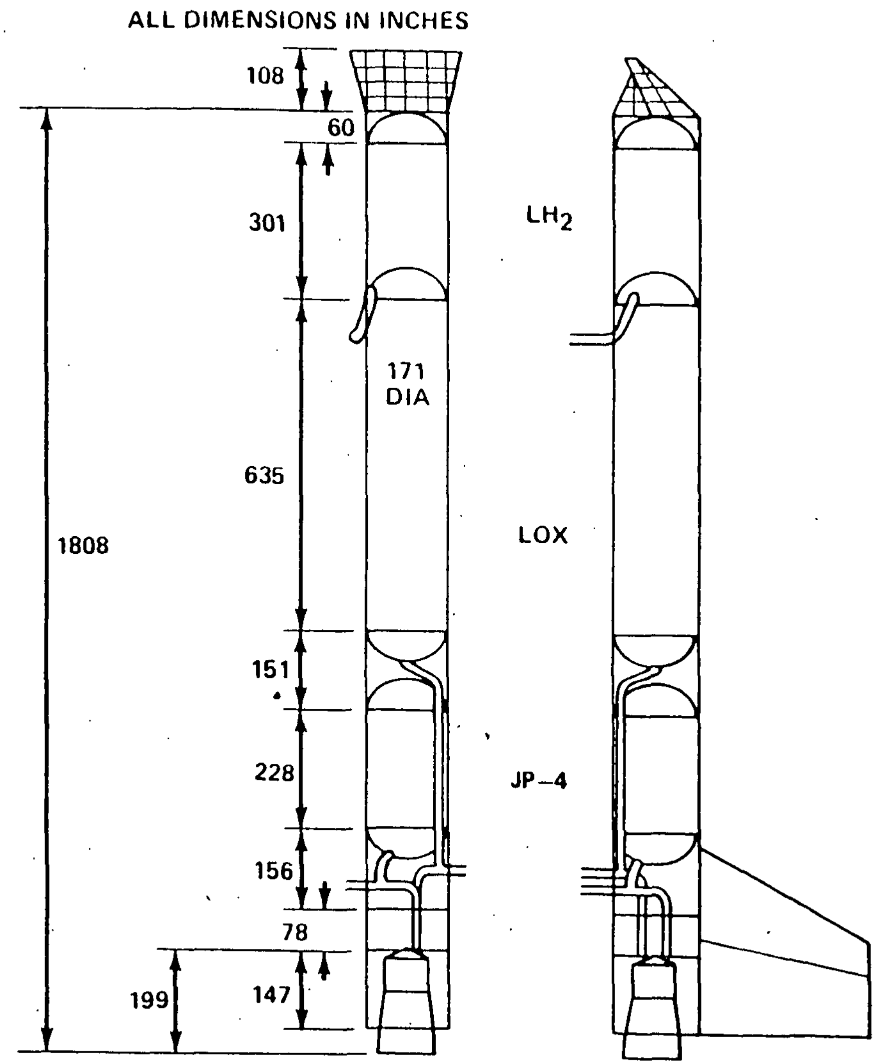


Figure 17. Single engine booster.

	WEIGHT ~ LBS
FWD STRUCTURES	8,255 LBS
SEPARATION	2,016
LH_2 CAPACITY	33,538
TANK	8,882
INSULATION	690
LOX CAPACITY	1,322,737
TANK	19,807
INTERTANK	7,110
PROPELLANT FEED SYS.	4,027
JP4 CAPACITY	400,466
TANK	7,942
AVIONICS	1,426
RECOVERY	4,725
THRUST AND CONNECTING STRUCTURES	24,196
PROPULSION SYSTEM	38,594
AFT SKIRT AND HEAT SHIELD	4,744
FIN	10,390
CONTINGENCY (15%)	20,976
TOTAL LAUNCH WEIGHT	1,920,522

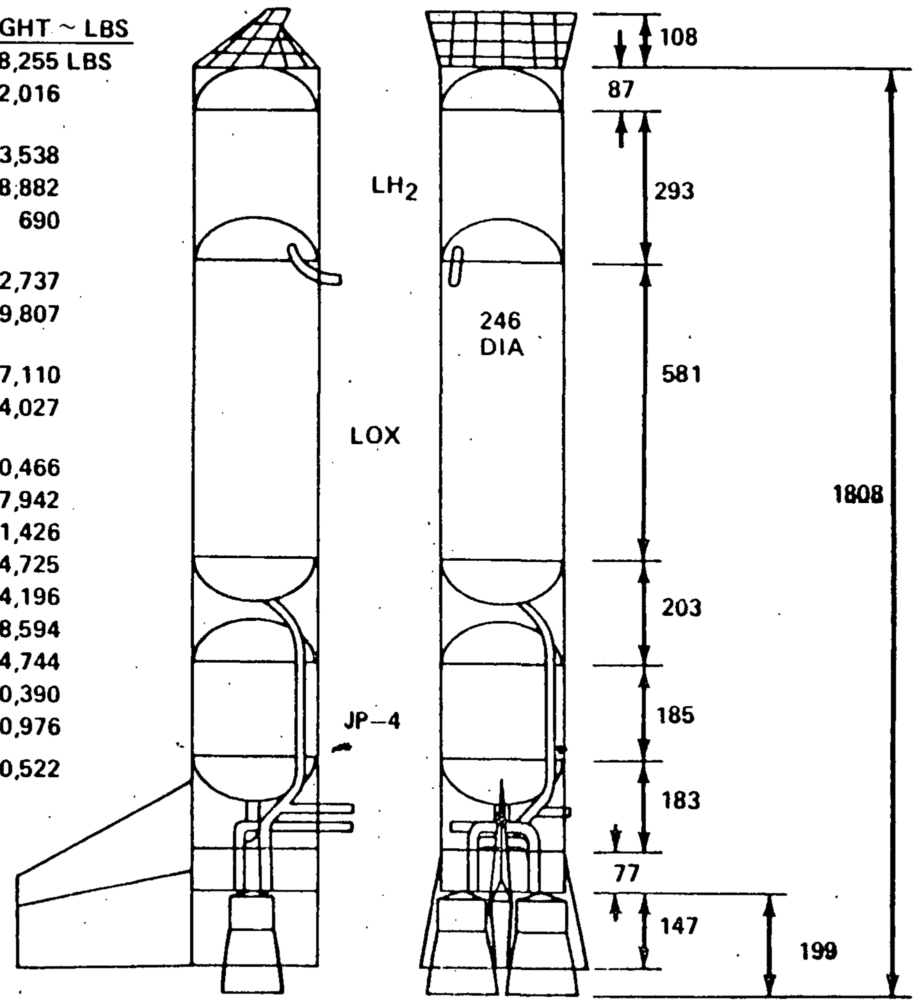


Figure 18. Two engine booster.

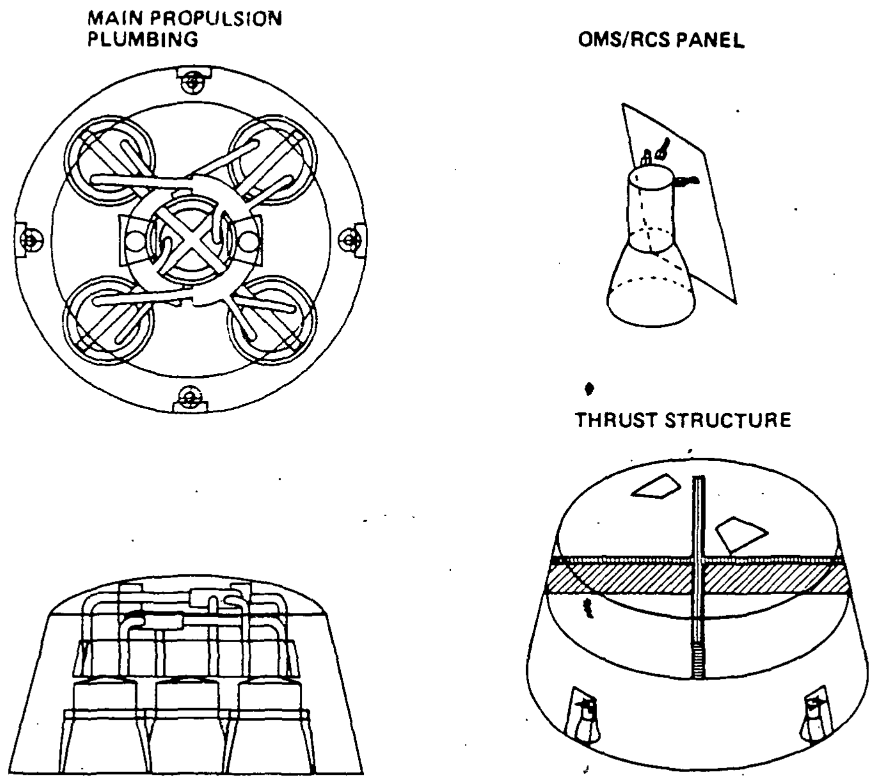


Figure 19. P/A Module structure and propellant lines.

ITEM	WEIGHT – LBS
FORWARD DOME WITH TPS	6084 LBS
AVIONICS AND POWER WITH THERMAL CONTROL	6294
THRUST STRUCTURES	10459
MAIN ENGINES (5)	35710
ANCILLARY SYSTEMS/APU	4925
PROPELLANT LINES	3933
RESIDUALS	6630
RCS/ACS	2064
RCS/ACS PROPELLANTS	10621
RECOVERY SYSTEM	4170
AFT SKIRT WITH TPS	7299
BASE HEAT SHIELD	2724
CONTINGENCY (15% ON NEW EQUIPMENT)	7282
TOTAL LAUNCH WEIGHT	108195

Figure 20. P/A Module component weights.

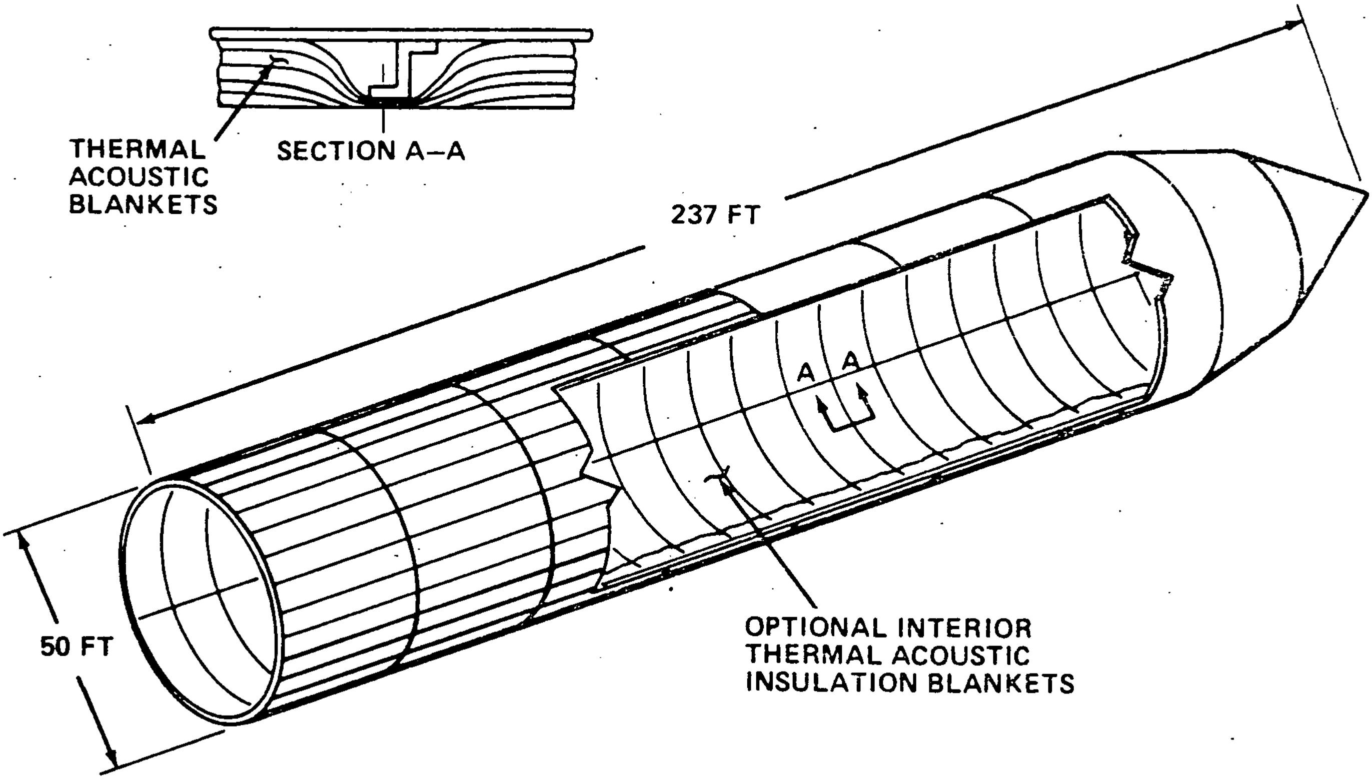


Figure 21. Payload fairing.

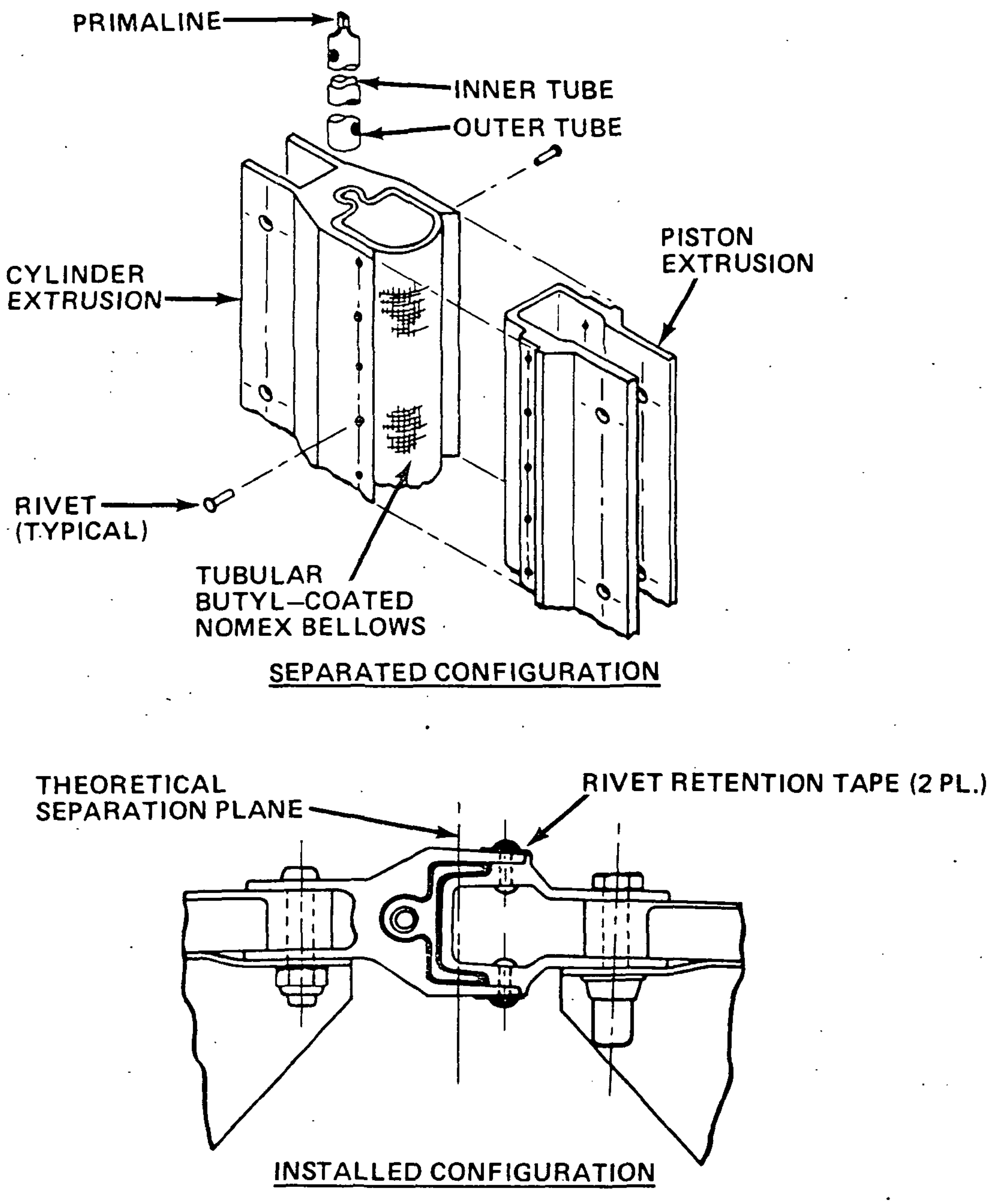


Figure 22. Payload fairing longitudinal separation system.

- STRUCTURAL
- PROPULSION
- FLIGHT CONTROL
- AVIONICS
- SEPARATION
- RECOVERY
- THERMAL PROTECTION
- RADIATION PROTECTION
- SOFTWARE
- GROUND
- PREFLIGHT

Figure 23. Test program elements.

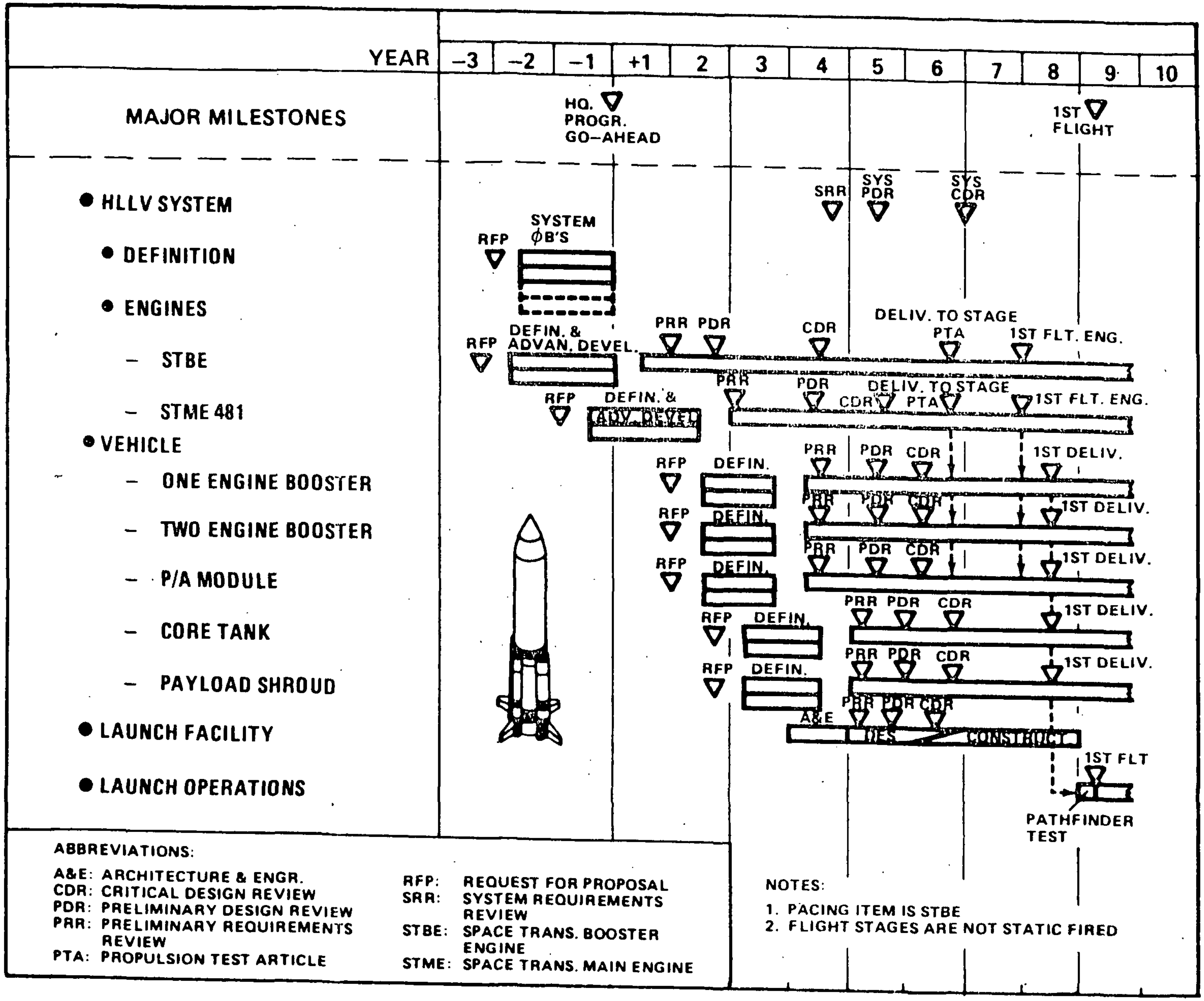


Figure 24. Direct approach development schedule.

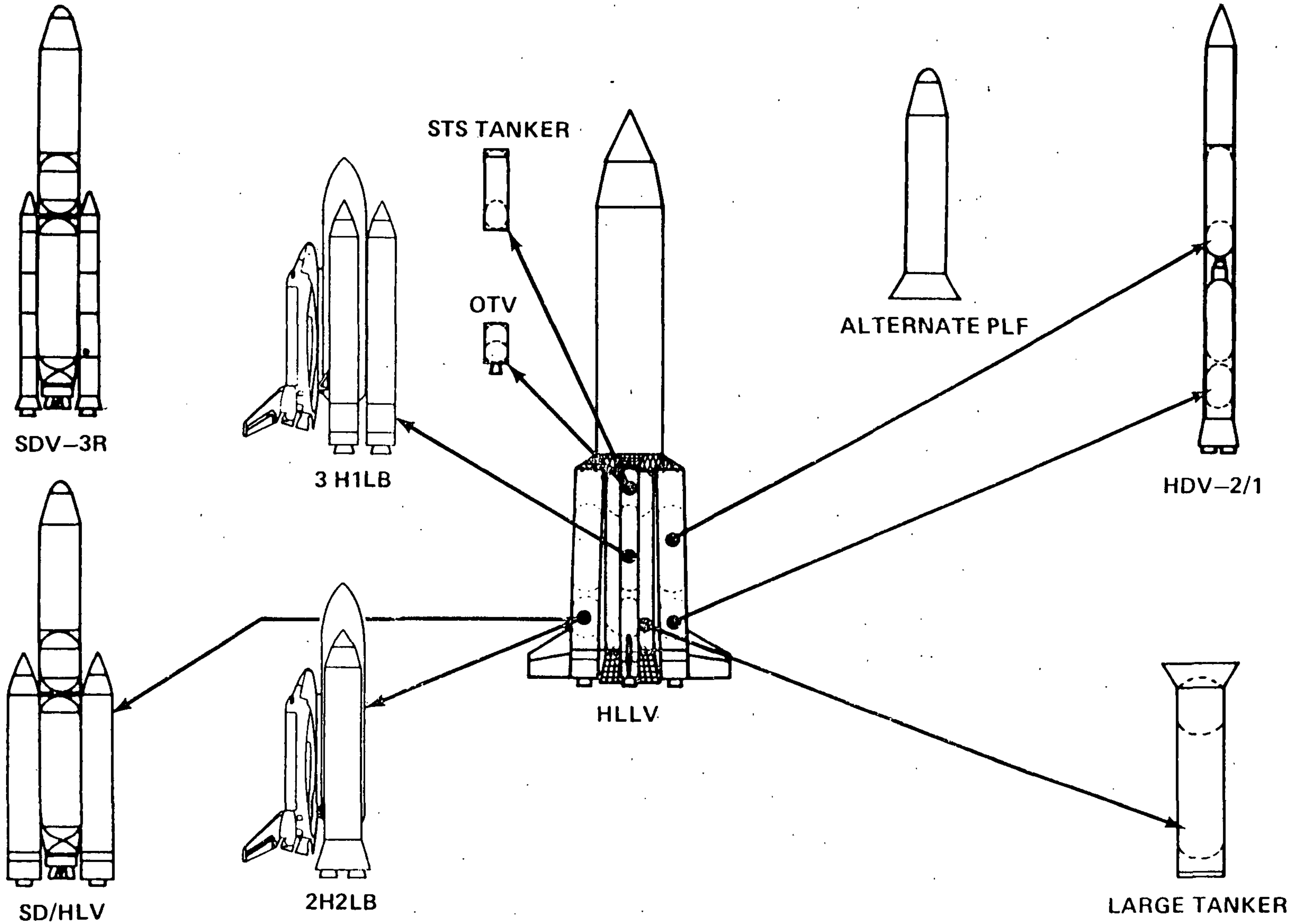


Figure 25. Derivative applications.

TABLE 1. HLLV WEIGHT SUMMARY

PAYLOAD	325,096
SHROUD	126,575
BOOSTERS (2) TWO ENGINE	3,841,044
BOOSTERS (2) SINGLE ENGINE	1,980,457
CORE STAGE 396 INCH DIAMETER	2,192,606
P/A MODULE 396 INCH DIAMETER	108,195
GROSS LIFTOFF WEIGHT	8,573,973

TABLE 2. HLLV DETAILED WEIGHT STATEMENT

UNITS IN LBS.	TWO ENGINE BOOSTER	SINGLE ENGINE BOOSTER	SECOND STAGE 396 IN CORE	SECOND STAGE P/A MODULE
SHROUD/STAGE ATTACH RING	4,465	5,200	14,744	0
FORWARD SKIRT/PL ADAPTER	3,790	2,100	14,588	5,037
LH_2 TANK	8,882	8,250	39,696	0
OXIDIZER TANK	19,807	17,635	15,747	0
INTERTANK	7,110	5,060	0	0
JP4 TANK	7,942	5,860	0	0
AFT SKIRT	3,880	2,680	0	5,991
THRUST STRUCT/CONNECTING	24,196	12,940	17,813	10,459
FINS	10,390	10,390	0	0
SUBTOTAL STRUCTURES	90,463	70,115	102,587	21,486
INSULATION/TPS	690	394	2,369	2,355
BASE HEAT SHIELD	864	318	0	2,724
AVIONICS THERMAL CONTROL	0	0	0	254
SUBTOTAL THERMAL CONTR	1,554	712	2,369	5,333
MAIN ENGINES	35,889	17,945	0	35,710
PROPELLANT FEED SYSTEMS	4,027	2,401	9,074	3,933
ANCILLARY SYSTEMS	2,705	1,415	0	4,925
OMS/RCS	0	0	0	2,064
DEORBIT PROPULSION	0	0	3,165	0
SUBTOTAL PROPULSION	42,621	21,761	12,239	46,632
AVIONICS (SEE TABLE 2a)				
SUBTOTAL AVIONICS	1,426	999	1,294	6,041
SEPARATION SYSTEM	2,016	1,344	0	0
RECOVERY SYSTEM	4,725	2,600	0	4,170
CONTIN (15% ON NEW EQUIP)	20,976	14,328	16,262	7,282
TOTAL DRY WEIGHT	163,781	111,858	134,751	90,944
RESIDUALS	12,293	6,146	3,860	6,630
TOTAL BURNOUT WEIGHT	176,074	118,004	138,631	97,574
USEABLE PROPELLANTS	1,744,448	872,224	2,053,975	10,621
TOTAL LAUNCH WEIGHT	1,920,522	990,228	2,192,606	108,195

TABLE 2a. DETAIL WEIGHT OF AVIONICS (lb)

	TWO ENGINE BOOSTER	SINGLE ENGINE BOOSTER	346 INCH CORE	P/A MODULE
TT&C	0	0	304	341
CENTRAL PROCESSOR & DATA STORAGE	0	0	0	340
GN&C	0	0	0	534
FLIGHT CONTROL	0	0	0	210
SEPARATION & LANDING CONTROL	374	374	350	300
AUXILIARY CONTROL	91	45	90	60
ENGINE CONTROL & TVC	560	280	0	142
COAST & DEORBIT CONTROL	0	0	190	217
PAYLOAD/GROUND INTERFACE	0	0	60	60
BOOSTER FOR ELECTRONICS	100	100	0	0
SAFETY	32	22	27	27
CABLES	254	168	208	627
ELECTRICAL POWER	15	10	35	1905
SUBTOTAL AVIONICS	1426	999	1294	6041

TABLE 3. PAYLOAD FAIRING WEIGHT SUMMARY (lb)

STRUCTURES	
FWD CONE SEGMENTS AND FRAMES	8,860
FRUSTUM CONE SEGMENTS AND FRAMES	12,780
FWD CYLINDRICAL SEGMENTS AND FRAMES	32,930
AFT CYLINDRICAL SEGMENTS AND FRAMES	51,440
THERMAL PROTECTION	
FWD NOSE CONE INSULATION	1,390
SEPARATION SYSTEM	
SEPARATION MECH	2,665
CONTINGENCY (15%)	16,510
TOTAL WEIGHT	126,575

APPENDIX A. AERODYNAMICS

The static aerodynamic characteristics of HLLV Configuration II are presented and were used in support of performance, structural, and control studies. Figure A-1 defines the coordinate system for the aerodynamics. The vehicle axial force is presented as a forebody coefficient (Fig. A-2) and base force (Fig. A-3). The base component is presented as a function of altitude and can be treated as a thrust component. The base force component is based on Saturn V base pressure measurements.

The vehicle normal force slopes (C_{N_α}) are presented in Figure A-4 for the body alone (no fins), with 300 ft^2 fins, and 675 ft^2 fins. The data are applicable to angles of attack to ±6 deg. The side force coefficients (C_{Y_β}) are presented (Fig. A-5) in the same format as the normal force.

The vehicle center of pressure locations are presented for the pitch (Fig. A-6) and yaw planes (Fig. A-7) for the body alone and body with each fin size. The CG location as a function of Mach numbers is shown on each CP chart. The difference between the CG and CP is the static stability margin.

The distribution of local normal force coefficient slope $(dC_{N_\alpha})/d(X/D)$ is presented in Figure A-8 at M = 1.55 which is the point of maximum dynamic pressure. These data were used to determine the maximum vehicle bending moment. Integration of the distributed load plus component loads equals the total normal force slope.

$$C_{N_\alpha} = \int \left(\frac{dCn}{d(X/D)}\right) d(X/D) + C_{N_{\alpha B}} + C_{N_{\alpha F}}$$

The local axial force coefficient distribution $[dC_A/d(X/D)]$ at M = 1.55 for q max shown in Figure A-9 is used to determine the maximum compressive loads for the shroud.

Booster Reentry Aerodynamics

Booster reentry drag curves are presented in Figures A-10 and A-11 for a side-first, tumbling and end-first reentry for each size booster. Reentry trajectories were run for each condition.

P/A Module Aerodynamics

The P/A Module axial force coefficient used in the reentry analysis is presented in Figure A-12. It was assumed that the Module would fly at an L/D of zero. Axial force (Fig. A-13) and normal force distribution (Fig. A-14) are provided at M = 1.0 for the maximum dynamic pressure of 458 psf, for use in structural analysis. An angle of attack of 10 deg was assumed for the normal force distribution to cover all angle of attack dispersions during reentry. These data were used in generating reentry trajectories required for thermal and structural designs.

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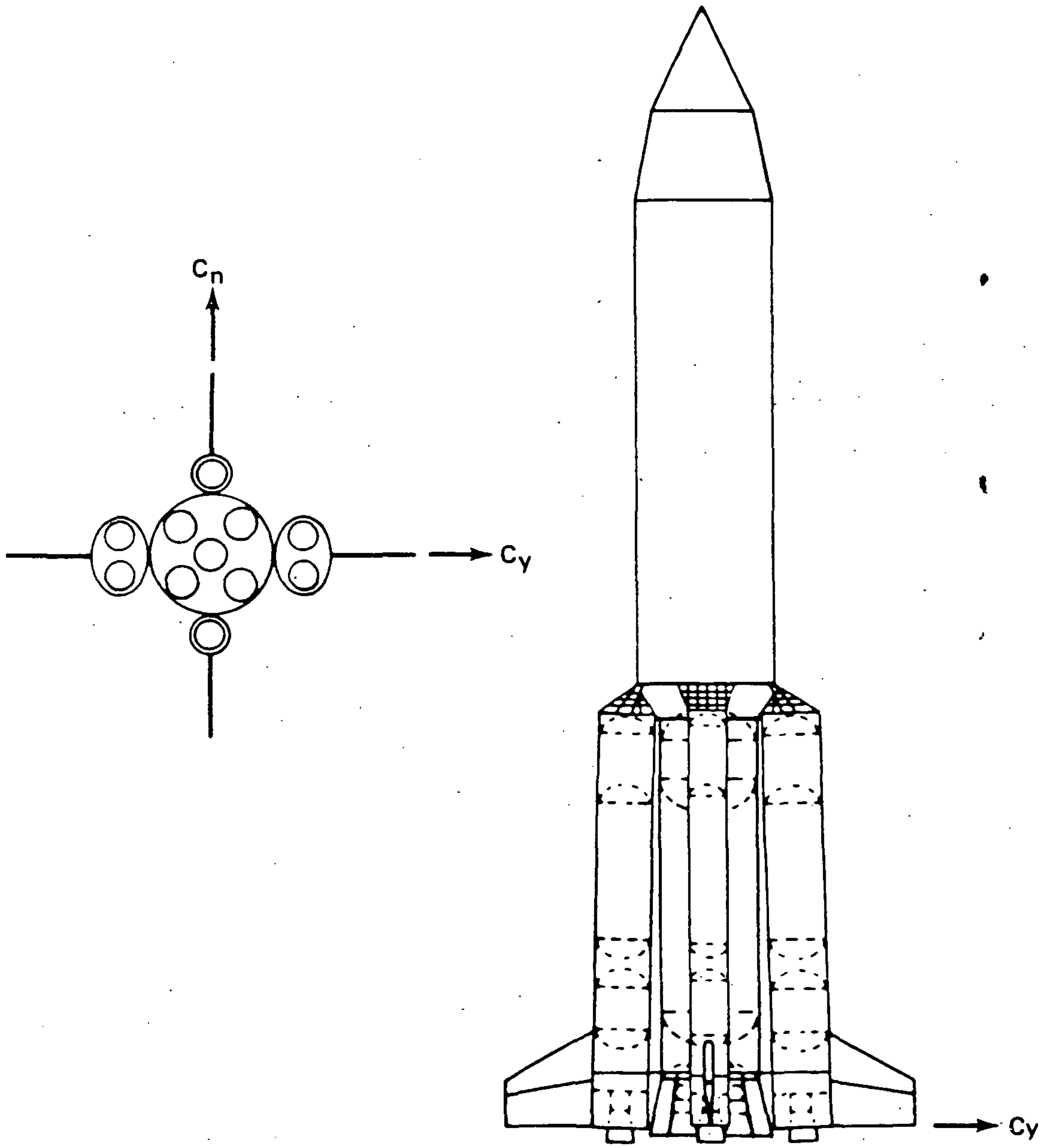

Figure A-1. HLLV aerodynamic coordinate system.

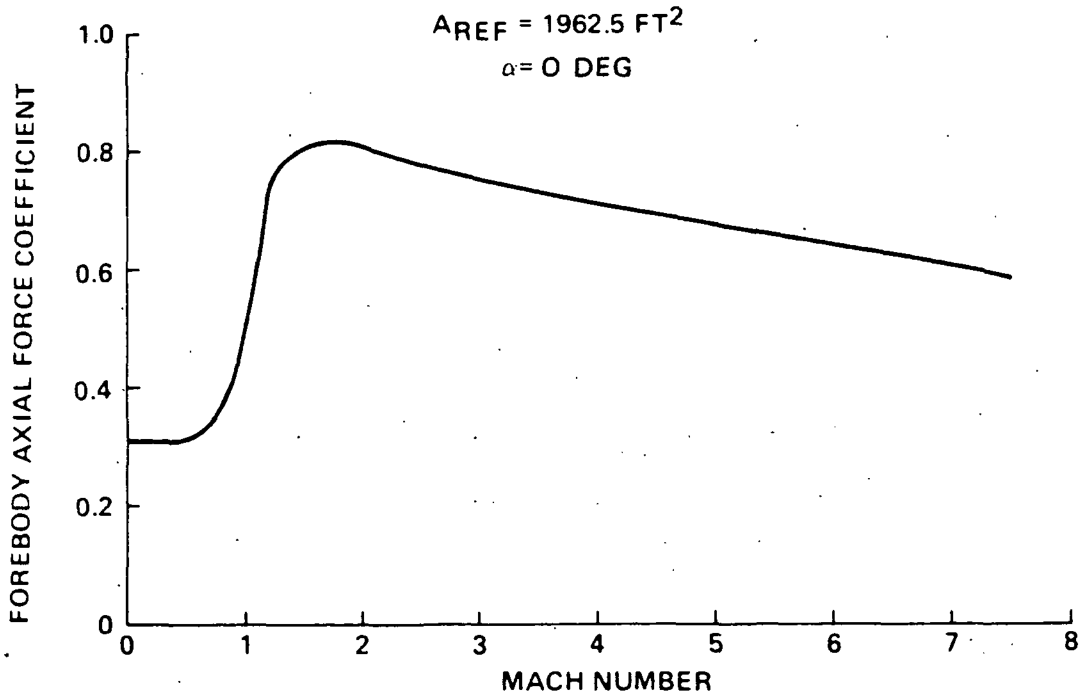

Figure A-2. Forebody axial force coefficient.

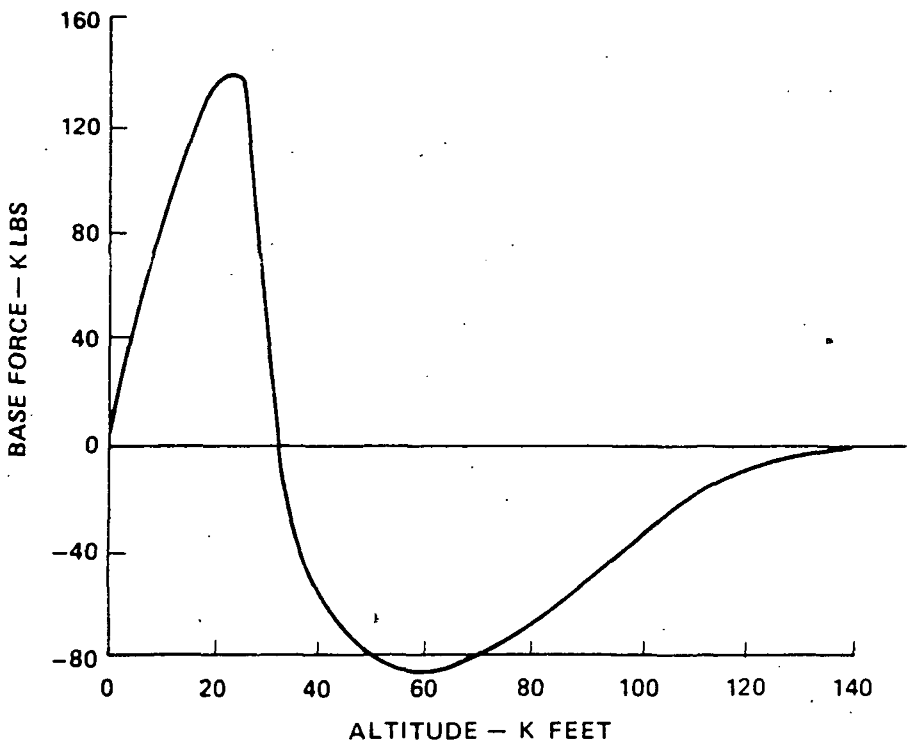

Figure A-3. Base axial force.

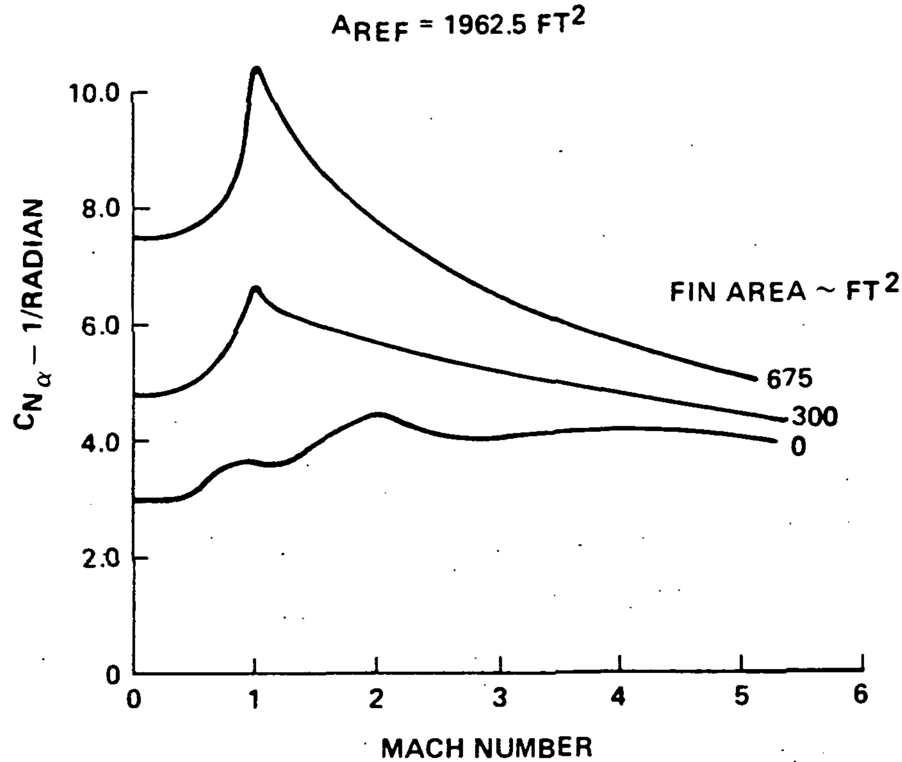

Figure A-4. C_{N_α} versus Mach number as a function of fin area.

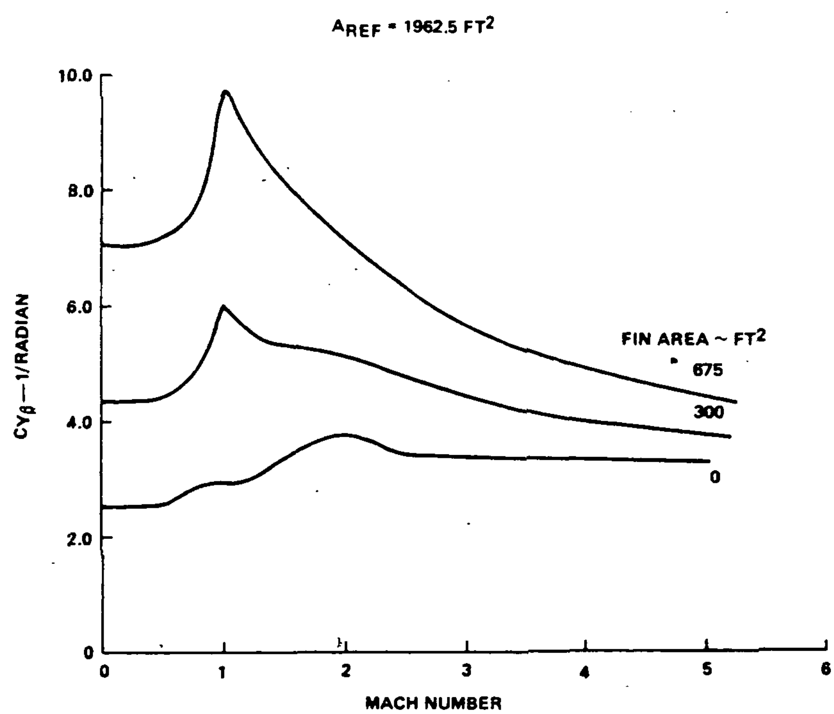

Figure A-5. C_{Y_β} versus Mach number as a function of fin area.

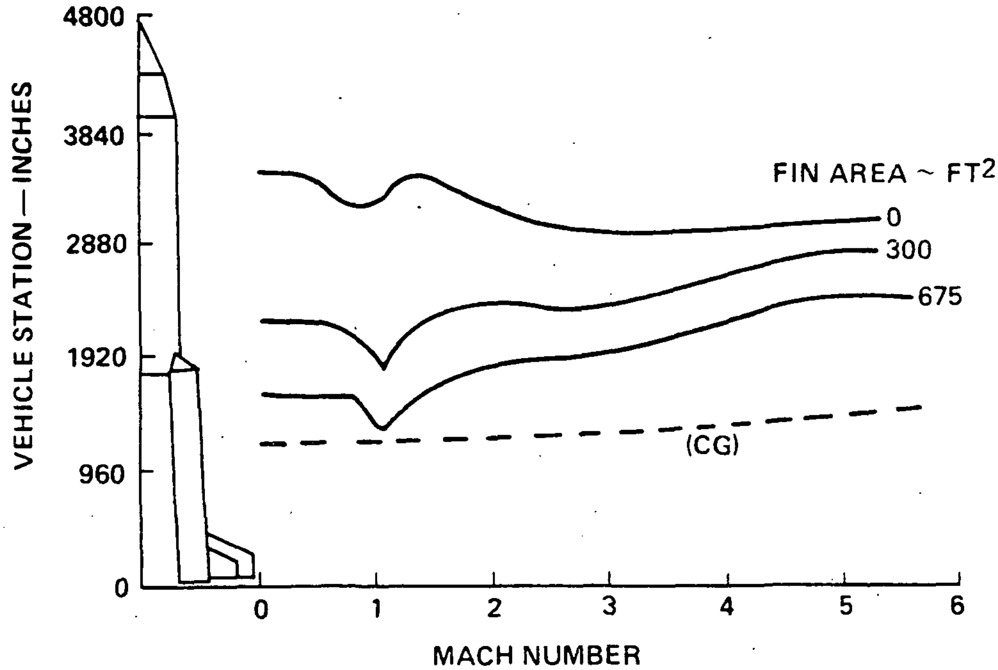

Figure A-6. Pitch plane center of pressure variation.

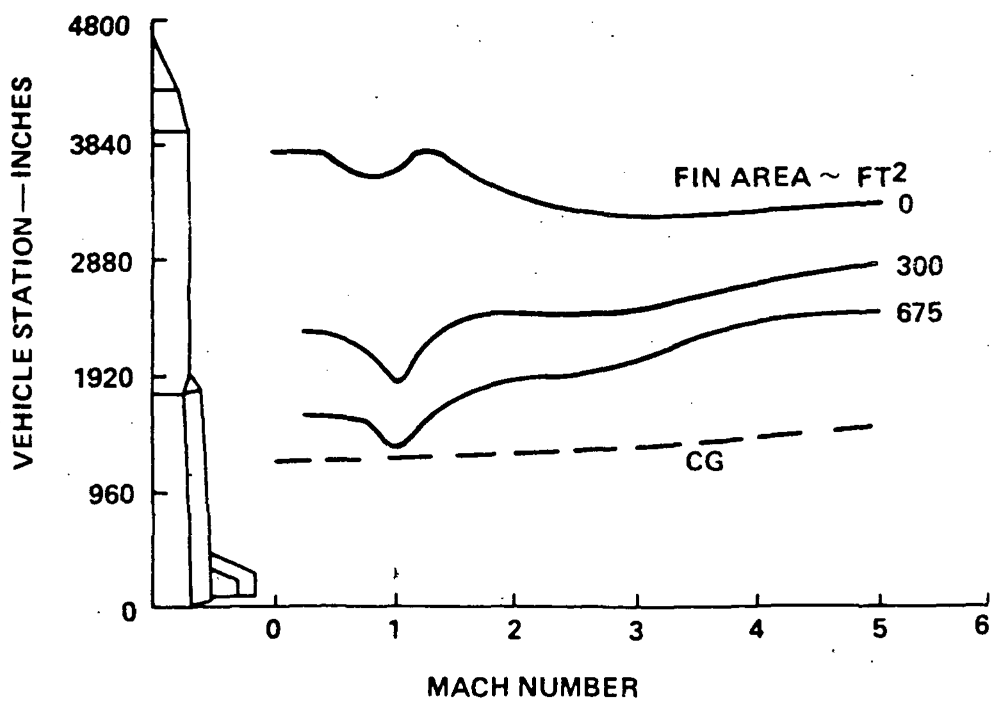

Figure A-7. Yaw plane center of pressure variation.

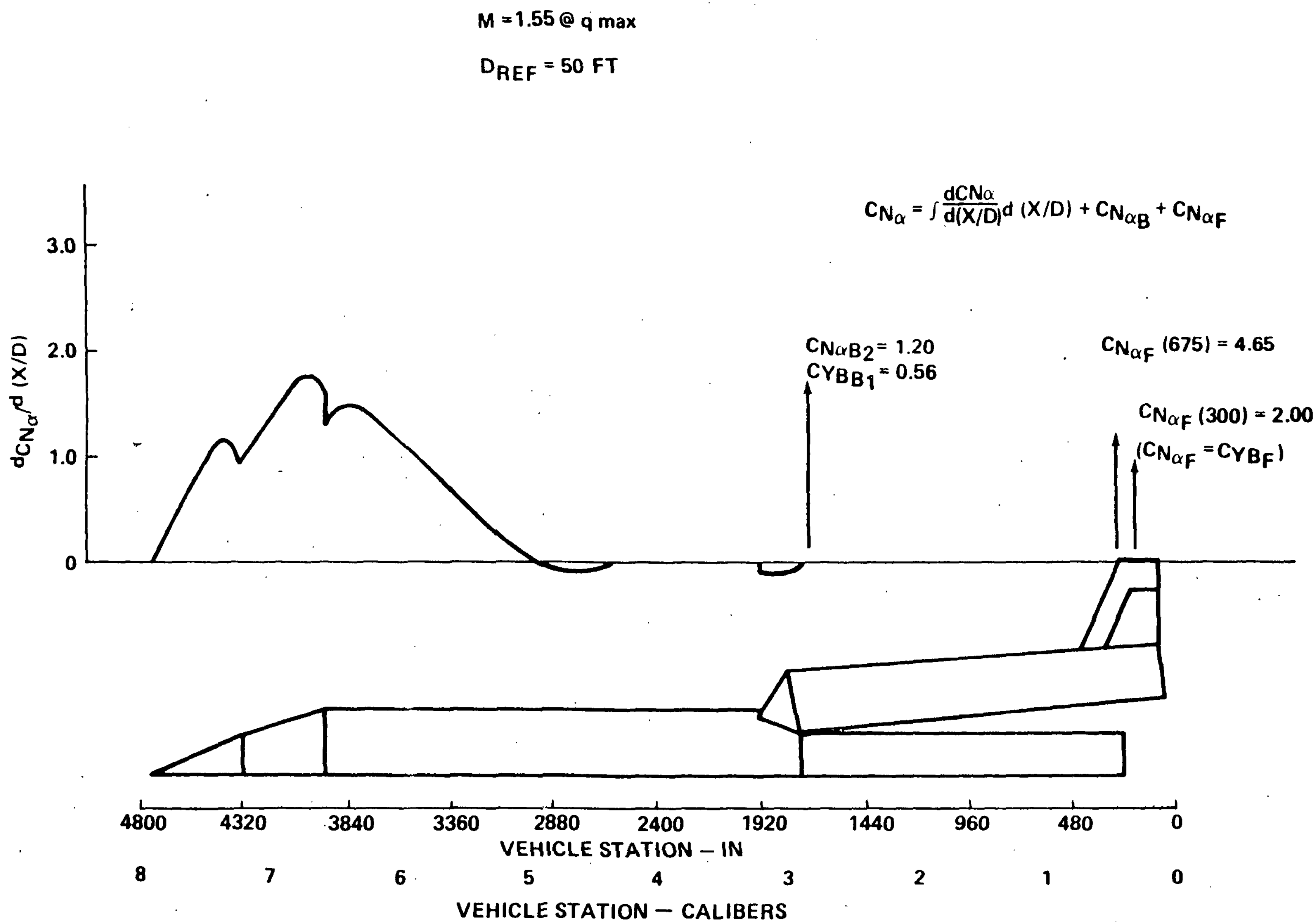

Figure A-8. Local normal force coefficient slope.

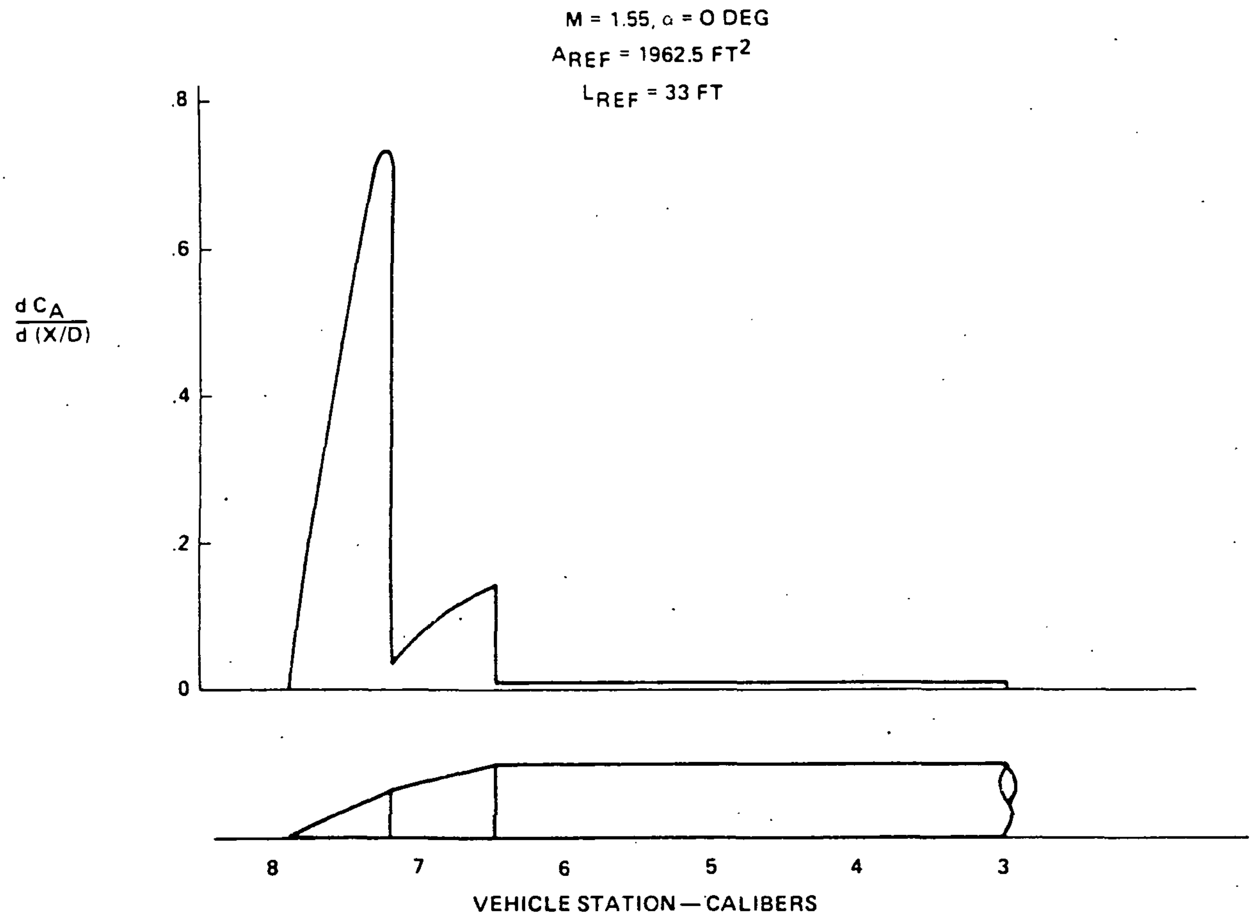

Figure A-9. Distribution of local axial force coefficient.

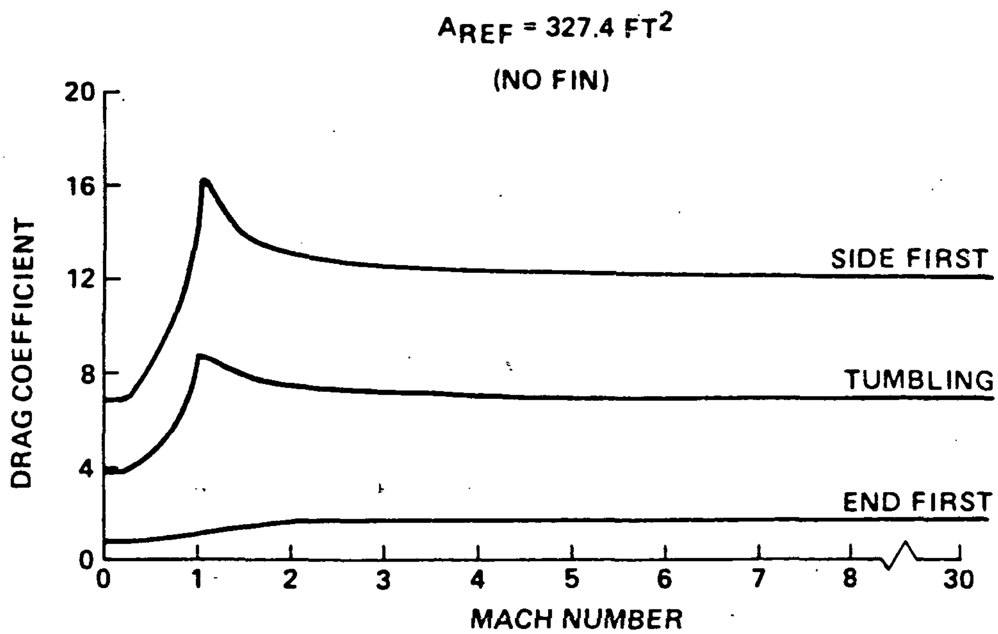

Figure A-10. Two engine booster reentry drag coefficient.

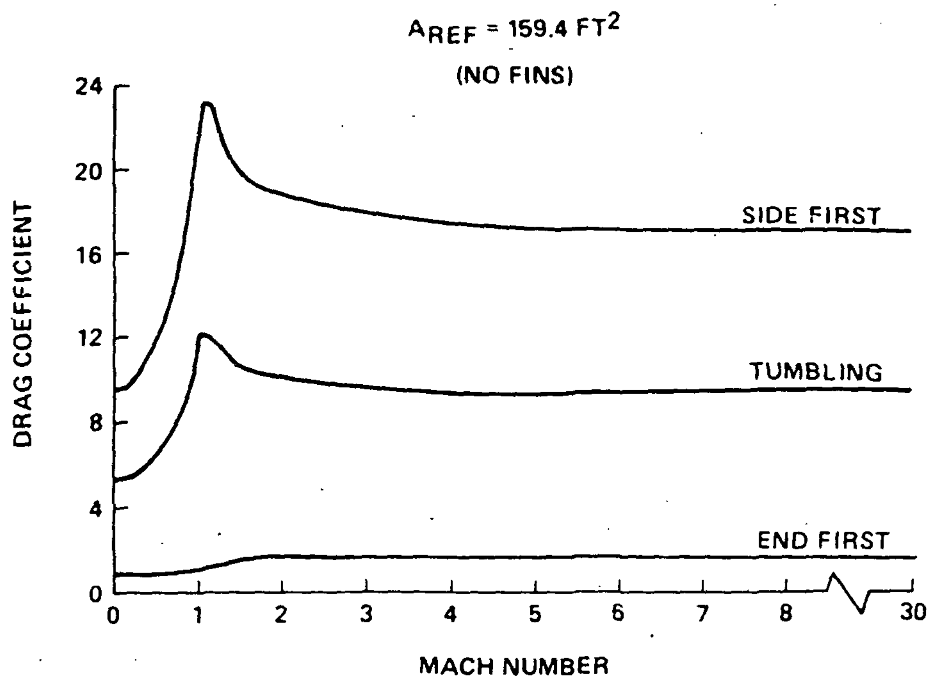

Figure A-11. Single engine booster reentry drag coefficient.

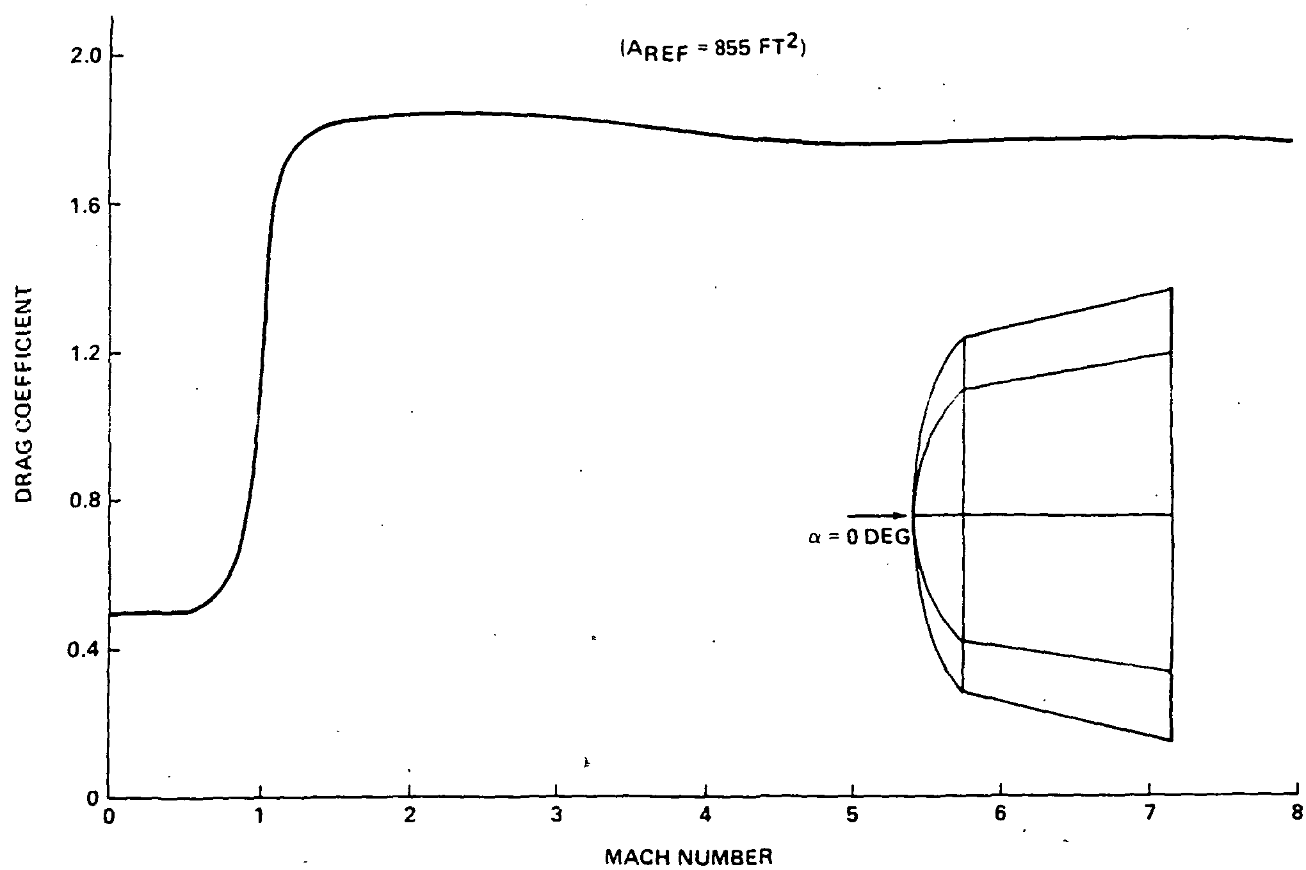

Figure A-12. Propulsion/avionics module reentry drag coefficient.

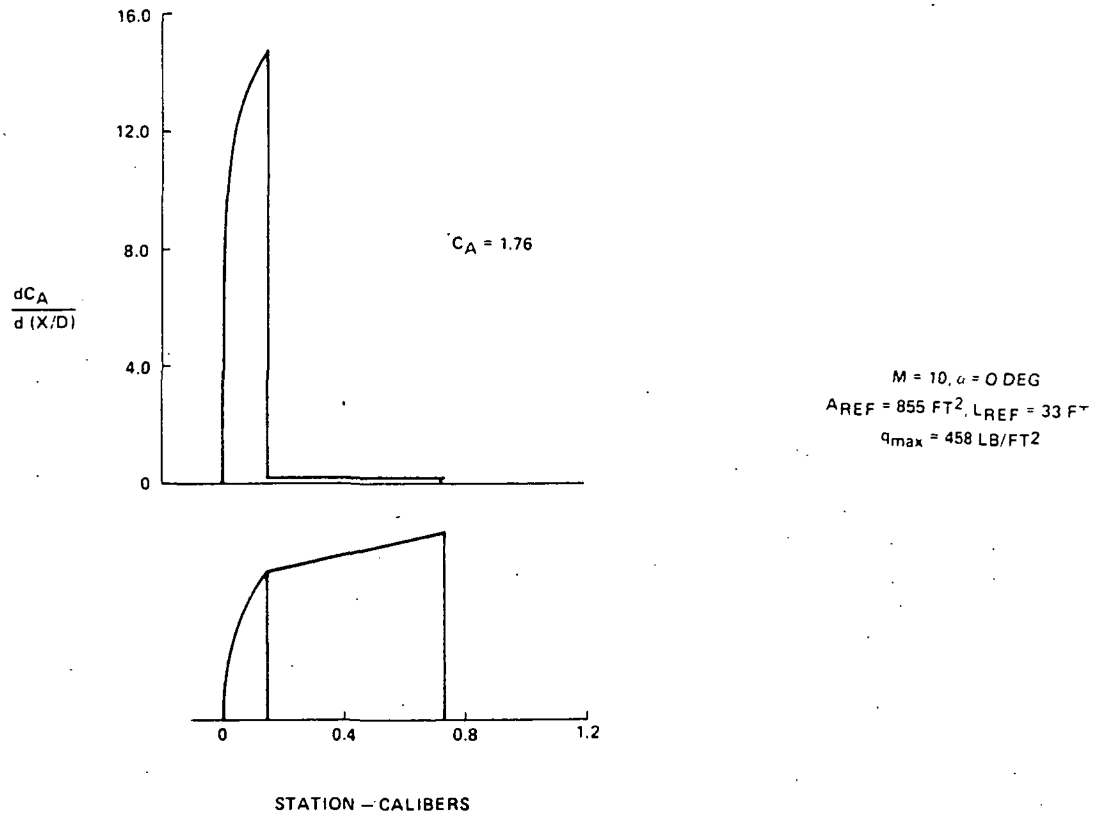

Figure A-13. P/A Module reentry local axial force coefficient.

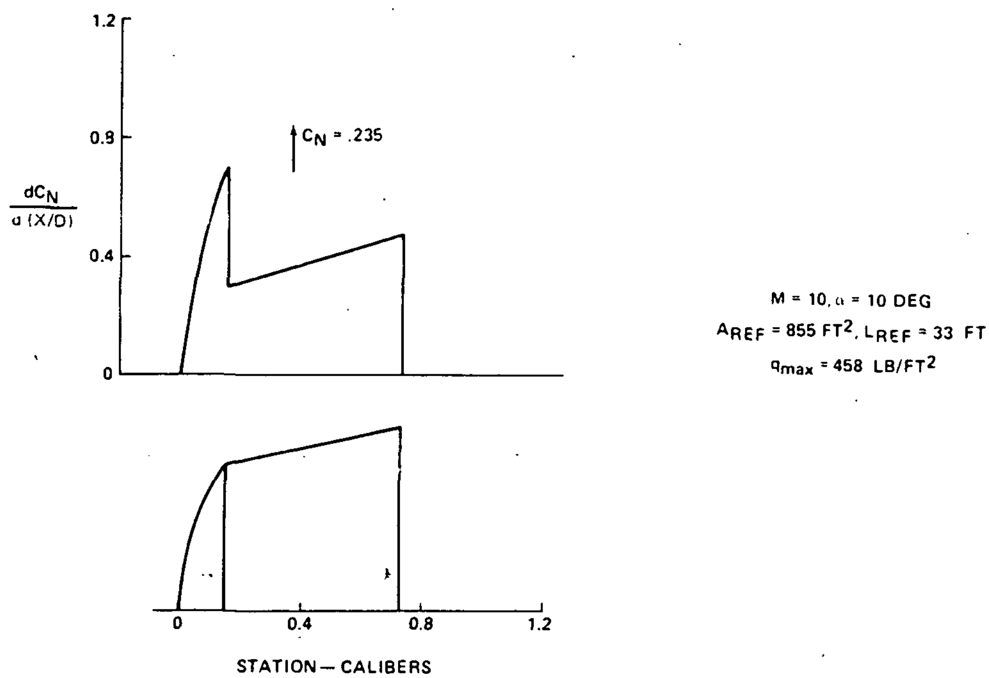

Figure A-14. P/A Module reentry local normal force coefficient.

APPENDIX B. AVIONICS AND SOFTWARE

Avionics

Launch vehicles for the mid-90s and beyond will require avionics and software that will lead to improved performance with reduced turnaround costs in comparison with current launch vehicles. Because the avionics for future launch vehicles will be reusable, new concepts and operational methods are required to enable rapid turnaround, reconfiguration, checkout, and launch with maximum assurance of success and minimum effort and cost. New concepts and operational methods are made possible by new technology, much of which already exists and some of which is in the pipeline. Cost savings can be realized by establishing commonality with other future NASA programs, including the Space Station, the Orbital Transfer Vehicle, and the Second Generation Shuttle. The following paragraphs describe the avionics functional requirements, discuss some of the desired attributes and characteristics, and identify studies needed to focus future planning and definition.

Avionics Functional Requirements

The HLLV avionics is composed of several major subsystems: Telemetry, Tracking, and Control; Data Processing; Guidance and Navigation; Flight Control; Propulsion Control; Auxiliary Control; Electrical Power; and Range Safety Control. Figure B-1 shows the major avionics interfaces.

The primary function of the launch vehicle avionics has been to provide guidance, navigation, control, and flight sequencing from liftoff through orbit insertion, earth return and recovery. Future avionic systems have many additional functions which are identified in Table B-1. Functions on HLLV may present a significant departure from past launch vehicle programs. The onboard checkout for HLLV, for example, is conceived to be much more capable and more automated than the current systems.

Avionics Desired Attributes

The nature of the HLLV program places high emphasis on low risk to the mission hardware and low operational costs, even at the expense of greater front end costs for design, development, and test. The reusable avionics will be designed with a great deal of flexibility for accommodating a variety of vehicle configurations and missions, and it will be easily reconfigured from mission to mission. Innovative concepts will be employed to minimize turnaround time and costs between missions. Checkout and launch schemes will be developed to minimize downtime, maximize readiness, and enable rapid launch from a standby mode with a high probability of success. To reduce risk to the mission and flight hardware, a high degree of fault tolerance will be designed into the avionics system. These and other desirable attributes are discussed further in the following paragraphs.

Avionics Technology

In order to meet the objective of improved performance at minimum risk and reduced turnaround time and costs, the HLLV avionics will capitalize on current and evolving technology. Many of the avionics technology needs for the HLLV are

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identified in Table B-1. Advanced technology in data processing will enable much higher levels of automation (approaching artificial intelligence) in design, analysis, mission planning, reconfiguration, checkout, and launch. Utilizing technology advances in microelectronics, self-test and checkout capability can be built into the onboard system to minimize test connections and reintegration complexities between missions. Advanced distributed fault tolerant processing architectures and methodologies can be utilized to provide very reliable flight systems that can be partitioned for vehicle modularity, contractual separability, and simplification of interfaces. Trends in industry and the military to standardize data bus protocols, software language, etc., offer an opportunity for savings and flexibility through commonality with other programs. Other advances, such as higher capacity energy storage and improved GN&C sensor accuracy and reliability are also highly beneficial.

As indicated by Figure B-2, the current Shuttle technology is already 10 to 15 years old and appears to offer very little heritage for the next generation of vehicles. However, there does appear to be an opportunity to share development with a possible second generation Shuttle program.

Avionics/Software Fault Tolerance

Fault tolerance in flight avionics and software is required to meet certain safety standards and to attain stringent avionics system reliability goals. Fault tolerance in systems for checkout, launch and other ground operations is needed to minimize interruptions, launch holds, and launch scrubs. Although efforts have been made to achieve completely automated fault tolerance in the past, e.g., Shuttle and IUS, these attempts have never quite fully reached their initial goals because of the high cost and complexity involved.

NASA is currently conducting a program to develop an Advanced Information Processing System (AIPS). The objectives of this program are: (1) design a distributed, fault tolerant and damage tolerant system which will capitalize on current and future developments in microelectronics, (2) develop supporting methodologies for system design evaluation and verification, and (3) demonstrate the viability of architecture commonality through proof of concept system development. The architectures and methodologies coming out of this program will be very useful in the design of a launch vehicle distributed data processing system.

Figure B-3 shows a fault tolerant network envisioned for the HLLV avionics, which utilzies some of the architecture concepts being studied by the AIPS program. The network consists of a number of processing sites which are physically dispersed, each processing site having a general purpose computer (GPC). GPC's can have varying levels of throughput, memory size, fault tolerance, and damage tolerance. All processing sites are linked together by a fault tolerant and damage tolerant communication network. Each processing site has access to one or more I/O buses to establish local networks. For the HLLV avionics, it is desirable that device interface units (DIUs) can be connected to either the global network or to the local networks.

The avionics conceived for HLLV employs triplex networks for data processing and power distribution. These networks are capable of interfacing with N-plex sensors, effectors, and other devices, the redundancy of which is dictated by reliability and safety factors. Techniques will be evaluated for averting software failures caused by mistakes in design or translation.

Avionics Reusability and Software Reconfigurability

The amount of savings that can be realized by reusing avionics depends on how efficiently the avionics and software can be turned around between missions. Figure B-4 shows the turnaround flow for avionics and software from mission to mission. The major cost drivers are hardware refurbishment, systems integration, and software reconfiguration.

The cost of hardware refurbishment can be minimized by: (1) utilizing long life components designed to survive launch and recovery environments, (2) protecting the hardware from launch and recovery environments, (3) designing equipment installation and interfaces for easy removal and replacement, and (4) providing means for damage assessment, cleanup, repair, and reverification without disassembly. Much of the avionics can be grouped together in a protective package. However, there are many items of equipment, in the engine area for example, which will be difficult to protect. This equipment will have to be designed to survive exposure to the launch and recovery environment.

System integration cost between missions is a function of the amount of equipment that must be changed out, the number and complexity of interfaces between major vehicle elements, the amount and complexity of software changes between missions, and the number of test connections. To reduce costs, steps will be taken to: (1) minimize the necessity for changing out equipment, (2) keep interfaces simple between major vehicle elements, and (3) minimize test connections. Ways to minimize software costs are discussed later. One way to minimize equipment changeout is simply to design and package the equipment for long-life survival over a large number of missions. Another way is to provide the capability for refurbishment, cleanup and verification without having to remove the equipment. Interfaces between major vehicle elements can be kept simple by utilizing a modular system approach. With this approach, complex dynamic interfaces are contained within a major vehicle element, requiring only simplified data traffic and power transfer between major vehicle elements. Each vehicle element has its own capability for data and command processing and power conditioning and distribution. Test connections at all phases of testing can be reduced practically to zero by implementing onboard built-in checkout capabilities. Checkout commands and data to any vehicle element or component can be transmitted via the data bus interface.

Probably the biggest mission-to-mission cost driver is in the area of software reconfiguration. As shown in Figure B-4, flight-to-flight activities begin with the definition of mission objectives and payloads. Requirements are then developed which in turn support mission planning and vehicle configuration definition, analyses are then performed to assure vehicle integrity and performance capability for desired mission. Mission planning and targeting is then performed, and targeting constants are generated and verified before being loaded into the various vehicle processors. Software/hardware integration tests are then performed at the various levels of integration. All of these steps are involved in the process of software reconfiguration between missions.

There are at least three levels at which software needs to be controlled: (1) the operating system software, which essentially remains fixed after the system is fully developed, (2) the applications software which needs to be changed only to accommodate changes in vehicle configuration and changes in the GN&C algorithms, and (3) flight data loads which accept targeting constants between missions. Launch vehicle mission-to-mission software costs can be minimized if ways can be found to eliminate or minimize changes to the operating system software and applications software and to simplify the changes to mission data loads.

Current approaches to mission planning, analyses, and targeting contain a number of inefficiencies. Data tends to be duplicated and converted several times. There is also duplication in simulations and analyses, which are labor intensive areas, further contributing to high costs. Significant cost savings can be realized by establishing a unified approach which eliminates duplication of effort. Such an approach can be implemented with the use of data base systems and local area networks which can be combined to produce central data libraries and common analyses tools.

High labor costs associated with mission planning, analyses, and targeting can be significantly reduced through automation. The technology of knowledge based expert systems, a form of artificial intelligence, is maturing to the point that it should be considered for this application.

Checkout and Launch Systems

Checkout and launch concepts will be chosen to minimize labor and time for integration and checkout, maximize readiness to meet launch opportunities, and enable rapid launch from a standby mode with high probability of success. Onboard checkout systems, knowledge based systems, and fault tolerant systems are attractive concepts for launch vehicle checkout and launch systems.

Advances in microelectronics make the concept of automatic onboard checkout feasible for large launch vehicles. With this concept, the onboard checkout capability is designed into the distributed avionic processors, effectors, sensors, and other equipment which exist for flight operations functions. The onboard checkout capabiltity is integrated with the fault detection, identificaiton, and recovery (FDIR) system, and compartmented for autonomous operation at the major vehicle element level. Each vehicle element, such as an engine, will have its own capability essentially independent of the rest of the vehicle. The onboard checkout concept minimizes umbilical connections and eliminates the need for drag-on cables.

High levels of automation for the checkout system can be achieved with the evolving technology of knowledge based expert systems. One use of this system is to perform problem diagnosis, develop solutions, and generate corrective procedures in real time. Another use is to generate checkout and launch procedures from requirements and information fed into the data base. To fully realize the cost effectiveness of expert based systems, the approach depicted in Figure B-5 is utilized. This approach provides for high levels of continuity and commonality through all phases of verification, checkout, and launch.

The information base and level of "intelligence" in the knowledge-based system would continue to grow throughout the program, incorporating experience from simulations, verification tests, and previous launches. As confidence in the system matures, more and more control authority can be transferred from the human operator to the expert system.

Launch and checkout systems are provided with fault tolerant reliability to minimize interruptions in checkout and standby operations and to minimize launch holds and scrubs.

Distributed Systems Modularity

Modularity is an inherent property of the distributed architecture to be utilized for data processing and electrical power subsystems. This modularity has several highly desirable attributes.

Avionics modularity enables logical separability of functions. Dynamic interfaces can be contained within functional groupings resulting in simplified data traffic and power transfer between groupings, and providing high levels of autonomy within the functional groupings. For example, each engine, as a function grouping, contains its own processing and power conditioning equipment, to provide all the necessary engine sequencing logic, fault detection and redundancy management logic, and automated onboard checkout capability.

Avionics modularity is also a desirable feature for project management. Modular avionics can be matched to modular vehicle concepts, thus establishing clean interfaces for contractual separability. It also enables a modular buildup in simulation, development, integration, and verification.

Another important attribute of modularity is the ability to add to or change system capability so that the system can be easily adapted to new requirements and upgrades in technology.

Avionics Commonality

Commonality, if properly implemented, can result in large cost savings in both development and operations. To work effectively, however, commonality objectives must be firmly established early in the program, and management controls must be established to assure implementation throughout the program.

Implementation of commonality presents a major challenge, particularly in the area of data processing. Rapid changes in technology, wide diversity of standards and rapid obsolescence of hardware have severely limited the success of commonality in the past. This problem has been recognized and there is a trend both in industry and the military to settle on standard interfaces, bus protocols, and software languages. The DOD, for example, has designated Ada as their standard programming language and NASA is seriously considering Ada for the Space Station.

However difficult the challenge may be, every effort will be made early in the launch vehicle avionics definition to establish commonality goals at all levels. Attempts will be made to establish commonality with other contemporary projects such as the Space Station, the Orbital Transfer Vehicle, and the next generation Shuttle. Commonality will be implemented within the launch vehicle program among the different contractors, different vehicle elements, and different support systems for development and operations.

A key area of potential cost savings through commonality is the Software Development and Support Environment (SDSE), which includes methodologies, software tools, and systems for systems analyses and simulation, mission planning and targeting, and software development and verification. The Space Station SDSE is just now in the planning stage, and the DOD is currently engaged in developing SDSE systems. These systems will employ the latest technologies, methodologies, software languages, and standards. The advanced launch vehicle program will utilize as much as possible from these programs with some tailoring and augmentation as necessary to meet the unique program needs.

Security

The avionics incorporates features to comply with security regulations requiring special handling or procedures for the storage or transmission of classified data. The launch pad umbilical is a secure communications link and RF communications links are encrypted. Classified data stored in memory is segregated from unclassified data and special provisions are made for erasure of all classified data before splashdown/ landing. Security is further enhanced by the utilization of fiber optic networks for data transmission on the vehicle and at the launch site.

Safety

High degrees of fault tolerance in both onboard and ground systems provide significant protection against hazards due to system failures. Additional hardware and software safeguards and interlocks protect against inadvertent errors by personnel and procedures.

Avionics Subsystem Design Concepts

The overall HLLV avionics architecture is shown in Figure B-6. Much of the avionics is distributed and physically dispersed to achieve modularity and partitioning objectives. The locations of distributed avionic functions are highly dependent on the vehicle configuration and the location of the associated subsystem interfaces. There are a number of avionic functions, however, that are relatively independent of the vehicle configuration. These functions tend to be centralized in nature and include such functions as guidance and navigation; telemetry, tracking and command; central data processing and control; and the central energy source. Subsystems with these centralized functions are integrated into a "central avionics package" located in the P/A module. This central avionics package can be treated as a major vehicle module, completely integrated within itself and having "clean" data bus and power bus interfaces. It is the central intelligence and control authority for the vehicle. Being a module, it can be relocated for alternative vehicle configurations.

Avionics Distributed Processing

The HLLV avionics distributed processing architecture is depicted in Figure B-7(a). It includes the vehicle segment and the ground segment. A global fault tolerant network runs throughout the vehicle to interconnect physically dispersed fault tolerant GPCs and device interface units (DIUs). The GPC may in turn connect to a local area network through which it can control attached subsystems and devices. DIUs in most cases connect to a local area network of a particular GPC, but, in some cases, they connect directly to the global bus. All processors, DIUs, and subsystem electronics have built-in capabilities for fault detection, identification, and recovery (FDIR) and onboard checkout. DIUs and subsystem electronics may have some programmable processing capability but on a smaller scale than the GPCs. Figure B-7(b) shows typically how the vehicle data processing segment may be partitioned to fit the HLLV configuration.

The ground segment of the distributed processing system as shown in Figure B-7(a) depicts separate systems for flight operations control and checkout and launch control. It is possible that these systems may be combined to form a single system for control of all operations, including checkout, launch, flight and recovery operations.

Systems for checkout, launch, and flight control include a ground based expert system which interacts with both the human operator and with the systems in performing diagnostics. Initially, it will be inhibited from issuing commands to the system and will be limited to requesting information through the GPC network. As confidence in the system matures, however, some level of control authority may be delegated to the expert system.

The third element of the ground support is the Software Development and Support Environment (SDSE). It is an off-line system which includes the methodologies, software tools, and systems for systems analyses and simulation, mission planning and targeting, and software development and verification. The end product of this element is all the flight and ground software for the on-line elements of the distributed processing system.

Telemetry, Tracking and Command (TT&C) and Central Data Management Subsystems

Figure B-8 shows the HLLV TT&C and central data management subsystems. Tracking and communications with each of the vehicle elements are required through all phases of flight. Therefore, each major vehicle element that separates from the primary system during flight must provide some means for communications and tracking either directly from the ground, through a relay satellite, or from some recovery system vehicle. All communications in both directions are encrypted. Onboard data storage is provided to store engineering data for post flight systems evaluation, damage assessment, and trend analyses. This post flight analysis is essential to establishing the flight worthiness of vehicle systems hardware for subsequent missions.

A central processor is provided for overall vehicle integration and control of the various distributed processors; a central fault tolerant mass memory is shared by the distributed processors and the central processor; and fault tolerant central timing is provided for synchronization of the global network system. Synchronization between redundant elements of a fault tolerant processor or device is required, but the various functional processing sites may operate asynchronously from one another.

Guidance, Navigation and Control Subsystems

The HLLV Guidance and Navigation Subsystem is shown in Figure B-9. It is located entirely within the central avionics package in the P/A Module. A dedicated G&N processor processes data from several sensors to compute the necessary parameters for vehicle guidance and navigation through all phases of flight. Initial studies do not indicate a need for separate G&N capability on the booster elements. Definition of the complement of G&N sensors remains to be studied but will probably be determined by requirements for reentry and landing/splashdown accuracy.

The HLLV flight control subsystem is shown in Figure B-10. A dedicated flight control processor is located in the central avionics package in the P/A Module. This processor receives data from the G&N processor, data from flight control sensors distributed throughout the vehicle, and feedback data from thrust vector actuator systems. Control laws within the flight control processor process the data to compute control information which is transmitted to the thrust vector control systems in operation at the time. Data is transmitted via the global data processing network. Number, types and locations of flight control sensors are highly dependent on vehicle configuration. Final selection will be determined after complete vehicle configuration definition and analysis.

In the GN&C subsystems there are several areas where technology advances could provide a significant improvement in capabilities and cost. An optimized adaptive guidance system would optimally retarget the vehicle either during flight or just prior to launch. This could reduce significantly or essentially eliminate preflight simulation requirements which has beneficial implications both for cost and turnaround time. Adaptive guidance would also reduce launch constraints and allow broader launch windows.

An adaptive control system would determine optimal control parameters during flight, increasing mission success while reducing requirements on structures, propellant loading, etc. The controller would optimaly accommodate off nominal events such as engine out, engine performance, atmospheric perturbations, center-of-gravity errors and other sources of off nominal performance.

For the HLLV, which is made up of recoverable modules (the propulsion/avionics module and booster modules), an improvement in technology allowing very accurate targeting for autonomous return to the ground is desirable. Precision targeting would reduce ground operations by allowing the reusable module to land near the launch site, thus speeding up the vehicle integration for a subsequent flight. Accurate targeting would reduce chances of losing a valuable vehicle module.

As discussed elsewhere, the GN&C subsystem would also benefit from increased automation in checkout and from optimum use of distributed processing. The increased automation would be in the area of built-in test for inertial measurement units, attitude sensors, control electronics and the like. Use of distributed processing would be at least in the areas of navigation processing, control interfaces and attitude update and sensing.

Propulsion Control Subsystem

The Propulsion Control Subsystem is shown in Figure B-11. Overall authority, integration, and coordination of propulsion control is the responsibility of the central processor in the P/A Module central avionics. Most of the processing, sequencing logic, etc., is in the distributed processors. Each main engine has a dedicated fault tolerant processor for engine control. Dedicated processors are also established for the APU and Hydraulic Control Subsystem and for the P/A Module Propulsion Subsystem for coast and deorbit control.

Auxiliary Subsystem

There are a number of unrelated vehicle functions, which independently do not justify a separate dedicated processor. These functions are serviced by "Auxiliary Subsystems Processors" as shown in Figure B-12. The central processor in the central avionics package has overall control of these auxiliary functions. In addition, there is an Auxiliary Subsystems Processor in each booster, which assumes control of the booster after separation from the main vehicle.

Electrical Power Subsystem

Independent power sources are located in the P/A Module central avionics package and in each booster. The P/A Module power source consists of fuel cells and/or batteries. Batteries are used for the booster power source and to provide power for the deboost system on the core stage. The power distribution and control subsystem is shown in Figure B-13. The possibility of electromechanical thrust vector actuators in lieu of hydraulic actuators will be evaluated for impact on the electrical power system design.

Concepts for highly distributed power sources will be evaluated in the trade studies. These concepts will employ a family of high energy density, long-life batteries which can be distributed and optimized to local load requirements. The family will include both primary and rechargeable batteries and special purpose batteries for applications such as high power, very short duration loads. The distributed source concept minimizes weight and complexity of the power transmission system and enables partitioning to match vehicle modularity and maintenance requirements. Concepts will be chosen to minimize cost and time of maintenance and refurbishment between missions.

Distributed power system concepts will be evaluated to: (1) determine impact to avionics data interfaces, (2) maintain an acceptable EMI/IMC environment through use of a hybrid grounding scheme, and (3) determine the performance and characteristics of the energy storage devices. Information gained will be used to identify the technology status and needs as they relate to the electrical power subsystem.

For larger reusable vehicles such as the HLLV, that have greater amounts of built-in test, autonomous systems, and redundancy, the energy and power needs will increase significantly. The use of a distributed power system to meet these needs may result in a more simplified electrical power system with higher reliability and the ability to fully utilize emerging technologies such as lithium and sodium sulphur batteries.

Range Safety System

The Range Safety Subsystem is shown in Figure B-14. Each major vehicle element contains a completely independent system. Cross strapping is provided as an additional assurance of complete vehicle destruction if necessary. With the exception of the cross strapping feature, the range safety system shown is essentially the same as flown currently on the STS.

HLLV Avionics Studies

The foregoing description of desired attributes and characteristics for an HLLV avionics system has been formulated without full benefit of a comprehensive set of studies, and should be treated as a reference for comparison. There are many areas open for trades, analysis, and further definition, that must be resolved very early before committing large expenditures to an advanced avionics development. A number of these areas for further study are identified in Table B-3.

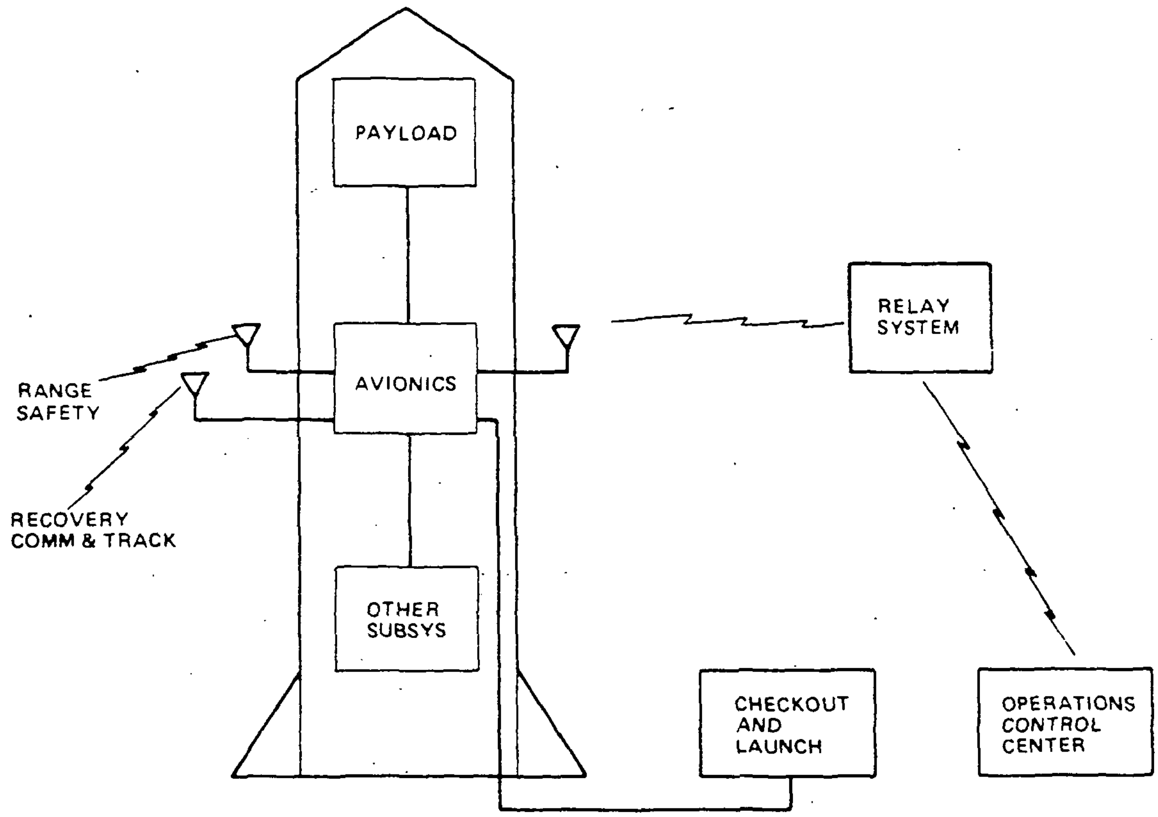


Figure B-1. HLLV avionics interfaces.

TABLE B-1. AVIONICS FUNCTIONAL REQUIREMENTS

GUIDANCE, NAVIGATION, CONTROL AND FLIGHT SEQUENCING	– PAYLOAD DELIVERY – RETURN/RECOVERY
SAFETY CONTROL	– RANGE SAFETY – CHECKOUT/LAUNCH OPERATIONS SAFETY – VEHICLE/PAYLOAD SAFETY
SECURITY	– INTERNAL/EXTERNAL COMMUNICATIONS – POST LANDING/SPLASHDOWN
VEHICLE/PAYLOAD SERVICES	– SYSTEM INTEGRATION – DATA PROCESSING – ELECTRICAL POWER
FAULT DETECTION/IDENTIFICATION, AND RECOVERY	– VEHICLE SUBSYSTEM FAULTS
FLIGHT INSTRUMENTATION	– REAL TIME FLIGHT STATUS – ENVIRONMENT/DAMAGE ASSESSMENT – TREND ANALYSIS – PROBLEM INVESTIGATION – OPERATIONAL READINESS VERIFICATION
ONBOARD CHECKOUT AND LAUNCH	– COMPONENT/SUBSYSTEMS VERIFICATION – SYSTEM INTEGRATION VERIFICATION – FLIGHT SIMULATION & COUNTDOWN DEMONSTRATION – COUNTDOWN AND LAUNCH

TABLE B-2. AVIONICS TECHNOLOGY NEEDS

TECHNOLOGY NEED	APPLICATION	RATIONALE
● KNOWLEDGE BASED EXPERT SYSTEMS	● HIGHLY AUTOMATED GROUND SUPPORT SYSTEMS FOR:	● REDUCES LABOR AND TIME BETWEEN MISSIONS
	– MISSION PLANNING AND TARGETING	● REDUCES REAL TIME SUPPORT PERSONNEL
	– SOFTWARE DEVELOPMENT AND VERIFICATION	● ENABLES RAPID PROBLEM DIAGNOSIS AND DECISION MAKING
	– VEHICLE CHECKOUT AND LAUNCH	
	– MISSION OPERATIONS	
● FAULT TOLERANT DISTRIBUTED PROCESSING	● VERY RELIABLE MODULAR PROCESSING SYSTEM FOR:	● ENABLES PARTITIONING OF PROCESSING TO FIT VEHICLE MODULARITY
	– VEHICLE AVIONICS	
	– CHECKOUT AND LAUNCH SUPPORT	● IMPROVES SAFETY AND MISSION RELIABILITY
● AUTONOMOUS CHECKOUT	● HIGH DEGREE OF ON–BOARD CHECKOUT AND SELF–TEST	● ENABLES RAPID CHECKOUT AND LAUNCH
		● REDUCES GROUND EQUIPMENT AND CABLING CONNECTIONS
● IMPROVED INSTRUMENTATION	● DETERMINATION OF CONDITION AND PERFORMANCE OF ENGINES, ETC.	● IMPROVES ASSESSMENT OF WEAR, DAMAGE, AND FLIGHT WORTHINESS OF REUSABLE HARDWARE
● ADAPTIVE GUIDANCE	● OPTIMAL RETARGETING DURING FLIGHT OR JUST PRIOR TO LAUNCH	● REDUCES/ELIMINATES PRE–FLIGHT SIMULATION
		● ALLOWS MISSION RETARGETING AT ANY TIME
● ADAPTIVE CONTROL	● DETERMINATION OF OPTIMAL CONTROL PARAMETERS DURING FLIGHT	● ACCOMMODATES OFF–NOMINAL EVENTS SUCH AS ENGINE OUT, ENGINE PERFORMANCE ATMOSPHERIC PERTURBATIONS, C. G. ERRORS, ETC.
● PRECISION REENTRY GN&C	● PRECISION TARGETING FOR RETURN OF RECOVERABLE MODULES	● REDUCES GROUND OPERATIONS
		● ENHANCES REUSABILITY
● HIGH POWER ELECTRO–MECHANICAL ACTUATORS	● MAIN ENGINE THRUST VECTOR CONTROL	● ELIMINATES NEE FOR APU AND HYDRAULICS
● IMPROVED SENSORS	● ATTITUDE, ATTITUDE RATES, ACCELERATION, ETC.	● IMPROVED LONG TERM STABILITY AND RELIAB. TO ACCOMMODATE LAUNCH ON DEMAND WITH MINIMAL RECALIBRATION AND CHECKOUT
● LONG–LIFE, HIGH–POWER LOW MAINTENANCE, PROPELLANT GRADE FUEL CELL	● CENTRAL POWER SOURCE FOR SEVERAL HOURS MISSION DURATION	● UTILIZES PROPELLANT BOILOFF
		● MINIMIZES MAINTENANCE AND REFURBISHMENT BETWEEN MISSIONS
● FAMILY OF HIGH ENERGY DENSITY, LONG–LIFE BATTERIES	● DISTRIBUTED SOURCES OPTIMIZED & DEDICATED TO LOCAL LOAD REQUIREMENTS	● MINIMIZES WEIGHT AND COMPLEXITY OF POWER TRANSMISSION SYSTEM
	– PRIMARY BATTERIES	● ENABLES PARTITIONING TO FIT VEHICLE OF POWER TRANSMISSION
	– RECHARGEABLE BATTERIES	
	– BATTERIES FOR HIGH POWER, VERY SHORT DURATION LOADS	● MINIMIZES MAINTENANCE AND REFURBISHMENT BETWEEN MISSIONS.

TABLE B-3. AVIONICS STUDIES

- TECHNOLOGY UTILIZATION AND ADVANCED DEVELOPMENT PLANNING
- FAULT TOLERANCE DEGREE AND METHODOLOGY
  - HARDWARE
  - SOFTWARE
- INNOVATIVE CONCEPTS FOR MINIMIZING TURN–AROUND COSTS AND TIME
  - HARDWARE
  - SOFTWARE
  - OPERATIONS
- DEGREE OF REUSABILITY
- DEGREE OF ONBOARD CHECKOUT AUTOMATION
- UTILIZATION OF KNOWLEDGE BASED EXPERT SYSTEMS
  - FOR CHECKOUT AND LAUNCH SUPPORT
  - FOR MISSION PLANNING AND TARGETING
  - FOR SOFTWARE DEVELOPMENT
- PARTS/COMPONENTS RELIABILITY APPROACH
- DEFINITION OF SOFTWARE DEVELOPMENT AND SUPPORT ENVIRONMENT
- DEGREE OF COMMONALITY AND STANDARDIZATION
  - FLIGHT HARDWARE AND SOFTWARE
  - DEVELOPMENT AND OPERATIONAL SUPPORT SYSTEMS
- SYSTEMS/SUBSYSTEMS DESIGN TRADES AND ANALYSES
  - DATA PROCESSING SUBSYSTEM ARCHITECTURE
  - SOFTWARE SIZING ANALYSIS
  - ELECTRICAL POWER SOURCES & DISTRIBUTION CONCEPTS
  - ELECTRICAL VS. HYDRAULIC THRUST VECTOR ACTUATORS
  - G&N SENSOR COMPLEMENT
  - GUIDANCE METHODOLOGY
  - GN&C PERFORMANCE/ACCURACY ANALYSIS
  - CONTROL LAWS, AND SENSORS

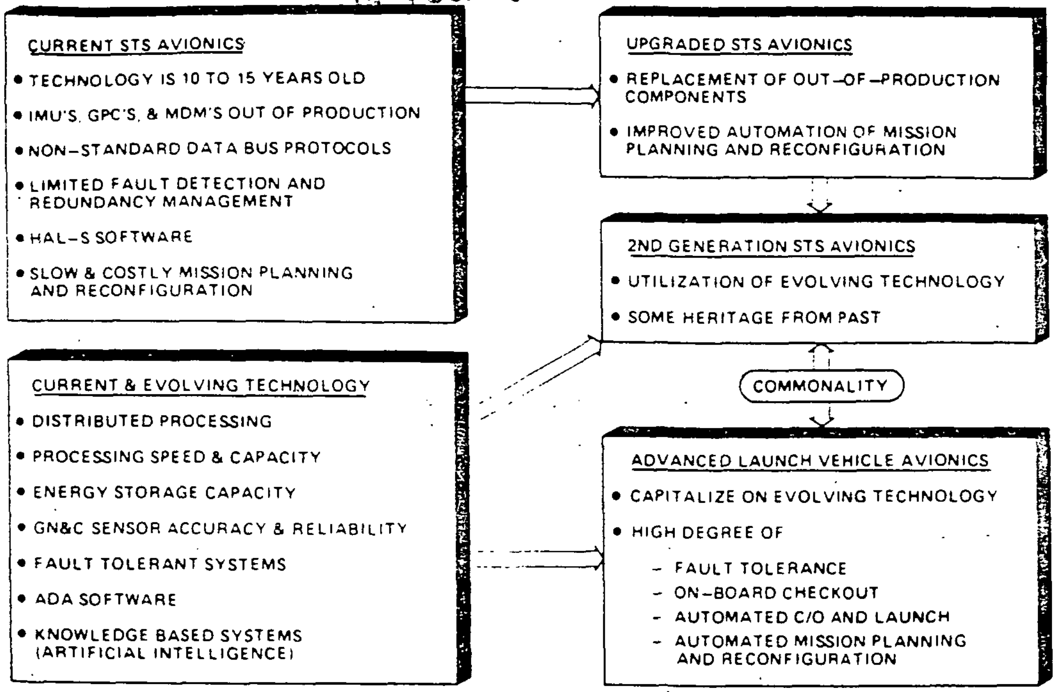


Figure B-2. Avionics/software technology.

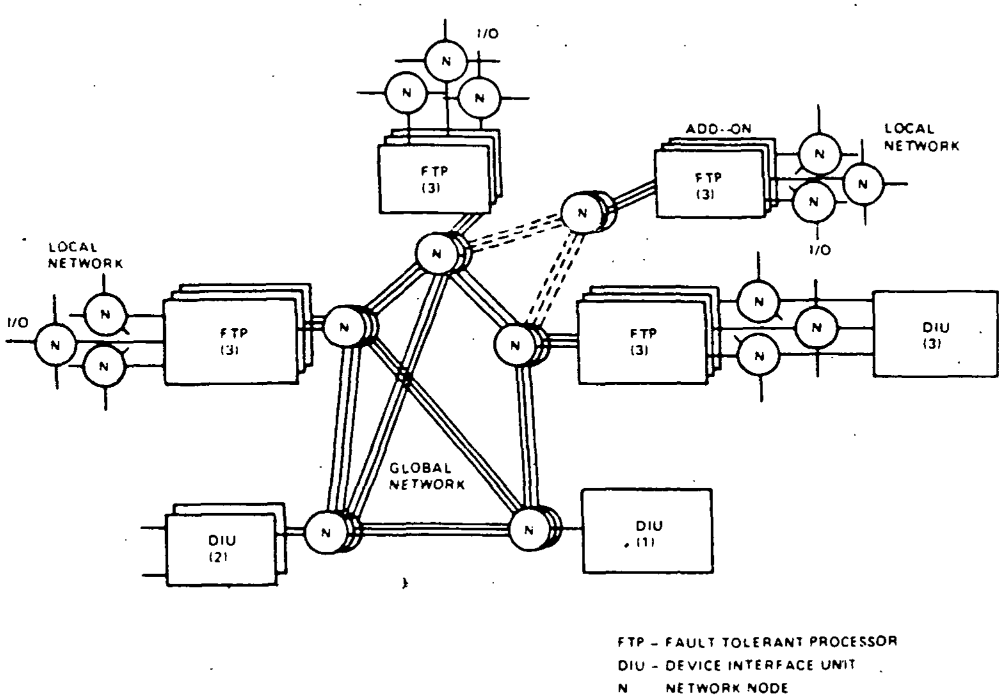


Figure B-3. Fault tolerant concept.

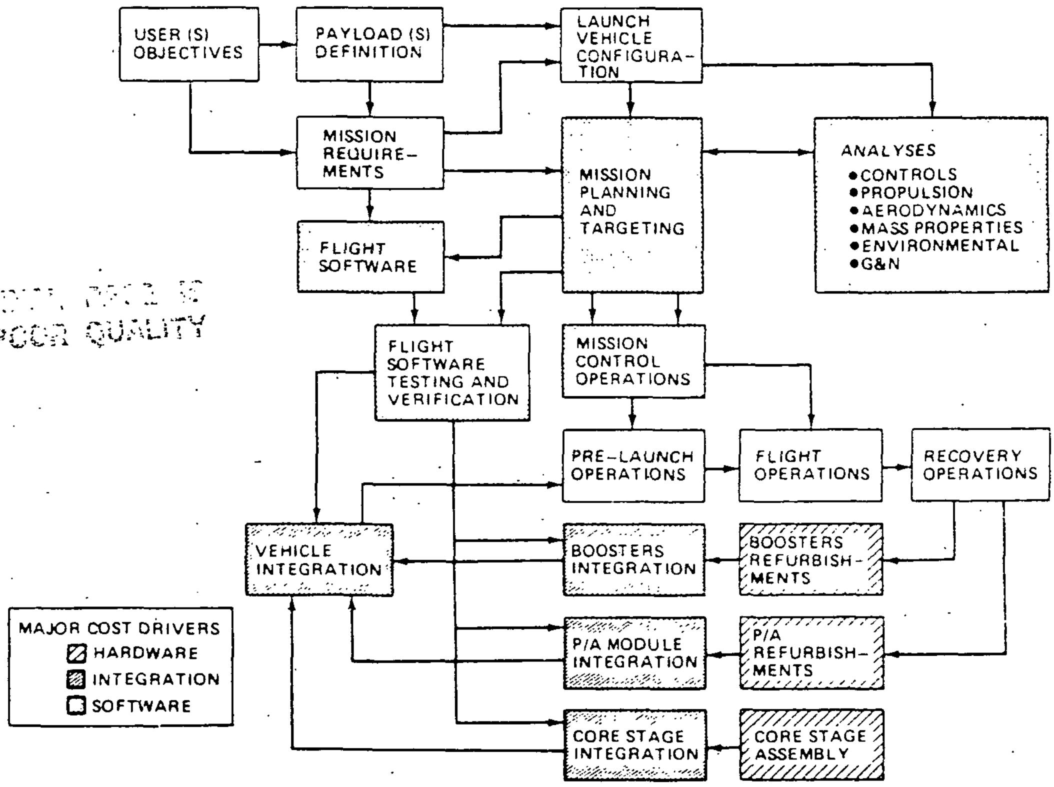


Figure B-4. Avionics/software turnaround flow.

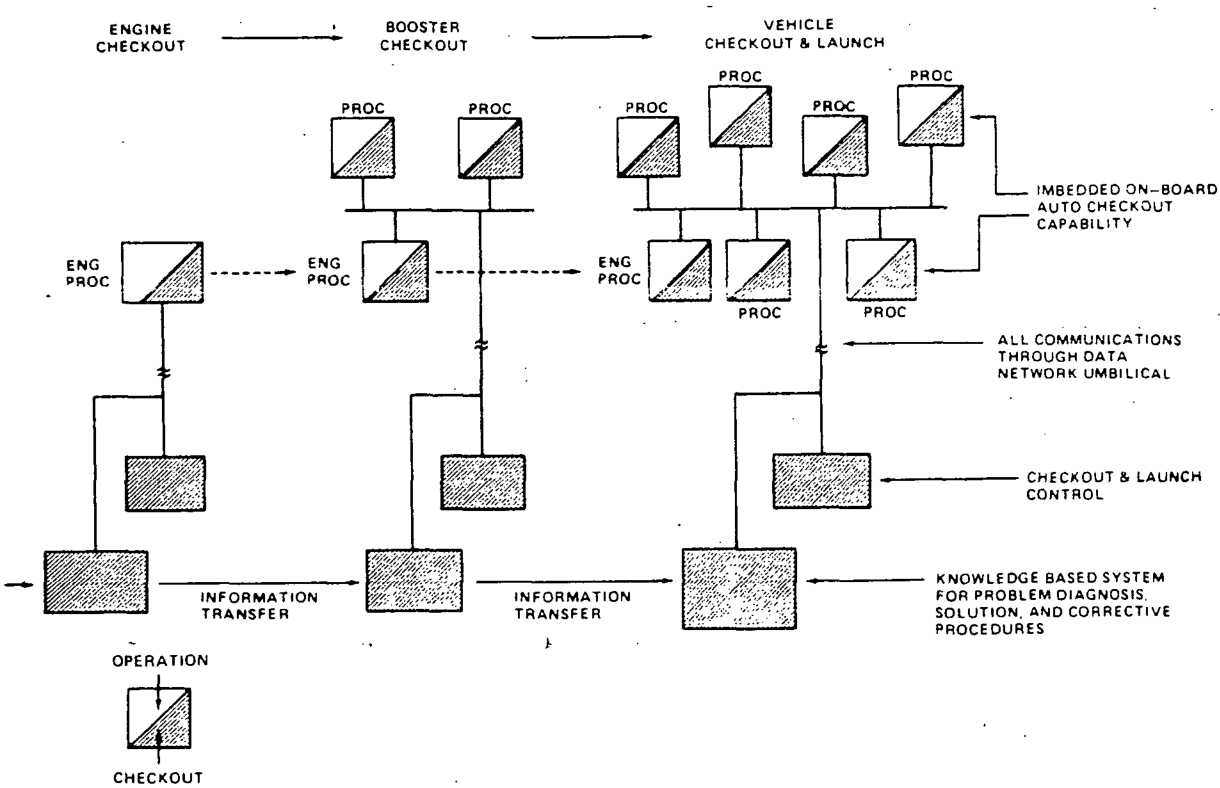


Figure B-5. Checkout and launch concept.

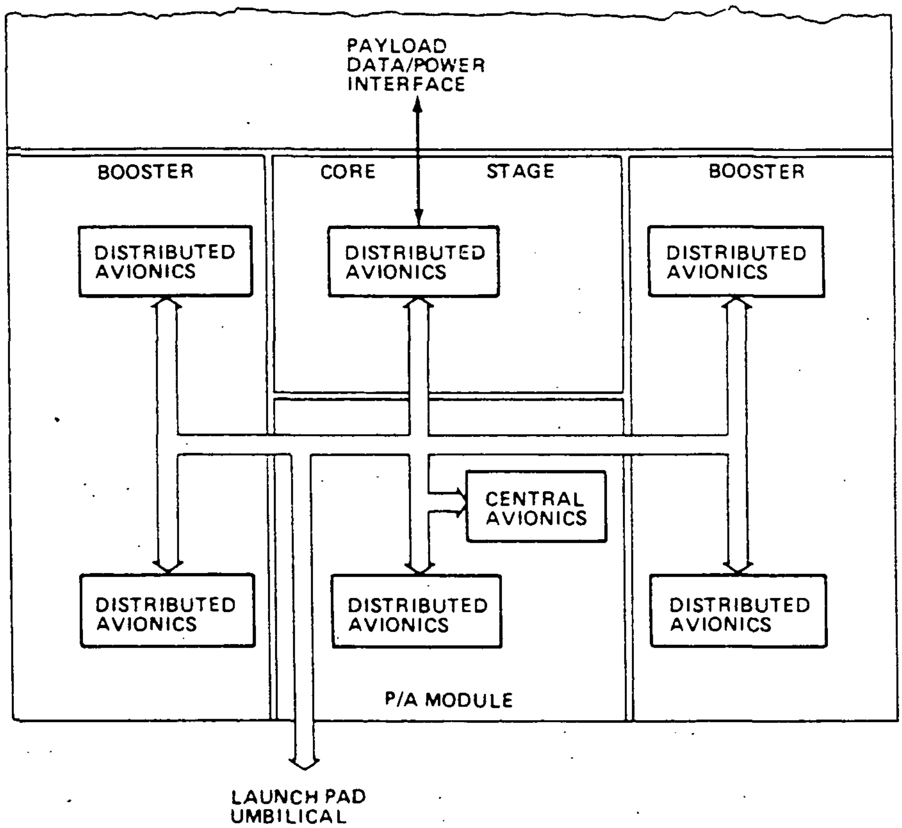


Figure B-6. Avionics system architecture.

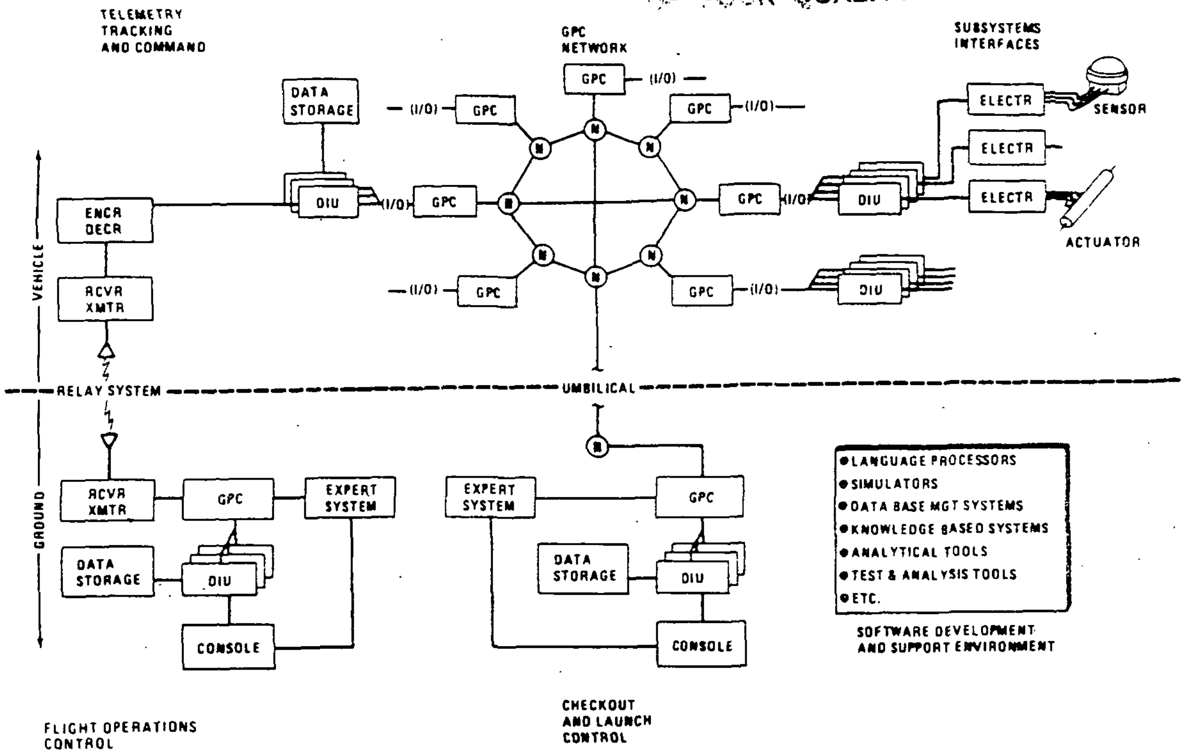


Figure B-7(a). Distributed processing system.

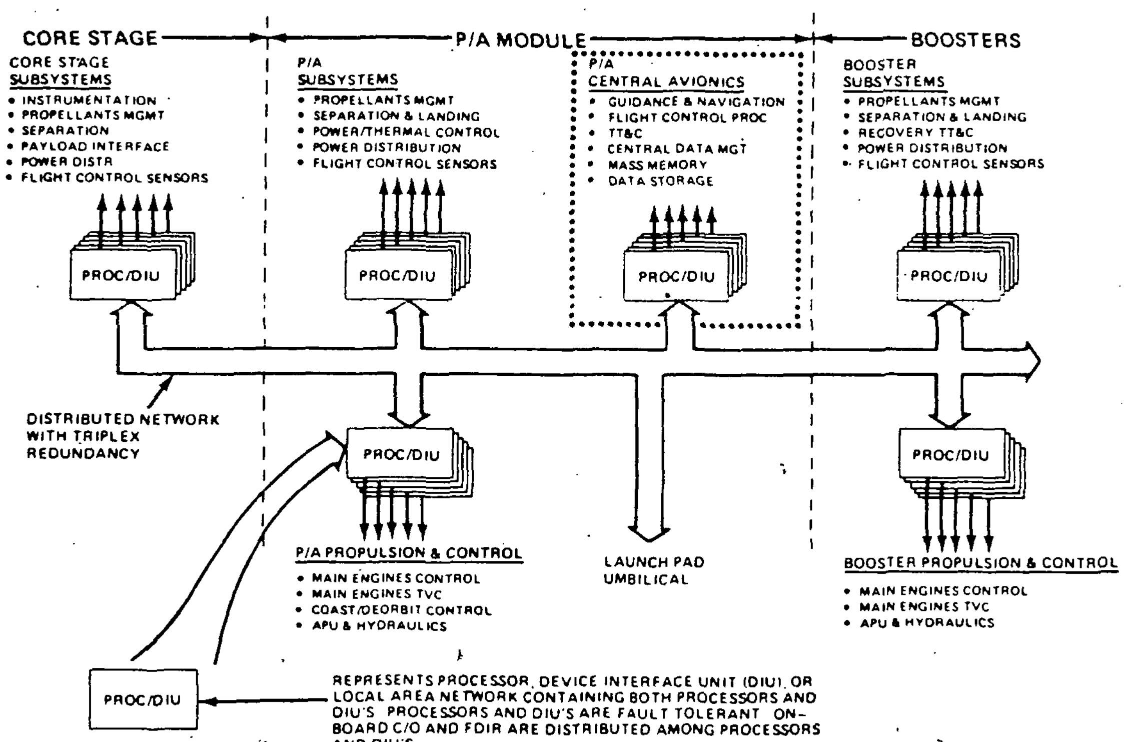


Figure B-7(b). Avionics distributed processing.

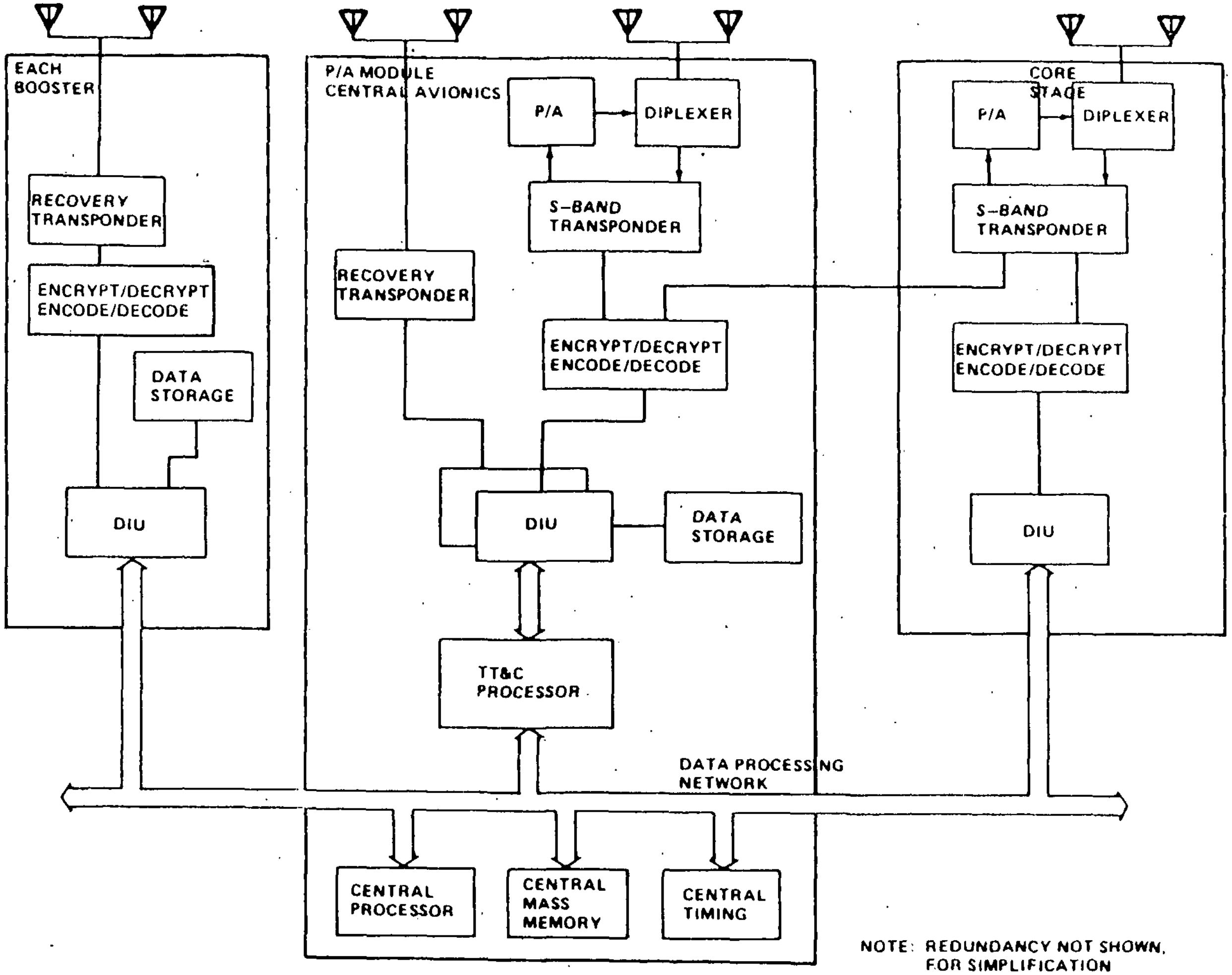


Figure B-8. Tracking, telemetry and command and data management subsystems.

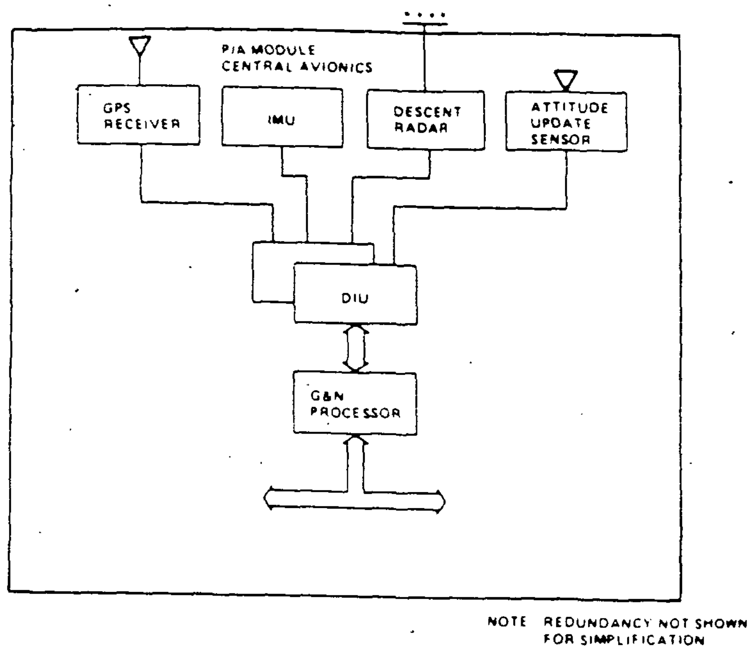


Figure B-9. Guidance and navigation system.

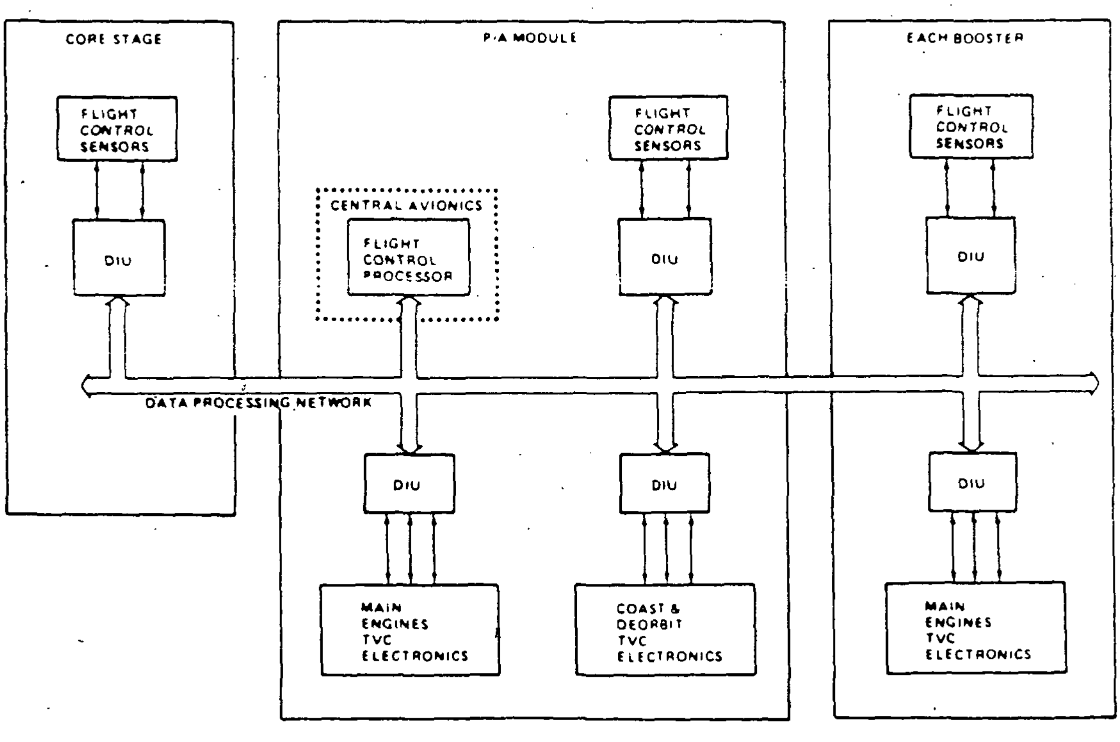


Figure B-10. Flight control subsystem.

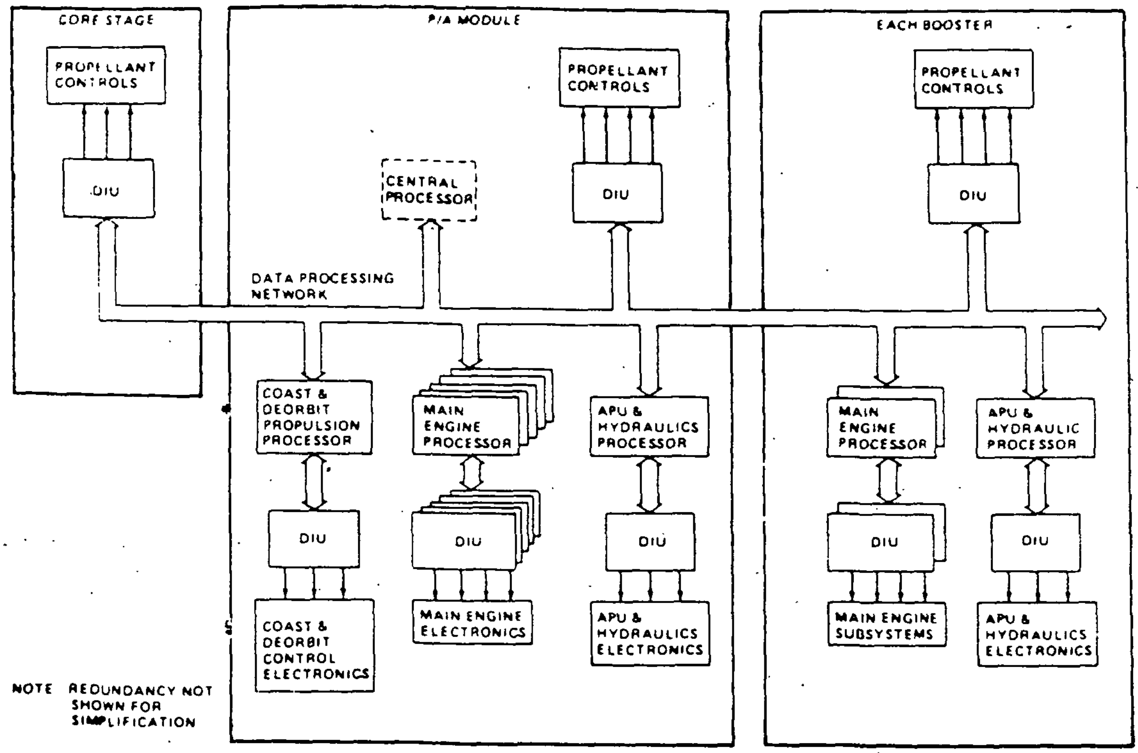


Figure B-11. Propulsion control subsystem.

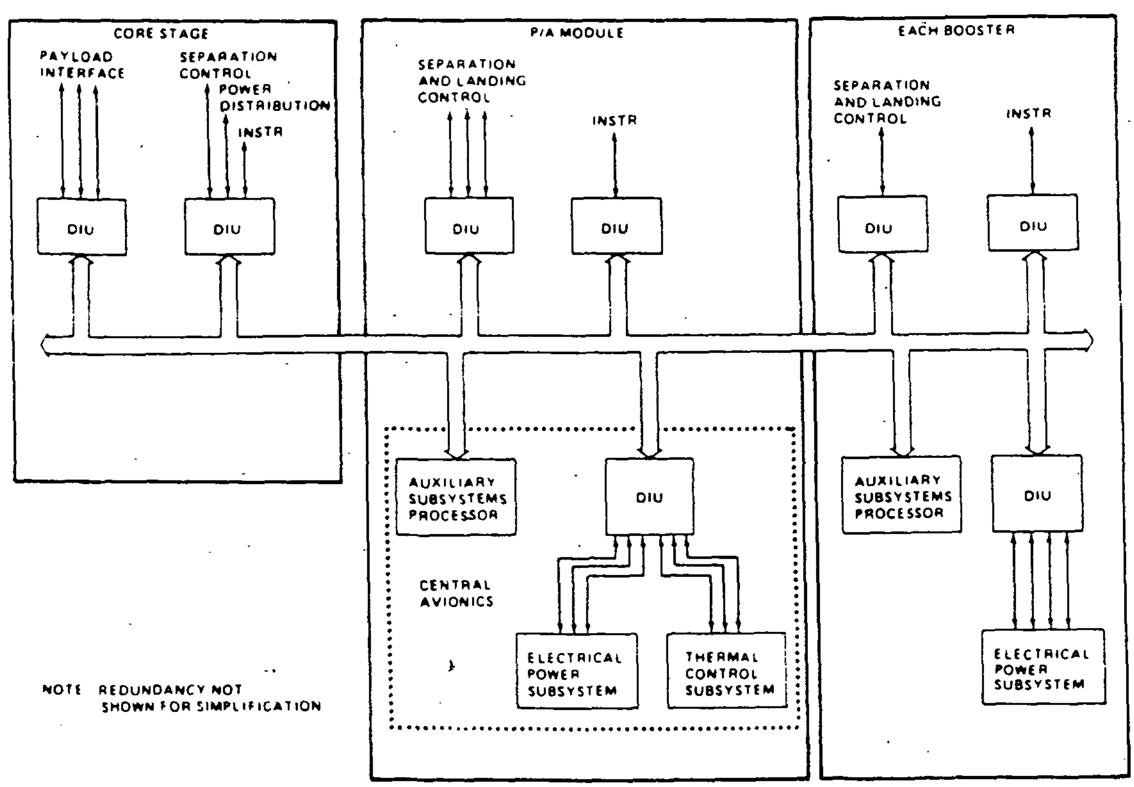
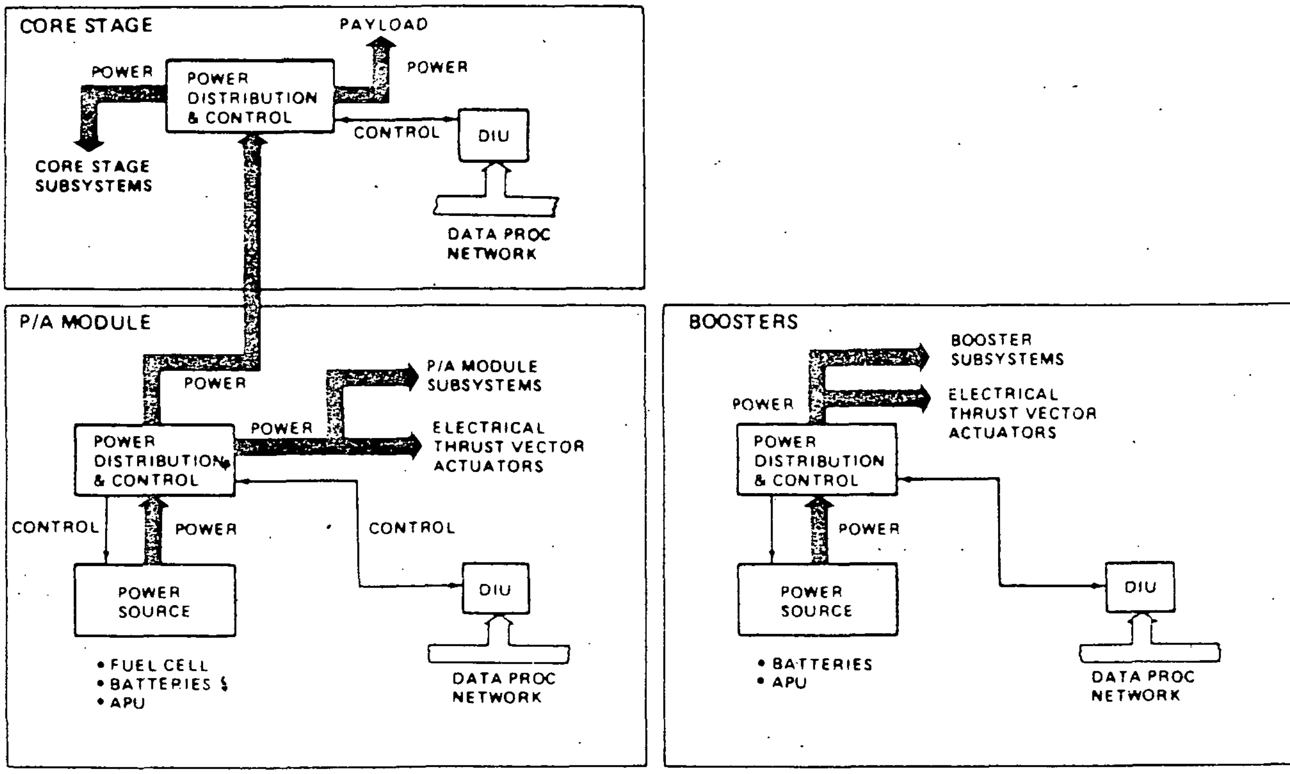
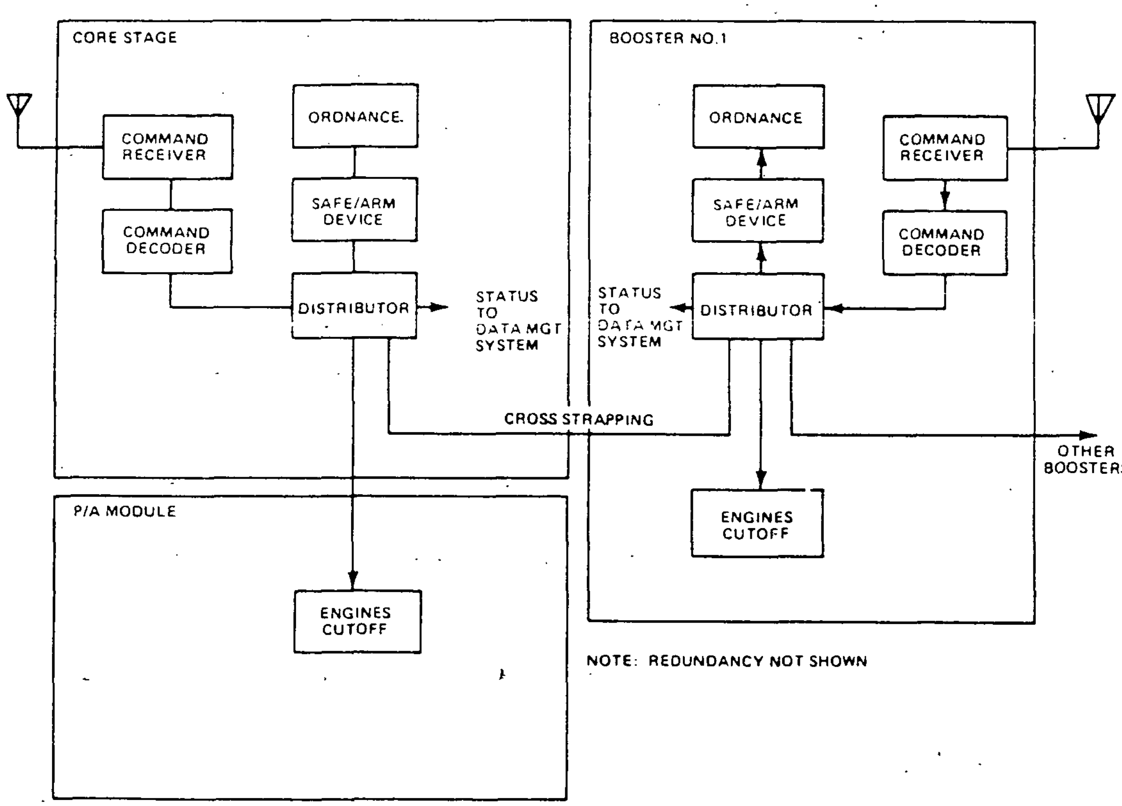


Figure B-12. Auxiliary subsystem.



NOTE REDUNDANCY NOT SHOWN FOR SIMPLIFICATION

Figure B-13. Electrical power subsystem.



NOTE: REDUNDANCY NOT SHOWN

Figure B-14. Range safety subsystem.

APPENDIX C. DEVELOPMENT SCHEDULES

Development schedules are estimated from the time of Headquarters "go-ahead." That is the date on which the NASA Center(s) is given the authority to release procurement documents such as an announcement of intent to contract to the "Commerce Business Daily," and later to release the Request for Proposal (RFP). This date is T = 0 on the following charts, and the numbers are cumulative years from T = 0.

Liquid rocket engine schedules are shown first. The LOX/LH_2 core stage engine can utilize any of three engines:

1) Existing SSME; this is off-the-shelf and has a three year procurement time. This is not shown on the chart.

2) The STME 481 represents a modified SSME, and its development time is relatively short. This would be an ideal way to configure the HLLV if a completely new Advanced Cryo Engine cannot be factored in early.

3) The Advanced Cryo Engine (ACE) represents the ultimate configuration goal. Much development is required, and thus the schedule is the longest of the three cryo engine possibilities.

Multiple studies are shown for each engine. These studies would terminate just prior to release of the Phase C/D RFP's. A schedule gap is then shown to accommodate the procurement cycle. The next milestones show deliveries of ground test engines — preliminary configurations good enough for static firings. The first of these would be used by the engine contractor, and later ones to be used by the stage contractor on the PTA (propulsion test article).

The engine contractor would begin flight certification tests as soon as a reasonably mature configuration is available. A preliminary certification could conceivably be granted after approximately 750 hot firings and burp-starts. Before declaring PFC (preliminary flight certification), some engines could be delivered to the stage contractor(s) to install on early flight vehicles. Historical precedents support this approach because:

1) Early flights are generally verification tests.

2) Some significant payloads can be flown on these early flights depending upon a risk assessment. No losses were incurred to boosters or payloads by using this scheme on the Saturn and Shuttle programs.

3) Engine performance characteristics are well defined by this point in time. Mean time between failure (MTBF) data is incomplete, but these will be new engines.

4) This allows the earliest possible schedule for first launch of the new booster.

Engines that are classified as FFC (final flight certified) could be delivered later upon completion of a long test program. These engines would be certified to deliver full specification life expectancy and MTBF.

The longest lead time problem on the HLLV Program is brought out on the Engine Development Schedules chart. The problem is the Hydrocarbon Engine (STBE) schedule for delivery of PFC engines to the stage contractor to be used for the first flight vehicle. This constitutes the critical path for the early years (7-8) of the HLLV program.

The stage contractors' schedules would be paced by the anticipated delivery dates of the first set of flight engines, to some extent by the deliveries of the PTA engines, and by the capacity to manufacture many engines the first year.

It should be emphasized on the schedules that a considerable time and monetary investment should be expended on engine studies and advanced development prior to T = 0. The schedules shown subsequent to T = 0 make this assumption. If sufficient time and resources are not forthcoming, then the Phase C/D schedules must be stretched accordingly.

In conclusion, if the HLLV program is adequately funded and staffed, it is reasonable to estimate that the first launch of a full-up vehicle could be accomplished in approximately eight years from Headquarters go-ahead. From contract awards, the time is shorter.

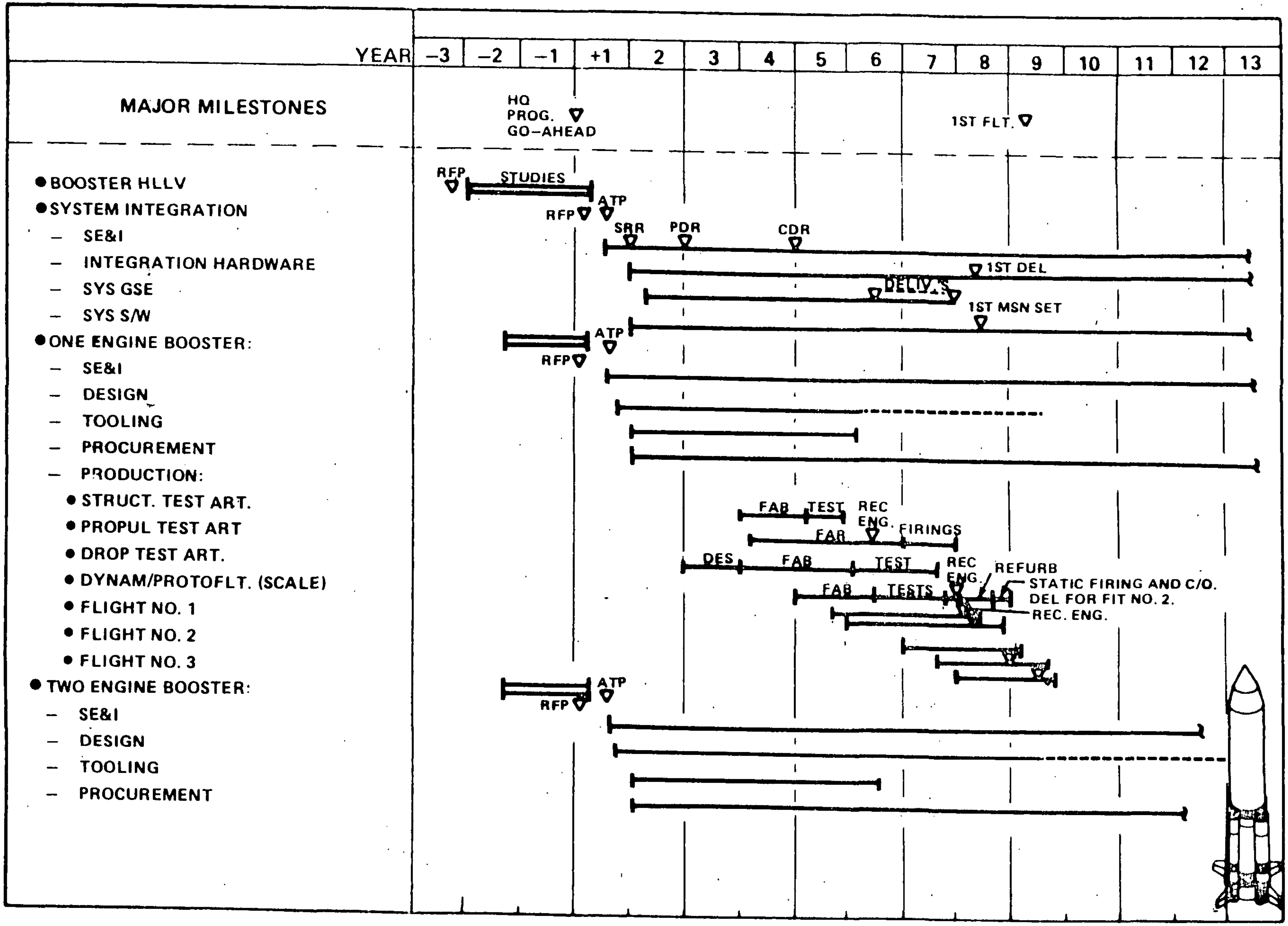


Figure C-1. Engine development schedules.

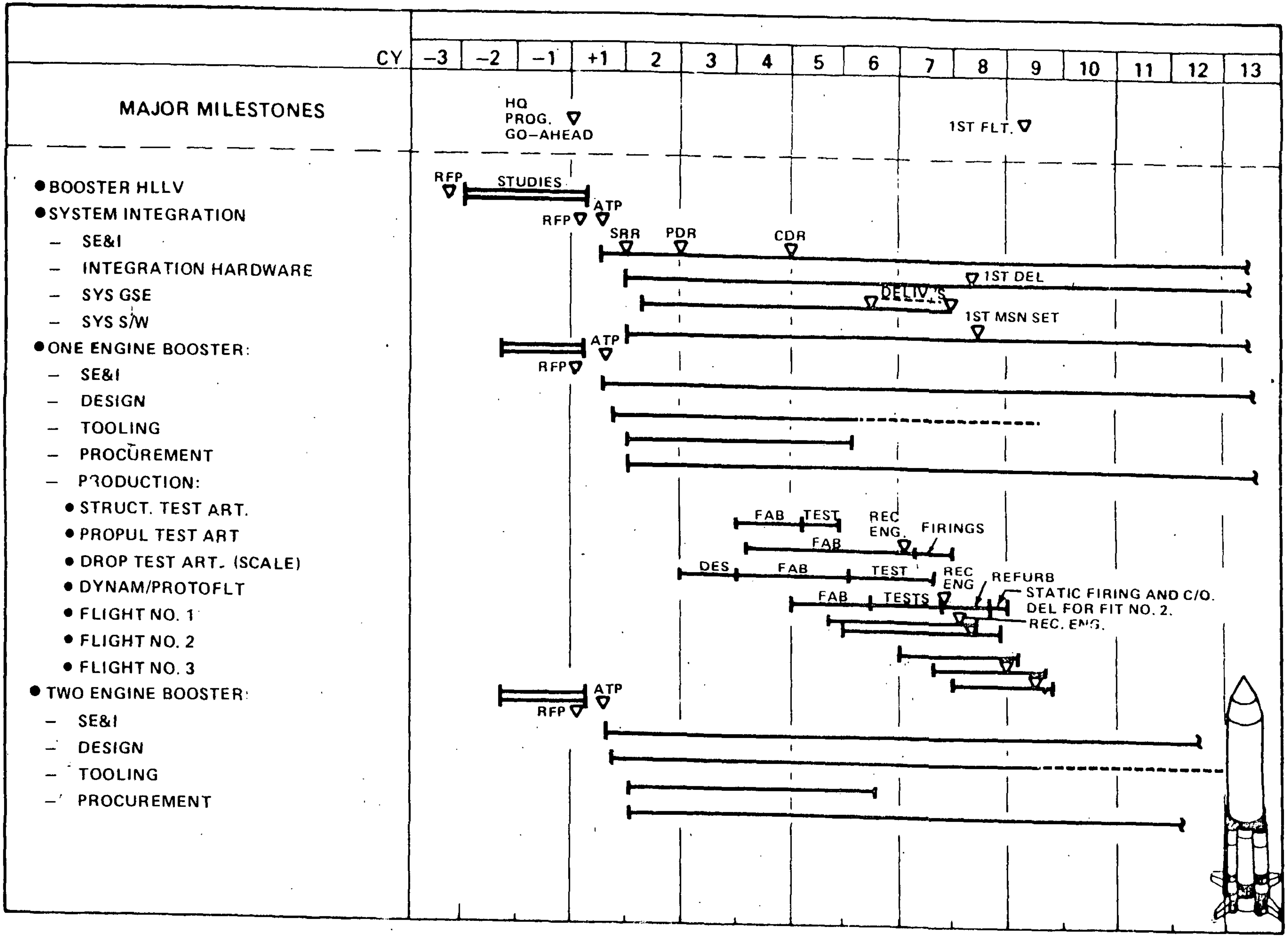


Figure C-2. Direct approach development schedule.

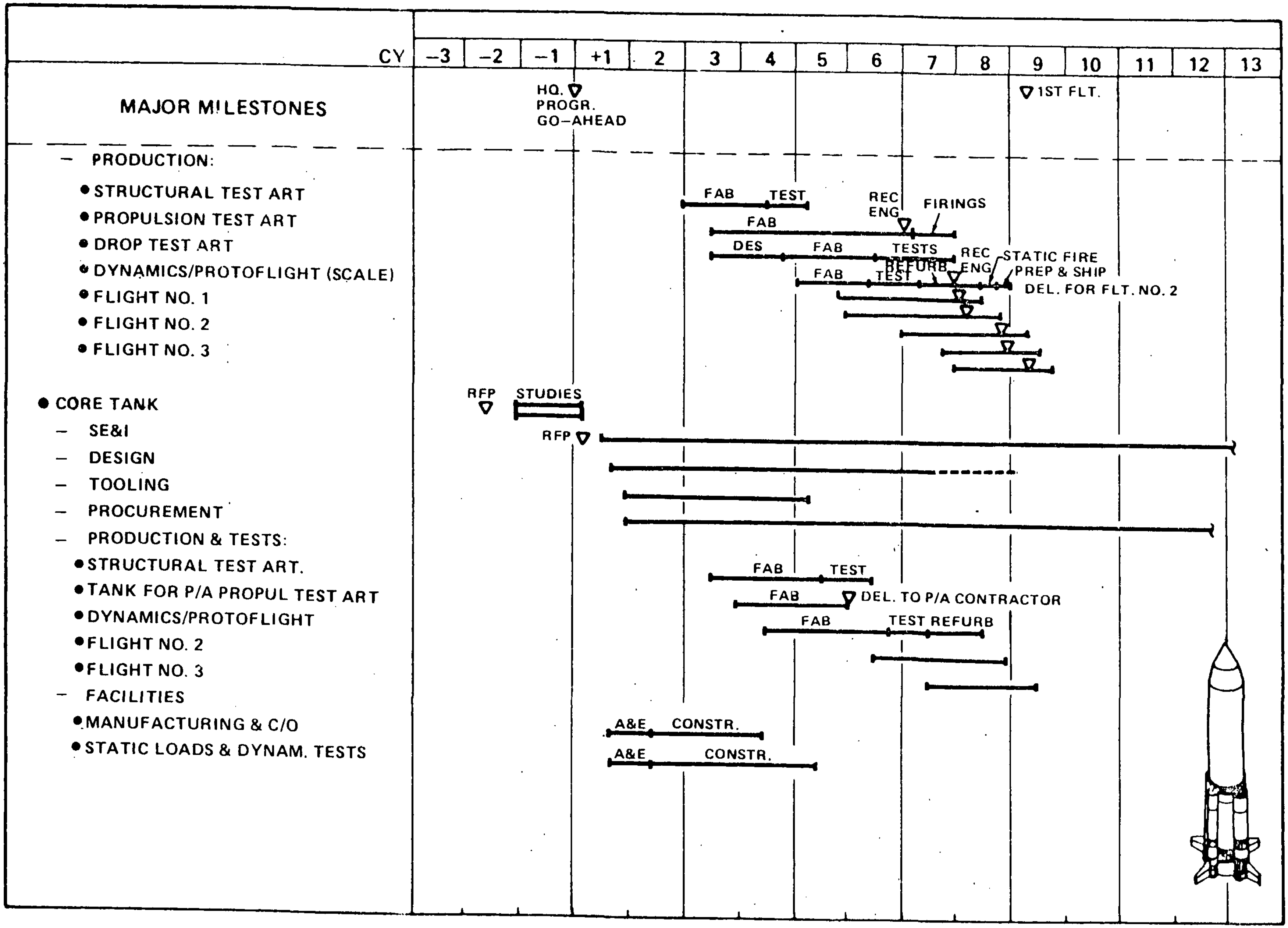


Figure C-3. Direct approach development schedule.

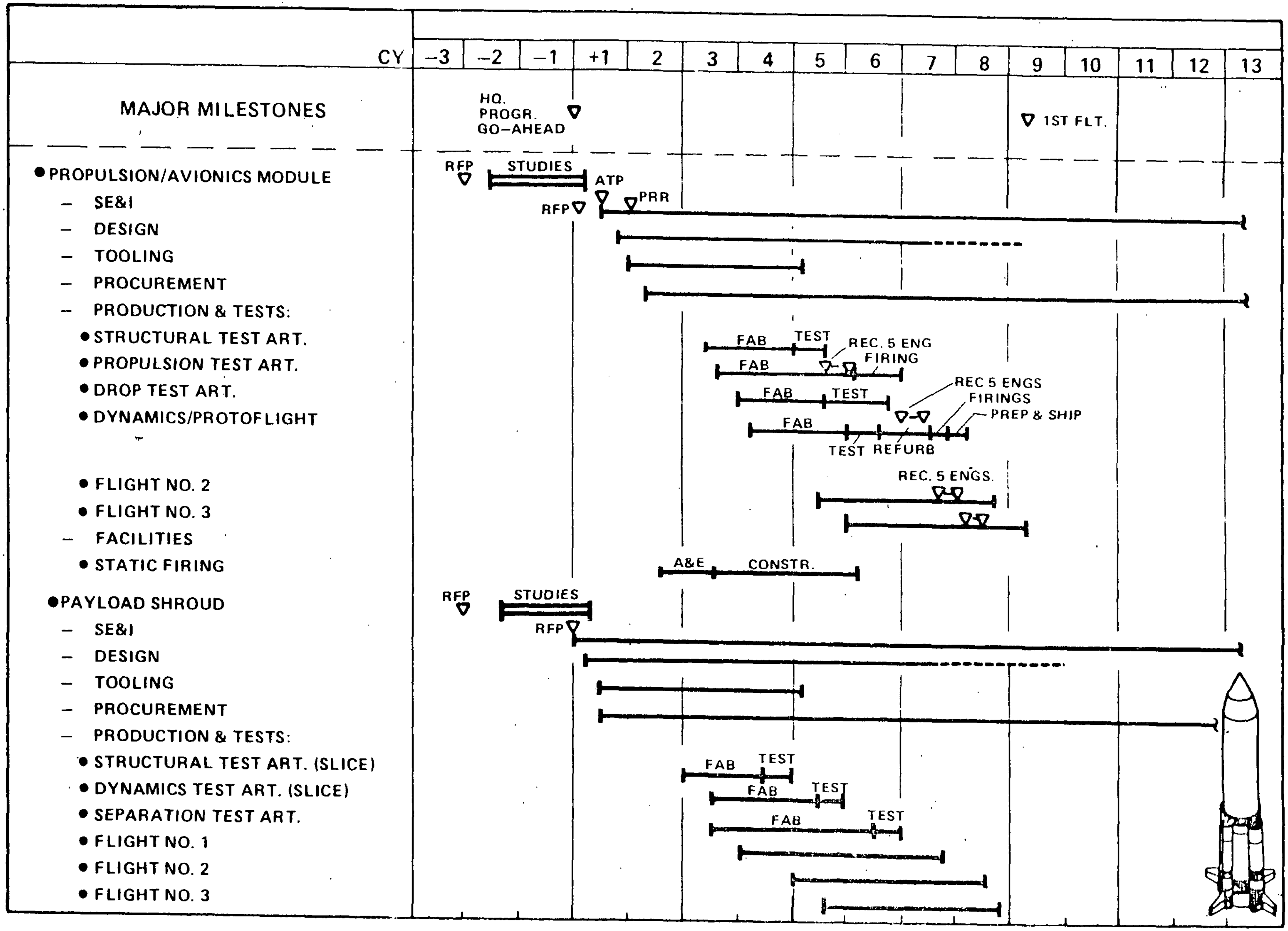


Figure C-4. Direct approach development schedule.

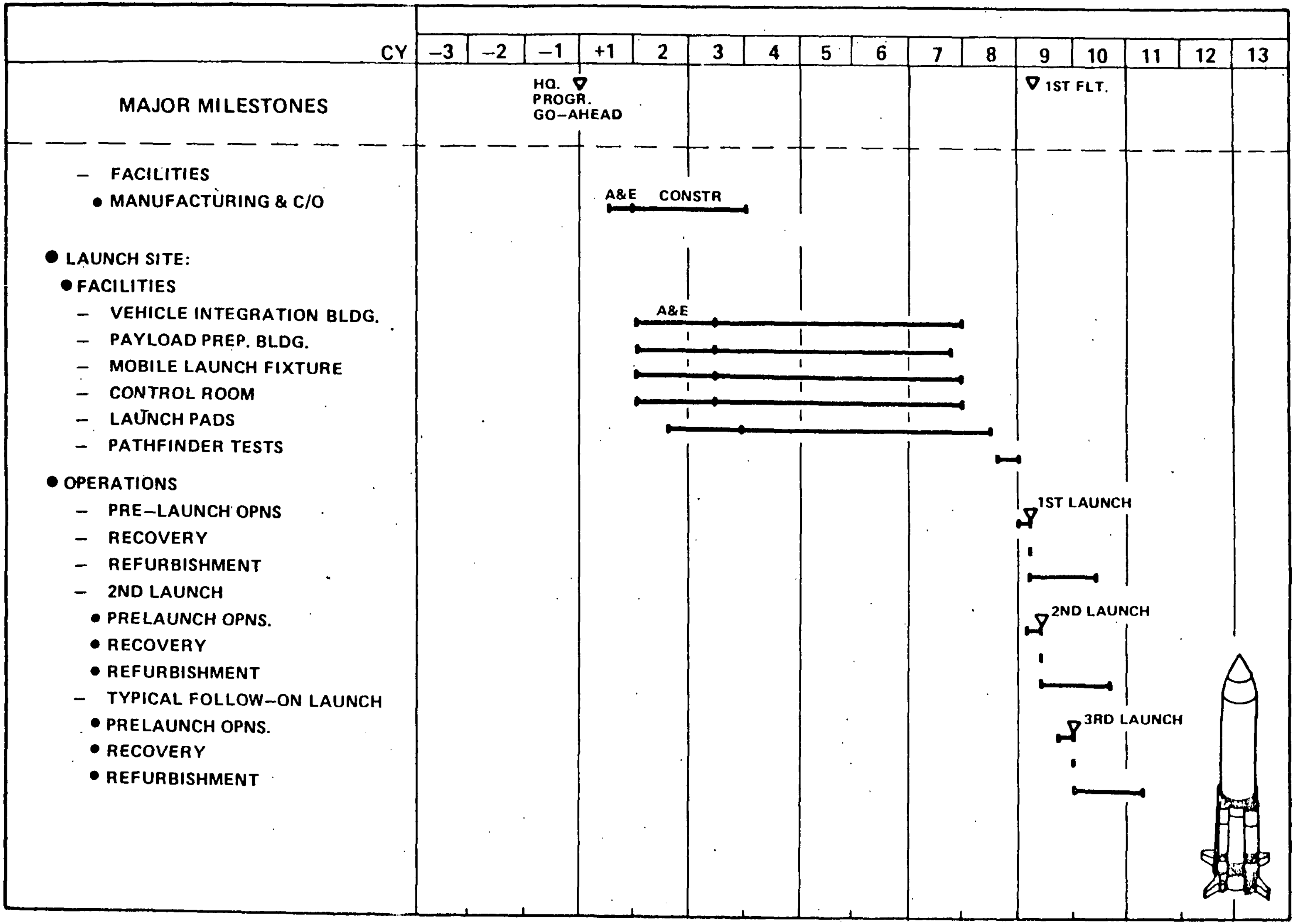


Figure C-5. Direct approach development schedule.

APPENDIX D. LAUNCH FACILITIES AND GROUND OPERATIONS

A new launch site is recommended because of the large overall size of the vehicle, the need for rapid buildup and payload changeout, potential interference with STS flight, and the required extensive modification to present facilities. The new facilities will provide secure processing and have no direct impact to the STS program or other schedules.

The baseline launch scenario is polar orbit (due south launch) which, because of the impact of expended components at staging, places certain constraints on launch site geographical location. Possibilities include the Vandenberg Air Force Base area in Southern California (site of WTR STS launch complex), Southern Alaska, Hawaii, and certain areas of New England. Although ideal in location and climate, Hawaii has logistical problems with transportation of vehicle components (both new and refurbished), propellants, and facility construction. Alaska has these problems as well as a comparatively disagreeable climate, and is close to the Soviet Union. New England is an attractive site, but has a rather harsh climate as well as a generally widespread dense population, and booster impact would be in heavily travelled shipping lanes. Continental United States sites typically will not support all azimuth launches due to present over-flight restrictions, so low inclination launches will require a separate launch site.

The Southern California area northwest of Los Angeles has some distinct advantages. Location of the site on VAFB provides the necessary security, has an acceptable climate, logistics for construction and operations is viable, and the mountainous terrain can be used advantageously to afford visual security and an acceptable degree of blast protection in case of launch pad accident while keeping the site reasonably compact.

The operation concept proposed here is generally similar to KSC in that the vehicle is built up in an assembly building and transported to the launch pad via a mobile launcher, as opposed to vehicle buildup on the pad. This concept provides more efficient use of the launch pads, allows parallel vehicle processing, isolates the launch pads from the buildup area, and readily permits launch vehicle changeout to a higher priority mission.

The functional facilities concept is shown in Figure D-1, and is applicable to any selected location. Facility size and quantity are determined by the launch rate. An advantage of this concept is that all payload, aerodynamic payload fairing, stage, P/A Module operations, vehicle buildup, and payload integration can be done under one roof. Dangerous or toxic propellant facilities should be isolated in separate buildings for safety. This concept provides increased security since the payloads are never exposed and vehicle operations are not externally obvious until rollout. Specialized transport canisters are minimized since there is no exposure to the weather at any time after payload delivery.

The launch pad area should include at least two pads for parallel launch capability with a single launch control center. Onboard prelaunch operations performed by the vehicle will reduce the size and complexity of the Launch Control Center. The launch pads can be judiciously located to take advantage of the mountainous terrain for blast protection. Otherwise, the pads would have to be 6 to 7 miles apart. Permanent cranes may be located at each pad for rapid payload changeout capability or the mobile launcher can be returned to the assembly building. The onboard checkout capability of the launch vehicle reduces the requirement of servicing electrical connections to the LCC. The vehicle can remain in a launch-ready condition except for propellant loads and can be removed for payload changeout. A detailed facility trade is required prior to hardware decision. Propellant (LOX, LH_2, and RP) storage facilities are needed at each pad complex, and acoustic suppression and pad protection water can be located at high elevation for better gravity feed.

Mobile Launcher Platform (MLP) design will incorporate the capability for maintaining vehicle readiness while enroute to the launch pad. This concept allows more efficient pad operations resulting in rapid launch, and lowers visible exposure time. MLP design will include the provision for other vehicles derived from the HLLV components, and therefore must have the proper hold down structures and flame port spacing for the vehicle family. Examples include versions with four large boosters, and a vehicle using only one of the first-stage boosters.

The overall operations sequence is similar to the STS processing flow at KSC, and some of the timelines (such as rollout, pad refurbishment, and MLP refurbishment) are derived from the STS processing assessment, STAR-027. A typical vehicle assembly/checkout timeline is shown in Figure D-2, and represents a total time of 720 hours for the pad, 476 for the assembly building, and 1076 hours for the P/A Module.

The booster stacking sequence, shown in Figure D-3, appears to be the most operationally feasible approach. The strap-on boosters are canted toward the core stage at the top because of the conical geometry of the P/A Module. Therefore, the core stage cannot be inserted from the top after booster erection without a major cantilever system tilting the boosters from the base attachment. The concept presented erects the core stage with the P/A Module attached, supported by a portable jack system attached to the MLP holddowns. The two-engine boosters are brought in from the sides with overhead cranes and attached to the MLP holddowns. The single engine boosters are erected in a like manner. Temporary GSE is required to stabilize the top of the boosters until the payload adapter is installed. The payload adapter is then lowered from the top and attached to the core stage, and the core stage is lowered and attached to the forward end of the four boosters, forming the major upper structural tie point. The aft struts are attached, completing the mechanical assembly. The portable jacks are removed and the vehicle is supported by the MLP holddowns. The propellant crossfeed plumbing is then installed. After all electrical and other connections are made, the basic vehicle stacking sequence is complete and ready for payload installation.

For a vehicle of this size and with such large volume and weight payloads, transportation of vehicle components and payloads to the launch site is an area of concern. These problems can be minimized if payload and vehicle component manufacturing or final assembly facilities are located in the immediate vicinity of the launch site. Reusable vehicle components would be recovered with refurbishment capability at the launch/buildup site.

The major launch site facility requirements and selected reusable vehicle components are shown in the table in Figure D-4. These requirements are derived from the ground processing timelines previously discussed using a parametric launch rate from 1 to 20 flights/year, with 24 hour-6 day processing. The parametric flight rate was used to define the thresholds in facility needs. Although a requirement for only a single launch pad is shown, at least two are recommended for backup and parallel launch capability.

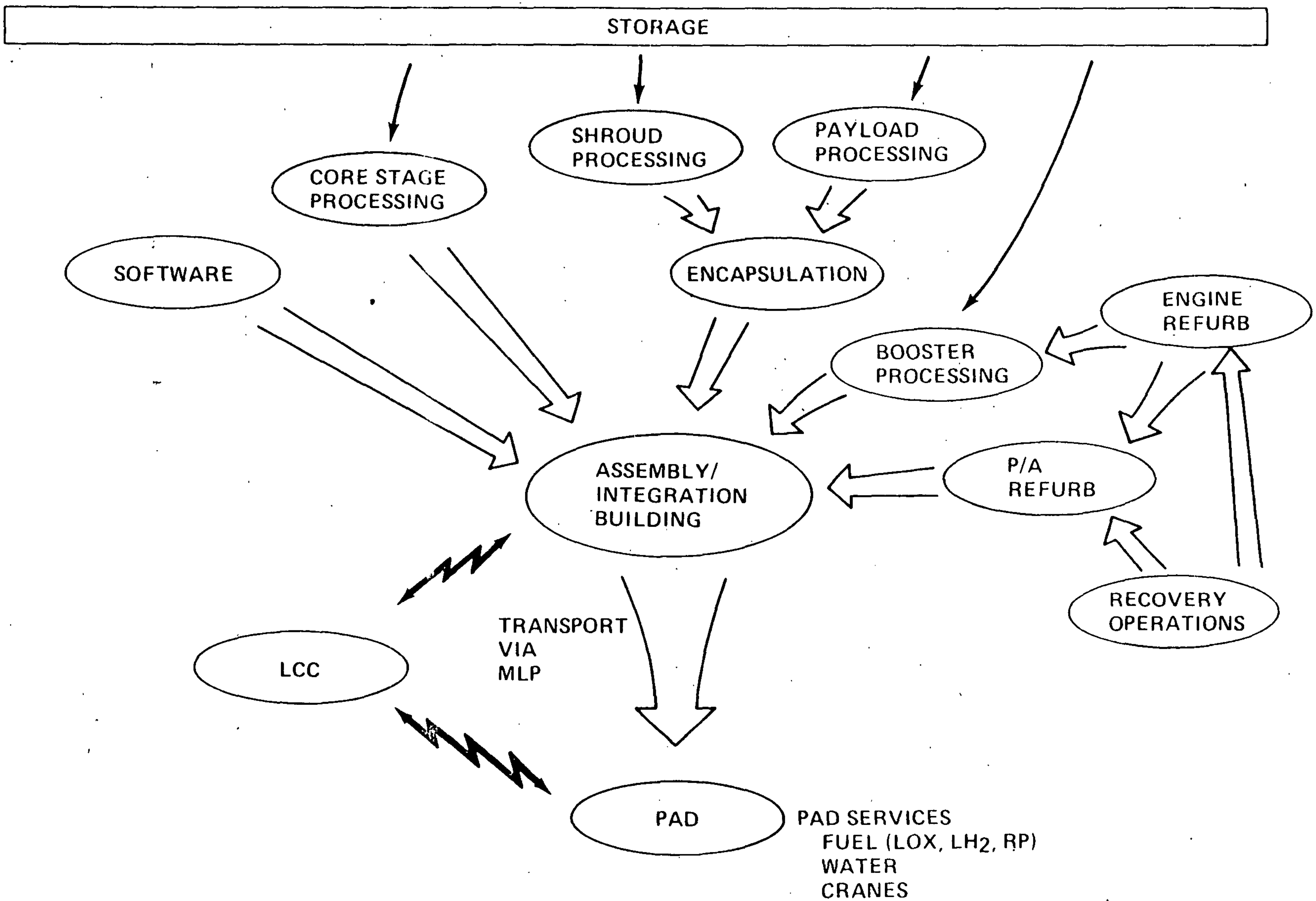


Figure D-1. Heavy lift launch vehicle functional facilities.

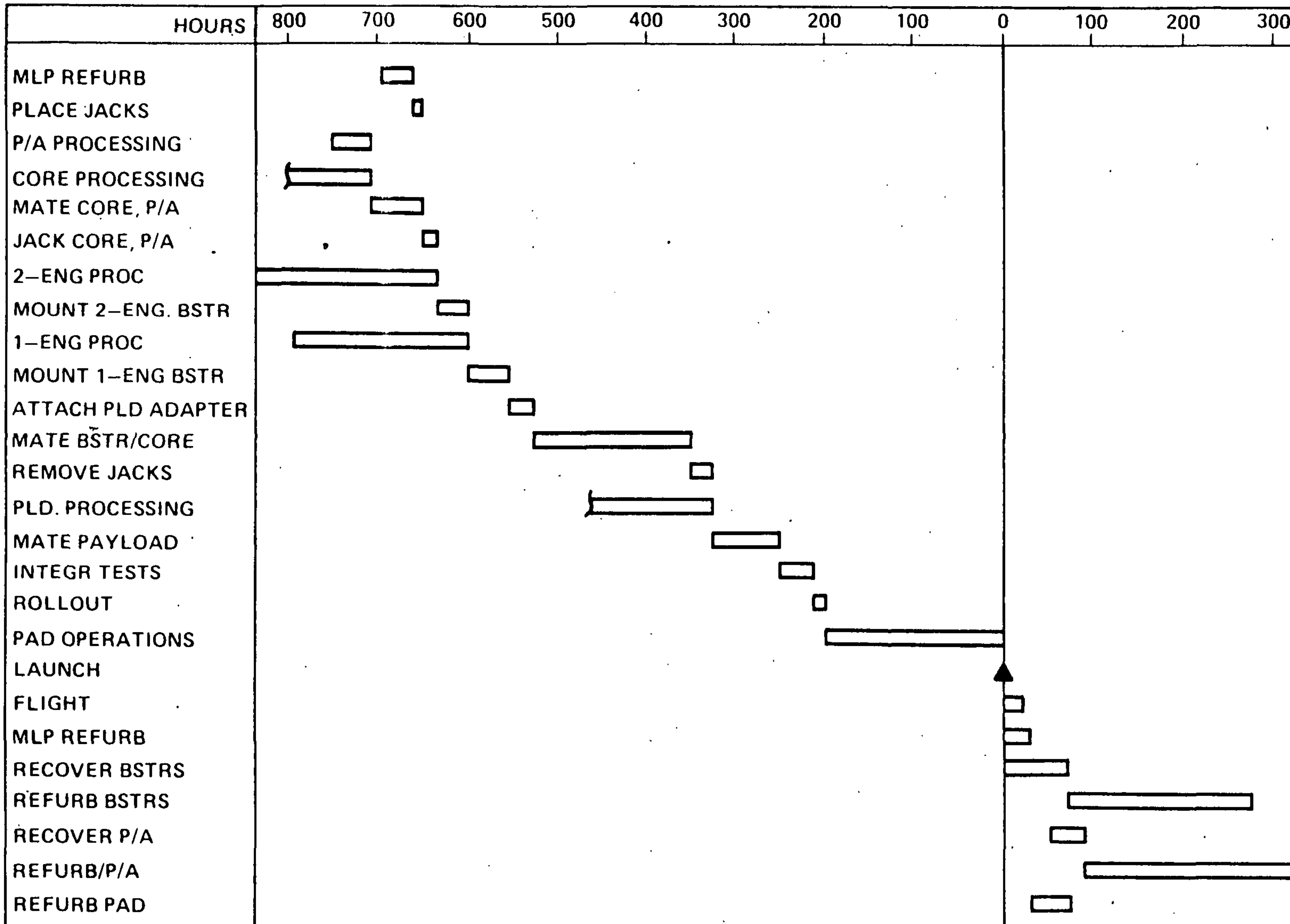


Figure D-2. Heavy lift launch vehicle ground processing timeline.

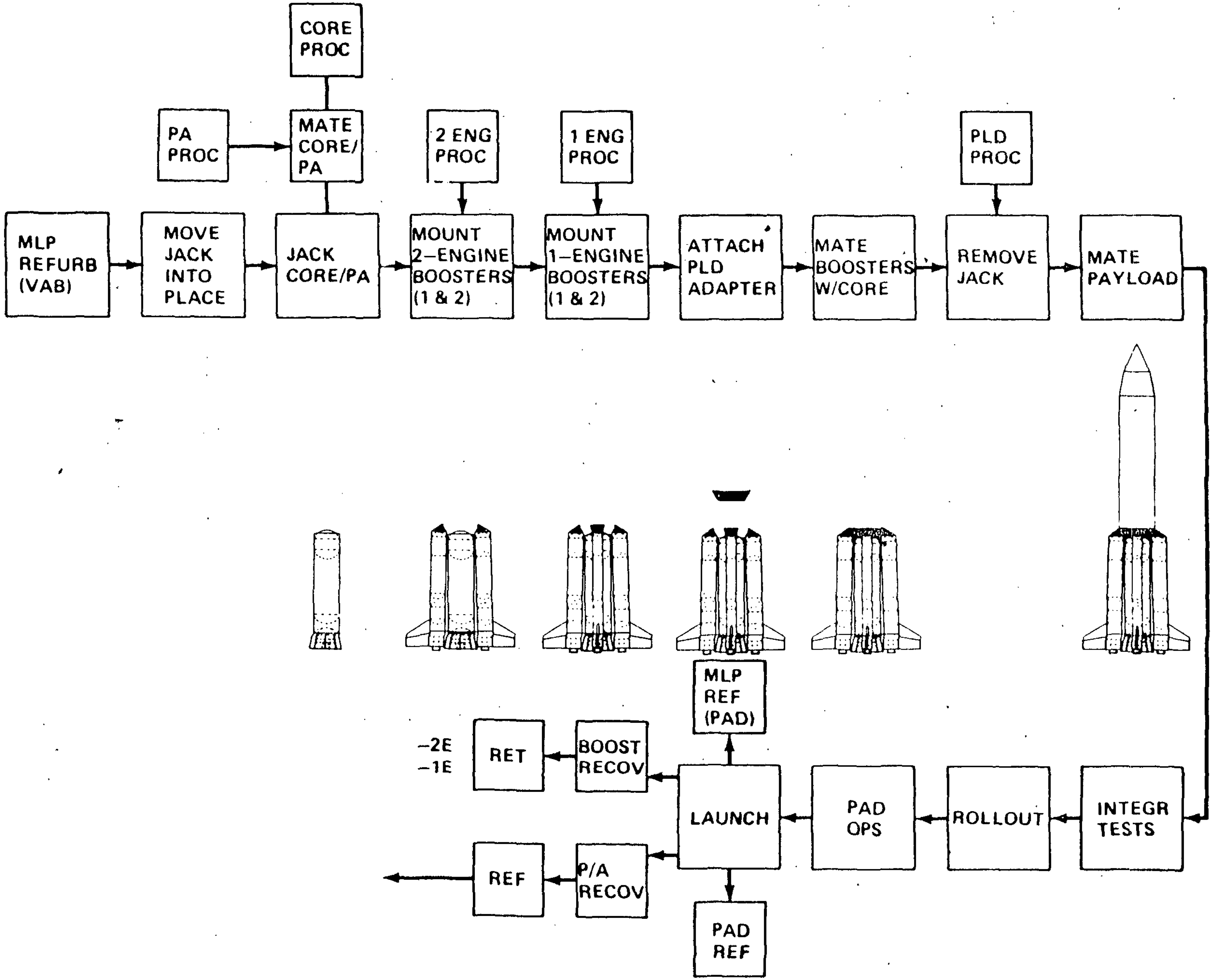


Figure D-3. Ground operations flow.

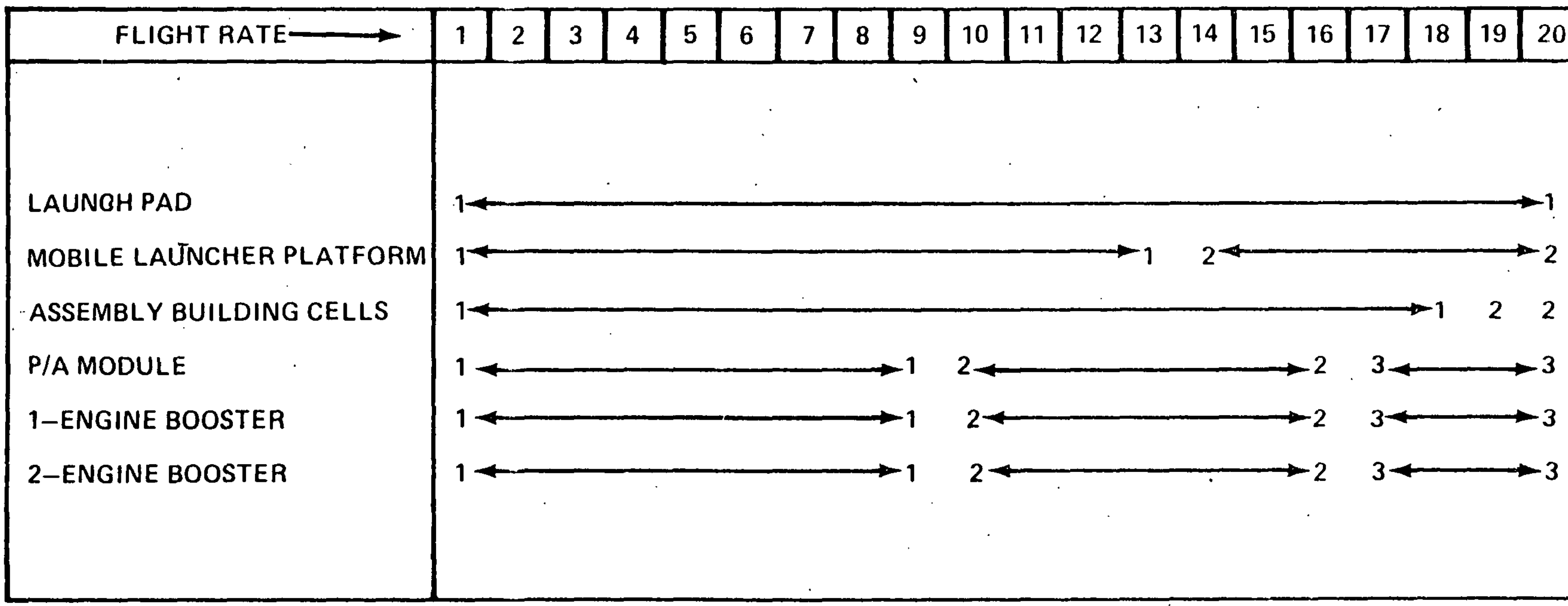


Figure D-4. Heavy lift launch vehicle facility/equipment requirements.

APPENDIX E. PERFORMANCE

The Design Reference Mission (DRM) is delivery of a 300,000 lb payload into a 540 n.mi. circular orbit inclined at 90 deg to the equator. Launch is from WTR on a flight azimuth of 179 deg measured from north. The HLLV is a two-stage parallel burn to perigee insertion of a 100 by 540 n.mi. orbit. Circularization at apogee is by the payload itself or use of an optional storable propellant kickstage. Propulsion data is detailed in Appendix F. Parallel burn profile was selected after various sizing analyses and resulted in a configuration using six LOX/hydrocarbon booster engines having 1.616 million lb sea level thrust each and five two-position nozzle LOX/LH_2 engines of 481,000 lb vacuum thrust each. The boosters contain all propellants consumed during the parallel burn portion of flight, i.e., LOX and LH_2 are crossfed to the second stage so the second stage tanks are full at booster separation.

The thrust to weight at liftoff is 1.36 g, and the boosters burn for 135 sec after liftoff where they are separated for partial recovery. The second stage engine bells are extended, and the vehicle is targeted for Main Engine Cutoff (MECO). The payload fairing (PLF) is jettisoned at 350,000 ft. The vehicle injects at perigee of 100 by 540 n.mi. orbit. The second stage separates from the payload and/or kickstage while coasting to first apogee of 540 n.mi. The kickstage/payload circularizes at the 540 n.mi. orbit.

The second stage coasts in an elliptical orbit until past second perigee. The Propulsion/Avionics (P/A) Module positions the second stage tanks at a correct deboost attitude, begins a slow roll for stabilization, and separates. The tanks are deorbited near second apogee for impact in a safe area of the Pacific Ocean by solid motors located in the conical adapter.

The P/A Module performs a phasing maneuver to adjust its groundtrack for a subsequent landing at Edwards Air Force Base. The phasing and deorbit maneuvers are performed with the P/A Module storable propellant engines. Three axis attitude control is provided by the RCS engines.

The performance ground rules and assumptions used for the trajectory simulation and vehicle performance are presented in Table E-1.

Table E-2 presents the baseline DRM weight sequence. Figure E-1 displays trajectory results of dynamic pressure, longitudinal acceleration, relative velocity versus flight time, and the ascent flight vehicle subpoint locus.

The HLLV configuration has growth capability by changing the booster assignment and the power level setting of the stage two engines. Replacing the single engine boosters with two engine boosters results in an increase of thrust and available propellant. The crossfeed propellant capacity is increased 33 percent, and to consume this requires increasing the stage two thrust 33 percent per engine during the booster burn. Two options for this case were investigated. The first option was to increase the core thrust 33 percent from liftoff to orbital insertion, and the second was a thrust increase from liftoff to booster burnout, then reduce thrust to nominal. The results are summarized in Table E-3.

A small payload two stage vehicle was derived from the two engine booster hardware. The first stage has two STBEs, and the upper stage, a single STME. The performance of this configuration is detailed in Table E-4. Table E-5 is a performance summary of HLLV options and derivatives to various orbits and inclinations. An option which uses three two-engine boosters is listed. This case would eliminate the development of the single engine booster. Performance of various Shuttle configurations in which the SRBs have been replaced with modified HLLV liquid rocket boosters (LRB) is presented in Table E-6. Initially, 2 two-engine LRBs with a full propellant load were investigated for the Shuttle. This results in a payload capability of approximately 167K lb. An attempt was made to reduce the payload of this configuration to the capability of the current Shuttle with SRBs (65K lb) by off-loading booster propellant. This results in excessive maximum dynamic pressure. A second option was then investigated in which three single engine LRBs were used with the Shuttle. This results in a payload capability of 95K lb which can be reduced to 65K-lb by off-loading 400K lb of LRB propellant.

Another configuration studied is the Shuttle Derived Heavy Lift Vehicle (SD/HLV). This configuration consists of a modified external tank (ET) from Shuttle derivative vehicle studies. Two engine LRB's replace the SRB's, a reusable propulsion/avionics module with three SSMEs is mounted beneath the ET, and a payload fairing (25 ft x 90 ft payload envelope) is mounted atop the ET. Payload capabilities of this vehicle are shown in Table E-7 to inclinations of 28.5 deg and 90 deg.

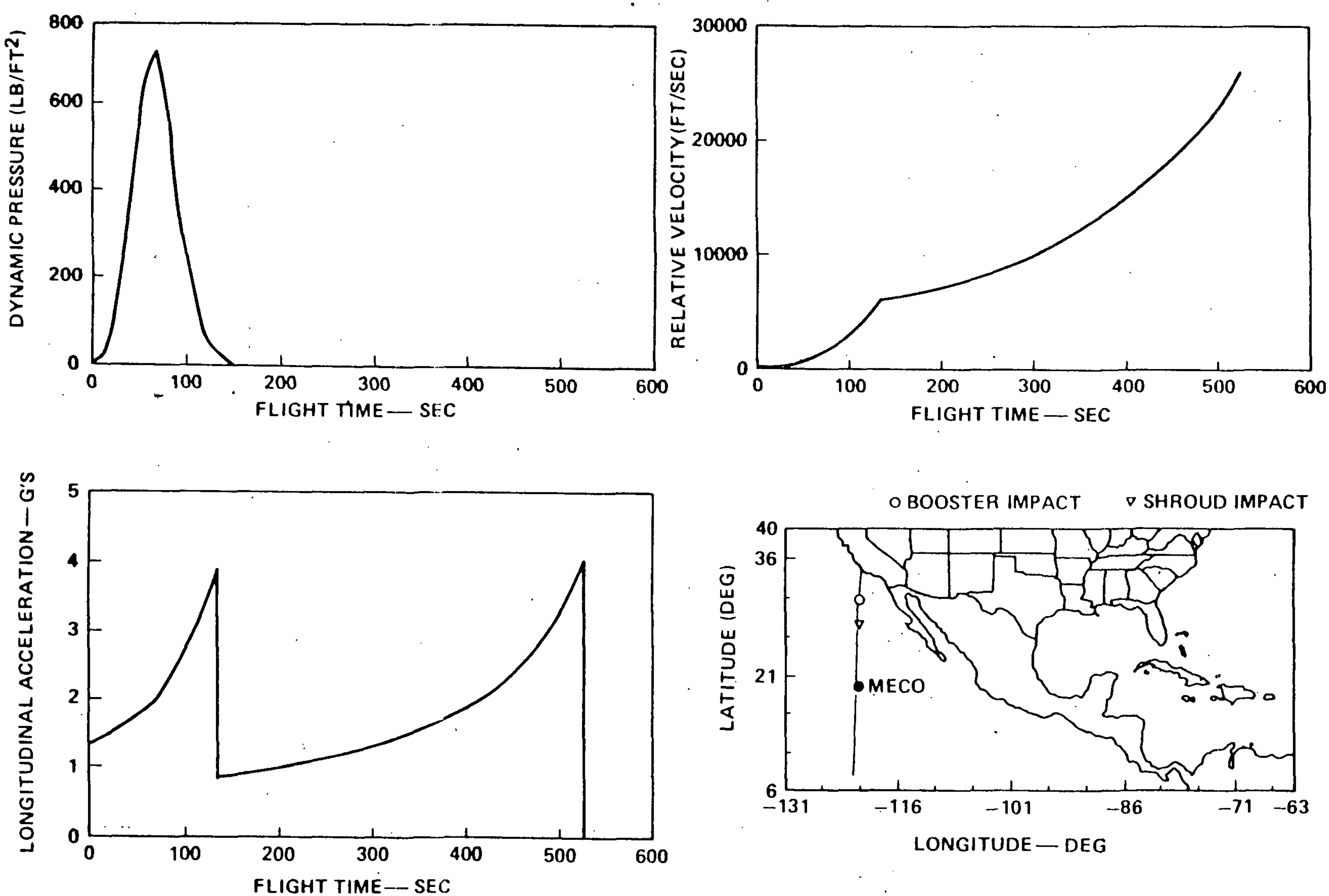


Figure E-1. Ascent flight environment.

TABLE E-1. GROUNDRULES AND ASSUMPTIONS

- LAUNCH SITE: WTR
- PAYLOAD DESIRED = 300 KLBS TO 540 NMI ALTITUDE/90–DEGREES INCLINATION ORBIT
- TWO STAGE TO 100 X 540 NMI WITH PERIGEE INJECTION
- THRUST–TO–WEIGHT AT LIFTOFF ≈ 1.35
- MAXIMUM ASCENT DYNAMIC PRESSURE: UNCONSTRAINED
- MAXIMUM ASCENT ACCELERATION: UNCONSTRAINED
- PAYLOAD FAIRING
  - 50 FT DIAMETER X 200 FT LENGTH
  - JETTISONED AT 350,000 FEET
- FLIGHT PERFORMANCE RESERVES = 3/4% TOTAL ΔV
- BOOSTER THRUST TO BE SELECTED
- STAGE 2 ENGINES
  - TWO POSITION NOZZLES WITH EXPANSION RATIOS OF 55 AND 150
  - VACUUM THRUST IS 481,000 LBS @ ϵ = 150
- KICKSTAGE WITH FOLLOWING CHARACTERISTICS USED TO CIRCULARIZE 300 KLBS PAYLOAD AT APOGEE:
  - MASS FRACTION (λ') = 0.90
  - ISP = 343 SEC
  - TOTAL STAGE WEIGHT = 22.5 KLBS: I.E., PAYLOAD + KICK STAGE WEIGHT IN 100 X 540 NMI ORBIT IS 322.5 KLBS.

TABLE E-2. HLLV WEIGHT SUMMARY

BRM REQUIREMENTS

- TWO STAGES PARALLEL BURN TO 100 x 540 NMI ORBIT
- INCLINATION = 90 DEGREES

BRM RESULTS

- 6 x STBE @ F_{SL} = 1.616 MLBF EACH
- 5 x STME @ F_{VAC} = 481 KLBF EACH

	WEIGHT ~ KLBS
GLOW	8,574
BOOSTER PROPELLANT CAPACITY	5,233
BOOSTER JETTISON WEIGHT	588
STAGE 2 LIFTOFF WEIGHT	2,753
STAGE 2 PROPELLANT WEIGHT	2,055
PAYLOAD FAIRING WEIGHT	126
STAGE 2 TANK JETTISON WEIGHT	139
PROPULSION/AVIONICS MODULE WEIGHT	108
PAYLOAD/KICKSTAGE WEIGHT IN 100 x 540 NMI ORBIT	325

F/W_{LO} = 1.36 g's

MAX_q = 725 LB/FT

MAX g's = 4.14

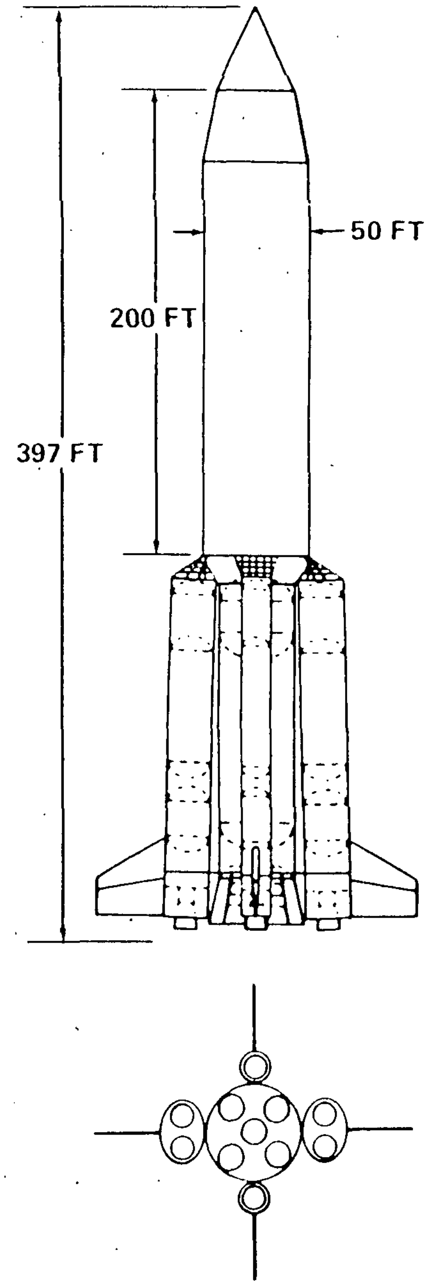


TABLE E-3. HLLV PERFORMANCE ENHANCEMENT

- PARALLEL BURN
- 2 STAGE TO ORBIT
- ORBIT 100 x 540 NMI
- INCLINATION = 90 DEG
- LOX/HYDROCARBON ENGINES BOOSTERS

		$8 \times F_{SL}$ = 1.616 MLBS/ENGINE 5 x SSME'S @ 133% F_{VAC} = 641.3 KLBS/ENGINE BOOSTERS = 4 x 246 IN WEIGHT ~ KLBS	$8 \times F_{SL}$ = 1.616 MLBS/ENGINE 5 x STME'S @ 133%/100% @ 135 SEC F_{VAC} = 641.3/481 KLBS/ENG. BOOSTERS = 4 x 246 IN WEIGHT ~ KLBS
GLOW	=	10,579	10,560
W_{PB} LOX/RP	=	6,039	6,039
W_P LOX/LH_2	=	939	939
W_{NB}	=	703	703
W_{02}	=	2,898	2,879
W_{P0} LOX/LH_2	=	2,054	2,054
W_{SHROUD}		126	126
W_N ET	=	139	139
W_N P/A MOD	=	108	108
W_{PLD} + KICKSTAGE	=	471	452
F/W_{LO}	=	1.47	1.47
MAX q LB/FT^2	=	921.	910.
MAX g's	=	4.85	4.87

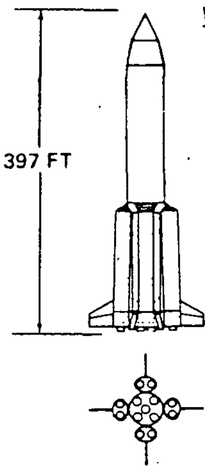


TABLE E-4. HLLV TWO STAGE DERIVATIVE (246 in. DIAMETER TANDEM STAGES)

- SERIES BURN
- 2 STAGE TO ORBIT
- ORBIT 100 NMI
- LOX/HYDROCARBON ENGINES BOOSTERS (2 @ FSL = 1.616 MLBS/ENGINE)
- 1 X STME @ F_{VAC} = 481 KLBS

		WEIGHT ~ KLBS INCL = 90 DEG	WEIGHT ~ KLBS INCL = 28.5 DEG
GLOW	=	2,322	2,341
W_{PB}	=	1,510	1,510
W_{NB}	=	142	142
W_{02}	=	670	689
W_{P0}	=	500	500
W_{PLF}*	=	13	13
W_{N2}	=	53	53
W_{PLD}	=	104	123
F/W_{LO}	=	1.39	1.39
MAX q LB/FT^2	=	849.	841.
MAX g'S	=	4.42	4.32

*PLF DROP @ 350,000 FT

TABLE E-5. ALTERNATE CONFIGURATIONS/ORBIT PERFORMANCE SUMMARY

• WTR LAUNCH • INCLINATION = 90 DEG

VEHICLE	INJECTION ORBIT NMI	INJECTION WEIGHT (LBS)	KICKSTAGE WEIGHT (LBS)	PAYLOAD (LBS)	MAX q (LB/FT^{-2})	MAX g (G'S)
BASELINE HLLV * ①	100 x 540	325,096	22,692	302,404	725	4.14
	100 x 540	322,344	22,493	299,751	727	3.00
	100 x 270	343,449	10,066	333,383	721	4.01
	160 x 160	318,434	0	318,434	696	4.19
ALT HLLV (133%) ②	100 x 540	471,029	32,878	438,151	921	4.85
(133%)	100 x 100	513,165	0	513,165	911	4.79
(100%) ③	100 x 540	452,312	31,571	420,741	910	4.87
ALT (TANDEM) ④	100 x 100	104,721	0	104,721	849	4.42
ALT HLLV ⑤	100 x 540	332,981	23,242	309,739	736	4.08

• ETR LAUNCH • INCLINATION = 28.5 DEG

VEHICLE	INJECTION ORBIT NMI	INJECTION WEIGHT (LBS)	KICKSTAGE WEIGHT (LBS)	PAYLOAD (LBS)	MAX q (LB/FT^{-2})	MAX g (G'S)
BASELINE HLLV * ①	100 x 540	392,928	27,427	365,501	718	3.84
	100 x 270	413,592	12,121	401,471	714	3.81
	160 x 160	383,206	0	383,206	689	3.85
	100 x 100	427,938	0	427,938	711	3.80
ALT HLLV (133%) ②	100 x 100	605,432	0	605,432	901	4.67
ALT (TANDEM) ④	100 x 100	124,025	0	124,025	841	4.32

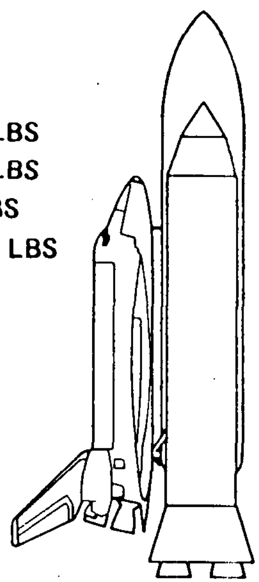
① HLLV CONFIG 2 (2 x 246 IN + 2 x 171 IN STRAP–ON BOOSTERS); STME'S @ 100%
② ALT HLLV (4 x 246 IN STRAP–ON BOOSTERS); STME'S @ 133%
③ ALT HLLV (4 x 246 IN STRAP–ON BOOSTERS); STME'S @ 133%/100% @ 135 SEC
④ ALT (TANDEM) (246 IN TANDEM STAGES); 1 x STME @ 100%
⑤ ALT HLLV (3 x 246 IN STRAP–ON BOOSTERS): STME'S @ 100%

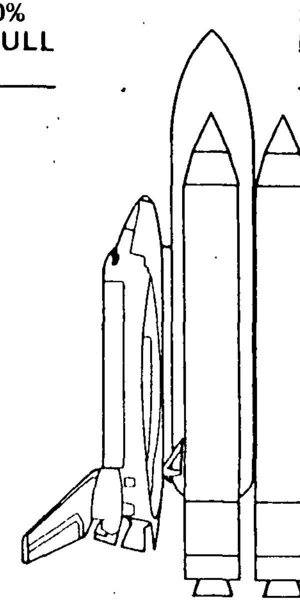
*SENSITIVITIES

∂ PAYLOAD/∂ BOOSTER JETTISON WEIGHT = 0.12 LB/LB
∂ PAYLOAD/∂ SHROUD JETTISON WEIGHT = 0.16 LB/LB

TABLE E-6. LIQUID BOOSTER APPLICATION TO SPACE SHUTTLE

- LOX/HYDROCARBON SEA LEVEL THRUST = 1.616 M LBS/ENGINE
- PARALLEL BURN
- NOMINAL INSERTION
- ALTITUDE = 150 NMI
- INCLINATION = 28.5 DEG





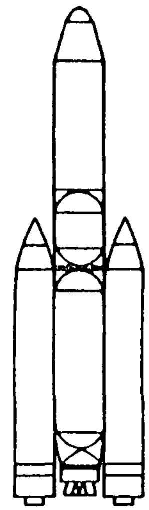
	2 TWO ENGINE LIQUID BOOSTERS	3 SINGLE ENGINE BOOSTERS	
	SSME'S @ 109% BOOSTERS FULL	SSME'S @ 100% BOOSTERS FULL	SSME'S @ 100% BOOSTERS OFF–LOADED 400K LBS
GLOW ~ LBS	5,350,649	4,505,534	4,074,940
LRB PROPELLANT ~ LBS	3,019,368	2,264,526	1,864,526
LRB JETTISON WT ~ LBS	291,928	276,141	276,141
ET PROPELLANT ~ LBS	1,582,918	1,582,931	1,582,928
ET + ORBITER WT* ~ LBS	289,923	286,864	285,608
PAYLOAD ~ LBS	166,512	95,072	65,737
F/W_{LO}	1.44	1.33	1.47
MAX q LB/FT^2	714	656	657
MAX g 'S	3.00	3.00	3.00
V_{STGB} FPS	6954	5617	4237
Q_{STGB} LB/FT^2	32	50	122

*INCLUDES OMS PROPELLANT

TABLE E-7. LIQUID BOOSTER APPLICATION TO SHUTTLE DERIVED VEHICLE

- 2 X TWO ENGINE BOOSTERS
- PARALLEL BURN WITH REUSABLE P/A MODULE (3 X SSME'S @ 109%)
- DIRECT INJECTION @ 57 X 160 NMI
- ALTITUDE = 160 NMI
- STANDARD SIZE EXTERNAL TANK
- PAYLOAD BAY = 25 X 30 FT
- SHROUD JETTISON ED @ 350,000 FT

	WEIGHT ~ LB INCL = 28.5 DEG	WEIGHT ~ LB INCL = 90 DEG
GLOW	5,384,595	5,333,381
LRB PROPELLANT	3,019,368	3,019,368
LRB JETTISON WT	291,928	291,928
SHROUD JETTISON WT	30,660	30,660
ET PROPELLANT α	1,582,878	1,582,882
ET JETTISON WT*	88,413	88,413
P/A MODULE WT**	64,697	63,765
PAYLOAD (LBS)	306,651	256,365
F/W_{LO}	1.43	1.45
MAX q LB/FT^2	948	956
MAX g	3.0	3.0



*INCLUDES RESIDUALS AND RESERVES
**INCLUDES OMS PROPELLANT

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FUNCTION	ENGINE SYSTEM
MAIN PROPULSION	6 STBE @ FSL = 1616K, 5 STME @ FVAC = 481K (PARALLEL BURN)
BOOSTER SEPARATION	40 STS BSM 6 FWD, 6 AFT ON EACH LARGE BOOSTER 4 FWD, 4 AFT ON EACH SMALL BOOSTER
DEORBIT CORE STAGE TANKS	6 STAR 26B RETROS LOCATED IN PAYLOAD ADAPTER
STG 2 P/A MODULE ON ORBIT & DEORBIT	12 RS-34 THRUSTERS (70 LBF) FOR ON ORBIT MANEUVERS 4 AMS ENGINES TO DEORBIT P/A MODULE FOR RECOVERY

Figure F-1. Propulsion system summary.

- PROPELLANTS — LOX/JP4
- NOZZLE AREA RATIO — 25.0
- THRUST (SEA LEVEL) LBF — 1500 TO 2000
- DELIVERED SEA LEVEL ISP SEC — 289
- CHAMBER PRESSURE PSIA — 2000
- MIXTURE RATIO (O/F) — 2.8
- LENGTH IN — 199 TO 226
- NOZZLE EXIT DIAMETER IN — 116 TO 131
- ENGINE INSTALLED WT LBM — 16340 TO 24160

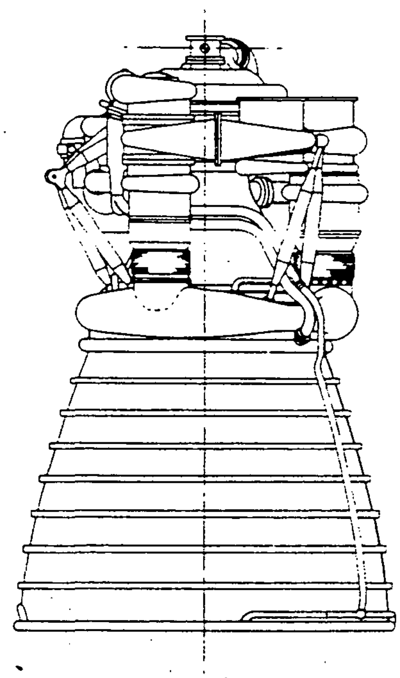


Figure F-2. Space Transportation Booster Engine (STBE).

• PROPELLANTS	LOX/LH_2
• NOZZLE AREA RATIO (STOWED/EXTENDED)	55/150
• VACUUM THRUST LBF	468K/481
• VACUUM ISP SEC	449/461
• CHAMBER PRESSURE PSIA	3006
• MIXTURE RATIO (O/F)	6.0
• LENGTH IN	139/219
• NOZZLE EXIT DIAMETER IN	76.2/126.3
• ENGINE INSTALLED WT LBM	7142
• SEA LEVEL THRUST LBF (STOWED)	397K
• SEA LEVEL ISP SEC	380.4
• FLOWRATE LB/SEC	1043.4

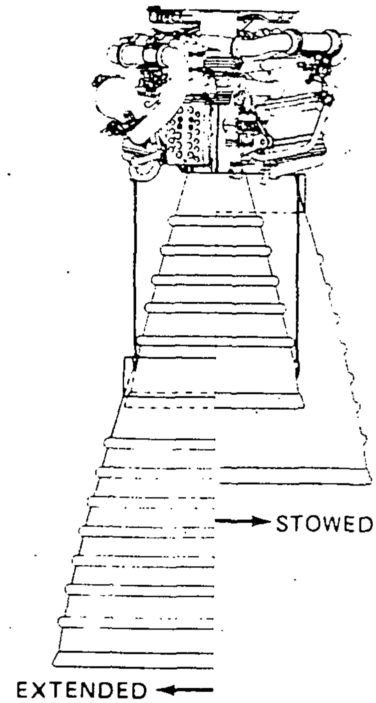
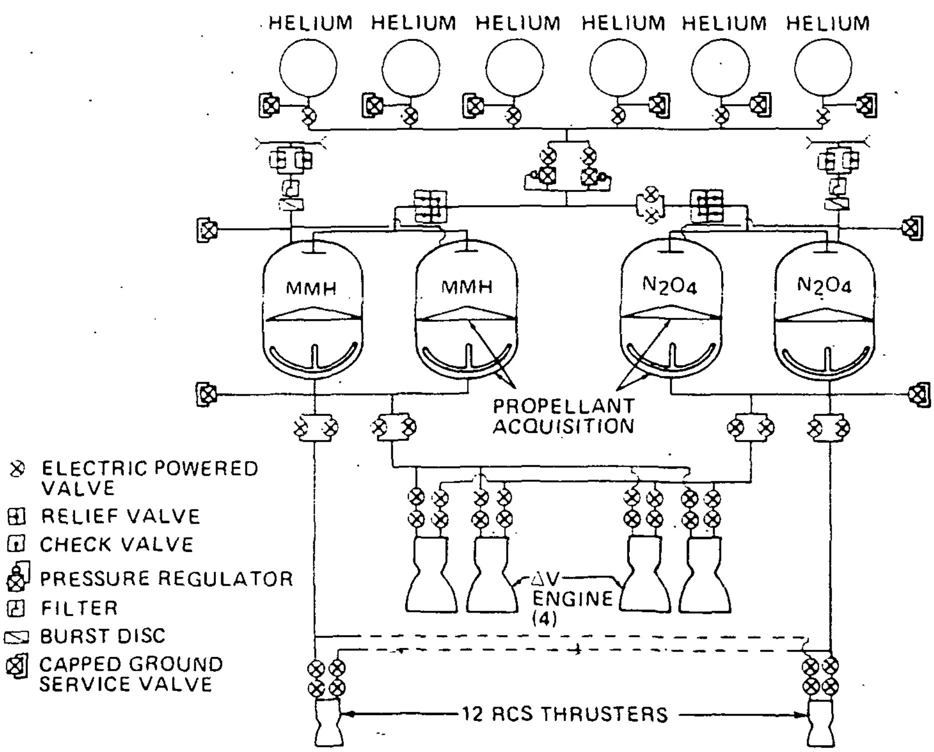


Figure F-3. Space Transportation Main Engine (STME 481).



- PROPELLANT TANKS (4)
  PSI (TDRSS MOD)
  46 FT3 WITH PMD
  150 CYCLE LIFE
- PRESSURANT BOTTLES (6)
  BRUNSWICK BLD 999020
  ORBITER MPS BOTTLES
  8200 IN3 EACH
  4500 PSI OPERATING PRESSURE
- ΔV ENGINES (4)
  2657 LB THRUST
  REUSABLE
  PC= 191 PSIA
- RCS THRUSTERS (12)
  70 LB THRUST
  RS-34
  I_{MIN}= 0.53 LB-SEC
- REGULATORS (2)
  FAIRCHILD /4340000
  DUAL REDUNDANCY
  ORBITER OMS UNITS
  300 SCFM CAPACITY

Figure F-4. P/A Module auxiliary propulsion system.

ITEM	WEIGHT (LB)
OMS/ACS PROPULSION	1944
ΔV ENGINES (4)	440
ΔV ENGINES TVC (4)	48
RCS THRUSTERS (12)	29
PROPELLANT TANKS	825
PROPELLANT FEED, FILL AND VENT	120
GHE BOTTLES	408
GHE FEED, FILL AND VENT, RELIEF	74

Figure F-5. P/A Module auxiliary propulsion dry weights.

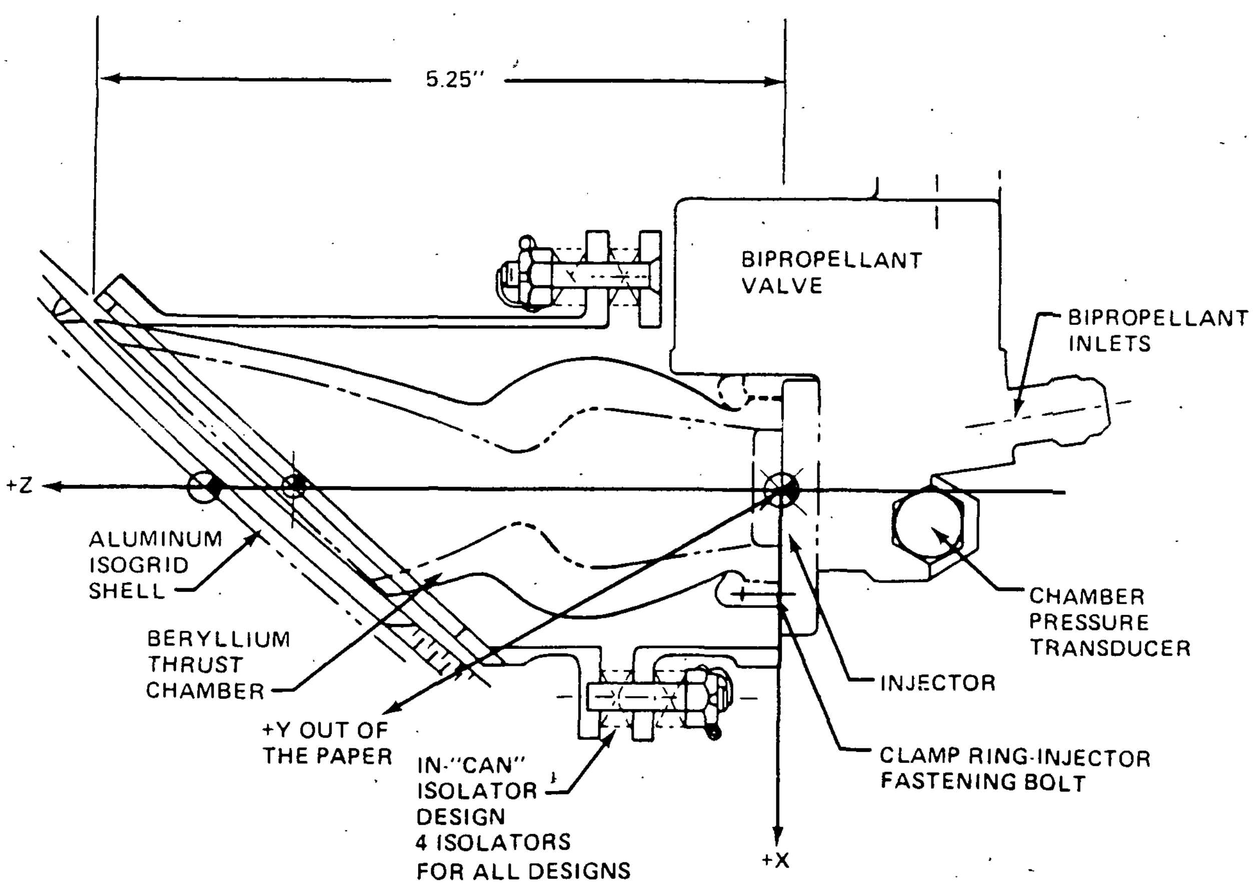
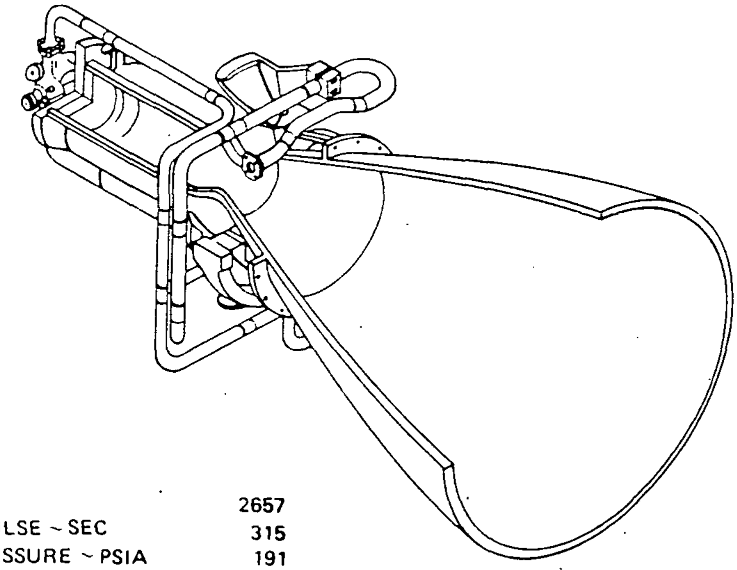


Figure F-6. RS-34 thruster.



THRUST ~ LBF	2657
SPECIFIC IMPULSE ~ SEC	315
CHAMBER PRESSURE ~ PSIA	191
MIXTURE RATION	1.63
AREA RATIO	90:1
WEIGHT ~ LBM	110
ENVELOPE ~ IN	31 DIA x 54 LONG

Figure F-7. OMS engine specifications.

	SPECIFICATION REQUIREMENT
WEB ACTION TIME (WAT) ~ SEC.	0.8 MAX.
TOTAL TIME ~ SEC.	1.05 MAX.
IGNITION INTERVAL, SEC.	0.030 – 0.100
PRESSURE, MAX. (P_C) PSIA	(2200 MEOP)
PRESSURE, WAT AVG. ~ PSIA	(1500 – 1850 FROM TEST REPORTS)
PRESSURE AT EWAT (P_{cEWAT}) ~ PSIA	2,000 MAX.
PEAK VACUUM THRUST ~ (FPEAK) LB	29,000 MAX.
F_{VAC}, WAT* AVG. (AVG. VACUUM THRUST) ~ LBF	18,500 MIN.
TOTAL IMPULSE, VAC. WAT* ~ LB–SEC	14,000 MIN.
TOTAL IMPULSE, VAC. ACTION TIME ~ LB–SEC	15,000 MIN.
TIME TO THRUST ~ MSEC	200 MAX
TIME (PC_{EWAT} TO $P_{cEWAT}/2$ ~ MSEC)	150 MAX.
ISP VACUUM = TOTAL I_{VAC}/PROPELLANT WEIGHT ~ LBF-SEC/LBM MASS	195.3
WEIGHTS ~ LBS	
MOTOR PREFIRE WEIGHT	167
PROPELLANT WEIGHT (B12002)	75.3 TO 78.8
TOTAL EXPENDED WEIGHT	76.5
EMPTY CASE (B12001)	43.1
LINED CASE (B12002)	45
LINED CASE WITH PROPELLANT (B12004)	123
IGNITER ASSEMBLY (B12011)	4.4
NOZZLE ASSEMBLY (B12003)	29
NOZZLE	
THROAT DIAMETER ~ IN	3.132
EXIT DIAMETER ~ IN	7.535
GRAIN	
GRAIN WEB THICKNESS ~ IN	0.549/0.543
GRAIN LENGTH ~ IN	20.69
PROPELLANT: UTP–19048	
HTPB	
AMMONIUM PERCHLORATE–%	83.77
HTPB ~	11.7
OTHER** %	4.53

*WAT = WEB ACTION TIME
**(SEE CSD SEO 722A)

Figure F-8. Booster Separation Motor (BSM) specifications.

- MOUNT BSM'S AT FORWARD AND AFT POSITIONS ON BOOSTERS
  - POINTED TOWARDS CORE STAGE
  - CANTAS REQUIRED
- LARGE BOOSTERS
  - LATERAL FORCE OF 152,361 LBF
  - BURN TIME OF 0.8 SEC
- SMALL BOOSTERS
  - LATERAL FORCE OF 95,858 LBF
  - BURN TIME OF 0.8 SEC

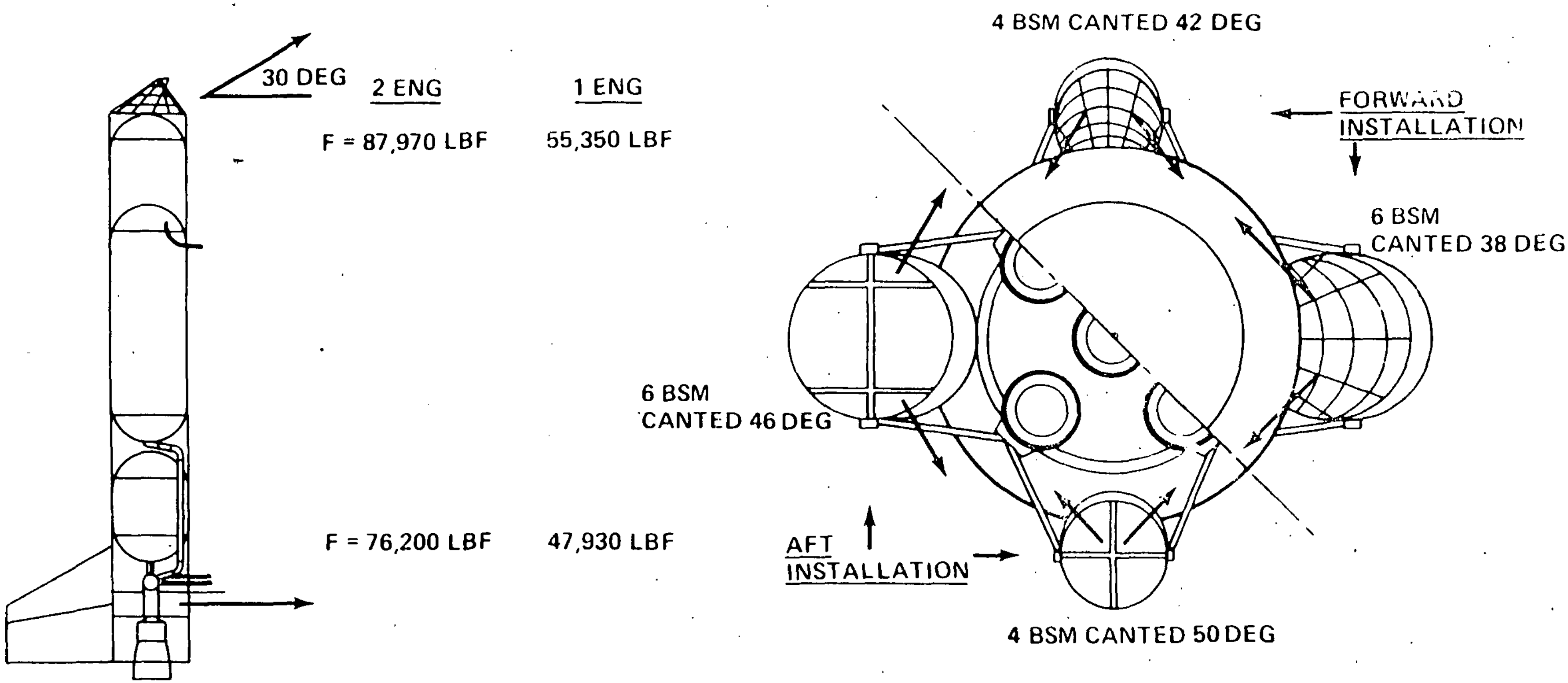
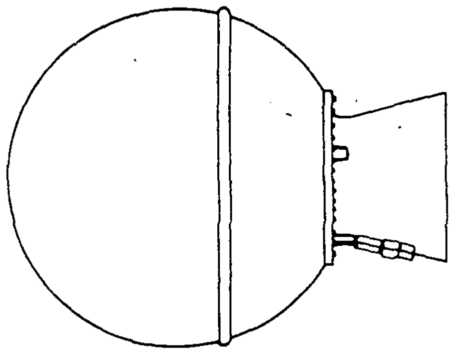


Figure F-9. Booster separation.



THE STAR 26B WAS QUALIFIED AND FLOWN AS AN UPPER STAGE ON THE BURNER IIA SPACECRAFT FOR BOEING AND THE USAF

MOTOR PERFORMANCE*

BURN TIME/ACTION TIME (t_b/t_a) ~ SEC	17.8/18.6
IGNITION DELAY TIME (t_d) ~ SEC	0.06
BURN TIME AVG. CHAM. PRESS. (P_b) ~ PSIA	623
MAXIMUM CHAMBER PRESSURE (P_{max}) ~ PSIA	680
TOTAL IMPULSE (I_T) ~ LBF-SEC	142,759
PROPELLANT SPECIFIC IMPULSE ~ LBF-SEC/LBM	272.4
EFFECTIVE SPECIFIC IMPULSE ~ LBF-SEC/LBM	272.7
BURN TIME AVERAGE THRUST (F_p) ~ LBF ~	7,784
MAXIMUM THRUST (F_{max}) LBF	8,751

TEMPERATURE LIMITS

OPERATION ~ DEG F	50 TO 90
STORAGE ~ DEG F	40 TO 100

WEIGHTS ~ LBM

TOTAL LOADED**	575.6
PROPELLANT (INCLUDING 0.4 LBM IGNITER PROPELLANT)	524.0
CASE ASSEMBLY	23.5
NOZZLE ASSEMBLY	19.3
IGNITER ASSEMBLY, EMPTY	1.4
INTERNAL INSULATION	5.8
LINER	0.6
MISCELLANEOUS	1.0
TOTAL INERT	51.6
BURNOUT	50.3

*70°F, VACUUM (I_{SP} BASED ON BURNER IIA FLIGHT DATA)
** PROPELLANT MASS FRACTION 0.910

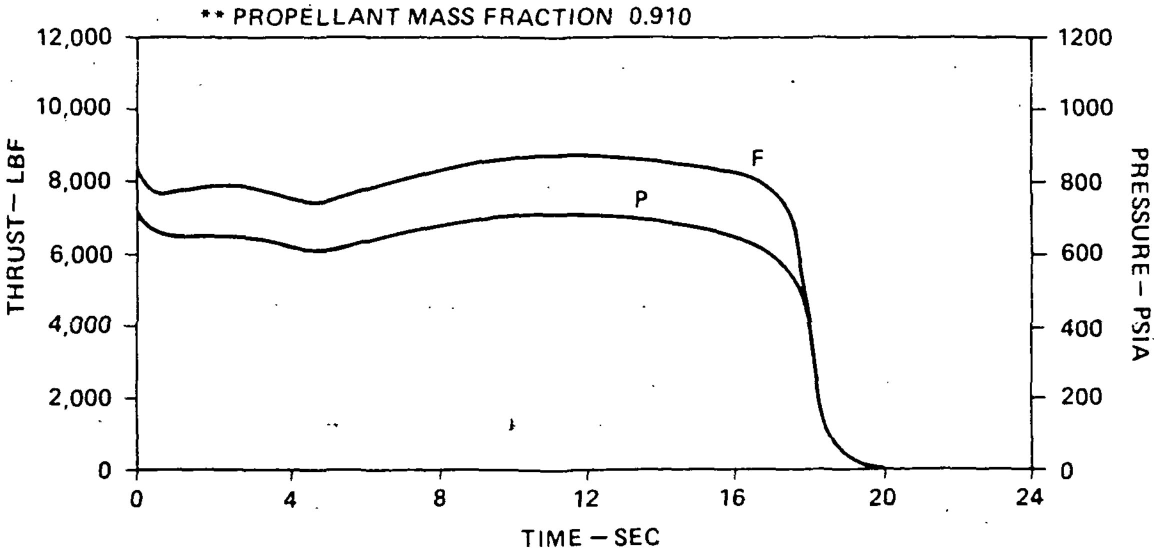


Figure F-10. Star 26B SRM.

APPENDIX G. REENTRY DATA

Each HLLV flight results in large pieces of equipment reentering the atmosphere. They are the first stage boosters, the payload fairing, the second stage core tanks, and the Propulsion/Avionics (P/A) Module. Return trajectory data and induced environment of the recoverable boosters and the P/A Module are discussed.

The mission profile for the Design Reference Mission and times of major events are listed on Table G-1.

Portions of the boosters are to be recovered for reuse. Analyses were performed to estimate the environment experienced during reentry. Table G-2 lists the ground-rules and assumptions used in modeling the reentry trajectories. Table G-3 displays results due to end-on, tumbling, and sideways orientations during reentry. The tumbling or end-on have the highest probability of occurrence and will experience the most severe reentry environment. Figures G-1 and G-1(a) display time varying trajectory parameters for the two-engine boosters. Figures G-2 and G-2(a) show the same data for the single engine boosters.

The P/A Module is to be recovered for reuse. After the payload is separated and the core stage disposed, the P/A Module will perform an orbital correction maneuver to insure the ground track passes over the Edwards Air Force Base landing site.

For reentry analyses, two targeted perigee altitudes were assumed for reentry, 30 n.mi. and 10 n.mi. Reentry data are shown for both. Table G-4 lists the ground rules for the reentry initiation. Figure G-3 displays the orbital ground track from launch to landing of the P/A Module. Figures G-4 and G-4(a) display time varying trajectory parameters of the two reentry trajectories. The data in Figure G-4 were used for the detailed thermal analyses in Appendix J.

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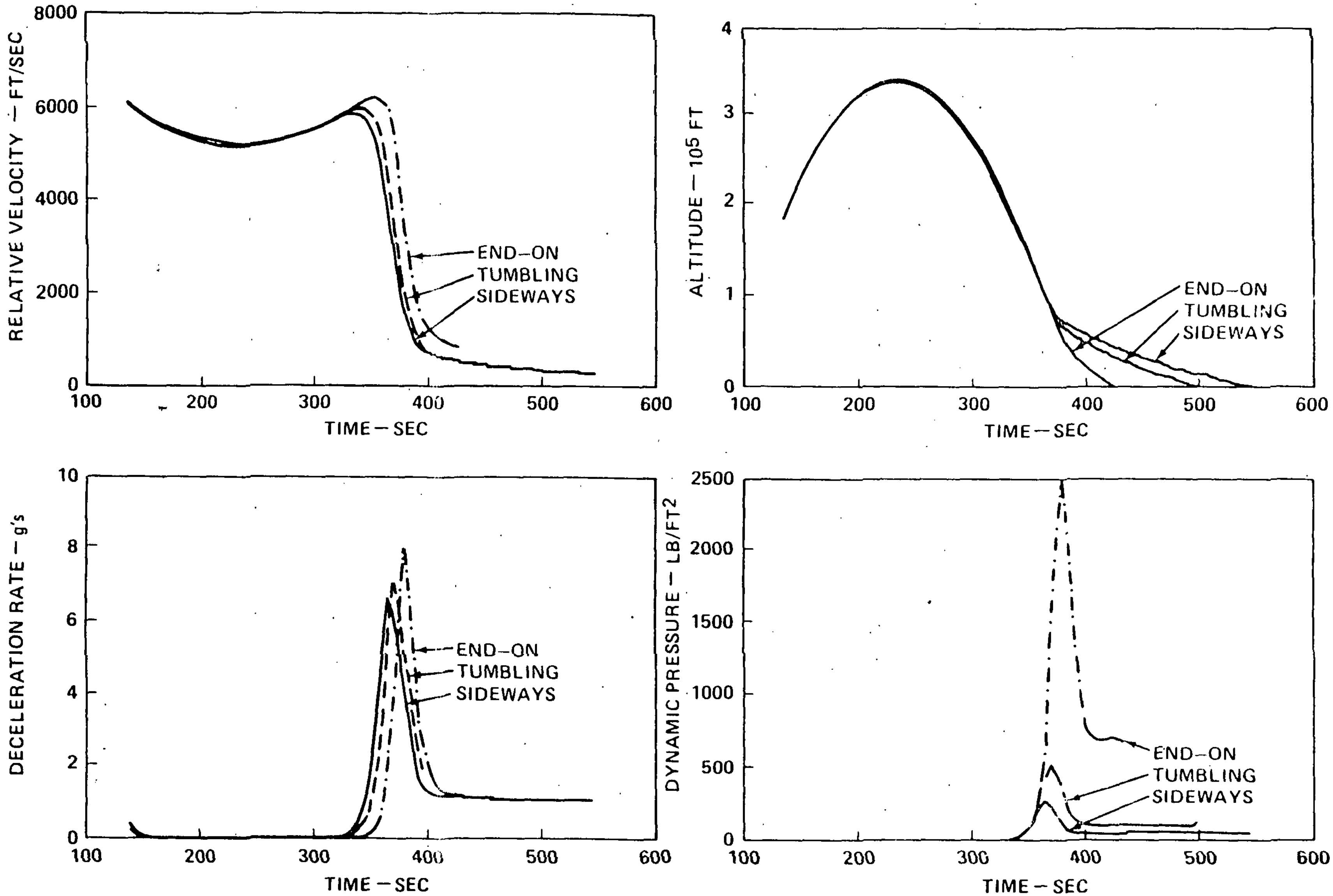
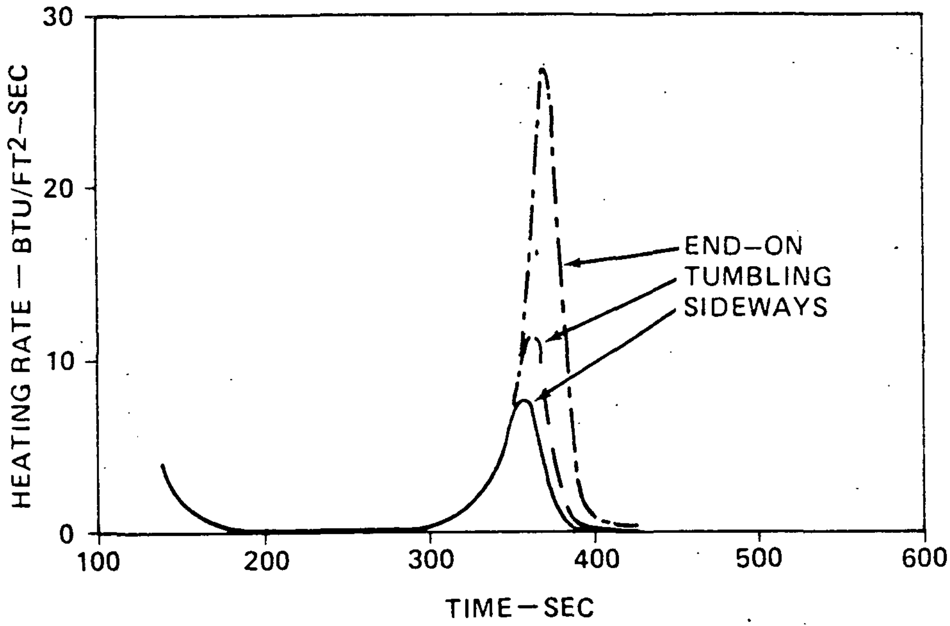


Figure G-1. HLLV two-engine booster reentry.



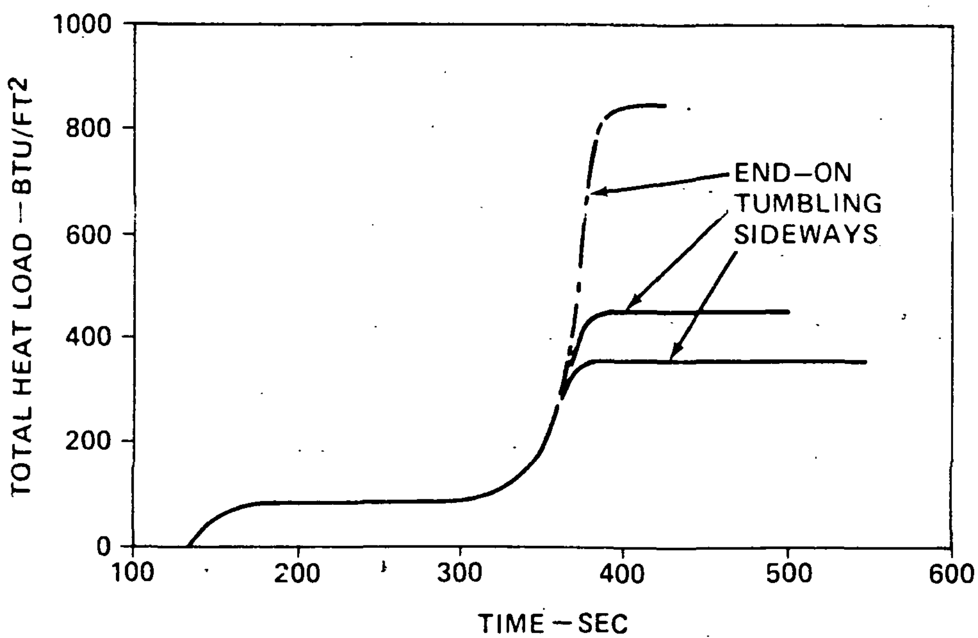


Figure G-1(a). HLLV two-engine booster reentry.

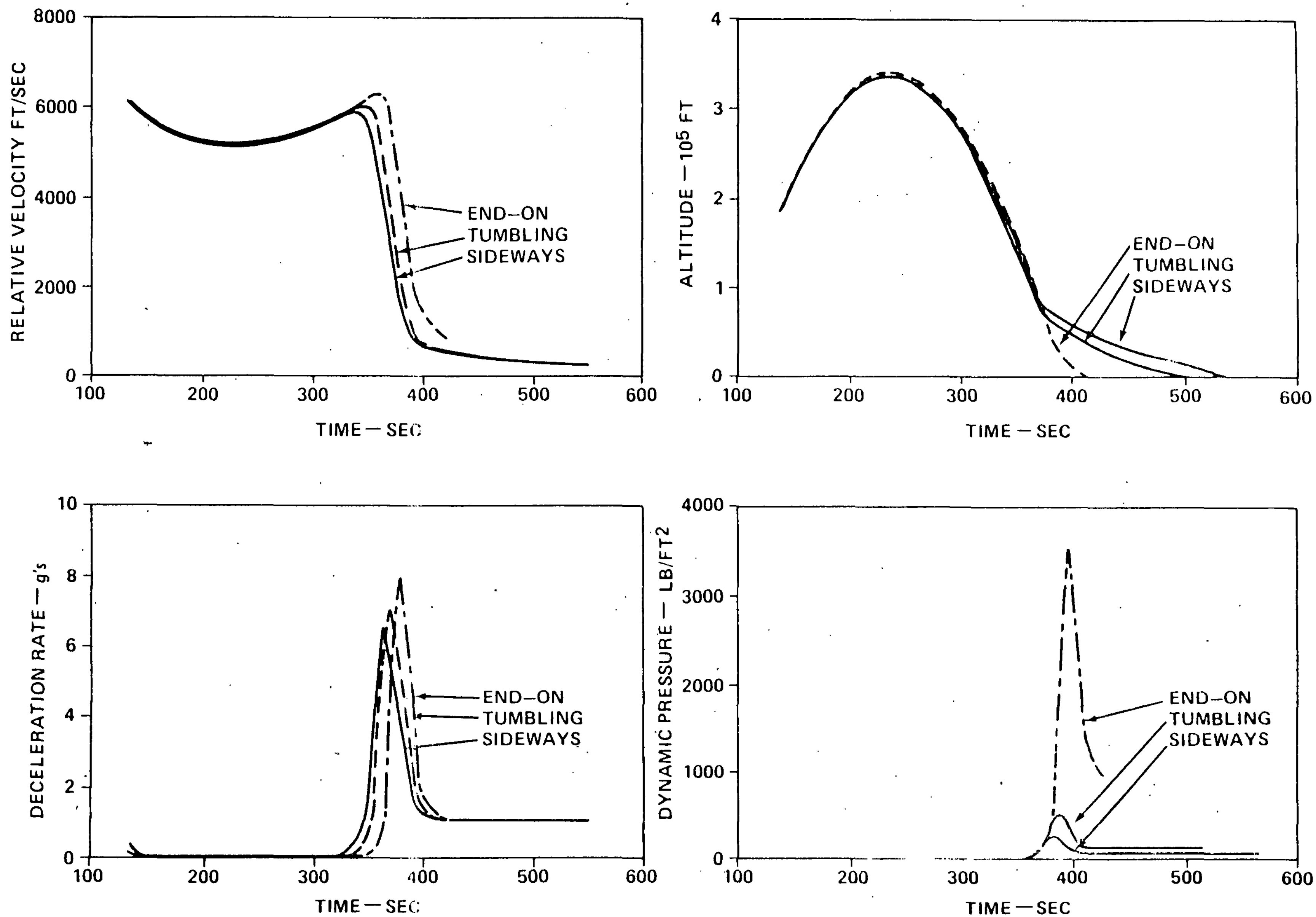
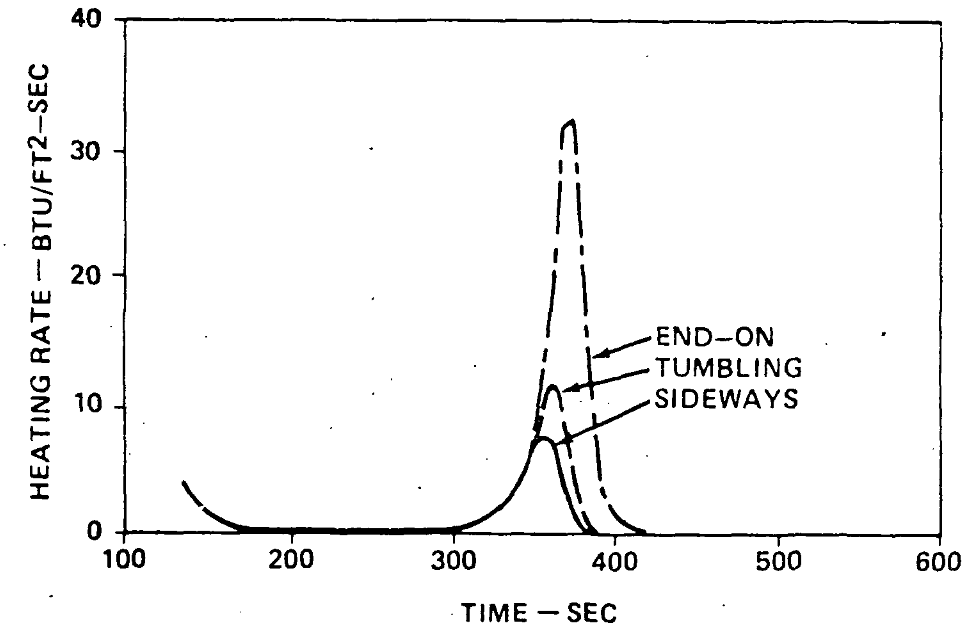


Figure G-2. HLLV single-engine booster reentry.



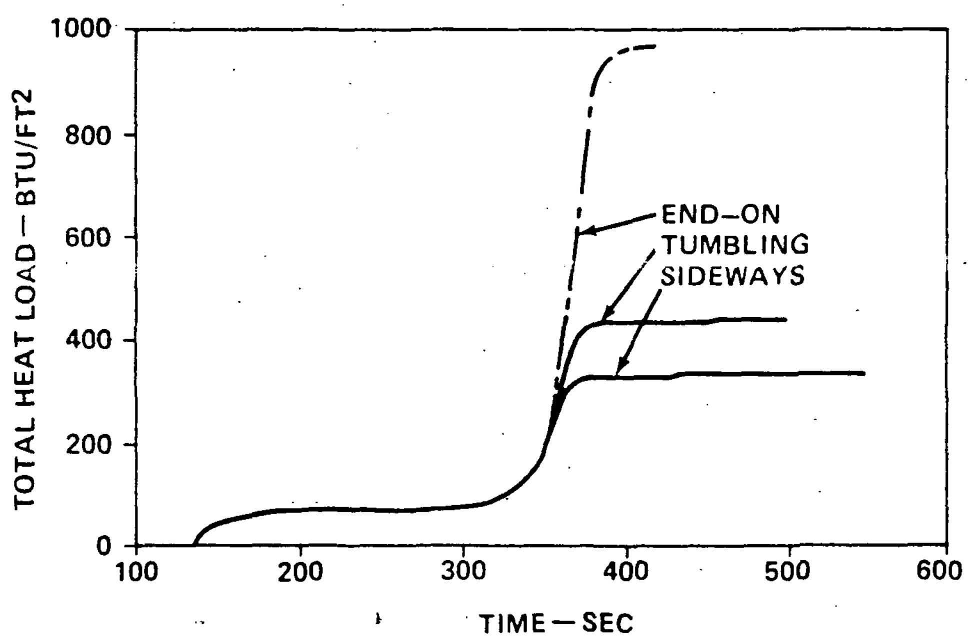


Figure G-2(a). HLLV single-engine booster reentry.

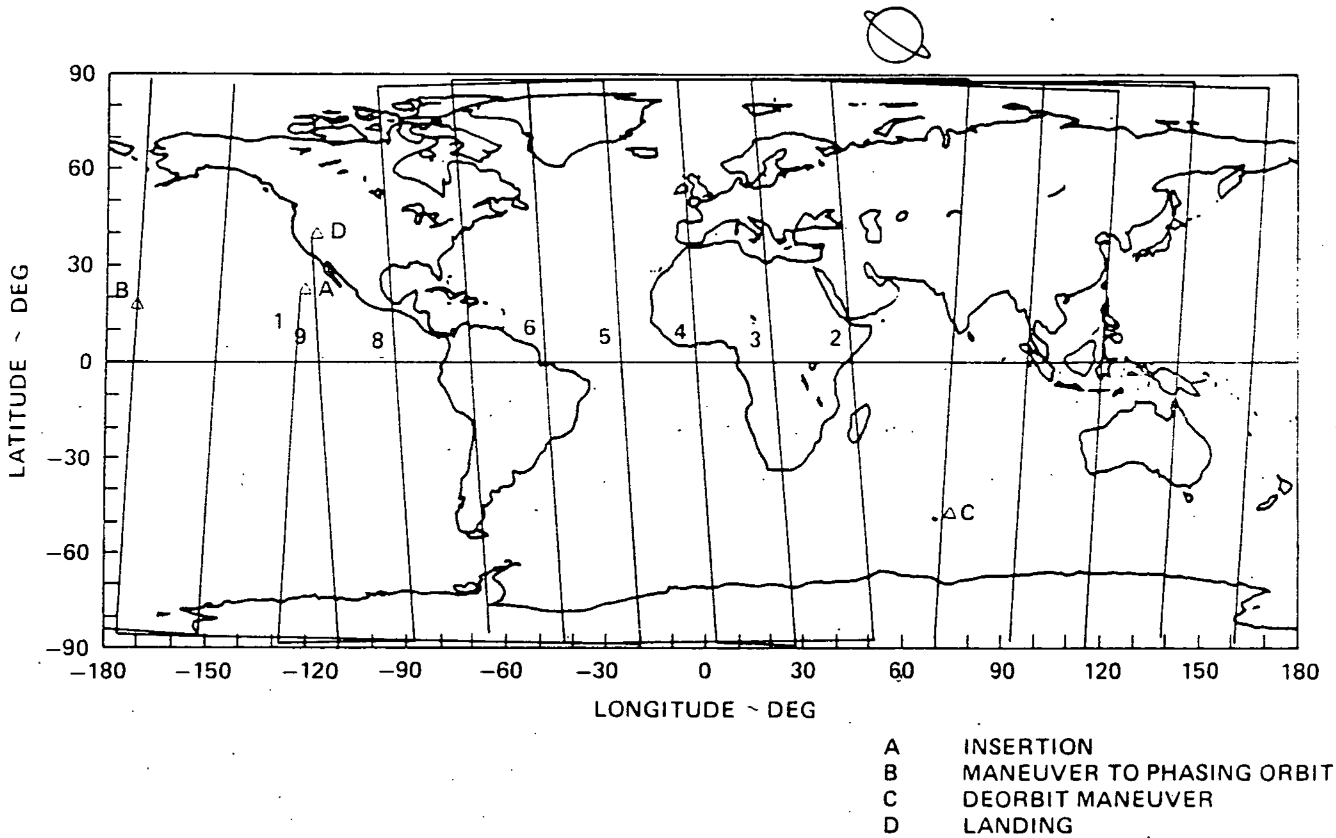


Figure G-3. HLLV P/A Module ground track.

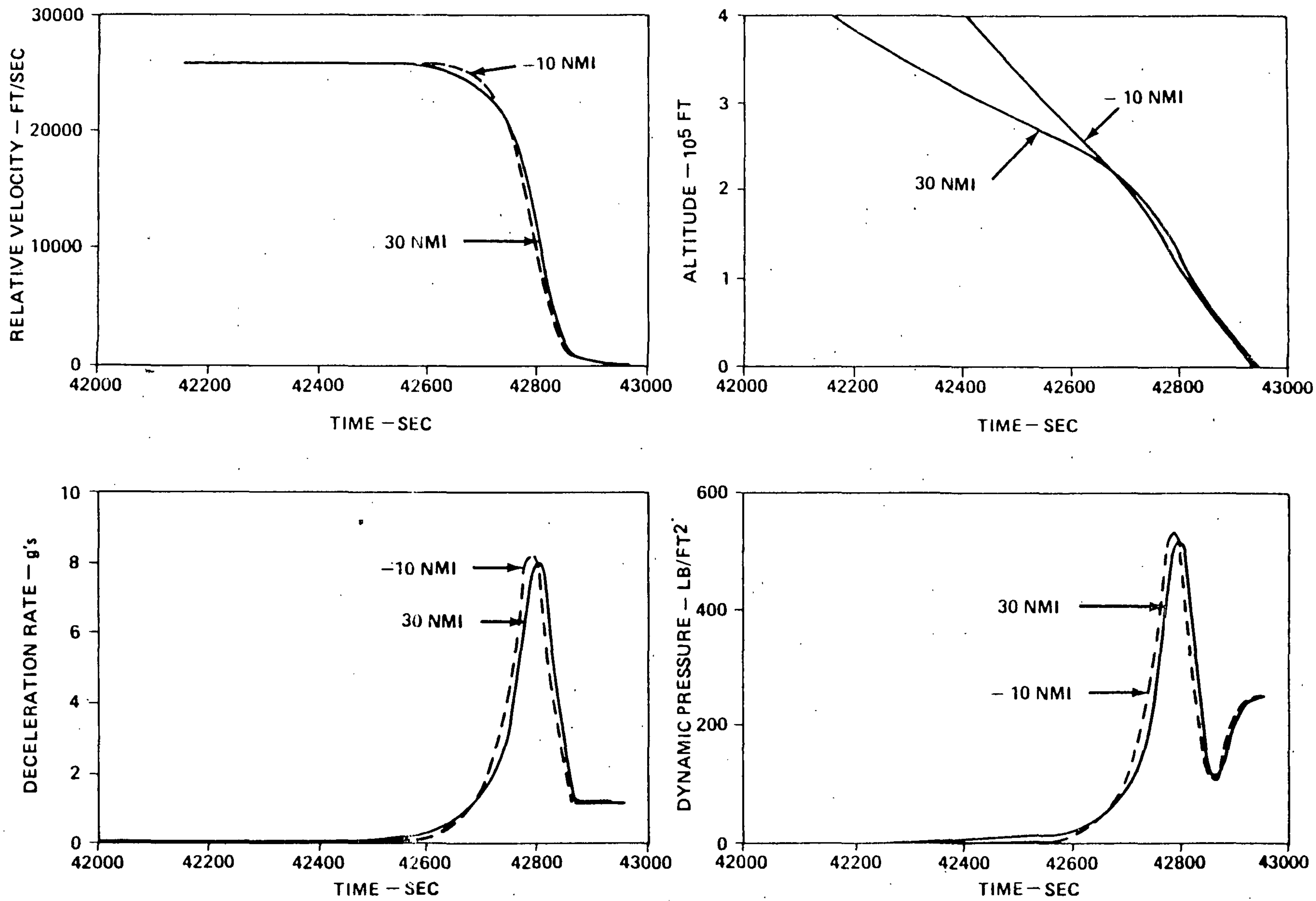
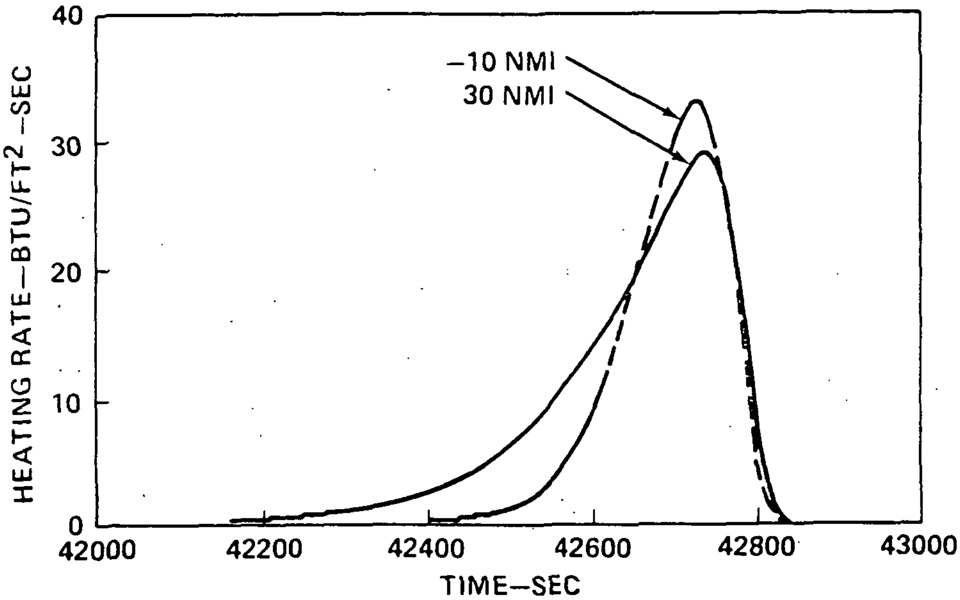


Figure G-4. HLLV P/A Module reentry targeted perigee variation.



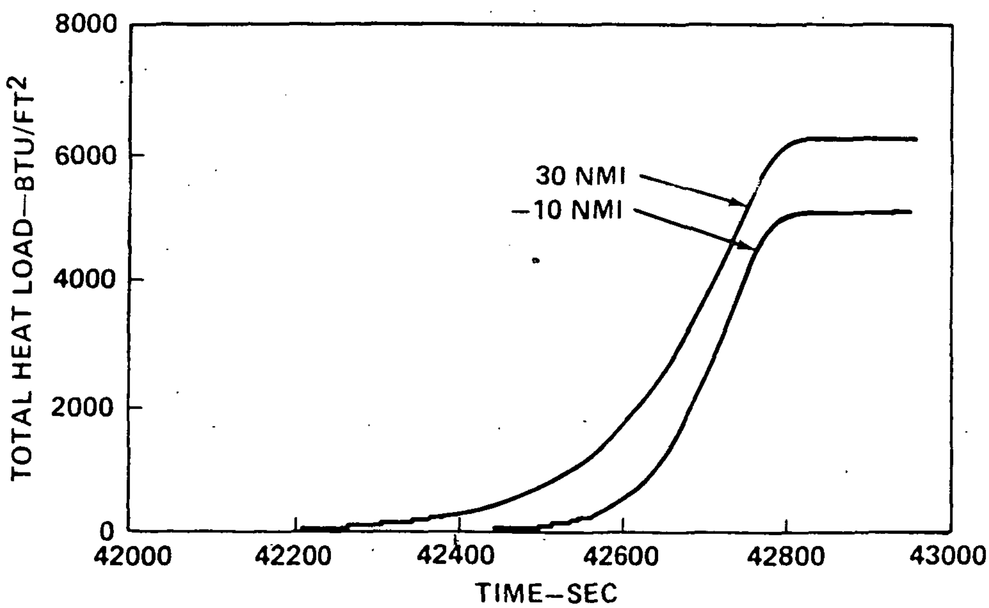
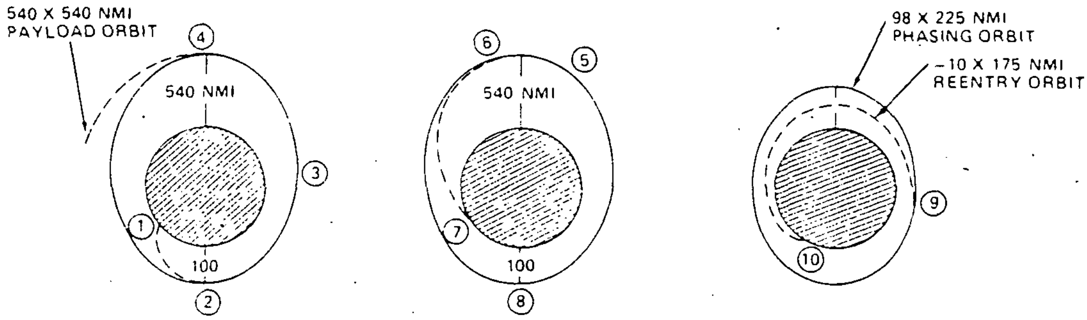


Figure G-4(a). HLLV P/A Module reentry targeted perigee variation.

TABLE G-1. MISSION PROFILE (INCLINATION = 90 deg)



EVENT	TIME ~ HRS
1 – LIFTOFF	0.000
2 – INJECT @ 100 X 540 NMI	0.146
3 – PAYLOAD SEPARATION	0.600*
4 – KICK STAGE CIRCULARIZES PAYLOAD @ 1st APOGEE	0.951
5 – SEPARATE P/A MODULE FROM STAGE 2	2.4*
6 – STAGE 2 DEORBIT NEAR 2ND APOGEE	2.7*
7 – STAGE 2 SPLASHDOWN	3.2*
8 – P/A MODULE PHASING BURN @ 2nd PERIGEE AFTER INSERTION	3.364
9 – P/A MODULE DEORBIT BURN	11.090
10 – P/A MODULE LANDING	11.933

*APPROXIMATE

TABLE G-2. HLLV BOOSTER REENTRY GROUNDRULES AND ASSUMPTIONS

- BALLISTIC REENTRY
- AERODYNAMICS CONTAINED IN APPENDIX A
- 175,831 POUNDS, TWO ENGINE BOOSTER WEIGHT
  118,062 POUNDS, SINGLE ENGINE BOOSTER WEIGHT
- BOOSTER SEPARATION CONDITIONS

ALTITUDE — FT	180,728
VELOCITY (REL) — FT/SEC	6,110
FLIGHT PATH ANGLE — DEG	30.56
LATITUDE — DEG	34.1
LONGITUDE — DEG	–120.7

- STAGNATION POINT HEATING RATE CALCULATIONS BASED ON 1.0 FT. RADIUS SPHERE
- TERMINAL LANDING DEVICE ANALYSIS NOT INCLUDED

TABLE G-3. HLLV BOOSTER REENTRY DATA

	TWO ENGINE BOOSTER			SINGLE ENGINE BOOSTER		
	SIDEWAYS	TUMBLING	END–ON	SIDEWAYS	TUMBLING	END–ON
MACH = 1 CONDITIONS						
*TIME – SEC	387	391	404	386	391	406
ALTITUDE – FT	65,147	52,981	18,210	65,968	53,104	9,884
IMPACT CONDITIONS**						
*TIME – SEC	547	500	426	550	501	413
VELOCITY – FPS	256	340	789	251	338	912
LATITUDE – DEG	30.8	30.7	30.6	30.8	30.7	30.5
LONGITUDE – DEG	–121	–121	–121	–121	–121	–121
RANGE – NMI	226	232	243	226	232	244
MAX. HEATING RATE – B/FT^2 – SEC	8.01	11.53	27.14	7.85	11.56	32.14
TOTAL HEATING – B/FT^2	346.8	445.6	843.1	341.2	445.9	978.9
MAX DYNAMIC PRES – LB/FT^2	287	544	2499	277	546	3518
MAX ACCELERATION (g's)	6.56	7.07	8.00	6.56	7.09	8.17

*FROM LIFTOFF
**NO PARACHUTES

TABLE G-4. P/A MODULE REENTRY GROUNDRULES

- **CONICAL P/A MODULE**
- **INSERTION ORBIT: 100 X 540 NMI/90°**
- **PHASING BURN @ SECOND PERIGEE AFTER INSERTION**
- **LAUNCH SITE: VAFB**
- **LANDING SITE: EAFB**
- **STAGNATION POINT HEATING RATE BASED ON 33 FT RADIUS SPHERE**
- **TERMINAL LANDING DEVICE ANALYSIS NOT INCLUDED**

APPENDIX H. STRUCTURAL ANALYSIS

The selected launch vehicle consists of a four-element booster and a core second stage. The booster is composed of two 171 in. tank diameter elements with one LOX/hydrocarbon engine and two 246-in. tank diameter elements with two LOX/Hydrocarbon engines. The second stage is a 396-in. diameter LOX/LH_2 stage powered by a recoverable Propulsion/Avionics (P/A) Module with five LOX/LH_2 engines. All engines are pad ignited. Propellants for the second stage burn are crossfed from the booster elements so the core stage is full when the first stage is jettisoned. Structural integration components are a forward adapter cone and booster attach ring with aft struts connecting the booster elements to the second stage.

A. Loads

The HLLV load characteristics were determined using trends and load analysis techniques developed on the Saturn and Space Shuttle programs. An early IOC was desired and new technology was not used in the design and materials selection. These may be studied using this design as a point of departure for improvements. A factor of safety of 1.25 was used on structural elements. Prelaunch on-pad, lift-off, maximum dynamic pressure (max q), max q times angle-of-attack (max q*alpha), maximum acceleration, staging, and P/A Module land impact recovery events were analyzed or considered in sizing the vehicle components. Structural design loads were generated at the trajectory point of max q*alpha assuming an aerodynamic angle-of-attack of 6 deg and all six booster engines gimballed 6 deg.

Aerodynamic Loads

The maximum shear loads were derived for the two worst conditions; first at lift-off prelease, and second at max q*alpha. The worst condition was determined to be the latter with a three sigma wind profile.

Maximum Bending Moment

The maximum bending moment distribution was calculated at the max q*alpha condition and based on aerodynamic load distribution with an angle-of-attack of 6 deg and all six booster engines gimballed 6 deg. The combination of these two separate loads yield the maximum limit structural bending moment at max q*alpha as shown in Figure H-1. The longitudinal acceleration history used for loads calculations is shown in Figure H-2.

The booster load paths are shown in Figures H-3 and H-4. The forward attachment, designed into the large adapter ring plus the aft attach struts, cause the bending loads to be reacted by the booster elements rather than the second stage. This penalty in booster weight is more than offset by the reduction in second stage tank weight.

B. Boosters

The single engine boosters are located in the pitch plane and will react a large percentage of the vehicle pitch and some of the yaw bending moments. The two engine boosters are mounted in the yaw plane, and react a large percentage of the yaw and a small amount of the pitch bending moments.

The booster forward adapters are unique structures and can be described as oblique cones, similar in shape to a "duckbill" (Fig. H-5). These are designed as aluminum skin and stringer shells that carry most of the loads in the stringers which are straight columns tied together by the skin and frames. The 50-ft diameter structural ring, that combines with the second stage adapter and the payload aerodynamic fairing base to integrate the components into a launch vehicle, is embedded in the "duckbill" section of each booster. Figure H-6 displays the forward integration hardware detailing the individual components and separation planes. The booster stage sections of the ring form a torque box/ring frame that separates from the upper section of the ring at booster burnout. The structural weight estimates for the boosters are detailed in Tables H-1 and H-2.

Propellant Tanks

The propellant tank walls must be designed to resist the sum of the following loads due to supporting mass, drag loads, bending compressive loads, engine thrust loads, and internal tank pressure loads. Using the worst case at max q*alpha, the required wall thickness and stiffness are established. All tanks are designed of integrally stiffened construction, 2219 aluminum alloy with stabilizing rings that serve as both prime structure and slosh baffles. The skin thickness is tapered according to pressure and compressive load requirements at all positions along the stage. The longerons and ribs are integral parts of numerically machined plates, formed to the correct radius and welded into the cylindrical sections. The welded ring frames are spaced for optimum stability.

The propellant tank domes are elliptical "square root of two" geometry. Except for the common bulkhead between the LOX and LH_2 tanks, all domes are made by mechanical and chemical milling of 2219 aluminum plate, formed into gore sections, and welded into a complete unit. The common bulkhead is of honeycomb construction with aluminum face sheets and a thermal barrier core. A phenolic core was selected for this design but trade studies may result in a more optimum material or concept. A "Y" ring is welded to the outer edge of each dome to allow connection to either the propellant tank wall or the intertank sections.

The intertank sections are non-pressurized load carrying cylindrical sections between tanks. They are designed as corrugated aluminum alloy with rings spaced for stability.

Thrust Structure

The thrust structure consists of aluminum box beams that support and react the engine thrust (Figs. H-7 and H-8). The thrust structure is supported by a cylindrical skin and stringer skirt with stabilizing ring frames. The stiffness was designed to minimize beam deflection during normal operation and engine out conditions. The aluminum cylindrical section and ring frames between the thrust structure and the fuel tank was sized to react the loads during all operating conditions, including transportation, on-pad stackup, prelaunch, and all flight environments. Attachments for the aft struts to react the kick loads to the second stage and posts for pad erection and holddown are located on the thrust structure ring (Figs. H-7, H-8, and H-9).

The booster aft skirt provides aerodynamic and thermal protection to the booster engine(s), thrust vector control system, and recovery system. Ring frames are added to provide shear support for the attachment of the aerodynamic fins.

Aerodynamic Fins

The aerodynamic fins are required to provide directional stability during atmospheric flight and reduce the required engine gimbal angle due to external disturbances. The all aluminum structure was sized for stiffness to react the worst wind shear experienced by any fin during ascent flight. Each fin is attached to the booster by 16 explosive bolts required for jettisoning from the booster after burnout. The structural weight estimates for the boosters are detailed in Tables H-1 and H-2.

C. Second Stage

The second stage consists of a 33-ft diameter LOX/LH_2 vehicle, nestled in the center of the four booster stages. It is composed of a conical forward adapter/payload attach ring, cryogenic propellant tanks, aft skirt, and a recoverable propulsion/avionics module.

Conical Adapter

The conical adapter is a structural transition between the 33-ft core stage to the 50-ft diameter booster attach ring/payload fairing adapter. The cone is an aluminum skin and stringer assembly with stabilizing ring frames. It is designed to take all first stage thrust tension loads minus the second stage thrust compressive loads, stage inertia loads, and bending moments. These loads are summed to establish the maximum conditions for both compressive and tension loads. The worst compression load condition will be during liftoff and the maximum tension load will occur at maximum booster acceleration. The adapter components are designed for the worst condition and therefore, can react any other flight loads.

The forward end of the conical adapter is a 50-ft ring which is an integral part of the payload fairing base as shown in Figure H-6. The lower half of this ring is formed from the booster forward adapters and the upper half from the payload attachment and aerodynamic shroud attachment. The upper half is a torque box, 25-in. high and 24-in. wide. The payload fairing is attached to the ring and is jettisoned by pyrotechnic devices during second stage flight.

Propellant Tanks

The propellant tanks of the second stage are 33-ft in diameter with a common bulkhead between the LOX/LH_2 (Fig. H-10). All domes are elliptical "square-root-of-two" geometry. The tanks are designed of 2219 aluminum alloy, integrally stiffened construction with stabilizing ring frames that also serve as slosh baffles. The basic membrane for both tanks tapers in thickness according to strength requirements. The longeron skin and ribs are numerically machined from thick plate, then formed to be welded into a cylinder.

The aft dome of the LH_2 tank is sized for the maximum ullage pressure of 24 psi plus the pressure head due to maximum longitudinal acceleration during booster burn. The common bulkhead is designed of honeycomb construction with aluminum face sheets and phenolic core for the thermal barrier. The LOX tank is designed for 27 psi ullage and the pressure head due to maximum longitudinal acceleration burn. "Y" rings are used to connect the tank sections with the forward and aft skirt sections.

The forward skirt is a 7-ft long cylinder of aluminum skin and stringer design with three stabilizing ring frames. It provides the transition between the LOX tank and the conical adapter.

Aft Skirt

The second stage aft skirt is a cylindrical section forming the transition between the LH_2 tank and the P/A module. It distributes all the P/A module longitudinal loads and the booster kick loads to the second stage (Fig. H-9). The booster strut attach ring is a welded aluminum assembly with four strut attach fittings. designed to withstand the maximum loads resulting from reaction forces of engine gimbaling and vehicle dynamics. These loads are introduced tangentially through the strut attachments.

The thrust loads from the P/A Module are transferred to the aft skirt at four hard points from the P/A Module thrust structure. These attach points must distribute thrust loads evenly to the second stage tanks. Table H-3 is a detailed structural weight summary of the second stage.

D. Propulsion/Avionics Module

The recoverable P/A Module is the heart of the HLLV. It houses the propulsion system composed of main LOX/LH_2 engines with two position nozzles, orbital maneuvering and attitude control engines (Appendix F), and major avionics systems (Appendix B). The geometric design of the P/A Module is a 33-ft diameter spherical dome with a 33-ft radius and a conical skirt which tapers to 41.7-ft diameter at its base (Fig. H-11). The forward section of the conical section attaches to the thrust structure.

The forward dome is designed as an aluminum honeycomb structure capable of withstanding the high aerodynamic pressure of reentry and absorbing the ground impact loads at landing. Several honeycomb face plates were investigated with different core thicknesses. It was determined, due to critical pressure, that the basic honeycomb core should be 5.2 in. thick with the outside face plate of 0.06 in. and the inner face plate 0.04 in. To reduce the nose cap weight, the honeycomb core was sized of two densities. To absorb the crushing loads of land impact, the center 10-ft diameter was designed using a density of 12 lb/ft^3. The remaining outer ring used a 6 lb/ft^3 density because of lighter loads. The spherical cap, including TPS, is considered expendable, i.e., removable from the basic structure and replaced after every flight.

The conical section was designed of isogrid construction to withstand the external reentry dynamic pressure. The thrust structure attaches directly to the forward conical section. Four thrust post fittings distribute the shear and compressive load to the second stage. These load attach points must be designed in such a way that they can be stowed and protected during reentry. Ring frames for stiffness have been designed into the lower portion of the conical section which could be used for attaching thermal curtains. Other main frame rings are located at the aft of the cruciform thrust structure to react the moments introduced by the engine gimbaling. OMS and RCS engines are mounted on removable panels located on each quadrant at the aft end of the conical skirt.

The thrust structure consists of all aluminum box beams that form a cruciform beam assembly that reacts the engine thrust and gimbal loads. The box beams are 52-in. high and 16-in. wide and designed for stiffness to minimize deflection. The aluminum box beam design was selected as a method of meeting an early IOC date, realizing that other materials and structure designs are available. Table H-4 summarizes the detailed structural weight summary of the P/A Module.

E. Aerodynamic Payload Fairing

The aerodynamic payload fairing (PLF) protects the payload during prelaunch and atmospheric flight. The PLF is jettisoned during second stage flight after the dynamic pressure is low enough to preclude damage to the payload from atmospheric heating. The fairing will have access ports for support services during ground processing. The aerodynamic, axial, bending, and shear loads were determined for several transient and quasi-steady-state conditions. The maximum loading occurs from the combination of these loads at max q*alpha. The maximum bending moment of 42 million ft-lb occurs at the fairing base.

Payload Fairing Adapter Ring

This 40-ft diameter ring is part of the second stage adapter ring. It consists of a bolted angle or machined flange that will be the base of the PLF. The fairing will be bolted or clamped to the base which will be the separation plane.

The basic cylindrical shell is 50 ft in diameter and 168-ft long as shown in Figure H-12. The shell is divided into seven longitudinal segments of 24 ft each, allowing for variable length payload fairings. The three forward segments are not heavily loaded. The limit bending moment at the forward segment is 4 million ft-lb. At the aft end of the third segment, the limit bending moment is 10 million ft-lb with a shear of 120,000 lb. The three forward segments are designed as aluminum honeycomb. The aft segments consist of aluminum skin and stringers with stabilizing ring frames. The forward segment honeycomb has face sheets 0.065 in. thick and a 2-in. thick aluminum core with a density of 2.5 lb/ft^3. The aft third segment has face sheets 0.090 in. and a core density of 3.5 lb/ft^3. The stabilizing rings for these segments are "Z" sections, 12-in. wide with 4-in. flanges, located every 4 ft. The four aft segments of the fairing are designed of aluminum skin and stringer construction with stabilizing ring frames. The aft segment is the most heavily loaded with a limit bending moment load of 42 million ft-lb at the base. The limit shear load at this location is 420,000 lb. The aft segment skin is 0.090-in. thick with stringer and rib spacing of 2 and 24 in. apart, respectively. Both the stringer and rib areas are 0.21 in.^{2} each. The stabilizing "hat section" ring frames will be located every 4 ft.

The PLF forward end is a double angle nose cone developed at Marshall Space Flight Center. The frustrum cone, 31.3 ft long with a 12.5 deg angle, is designed with a 2-in. thick honeycomb having 0.040-in. thick face sheets. The core density is 2 lb/ft^3. The shell is designed to withstand the external pressure introduced by shock waves. Four stabilizing rings are included in the frustrum design. The forward cone is 37.4-ft long with a 25 deg angle. The design is aluminum skin and stringer with stabilizing ring frames. The skin thickness is 0.042 in. and the stringers are at 2-in. spacing. The stringer required cross section area is 0.11 in.^{2} The rib spacing is 40 in. and has a required cross section area of 1.0 in.^{2}. Eight stabilizing rings are required. The stabilizing rings are "Z" sections 12-in. wide, having 34-in. flanges, spaced every 4 ft. Table H-5 lists the structural weight estimates for the PLF.

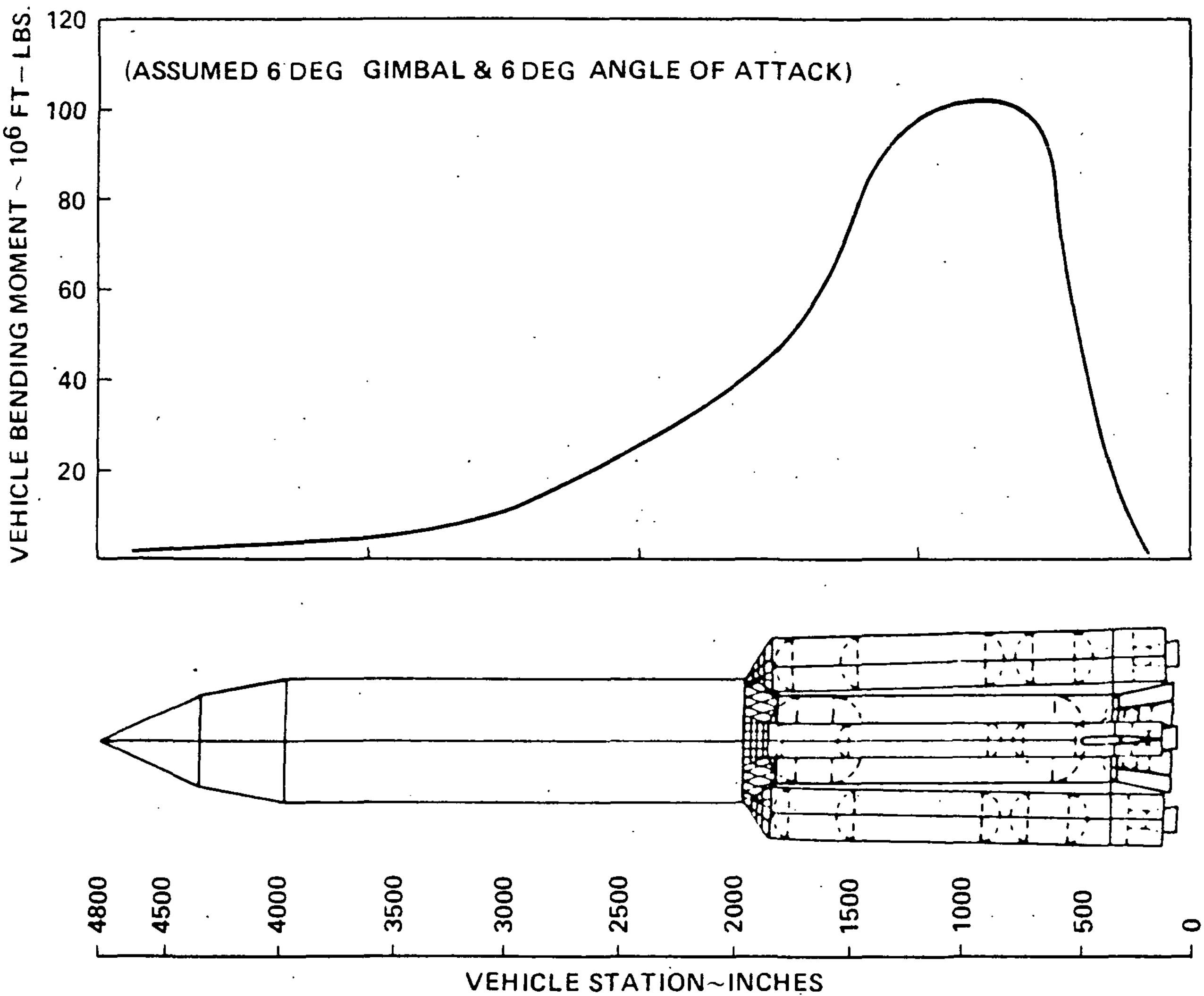


Figure H-1. Vehicle bending moment at maximum dynamic pressure.

FIRST STAGE	Ng
LIFTOFF	1.349
MAX q	1.96
MAX ACCELERATION	3.827
SECOND STAGE	
IGNITION	.826
MAX ACCELERATION	4.263

Figure H-2. Longitudinal acceleration factor for preliminary design.

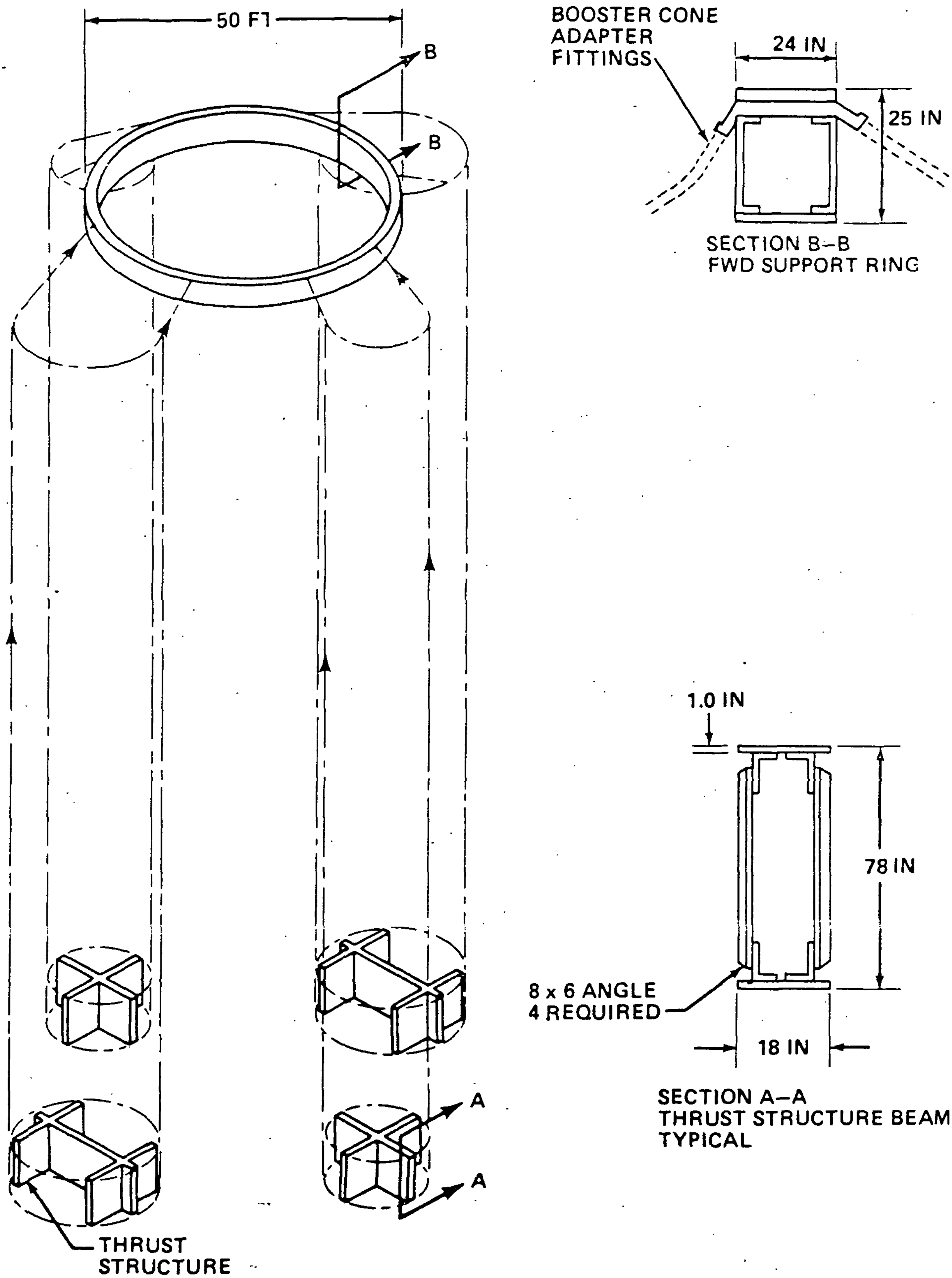


Figure H-3. First stage structural main frames and load paths.

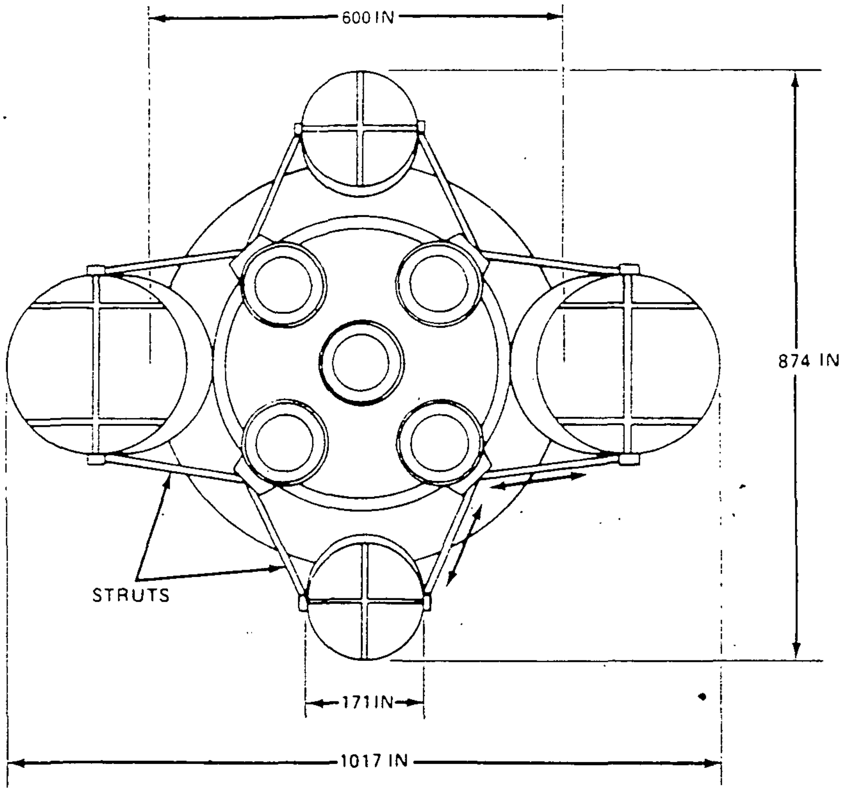


Figure H-4. Aft strut end view.

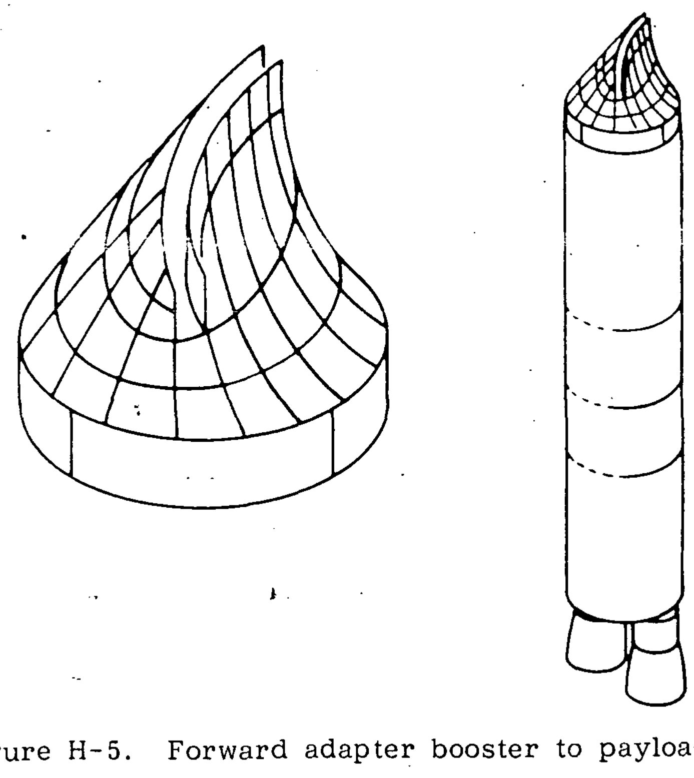


Figure H-5. Forward adapter booster to payload ring.

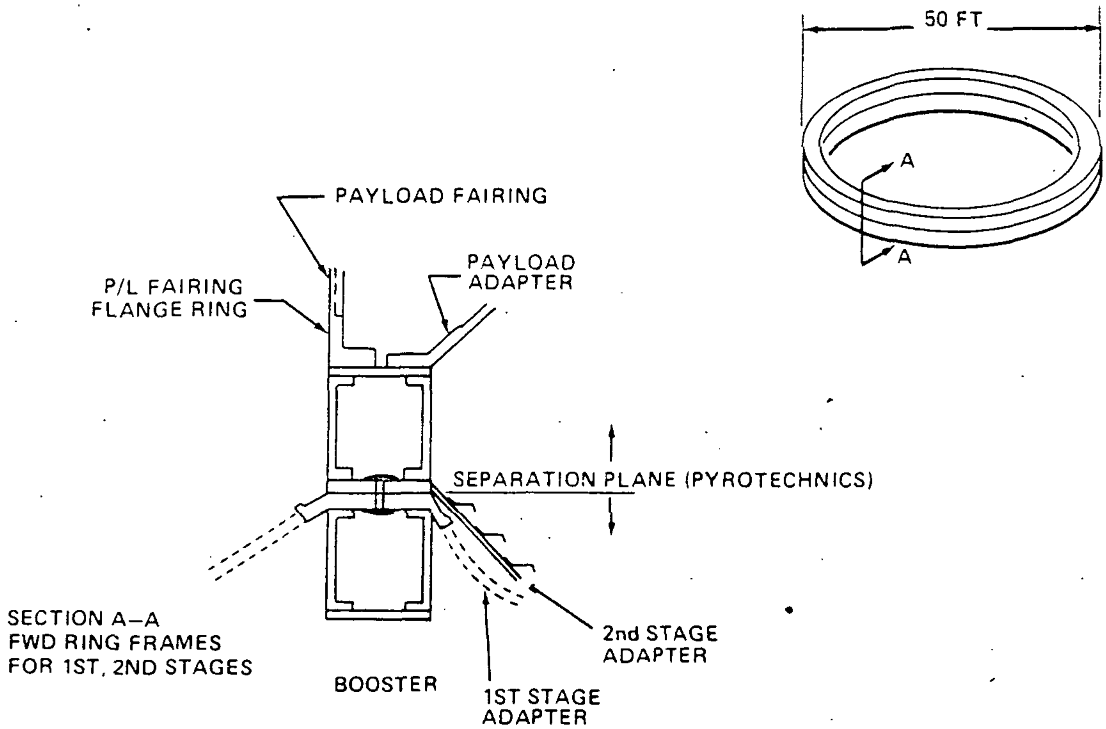


Figure H-6. Ring structural details.

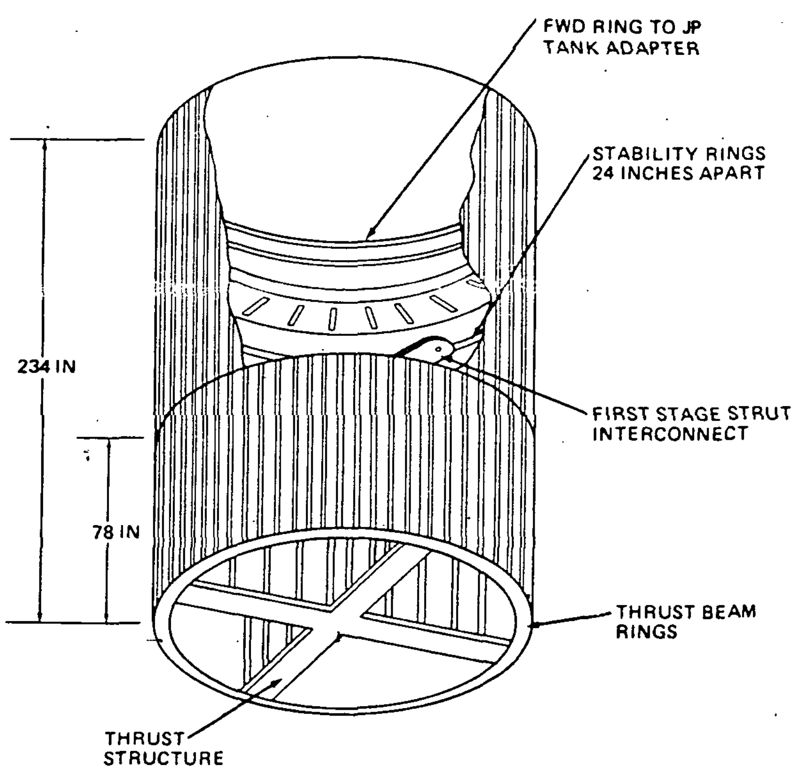


Figure H-7. Single engine booster structure and tank adapter skirt.

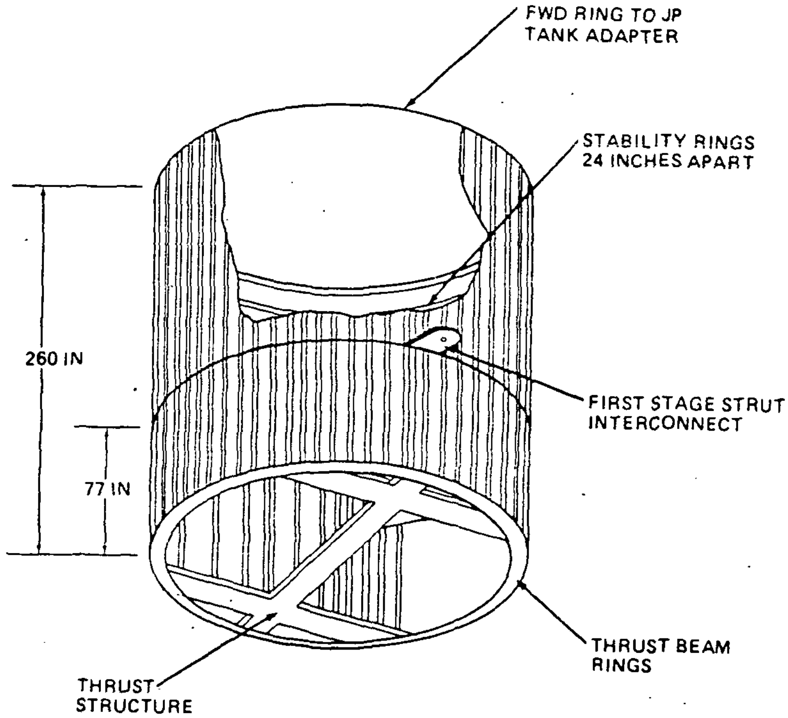


Figure H-8. Two engine booster structure and tank adapter skirt.

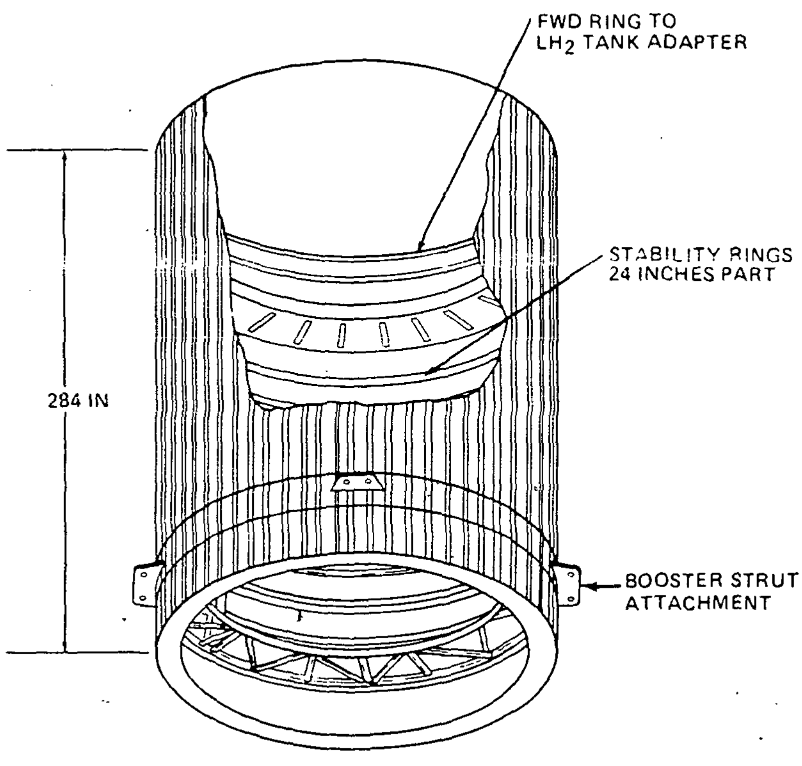


Figure H-9. Structural description of second stage aft adapter skirt.

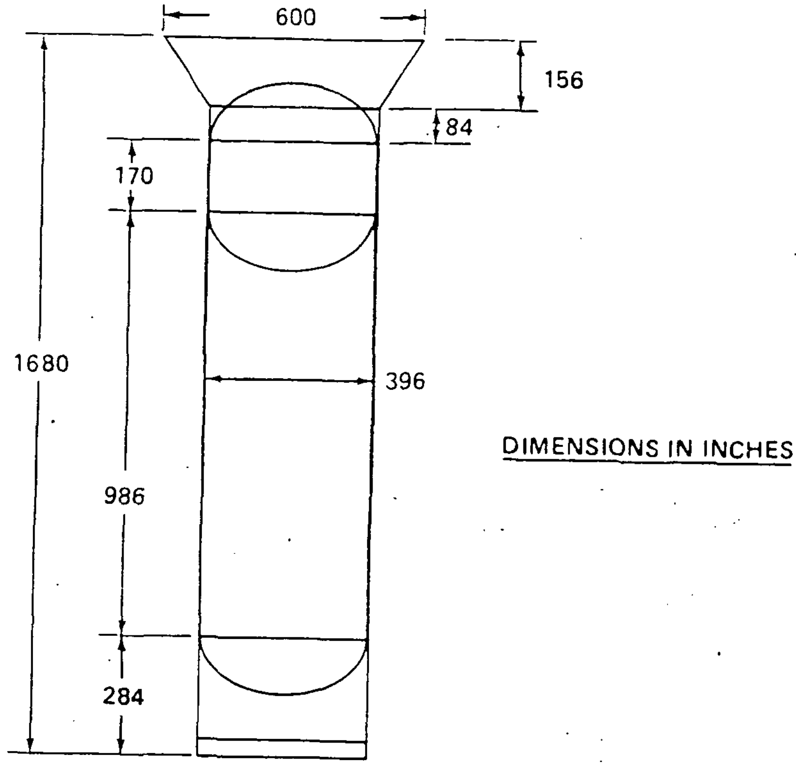


Figure H-10. Second stage configuration.

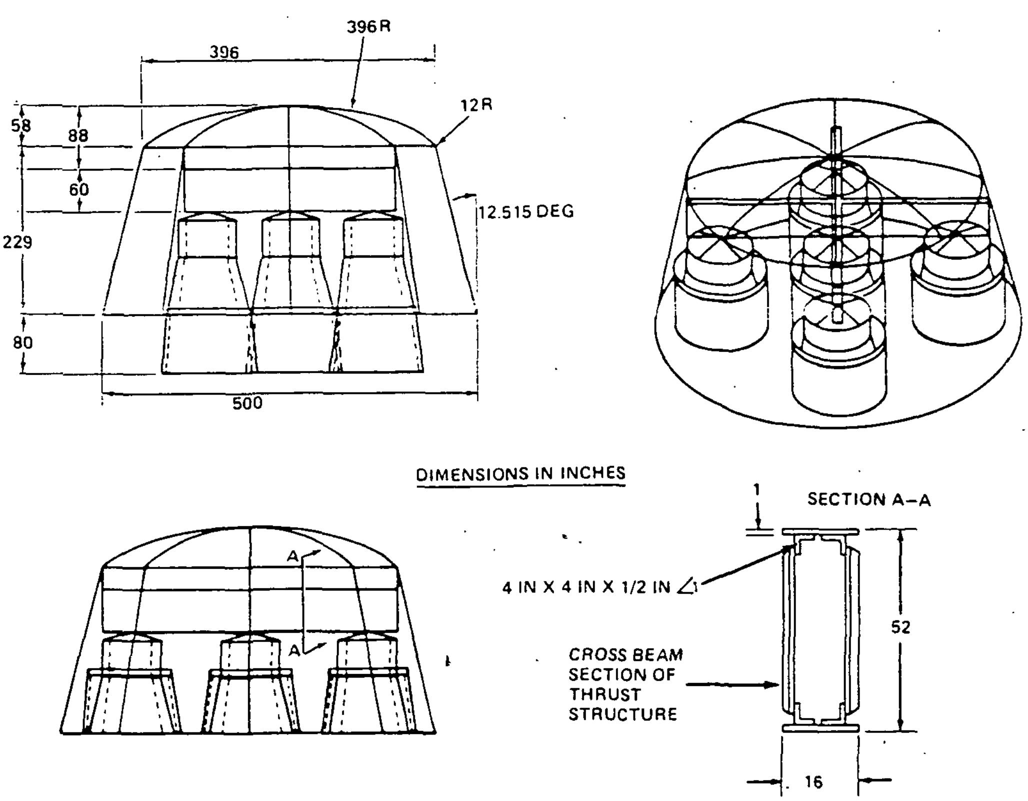


Figure H-11. Propulsion/avionics module.

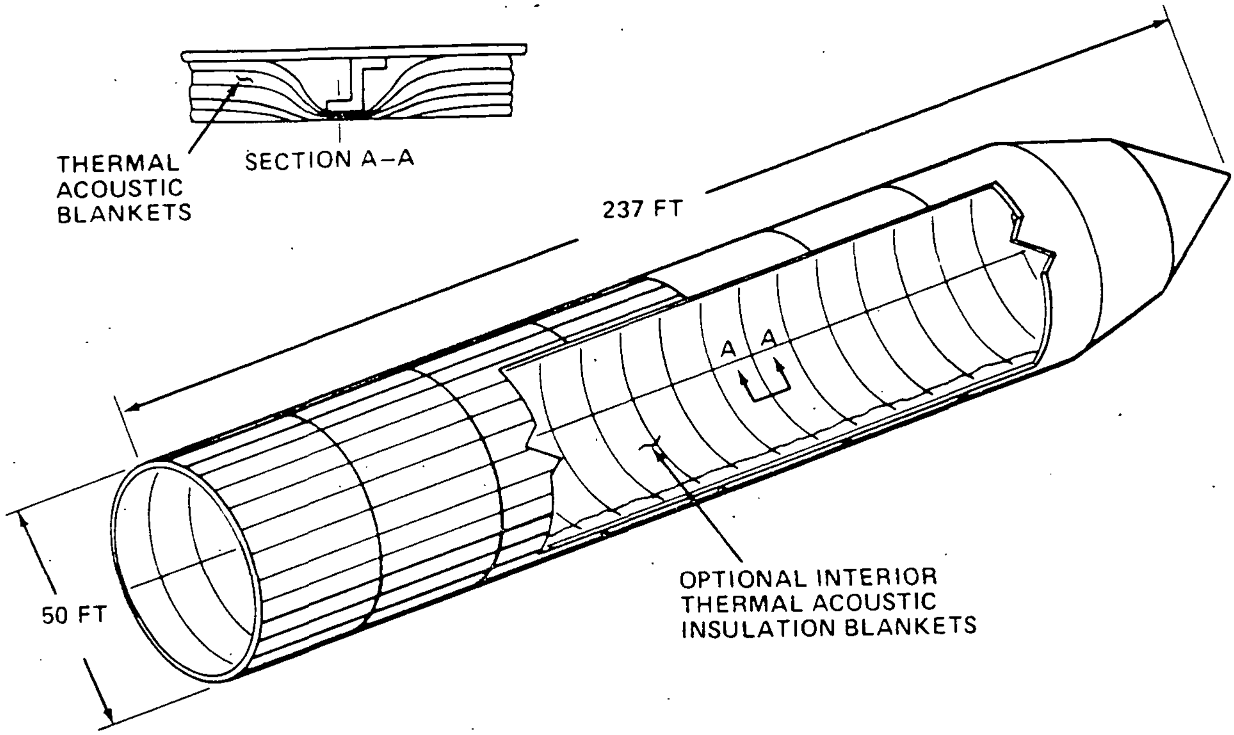


Figure H-12. Aerodynamic payload fairing.

TABLE H-1. ONE ENGINE BOOSTER STRUCTURAL COMPONENT WEIGHTS

ITEM	WEIGHT/LBS.
FORWARD SKIRT	5200
CYLINDRICAL SECTION SKIRT FWD	2100
LH_2 FWD DOME	540
LH_2 CYLINDRICAL DOME	7350
LH_2 COMMON DOME	720
LOX CYLINDRICAL DOME	16390
LOX AFT DOME	935
INTERTANK	5060
RP FWD DOME	540
RP CYLINDRICAL SECTION	4680
RP AFT DOME	640
AFT ATTACH. STRUTS	3000
AFT SKIRT CYLINDRICAL SECTION	6640
THRUST STRUCTURE	3300
AFT SKIRT	2680
FINS	10390

TABLE H-2. TWO ENGINE BOOSTER STRUCTURAL COMPONENT WEIGHTS

ITEM	WEIGHT ~ LBS.
FORWARD SKIRT (CONICAL SURFACE)	4465
CYLINDRICAL SECTION (FWD. SKIRT)	3790
LH_2 DOME	1042
LH_2 CYLINDRICAL SECTION	7065
LH_2/LOX COMMON DOME	1550
LOX CYLINDRICAL SECTOR	16352
LOX AFT DOME	2680
INTERTANK	7110
RP FWD DOME	1042
RP CYLINDRICAL SECTION	5280
RP AFT DOME	1620
AFT STRUTS	3000
CYLINDRICAL SECTION	8060
THRUST STRUCTURE	13136
AFT SKIRT	3880
FINS	10390

TABLE H-3. SECOND STAGE STRUCTURAL COMPONENT WEIGHTS

ITEM	WEIGHT ~ LBS.
P/L FAIRING ADAPTER RING	15520
STAGE PAYLOAD ADAPTER	10325
FWD SKIRT	5030
LOX FWD DOME	4505
LOX CYLINDRICAL SECTION	8085
LOX/LH_2 COMMON DOME	7970
LH_2 CYLINDRICAL SECTION	34100
LH_2 AFT DOME	3700
AFT SKIRT	17950
AFT ATTACH STRUTS FITTINGS	800

TABLE H-4. P/A MODULE STRUCTURAL COMPONENTS WEIGHTS

ITEM	WEIGHT – LB
THRUST BEAMS 52 X 16 IN BOX ALUMINUM	8960
DOME–SPHERICAL CAP 6" THICK ALUMINUM HONEYCOMB	4760
CONE – SKIRT ISOGRID ALUMINUM	5776
UPPER CONE RING	532
LOWER CONE RING	530
THRUST BEAM RING LOWER	712
THRUST BEAM UPPER	685
THRUST POSTS	252
BRACKETRY	300
FASTENERS & MISCELLANEOUS	100

TABLE H-5. PAYLOAD FAIRING STRUCTURAL COMPONENTS WEIGHTS

NO OF ITEM SEGMENTS	DESCRIPTION	TOTAL WEIGHT POUND
ONE/4	FWD CONE SEGMENTS	8970
	FWD CONE FRAMES	1220
ONE/4	FRUSTUM CONE SEGMENTS	9900
	FRUSTUM CONE FRAME	4800
FOUR/4	FWD CYLINDRICAL SEGMENTS	25070
	FWD CYLINDRICAL SEGMENT FRAMES	12800
FOUR/4	AFT CYLINDRICAL SEGMENTS	40360
	AFT CYLINDRICAL SEGMENTS FRAMES	18800

APPENDIX I. TESTS

Test Program

The major details of the test program are shown graphically in Figures I-1 through I-6. Verbal discussion on upper level structure of program is given in the following paragraphs.

Structural Testing

The building block approach is to be used for the HLLV structural test program. The smaller module such as the small booster LH_2 and LO_2 tanks, RP tank, and interstage structure will be tested. The LH_2 and LO_2 tanks will be tested using cryogenics; after this, all the pieces will be assembled into a small booster and used for the all-up cryogenic test and the all-up dynamic tests. The same sequence of events will follow for the medium booster and the core stage as well as the other major structural items.

Propulsion Testing

The propulsion test program may be broken down into two phases — Engine Development Testing and Static Engine Firing Testing. In the first phase, all of the subassemblies of the engine are developed and tested (i.e., turbo pumps, gas generators, etc.). In the second phase, the assembled engines are fired as a single engines, or multiple engines based on intended use. Also, the engines singularly and clustered are fired in the expected environment (sea level or altitude, or both). The engines are then integrated with the structural test items to form an all-up vehicle which will then be static fired.

Flight Control Test

Flight control equipment is tested in two phases. The first phase is development testing of individual items of equipment and also testing at the systems level. The second phase is all-up systems level verification where the equipment is tested to its outer limits and, where possible, to designed life. The second phase could be from 75 to 250 flight simulations.

Avionics Testing

The objective of avionics testing is to assure that testing of critical items at all levels of assembly is sufficient to validate the design approach. The majority of avionics tests are started at the component and breadboard levels. Then, using a building-block approach, the testing is carried to all levels (board, box, subsystem, and system). The process ends with the testing necessary to integrate the system avionics with the vehicle.

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Separation Testing

Because of the complexity and size of the components that are to be separated on the HLLV, a number of tests will be required to assure its success. The pyrotechnic components must be verified and the shock loads introduced by them must be designed for them. The separation motors must be tested in an all-up configuration. The separation of the aerodynamic shroud, because of its size, cannot be tested in any facility presently available. This test might best be accomplished on a test flight.

Recovery Testing

Because of the staging velocity of the boosters and their size, it is not possible to conduct a full-scale test of their recovery system. Testing will have to be done on modules and possibly on expendable rocket vehicles. This is also true of the P/A Module and the shroud; however, it appears that a scaled-down P/A Module could be tested on a STS flight.

Thermal Protection Testing

Samples of proposed materials would first be tested in the expected environments simulated on the ground. The scale modules with the materials applied would be subjected to expected environments as well. In the case of the P/A Module, a flight demonstration from Shuttle could proof the TPS and their recovery system in the same test.

Software Testing

Software testing is divided into four phases — code and debug, verification, validation, and systems integration. After the software is coded, testing begins. This level of testing is commonly referred to as debugging. Next, a group independent of the coding group performs verification listing on the software. The software is checked in a simulation facility which simulates a closed-loop system using as much system or protoflight hardware as feasible. The validation tests are performed with maximum system hardware possible, and the emphasis is on system/software compatibility. The systems integration is performed on an all-up systems facility or on a protoflight vehicle.

Ground Testing

Ground testing are those tests necessary to prove that an item is completely functional. Each booster, core stage, and P/A Module will be tested at completion of assembly and then stored until use. The items will be stored with power on and, periodically, the onboard computers will run a redundant test to assure the item is functional and ready for flight. If a failure occurs, the on-board system will alert the ground personnel and will be repaired.

Preflight Testing

When the HLLV is assembled on the launch transporter, a complete check will be made by the on-board test system. This test will primarily be an interface check between the four boosters, the core stage, and the P/A Module. While at the launch position, the onboard system will perform periodic checks to determine flight readiness.

Facilities Testing

Facilities must be fit checked using a protoflight vehicle before use. Also, all fluid and signal interfaces must be checked and corrected if wrong The final check is that of the special test equipment and the supporting software.

STRUCTURAL

STRUCTURAL LOAD TEST

- SMALL BOOSTER LH_2 & LO_2 TANK
- SMALL BOOSTER RP TANK
- SMALL BOOSTER INTERSTAGE ADAPTER
- MEDIUM BOOSTER LH_2 & LO_2 TANK
- MEDIUM BOOSTER RP TANK
- MEDIUM BOOSTER INTERSTAGE ADAPTER
- P/A MODULE
- PAYLOAD ADAPTER
- PAYLOAD FAIRING

CRYOGENIC STRUCTURAL TEST

- SMALL BOOSTER LH_2 & LO_2 TANK WITH LH_2
- MEDIUM BOOSTER LH_2 & LO_2 TANK WITH LH_2
- CORE STAGE LH_2 & LO_2 TANKS WITH LH_2
- ALL-UP LH_2 & LO_2 TANKS WITH LH_2

DYNAMIC STRUCTURAL TEST

- ALL—UP WITH PAYLOAD FAIRING AND PAYLOAD MODULE

(2 SMALL BOOSTERS, 2 MEDIUM BOOSTERS, 1 CORE STAGE, 1 P/A MODULE, 1 PAYLOAD ADAPTER)

Figure I-1. HLLV test program required tests.

REQUIRED TESTS

- PROPULSION
  - BOOSTER ENGINE DEVELOPMENT TESTS
    - TURBO PUMP TEST
    - GAS GENERATOR TEST
    - COOLING AND COKING TEST
    - THRUST CHAMBER CALORIMETER TEST
    - INJECTOR TEST
    - LOX COOLING TEST
    - IGNITION TEST
    - NOZZLE INTEGRITY TEST
    - FULL SCALE TEST
      - LIFE CYCLE TEST
      - THROTTLING TEST
  - STATIC ENGINE FIRING TEST
    - SMALL BOOSTER FIRING TEST (1 ENGINE)
    - MEDIUM BOOSTER FIRING TEST (2 ENGINES)
    - P/A MODULE STATIC FIRING TEST (5 ENGINES) SEA LEVEL
    - P/A MODULE STATIC FIRING TEST (5 ENGINES) ALTITUDE
    - CORE STAGE AND P/A MODULE STATIC FIRING (5 ENGINES)
    - SEA LEVEL
    - CORE STAGE AND P/A MODULE STATIC FIRING (5 ENGINES)
    - ALTITUDE
    - ALL-UP STATIC FIRING (11 ENGINES) SEA LEVEL
    - ALL-UP STATIC FIRING (5 ENGINES) ALTITUDE

Figure I-2. HLLV test program.

FLIGHT CONTROL

- FLIGHT CONTROL DEVELOPMENT TEST
  - HYDRAULIC PUMP TEST
  - SERVO-ACTUATOR TEST
    - SERVO-VALVE TEST
    - FLUID SHARING TEST
  - ACCUMULATOR OR BOOST PUMP TEST
  - FILTER TEST
  - HYDRAULIC MANIFOLD TEST
  - TURBINE DRIVE TEST
  - FLUID COOLER TEST
  - SYSTEM ALL-UP TEST
- FLIGHT CONTROL VERIFICATION TEST
  - ALL-UP TEST USING WORST CASE CONDITIONS (75-250 TESTS)

AVIONICS

- COMPONENT TESTING
- DEVELOPMENT ARTICLE TESTING
- VERIFICATION TESTING
- BOX LEVEL TESTING
- SUBSYSTEM TESTING
- SYSTEM LEVEL TESTING
- SYSTEM INTEGRATION TESTING
- PRODUCTION TESTING

Figure I-3. HLLV test program required tests.

REQUIRED TESTS

- SEPARATION
  - SEPARATION SYSTEM TEST
    - SMALL BOOSTER SEPARATION TEST
    - MEDIUM BOOSTER SEPARATION TEST
    - SHROUD SEPARATION TEST
    - PAYLOAD SEPARTION TEST
    - P/A MODULE SEPARATION TEST
- RECOVERY
  - RECOVERY SYSTEMS TESTS
    - SMALL BOOSTER RECOVERY TEST
    - MEDIUM BOOSTER RECOVERY TEST
    - SHROUD RECOVERY TEST
    - (POSSIBLE FLIGHT DEMONSTRATION SUBSCALE)
    - P/A MODULE RECOVERY TEST
    - (POSSIBLE FLIGHT DEMONSTRATION SUBSCALE)
- THERMAL PROTECTION
  - THERMAL PROTECTION SYSTEM TEST

    MATERIAL SAMPLE TEST
    SUBSCALE MODULE TEST
    RE-ENTRY TEST
    (FLIGHT DEMONSTRATION SUBSCALE)

Figure I-4. HLLV test program.

SOFTWARE

SOFTWARE TEST

- DEBUG AT CODING LEVEL
- VERIFICATION TEST
- VALIDATION TEST
- SYSTEMS INTEGRATION TEST

GROUND

GROUND TEST

- COMPONENT TEST
- BOX LEVEL TEST OR ITEM LEVEL TEST
- SUBSYSTEM TEST
- SYSTEMS TEST
- INTEGRATED SYSTEMS TEST
- STAND-BY REPEAT TESTING

PREFLIGHT TEST

PREFLIGHT TEST

- ALL SYSTEMS TEST
- REDUNDANCY TEST

Figure I-5. HLLV test program required tests.

REQUIRED TEST

- FACILITIES
  - FACILITIES TEST
    - CHECK TO DESIGN DRAWING AND SPECIFICATIONS
    - FIT CHECK WITH HARDWARE
    - CHECKOUT OF STS AND SOFTWARE
    - INTERFACE TEST
    - VEHICLE CHECKOUTS
    - SOFTWARE OPERATIONS AND MAINTENANCE

Figure I-6. HLLV test program.

APPENDIX J. THERMAL ANALYSIS

HLLV Thermal Protection System

Thermal Protection System (TPS) design for the propellant tanks, protuberant surfaces, and heat shield was based on the Shuttle and Saturn vehicle programs. Prelaunch and ascent flight environments were assessed. An extensive analysis was performed to define the re-entry environment and TPS requirements for the P/A Module. A summary of the HLLV TPS is shown on Figure J-1.

The liquid hydrogen tanks must be insulated to limit the boil-off rate while on the pad (Fig. J-2). The LOX tanks prelaunch boil-off rate is acceptable without insulation. The P/A Module skirt TPS is a durable material that will not be damaged due to ice falling off the LOX tank at lift-off.

The Payload Fairing (PLF) nosecone and the booster forward adapter will be protected from aerodynamic heating during ascent by an application of Marshall Sprayable Ablator (MSA). The thickness of the MSA on the PLF nosecone will average about 0.5 in. On the booster forward adapter the thickness ranges from 0.125 to 0.250 in. The booster protection is estimated at 412 lb. The large fins will not be thermally protected.

The tail section of the boosters and P/A Module must be protected from the rocket exhaust heat during ascent. The protection will be provided primarily by heat shields located at the engine gimbal planes. The heat shields can be made of either titanium or stainless steel honeycomb with insulation on the exposed surface. The engines can be protected by wrapping them with blankets consisting of fibrous silica insulation enclosed in inconel foil, except for the nozzles where wire reinforced asbestos may be used. Heat shield penetrations for fuel and oxidizer lines must have curtains that allow the engines to gimbal while preventing flow-through of gases in the tail section.

Aft skirts on all the boosters will require localized applications of ablator cork and phenolic glass, the phenolic glass being applied at the lower extremeties of the skirts near the engine exhaust. Detailed analyses will be required to determine material thicknesses and weights. The amount of protection required will depend on the location of the skirts relative to the engine exhaust.

P/A Module Re-Entry

The design approach for the P/A Module TPS was to use a state-of-the-art inexpensive ablator material that would be replaced after each flight separately from the P/A Module refurbishment. A cork ablative material was selected for this purpose. The cork thickness was optimized so that the aluminum back-up structure temperature did not exceed 300°F during re-entry. A Langley Research Center update of the MINIVER aeroheating computer program (LANMIN) was used to estimate the thermal environments. A heat transfer subroutine of the program (EXITS) was used to calculate the material ablation and temperature histories.

Two P/A Module descent trajectories were used to generate the maximum expected envelope for the re-entry aerothermal environments. One was a targeted perigee of 30 miles and the other a perigee altitude of -10 miles. Starting at the entry interface altitude of 400,000 ft, altitude versus time is shown on Figure J-3. Thermal environments for each trajectory were calculated for three different regions on the P/A Module — the forward dome, upper conical sidewall, and lower conical sidewall (Fig. J-4). Shown on Figure J-5 are the re-entry heating rates versus time. Figure J-6 shows the heat loads versus time.

The trajectory with the slower descent rate requires the greatest insulation thickness due to the higher total heat load. Temperature versus time at specified insulation depths are shown for the three P/A Module regions on Figures J-7, J-8, and J-9. The cork ablation temperature is 760°F. The material at increasing depths goes to 760°F as the ablation proceeds with time. To maintain the structure temperature to 300°F or less, the forward dome, upper cone sidewall, and lower cone sidewall insulation thicknesses were 0.4, 0.25, and 0.2 in., respectively.

Temperature gradients through the cork ablator at specified times during the descent are shown on Figures J-10, J-11, and J-12. At each successive time, the surface temperature is shown to be at increasing depths in the cork, indicating the amount of material that has ablated. The last gradient is shown at a time near the end of the descent. Ablation has ceased and the surface temperature has dropped below the 760°F ablation temperature. The temperature increases with depth, showing that the inside of the material is warmer than the external surfaces. The backside temperature (structure) is 300°F. The ablator thicknesses for the three P/A Module regions are shown as a function of descent time on Figure J-13.

Avionics Thermal Control System (TCS)

Avionics and power systems for the HLLV will be located in the P/A Module. The TCS was sized to accommodate a 6 kW bus load and a 4 kW waste heat from the fuel cells. The equipment, with the exception of the fuel cells, was assumed mounted on cold plates. The internally cooled fuel cells would be directly connected into the TCS coolant (Freon) loop. A schematic for the TCS coolant loop is shown on Figure J-14. To minimize pump power and provide constant cold plate fluid inlet temperatures, the cold plates were placed in parallel. The fuel cells operate at the highest temperature range. Therefore, the fuel cells were placed downstream of the component cold plates.

A temperature rise of 40°F was assumed for calculating the coolant flow rate and pump power. A flash evaporator serves as a sink for rejecting the heat. A water unit of the type used on the Orbiter would be more than adequate. The flash evaporator would be effective at altitudes above 100,000 ft. A GSE heat exchanger would cool the equipment before launch. Little power is expected to be consumed below the 100,000 ft re-entry level. Should the requirement exist for cooling at times the GSE heat exchanger or water flash evaporator are not available, an ammonia flash evaporator could be added. A list of the TCS equipment with weight, volume and power is shown on Table J-1.

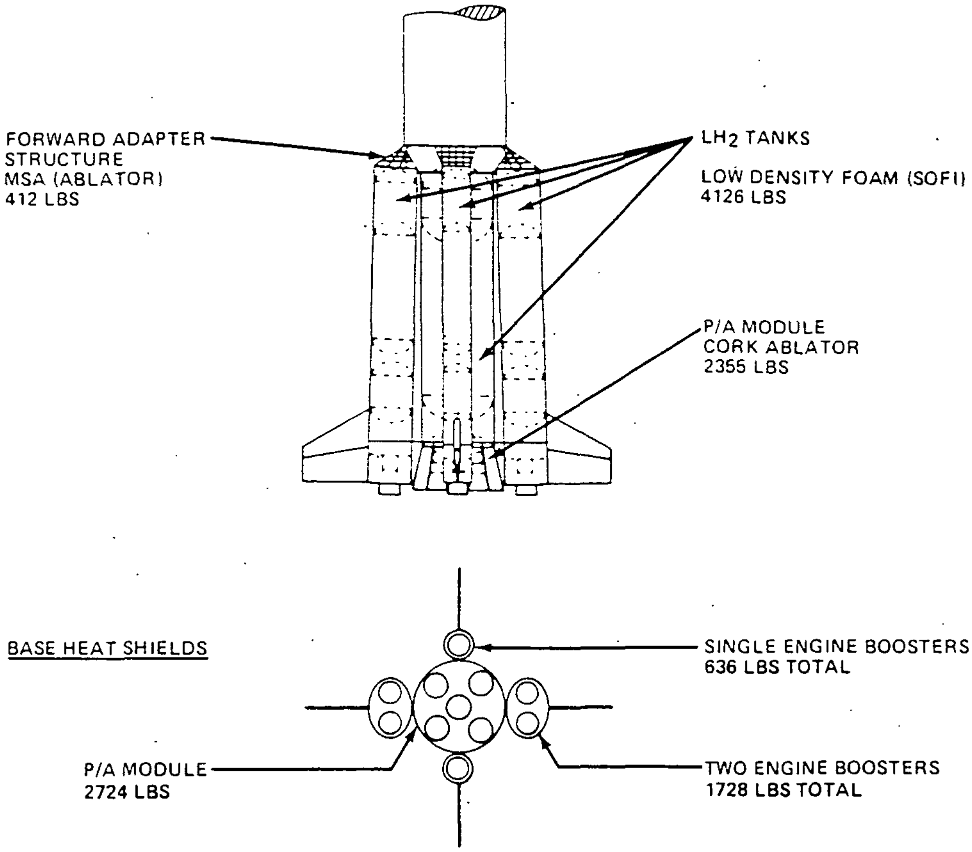

Figure J-1. TPS summary.

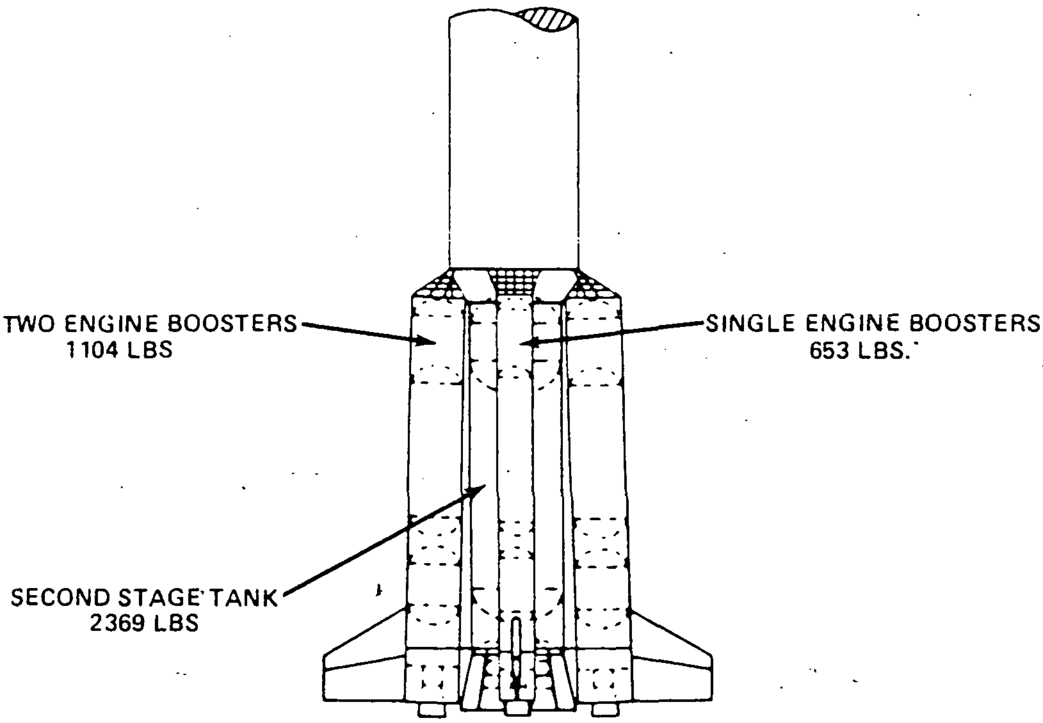

Figure J-2. LH_2 tanks insulation.

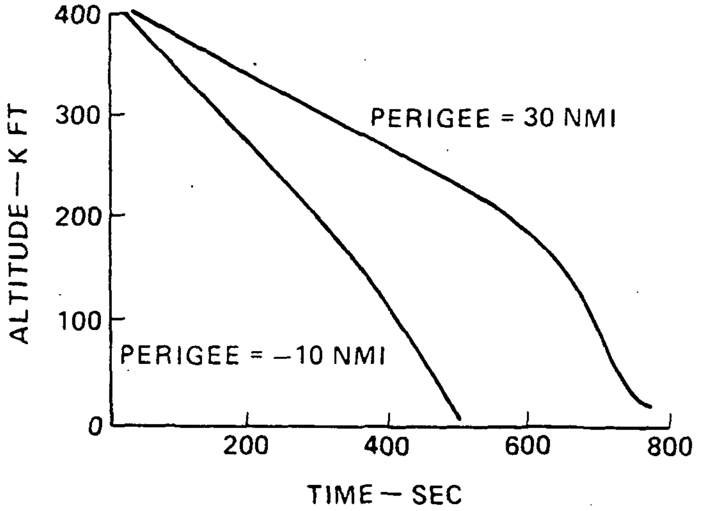

Figure J-3. P/A Module reentry trajectory.

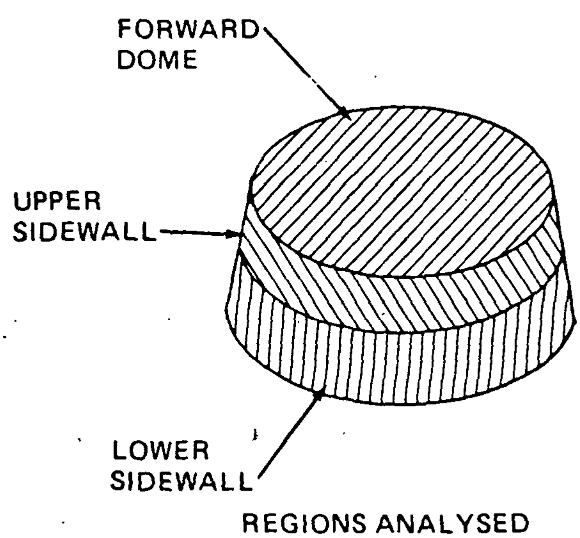

Figure J-4. P/A Module configuration.

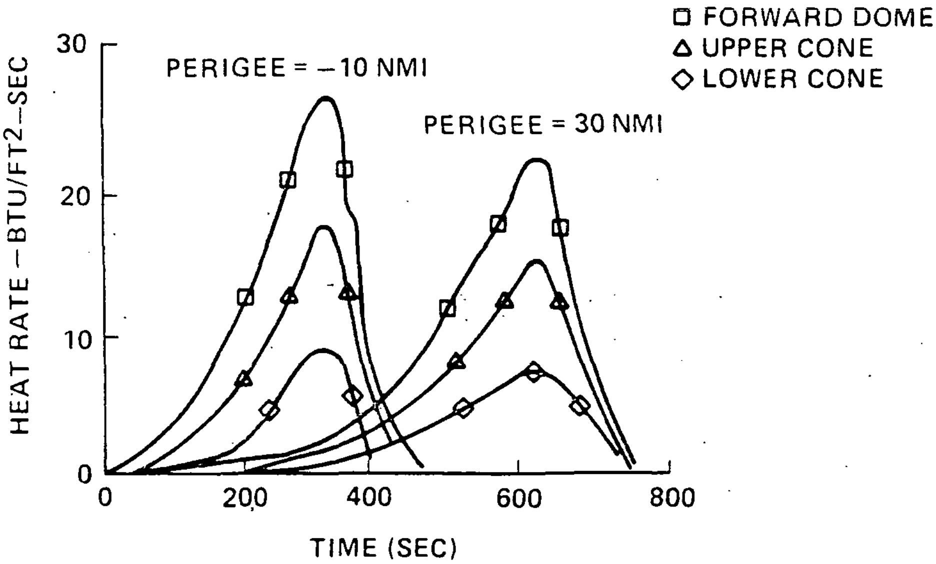

Figure J-5. P/A Module reentry heat rates.

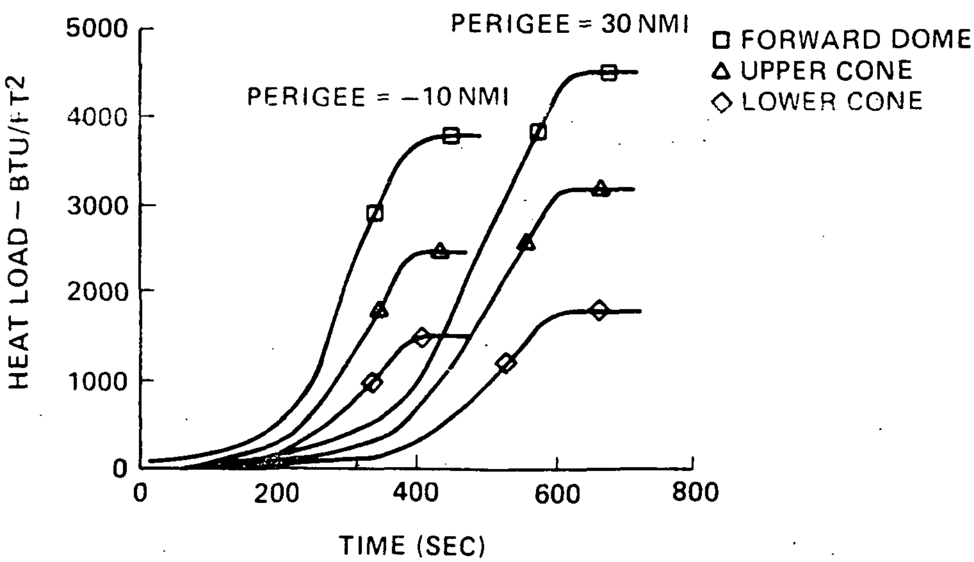

Figure J-6. P/A Module reentry heat loads.

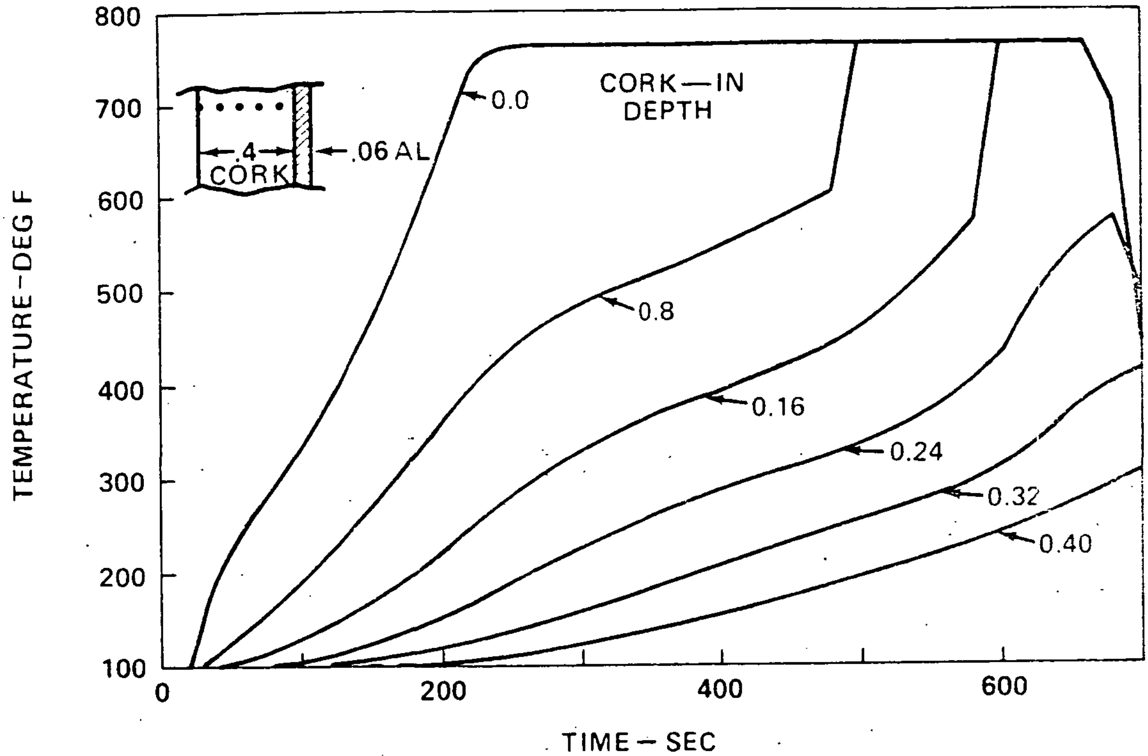

Figure J-7. Forward dome ablator temperature history for P/A Module reentry.

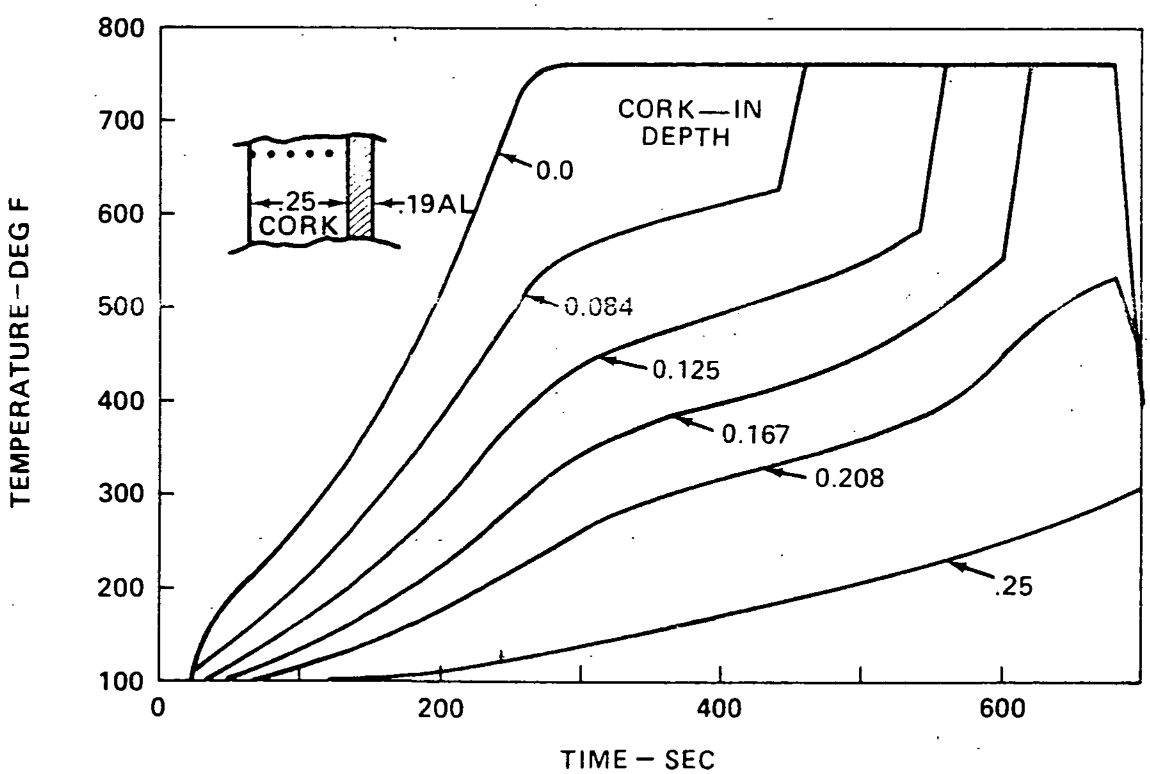

Figure J-8. Upper cone ablator temperature history for P/A Module reentry.

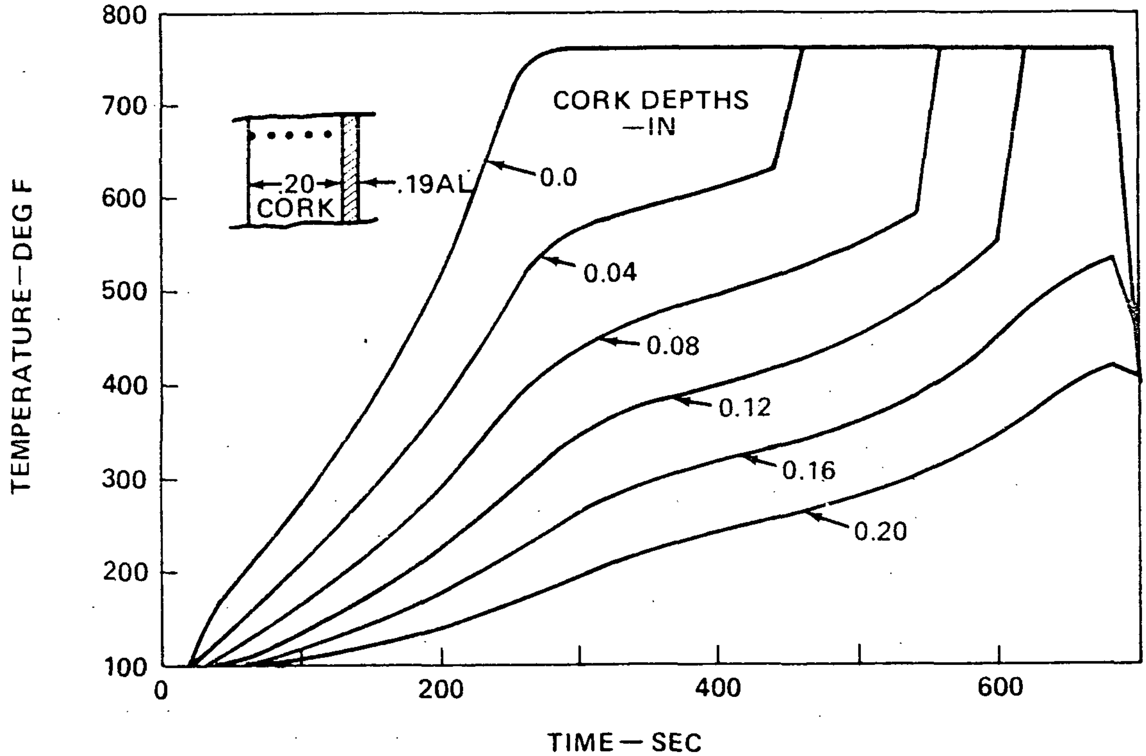

Figure J-9. Lower cone temperature history for P/A Module reentry

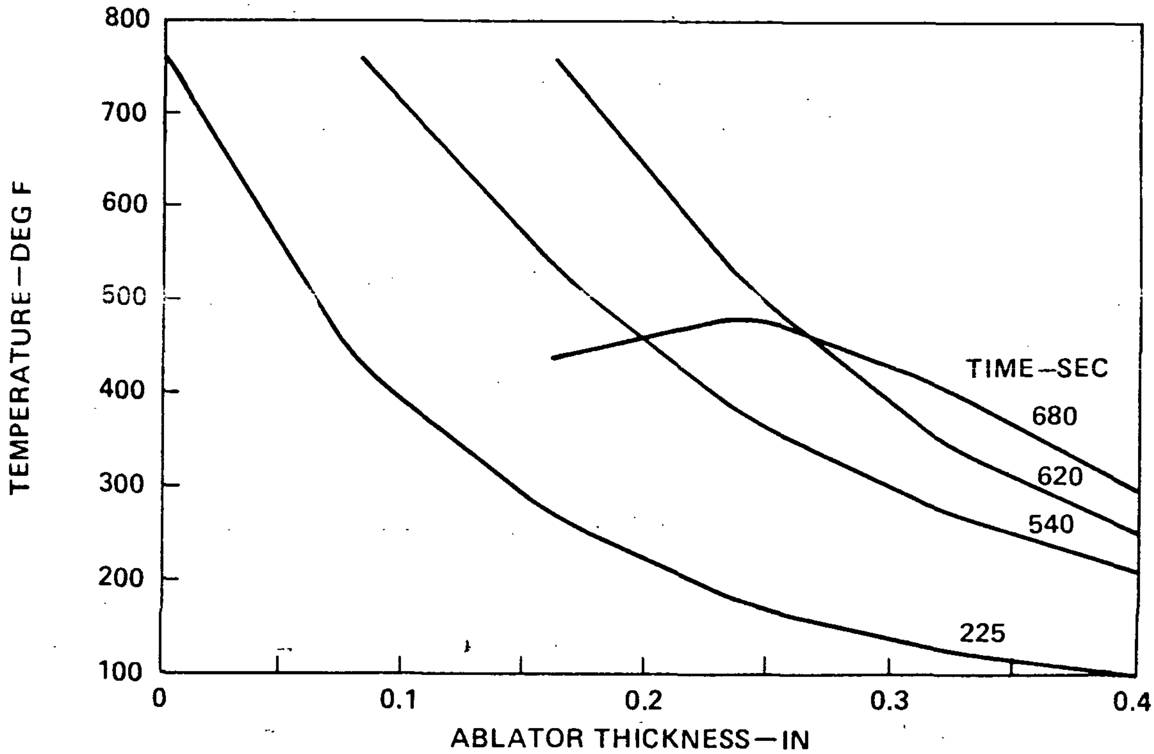

Figure J-10. Forward dome ablator temperature gradient for P/A Module reentry.

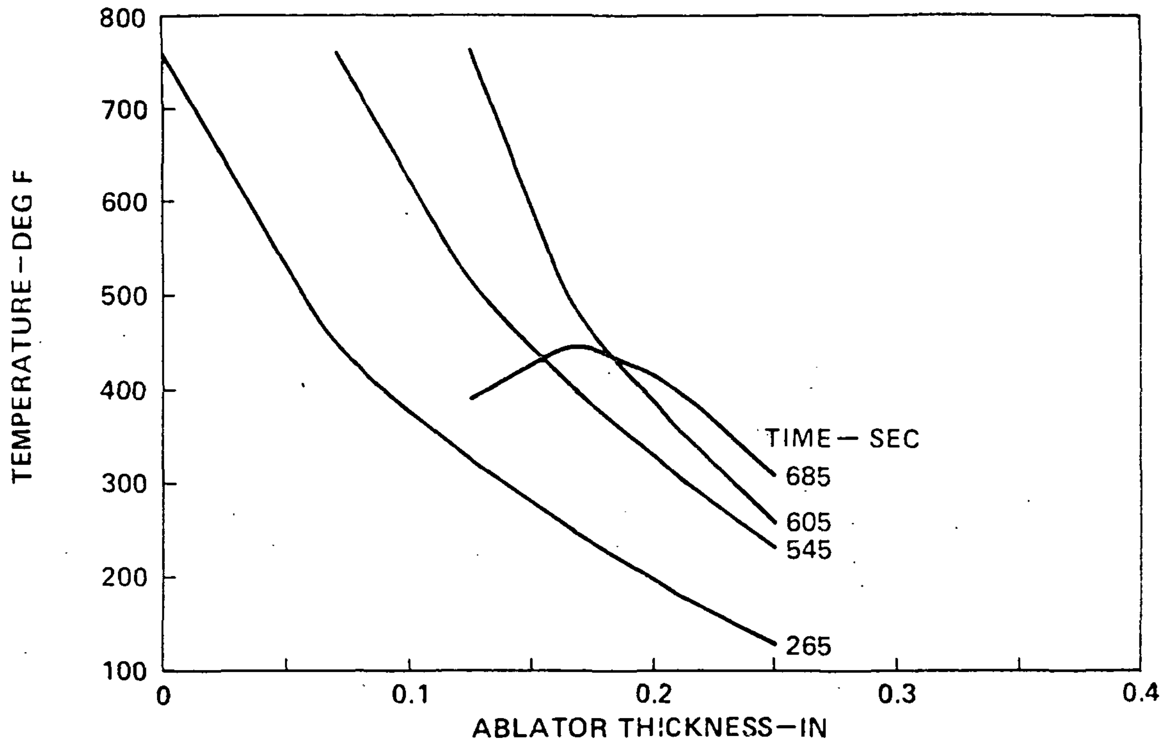

Figure J-11. Upper cone ablator temperature gradient for P/A Module reentry.

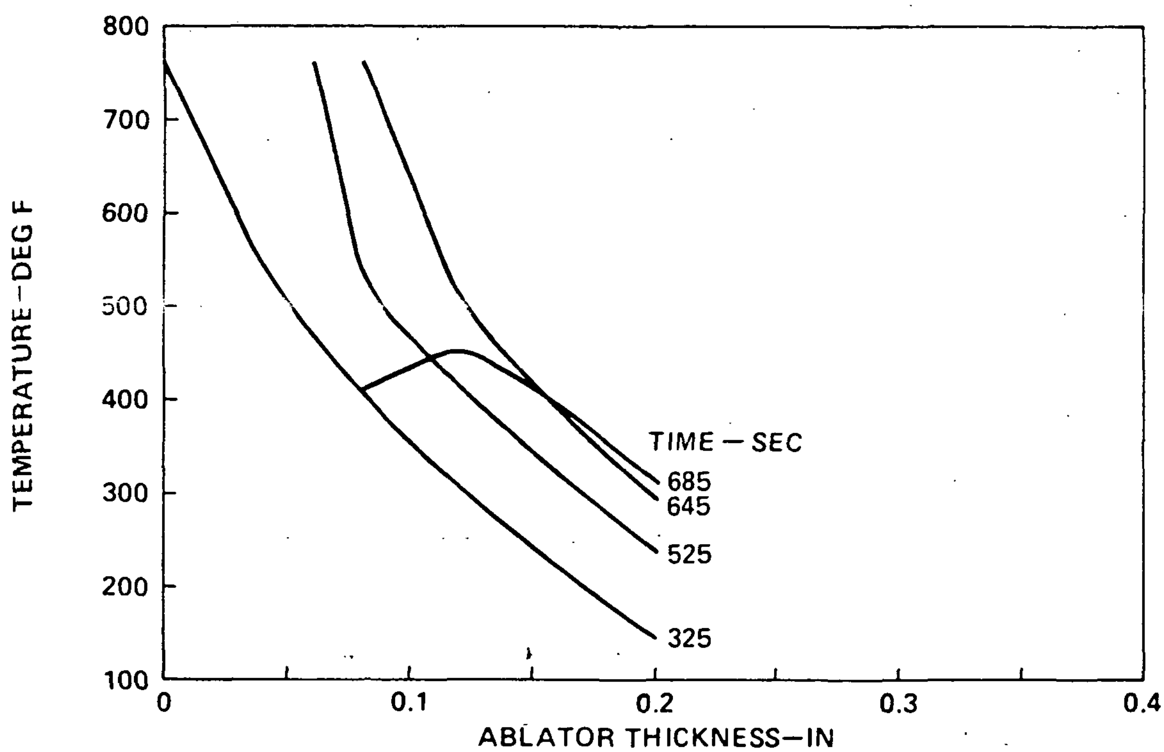

Figure J-12. Lower sidewall ablator temperature gradient for P/A Module reentry.

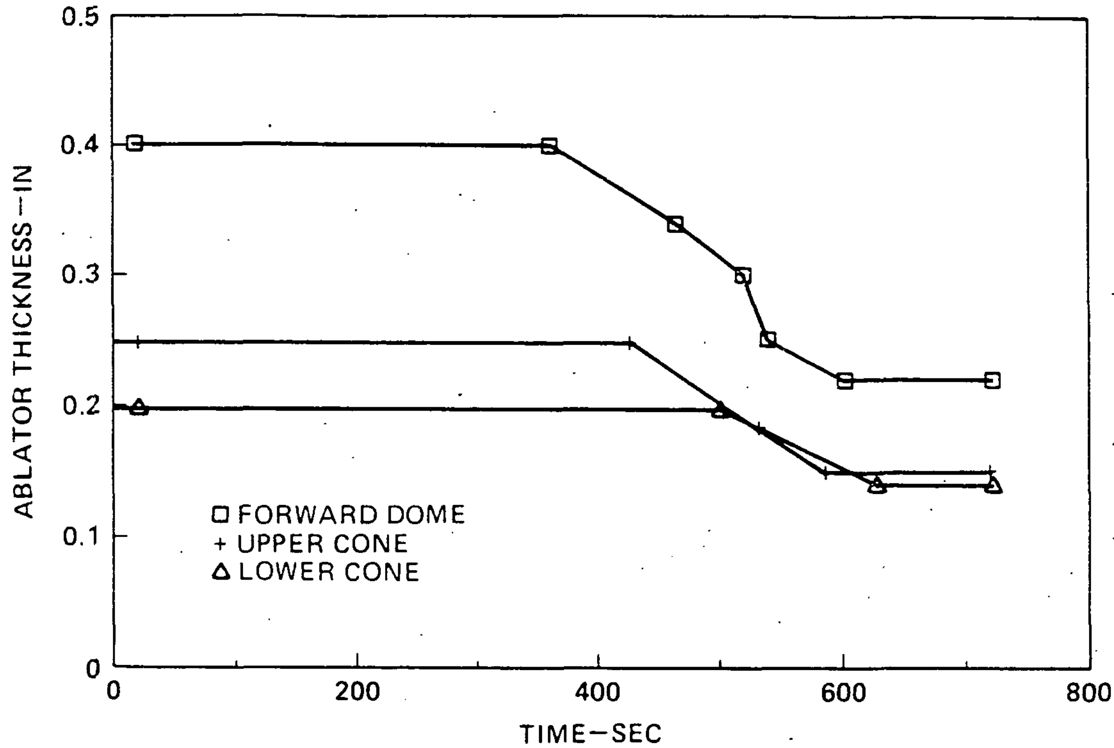

Figure J-13. Ablator thickness for P/A Module reentry.

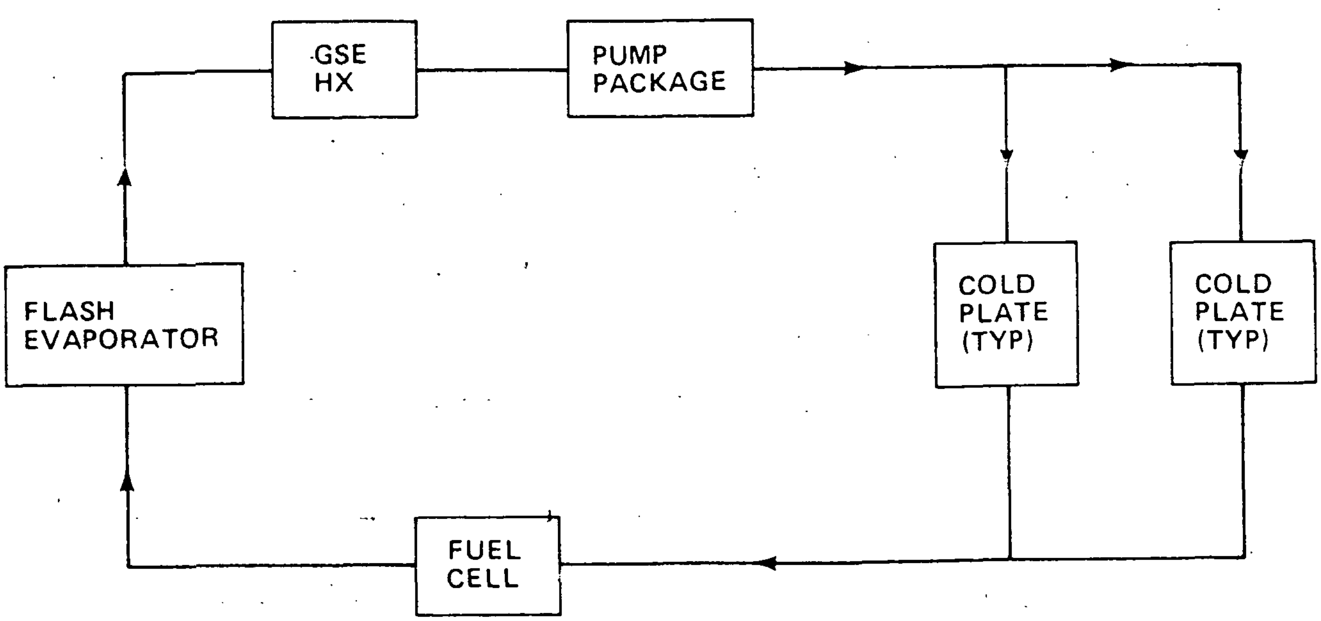

Figure J-14. P/A Module thermal control system equipment schematic.

TABLE J-1. P/A MODULE THERMAL CONTROL SYSTEM

EQUIPMENT	QUANTITY	WEIGHT LBS	VOLUME FT^3	POWER WATTS
PUMP PACKAGE	1	32	1.4	150
COLD PLATES	12	120	0.4	
GSE HEAT EXCHANGER	1	14	0.3	
FLASH EVAPORATOR	1	58	4.5	
CONTROLS AND INSTR		10		
VALVES		20		
FLUID LINES		50		
TOTAL WEIGHT (DRY)		304		
WATER		35		
FREON		50		
TOTAL WEIGHT (WET)		389		

APPENDIX K. CONTRIBUTING PERSONNEL

The following personnel of the Program Development Directorate have made significant contributions to the preparation of this report.

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NOTE: Organizational listings reflect personnel assignments in effect at the time of this study effort.

APPROVAL

HEAVY LIFT LAUNCH VEHICLES FOR 1995 AND BEYOND

Compiled by Ronald Toelle

The information in this report has been reviewed for technical content. Review of any information concerning Department of Defense or nuclear energy activities or programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

William R Marshall

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