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VOLUME II
FINAL REPORT

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LIQUID ROCKET BOOSTER STUDY
FINAL REPORT

GENERAL DYNAMICS
Space Systems Division

LIQUID ROCKET BOOSTER STUDY-FINAL REPORT

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List of Abbreviations and Acronyms

ALS	Advanced Launch Systems
AOA	Abort Once Around
AR	Area Ratio
ATO	Abort to Orbit
BECO	Booster Engine Cutoff
BLOW	Booster Liftoff Weight
BSM	Booster Separation Motor
CH4	Methane
CMD	Command
CDR	Critical Design Review
DC	Dual Channel
DDT & E	Design, Development, Test & Engineering
ECO	Engine Cutoff
ECS	Environmental Control System
EMA	Electrical Mechanical Actuator
ERB	Engineering Review Board
ET	External Tank
FASTPASS	Flexible Analysis for Synthesis, Trajectory and Performance for Advanced Space Systems
FCV	Fuel Cooldown Valve
FSD	Full Scale Development
FSOV	Fuel Shutoff Valve
ft	Feet
G,g	Acceleration of Gravity
GDSS	General Dynamics Space Systems

GLOW	Gross Liftoff Weight
GLP	Ground Launch Processing
GOX	Gaseous Oxygen
GSE	Ground Support Equipment
HYPRS	A Pressurization Model Computer Program
IRD	Interface Requirements Document
ICD	Interface Control Document
ISP	Specific Impulse
JBV	Jacket Bypass Valve
JSC	Johnson Space Center
KLB	Thousands of Pounds
KSC	Kennedy Space Center
LCC	Life Cycle Cost
L/D	Length to Diameter Ratio
LEMSCO	Lockheed Engineering and Management Services Company
LOX,LO2	Liquid Oxygen
LPS	Launch Processing System
LRB	Liquid Rocket Booster
LSOC	Lockheed Space Operations Company
MECO	Main Engine Cutoff
MLP	Mobile Launch Platform
MMC	Martin-Marietta Corporation
MMH	Mono Methyl Hydrazine
MOV	Main Oxygen Valve
MR	Mixture Ratio
MSFC	Marshall Space Flight Center
N2	Nitrogen
NASA	National Aeronautics and Space Administration
nm	Nautical Miles

NPSH	Net Positive Suction Head
NPSP	Net Positive Suction Pressure
NSTL	National Space Technology Laboratories
NSTS	National Space Transportation System
NTO	Nitrous Oxide
OCV	Oxygen Cooldown Valve
ODC	One Dimensional Equilibrium
OMS	Orbital Maneuvering System
ORB	Orbiter
P/A	Propulsion/Avionics
Pc	Engine Combustion Chamber Pressure
PDR	Preliminary Design Review
PRC	Planning Research Corporation
PRR	Preliminary Requirements Review
PQR	Pitch, Yaw, Roll
psia	Pounds Per Square Inch Absolute
P/U	Propellant Utilization
P&W	Pratt & Whitney Company
Q	Dynamic Pressure
Q-Alpha	Dynamic Pressure x Angle of Attack
R	Rankine
RP-1	Rocket Propellant
RSS	Rotating Service Structure
RTLS	Return to Launch Site
SC	Single Channel
secs	Seconds
SE&I	Systems Engineering and Integration
SEPP	Systems Effectiveness Program Plan
SIMS-II	Space Integrated Management System II (A Program Plan Progress Accounting System)
SIT	Shuttle Integrated Test

SL	Sea Level
SLA	Super Light Ablative
SOFI	Spray On Foam Insulation
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor
SSME	Space Shuttle Main Engine
STBE	Space Transportation Booster Engine
STME	Space Transportation Main Engine
STS	Space Transportation System
TBV	Turbine Bypass Valve
TPS	Thermal Protection System
TVC	Thrust Vector Control
T/W	Thrust to Weight Ratio
V	Velocity
VAB	Vertical Assembly Building
VAC	Vacuum
WBS	Work Breakdown Structure

SECTION 1
RECOMMENDED CONCEPTS

Figure 1-1 shows our recommended LRB concept using a common main engine with the Advanced Launch System (ALS) which burns liquid oxygen and liquid hydrogen. The central rationale is based on our belief that the USA can only efford one big new rocket engine development in the 1990's. A LO2/LH2 engine in the half million pound thrust class could satisfy STS LRB, ALS, and Shuttle-C (instead of SSMEs). Development costs and higher production rates can be shared by NASA and the USAF. If the ALS program does not occur, then LO2/RP-1 propellants would produce slightly lower costs for an STS LRB. When the planned Booster Engine portion of the Civil Space Transportation Initiatives (CSTI) has provided data on large pressure fed LO2/RP-1 engines, then the choices should be reevaluated.

Some basic LRB features which we recommend:

1. The LRB should be expendable. At the projected flight rates of 6 to 14 STS missions per year, downstream economic benefits that would accrue from recovery and reuse of LRBs do not appear to justify the added risk and up-front program cost associated with developing a recovery capability. An inherently reusable concept for limited engine reuse deserves further study.

2. Four engines on each LRB gives the best combination of engine-out capability, reliability, and cost. Note that these are much less complex engine types than the Space Shuttle Main Engine (SSME).

3. A new low cost expendable engine should be developed for the LRB. This option is considerable more cost-effective than adapting an existing engine, such as the Space Shuttle Main Engine (SSME) or the F-1 of the Saturn first stage, to the LRB. In fact, a new LO2/LH2 engine could be common with the ALS program and Shuttle-C.

4. Liquid oxygen and liquid hydrogen is the propellant combination recommended for LRBs. LO2/LH2 has the least environmental impact and uses propellants common with the ET. Sharing a

common engine with ALS makes sense for the country. Second choice is LO2/RP1 which has been used since the 1950's with the highly reliable Atlas, Delta, and Saturn launch vehicles. It has significant safety and environmental advantages over storable propellants (N204/MMH).

5. Pump-fed and pressure-fed LRBs are both viable options. LRB safety and performance requirements can be met with either pump-fed or pressure-fed boosters. Pressure fed systems need technology development in engines hot gas pressurization and light weight tanks to reduce LRB risks. Further efforts are warranted to reduce the costs of all LRB concepts. The split-expander cycle concept offers the hope of higher reliability at lower costs.

6. LRB will impact KSC moderately. The major challenge is scheduling the phase-in with minimum disruption to ongoing launch operations.

RECOMMENDED LRB CONCEPT

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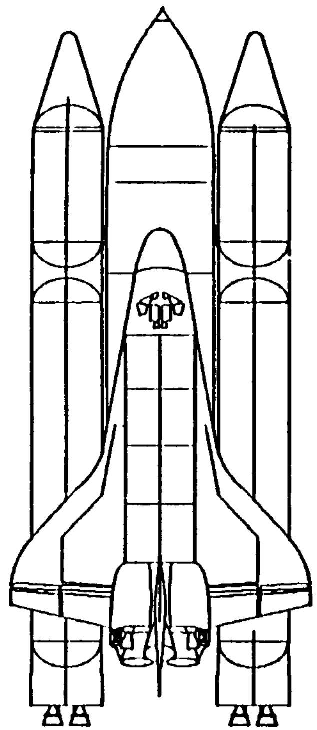


PROPELLANTS: LO2/LH2

EXPENDABLE (LIMITED ENGINE REUSE TBD)

ENGINE THRUST: 515/558 LB (SL/VAC)

DRY WEIGHT: 121,935 LBS.

BLOW: 820,531 LBS

LENGTH & DIAMETER: 178 X 18 FT

- NEW BOOST PHASE ABORT CAPABILITY
- INCREASED STS PERFORMANCE (70.5klb TO 150nmi)
- MISSION SUCCESS WITH ONE ENGINE OUT
- SHORTER ON-LINE PROCESSING AT KSC
  NO HAZARDOUS PROPELLANTS IN VAB
- CLEAN, NON-TOXIC EXHAUST PRODUCTS
- POTENTIAL FOR COMMONALITY:

COMMON ENGINE FOR STS AND ALS BOOSTERS
REPLACE SHUTTLE-C SSME AND ALS CORE

ALLOWS USA TO CONCENTRATE ON 1 BIG NEW ROCKET ENGINE IN 1990'S

Figure 1-1 Recommend LRB Concept

SECTION 2
STUDY APPROACH

Our study approach was to "start with a clean sheet of paper", perform basic trades such as propellant selection from which concepts would be sized, and then select the best using an approved list of criteria.

2.1 BASIC REQUIREMENTS, GROUND RULES, AND ASSUMPTIONS

- Each concept sized for 70.5 KLB payload to 150NMI due east from KSC
- Safe abort with one LRB engine (or 1 SSME) out
- GD Goal: Full payload ATO (105 nmi) with 1 engine out
- High Reliability/probability of mission success (approximately 0.99)
- Virtually no hardware changes to Orbiter
- Use STS trajectory constraints on Max Q, Max G, etc.
- Orbiter wing loads limited to current levels
- Minimize changes to ET
- Reasonable changes to KCS facilities and GSE (need new MLP)
- LRB may or may not be reusable, depending on trade results
- IOC depends on concept but 1996 is an approximate target
- Growth and evolution being considered

2.2 METHODOLOGY

The breadth of concept options for which an LRB can be considered: propellant combinations, new and existing engines, pump- and pressure-fed alternatives, alternative recovery modes, and evolutionary options, led us to a concept selection approach in which selections were made in three stages.

In our first selection, booster concepts were initially evaluated on the basis of safety, performance, compatibility, and other criteria independent of recovery considerations. The intent was to identify 3 to 5 LRB concepts that include, as a minimum, one pump-fed system with a new engine, one pump-fed system with an existing engine, and one pressure-fed system. After this is accomplished, various recovery options for those selected LRBs were examined (including the consideration of not recovering the boosters should economics so justify). A second selection was then made in which the best recovery system was coupled with each of the LRB concepts. In the third stage of this process, each candidate was examined for evolution and growth approaches through the analysis of alternative growth paths.

The current engines include those candidates judged suitable for LRB application which are either in production or can be readily brought into production. In the case of the other pump-fed and pressure-fed alternatives, propellants were considered that exhibited various desirable features for LRB applications. New engine designs were based on NASA/MSFC STBE (Space Transportation Booster Engine) and STME (Space Transportation Main Engine) studies. Pressure-fed engine data was provided by engine subcontractors, Rocketdyne and TRW. Also looked at were metalized propellant systems which offer high density-impulse characteristics, but require technological advancements.

Figure 2.2-1 shows that the original 15 concepts were refined and evaluated by a number of trades and analyses. Initially attention was focused on propellant safety/environmental impact and Orbiter wing loading problems caused by large LRBs. Before the middle of the study, some concepts had been eliminated.

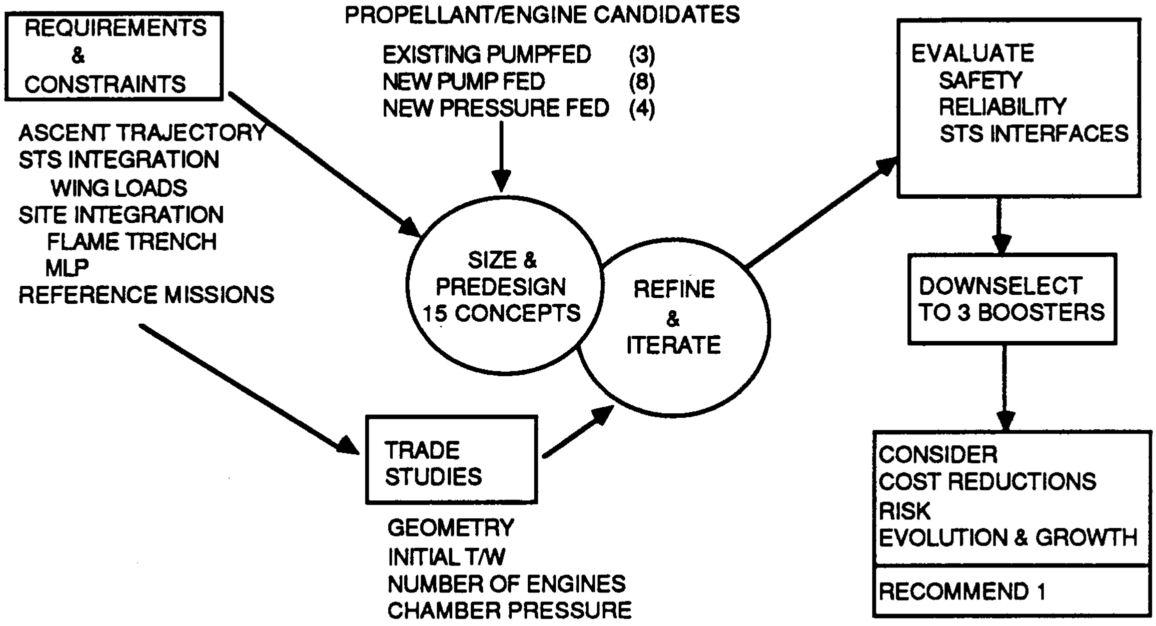


Figure 2.2-1 Approach to LRB concept selection

Sizing was initially performed using typical propellant density, mixture ratio and Isp data from our files. As the engine subcontractors provided data tailored to LRB and as Shuttle trajectory constraints became better defined, more accurate sizing was performed using our pre-design synthesis model "FASTPASS". Late in the study we resized the selected concepts for Abort to Orbit (ATO) with one LRB engine out and engine throttling to avoid overloading the ET LOX tank aft bulkhead.

The LRB evaluation and selection process provides insight into candidate LRB attributes and weaknesses. When comparing LRB candidates, the evaluation process provides a relative measure of goodness, and helps to select LRB concepts. While cost is an important criterion, our approach ensures that candidate LRB concepts will also be evaluated on such critical factors as safety, reliability, and STS vehicle/facility impacts. Using the ground rules and assumptions shown earlier, and preliminary propulsion performance estimates, the 15 vehicle concepts were sized and outline drawings made. This data pack was submitted to 6 specialists, who were responsible for screening all candidate LRB concepts with respect to his assigned criteria, such as safety. This is to ensure evaluation accuracy and consistency. The evaluator ranked each candidate LRB concept relative to the others. A detailed written rationale was submitted that explained the attributes and detriments of each candidate LRB and why the evaluator ranked them in a specific order.

The Technical Review Board including the LRB program manager and chief engineer plus 4 senior management engineers representing design, technology, Atlas/Centaur and advanced technology reviewed the evaluations, and made their own ranking. Subsequently, new information has eliminated more concepts.

Substantial differences exist among generic classes of boosters that can meet LRB goals. To prevent prematurely eliminating a booster class from further consideration, we structured out approach to ensure that the following concepts, at a minimum, carried through the evaluation process for further definition. These include one pressure-fed concept, one pump-fed concept with existing engines, and one pump-fed concept using new engines. Late in the study an additional concept was added by contract change using a new pump-fed engine of the split expander type. Another contract change focused on requirements and planning technology demonstration work planned at NASA/MSFC to test large LOX/RP-1 pressure-fed engine systems.

A key element in performing the study is the Technical Working Group. This consists of NASA/MSFC, JSC, and KSC personnel and their contractors. We met frequently, exchanged data and talked on the phone. Together we are an LRB team. Our trade studies, analyses, and concept selections were strongly influenced, and in some cases depended on, inputs from the team. For instance: wind tunnel data from MSFC and wing loading analyses by LEMSCO were the basis of our geometry trade.

Costs were initially considered a secondary selection criteria. As concepts became better defined, the accuracy of the cost estimates improved. Cost was the key to the recoverability trade and was a strong consideration in the final selection.

Evolution and Growth including commonality with ALS became a major issue late in the study.

The final recommendation is shown in Section 1 with rational throughout this report.

2.3 SELECTION CRITERIA

We have used the same criteria for evaluating trade study data and for selecting concepts. Table 2-1 lists the primary and secondary criteria. They were grouped into first-order criteria -- those factors determined to be of overwhelming importance, which if not met would likely preclude acceptance of an LRB - and second-order criteria, which are also important but applied to discriminate further between acceptable LRB candidates. Within the selection process, those concepts found to be unacceptable with respect to first-order criteria were discarded and not further considered.

Improved safety/environmental acceptability is the prime reason for considering LRBs as potential SRB replacements. Any viable LRB must therefore well exceed the behavior of solids in this category. Reliability is also of particular interest, not only because of the considerable expense of aborted or curtailed missions, but because the greater complexity of liquids over solids requires added attention to this parameter. The two subsequent items, STS and launch site compatibility, are crucial in that they establish the feasibility of LRBs; difficult integration or incompatibility with existent constraints rapidly escalates program cost and risk. The final first-order criterion, performance, is interpreted not as a measure of payload capability - as all LRBs were designed to satisfy the same requirement of enabling delivery of 70,500 lb to 150 nmi - but instead is determined by booster size and weight.

When concepts rank high against the primary criteria, then secondary criteria may be the deciding factor. Thus costs, risk and growth potential become major considerations in comparing concepts which were approximately equal in regard to safety, reliability, etc.

Table 2-1 LRB selection criteria

PRIMARY	
•Safety/Environmental Acceptability	•STS and Crew •Launch Facilities and Personnel •Transportation Infrastructure •Public
•Reliability	•Engine-Out Capability •Complexity •Redundancy
•STS compatibility	•Interface Modifications •Aerodynamics •Loads
•Launch Site Compatibility	•Required Modifications •Processing Times •Commonality
•Performance	•Weights/Volume •Margins
SECONDARY	
•Costs	•DDT & E •Recurring •Life Cycle
•Development Risk	•Cost Risk •Schedule Risk •Technical Risk
•Operational Availability	•Readiness •Environmental Sensitivity •Turnaround
•Operational Complexity	•Hazardous Operations •Accessibility •Processing Complexity
•Growth Potential	•Versatility •Subsystem Applicability

2.4 LRB DESIGN REQUIREMENTS

The LRB Design Requirements are listed in Table (TBD). The sources used in compiling the list of LRB goals and requirements were, the LRB Statement of Work, the SRB End Item Specification, NSTS 07700, Vol X, Space Shuttle Flight and Ground System Specification and GDSS inputs. The emphasis of the LRB Goals and Requirements was directed toward the safety, reliability and SRB commonality factors.

The table is divided into the Goals and Assumptions section and the Requirements section. The Requirements section is further organized by program level and addresses the requirements applicable to the particular level. The requirements fall into four different levels; Space Transportation System (Level 1), Space Shuttle Vehicle (Level 2), Liquid Rocket Booster (Level 3), and LRB Subsystems (Level 4).

TABLE TBD LRB GOALS AND REQUIREMENTS

No.	TITLE AND DESCRIPTION	SOURCE
	GUIDELINES, GOALS AND ASSUMPTIONS	
001	Impact/Changes to the STS As a primary program guideline, the use of the LRB on the STS shall cause minimum impact/changes to the ET, Orbiter. Launch site and GSE	LRB SOW Para D p. j-1-3
002	Safety, Reliability and Performance The safety, reliability and performance of the STS shall be improved compared to the current STS by the development of new or revised abort sequences that are based on capability of LRB engines being throttled and/or shutdown on demand in cases of LRB or Orbiter contingencies.	LRB SOW Para c p. j-1-3
003	SRB/LRB Transition Transition from SRBs to LRBs shall be on a non or minimal interference basis with the planned STS launches.	MSFC
004	Initial Operating Capability (IOC) IOC for STS with LRBs is 1995 assuming start of definition/development in FY-89. For the "best" concept; however, this date may not be feasible.	MSFC/GDSS
005	Maximum Usage The maximum use rate of the STS with LRBs will be 14 flights per year, including any possible unmanned flights. Current fleet use in operation until year 2005.	MSFC
006	Engine-Out Missions The STS with LRBs is to have ATO capability despite failure of one engine on the LRB at the time of liftoff.	GDSS
007	Recovery/Refurbishment For the recovery/refurbishment/ reuse LRB option, maximize use of existing SRB refurbishment facilities and water recovery systems and equipment.	MSFC

008	Ignition and Liftoff LRB engines start and vehicle liftoff to be compatible with existing launch facility and sequencing.	MSFC
009	LRB Inspection and Checkout Ease of inspection and checkout of the LRB throughout processing and launch operations will be incorporated into the design of the LRB.	GDSS
010	Flame Trench The STS vehicle with LRBs should, as a goal, be compatible with current launch pad flame trench dimensions.	GDSS
011	MLP Flame Holes The STS Vehicle System should maintain the current Mobile Launch Platform flame hole dimensions as a goal.	GDSS
012	Ground Access The LRB should maintain existing VAB and RSS access as a goal.	GDSS

Requiremenrts
Level I: Space Transportation S.:tem

1.1(a)	Performance Requirement 1 The performance requirement for the nominal case STS with LRBs shall be to place a 70,500 pound payload, in a 160 nautical mile circular orbit at 28.5 degrees inclination from a KSC launch with the Orbiter SSMEs limited to 104% power level (109% for abort).	LRB SOW Para. E p. J-1-3
1.1(b)	Alternate Performance Requirement As an alternate case, LRBs shall be sized to meet the requirement that the Space Transportation System be capable of placing a 59,000 pound payload in a 160 n.m . circular orbit with 28.5 degrees inclination from a KSC launch with Space Shuttle Main Engine thrust levels limited to 104%.	LRB SOW Para. E p. J-1-3
1.2	Launch Sites The STS with LRBs shall initially be launched from the Kennedy Space Center (KSC). There may be a requirement to also conduct future launches from the Western Space and Missile Center (Vandenberg AFB).	BRM-1

1.3	<u>Day & Night Operations</u> The Shuttle System shall have the capability to launch and land the flight vehicle in daylight or darkness.	NSTS 07700 Vol. X Para 3.2.1.1.7
1.4	<u>24 Hour Notification for Launch</u> To fulfill the space rescue role, the Shuttle System shall be capable of launching within 24 hours after notification and the flight vehicle mated and ready for transfer to the pad. This time includes retargeting to a dissimilar mission, loading a validated flight program and filling the OMS and RCS propellant tanks.	NSTS 07700 Vol. X Para 3.2.1.2.1
1.5	<u>Pad Stay Time</u> The Space Shuttle System shall accommodate the mated vehicle on the launch pad for durations up to 180 days. Exposure to natural and induced environments for the pad stay time duration shall not invalidate the design performance or operational capability of the flight vehicle.	NSTS 07700 Vol. X Para 3.2.1.2.11
1.6	<u>Debris and Ice Prevention</u> The Shuttle System, including the ground systems, shall be designed to preclude the shedding of ice and/or other debris from the elements during prelaunch and flight operations that would jeopardize the flight crew and/or mission success.	NSTS 07700 Vol. X Para 3.2.1.2.14
1.7	<u>Range Safety</u> The STS shall conform to the range safety documents, AFTREM 127-1, SAMTECM 127-1, MSFC SPEC 30A90506.	NSTS 07700 Vol. X Para.- 3.5.4.1.12 3.5.4.2.7
1.8	<u>Hold-down & Release</u> The STS Ground Ops Sys shall have the capability of holding down the Shuttle Vehicle on the launch pad following ignition through thrust buildup to 100% RPL of the Orbiter and LRB engines. This delay is for also conducting health checks of the vehicle sub-systems prior to release.	NSTS 07700 Vol X Para 3.4.6.2.2
	<u>LEVEL 2: SPACE SHUTTLE VEHICLE</u>	
2.1	<u>Engine-out Performance</u> Intact abort must be possible with one engine out on the Orbiter and/or one LRB.	NSTS 07700 Vol X Para 3.4.6.2.2

2.2	Lift-off Clearances Minimum position clearances shall not be violated that exist between the Space Shuttle vehicle and all ground launch facility hard points from booster ignition through tower clearance for both nominal and intact abort modes. Vehicle clearance and drift during lift-off shall be within the envelops specified in applicable ICD.	NSTS 07700 Vol. X Para 3.2.1.16.1
2.3	Thrust/Weight at Lift-off The initial thrust to weight ratio of the STS vehicle at lift-off with engine out must not be less than 1.2 to ensure that the Shuttle Vehicle clears the launch tower safely.	JSC/LEMSCO
2.4	Launch from Standby The Shuttle System shall have the capability to launch the flight vehicle from a standby status within 2 hours. Vehicle access shall be permitted for not less than 45 minutes of consecutive time within the 2 hours to accommodate flight crew ingress and final prelaunch closeout. The Shuttle System shall have the capability to hold in a standby status up to 24 hours.	NSTS 07700 Vol. X Para 3.2.1.2.2
2.5	Flight Performance Reserves. Flight performance reserves shall be based on + or - 3 sigma systems and environment dispersions, except during the Abort-Once-Around/ Abort-To-Orbit (AOA/ATO) abort portion of the missions. The flight performance reserves during the AOA/ATO abort portion of the missions shall be based on + or - 2 sigma systems and environmental dispersions.	NSTSD 07700 Vol . X Para 3.2.1.1.4
2.6	Vehicle Acceleration For manned STS missions, a load limit factor of 3g acceleration must not be exceeded.	NSTS 07700 Vol. X Para 3.2.1.1.11
2.7	Ordnance Control All ordnance circuits shall utilize Pyro Initiator Controllers per Rockwell International Space Division specification MC450-0018 and shall meet the requirement of Pyrotechnic Specification JSC-08060 and AFETRM 127-1.	1) NSTS 07700 Vol. X Para 3.6.20 2) SRB End Item spec CPO13M00000B Para 3.2.1.5.5

2.8	POGO The Space Shuttle Vehicle Shall be free of instabilities resulting from dynamic coupling of the structure, propulsion and flight control subsystems during all phases of powered flight with all payload variations.	NSTS 07700 VOL X Para3.2.2.1.4

LEVEL 3: LIQUID ROCKET BOOSTER

3.1	Launch Readiness Checkout The LRB shall be capable of launch readiness checkout with support from the Orbiter after ground system connection on the launch pad.	SRB End Item Spec CPO13M00000B Para 3.2.1.1.5
3.2	LRB/External Tank Buildup & Mating The LRB shall be capable of buildup, servicing, verification and assembly on the MLP prior to ET mating. The LRB shall be capable of alignment, connection, and verification of mechanical and electrical interfaces during mating operations.	NSTS 07700 Vol. X Para 3.4.5.1
3.3	Prelaunch Loads The LRBs must have free standing capability and support the entire STS vehicle system (LRBs, ET & Orbiter) on the MLP during ground operations. The combined loads it must withstand are propellant loading, vehicle body and launch pad/MLP flexibilities, effects of vortex shedding and other unsteady flow phenomena	SRB End Item Spec CPO13M00000B Para 3.2.1.1.7.1
3.4	Propellant Interface Accessibility LRB interfaces for servicing propellants on the MLP must be physically accessible.	NSTS 07700 Vol. X Para 3.4.5.1
3.5	Launch Process Interface The LRB shall interface with the launch processing system for checkout, integration with STS, countdown and launch.. The start sequence, propellant loading and topping functions are included. The instrumentation on the LRB should support a go/no go decision by an automated LPS. transients (following vehicle release) have damped out.	NSTS 07700 Vol. X Para 3.4.16

3.6	Launch Loads The LRBs shall be capable of withstanding induced loads during the launch period. The launch period extends from the initiation of LRB ignition until all structural response transients (Following vehicle release) have damped out.	NSTS 07700 Vol. X Para 3.2.2.1.17.2
3.7	Icing Conditions The LRB design shall preclude the possibility of icing formation (cryogenic) that could be potentially hazardous to the Orbiter or ET.	NSTS 07700 Vol. X Para 3.2.1.2.14
3.8	Storage After acceptance, the LRB including all components and subsystems shall be capable of the required performance after a maximum storage period of 5 years at a temperature range of +32 to +95 deg. F.	SRB End Item Spec CP013M00000P Para 3.2.1.6.9
3.9	Accessibility The LRB shall provide access to each area of the LRB containing components or items requiring access. This access and clearance envelope shall permit the use of access and handling equipment if such GSE is required.	BRM-1
3.10	Maximum Dynamic Pressure A maximum dynamic pressure of q = 733 psf shall be used as the nominal upper design limit based on orbiter wing loading, system tolerances and seasonal winds.	BRM-1
3.11	Maximum Dynamic Pressure at LRB Separation The maximum dynamic pressure at LRB separation shall not exceed 75 psf.	BRM-1

LEVEL 4: AVIONICS AND FLIGHT CONTROL SUBSYTEMS

4.1.1	Ascent Control The thrust vector control (TVC) on the LRB, in conjunction with the TVC on the SSME shall provide ascent control authority in roll, pitch, and yaw.	SRB End Item Spec CP013M00000B Para 3.2.1.1.1
4.1.2	Multiplexing Pulse Code Modulation (PCM) shall be primary technique for multiplexing LRB data. FM multiplexing shall be provided in cases where PCM cannot satisfy development data bandwidth requirements. Maximum programming flexibility shall be provided in order to facilitate measurement changes. Multiplexer design shall be optimized for system compatibility with the data storage equipment.	SRB End Item Spec CP013M00000B Para 3.2.1.5.8.5
4.1.3	Commands All critical commands shall be provided to the LRB via hardwire. Noncritical commands shall be multiplexed from the Orbiter to the LRB. The Electrical & Istrumentation (E&I) subsystem shall provide capability to demultiplex digital commands as defined by ICD TBD.	SRB End Item Spec CP013M00000B Para 3.2.1.5.8.6

LEVEL 4: LRB SEPARATION SUBSYSTEM

4.2.1	LRB Separation Subsystem/Orbiter Interface The LRB separation subsystem shall include the capability to: a) accept and respond to separation commands transmitted by the Orbiter over hardwire and (b) release and separate the LRBs away from the Orbiter/ET. The release and separations hardware shall be the responsibility of the LRB contractor.	SRB End Item Spec CP013M00000B Para 3.2.1.3

4.2.2	Separation Damage The LRB separation subsystem shall provide for separation of the LRB from the External Tank (ET) without damage to or recontact with the ET or Orbiter during or after separation. Damage to the LRB/ET connectors on the aft upper struts at the LRB/ET interface during LRB separation after the TVC power is deadfaced is acceptable. The plumes from the booster shall not impinge on the Orbiter. The LRB separation subsystem shall not release any debris which could cause damage to any Orbiter/ET system or subsystem.	NSTS 07700 Vol. X Para 3.2.1.1.9.1
4.2.3	Separation Torque Any component disconnect or breakwire at release shall not include an impulse torque in excess of 700 ft-lb-sec about the LRB center of gravity at separation.	NSTS 07700 Vol. X Para 3.2.1.1.9.1.1.2
4.2.4	Separation Signal Interlock The LRB separation subsystem shall incorporate signal interlocks to prevent LRB release due to stray signals.	NSTS 07700 Vol. X Para 3.2.1.1.9.1.1
4.2.5	Separation Body Rate Limits Separation of the LRBs from the Orbiter/ET shall occur only after LRB shutdown. The separation shall be automatically inhibited if vehicle body rates and/or dynamic pressures exceed those values for which the separation system has the capability to perform a separation without causing shuttle element damage. The crew shall be provided the capability to manually override these body rate dynamic peressure inhibits.	NSTS 07700 Vol. X Para 3.2.1.1.9.1
4.2.6	Design LRB Staging Conditions The LRB separation system shall be designed to provide a safe separation for staging conditions which comprise any combination of values, within the separation limits, of these parameters: a. Roll rate between -5 degrees/sec and +5 degrees/sec. b. Pitch rate between -2 degrees/sec and +2 degrees/sec. c. Yaw rate between -2 degrees/sec and + 2 degrees/sec. d. Dynamic pressure less than or equal to 75 psf.	NSTS 07700 Vol. X Para 3.2.1.1.9.1.3

The separation system shall be designed to provide a safe separation for pitch and sideslip angles at staging which do not exceed plus or minus 15 degrees.

SECTION 3
TRADES & ANALYSES SUMMARIES

3.1 SAFETY AND RELIABILITY

The accident with ST-5IL has focused attention on shuttle safety. Solid rocket systems do not offer the advantages inherent in liquid engines which can be throttled or cut off on command. This provides flexibility to work around failures and opens the possibility of new improved abort modes. There is the general perception, however, that solid rocket motors are simple and therefore more reliable than liquids. Throughout our LRB study, safety and reliability have been the top priority driving trade studies and concept selections. LRB's offer several major safety, reliability, and environmental impact advantages over SRB's.

It was an original study requirement that safe abort must be possible with one LRB engine out. This meant that at least two engines are required per LRB. Section 3.2 discusses further the number of engines recommending four.

Section 8 discusses enhanced abort modes. We believe that mission performance with one engine out will pay for itself in added reliability. Therefore we have sized all concepts for Abort-to-Orbit (ATO) with one engine out. Section 3.2 shows that with engine out capability, LRB propulsion system reliability should be better than a single SRM.

Safety during propellant loading and flight was a prime consideration in propellant selection as discussed in Section 3.5. Storable propellants (NTO/MMH) were considered too great a risk for LRB. Storable propellants are currently used on the Shuttle OMS requiring safety clothing and special procedures. The Titan launches USAF payloads with a storable propellant core. A single Shuttle launch using storable propellant LRB's would involve five times the amount used in a Titan launch. Nitrogen tetroxide is an acute toxic waste, highly toxic at low concentrations. Hydrazines are a suspected carcinogen and very reactive. Such a large quantity spilled would be a high risk to ground and/or flight personnel, and the environment for miles around. Therefore we have selected liquid oxygen, RP-1, methane and liquid hydrogen as much safer and more environmentally compatible propellants. Liquid propellants would be loaded into the LRB at the launch pad.

Therefore propellants offer no safety hazard when LRB is in the VAB. Currently parts of the VAB are evacuated when SRB segments are stacked. Work time would be significantly increased with LRB's. An SRB accident could raze the VAB.

3.2 NUMBER OF ENGINES TRADE

The primary reason for considering LRBs for replacement of SRBs is to enhance safety and reliability. An LRB, because of its capability to shut down on command, is inherently safer than a SRB. It is shown in this trade that : (1) a multiple engine booster configuration with an engine out capability increases safety and reliability, and (2) a four-engine configuration for each LRB is preferrable.

The number of engines and their arrangements considered in this trade are depicted in Figure 3.2-1. Although five engines and more than six engines were not considered, it can be shown that conclusions drawn and trends developed with the 1, 2, 3, 4, and 6 engine configurations are applicable to any number of engines.

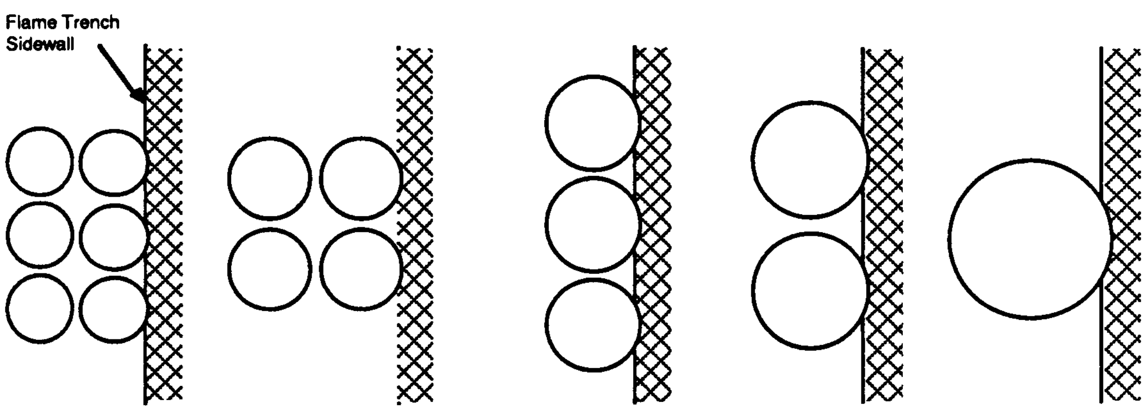


Figure 3.2-1 Engines Layout Considered

3.2.1 NEED FOR MULTIPLE ENGINES ON LRB. Table 3.2.1-1 shows historical data on selected flight vehicles. One can readily conclude from this table that engine-out capability is essential for an excellent launch record. In the case of solid rocket motors, most of the failure modes are catastrophic, and hence the engine-out concept is not meaningful. Flight data, and Failure Mode and Effect Analysis (FMEA) indicates that liquid motors, on the other hand, can be safely shut down in most failure cases. Hence engine-out capability (meaning multiple engine configurations), and better safety, can be achieved using liquid rocket boosters. Therefore the single engine configuration was dropped from further consideration.

Table 3.2.1-1 Historical data showing need for multiple engines on LRB

	TYPE OF PROPULSION SYSTEM	VEHICLE	FLIGHT ENGINES	FAILURES	LOSS OF VEHICLE
NO ENGINE-OUT CAPABILITY	SRM (SEGMENTED)	TITAN	174	1	YES
		STS	50	1	YES
	LIQUID ENGINES	THOR	316	3	YES
		DELTA	76	0	NO
		ATLAS	839	14	YES
ENGINE-OUT CAPABILITY WITH REDLINES	LIQUID ENGINES	SATURN (H-1, F-1, J-2)	303	3	NO
		STS	75	1	NO

ENGINE-OUT CAPABILITY ENHANCES SAFETY/RELIABILITY

3.2.2 NUMBER OF ENGINE(S)-OUT CAPABILITY. The basic assumptions made here are that LRB engines will have a demonstrated reliability of 99% at the time of first flight with a correlation of failure between various engines of 0.05. This reliability is an accepted number in the industry, and it represents a compromise between DDT&E costs (which are dependent on the number of tests needed to demonstrate the required reliability), time of development, and having sufficient confidence level at first lift-off. It should be pointed out that this reliability level is the demostrated reliability level before the first flight, and has no direct bearing with inherent reliability of the engines (a pressure fed may be more reliable than a pump fed) or historical reliability.

As shown in Figure 3.2.2-1, the propulsion system reliability without engine-out decreases rapidly as the number of engines increases.

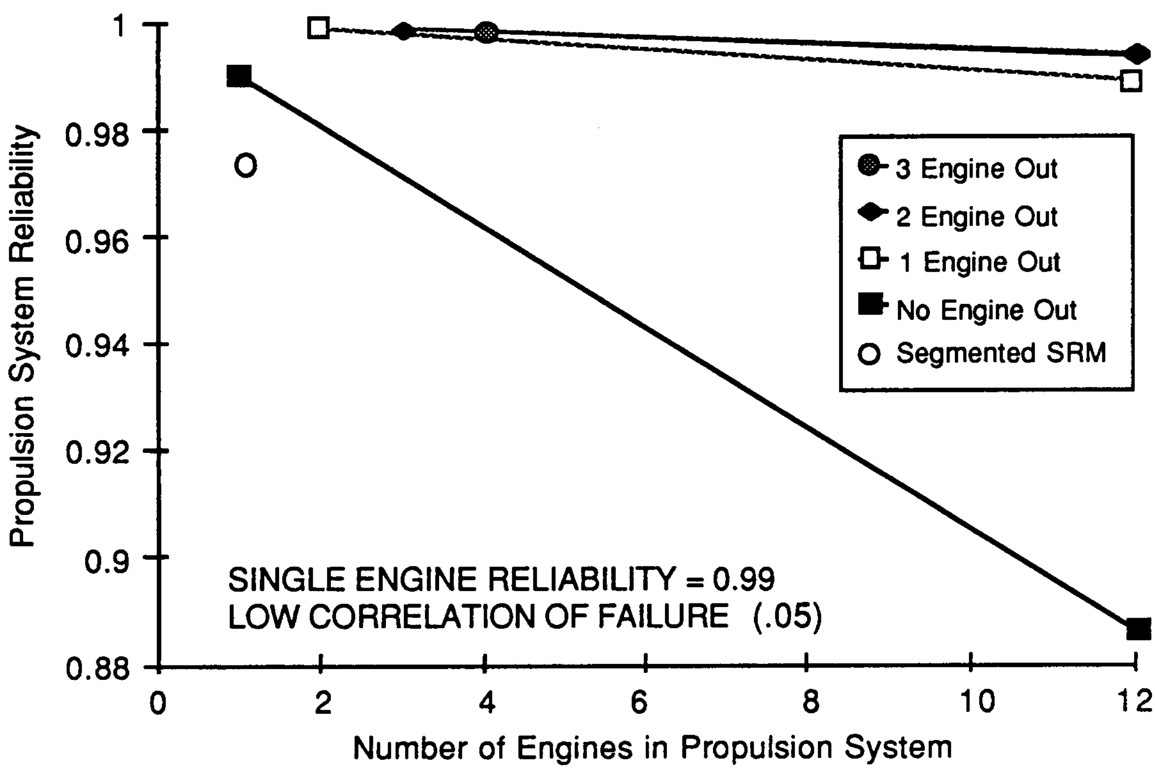


Figure 3.2.2-1 Number of Engine(s)-out capability needed for LRB

With engine-out, the propulsion system reliability achieved for a smaller number of engines system is higher than for a single engine system. There is diminishing returns after single engine out capability. Without a payload mission model, (to provide payload data), it is not possible to determine whether it pays to have complete mission success with one engine out. However, it is clear that from a safety consideration, which is the main reason for considering LRBs, that an abort to orbit (ATO) with one engine out should be the basic goal of the LRB.

Also shown for reference is the historical reliability of the segmented SRM data. This shows that the demonstrated reliability of STS with LRB can be higher than that with the current SRB.

3.2.3 OPTIMUM NUMBER OF ENGINES PER BOOSTER. Flight control analysis showed that for two engines, unacceptable throttling capability of the engine, and large gimbal angles and rates were required to have proper control of the vehicle in case of engine out under worst case conditions (Examine Figure 3.2.3-1). Here the peak LRB yaw TVC deflections and rates required to counteract an engine out at max Q for various numbers of engines are shown. Pitch plane results are not shown as the crosswind disturbances primarily affects the yaw plane, and hence gimbal requirements in that plane are smaller.

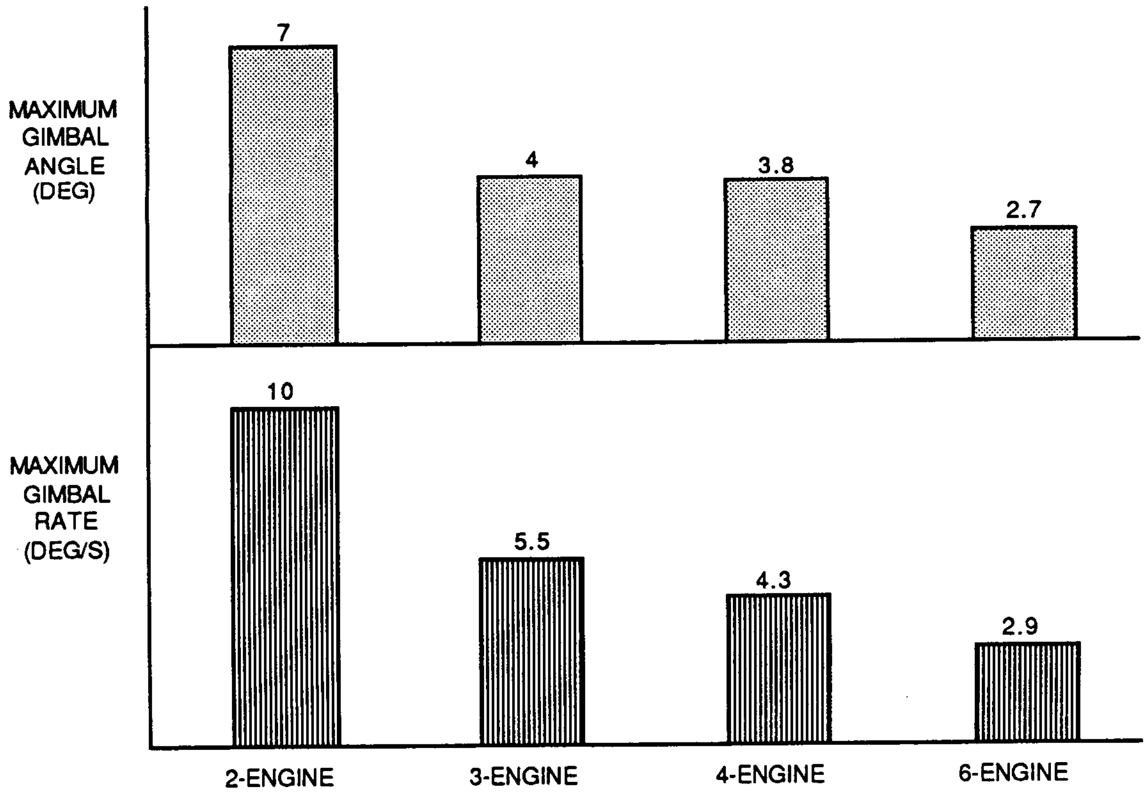


Figure 3.2.3-1 Engine-out Gimal requirements

Hence three, four, and six engines were identified as the most viable alternatives based on the current rectangular shape of the flame trench.

The top level assumptions/requirements made for determining the optimum number of engines are given below:

1. Vehicles sized should have ATO capability with one engine out. The rationale for this requirement has been discussed previously.

2. Flame trench constraint does not impact trade results. This is based on a sensitivity study done for the pressure-fed concept regarding impact of exit diameter on vehicle size. The pressure fed concept is most sensitive to the exit diameter constraint because of its lower operating pressure. As shown in Figure 3.2.3-2, LRB size changed by less than 0.5% between 90 inches and 114 inches exit diameter.

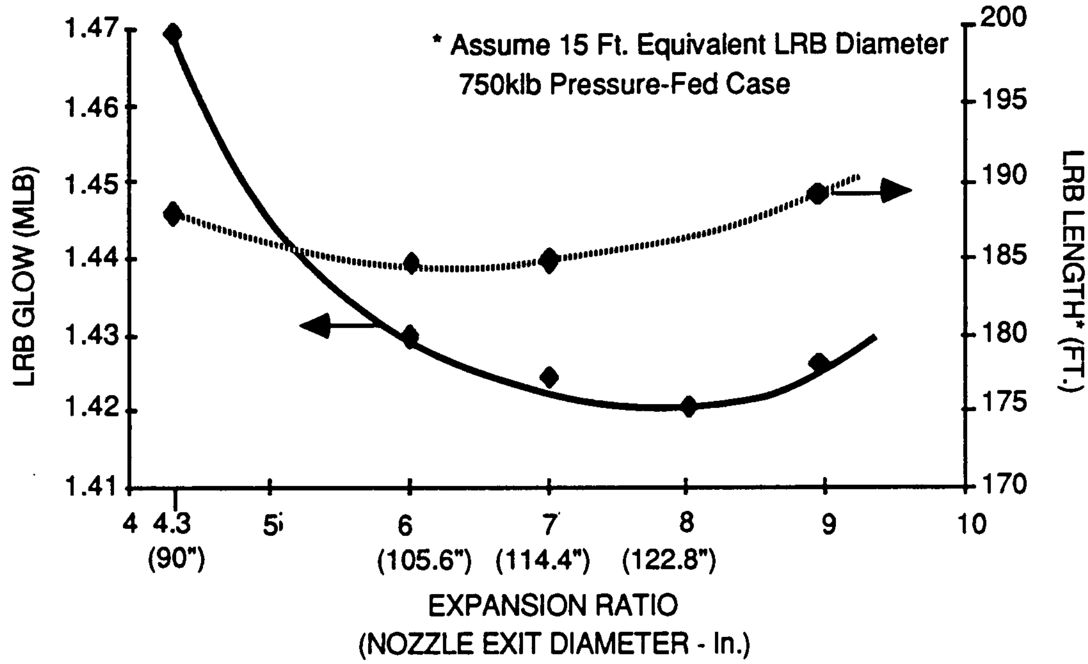


Figure 3.2.3-2 LRB Length & Weight vs Expansion Ratio for 4-Engine Pressure-Fed LO2/RP1 LRB

3. Vehicle T/W with one engine out is 1.25 for ATO capability. This was based on earlier studies which showed that with a engine-out T/W of 1.25, one can make an ATO without significant change in vehicle size (Current T/W with one engine out is 1.2. This is based on minimum cost. This change in assumption should not affect our conclusions).

4. All engines will have 99% demonstrated reliability before first flight (as discussed before).

5. The LRB with no engine out will throttle its engines to balance propellant consumption. This is based on the rationale that flying with large differential thrust, or shutting down a good engine is not desirable.

Trade results are shown in Table 3.2.3-1 are mainly based upon analyses conducted using the LO2/RP1 pump-fed booster. However, trends presented here are considered valid for other selected LRB concepts, (any significant difference between concepts is discussed).

Table 3.2.3-1 Number of Engines Trade Results

CRITERIA	3 ENGINES PER LRB	4 ENGINES PER LRB	6 ENGINES PER LRB
SAFETY/RELIABILITY: • Nominal Mission • ATO - One Engine-Out	.9414 .9957	.9227 .9935	.8864 .9884
STS COMPATIBILITY (OF THE ENGINE SYSTEM): • Complexity (Ground/Flight Operations) • Base Heating (Heat Load To Orbiter Body Flap)	LOWEST About 10% Increase In Heat Load Compared To 4 Engine Case; Still Less Than SRBs	MEDIUM	HIGHEST About 10% Increase In Heat Load Compared To 4 Engine Case; Still Less Than SRBs
PERFORMANCE: • Total Engine Weight Per LRB	Approx. Same	Approx. Same	Approx. Same
COST • LRB Engine DDT&E Cost • Engines Recurring Cost Per LRB	High (~$1100 M) Approx. Same	Medium (~$830 M) Approx. Same	Low (~$640 M) Approx. Same
TECHNICAL RISK • Throttleability	~49%	~35%	~25%

The criteria by which the number of engines was chosen is summarized below. These criteria are the same as those used for the configuration trade studies, and are listed in order of importance.

Safety/Reliability. The reliability of the propulsive system to accomplish a given mission diminishes as the number of engines increases. To improve safety, or better the chances of saving the crew and payload in the event of an engine failure, it is desirable to have engine-out capability. If engine out capability is designed into the booster, the reliability of the propulsion system to meet the desired mission is improved; examine Figure 3.2.2-1. The GD goal is to size the LRBs such that if a booster engine fails during ascent, it is still possible for the orbiter to deliver full payload to a reduced "safe" orbit and return the crew. Table 3.2.3-1 shows reliability values with and without engine-out capability. Because high reliability is desired, the basic conclusion can be drawn that a four engine arrangement is preferred over a six engine arrangement.

STS Compatibility. The quantity of LRB engines used affects the MLP/flame trench, plume/base heating, aerodynamic drag, control of the mated vehicle, and ground/flight operations.

For our initial trade studies, free plume expansion in the MLP was assumed to be similar to the SSMEs, and the LRB nozzle diameter was constrained such that the plume from the LRB engines struck the flame deflectors located over the flame trench in the same manner as the SRBs. This low risk approach allowed a maximum exit diameter of 90 inches. Optimum pump-fed engine performance can be achieved within this limitation. However, the pressure-fed engine performance (for 4 engine LRBs) optimizes with nozzle diameters over 90 inches (see Figure 3.2.3-2); if 6 engines are used on the LRBs it is easier to optimize engine performance within the 90 inch nozzle limit. Because the 4 engine pressure-fed booster optimizes with nozzle diameters greater than 90 inches, we have asked our subcontractors, PRC and Rocketdyne, to assess the possibility of using nozzle diameters greater than 90 inches. We feel that by shaping the MLP flamehole side walls and modifying the flame deflectors it will still be possible to channel the exhaust into the flame trench. However, scale model testing will be required to verify/prevent overpressure wave impingement on the engines or interference with their operation. Hence, although 6 engines are better suited for for the 90 inch diameter limit, currently no major impact is foreseen in increasing the exit diameter beyond 90 inches to get optimum size/performance using 4 engines.

An initial assessment made by Eagle Engineering suggests that the plume radiative heating to the orbiter body flap with engines aligned in a vertical row, rather than a clustered about the booster centerline is more severe (~10%). To fit within the geometry of the flame trench, the row layout is better suited for the 6 engine case (examine Figure 3.2-1). However, for either engine layout (in a row or clustered around the centerline), the LRB base heating rate will be less than the current SRBs.

The aerodynamic drag of a 3 or 6 engine LRB is expected to be greater than that of the same booster using 4 engines due to the larger aft skirt area (assuming the 6 engines are aligned in a row as presented in Figure 3.2-1). Presently vehicle control does not pose any problem for all three number of engine options. For comparison, engine out gimbal angle were calculated using the RP1 pressure-fed booster with 3,4, and 6 engines (see figure 3.2.3-1). The worst case was the three engine case, and the largest gimbal angle for engine out at maximum dynamic pressure was less than 5 degrees.

Ground/flight operational complexity will increase with increasing number of engines. In terms of ground operations, additional test and checkout will be required for additional engines, actuators, feedlines and avionics. In terms of flight operations, additional software development, will be required as the number of engines increases. Additional costs due to increased operational complexity as the number of engines multiplies have not yet been evaluated.

Performance. The weight of the engines increases slightly with increasing number of engines (after 4). Yet even with inclusion of accessories, the difference in weight is quite small. Thus the impact on the vehicle weight and size for a given payload requirements is negligible.

Cost. The approximate change in engine DDT&E cost and manufacturing cost with change in number of engines are shown in Table 3.2.3-1. As expected, DDT&E cost per engine decreases with an increase in the number of engines used per booster, because of decrease in engine thrust. There is not much of a change in engine manufacturing cost per LRB as the number of engines changes.

Technical Risk. The approximate throttling range for various numbers of engines (with and without engine-out) are shown in Table 3.2.3-1. An accepted rule of thumb in the industry is that 35-40% throttling is readily achievable. Any higher range imposes significant technological risk and cost. For the RP1 pump-fed booster used in this comparison, throttle ranges for both the 4 and 6 engine configurations fall within this range, but the 3 engine case requires ~49% throttling.

Conclusion. Safety and reliability are improved if the minimum multiple number of engines is used per LRB (while still retaining engine-out capability). Three, and lower number of engine configurations, pose unacceptable technical risk because of throttling requirements. A 6-engine configuration is poorer than 4 engines in terms of safety/reliability, overall vehicle complexity, and STS compatibility. As safety, reliability, and STS compatibility are the premier criteria for judging options on this program, we conclude that 4 engines per LRB is the best number of engines to use.

3.3 THRUST TO WEIGHT

Thrust to weight for nominal mission is determined by the minimum value of ≥ 1.2 required to clear the tower with a single LRB engine failure at liftoff, the propellant combination, and the type of engine. (Our cost optimizatioon trade in section 3.7.1 shows that for minimum cost vehicle, thrust to weight with one engine out should be 1.2). Typically the vlaue is between 1.5 and 1.6 for the combinations studied. The process is presented in section 8 of this volume.

3.4 AVIONICS ARCHITECTURE TRADE

The top objectives for the LRB avionics are to provide an avionics system which will command and monitor the more complex engine systems, thrust vector control systems, pressurization systems, and fluid systems, which are associated with an LRB, improve STS system reliability, minimize Orbiter software/hardware impacts, and provide for LRB growth potential. The Orbiter impacts are minimized by making the LRB avionics autonomous, thus satisfying the requirements for engine control and fuel/oxidizer pressure control, and by making the LRB avionics "smart" such that information sent to, and commands received from, the Orbiter avionics are in a format similar to SRB. The reliability increase can be achieved with the use of redundancy within the LRB avionics system. The growth potential will be provided by implementing a flexible system which may be used to support other potential vehicle applications.

The avionics downselection was performed on three architectures and was based on how well each architecture met the trade criteria of LRB subsystem control, avionics system reliability/redundancy, and Orbiter impact minimization. Other criteria such as cost, operability, maintainability, weight, power, etc. were considered but were not deemed to be significant discriminators for this trade. The three major requirements which had to be met by the LRB were (1) The system shall be single fault tolerant in accordance with document NSTS-07700, Vol. 10, Para. 3.5.1., (2) Impacts to the Orbiter and Orbiter systems are to be minimized, (3) The system shall have an initial operation date of 3rd quarter 1994 for the pump fed, and 3rd quarter 1995 for the pressure fed concept.

One of the avionics system architecture concepts that was investigated used the equipment on, and an architecture incorporated from the SRB system making it compatible with functional requirements of the LRB. The second architecture concept uses current technologies and units that are being developed at this date. This system uses a fully autonomous (from the Orbiter) fault tolerant architecture to yield high reliability and full control of the LRB subsystem. The third architecture investigated uses advanced technology modular units and architectures, such as the Multi-Path Redundant Avionics Suite (MPRAS), to obtain a highly fault tolerant and autonomous architecture. These systems are presently not available but are under development.

The first concept of using SRB based equipment and architecture (Figure 3.4-1) was not selected because of its inability to provide LRB vehicle autonomy and because of the outdated technology of the avionics units. The lack of LRB autonomy in the first architecture would have dictated impacts to the Orbiter for engine control software, engine shutdown events, pressurization control, etc. Each of these functions would require data to be sent to the GPC's, hence increasing the electrical interface requirement. Orbiter design would be affected as well as the increased number of Orbiter vehicle validation tests. The third concept (Figure 3.4-2) was not selected due to its larger developmental and schedule risk (the IOC for MPRAS technology is 1996) in relation to the LRB development and launch schedule (IOC 1995).

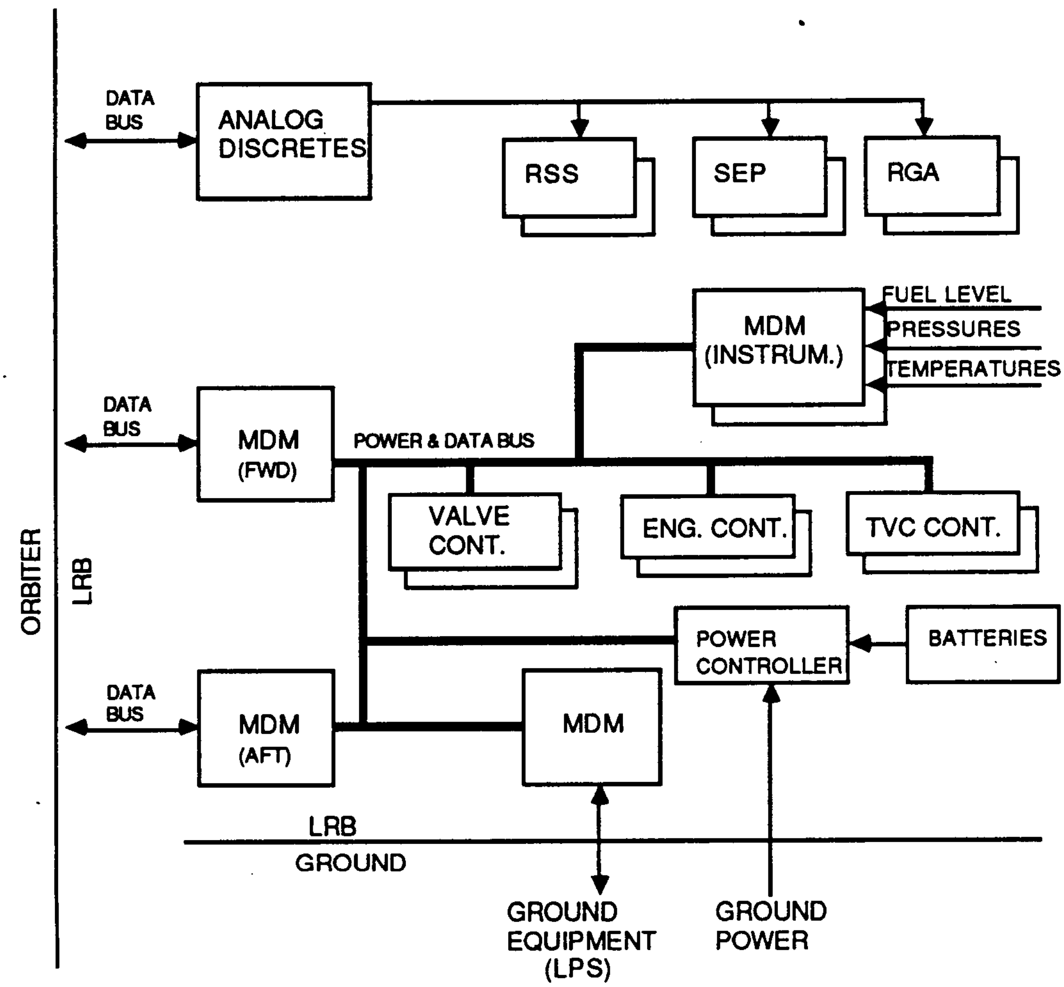


Figure 3.4-1 LRB Avionics Concept Based on SRB Architecture

The second architecture (Figure 3.4-3) was selected because it minimizes changes to the Orbiter interfaces through system autonomy. Changes are further minimized by incorporating "smarts" into the system such that data going to the Orbiter avionics and commands coming from the Orbiter avionics can be manipulated to insure consistency with the present Orbiter data protocol. Because of the improved technologies and implementation of failure tolerant techniques, high reliability is achieved. Flight control commands such as engine start/shutdown, TVC, and separation will come from the Orbiter avionics as is presently done for the SRB's. This will allow the use of the

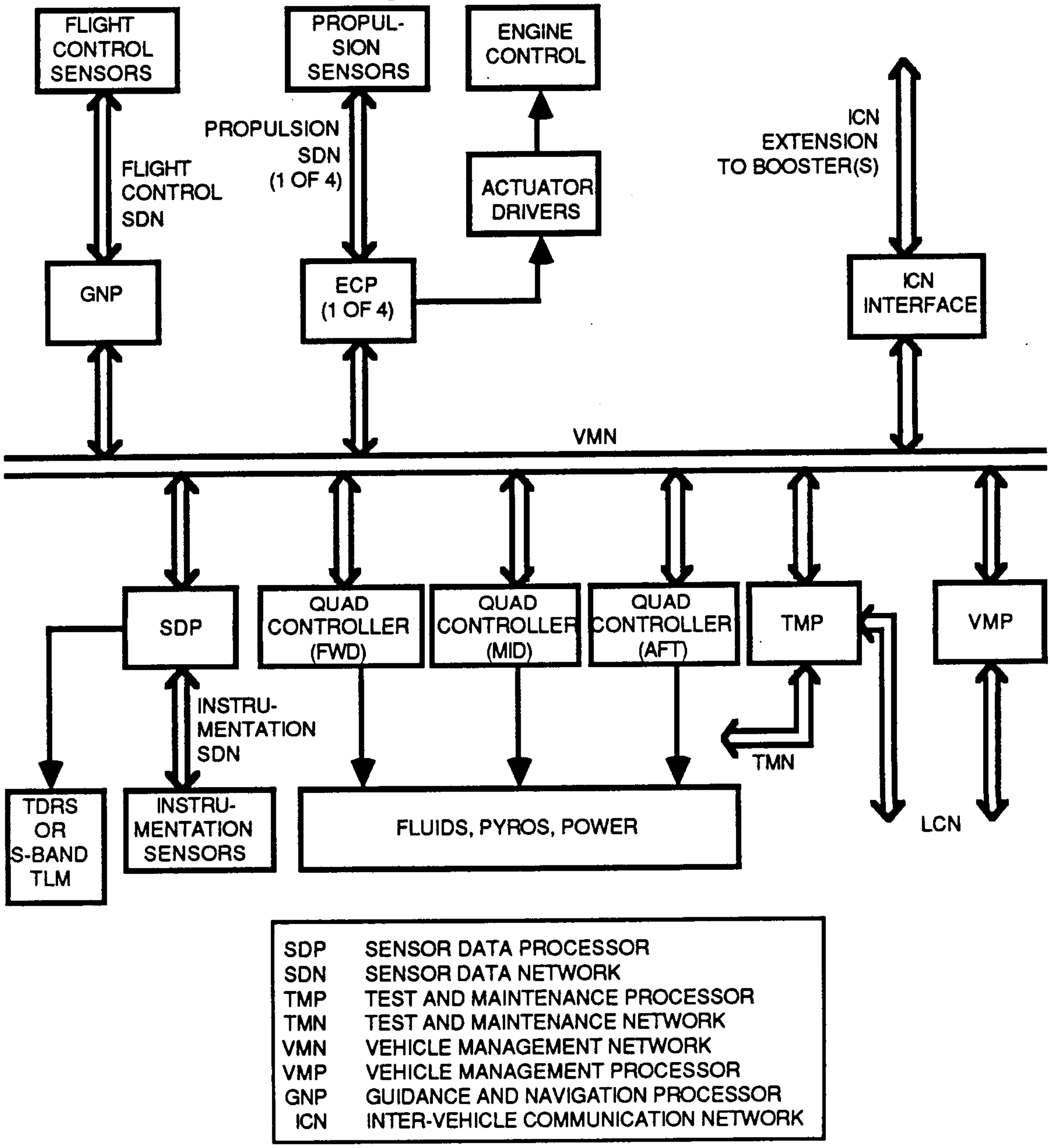


Figure 3.4-2 LRB Avionics Concept Based on Multi Path Redundant Avionics

existing command lines. The LRB avionic system will perform all LRB subsystem command and monitoring, advising the Orbiter of problems by providing flight critical data to the Orbiter. Therefore, proper action can be taken if failures occur. The LRB will transmit telemetry data independently to the ground via an on board RF telemetry system.

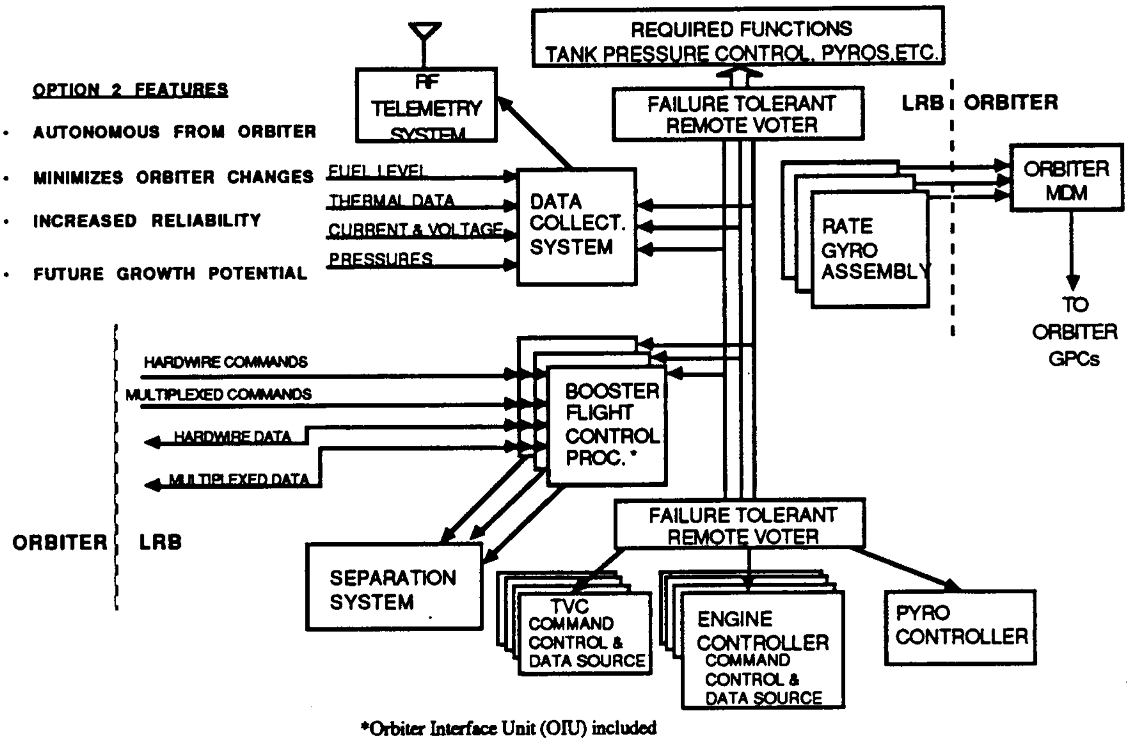


Figure 3.4-3 Baselined LRB Avionics Architecture Derived From Modern Centaur Avionics

The features of each architecture, along with a relative evaluation, are summarized in Figure 3.4-4.

During the Low Cost LRB study several options that arose for a lower cost avionics suite were.

1) Modified Triple String
2) Primary/Backup - BCPs on LRBs
3) Primary/Backup - BCPs on Orbiter
4) Dual/Dual - BCPs on LRBs
5) Dual/Dual - BCPs on Orbiter

The above listed options were used to perform a trade study to determine the most cost effective system.

The Dual/Dual system and the modified three string system, shown in Figure 3.4-5 and 3.4-6 respectively, were the two most promising candidates. Option numbers 3 and 5 (BCPs on Orbiter)

CRITERIA / OPTIONS	SRB BASED	MODERN AVIONICS BASED	MPRAS BASED
VEHICLE INTERFACE IMPACTS	H/W - HIGH • MORE CONTROL LINES • MORE ATVC CHANNELS • MORE DATA LINES • HIGHER POWER REQS. S/W - HIGH • COMPLEX BOOSTER ENGINE CONTROL/ MONITOR • PRESSURE CONTROL • CAUTION & WARNING	H/W - LOW • AUTONOMY INCORP- ORATED AS NEEDED S/W - MEDIUM • CAUTION & WARNING • BOOSTER TO BOOSTER COMMUNICATION	H/W - LOW • AUTONOMY INCORP- ORATED AS NEEDED S/W - MEDIUM • CAUTION & WARNING • BOOSTER TO BOOSTER COMMUNICATION
GROUND INTERFACE IMPACTS	H/W - LOW • THROUGH ORBITER S/W - MEDIUM • BOOSTER CHECKOUT • BOOSTER MONITOR/ COMMAND	H/W - MEDIUM • BOOSTER UMBILICALS • BOOSTER COMM. S/W - MEDIUM • BOOSTER CHECKOUT • BOOSTER MONITOR/ COMMAND	H/W - MEDIUM • BOOSTER UMBILICALS • BOOSTER COMM. S/W - MEDIUM • BOOSTER CHECKOUT • BOOSTER MONITOR/ COMMAND
OPERATIONAL COMPLEXITY	HIGH • ORBITER DEPENDANT	LOW • AUTONOMOUS	LOW • AUTONOMOUS
PROGRAM RISK			
SCHEDULE	LOW	LOW	HIGH
AVAILABILITY	MEDIUM	LOW	MEDIUM
RELIABILITY	MEETS REQUIREMENT	MEETS REQUIREMENT	MEETS REQUIREMENT
COST			
DEVELOPMENT	MEDIUM	LOW/MEDIUM	HIGH
RECURRING	LOW	LOW/MEDIUM	LOW
GROWTH POTENTIAL	LIMITED	GOOD	EXCELLENT

Figure 3.4-4 Avionics Architecture Trade Evaluation Summary

were attractive from a recurring cost standpoint, however these approaches conflicted with our goal of minimized Orbiter impacts and hence were not considered further. The two string option with the BCPs on the LRB was discarded due two the higher implementation complexity especially in the area of redundancy management software. The Dual/Dual system had a lower recurring cost of the remaining two choices, however the modified three string system had commonality with developments presently under way for other launch systems. The commonality factor significantly lowers the non-recurring cost of the system therefore the three string system as shown in Figure 3.4-6 was chosen. Figure 3.4-7 summarizes the evaluation of the Low Cost LRB avionics trade.

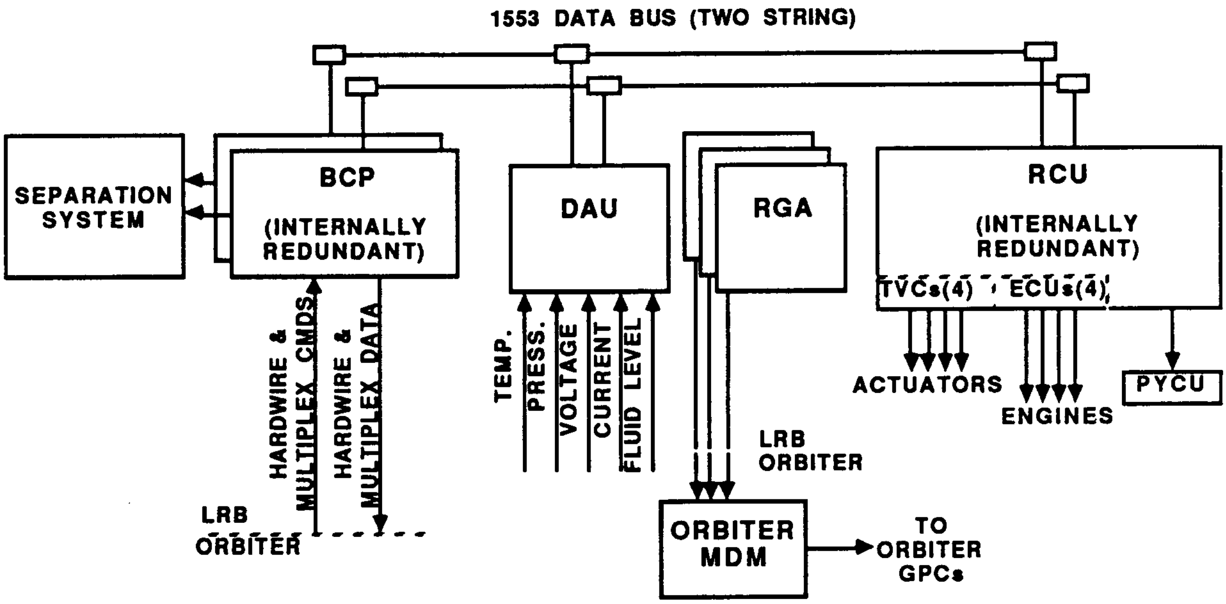


Figure 3.4-5 Low Cost Avionics Trade Option 4: Dual/Dual Redundancy with BCPs on the LRB

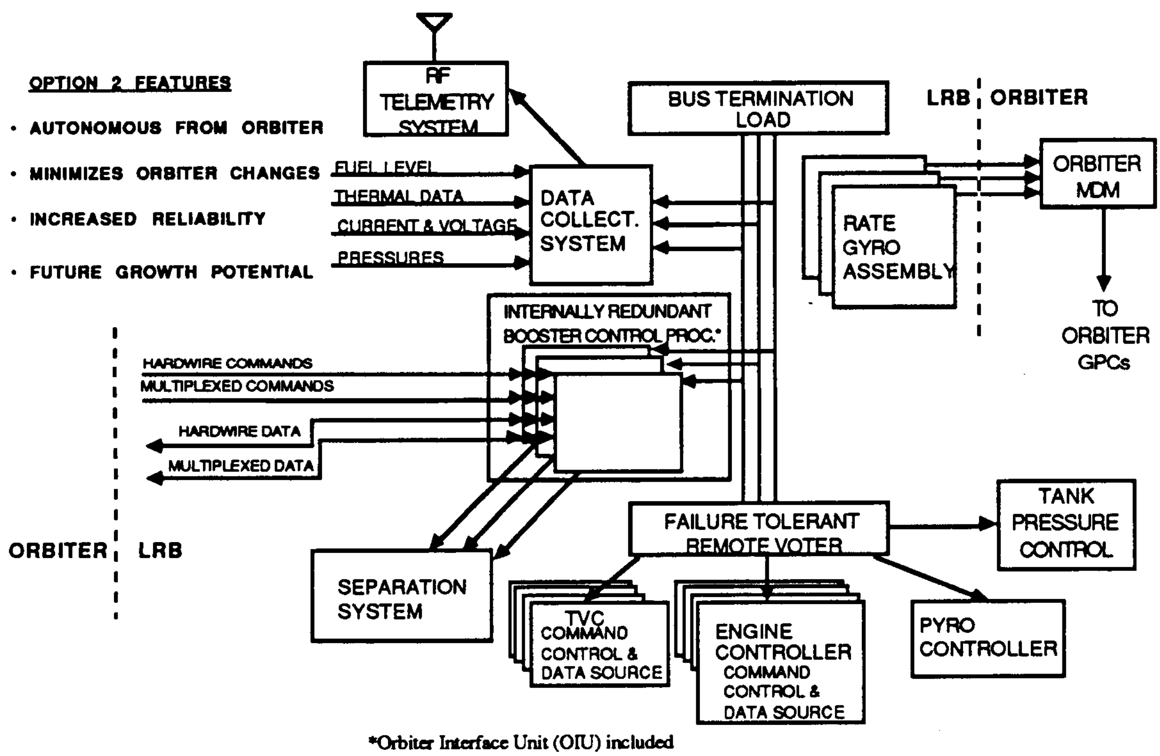


Figure 3.4-6 Low Cost Avionics Trade Option 1: Modified Three String System

OPTIONS \ FACTORS	IMPLEMENTATION COMPLEXITY	RELIABILITY	AVG UNIT COST	AUTONOMOUS FROM ORBITER	WEIGHT PENALTY	RECOVERABLILITY	COMMONALITY WITH OTHER VEHICLE SYSTEMS
TRIPLE STRING	LOW	0.99999	2.64M	YES	NONE	NONE	YES
PRIMARY/BACKUP BCPs ON LRBs	MODERATE	0.99977	1.97M	YES	NONE	NONE	NO
PRIMARY/BACKUP BCPs ON ORBITER	HIGH	0.99977	1.71M	NO	50lbs	BCPs(2)	NO
TWO STRING BCPs ON LRBs	LOW	0.99999	2.22M	YES	NONE	NONE	NO
TWO STRING BCPs ON ORBITER	HIGH	0.99999	1.77M	NO	100lbs	BCPs(4)	NO

SELECTED BASELINE DESIGN

Figure 3.4-7 Low Cost LRB Avionics Trade Evaluation Summary

3.5 PROPELLANT SELECTION TRADE

In the initial phase of the LRB study, we carried out several trade studies to select the best propellant/engine candidates for the pump-fed and pressure-fed LRB concepts. Environmental acceptability/safety was one of the primary selection criteria. Initially, LRB size was also considered as one of the top criteria, as it was thought to be the barometer for STS compatibility; our work in later phases did not show such a direct correlation. Operational complexity and compatibility with the current facilities, operational availability or technology risk, and cost were the other relevant criteria for the propellant selection.

All the propellant concepts considered are shown in Figure 3.5-1. The selection of these propellant concepts was based on the past and current engine/vehicle studies sponsored by NASA/MSFC and AFAL. As shown in the trade tree, the pump-fed concepts were considered in three different groups, viz., existing pump-fed engines, new conventional (stage-combustion and gas-generator) pump fed engines, and split expander engines; each group had its own unique advantages and disadvantages. The existing engines seemed to have early IOC capability, but they might not be compatible with LRB goals; conventional pump-fed engines could be designed to completely satisfy the LRB goals; and the split expander cycle, which is an innovative modification to the expander cycle, was added to the study by MSFC because of its potential as a low cost and highly reliable engine.

Environmental impact assessments for the various propellants, with a comparison to the current SRB, are provided in Table 3.5-1. Except for NTO/MMH, all other LRB propellants were clearly more acceptable in this prime category. Also note that a single Shuttle launch using storable propellants would require five times the amount used in a Titan launch, and this posed concerns on propellant availability. Therefore, NTO/MMH propellant concept was rejected very early eventhough it gave a booster size about the same as the current SRB.

During the basic contract phase of the study, in consensus with MSFC, four propellant/engine concepts were selected. These were: LO2/RP1 and LO2/LH2 with GG cycle engine, LO2/CH4 with split expander cycle engine, and LO2/RP1 with pressure fed engine.

During the extension phase, these four concepts were optimized. Engine data was updated based on preliminary analysis of combustion stability, bottoms-up weight and cost estimates, and work done on STME/STBE contracts. It was found that LO2/LH2 Split Expander engine gives almost same size vehicle as LO2/LH2 GG engine. This is because of no cycle losses and lower weight of the split expander engine, even though the operating pressure of this engine is about half of the GG

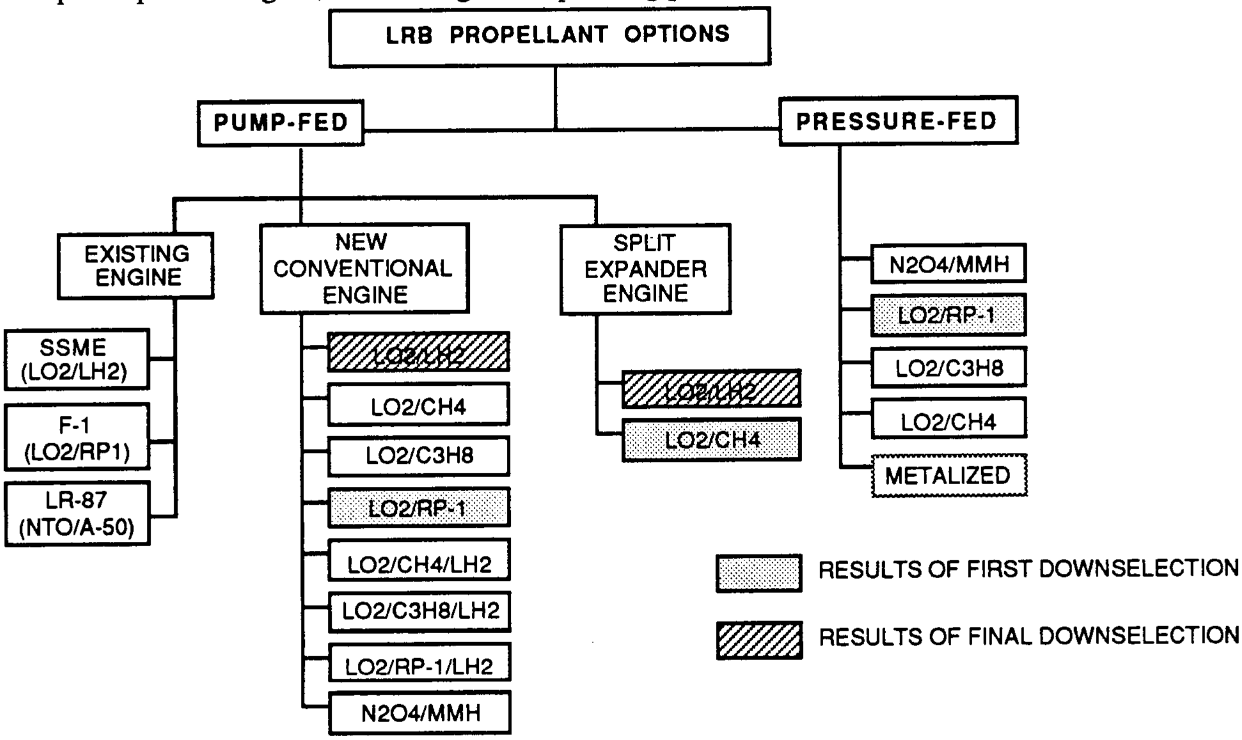


Figure 3.5-1. LRB Propellant Trade Tree

engine. The Split Expander Cycle engine is kept as an alternate engine to GG engine as it shows promise of lower cost and higher reliability, but needs technology demonstration. A final downselection to LO2/LH2 propellant was carried out at the end of extension phase. The rationale for initial and final downselections are discussed below.

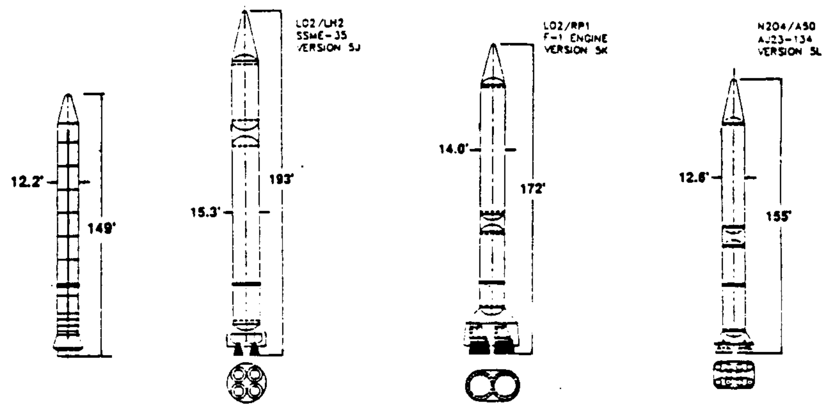
Existing Engine Pump-Fed LRB Concepts. Existing pump-fed engines were considered as a possibility because of achieving earlier IOC and lower DDT&E costs. Figure 3.5-2 shows the size and characteristics of various concepts in this category. The vehicles were sized with a length to diameter ratio of 12.3, same as the current SRB. These vehicles were sized to payload requirement in the earlier part of the study (70 klbs. to 150 nm). The slight change to current requirement and current L/D should not have any impact on the results of this trade study.

The results of the engine evaluation are summarized on Table 3.5-2. None of the engines in this category was found suitable for LRB as discussed below.

Table 3.5-1 Environmental Impacts Evaluation for LRB Propellant Options

PARAMETER	SRB	LO2/LH2	NTO/MMH
ATMOSPHERE QUALITY	HARMFUL PRODUCTS: AL, AL2O3, HCL; AFFECT OZONE LAYER	MINIMAL IMPACTS - BENIGN PROPELLANTS & PRODUCTS (H2O)	SEVERE IMPACTS - TOXIC & CORROSIVE PROPELLANTS; BENIGN PRODUCTS
WATER QUALITY	ACIDIC WATER & SOLID PARTICULATE CONTAMINANTS	MINIMAL IMPACTS	SEVERE IMPACTS - WATER & PROPELLANT REACTIONS FORM TOXIC BY-PRODUCTS
PLANTS & ANIMAL LIFE	HIGH IMPACTS	MEDIUM IMPACTS	SEVERE IMPACTS - ESPEC. IN CASE OF EXPLOSION OR ACCIDENT
NOISE/ACOUSTICS LEVEL	HIGH IMPACTS	HIGH IMPACTS • PUMPFED SLIGHTLY BETTER • SMALLER OVER-PRESSURE	HIGH IMPACTS • PUMPFED SLIGHTLY BETTER • SMALLER OVER-PRESSURE
SPILLS	NONE	EXPLOSIVE HAZARDS; (LH2 TO FLAME STACK)	SEVERE HAZARDS: TOXIC, HYPERGOLIC & CORROSIVE; EXT. SAFETY PROCEDURES
TRANSPORTATION	NO IMPACT; ON-SITE PRODUCTION	DOT APPROVAL REQ'D FOR LARGE QUANTITY	DOT APPROVAL REQ'D; STRICTLY ENFORCED RULES

PARAMETER	LO2/CH4	LO2/C3H8	LO2/RP1
ATMOSPHERE QUALITY	LOW CONC. OF SMOG FROM CO & CO2; MAY AFFECT OZONE LAYER	SMOG FROM CO & CO2 PRODUCTS; MAY AFFECT OZONE LAYER	SMOG FROM CO & CO2 PRODUCTS; MAY AFFECT OZONE LAYER
WATER QUALITY	SMALL EFFECT DUE TO CO & CO2 (SOOTS) PRODUCTS	SOME EFFECT DUE TO CO & CO2 (SOOTS)	SOME EFFECT DUE TO CO & CO2 (SOOTS); RP1 CONTAMINATION
PLANTS & ANIMAL LIFE	HIGHER IMPACTS THAN LO2/LH2, LESS THAN OTHER LRB OPTIONS	HIGHER IMPACTS THAN LO2/LH2, LESS THAN SRB & HYPERGOLS	HIGHER IMPACTS THAN LO2/LH2, LESS THAN SRB & HYPERGOLS
NOISE/ACOUSTICS LEVEL	HIGH IMPACTS • PUMPFED SLIGHTLY BETTER • SMALLER OVER-PRESSURE	HIGH IMPACTS • PUMPFED SLIGHTLY BETTER • SMALLER OVER-PRESSURE	HIGH IMPACTS • PUMPFED SLIGHTLY BETTER • SMALLER OVER-PRESSURE
SPILLS	EXPLOSIVE HAZARDS	EXPLOSIVE HAZARDS	CLEAN-UP OF RP1 REQ'D; FIRE HAZARDS
TRANSPORTATION	DOT EXEMPTION REQ'D	DOT APPROVED	DOT APPROVED



CHARACTERISTICS	SRB REF.	SSME ENGINES	F-1 ENGINES	AJ-23 ENGINES
GLOW (LRB/ORB/ET)	4,500K	3,214K	4,242K	4,132K
LRB PROP. WT. (lb)	1,100K	549K	1,140K	1,043K
LRB ENG. THRUST (SL, lb)	3,000K	454K	1,309K	430K
Isp (VAC., sec)	265	442	304	285
MIXTURE RATIO	-	6.0	2.3	1.9

Figure 3.5-2 Size Comparison for Pump-Fed LRB with Existing Engine Options

Table 3.5-2 Evaluation for Pump-Fed LRB with Existing Engine Options

CRITERIA	SSME-35 (LO2/LH2)	F-1 (LO2/RP1)	LR87 (NTO/A50)
SAFETY	HIGH EXPLOSIVE HAZARD; CRYOGENS	LEAST EXPLOSIVE HAZARD; STORABLE FUEL	SEVERE HAZARDS: TOXIC, HYPERGOLIC, CORROSIVE, CARCINOGEN; STORABLES
DESIGN CONSIDERATIONS			
• **No. of Engines/LRB**	4 - GOOD FLT CONTROL	2 - MARGINAL FLT CONTROL	5 PAIRS - TOO COMPLEX
• **Ignition Complexity**	MEDIUM	HIGH	LOW
• **Throttling Range**	65 TO 109% RPL	80 TO 120%	-
• **Reusability**	55 FLIGHTS (DESIGN VALUE)	20 FIRINGS	-
• **Engine Complexity**	HIGH	LOW	LOW
STS COMPATIBILITY	LARGEST SIZE; EXPERIENCED IN GRND OPS; HIGH COMPLEXITY W/LH2, EG. PURGE,LEAK DETECTION	MEDIUM SIZE; EXPERIENCED IN GRND OPS; SIMPLE RP1 LOADING OPS & FACILITY SYSTEM	SMALLEST SIZE; VERY COMPLEX GRND OPS & TIME CONSUMING DUE TO SAFETY
PERFORMANCE, GLOW	3214 KLB	4242 KLB	4132 KLB
ENGINE COST/LRB	LOW NON-REC. VERY HIGH RECUR (~$100M)	TOOLING COST REQ'D; LOW RECUR (~$24M)	LOW NONRECURRING MEDIUM RECUR (~$40M)
PROGRAM RISK	LOW - CURRENT STS ENGINE	MEDIUM - OUT OF PRODUCTION NEED REQUALIFICATION	LOW - CURRENT TITAN ENGINE
IOC/OPERATIONAL AVAILABILITY	48 MONTHS	48 (BASIC) TO 72 MONTHS (THROTTLEABLE)	48 MONTHS; PROPELLANT AVAILABILITY CONCERN

• F-1 (LO2/RP1), requiring 2 engines per LRB, offered small booster size and low recurring cost.

However, it demanded a large throttling capability to accommodate engine-out, and this would require design changes that can result in longer lead time and high DDT&E cost. In addition, control requirements in case of engine-out were very high, even with throttling capability.

• SSME (LO2/LH2), 4 engines per LRB, seemed compatible with current STS and facilities, offered low risk and minimal environmental impact. However, it was not selected because of high recurring cost and operational complexity.

• AJ-23 (NTO/A-50), 5 pairs of engines per LRB, showed smallest booster size, low risk, high reliability and low cost, but it was eliminated based on propellant safety and environmental concerns, as previously mentioned.

<u>New Conventional Pump-Fed Engine Concepts.</u> New pump-fed engine concepts selected for LRB application were narrowed down to the gas-generator cycle for a low-risk conventional engine concept. This was based on STME and STBE studies, and our evaluation of the differences between the LRB requirements and these studies. The basic assumptions used are shown in Table 3.5-3. The relative size of the various propellant concepts is not only a function of propellant type, but also function of bulk density, (MR), and engine performance (chamber pressure, mixture ratio, and exit area). Current LRB designs have slightly different values of the parameters than shown in Table 3.5-3. The rationale for these assumptions and their impact on propellant/engine selection are discussed below.

An exit diameter of 50-in was assumed because this would ensure no impact to either the flame trench or the MLP hole. Our current engines have optimized nozzles, because it was later realized that a new MLP or major mods to the current MLP would be required anyway. Flame trench impact is not anticipated with these larger exit diameter nozzles. Results from the exit diameter sensitivity study are shown in Figure 3.5-3.

Table 3.5-4 summarizes the evaluation of all concepts. Figure 3.5-4 shows the size and characteristics of various options. Reasons for selection and rejection are discussed below.

• The storable and hypergolic propellant combination, NTO/Hydrazines, had several major advantages such as small booster size, low risk and high vehicle reliability. However, it was eliminated early in the study because of its highly toxic and corrosive nature which posed serious safety and environmental concerns, as indicated in Table 3.5.1.

Table 3.5-3 Major Assumptions in New Pump-Fed Engine/Propellant Trade

CONFIGURATION	=	CONVENTIONAL CYLINDRICAL (CURRENT SRB)
VEHICLE L/D RATIO	=	12.3
NUMBER OF ENGINES	=	4
TANK MATERIAL	=	AL-LI (WEIGHT OF TANK BASED ON LOAD CONSIDERATION)

FOR NEW ENGINES:

EXIT DIAMETER	=	50" (NO MODIFICATIONS IN MLP OR FLAME TRENCH)
	=	OPTIMIZED NOZZLE FOR 6 PSIA BACK PRESSURE (NO MODIFICATIONS TO FLAME TRENCH)
CHAMBER PRESSURE	=	BASED ON STBE NORMAL POWER LEVEL
MIXTURE RATIO	=	BASED ON STBE STUDY
ENGINE CYCLE	=	GAS GENERATOR

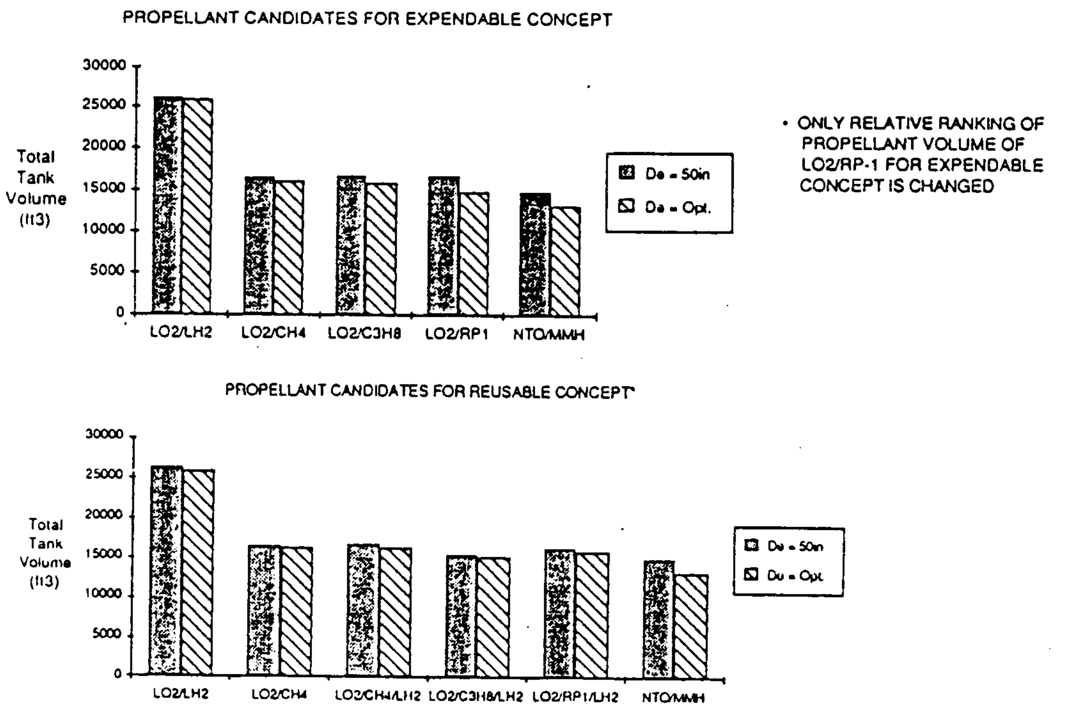


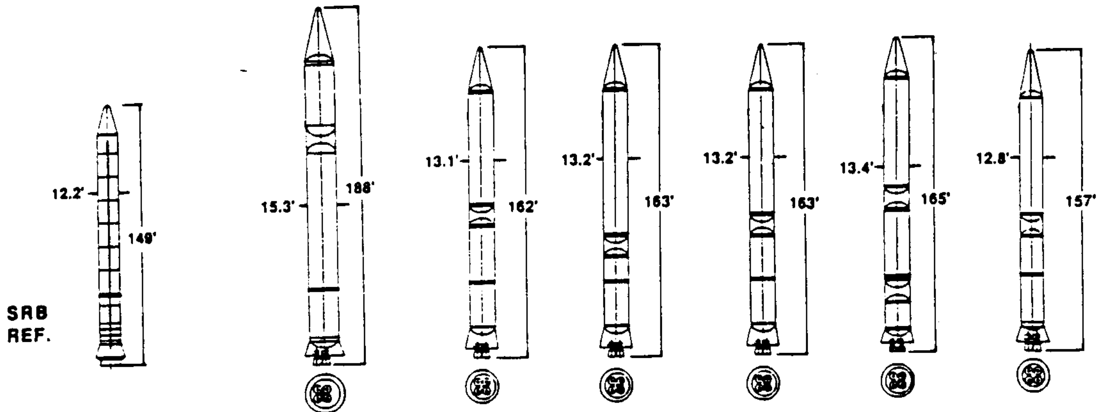
Figure 3.5-3 Effect of Nozzle Exit Diameter

Table 3.5-4. Evaluation for Pump-Fed LRB with New Conventional Engine Options

CRITERIA	LO2/CH4	LO2/C3H8	LO2/RP1
SAFETY	EXPLOSIVE HAZARD; CRYOGENS	HIGHER EXPLOSIVE HAZARD DUE TO HEAVY VAPOR	LOW EXPLOSIVE HAZARD; STORABLE FUEL
DESIGN CONSIDERATIONS			
• Chamber Pressure	2333 PSIA	2333 PSIA	1400 PSIA
• Nozzle Exit Diameter	73.4 IN	73.4 IN	101 IN
• Ignition Complexity	MEDIUM	HIGH	HIGH (BUT HAVE EXPERIENCE)
• Combustion Stability	HIGH	LOW	LOW
• Reusability	MEDIUM	LOW (COKING, CARBON DEP.)	LOW (COKING, CARBON DEP.)
• Engine Complexity	MEDIUM	LOW	LOW
STS COMPATIBILITY	MEDIUM SIZE; CRYOGENS - COMPLEX GROUND OPS; NO AEROSPACE EXPERIENCE	MEDIUM SIZE; LESS COMPLEX GROUND OPS; NO AEROSPACE EXPERIENCE	MEDIUM SIZE; EXPERIENCED IN GRND OPS; SIMPLE RP1 LOADING OPS & FACILITY SYSTEM
PERFORMANCE, GLOW	3642 KLB	3892 KLB	4129 KLB
PROGRAM RISK	MEDIUM - VERY LIMITED TEST & NO FLT EXPERIENCE	HIGH - VERY LIMITED TEST & NO FLIGHT EXPERIENCE	LOW - EXTENSIVE TEST & FLIGHT EXPERIENCE
IOC/OPERATIONAL AVAILABILITY	LATE	LATEST	EARLY (72 MONTHS)

CRITERIA	LO2/LH2	LO2/HC/LH2	NTO/MMH
SAFETY	HIGH EXPLOSIVE HAZARD; CRYOGENS	HIGH EXPLOSIVE HAZARD; CRYOGENS	SEVERE HAZARDS: TOXIC, HYPERGOLIC, CORROSIVE, CARCINOGEN; STORABLES
DESIGN CONSIDERATIONS			
• Chamber Pressure	2500 PSIA	3067 PSIA	1000 PSIA
• Nozzle Exit Diameter	82 IN	72.6 IN	LIMITED TO 90 IN
• Ignition Complexity	MEDIUM	MEDIUM	LOW
• Combustion Stability	HIGH	HIGH	HIGH
• Reusability	HIGH	HIGH	HIGH
• Engine Complexity	MEDIUM	HIGH	LOW
STS COMPATIBILITY	LARGEST SIZE; EXPERIENCED IN GRND OPS; HIGH COMPLEXITY W/LH2, EG. PURGE,LEAK DETECTION	MEDIUM SIZE; COMPLEX GRN OPS & ENGINE SERVICING DUE TO 3 PROPS	SMALLEST SIZE; VERY COMPLEX & TIME CONSUMING GRND OPS DUE TO SAFETY
PERFORMANCE, GLOW	3214 KLB	3500 KLB	4204 KLB
PROGRAM RISK	LOW - EXTENSIVE TEST & FLIGHT EXPERIENCE	HIGH- NO TEST OR FLIGHT EXPERIENCE	LOW - EXTENSIVE TEST & FLIGHT EXPERIENCE
IOC/OPERATIONAL AVAILABILITY	EARLY (~72 MONTHS)	LATEST	EARLY; PROPELLANT AVAILABILITY CONCERN

• All tri-propellant GG cycle engine options, which include LO2/CH4/LH2, LO2/C3H8/LH2 and LO2/RP1/LH2, were undesirable because of the disadvantages associated with a three propellant system, i.e. high operational complexity, engine technology risks, and cost.



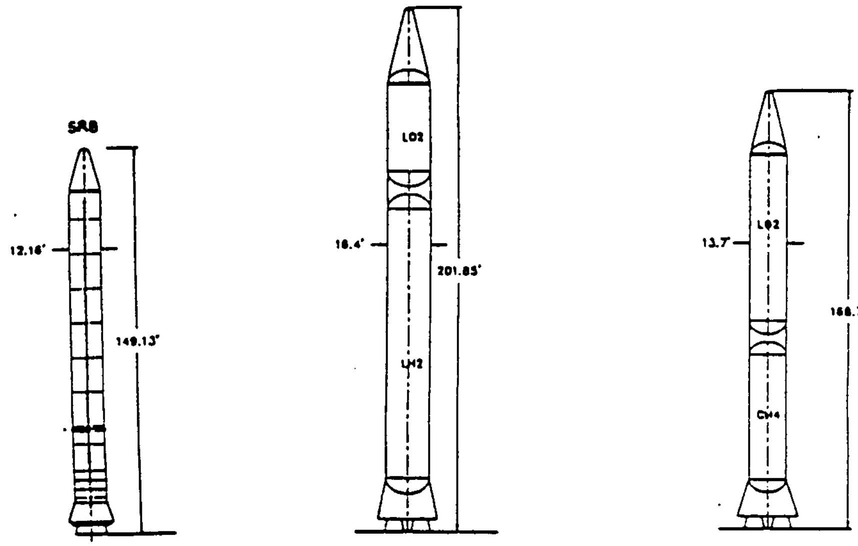
CHARACTERISTICS	O2/H2	O2/CH4	O2/RP1	O2/C3H8	O2/CH4/H2	MMH/N2O4
GLOW (LRB/ORB/ET)	3,227K	3,642K	4,129K	3,892K	3,512K	4,204K
LRB PROP. WT. (lb)	567K	794K	1,045K	921K	722K	1,090K
LRB ENG.THRUST (SL, lb)	404K	480K	569K	526K	456K	583K
LRB Isp (VAC., sec)	430	333	282	315	354	265
LRB MIXTURE RATIO	6	3	2.7	2.7	3.5	2.2

Figure 3.5.4 Size Comparison for Pump-Fed LRB with New Conventional Engine Options

• Among the hydrocarbon bi-propellants with GG cycle engines, i.e. LO2/CH4, LO2/C3H8 and LO2/RP1, only LO2/RP1 was selected based on STS compatibility, operational availability and simplicity, cost, and particularly, low engine development risk.

• LO2/LH2 with a GG cycle engine was also selected because of the propellant compatibility with current ET, low technology risk, and minimal environmental impact. But more importantly, its commonality in terms of engine requirements with other programs such as STME and ALS offers possible reduction of cost, i.e. rate-effect, and its application in the ALS program. The original concern regarding its large booster size, which can affect STS integration and aerodynamic wing loading, was resolved by results of facility assessments and load analyses.

Split Expander Pump-Fed Engine Concepts. A split expander cycle, a modified version of the expander cycle (RL-10), needs a low boiling point and high heat capacity fuel for its operation, and LO2/LH2 and LO2/CH4 were identified as the most viable concepts. These two propellant systems were sized using Haynes 230 for thrust chamber material, and the resulting sizes and characteristics are shown in Figure 3.5.5. Table 3.5.5 summarizes the evaluation of the two concepts. LO2/LH2 vehicle size was similar to that for the GG version. It was kept as an alternate engine concept because of : (1) need for technology demonstration, and (2) The promise of a lower cost and higher reliability engine. LO2/CH4 was selected as the baseline propellant for this cycle.



CHARACTERISTICS	SRB	LO2/LH2 SPLIT EXPANDER	LO2/CH4 SPLIT EXPANDER
STS GLOW (KLB)	4,500	3,561	3,864
PROPELLANT WT/LRB (KLB)	1,100	688	847
SL THRUST/LRB ENG. (KLB)	3,000	512	624
Isp,vac (SEC)	265	410	338
MIXTURE RATIO		6.0	3.5

Figure 3.5.5 Size Comparison for Pump-Fed LRB with Split Expander Engine Options

Pressure-Fed LRB Concepts. The propellant selection process for the presure-fed LRB concept was similar to that discussed for the pump-fed concept. Only hydrocarbon and hypergolic bi-propellant combinations, for higher bulk density and lower system weight, were considered for the pressure-fed application. Also included as a side option is the metalized propellant concept.

The basic assumptions made for Pressure fed concept trade are shown in Table 3.5-6. The chamber pressure and exit diameter were based on optimzation runs made for grapite-epoxy tanks. Our current baseline is Aluminum-2219, and hence values of these parameters are different. However, because of the similar impact on each vehicle, the trade results are still valid.

The relative sizes of all propellant candidates and the vehicle characteristics are shown in Figure 3.5-6, and their sizing assumed 400 psia in chamber pressure, graphite-epoxy tanks, and 90-in nozzle exit diameter.

Table 3.5.5 Evaluation for Pump-Fed LRB with Split Expander Engine Options

CRITERIA	LO2/LH2	LO2/CH4
SAFETY	EXPLOSIVE HAZARD; CRYOGENS	EXPLOSIVE HAZARD CRYOGENS
DESIGN CONSIDERATIONS		
• **Chamber Pressure**	968PSIA	758 PSIA
• **Nozzle Exit Diameter**	67 IN	107 IN
• **Ignition Complexity**	LOW	LOW
• **Combustion Stability**	HIGH	MED-HI
• **Reusability**	N/A	N/A
• **Engine Complexity**	LOW	LOW
STS COMPATIBILITY	LARGEST SIZE; CRYOGENS - COMPLEX GROUND OPS; EXTENSIVE AEROSPACE EXPERIENCE	LARGE SIZE; CRYOGENS - COMPLEX GROUND OPS; NO AEROSPACE EXPERIENCE
PERFORMANCE, GLOW	3425 KLB	3864 KLB
PROGRAM RISK	MEDIUM - RL-10 SIMILARITY; NO TEST OR FLT EXPERIENCE	MEDIUM - RL-10 SIMILARITY; NO TEST OR FLT EXPERIENCE
IOC/OPERATIONAL AVAILABILITY	LATE	LATEST

Table 3.5-6 Assumptions for Pressure-Fed Propellant Selection Trade

CONFIGURATION	= CONVENTIONAL CYLINDRICAL (CURRENT SRB)
VEHICLE L/D RATIO	= 12.3
NUMBER OF ENGINES	= 4
TANK MATERIAL	= GRAPHITE-EPOXY
PRESSURIZATION SYSTEM	= GAS GENERATOR HEATED HELIUM SYSTEM
EXIT DIAMETER	= 90 IN (NO MODIFICATION TO FLAME TRENCH)
CHAMBER PRESSURE	= 400 PSIA
MIXTURE RATIO	= FIXED (ISP OPTIMIZED) - SENSITIVITY RUNS SHOWED ONLY HIGHER ORDER EFFECT ON PROPELLANT VOLUME

Their assessments based on safety, design considerations, operations, performance, risk and availability, are shown in Table 3.5-7. Results from the pressure-fed propellant trade study are discussed below.

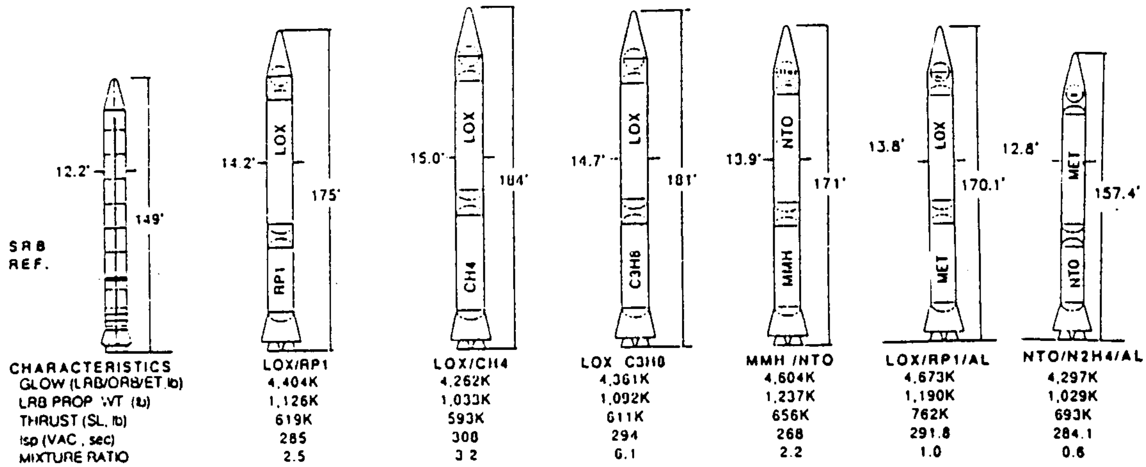


Figure 3.5-6 Size Comparison for Pressure-Fed LRB Propellant Options

Table 3.5-7 Evaluation for Pressure-Fed LRB Propellant Options

CRITERIA	LO2/CH4	LO2/C3H8	LO2/RP1	NTO/MMH
SAFETY	EXPLOSIVE HAZARD; CRYOGENS	HIGHER EXPLOSIVE HAZARD DUE TO HEAVY VAPOR	LOW EXPLOSIVE HAZARD; STORABLE FUEL	SEVERE HAZARDS: TOXIC, CORROSIVE, HYPERGOLIC; STORABLES
DESIGN CONSIDERATIONS				
• **Cooling Δ P (Regen)**	LOW	MEDIUM	HIGH	MEDIUM
• **Ignition Complexity**	MEDIUM	HIGH	HIGH	LOW
• **Injector Design**	MAY REQUIRE GASIFICATION OF CH4	MINIMAL REQUIREMENTS	MAY NEED INSUL. BETWEEN LO2/RP1	NO REQUIREMENTS
• **Reusability**				
- **Ablative**	SAME	SAME	SAME	SAME
- **Regenerative**	HIGH	MEDIUM	LOW	HIGH
• **Combustion Stability**	HIGH; BUT NO EXPERIENCE	LOW; MINIMAL EXPERIENCE	LOW; BUT HAVE LARGE EXPERIENCE	HIGH
STS COMPATIBILITY	LARGEST SIZE; CRYOGENS - COMPLEX GRND OPS; NO AEROSPACE EXPER.	MEDIUM SIZE; LESS COMPLEX GRND OPS; NO AEROSPACE EXPERIENCE	SMALL SIZE; LARGE EXPERIENCE SIMPLE GRND OPS & FACILITY SYSTEM	SMALLEST SIZE; VERY COMPLEX GRND OPS & TIME CONSUMING DUE TO SAFETY
PERFORMANCE, GLOW	4262 KLB	4361 KLB	4404 KLB	4604 KLB
ENGINE COST/LRB	~SAME	~SAME	~SAME	~SAME
PROGRAM RISK	HIGH - LIMITED EXPERIENCE	HIGH - LIMITED EXPERIENCE	LOW - EXTENSIVE TEST EXPERIENCE	LOW - EXTENSIVE TEST EXPERIENCE
IOC/OPERATIONAL AVAILABILITY	LATEST	LATER	LATE (48 MONTHS)	EARLY; PROPELLANT AVAILABILITY CONCERN

• NTO/MMH, similar to pump-fed case, offered small booster size, high reliability, low development risk and cost, but it was not selected due to severe safety and environmental impacts.

• Among the three hydrocarbon options, LO2/RP1 was singled out as the best propellant combination, because it gave relatively smaller booster size, higher operational flexibility, lower development risk and cost than LO2/CH4 and LO2/C3H8 options.

• Metalized or gel propellants have been considered as a side option, since they have some advantages that makes a good propellant for the pressure-fed LRB system, i.e. storable, high bulk density, minimal explosive hazard or safety concerns. However, metalized propellant is a relatively new concept which has never been tested or produced in large scale, and its rheologic properties still are not well understood. Therefore, it was decided that the technology level of the metalized propellants is not yet ready for STS application, and they should only be considered as an option for the future applications. Table 3.5.8 summarizes our assessments on metalized propellants. At present we do not see any major advantage of this propellant type for the LRB

<u>Validity of Trades.</u> After downselection of the concepts, a number of trades and analyses were carried out which resulted in vehicle size changes. As indicated in Tables 3.5.9 and Tables 3.5.10, these changes do not impact the validity of the above trade.

Final Downselection: A comparison of the three leading propellants LO2/RP1, LO2/CH4, and LO2/LH2 is shown in Table 3.5-11. All these propellant/engine concepts are basically suitable for LRB and are very close in overall merit. LO2/LH2 propellant was downselected (see section 3.13 which follows) because of no environmental concern, commonality with STS main stage (existing propellant system), and for being most suitable for alternate applications.

Table 3.5.8. Metallized Propellants Evaluation with Growth Assessments

DESCRIPTION

- COLLOIDAL MIXTURES - PULVERIZED METAL SUSPENDED IN LIQUID PROPELLANTS; FUEL AND OXIDIZER MIXTURES ARE "GELLED" WITH GELLING AGENTS
- TYPES OF METALS - POSSESS HIGH HEAT OF COMBUSTION, e.g. Be, Li, Al, etc. HIGH DENSITY IS DESIRABLE, e.g. Fe, Al; RECOMMENDED - Al
- TYPES OF LIQUID PROPELLANTS - ANY COMBINATIONS, e.g. LO2/RP1, NTO/MMH, etc.

ADVANTAGES

- SAFETY - LESS EXPLOSIVE HAZARD THAN BOTH LIQUIDS & SOLIDS IN HANDLING & STORAGE
- HIGHEST Isp DENSITY COMPARED TO ALL CONVENTIONAL LIQUIDS
- STORABLE - FLEXIBILITY IN LOADING TIME

DISADVANTAGES

- TRANSFER - HIGH VISCOSITY AS GEL, RHEOLOGY IS NOT WELL UNDERSTOOD, EVACUATED TANKS MAY BE REQUIRED TO AVOID BUBBLE ENTRAPMENT
- UNLOADING OF PROPELLANT IN CASE OF ABORT MAY NOT BE POSSIBLE
- CORING IN TANK - POSITIVE EXPULSION DEVICE (EG. PISTON) MAY BE REQUIRED
- AVAILABILITY - ONLY PRODUCED IN SMALL QUANTITIES SO FAR. FOR LARGE QUANTITIES, NEW PRODUCTION PLANTS MAY BE REQUIRED
- COST - PROPELLANT COST WOULD PROBABLY BE HIGHEST. TRANSFER WILL BE EXPENSIVE.
- ENVIRONMENTAL IMPACTS - SOLID PARTICULATES (Al_2O_3) IN EXHAUST PRODUCTS
- TECHNICAL AND SCHEDULE RISK - NEW DEVELOPMENT WHERE MANY PROBLEMS ARE IDENTIFIED AND NOT YET RESOLVED
- OPERATIONAL AVAILABILITY & COMPLEXITY - NEW FACILITY, TRANSFER, ETC

METALIZED/GEL PROPELLANTS OFFER HIGHER SAFETY & Isp DENSITY BUT ARE CONSIDERED AS GROWTH OPTION ONLY DUE TO:

- RHEOLOGY NOT WELL UNDERSTOOD AT THE PRESENT
- SIGNIFICANT GAINS WITH TOXIC METALS AND STORABLE PROPELLANTS. MODERATE GAINS WITH PREFERRED PROPELLANT COMBINATION (AL/LO2/RP1)
- MOST LIKELY IOC IS YEAR 2000.

	LO2/RP1	AL/LO2/RP1
BLOW, LBM	1,288,468	1,432,734
LRB LENGTH/DIAMETER, FT	175.0 / 14.2	170.1 / 13.8
PROPELLANT VOLUME/LRB, FT3	18,068	16,550(8.4% CHANGE)
VACUUM Isp, SEC	285.2	261.8

- GROWTH POSSIBILITY FOR PREFERRED PROPELLANT CONCEPT (AL/LO2/RP1)

Table 3.5.9 Impact of Change of Assumptions on Pump-fed Propellant Trade

TRADE ASSUMPTIONS & REQUIREMENTS	CURRENT DESIGN & IMPACT
• VEHICLE L/D = 12.3	• CONSIDERATION GIVEN TO ET/LRB ATTACHMENT - NO IMPACT
• VEHICLE SIZED WITHOUT ENGINE-OUT CAPABILITY	• VEHICLE SIZED FOR ATO WITH ONE ENGINE OUT - ALL VEHICLES EQUALLY AFFECTED
• ENGINE DATA BASED ON STME & STBE STUDIES	• LO2/LH2 CHAMBER PRESSURE LOWER DATA BASED ON ROCKETDYNE STUDY - MINOR IMPACT
• NOZZLE EXIT DIAMETER = 50 IN	• OPTIMIZED NOZZLE FOR ABOUT 12 PSIA - LO2/RP-1 VOLUME RELATIVELY REDUCED

Table 3.5-10. Impact of Change of Assumptions on Pump-fed Propellant Trade

TRADE ASSUMPTIONS & REQUIREMENTS	CURRENT DESIGN
• VEHICLE L/D = 12.3	• VEHICLE L/D = 15 FT - NO IMPACT
• GRAPHITE-EPOXY TANK MATERIALS	• ALUMINUM TANKS - VEHICLES BIGGER SAME IMPACT
• NOZZLE EXIT DIAMETER = 90 IN	• NOZZLE EXIT DIAMETER = 108 IN - SAME IMPACT
• GAS GENERATOR-HEATED HELIUM PRESSURIZATION	• HE/H2/O2 WITH CATALYST - SAME IMPACT
• VEHICLE SIZED WITHOUT ENGINE-OUT CAPABILITY	• VEHICLE SIZED FOR ATO WITH ONE ENGINE OUT - BIGGER VEHICLES; SAME IMPACT
• CHAMBER PRESSURE = 450 PSIA	• CHAMBER PRESSURE = 334 PSIA - SAME IMPACT

Table 3.5-11. Comparison of Leading LRB Propellant Candidates

	LO2/RP-1	LO2/CH4	LO2/LH2
PERSONNEL SAFETY	BEST - LEAST EXPLOSIVE	GOOD - 20% TNT EQUIV.	MEDIUM - 60% TNT EQUIV. VERY FLAMMABLE
ENVIRONMENTAL IMPACT	SOME - IGNITABLE WASTE, CO2 & CO IN EXHAUST	SOME - CLEAN BURNING BUT, CO2 & CO IN EXHAUST	BEST - BENIGN
RELIABILITY	BEST-SIMPLE, PROVEN	LOWEST - 2 CRYOGENS, NEW TECHNOLOGY	MEDIUM - COMPLEX, PROVEN (2 CRYOS) BUT SIMILAR STEP THROTTLING
STS COMPATABILITY	GOOD - CLOSEST TO SRB SIZE	GOOD - NEXT SMALLEST	GOOD - LARGE SIZE OFFSET BY LOW THRUST/WEIGHT
KSC FACILITY IMPACT	MAJOR - DUAL FUEL, REBUILD SATURN SYSTEM	MAJOR - ALL NEW FUEL SYSTEM	SOME - SIZE IMPACTS TOWER, ADD TO EXISTING PROPELLANT SYSTEM
GROWTH POTENTIAL & OTHER APPLICATIONS	FAIR - STANDALONE OR ALS BOOSTER, BUT LIMITED REUSABILITY	GOOD - STANDALONE OR ALS BOOSTER, BETTER REUSABILITY	BEST - COMMON ENGINE WITH ALS, SHUTTTLE-C, STANDALONE BOOSTER AND CORE
RISK	LOW - PROVEN, BUT COMBUSTION INSTABILITY AND CONCERNS	MEDIUM - NO FLIGHT EXPERIENCE	LOW - PROVEN
LARGE THROTTLING VEHICLE COSTS	LOWEST FOR STS LRB	HIGH-RISK	CLOSE TO LO2/RP-1, LOWER WITH COMMONALITY

3.6 VEHICLE CONFIGURATION

The choice of LRB geometry is a complex problem involving aerodynamics, control capability, and structural loads on the whole STS stack as well as LRB itself. Due to increased payload requirement and propellant density differences, all LRB concepts are larger than the current SRB.

In the fall of 1987, NASA/MSFC initiated wind tunnel tests on STS configurations with LRBs because of serious concerns about Orbiter wing loads. Cylindrical test shapes simulated LRBs up to 21 feet in diameter and 190 feet long. Multi-diameter (such as hammerhead) and non-symmetrical (clocked) arrangements were also tested.

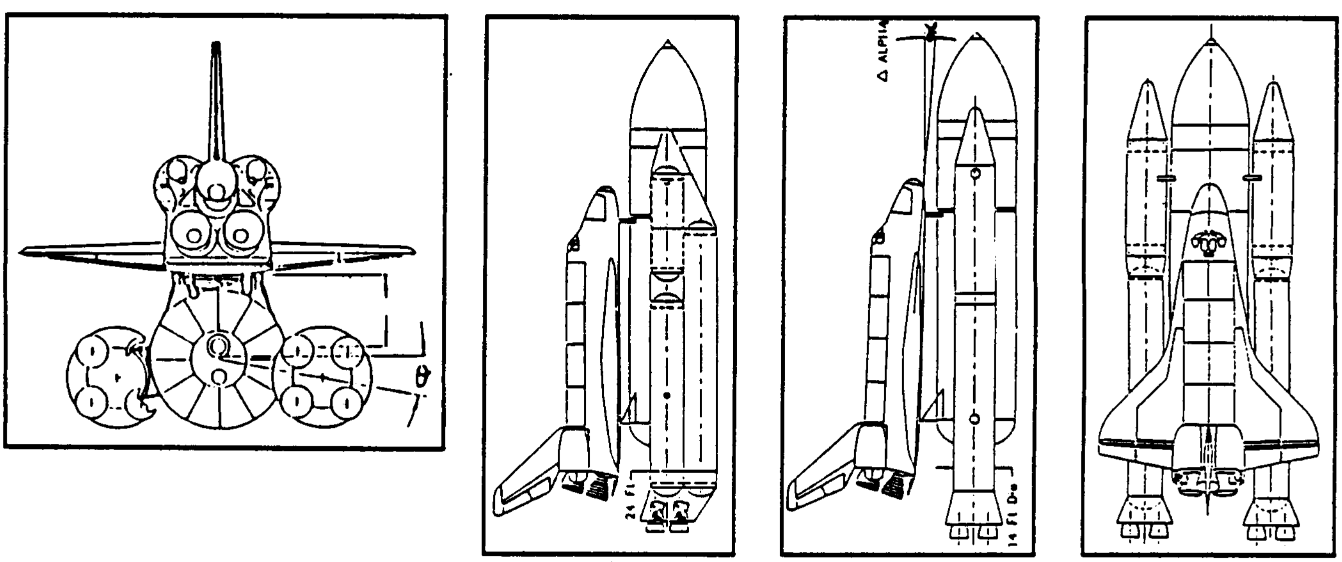
Figure 3.6-1 illustrates four alternate configurations. The first one involves increasing the length and diameter beyond that of the current SRB, and rotating the booster around the ET. This presents minimum impact to the Orbiter and Launch schedule. Small rotation angles would be required, for example, one to three degrees for pump-fed hydrocarbon concepts. However, the ET structure or struts and launch site facilities would need to be modified to accomodate the new booster positions. Wind tunnel results showed rotation to be of little benefit, so it was discarded.

The second configuration, tandem or parallel tanks, allows greater propellant volume than a conventional stacked tank arrangement. The stiffer side-by-side tanks may help to alleviate the "twang" problem at ignition. Considering evolutionary growth paths, this concept would be a poor choice for a stand-alone booster. If cryogenic propellants are used, the problem of heat transfer between parallel tanks would need to be addressed.

The third concept involves modifying the Orbiter standoff mounts to decrease the Orbiter wing loads via two effects–separation distance and angle of attack. This option would affect the entire trajectory, from lift-off until ET separation, and would necessitate changes to the Orbiter propellant feedlines, and MLP masts.

Finally the hammerhead configuration strives to increase available propellant volume by increasing the diameter of the booster ahead of the Orbiter wing. This shape would probably still be inadequate for the lower-density propellants such as hydrogen, and the contour change may require more difficult LRB design. Wind tunnel tests of this configuration showed reduced drag for the whole stack.

CONCEPT	ADVANTAGES	DISADVANTAGES
ROTATED	• Miminum Impact To Orbiter • Small Rotation Angles Required	• ET Structural Mods • VAB Work Platform Mods • Disturbs Lateral Aerodynamics
TANDEM TANKS	• Large Performance Margins • Increases Bending Stiffness • Reduces Length	• Complex To Design & Manufacture • Heat Transfer Between Parallel Tanks • Poor Stand-alone Booster • New Work Platforms Required
ORBITER MOUNTS	• Decreases Wing Loads Via Two Effects (Separation Distance, Angle Of Attack)	• Affects Entire Trajectory • Impacts Launch Schedule • Orbiter Propellant Feedlines • Changes MLP Masts
HAMMERHEAD	• Concept Flight-proven • Wind Tunnel Tests Showed Reduced Wing Loads	• Probably Inadequate For LH2 • Contour Change May Not Match Intertank Structure



ROTATED | TANDEM TANKS | ORBITER MOUNTS | HAMMERHEAD

Figure 3.6-1 Alternate Configurations

Based on the results of these wind tunnel tests and our analysis, we recommend flying q-alpha profiles which produce acceptable wing loading rather than utilizing unconventional geometries on configuration layouts.

3.6.1 LENGTH AND DIAMETER. KSC facility derived limits on diameter (19 ft maximum) and length (200 ft maximum) have been identified. There is a KSC preference for LRB lengths less than 170 ft to avoid interference with the ET GOX Vent Arm. Aerodynamic and aerothermal effects have been examined, and LRB lengths between 175-185 ft should be avoided to reduce drag and heating. These constraints are illustrated in Figure 3.6-2. To allow for more straightforward design and to reduce system complexity, the forward attachment for pump-fed LRBs should be located either in an intertank or on the forward adapter.

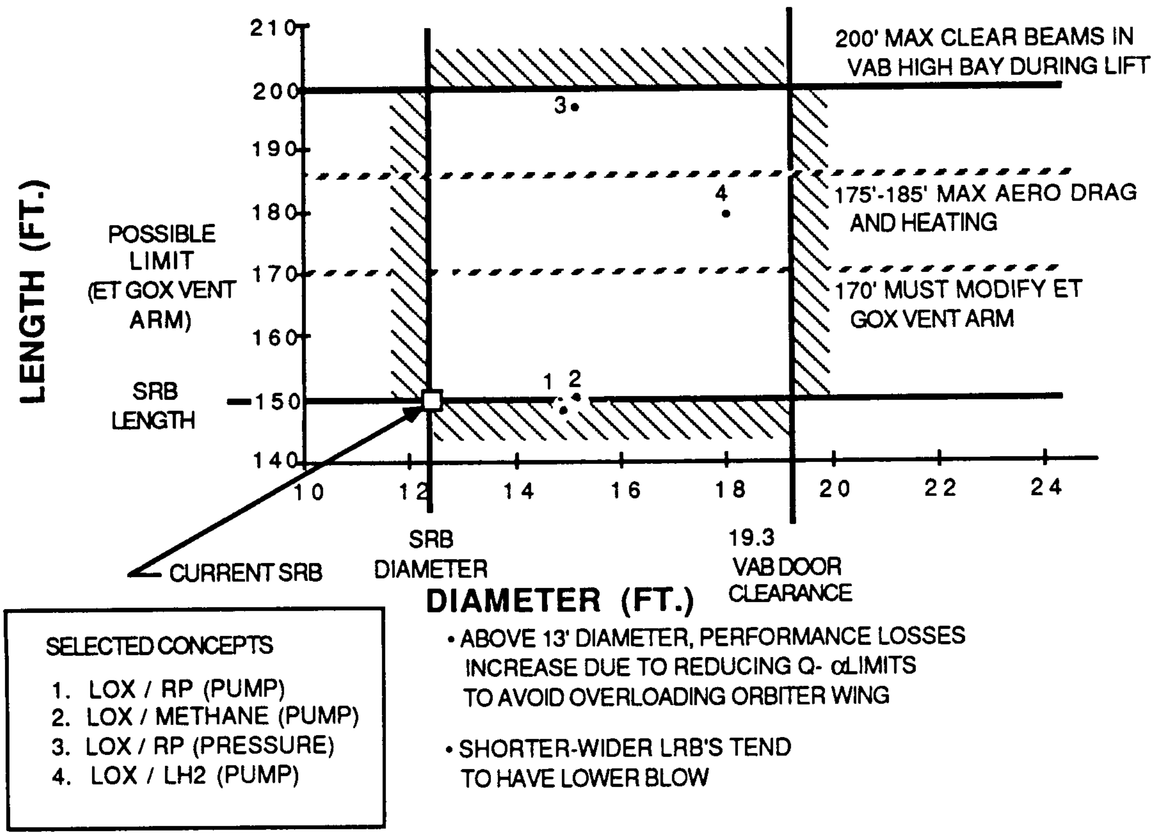


Figure 3.6-2 LRB length and diameter constraints

The VAB doors are 871.5 inches wide. When the ET diameter (331in) is subtracted from this distance and provisions are made for dynamic clearances, the maximum LRB diameter possible is 19 ft.

The only constraining limit on LRB length in the VAB is due to a clearance above the support beams which separate highbays 2 and 4 from their associated transfer aisles. A gully assembled LRB will have to be lifted through one of these openings prior to stacking on an MLP. The height (considering crane apparatus requirements) is about 257 ft – we recommend limiting the LRB to 200 ft. Taller LRBs would force modification to internal VAB structural beams. Because these beams are major structural members, their removal or modification is unlikely. In addition, inside the VAB numerous changes will have to be made to service platforms for all our LRB designs. The number and severity of modifications required increases as the length and diameter increase, but it is not felt that work platform impacts should be considered to constrain LRB size.

At the launch pads (39A and B) the ET GOX vent arm is located at elevation 265, and the maximum distance an LRB could attain and fit under the arm is approximately 170 ft. LRBs taller than this are acceptable, but necessitate rerouting the ET GOX arm, which is a slight cost increase.

Aerodynamic considerations are discussed in Section 3.6.2 and indicate increased drag for lengths between 175 and 185 ft. Concepts can be arranged to avoid these lengths. Structural analysis of E7 interface loads are discussed in Section 8. Our estimates indicate that configurations up to 200 ft. long and 16 ft. in diameter do not exceed allowable loads. It is desirable to arrange pump-fed LRB's such that the forward attach fittings are located in intertank or forward adapter structures rather than on the propellant tanks. From a gross weight standpoint, smaller L/D's are slightly better although the difference is on the order of 1%. In summary, it is concluded that L/D's between 10 and 14 are generally acceptable.

3.6.2 AERODYNAMICS ANALYSIS. The aerodynamic concerns associated with the incorporation of liquid rocket boosters are focused on the physical characteristics of the larger boosters. Specifically, the Orbiter wing and elevon loads, and the interface loads between each element of the mated vehicle are highly dependent upon the size of the booster. Additionally, development of longitudinal (C_A, C_N, and C_m) and lateral-directional (C_Y, C_n, and C_l) aerodynamic coefficients as a function of the various physical and flight parameters (Length, Diameter, Mach number, angle of attack, sideslip angle, etc.) is required to support vehicle sizing, loads and controls studies.

To support initial trade studies with a rapidly evolving configuration, the development of an aerodynamic prediction computer routine was undertaken. This resulting code rapidly predicts wing loads, elevon loads, and element and mated vehicle aerodynamics for a wide range of booster lengths and diameters. A high degree of flexibility has been incorporated into the prediction routine, allowing for future growth in the support of more advanced trade studies.

The aerodynamic prediction capability for the liquid rocket booster configurations has been sufficient to support the analyses conducted to date. The longitudinal force and moment coefficients have been provided to the trajectory/sizing study, the longitudinal and lateral-directional force and moment coefficients have been provided to the aborts and controls studies, and wing and elevon loads have been analyzed sufficiently such that an understanding of the flight constraints for each booster diameter and length has been achieved. Alternate configurations (see section 3.6, Figure 3.6-1) have been analyzed, with the hammerhead configurations proving to be the most promising at reducing wing loads, and the gap and aft skirt designs have proved to be minor influences on both the wing and mated vehicle loads.

<u>Modeling Development/Capabilities</u>. The aerodynamic predictions for a configuration employing LRB's are largely based upon the series of wind tunnel tests performed by Charlie Dill of MSFC. The first test (9/87) investigated wing loads, elevon loads, and the mated vehicle longitudinal aerodynamic characteristics as a function of Mach number and angle of attack. Booster diameters between 12.2 and 21 feet and lengths between 144 and 190 feet were tested. The second test (2/88) investigated the wing and elevon loads, and the longitudinal mated vehicle characteristics for a series of "alternate" booster configurations. These configurations included hammerhead boosters, boosters rolled relative to the external tank (away from the Orbiter), and twin tank boosters. The third test (3/88) investigated gap (between booster and external tank) and booster aft skirt designs. All wind tunnel data was presented as deltas from the current STS configuration, which is described in the IVBC-3 document.

Rapid production of the aerodynamic characteristics for LRB configurations with variable length and diameter boosters commanded the development of a computer tool directly accessible by the requesting design groups. The resulting computer code SAPT (Shuttle Aerodynamic Prediction Tool) generates the current STS 6-DOF aerodynamic characteristics along with the wing and elevon loads. In addition to Mach, angle of attack, and sideslip angle, the effects of off-nominal elevon deflections, altitude, and dynamic pressure are accounted for. LRB configuration aerodynamic characteristics are generated by adding the LRB deltas from the wind tunnel data to the current STS configuration data. Additional provisions have been made to develop the element aerodynamics for the LRB configurations, where no wind tunnel data exists. Each aerodynamic coefficient and the variables with which it is a function of, are presented in Table 3.6.2.1-1.

Table 3.6.2-1. Summary of aerodynamic prediction capabilities

MATED VEHICLE

Longitudinal (C_A, C_N, and C_m) = f(Mach number, angle of attack, sideslip angle, booster length†, booster diameter†, altitude, elevon deflections, dynamic pressure)

Lateral-dir. (C_Y, C_n, and C_l) = f(Mach number, angle of attack, sideslip, altitude, elevon deflections, dynamic pressure)

ELEMENT LOADS

Longitudinal (C_A, C_N, and C_m)= f(Mach number, angle of attack, sideslip angle, booster length‡, booster diameter‡)

Lateral-dir. (C_Y, C_n, and C_l) = f(Mach number, angle of attack, sideslip angle, booster length*, booster diameter*)

† length and diameter variations were wind tunnel tested, but only as a function of Mach number and angle of attack. No sideslip, elevon, altitude, etc. were tested. These variations (i.e. $\partial CA/\partial$altitude) for variable length and diameter boosters are assumed equal to the current STS value.

* Empirical relations only. See Section 3.6.2.1 Element Aerodynamics

‡ Total mated vehicle deltas due to increased length and diameters are known, but the distribution of these deltas among the various elements is estimated by empirical relations. See Section 3.6.2.1 Element Aerodynamics

The SAPT computer code resides as a subroutine in each engineering group's analysis routine, thereby automating the generation of the aerodynamic characteristics for variable length and diameter boosters. This one prediction tool provides each design group with the necessary data. This relationship is illustrated in Figure 3.6.2-2. Note that the flow of information (down) is structured such that the design groups in the inner design loop are not dependent upon aerodynamics to generate the data for each new booster configuration. Aerodynamics has been removed from a position directly inside the design loop, thus accelerating data transmittal.

One comment regarding the wind tunnel test models used to represent the LRB configurations should be noted. The external surface of the SRB's contain more protuberances than will the candidate LRB configurations. The external structure (attachment) rings of the SRB's will be replaced by a smooth surface, and external protuberances such as the aft integrated electronics assembly may be relocated. These combine to reduce the "effective" diameter of the boosters up to 10 inches. Thus, the current diameter limit is 18ft Dia. without perturbences. This topic is mentioned here, because the models as tested in the MSFC wind tunnel tests retained these protuberances - and the removal of these protuberances reduces wing loads proportional to the decrease in the effective diameter.

Element Aerodynamics. The calculation of interface loads requires the generation of the complete six degree of freedom aerodynamic force and moment coefficients for each "element" of the mated vehicle: orbiter, external tank, left and right booster. Interface loads could prove to be a configuration discriminator, and, while the estimation of the element aerodynamics is not exact, the predictions of the element aerodynamics of each configuration is handled in a consistent manner for the development and parametric evaluation of each configuration. This technique allowed a fair aerodynamic assessment of one configuration relative to another. Note that because length and diameter where the only geometric variables of interest (at least at this point), it was desired to create a prediction routine using only these two variables in conjunction with the mated vehicle and element aerodynamic database.

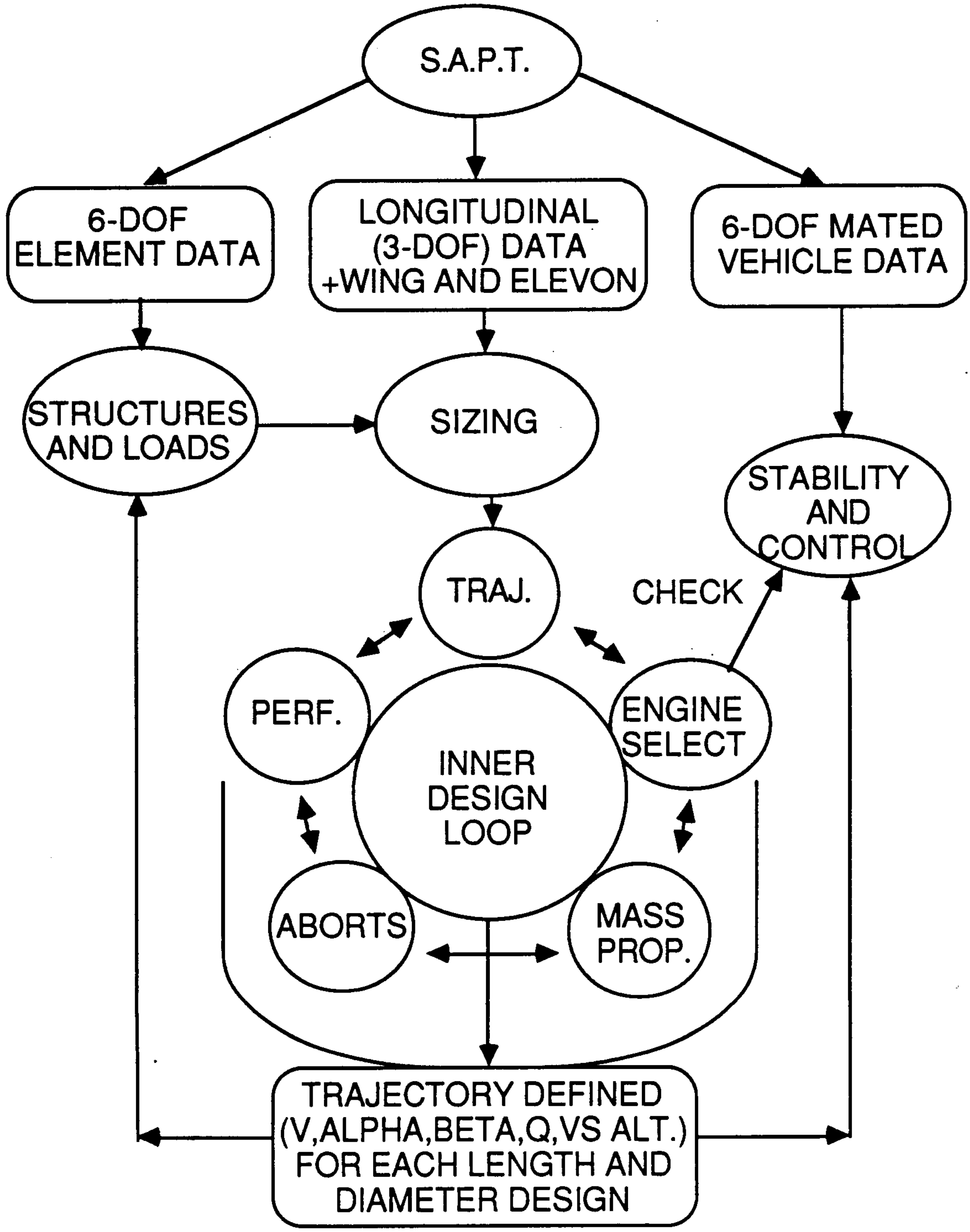


Figure 3.6.2-2. Aerodynamics - trade study relationship

Axial Force Coefficient. The axial force coefficient, C_A, for variable length and diameter mated vehicle configurations was provided from wind tunnel tests. Therefore, the delta mated vehicle axial force coefficient (LRB - SRB baseline) is known - only the distribution of this additional force between the elements is unknown. The method chosen to distribute the axial force is a function of the geometrical area variation ratio (in the longitudinal - Y-Z plane). Thus, the Orbiter (no area change) received no addition, the boosters were allotted 45% (each) of the delta, and the external tank received the remaining 10%. While the area of the external tank did not vary (obviously), the increased length and diameter of the boosters displace an additional portion of the pressure load onto the forebody of the external tank, consequently increasing its axial force.

Normal Force Coefficient. The normal force coefficient, C_N, for larger length and diameter mated vehicle configurations was also provided from wind tunnel tests. With the normal force however, the portion of the additional normal force carried by the Orbiter is presumed to lie in the increase in the wing shear force coefficient (also provided from wind tunnel data). The remaining portion of the delta normal force coefficient is distributed via a 40/40/20 split between the boosters and the external tank, respectively. This ratio was also determined by considering the revised geometry and the corresponding distribution of the pressure load (this time in the lateral directional - XY - plane).

Side Force Coefficient. Unlike the axial and normal force, the side force coefficient, C_Y, for the larger length and diameter configurations was not measured in the wind tunnel tests (to date). Thus, the variation from the baseline as a function of length and diameter was estimated. It was decided that, in order to facilitate estimation of the yawing moment coefficient, C_n, of each booster, the sideforce coefficient of each booster would be divided into a component fore and aft of the moment reference point ($x_{MRC} = 976''$).

Bifurcation of the sideforce coefficient aids estimation of the boosters sideforce and yawing moment coefficients two ways. First, some portion of the increased sideforce coefficient will be due to the length variation, and some part due to the diameter variation. Because the length of the boosters vary fore of the moment reference center, but not aft, the length variation effect is distributed to the forebody term alone. Each term varies with booster diameter. Second, the variation in the location of the center of pressure of the forebody component can be assumed to vary proportional to the length, while the aft component variation can be assumed negligible. Only the aft skirt geometry will play a significant role in the aft component center of pressure variation, but this geometry had not been finalized at the time of this analysis. The bifurcation of the side force coefficient will also aid the side force and yawing moment calculations for the external tank.

The current SRB sideforce and yawing moment coefficients are known from the IVBC-3 database - only the fore and aft distribution is in question. To estimate this distribution, the center of pressure of the fore and aft components must be assumed. Two equations and two unknowns

($C_{Y_{Fore}}$, and $C_{Y_{AFT}}$) remain:

$$C_{Y\,TOTAL,\,SRB} = C_{Y\,Fore} + C_{Y\,AFT}$$

$$C_{n\,TOTAL,\,SRB} = C_{Y_{Fore}}*(DX1/LREF)+C_{Y_{AFT}}*(DX2/LREF)+C_{A_{SRB}}*(DY/LREF)$$

Where: DX1 = Xmrc - Xcp_{Fore}

Xmrc = 81.33 (Feet)

Xcp_{Fore} = 77.5 (Feet) - *assumed*

DX2 = Xmrc - Xcp_{Aft}

Xcp_{Aft} = 175 (Feet) - *assumed*

LREF = 107.525 (Feet)

DY = Ycp - Ymrc (Ycp = centerline of booster)

Ycp = Radius of ET + 1" (GAP)+ Radius of Booster

= 13.79 + 1 + Dia/2 (Feet)

The location of the center of pressure for the fore and aft components was chosen based on slender body theory for a cylinder at small angles of attack.

These two equations can be solved simultaneously to yield the fore and aft sideforce components of the solid boosters. These components are then altered based on the cross-sectional area variation of the larger boosters:

$C_{Y\ Fore,\ LRB}$ = $C_{Y\ Fore,\ SRB}$ * [(DIA*DIA/SIN(18°)/2.0+ DIA*DL]/ASRB

$C_{Y\ Aft,\ LRB}$ = $C_{Y\ Aft,\ SRB}$ * [1.0 + (DIA-12.2)/12.2]

Where: DIA = LRB Diameter
DL = LRB Length - 144.0
ASRB = cross-sectional area of the forebody of the SRB (288.61 ft^2)

The total sideforce coefficient for the larger boosters is then the sum of the revised fore and aft components.

The variation in the sideforce coefficient of the external tank is assumed to be the negative of the forebody variation in the booster sideforce term. No variation to the Orbiter sideforce is predicted. As wind tunnel data becomes available, the step generating the LRB booster coefficients from the SRB data will be eliminated, and the LRB booster fore and aft sideforce coefficients will be estimated similar to the current pitching moment and normal force coefficients technique. This would remove the limitation associated with the current technique of estimating the total booster sideforce variation as a function of booster length and diameter, and will introduce the dependence on sideslip angle more directly.

Pitching Moment Coefficient. The increment in the pitching moment coefficient, C_m, for the mated vehicle and the Orbiter wing torsion moment are known from wind tunnel data. The variation in the Orbiter's pitching moment is assumed equal to the variation in the total (left+right) wing torsion plus the appropriate reference center shift of the wing shear.

$DC_{m\ Orbiter,LRB}$ = 2*(DCSR*(XMRC-XMRCWNG)/LREF+DCTR*LREFWT/LREF)

Where: DCSR = change in wing root shear force coefficient due to the presence of the larger boosters.
DCTR = change in wing root torsion moment coefficient due to the

presence of the larger boosters.

XMRCWNG = moment reference center of the wing (108.92 Feet)

LREFWT = reference length for the wing torsion moment coefficient (40.39 Feet)

The remaining portion of the delta pitching moment coefficient is used, along with the known SRB configuration booster normal force and pitching moment, to predict the pitching moment variation for the liquid boosters and the external tank.

The equations for the estimate of the pitching moment coefficient for the LRB configuration are formulated below.

$$Xcp_{\text{booster, LRB}} = (-C_{m\,\text{booster,SRB}}/C_{N\,\text{booster,SRB}}) * LREF + DX)*(1.0 + DL/L)$$

Where: DX = 20.22 (Feet) - Length from the nose of the booster to the moment reference center

L = Length of the LRB

This equation calculates the center of pressure location of the booster from the nose of the booster. This value is then multiplied by the percentage increase in the length of the booster, thus holding the ratio of Xcp/Length constant. The effect of the variation in the axial force is included. The change in the booster pitching moment coefficient is:

$$DC_{m\,\text{booster,LRB}} = - DC_{N\,\text{booster,LRB}}*(Xcp-DX)/LREF$$

where: ZMRC = 33.33 (feet)

DCAL = change axial force coefficient of corresponding booster

The change in the pitching moment of the external tank is estimated by:

$$DC_{m\,\text{ET,LRB}} = C_{m\,\text{LRB}} - C_{m\,\text{SRB}} - DC_{m\,\text{right booster,LRB}} - DC_{m\,\text{left booster,LRB}} - DC_{m\,\text{Orbiter,LRB}}$$

This formulation assures that the sum of the element components will equal the total pitching moment for an LRB configuration, as measured in the wind tunnel test.

Yawing Moment Coefficient. The estimation of the variation in the yawing moment coefficient for the boosters was eluded to previously. The resulting relation is written as:

$$DC_{n\ booster,\ LRB} = DC_{Y\ Fore,\ LRB} *(DX1/LREF)C_{Y AFT,LRB}*(DX2/LREF) + C_{A\ booster\ LRB}*(DY/LREF)-C_{n\ booster,\ SRB}$$

The relation for the external tank is:

$$DC_{n\ ET,\ SRB} = (C_{Y\ FORE,\ SRB} - C_{Y\ FORE,\ LRB})*(DX1/LREF)*SIN(BETA)$$

Note that the yawing moment of the external bank is decreased by the increase in one booster's yawing moment. The variation in the yawing moment of the Orbiter is presumed negligible.

Rolling Moment Coefficient. The rolling moment coefficient, C_l, of the boosters about the moment reference center is dominated by the translation of the normal force to the moment reference center. Thus, $C_{l\ booster,\ LRB} = C_{N\ booster,LRB}*DY/LREF$

The rolling moments of the Orbiter and External Tank remain unaltered.

3.7 ENGINE SYSTEM

Early in the study, the LRB engine candidates included both existing and new engines with a wide range of propellant combinations. We considered pump-fed and pressure-fed, expendable and reusable concepts for the LRB vehicle. STME and STBE study results constituted a large database for pump-fed engine trade studies. Our contractors, TRW and Rocketdyne, provided engine parametric data and assisted in engine trades and analyses to define the large pressure-fed propulsion system. We also subcontracted Pratt & Whitney and Rocketdyne in the study of new pump-fed engine concepts tailored for LRB application.

From a wide range of propellant combinations, existing and new engines, pressure-fed and pump-fed vehicle concepts, we have downselected to a LO2/LH2 pump-fed vehicle with new gas-generator engines as baseline and split expander cycle engine as an alternative. Expendable mode was selected over reusable.

TABLE 3.7-1 SELECTED LRB ENGINE CHARACTERISTICS

Engine	LO2/RP1 Pump	LO2/LH2 Pump	LO2/LH2 Pump	LO2/CH4 Pump	LO2/RP1 Pres
Type	Gas-Generator	Gas Generator	Split-Expander	Split-Expander	Pressure-fed
Status		Baseline	Alternate		
Chamber Press	1275	2250	968	758	334
Vac Isp, sec	310.5	410.5	409.5	337.5	285
Area Ratio	16	20	10.6	16.5*	4.96
Mixture ratio	2.53	6	6	3.5	2.5
Throttling	Continuous	Step	Step	Continuous	Continuous
	+10%, -25%	-25%	-25%	-35%	-40%
Cooling	Regen Fuel	Same	Same	Same	Same
Gimbal	Head-end	Same	Same	Same	Same
Bleed	No Bleed	Same	Same	Same	Same
Vac Thrust(k lb)	630	558	564	756	972
Weight (lb)	6216	5480	3560	5640	7017

* Not cost optimized

Recent work has focused on simple, reliable and hopefully low cost engines for expendable but man-rated LRBs for the STS. The engine parameters such as engine thrust, area ratio, chamber pressure, and inlet pressures were optimized, and selections were based on overall vehicle system trades and are described within this Section. The engine design and analysis, like combustion stability in pressure fed and injector selection are covered in details in the final reports of the engine subcontractors, Rocketdyne, Pratt and Whitney, and TRW. These reports are included as appendices to our final report. Previously selected engine types, as from June 1988 Final Report, and characteristics are shown in Table 3.7-1.

Among the selected engine features, four areas have been particularly controversial:

a) Throttling: Continuous throttling sharply increases engine complexity and cost. For the LO2/LH2 booster vehicle, it is possible to have a single step throttling with 100% and 75% thrust levels and still satisfy all the vehicle constraints. Open-loop engine control is bvaselined for this vehicle because of simple throttling control requirements. However, open-loop control causes the vehicle to grow slightly because of higher mixture ratio and thrust dispertion.

For other concepts, multistep throttling is needed, and hence there is no major savings in cost or complexity over the continous throttling. Therefore, continous throttling with closed loop control is baselined. For the LO2/RP-1 pump-fed LRB case, a combination of throttle up and down seems preferable based on the trands predicted by Rocketdyne. For pressure-fed LRB, only throttling down is considered because of direct correlation of chanber pressure wiht tank pressure, and hence impact on tanks weight. ATO capability with one engine out from lift off increases throttling to about 40%.

b) Engine Bleeds: The engines baselined have no engine bleeds in order to simplify the ground operations. However, the turbopump assembly is often a major heat source that will cause coontinous bubble formation in cryogenic propellants while the vehicle is on the pad. Although preliminary analysis by STME/STBE contractors indicates that no problem is anticipated during engine start-up, more intensive vehicle and engine analysis is needed in this area.

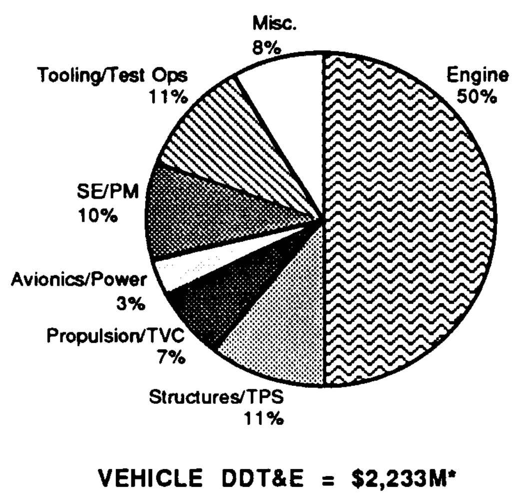
c) Cooling for pressure-fed engines: Regenerative cooling vs ablative coatings for expendable pressure fed engines is not a clear cut choice. We chose regenerative cooling because there is no experience with ablative LOX/RP-1 chambers. Test data from the MSFC pressure-fed test bed would help.

d) Costs: Average unit costs are a judgement call by General Dynamics defined by modifying inputs from several engine contractors. This is a key to the reusability question, to the viability of pressure-fed concepts, and to the eventual success of any LRB program.

For details of the main engine trades and analyses, see the appendices containing final reports from the subcontracts to Rocketdyne, Pratt and Whitney, and TRW.

3.7.1 VEHICLE OPTIMIZATION PARAMETER. The optimization parameter selected is of prime importance to the size and cost of the vehicle and its subelements. In the early part of the study, it was assumed that the size of the vehicle is directly related to the impact on the STS system, and to optimize the system, one should minimize the booster size. Hence the volume of the propellants or GLOW was considered as the optimization parameter. However from later analysis of the impact of length and diameter of the booster on the STS system, we have come to the conclusion that there is a large flexibility in selecting the diameter and length of the booster, even with the current constraints of the STS system. This analysis is based on the subscale wind tunnel tests performed at MSFC, interpretation of that data by JSC, and study of impacts by KSC on the facilities. We do not foresee any major constraint for booster diameter as large as 18 ft and length as high as 200 ft.

With these liberal constraints on length and diameter, life cycle cost of the vehicle becomes the most important optimization parameter. Because costs are seldom integrated into a vehicle synthesis program, it is a common assumption in vehicle optimization to assume that minimizing of inert weight will result in minimum cost. Our trade on the optimization parameter shows that this is too simplistic an assumption, and is not always true. Considering two extreme cases of engine costs, LO2/LH2 gas generator pump-fed and LO2/RP1 pressure-fed (Figures 3.7.1-1 and 3.7.1-2), it is shown here that for an expendable LRB with engine out capability (and with a



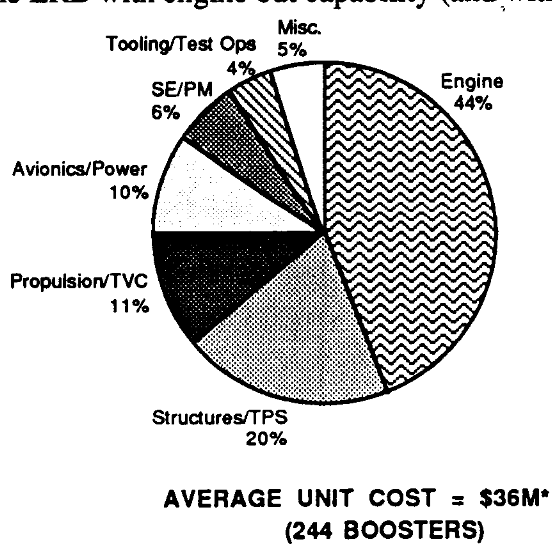
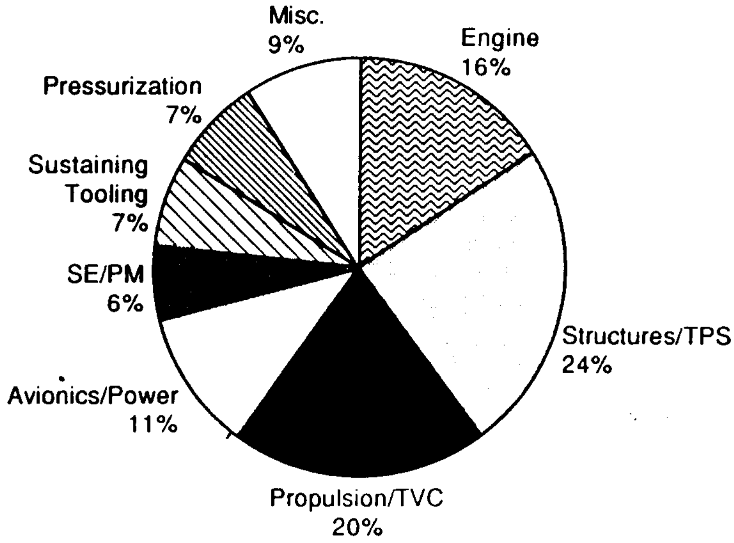


Figure 3.7.1-1. LO2/LH2 Pump-Fed LRB Cost Breakdown



AVERAGE UNIT COST = $33.8M*

Figure 3.7.1-2. LO2/RP1 Pressure-Fed LRB Cost Breakdown

minimum thrust/weight of 1.2 at lift-off with engine out) minimizing of engine thrust results in minimum life cycle cost. The difference in life cycle cost between a minimum inert weight optimized vehicle, and a minimum engine thrust vehicle is about $0.3B for the LO2/LH2 vehicle and insignificant for LO2/RP1 pressure fed vehicle. Hence we recommend, and have used, minimum engine thrust as the optimizing parameter in the vehicle synthesis runs and in our propulsion system sub-trades.

3.7.1.1 LO2/LH2 VEHICLE TRADE.

Assumptions

1. Minimum T/W with one engine out ≥ 1.2 for tower clearance etc.
2. ET vent arm needs to be modified for LRB lengths >170 ft. However the cost of modifications is insignificant.
3. LRB diameter is 18 ft (see length and diameter trade).
4. Engine chamber pressure at EPL is 2538 psia (Note that this is based on maximum performance. Trade made later shows that chamber pressure of 2250 psia results in minimum cost).
5. Engine nozzle area ratio is 40.1 (Note this is based on performance. Trade made later shows that expansion ratio of 20 results in minimum cost).
6. Life cycle cost excludes contractor fees, government support and contingency.

Results. Figure 3.7.1-3 depicts the variation of some of the important parameters with change in lift-off thrust to weight with one engine out. As shown, GLOW and length of the vehicle decrease with increase of thrust to weight, basically due to decrease in gravity losses. However at much higher thrust to weight (at about 1.4 - not shown here), A minimum in GLOW is reached at about 1.4 (not shown) because of large throttling required in most part of the flight because of q-alpha and 3-g constraints resulting in an oversize engine, and thrust structure.

LRB dry weight is a very weak function of thrust to weight, and reaches a minima at a thrust to weight of about 1.26 for this case. The inert weight increases with a decrease of thrust to weight because of increases in vehicle size and hence in tank weight, and increases with increase of thrust to weight because of increase in engine thrust and thrust structure.

The engine thrust monotonically decreases with decrease of thrust to weight of the vehicle, in the range shown (engine thrust reaches a minima around thrust to weight of 1.1 - not shown). The life cycle cost follows the engine thrust trend. This is because, as shown in Figure 3.7.1-1, engine cost is a major portion of the total booster cost (about 44%). As shown in Figure 3.7.1-3, the booster optimized to give minimum engine thrust is cheaper by $0.3 B over a booster sized for minimum inert weight, and by about $1.0 B over a booster having smallest dimensions (not shown).

3.7.1.2 LO2/RP1 PRESSURE FED BOOSTER. Two runs were made for a typical chamber pressure of 500 psia. Exit diameter of the nozzle was assumed to be 108 inches. In the first case, the vehicle was optimized for minimum inert weight, and in the second case , the vehicle was optimized for minimum engine thrust. As shown in Table 2.2.1, the difference in life cycle cost is insignificant. This is because, as shown in Figure 3.7.2-2, engine cost forms comparatively a smaller part (about 16%) of the total booster cost for a pressure-fed vehicle. Hence choice of either minimum inert weight or minimum thrust will result approximately minimum life cycle cost of the vehicle. For consistency, minimum engine thrust is recommended as the optimization parameter.

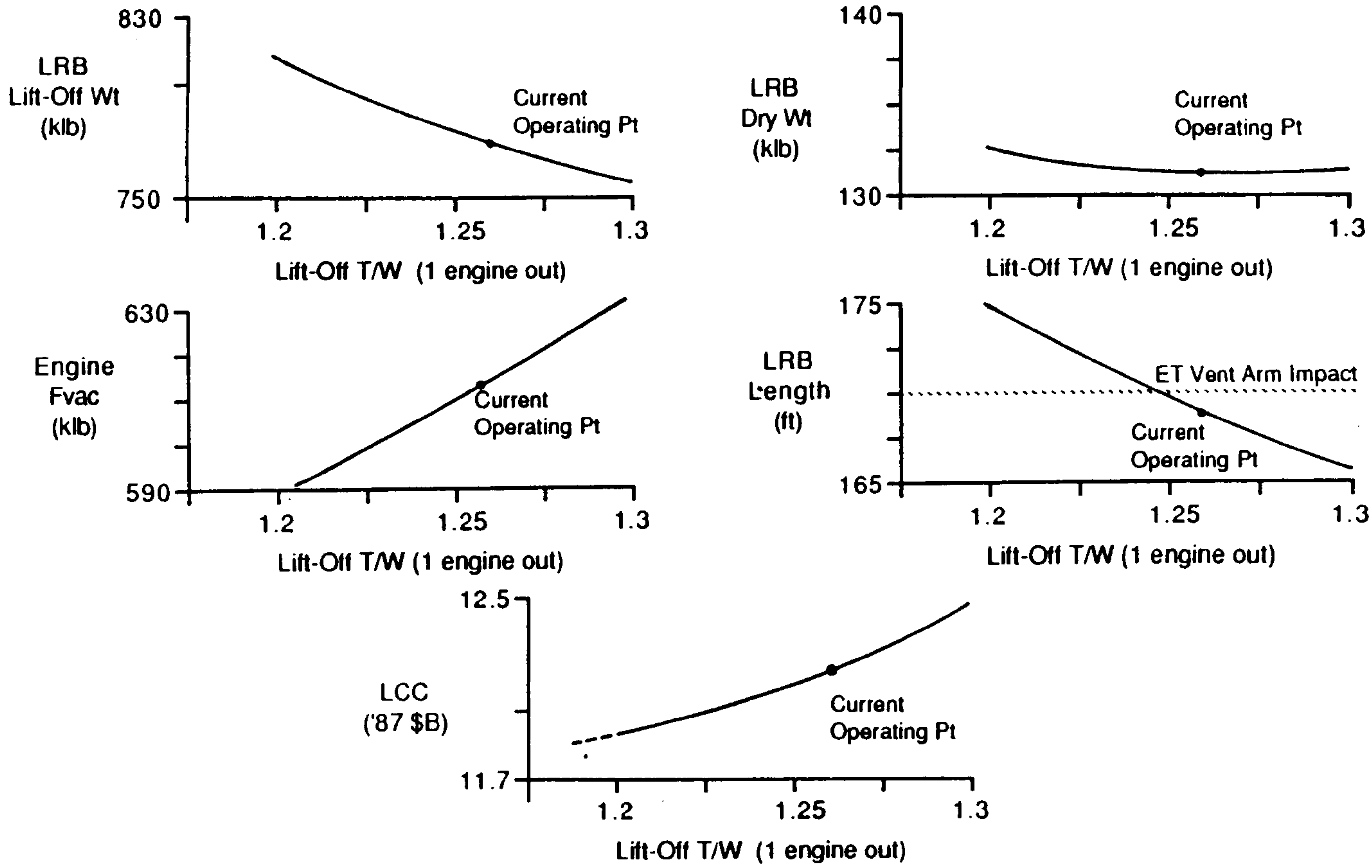


Figure 3.7.1-3 Impact of Thrust/Weight on LO2/LH2 Vehicle Parameters

Conclusion. Minimum engine thrust results in minimum vehicle cost, and hence should be used as an optimization parameter in the vehicle synthesis program runs, and propulsion system sub-trades.

Table 3.7.1-1 LO2/RP1 Pressure-Fed LRB Optimization Parameter Impact

Parameter	Minimum Inert Weight	Minimum Engine Thrust
T/W	1.22	1.20
Inert Weight	258 K lb	259 K lb
Engine Thrust	959 K lb	950 K lb
LCC	$11.8 B	$11.8 B

3.7.2 CHAMBER PRESSURE SELECTION. Here the rationale for chamber pressure selection for the LO2/LH2 (GG) pump-fed vehicle, LO2/RP1 (GG) pump-fed vehicle, and LO2/RP1 pressure-fed vehicle are discussed.

3.7.2.1 LO2/LH2 VEHICLE. There is extensive experience with LO2/LH2 over a wide range of chamber pressures. J-2 was operated at pressure of 750 psia, SSME is operated at 3000 psia, and RL-10 at about 450 psia. Hence there is not much of a technology issue in selection of chamber pressure. STME Engine contractors have recently taken an in-depth look at the impact of chamber pressure on engine cost and complexity. A typical qualitative curve of their current understanding is shown in Figure 3.7.2-1.

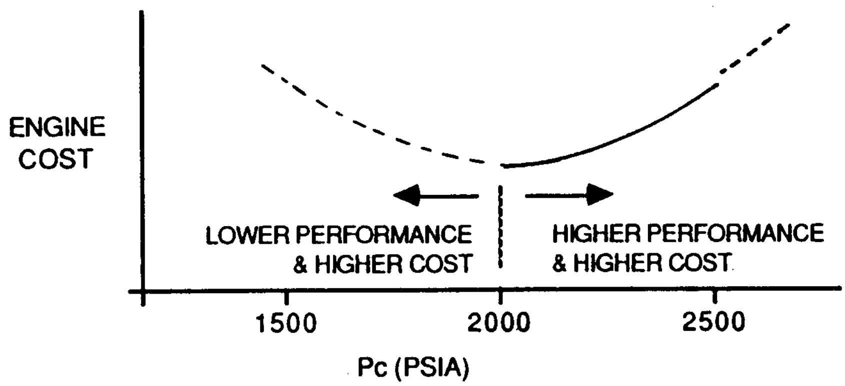


Figure 3.7.2-1. Effect of chamber pressure on LO2/LH2 engine production cost

With increase of the chamber pressure, the weight and cost of the turbopumps increase while the weight, size and cost of the thrust chamber decrease. A minimum is predicted near 2000 psia. At about 2500 psia, it is advantageous to go from two stage hydrogen pump to three stage pump because of large decrease in pump efficiency for a two stage configuration. Hence an increase in complexity and jump in cost is expected near this pressure.

However lowest engine cost at lower performance does not automatically mean lowest vehicle cost. As indicated in Figure 3.7.2-1, the chamber pressure range of interest is above 2000 psia as chamber pressures below this pressure result in lower performance of the engine plus higher cost.

Our chamber pressure trade shows that the booster cost is a weak function of chamber pressure. A chamber pressure of 2250 psia gives the lowest life cycle cost. Interestingly, this is the current baseline pressure for the STME program. This choice of pressure also provides about 10% margin from 2500 psia, where there is added complexity of another stage on the hydrogen pump. Hence a chamber pressure of 2250 psia is recommended for the LO2/LH2 booster.

Assumptions

The specific assumptions made for this trade are:

1. Engine nozzle area ratio is 20 (from nozzle area ratio trade).
2. LRB diameter is 18 ft (from length and diameter trade).
3. Lift-off thrust to weight with one engine out is 1.2.
4. Life cycle cost excludes contractor fees, government support, and contingency.

Results. Figure 3.7.2-2 shows the impact of chamber pressure on various vehicle parameters of interest. The vehicle synthesis runs at each chamber pressure are made with minimum engine thrust as the optimizing parameter (with thrust to weight with one engine out constraint of 1.2) as discussed in the selection of optimization parameter.

There are three main points of interest shown in Figure 3.7.2-2. First, all the vehicle parameters are weak functions of chamber pressure in the range of chamber pressure considered. Second, GLOW, LRB dry weight, LRB length, and engine thrust decrease with increase of chamber pressure because of increased engine performance. And third, although the engine thrust decreases with increase of chamber pressure, a flat minima in the life cycle cost occurs at chamber pressure of 2250 psia, because of increase of cost with chamber pressure as shown in Figure 3.7.2.-1.

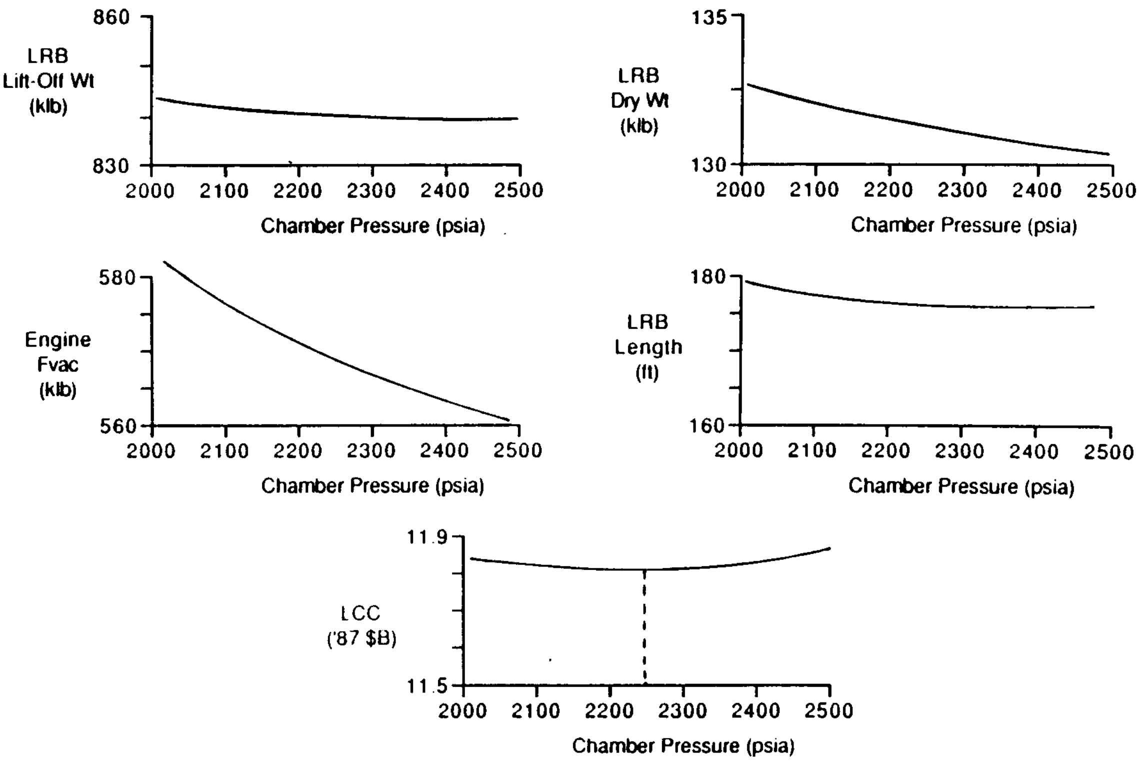


Figure 3.7.2-2. LO2/LH2 Pump-Fed LRB Chamber Pressure Trade Results

Conclusions. Chamber pressure of 2250 psia is recommended for the LO2/LH2 vehicle because:

1. Life cycle cost shows a flat minima at 2250 psia.

2. Chamber pressure of 2250 psia provides 10% margin from 2500 psia at which point there is increased complexity in the LH2 pumps (2 to 3 stages), and jump in cost.

LO2/RP1 PUMP-FED VEHICLE: The chamber pressure for the LO2/RP1 GG cycle is recommended not to exceed 1600 psia because of potential coking problem in thrust chamber cooling passages associated with RP-1 at high heat flux conditions. The cost of the engine follows a curve similar to Figure 3.7.2-1, and decreases as the chamber pressure increases due to decrease in thrust chamber/nozzle size. The maximum chamber pressure selected for this engine at EPL is about 1400 psia, to allow some development risk margin.

LO2/RP1 PRESSURE-FED VEHICLE. The experience with LO2/RP1 pressure fed propulsion system is very limited. There are number of questions which need to be resolved in the technology program. Because of absence of the pumps, there is direct coupling between the combustion chamber and the propellant tanks. Experience in running LO2/RP1 engine at these low pressures is almost non-existent. In order to perform a trade on chamber pressure, it is essential to have an understanding of these issues. Rocketdyne has done preliminary analysis on the impact of chamber pressure on combustion stability, POGO system, cost and weight. Details of these analysis are given in the Engine Report, Appendix-9. The main conclusions drawn from the analysis and used in this trade are:

1. Preliminary POGO system analysis indicates no impact of chamber pressure.
2. Preliminary combustion stability analysis suggests:
    - Injector pressure drop should be 25%;
    - Combustion efficiency of 96% is achievable;
    - There is higher stability margin at higher chamber pressure for the same combustion efficiency; alternately, it can be interpreted that for the same stability margin, one can have higher combustion efficiency.

    Hence for this trade, we ran two cases; Case 1: with constant combustion efficiency of 96%, and Case 2: with combustion efficiency increasing with chamber pressure, indicative of constant stability margin (Figure 3.7.2-3).
3. The cost and the weight of engine, for the same thrust, decrease with increase of chamber

pressure. This is the thrust chamber and injector size decrease with increase in chamber pressure.

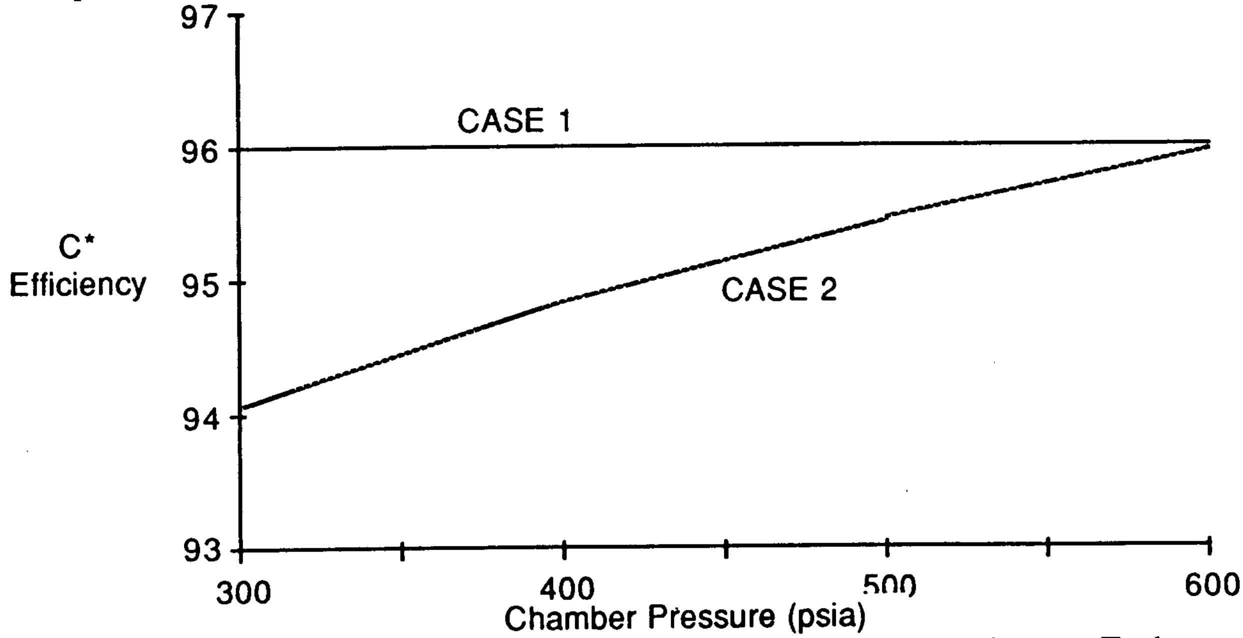


Figure 3.7.2-3. Combustion Efficiency Assumed for Chamber Pressure Trade

Assumptions

Specific assumptions made for this trade are:

1. The tanks are made of 2219 Al alloy (see materials trade);
2. Tank pressurization is achieved using He/O2/H2 catalytic pressurization system (see pressurization trade);
3. For vehicle sizing, minimum LCC is used as an optimization parameter at each chamber pressure;
4. Exit nozzle diameter is 108 in.;
5. Minimum combustion chamber feasible with 60% throttling, without undue engine development risk, is 334 psia. It should be noted that engine development risk increases with decrease in chamber pressure, and this has taken into account in engine DDTE costs;
6. LCC excludes contractor fees, government support and contingency.

Results. The effect of chamber pressure on various vehicle parameters is shown in Figure 3.7.2-4. The runs were made for constant combustion efficiency of 96%, and with varying combustion efficiency indicative of constant combustion stability margin.

The main points to be noted from Figure 3.7.2-4 are:

1. Qualitative difference between the shape of constant combustion efficiency case and varying combustion efficiency case curves is negligible;
2. GLOW and length of the vehicle show flat minima at about 400 psia;
3. There is slight increase in engine thrust with increase of chamber pressure;
4. Dry weight and life cycle cost (LCC) increase monotonically with increase of chamber pressure in the pressure range considered. However change in LCC between 334 and 400 psia is quite small.

Qualitative considerations.

1. Lower tank pressure results in thinner tank walls. The thickness of the walls here is of the order of 1 in. Thinner material thickness allows:
   - -Single pass VPPA welding
   - -More accurate inspection of materials and weld (ultrasonics & photographics)
2. Lower tank pressure results in smaller and lighter pressurization system. The amount of helium required for pressure fed booster is of the order of 5000 lbs.
3. Lower tank pressure, which results in thinner tank walls, puts lower demand on tank manufacturing technology, and hence lower risk.

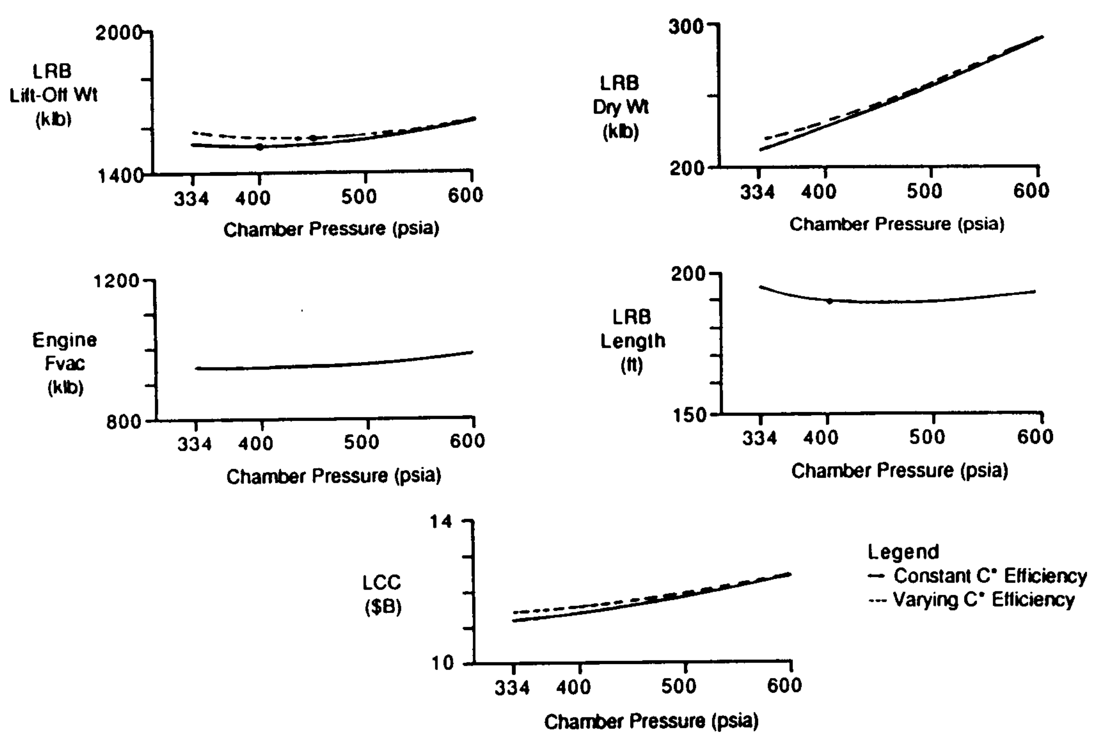


Figure 3.7.2-4. LO2/RP1 Pressure -Fed LRB Chamber Pressure Trade Results

4. Higher chamber pressure results in smaller diameter injectors and combustion chambers, and hence easier engine fabrication.
5. Lower tank pressure results in lower stored explosive energy. Table 3.7.2.-1 shows TNT equivalent of pressure energy at some typical pressures. The three main points to note here are: a) The pressure energy increases faster than the pressure, b) There is order of magnitude difference between the stored pressure energy between pump and pressure-fed vehicles, and c) The tanks, if possible, should be designed for leak before burst considerations.

Table 3.7.2-1. TNT Equivalent of Pressure Energy Variation with Tank Pressure

TANK PRESSURE PSIG	TNT EQUIVALENT PRESSURE ENERGY (LB)
50	133
70	210
500 (Pc ~334psia)	2550
700 (Pc ~500psia)	3750

Conclusions. It should be emphasized that there is lack of experimental data for the pressure-fed vehicle to anchor analytical work. Technology work is needed in the areas of the POGO system, combustion stability and design, and the pressurization system. Based on our present understanding of the subsystems, we conclude that:

1. Ease in tank fabrication at lower tank pressures balances out the ease of engine manufacture at higher chamber pressures.
2. Chamber pressure of 334 psia results in minimum life cycle cost without undue technical risk, and is recommended for the pressure-fed vehicle.

3.7.3. AREA RATIO OPTIMIZATION. The trade study described here is for the LO2/LH2 vehicle. However the conclusions drawn for for this vehicle are found to be applicable to other other pump fed boosters. Unlike the pump-fed vehicles, the LO2/RP1 pressure-fed vehicle area ratio is limited by maximum nozzle exit diameter.

LO2/LH2 VEHICLE. The average back pressure during the booster flight is approximately 6.0 psia. If booster size is the consideration, nozzle area ratio should be chosen to give nozzle exit pressure close to this pressure. But for minimum life cycle cost for an expendable LRB and thrust to weight constraint of 1.2 with one engine out, it is found that nozzle exit pressure of about 12 psia is optimum (area ratio of 20). The LCC difference between minimum physical size vehicle and minimum cost vehicle is about $0.5 B.

Assumptions:

The specific assumptions made for this trade are:

1. Chamber pressure at EPL is 2250 psia (see chamber pressure trade).
2. Maximum nozzle exit diameter is 108 in.
3. LRB diameter is fixed at 18 ft.
4. Lift-off thrust-to-weight with one engine out is 1.2.
5. Vehicle synthesis run at each area ratio is optimized for minimum engine thrust (see optimization parameter trade).
6. Life cycle cost excludes contractor fees, government support, and contingency.

Results. The impact of nozzle area ratio on various vehicle parameters is shown in Figure 3.7.3-1. As expected, GLOW minimizes at the nozzle area ratio where nozzle exit pressure is approximately equal to flight average pressure. The LRB dry weight mainly consists of engine dependent weights and tank dependent weights. Although tank weight minimizes at about the same point as GLOW, the sum reaches a minima at lower expansion ratio because of change in engine weight.

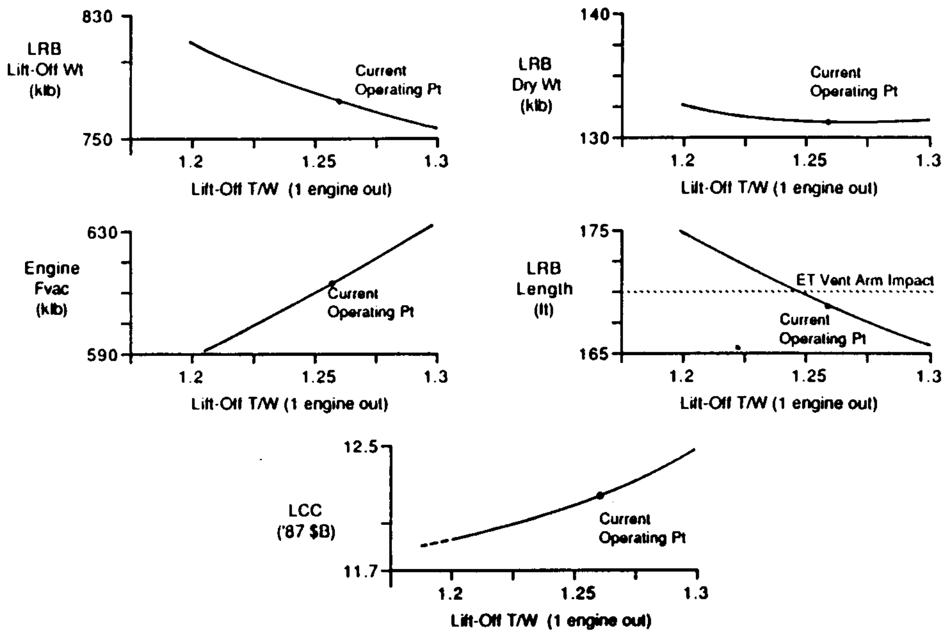


Figure 3.7.3-1 LO2/LH2 Pump-Fed LRB Area Ratio Optimization Results

Engine vacuum thrust continuously decreases with decrease of area ratio. However it is not completely indicative of engine size because of change of specific impulse with area ratio. Engine mass flow rate is true indicator of the engine size. A minima in engine size is expected between 14.7 psia and 6 psia because of two opposing effects as shown in Figure 3.7.3-2. These effects are:

1) Decrease in engine performance, i.e, average specific impulse with decrease of area ratio (for area ratio less than about 35), and
2) Increase in sea level specific impulse with decrease in area ratio (for area ratios greater than about 15) Because of lift-off constraint of T/W, i.e, sea level T/W = 1.2 with one engine out, this impacts the size of thrust chamber.

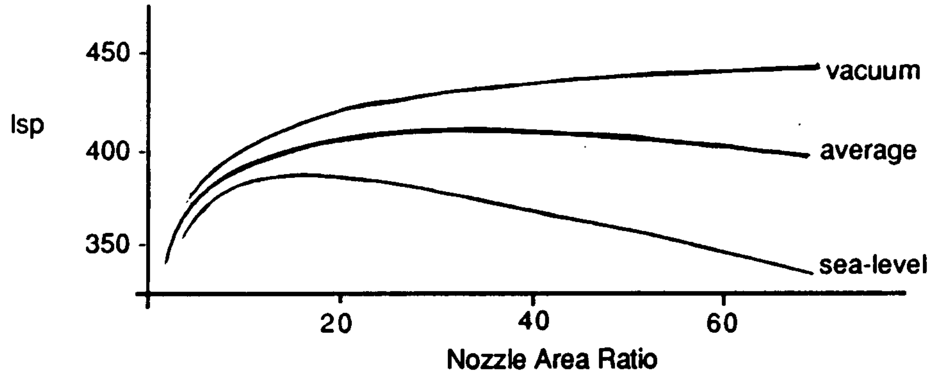


Figure 3.7.3-2 Specific Impulse vs. Area Ratio

As seen in Figure 3.7.3-1, engine mass flow rate reaches minimum between 20 and 30, and starts rising again below expansion ratio of 20. Cost of the engine is function of thrust chamber size, which is proportional to propellant flow rate, and to nozzle size, which is related to the area ratio. Although there is no change in engine mass flow rate between area ratio of 20 and 30, LCC reaches minimum at lower expansion ratio, i.e., expansion ratio of 20, because of decrease in nozzle size. This area ratio corresponds to nozzle exit pressure of about 12 psia.

Conclusion. Minimum life cycle cost of vehicle occurs for nozzle exit pressure of about 12 psia for all LRB booster. This corresponds to an expansion ratio of 20 for the LO2/LH2 vehicle with a chamber pressure of 2250 psia.

3.7.4. ENGINE INLET PRESSURE.

Initially the inlet pressures chosen for the LO2/LH2 concept were the same as for the STME. In the later part of the program, when better definition of the vehicle was available, a study was made to determine the influence of propellant inlet pressures on the vehicle/engine weight and cost for the LO2/LH2 gas generator concept. The skin-stringer type of construction is chosen for the propellant tanks, based on material/manufacturing trade. Tanks designed for stiffness and loads can take 70 psi of differential tank pressure. Data generated by Rocketdyne showed cost, weight and performance benefits with increase of operating inlet pressures. Figures 3.7.4-1 gives the impact of inlet pressures on engine weight. The cost sensitivity follow the weight curves. Other factors like heat exchangers, pressurization lines, pressurants, etc. have secondary effects. The recommended inlet pressures based on the data is 45 psia for the hydrogen side and 65 psia for the oxygen side. The corresponding maximum tank bottom pressures, with vehicle designed to supply these minimum inlet pressures as discussed in Section 5.2.2.3, are 55 psig and 70 psig respectively. And hence no extra requirement is imposed by this change in inlet pressures. The recommeded change in inlet pressures results in a life cycle cost savings of about $100 M over the initially chosen STME inlet pressures.

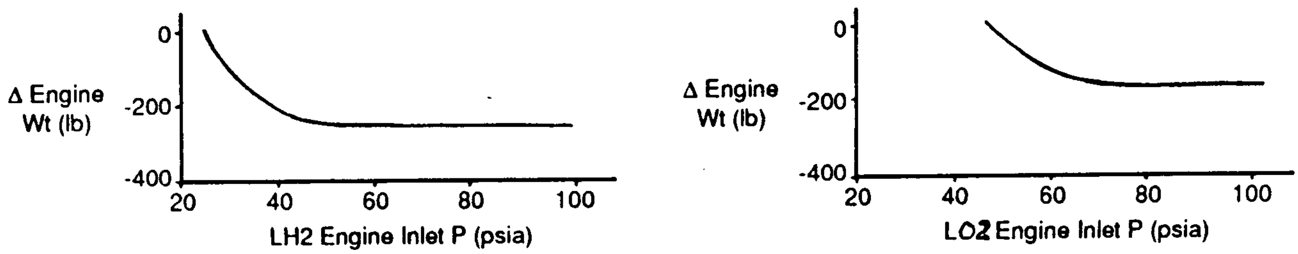


Figure. 3.7.4-1 Impact of Engine Inlet Pressures on LO2/LH2 Engine Weight

A similar trend is shown by the LO2/RP1 gas generator engine concept. For the of split expander cycle, an increase in inlet pressures also benefits the chamber operating pressure. The vehicle cost benefits for pessures greater than above-mentioned pressures are marginal. It was decided to keep the same interface conditions as for the gas generator cycle, as split expander engine is selected as an alternate engine concept. Hence 65 psia and 45 psia inlet pressures were chosen for the oxidizer side and fuel side respectively for all the pump fed concepts.

3.8 PRESSURIZATION SYSTEM

The design of the pressurization system for the main propellant tanks is more critical and complex for pressure-fed engines than for pump-fed engines, because tank pressures are an order of magnitude greater and autogenous pressurization can not be used. The goals are to design a safe, reliable, light weight, minimum volume pressurization system. Impact on launch operations is also important.

General Assumptions.

LO2 Tank	
Volume	11,280 $ft.^3$
Pressure	700 psia
Ullage Temperature	800 ^0R (max)
RP-1 Tank	
Volume	6,330 $ft.^3$
Pressure	700 psia
Ullage Temperature	800 ^0R (max)
Helium Storage Bottle (sphere)	
Pressure	4,000 psia
Ambient He Temperature	520 ^0R
Cold He Temperature	150 ^0R

Systems Evaluated. Twelve different options were evaluated and the results are listed in Table 3.8-1. Most of the systems investigated used helium pressurant, since it is the lightest inert gas and is compatible with LO2 and RP-1.

Option A. This system uses ambient helium (520 ^0R) stored at 4000 psia to pressurize the main propellant tanks. It is the simplest system considered, but it weighs 79,219 lb. and has five large helium bottles as shown in Table 3.8-1. Option A was eliminated based on weight and size.

Table 3.8-1 (A-C) Pressurization system selection

(Part A)

SUMMARY OF PRESSURIZATION SYSTEM OPTIONS

	A Ambient Helium (He)	B Cold He Heated With LO2/RP-1 Gas Generator	C Cold He Heated With LO2/RP-1 Gas Generator and Hot He Coil passing through He storage bottle	D Cold He heated with LO2/RP-1 Gas Generator and Three Storage Bottle Cascade
Pressurant Storage Bottle Components Main Propellant Total Weight	18,529 He 60,690 (Five 14.9 Dia. He) 79,219 Lbs.	11,725 He 14,600 (14.9 Dia. He) 5,000 5,954 37,279 Lbs.	6,846 He 8,707 (12.4 Dia. He) 8,525 5,874 29,952 Lbs.	6,069 He 7,558 (11.7 Dia. He) 667 (4.6 Dia. He) 168 (2.6 Dia. He) 5,398 6,431 26,291 Lbs.
Advantages	• Proven technology • Simple	• Proven technology	• Same as B, but less residual He makes it lighter	•Same as B but less residual He makes it lighter.
Disadvantages	• Very Heavy • Very large volume	• Heavy	• Approximately 600 ft of 3 inch tubing in storage bottle.	• Complex with three different storage bottles.
Safety and Reliability	• Very High	• High	• High	• Medium

(Part B)

SUMMARY OF PRESSURIZATION SYSTEM OPTIONS

	E Cold He heated with Heat Exchanger which is part of Engine Cooling System	F LN2 heated with LO2/RP-1 Gas Generator	G LN2 to pressurize fuel tank and Cold He to pressurize Oxidizer Tank; Both heated with LO2/RP-1 Gas Generator	H LH2 to pressurize fuel tank and Cold He to pressurize Oxidizer Tank; Both heated with LO2/RP-1 Gas Generator
Pressurant Storage Bottle. Components Main Propellant Total Weight	11,725 He 14,600 (14.9 Dia. He) *397 2,142 28,864 Lbs.	44,978 He & LN2 5,750 (10.7 Dia. He) 738 (11.5 Dia. LN2) 10,601 11,967 74,034 Lbs.	23,679 He & LN2 11,218 (13.6 Dia. He) 337 (8.2 Dia. LN2) 6,857 8,116 50,207 Lbs.	9,832 He & LH2 10,885 (13.5 Dia. He) 290 (7.7 Dia. LH2) 6,166 7,346 34,519 Lbs.
Advantages	• Light weight	• Proven Technology	• Proven Technology	• Proven Technology
Disadvantages	• Makes engine more complex and possibly less reliable	• Very heavy	• Very heavy	• More complex than B • LH2 on board
Safety and Reliability	• Medium	• High	• High	• Medium

*Net increase from weight of ablative thrust chamber

(Part C)

SUMMARY OF PRESSURIZATION SYSTEM OPTIONS

	J High Press LO2/LH2 Gas Generator Combustion Products to Pressurize Fuel Tank and to Heat LO2 to Pressurize Oxidizer Tank	K High Press LO2/LH2 Gas Generator Combustion Products to Pressurize Fuel Tank and to Heat Cold He to Pressurize Oxidizer Tank	L N2H4 Decomposition Products to pressurize Fuel Tank and to Heat Cold He to Pressurize Oxidizer Tank	M Cold He Mixed with Small Amounts of H2 and O2 Heated by Catalytic Reaction
Pressurant Storage Bottle	44,130 He,LO2 & LH2 8,060 (11.9 Dia. He) 1,771 (11.4 Dia. LH2) 1,111 (10.0 Dia. LO2)	13,299 He,LO2 & LH2 13,261 (14.3 Dia. He) 746 (8.3 Dia. LH2) 70 (3.3 Dia. LO2)	17,125 He & N2H4 9,570 (12.9 Dia. He) 309 (6.6 DIA N2H4)	8,216 He/H2/O2 Mix 11,740 (13.9 Dia. Mix) 4,701 (9.9 Dia. Mix)
Components Main Propellant	2,792	1,082	5,289 - 1,256	400
Total Weight	84,573 Lbs.	38,913 Lbs.	31,037 Lbs.	25,057 Lbs.
Advantages		•Lighter than B	• Proven Technology • Light weight	• Lightest system • Simple
Disadvantages	• Very heavy • Complex with three different storage bottles • LH2 on board	• Complex with three different storage bottles • LH2 on board	•Requires 10,000 pounds of N2H4 which is toxic	• Needs development for LRB operating conditions
Safety and Reliability	• Low	• Low	• Medium	• High

Option B. Helium is stored cold (150 ^{0}R) to decrease the storage volume. The helium pressurant is heated with hot gas combustion products from a gas generator burning LO2 and RP-1 from the main tanks. The heaviest item in this system is the helium storage bottle which is 14.9 ft.in diameter and weighs 14,600 lb. as shown in Table 3.8-1. This storage bottle will be aluminum with Kelvar overwrap and is quite thick, to withstand the 4000 psia operating pressure. This system requires 11,725 lb. of helium, which is stored at 4000 psia and 150 ^{0}R. As the helium is used, the pressure decreases to 900 psia and the temperature to 83 ^{0}R. The helium is so dense at these conditions that half of the helium remains as residual. The heat exchanger for this system will be quite large. For a shell and tube heat exchanger, the estimated size is 4 ft. diameter and 11 ft. long. The gas generator (GG) plus heat exchanger will weight about 5000 lb. Approximately 18,000 lb. of propellant will be required to operate the GG. The GG combustion products will be used to heat the helium pressurant and will then be dumped through a nozzle which will create approximately 20,000 lb. of thrust. This additional thrust will decrease the required propellant to the engines such that the net increase in propellant is 10,700 lb. This value was converted to equivalent inert weight, since the propellant is being expelled through-out the flight, and the extra tank weight added to determine the net weight of 5954 lb. listed in Table 3.8-1. Option B is fairly heavy (37,279 lb.), but has proven technology.

Option C. This system is the same as option B, except that after the helium is heated, it passes through a coil in the helium storage bottle before being used to pressurize the main propellant tanks. This warms the helium in the storage bottle, which reduced the amount of residual. This system has a total weight of 29,952 lb., which is about 7000 lb. less than option B. It is estimated that it will require 600 ft. of 3 in. tubing in the helium storage bottle. This will make fabrication more complex which will increase the cost.

Option D. This system is similar to option B except that it is a cascade system with 3 helium storage bottles in series. Helium from the large bottle passes through a heat exchanger (Hx) and is used to pressurize the main propellant tanks. Helium from the middle sized bottle passes through a Hx and is used to warm the large bottle. Helium from the small bottle passes through a Hx and is used to warm the middle sized bottle. Thus, only the small bottle has cold residual helium. This system weighs less than either option B or C, but would be more complex to operate.

Option E. This system is similar to option B except that the Hx is part of the engine. Engine heat is used to heat the helium pressurant. This system is light weight (28,864 lb.) and does not have a bulky Hx as does option B. However the engine will be more complex, which will significantly increase the cost. Since it is better to have the pressurization system independent of the engine, this system was eliminated.

Option F. Nitrogen was used as a pressurant, since it can be stored as a liquid which reduces the storage volume. Helium is used to pressurize the liquid nitrogen tank. The nitrogen is heated using a GG burning LO2 and RP-1. This system is very heavy (74,034 lb.) and therefore was eliminated.

Option G. This system is similar to option F except that the LO2 tank is pressurized with helium. The system weight is 50,207 lb. and so this system was eliminated.

Option H. This system uses combustion products from the LO2/RP-1 GG to heat LH2 to pressurize the RP-1 tank and to heat helium to pressurize the LO2 tank. The system weight is 34,519 lb. With two different pressurants, this system is more complex and less reliable than options B and C, and was therefore eliminated.

Option I. This system uses a high pressure GG which burns LO2 and LH2 to form combustion products (H_20 and H2) which are used to pressurize the RP-1 tank. These combustion products also heat LO2 to pressurize the main LO2 tank. The high pressure (1000 psia) LO2 and LH2 tanks are pressurized with helium. This system is very heavy (84,753 lb.) and with three different pressurants is quite complex, therefore this system was eliminated.

Option J. This system is similar to option J except that the LO2 tank is pressurized with helium. The system weight is 38,913 lb. This system was eliminated because with three different pressurants, it is more complex and less reliable than options B and C.

Option K. This system uses hydrazine which passes through a catalyst bed and the decomposition products are used to pressurize the RP-1 tank. Helium is heated by the catalyst bed and used to pressurize the LO2 tank and the hydrazine tank. The system weight is 31,037 lb. This system was eliminated because hydrazine is toxic and more difficult to handle during launch operations.

Option L. This system uses a mixture of helium with small amounts of hydrogen and oxygen and is heated by catalytic reaction. Pressurant from the storage bottle passes through a catalyst bed, where it is heated before entering the propellant tanks (Figure 3.8.1). Pressurant from the make-up bottle passes through a catalyst bed, where it is heated and used to warm the pressurant in the storage bottle, thereby reducing the residual pressurant. Nominal operating conditions are listed in Figure 3.8-1. Temperatures in the storage bottle and the make-up bottle are selected so that the oxygen remains a gas. The volume percent of hydrogen and oxygen is a nonignitible mixture and is designed to give the required pressurant temperature. The system weight is 25,057 lb., which is less than any other pressurization system investigated.

Selected Pressurization System. The catalytic heated helium system (option M) was selected as the potentially best pressurization system for LRB. It is light weight, simple and does not use LO2 or RP-1 and so is completely independent of the vehicle. It is shown as a single system (Figure 3.8-1), but two smaller systems could be used, one located near the LO2 tank and the other located near the RP-1 tank. This would reduced the length of the pressurization lines. The catalyst bed is much smaller than the heat exchanger used in options B and C. The disadvantage of this system is that it has never been used on a large scale as will be required for LRB. The pressurant, which is helium with about 4 percent water vapor, will enter the main propellant tanks at approximately 800 ^{0}R. There may be some frost formed at the LO2 surface. The amount of frost should be quite small, as only a

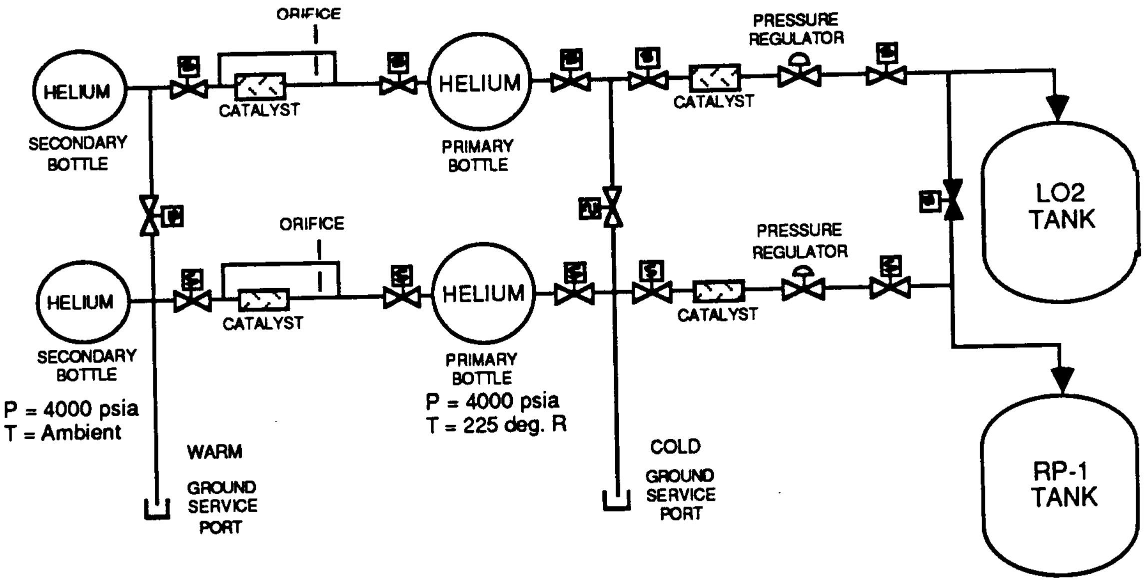


Figure 3.8-1 Selected pressurization system pressurant heated by catalytic reaction

small fraction of pressurant will come in contact with the LO2. This concern will be addressed early in the system design.

The gas generator/heat exchanger system (option B or C) was selected as a backup system for LRB. This system is heavier and larger than the catalytic heated helium system (option M), but is of proven technology. The challenge will be to design a light weight, compact efficient heat exchanger.

3.9 TVC & CONTROL

3.9.1 THRUST VECTOR CONTROL. Thrust Vector Control is a critical capability for controlling STS vehicle ascent. There are several basic methods available for TVC including: moveable engine/nozzle, secondary injection, mechanical exhaust deflection, and differential throttling. These are indicated in Figure 3.9.1-1. Of these possibilities, the moveable engine/nozzle approach has been selected. Secondary injection and mechanical exhaust deflection are considered impractical for a liquid propellant booster of this size. Our rationale for eliminating differential throttling is given in section 3.9.5.

THRUST VECTOR CONTROL METHOD

MOVEABLE ENGINE/ NOZZLE
- Hydraulic
- Blowdown Electrohydraulic
- Turbomechanical
- Electromechanical

SECONDARY INJECTION
- Cold Liquid Injectant
- Liquid Propellant Thrusters
- Combustion Chamber Bleed Gas

MECHANICAL EXHAUST DEFLECTION
- Hydraulic
- Electromechanical

DIFFERENTIAL THROTTLING

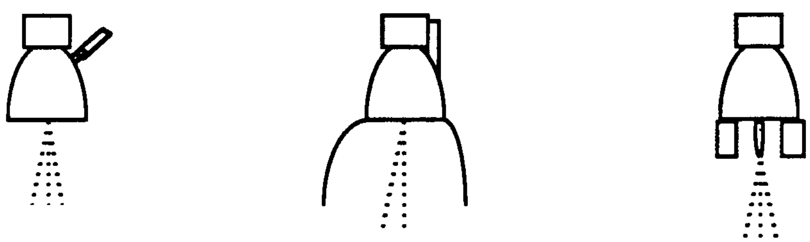


Figure 3.9.1-1 We have selected hydraulic and electromechanical actuation of moveable engines as our candidates for thrust vector control.

The candidate methods selected for the moveable engine/nozzle were hydraulic and electromechanical actuation. Other systems either had little promise for cost and weight reduction or were undeveloped and unproven. Lately, there has been considerable attention given to the EMA vs. hydraulic systems for TVC as well as aerodynamic flight control surfaces. Some of the pros and cons of EMA vs. hydraulic are given in Table 3.9.1-1. Most of these criteria are qualitative in nature. They will remain so until further vehicle systems definition and cost modeling reflects the differences between systems in cost figures, and EMA technology for large systems has been demonstrated.

Table 3.9.1-1. EMA vs. TVC Comparison

ISSUES/CRITERIA	ELECTROMECHANICAL	HYDRAULIC
DDT&E Costs	- High for developing new technology	+ Lower, for dealing with established technologies
Operations Costs	+ Much lower testing and checkout costs, much simpler GSE for test of TVC components	- Time consuming checkout, elaborate GSE and test procedures required, with regular maintenance.
Power level req	- large motors are new technology	+ Within existing technology
Power Distribution	270 Volt system is demonstrated	Existing technology
Maintainability	+ Electrical components/interfaces are simple to repair or replace	- Repair/replacement of hydraulic assemblies requires disassembly of other major subsystem interfaces
System Weight	+ Somewhat less than hydraulics	- Slightly more than EMA
Packaging Flexibility	+ Good flexibility, high density packaging	- Packaging flexibility limited due to fluid lines and APU
Storability	+ Good storability	- Regular visual & instrumentation checks required to detect leaks & contamination
Redundancy	+ Can provide power redundancy through torque summing	- Redundancy provided for actuators, difficult with power source
Control Electronics	Fault tolerant Controls	Majority Voting Fault tolerance
EMI Characteristics	- Can be a problem	+ No contribution
Proven Flight Capability	- Small scale only, a new technology for large scale systems	+ Many systems flight qualified & flown
Simplicity	+ Reduction of complex plumbing, APU, and checkout procedure	- Established and understood procedures, but complex systems
Reliability	Excellent reliability proven for small systems, & expected for large system designs	Never had a flight failure
Environmental Suitability	Problem with high temperatures.	Suitable for high temperature and temperature delta's although high vibration and shock can loosen mechanical connections, causing leaks and degrading system performance.

+	Advantage relative to other system
-	Disadvantage relative to other system
No Mark	No major difference

C-2

We have come to the following conclusions regarding the EMA vs. hydraulic TVC trade:

- EMA technology is advancing such that high power EMA is currently feasible.
- Weight of the EMA vs Hydraulic system is comparable, with the EMA being slightly less.
- DDT&E is higher for EMA than Hydraulic.
- Recurring costs are lower for EMA, principally due to lower operations costs from a much simpler system in terms of test, checkout, and ground operations.
- The LCC of EMA will be less than Hydraulic, because the savings in recurring cost will more than offset the increases required in DDT&E.
- Many of the qualitative discriminators presented in the comparison will become quantitative in terms of cost, when a complete detailed cost model of the subsystem components and operations can be made.

Our recommendation is to continue to baseline EMA TVC, but retain hydraulics as an option, pending further trades and more detailed system and cost definition in Phase B.

3.9.2 TVC ANALYTICAL REQUIREMENTS. There are three primary analyses that must be done in order to define TVC requirements for an EMA or hydraulic system. These include: establishing vehicle mass properties and performance characteristics, performing vehicle ascent control simulations to establish TVC gimbal angle and rate requirements, and calculation of TVC torque requirements based on results of the ascent simulations.

To evaluate the control aspects of the candidate vehicles, suitable models were developed early in the study. These models were used to make preliminary assessments of the controllability, stability margins, and actuator requirements of typical LRB designs. GDSS developed and utilized two software models to support preliminary investigations into the dynamics and control of the STS outfitted with proposed LRB configurations. These models were a mass properties model and a six-degree-of-freedom ascent trajectory model. The mass properties model computes the center of gravity and the principal moments of inertia about the center of gravity from the weights and physical configurations of the various elements of a booster.

The six-degree-of-freedom model was developed in sufficient detail to permit an assessment of basic forces and moments on the vehicle, aerodynamic effects, and actuator motion. The model generates an STS trajectory from lift-off through booster burn-out and separation. The mass properties are derived from the mass properties model for identical configurations. A KSC wind profile (95%) is implemented, which includes gusts and shears. This simulation correlates well with NASA published results.

A spreadsheet model was developed to generate TVC performance requirements based on the results of our ascent simulation, and the mass and performance characteristics of the LRB engines. This spreadsheet calculates the nine torque requirements imposed on the TVC actuator, and defines the power requirements placed on the system. The results of this analysis are given in the TVC section for each engine concept under consideration (sections 4.2.3, 5.2.3, 6.2.3, and 7.2.3).

3.9.3 SCENARIOS MODELLED. Four candidate configurations for the proposed LRB system were investigated with these models. Although three of the configurations were not carried forward to the final selection, the results obtained still provided insight into the basic capabilities of the designs. One of the configurations (the pressure-fed engine) was carried forward as a candidate until the final downselection during the study extension phase.

For each of the configurations examined, several critical periods during ascent were investigated. These included the nominal LRB separation time, along with LRB separations for aborts, and LRB engine out conditions near liftoff, at max-Q, and near burnout.

A significant difference in the adoption of LRBs as replacements for the SRBs is the introduction of propellant sloshing. Although this feature has not been incorporated into simulation programs, a review of the dynamics indicates that the frequencies associated with sloshing are somewhat higher than the natural frequency of the STS vehicle, and consequently this is a manageable problem.

3.9.4 RESULTS & OBSERVATIONS. The results obtained from these simulations indicate that the Orbiter gimbal motions required (in position and rate) to maintain control for the conditions listed above are within current STS limits (less than 10.5 deg deflection from null and less than 10 deg per second rates). This indicates that the gimbal requirements for the LRB configuration are no more severe than for the SRB and because there are several engines per LRB, additional flexibility in distributing the motion is offered (see Figures 3.9.4-1 and 3.9.4-2). A summary of SSME, SRB, and LRB TVC (with nominal, engine out, the adopted specification) performance requirements are provided in Table 3.9.4-1

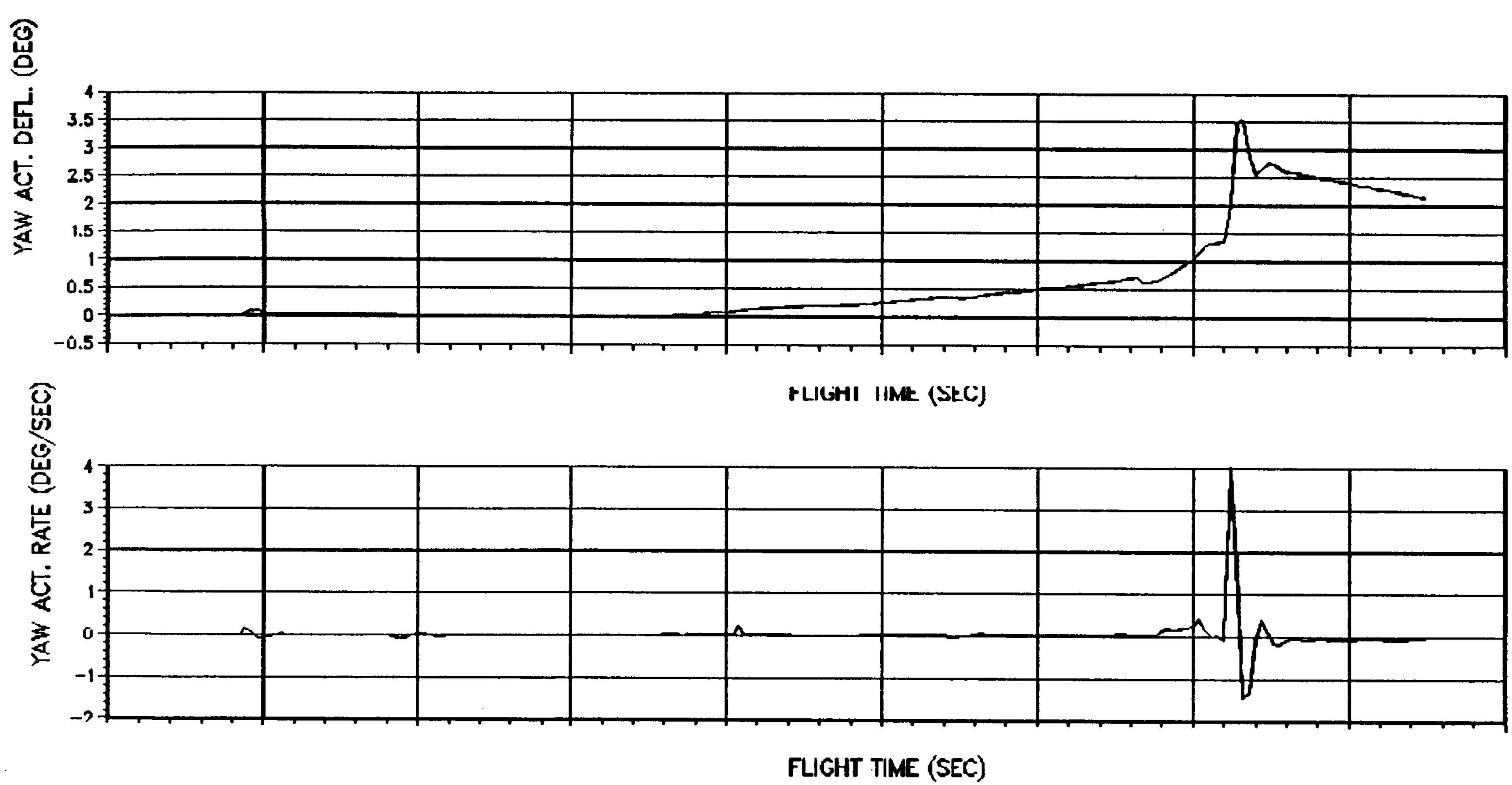


Figure 3.9.4-1. Typical LRB Gimbal Deflection And Rate (Yaw Plane)

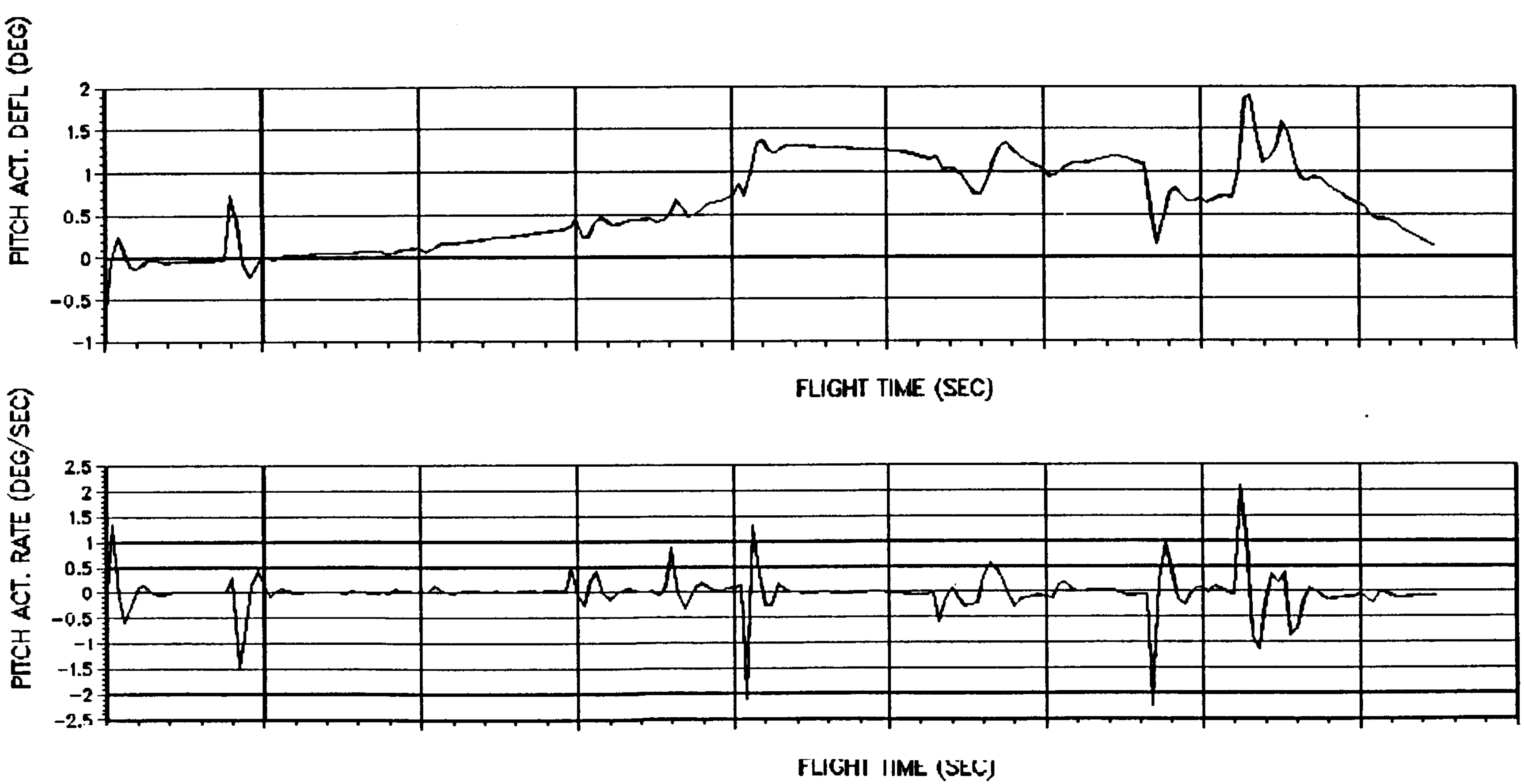


Figure 3.9.4-2 Typical LRB Gimbal Deflection And Rate (Pitch Plane)

Table 3.9.4-1 LRB specifications for TVC performance requirements provide a good safety margin and are in line with current SSME and SRB requirements.

	SSME	SRB	LRB Nominal*	LRB Engine-out*	LRB Specification
Max Gimbal Angle (deg)	±10.5 ±8.5	±5.0	±1.6	±3.8	±6.0
Max Slew rate (deg/sec)	10.0	5.0	2.8	4.3	10.0

*from our 6-DOF ascent simulation

To insure that the integration of LRBs is compatible with the STS, the flight control system needs to have an aerodynamic stability margin similar to the current SRB or more. Aerodynamic stability directly influences the flight control system gains; as the vehicle tends to become unstable, the gains must be increased. This is undesirable since it makes both system design and vehicle control more difficult.

In the pitch axis, the configurations examined were stable, or very nearly so (see Figure 3.9.3-3). For the pressure-fed configuration, the LO2 tank forward alternative is slightly more stable than having LO2 aft. Although this analysis was done early in the design process, the required movement of the center of gravity to insure stability is small enough (on the order of 2-4 feet) that it can most likely be accommodated by minor design changes for any of the candidate configurations.

3.9.5 STEERING BY DIFFERENTIAL THROTTLING. The possibility of steering the STS vehicle by differential throttling of the LRB engines was investigated. Such a technique would alleviate the need for gimballed engines, with the inherent complexities of actuators and flexible feedlines (which is of particular interest for the pressure-fed option).

While it might be possible to design and build a system which steers by throttling, the engineering assessment at this time indicates it is not practical to do so.

Several significant difficulties were identified. First, roll control authority is difficult to achieve. It probably requires that some of the engines be canted away from the centerline of the booster, thereby producing thrust losses. While the SSMEs may provide roll stabilization for portions of the trajectory, the boosters are necessary to accomplish planned roll maneuvers characteristic of the early portion of the trajectory.

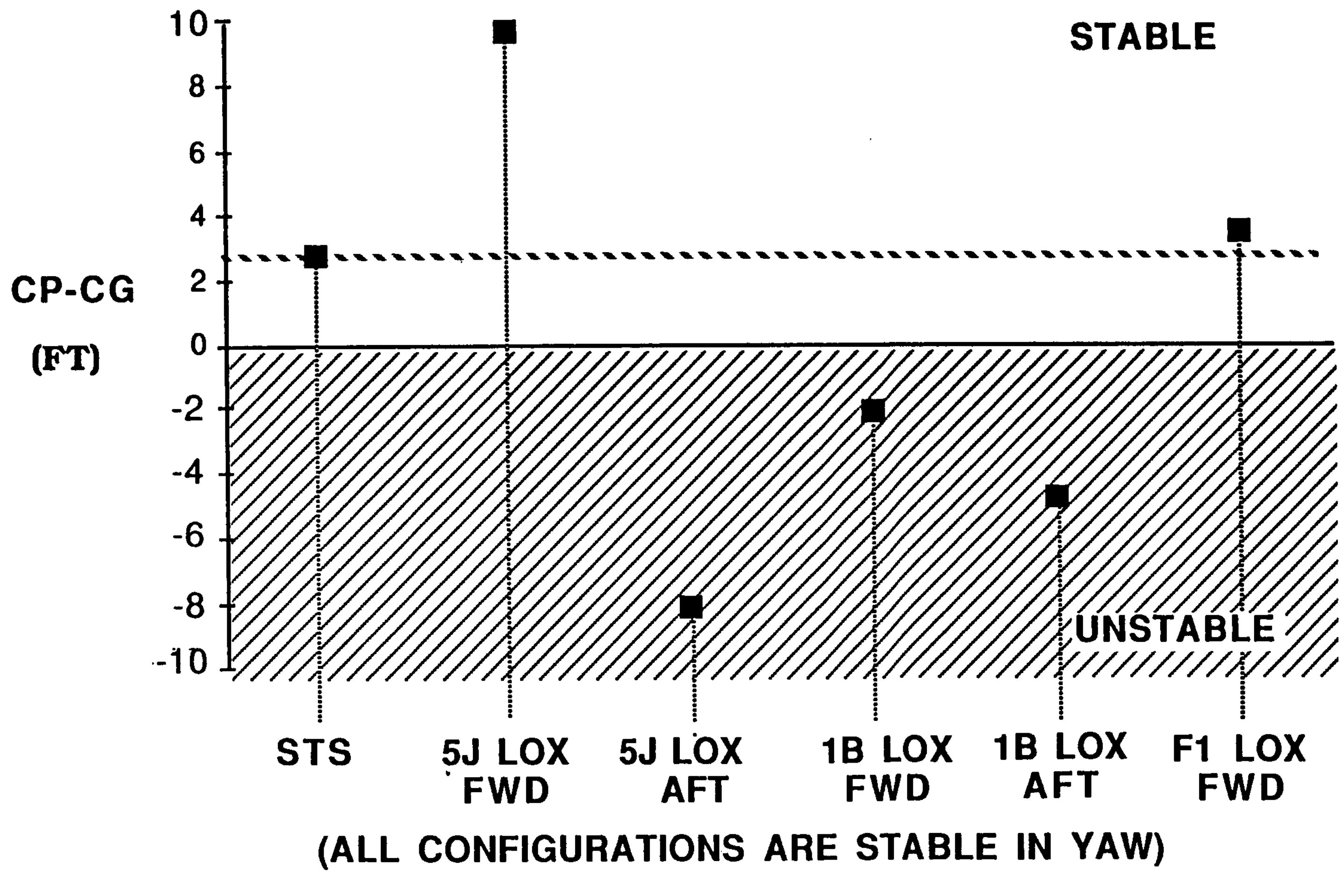


Figure 3.0-4.3 Pitch Plane Stability Margin

Secondly, the throttle rates for typical candidate engines probably are insufficient to provide timely response to control commands. For example, the SSMEs can be throttled at about 10% per second; preliminary analysis shows that rates greater than 30% per second are required for worst-case conditions.

Finally, propellant management for a dual-booster system cannot be accomplished so that the efficient use of fuel is assured. Any continuous bias in the mass flow rate between the boosters contributes to non-synchronous depletion of the propellants. Whatever propellant is left in the tank represents loss of performance capability. Biased flow rates are very likely in the presence of cross winds or other non-symmetrical disturbances.

Differential throttling for steering has been judged impractical at the present time. As the LRB program matures in Phase B, further analysis could be performed to re-assess the magnitude of the problems identified here in the presence of newer technology or revised LRB design.

3.10 TANK MATERIALS SELECTION

The objective of the LRB tank selection was to develop the optimum high performance yet low risk tank design for both pump and pressure fed Liquid Rocket Booster Systems. Light weight high-strength tanks could significantly reduce the weight and size of pressure fed LRBs. For this reason, advanced composite materials have been evaluated. Emphasis was placed on a composite high pressure liquid oxygen and RP-1 tank.

Graphite/epoxy was the material chosen for the highest performance high pressure tank design based on the results of the tank weight vs. tank pressure data as shown in Figure 3.10-1.

Due to LOX being incompatible with graphite/epoxy, a metallic liner is required in the system. This is acknowledged to be an advanced technology which involves risk at this time.

The composite materials recommended for the liquid oxygen tank are as follows: T300/934 prepreg assuming good cryogenic properties data. The driver for the resin system is the long outlife requirement for fabrication of large full scale (≈ 14.0' diameter tanks). Another driver is a low temperature cure requirement. The long outlife requirement drives us to a prepreg resin system due to large wet wound composite structures having delaminations and high void content. A low temperature cure resin is desirable to match the cure requirement of the liner-to-composite FM 1000 film adhesive and to minimize thermal stresses.

In March 1988, it was believed that a load sharing aluminum-lithium liner would be optimum due to a slight weight savings. However, recent IRAD studies at GDSS have shown that aluminum 1100-0 as a non-load sharing liner would be optimum due to better workability and thermal properties.

For the RP-1 tank, graphite/epoxy system T300/934 would be optimum for a high pressure composite tank. A liner would not be required as shown in Figure 3.10-2.

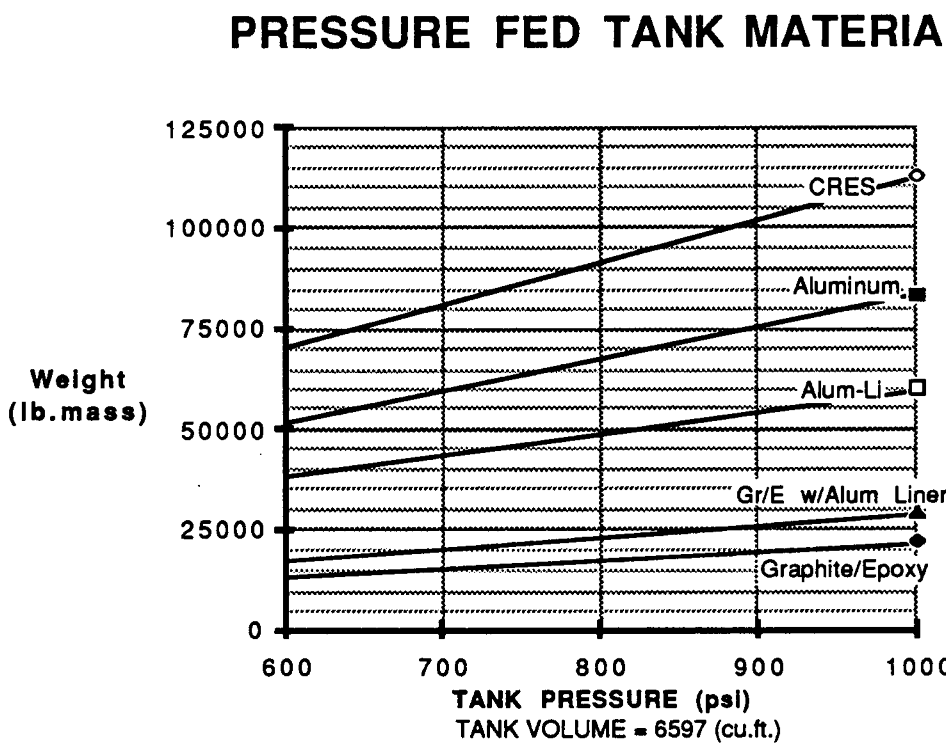


Figure 3.10-1 LRB tank weight verses tank pressure

		MATERIAL			TANK CONSTRUCTION						TANK LINER		
	CONFIGURATION / PROPELLANT	AL-LI	ALUMINUM	GR/EPOXY	STRETCH FORMED	SPIN FORMED	FILAMENT WOUND	MONOCOQUE	ISOGRID	RING FRAMES	ALUMINUM	LOAD SHARING	RIGID INNER SHELL
COMPOSITE	LOX		X	X		X	X	X			X	X	X
	RP1			X			X	X					
METALLIC	LOX	X			X					X			
	RP1	X			X					X			

Figure 3.10-2 Pressure fed comparison matrix

For a pump-fed liquid oxygen tank, an all metallic tank design would have the lowest risk verses an advanced technology composite tank design. A composite design would offer only a slight weight advantage over the metallic design due to reduced allowables caused by the manufacture of large tank structures using thin laminates and a metallic liner. However, a composite design would be preferred for the RP-1 tank due to a metallic liner not being required, but it was not selected to minimize risk.

For a pressure-fed LRB the highest performance may be seen by a tank design incorporating an advanced composite, such as graphite/epoxy filament wound tow. Using filament winding technology from solid rocket motor cases, an all composite tank for the RP-1 propellant would be possible. Fiber-overwrapped metallic pressure vessel technology could also be applied to the design of the liquid oxygen tank, providing liquid oxygen compatibility. Composite tanks were not selected for any LRB concepts due to the risk of advanced technology. At some future date, when programs such as NASP have advanced the state of the art, then composites should be reconsidered. Both pump and pressure fed vehicles will use 2219 aluminum alloy for the tank material. There is a large data base on 2219, it is low cost, reliable and readily welded.

3.11 RECOVERY

The recovery trade study analyzed a broad array of recovery concepts and approaches. This was followed by a downselection analysis that culminated in the selection of the most feasible/practical concepts for LRB application. The final part of the trade study was determining the cost effectiveness of recovery/reuse compared to the expendable LRB.

In the evaluation of the LRB recovery concepts there were two fundamental criteria used; viz, technological considerations and resulting costs. Applicable recovery concepts were first analyzed, evaluated, and then downselected based on technological reasonableness, employing parameters of reliability, risk, and operations impacts. The technological selected concepts were then costed as to development, production, and operations as compared to the expendable LRB.

Both downrange and return-to-launch-site (RTLS) recovery modes were considered in the study. In studying the recovery/reuse of LRBs consideration was given to the option of recovery of the complete LRB or because of the relative cost of high technology components just the propulsion/avionics (P/A) module. Recovery of the P/A module only compared to the recovery of the complete LRB requires the separation of the P/A module from the tankage prior to deployment of the recovery system. A drogue parachute must first be used to slow the descent of the recovered item to a speed that will permit the deployment of either a full parachute(s) or a wing system.

For the downrange recovery of the LRB or P/A module by parachute, the recovered item would simply drop into the ocean for marine recovery, similar to the present SRB operation. Use of a "sock" or "clamshell" protection system in conjunction with the recovery system for a dry water recovery is possible particularly in the case of the P/A recovery. A wing recovery system directing the recovered item to an ocean-going platform, etc, for a conventional deck landing or into a net were options that were also considered in downrange recovery concepts evaluation, as shown in Figure 3.11-1.

The RTLS modes of recovery as shown in Figure 3.11-2, were all considered technically possible.Because of their complexity, however, the reliability of the systems would be low.The RTLS systems considered were of two different techniques; the "toss-back" and the "tow-back. The "toss-back" technique requires, after separation from the Shuttle, reorientation toward the launch site, a propulsive maneuver, followed by slowdown and the deployment and use of lifting surfaces to glide the LRB to a controlled landing in the launch area. The "tow-back" technique uses a ballistic descent after separation from the Shuttle, followed by slowdown and the deployment of

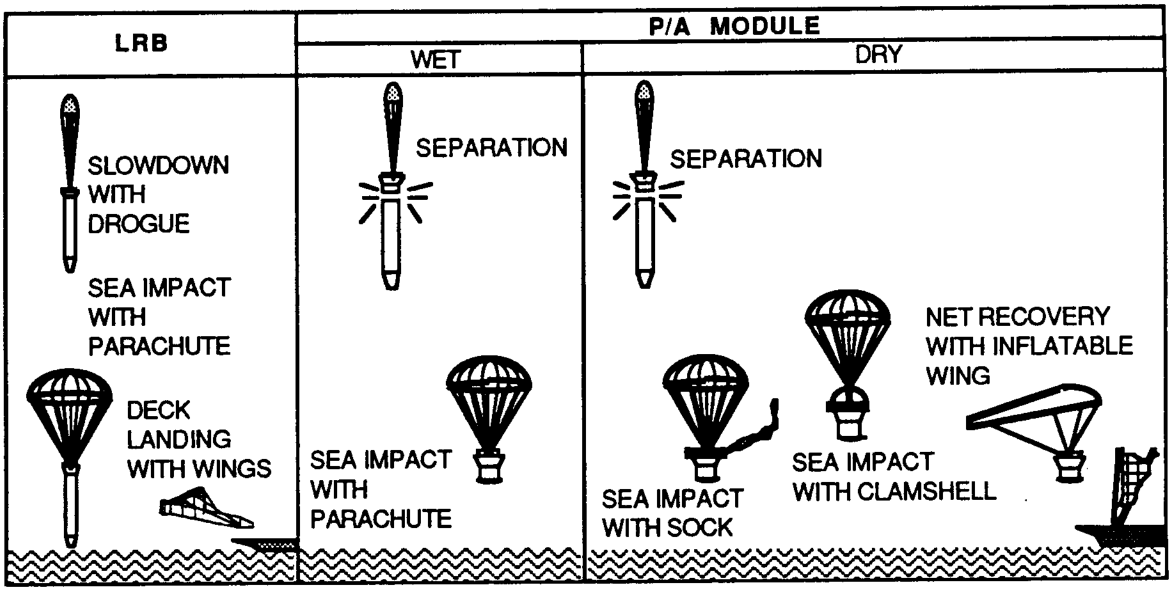


Figure 3.11-1, Downrange Recovery Concepts

lifting surfaces (such as a parawing). After the LRB reaches the lower altitudes and stabilized, a plane of the DC-9 or C-130 class maneuvers to fly in formation with the LRB and capture it with a tail boom. The LRB is then towed back to the launch area where the aircraft releases it for a controlled landing.The landing site would be a dedicated pond close to the launch site, instead of an air strip, allowing a lighter controlled landing and not requiring precise lateral aerodynamic control. Use of a parawing is also applicable for recovery of the P/A module concept.

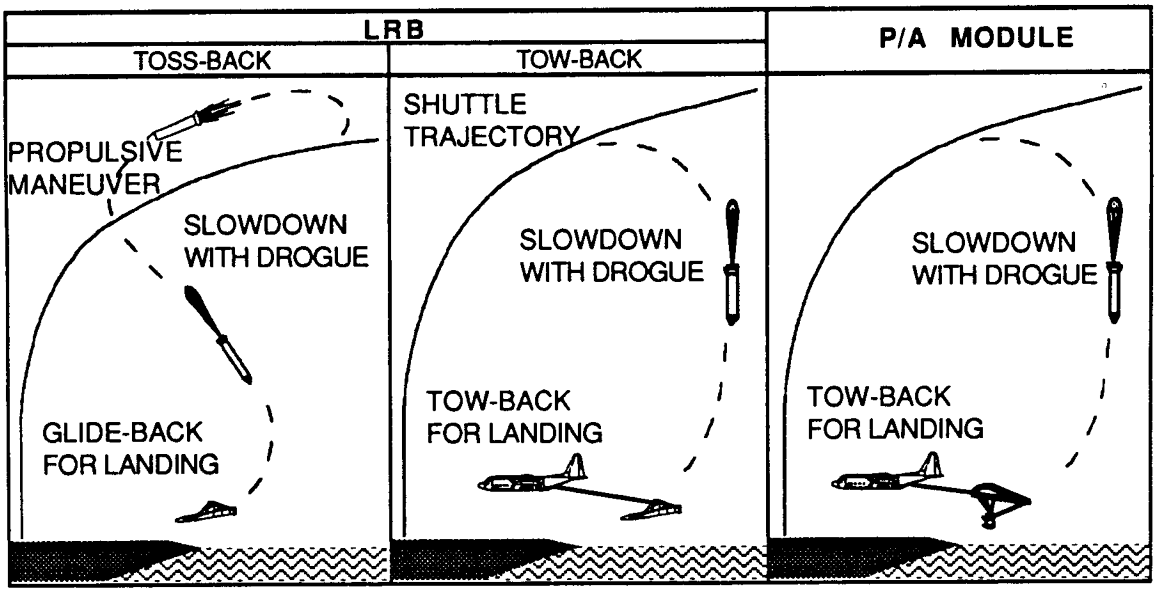


Figure 3.11-2, Return to Launch Site Recovery Options

Four stowable/deployable recovery concepts with lift/drag (L/D) ratios ranging from 0 to 10 for recovery of the LRB were studied; parachute, ram air inflatable, inflatable wing (Rogallo, parawing), and swing-out rigid wing. The parachute recovery system is the most highly developed of the four concepts having been used on past NASA programs as well as the present for SRB recovery. The parachute only concept is also the lightest in weight. The low L/D of the parachute system, however, means that it will have the highest landing velocities of the four concepts. Therefore, the use of only a parachute for recovery means that the vehicle to be recovered has to be structurally strong to withstand the high impact loads as it hits the water going 75 feet per second. The impact speeds could be reduced by utilizing retro-rockets, but this introduces undesirable complexities and additional weight. Although the ram air inflatable and the inflatable wing concepts have higher L/Ds than the parachute which means lower landing velocities the technology development of these concepts is extremely limited particularly for weights above 20K lbs. The one exception is the semi-rigid parawing concept being developed by United Technology Corp. However, there still exists significant development associated with this concept. The developmental risk concern also applies to the swing-out rigid wing concept; the method of swinging out the wings of the size needed as well as the method of stowing the wings would require an extensive development program.

A total of six LRB/recovery-concept configurations (four LRBs with pump fed engines and two LRBs with pressure fed engines) were evaluated for downselection as well as compared to the LRB expendable versions.

Pump fed LRB:

1. Downrange, parachute, P/A module only recovery
2. RTLS, parawing, tow-back, full LRB recovery
3. RTLS, parawing, toss-back, engine restart, full LRB recovery
4. RTLS, rigid wing, tow-back, full LRB recovery

Pressure fed LRB:

1. Downrange, parachute, full LRB recovery
2. RTLS, parawing, tow-back, full LRB recovery

Each recovery concept was evaluated to the following set of parameters:

Technological: reliability, risk, and operations

Cost: development, production, and operations

Assessment of downrange dry recovery concepts revealed them to be of high risk. An extended development program primarily with developing the complex landing systems associated with them would be required. The RTLS parawing toss-back concept was also rated high risk. The requirement for a separate auxiliary propulsion system or restarting the LRB main engines for executing the toss-back maneuver adds unreasonable high risk and cost to the program. Among the RTLS tow-back concepts the parawing is preferred over the rigid wing based upon least complexity, the greater data base on parawings which offers less development time, and lower cost and risk. This concept, however, requires very precise timing and coordination with the tow plane to successfully execute which severely impacts the reliability of this concept. Of all the different recovery concepts evaluated the downrange parachute (without the added complexity of a "clamshell" or "sock" protective system to keep the engines dry) is considered the the most feasible and practical for LRB recovery. The present downrange sea operations system and equipment in use for SRB recovery can also be used essentially intact for the LRB which would have a significant effect on lowered costs for the recovery system development and operations.

From a technological basis the recovery concepts evaluation and downselection identified two recovery concepts to be the most desirable as shown in Figure 3.11-3; the downrange parachute recovery concept (without any "dry" protection scheme) of a pressure fed LRB and the RTLS parawing tow-back recovery concept of the pump fed or pressure fed LRB.The parachute recovery system of the pressure fed LRB is comparable to the current SRB downrange recovery system. It is the least complex of all the candidate recovery systems which, therefore, makes it the most reliable. The parawing tow-back concept for RTLS recovery is lighter than the rigid wing and is lower risk because of its more advanced development. Both downselected recovery concepts employ water landing which makes them significantly lighter than the airstrip or platform landing configurations.This characteristic, in turn, reduces complexity and risk with higher reliability.

A cost comparison analysis was conducted of the downselected recovery options versus the expendable for both the RP1 and LH2 pump fed engines LRB and the RP1 pressure fed engine LRB.The DDT&E (nonrecurring) costs computed in costing were assumed to be for development programs that have 100% achievement. In reality, as recovery and refurbishment systems are added to LRB development there are greater risks and increased probability of cost overrun in comparison with the expendable vehicle.However, even when 100% achievement is assumed, the payback on the DDT&E investment is either marginal or negative. For both the pump fed LRB with downrange parachute recovery of the P/A module and the pressure fed LRB with parachute recovery the cost effectiveness compared to the expendable is only marginal. The DDT&E for these parachute recovery concepts are 13% higher than the cost of the expendable . Based on a LRB

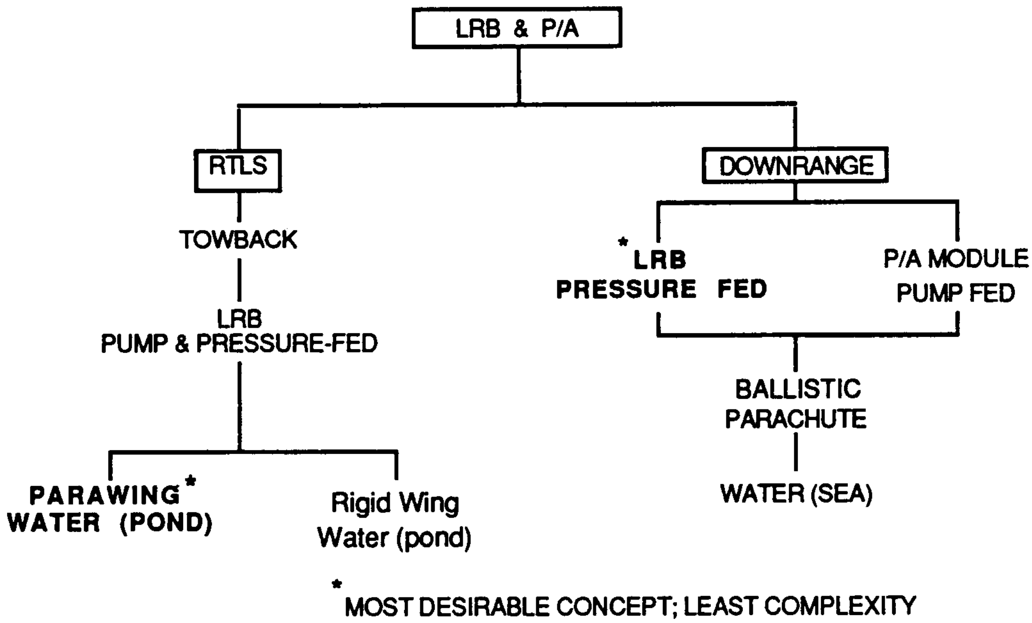


Figure 3.11-3, Technological Downselected Recovery Concepts

nominal model of 81 flights and a 90% reliability for recovery the best life-cycle-cost (LCC) gain that can be expected is 13% (pressure fed LRB parachute downrange recovery).Therefore, from cost and risk considerations based on the relatively low number of flights, it was concluded that recovery-refurbishment-reuse is not warranted.

3.12 SEPARATION SYSTEM TRADE STUDY

To support LRB design, a trade study (number 1.16) was conducted to select the best booster separation system. Current SRB Booster Separation Motors (BSMs) were chosen because they provide the least development risk, lowest cost, and proven reliability. Also, as part of the trade study, several related subjects were examined such as LRB separation for aborts, control requirements during booster separation, and definition of the separation sequence.

3.12.1 SEPARATION SYSTEM OPTIONS EXAMINED. Initially, a large number of separation system options were considered including such approaches as aerodynamic surfaces and slide rails. Of all options identified, the subset below represents those deemed most appropriate for LRBs. LRB separation system options were then examined and evaluated per the criteria shown.

Options Considered
- Separation Motors Used On SRBs
- New Solid Propellant Separation Motors
- A Liquid Propellant Separation System
- New Liquid Propellant Separation Motors
- Spring Thrusters
- Pneumatic Thrusters
- Pressure Bleed (From Pressurization Tanks)

Evaluation Criteria
- Safety
- Reliability
- STS Integration
- System Weight
- Orbiter/ET Impingment
- Cost

Section 1.16 (pp. 320-346) in Appendix 1 to Volume II of the final report contains further description of these systems.

Liquid propellant separation systems were investigated because they can be designed to provide variable thrust and burn times. This is beneficial for abort considerations. It might be possible to separate a non empty LRB for a RTLS abort using a system sized for nominal conditions by adjusting the burn time and impulse delivered. Thus, upscaling for abort requirements might not be necessary. However, liquid systems tend to be rather complex.

Pneumatic and spring thrusters were originally considered for the Shuttle in 1973-4 and are in use on launch vehicles such as the Delta. Thruster separation systems are attractive because they do not produce debris which might impact the Orbiter or ET TPS. However, such systems require excessive structural modification to the ET for attachment and reinforcement.

Systems using pressure bleed from pressurization tanks (for the pressure-fed LRB) were eliminated due to feasibility concerns.

Although the current SRB separation motors were chosen for the LRB separation system, new motors are nearly as acceptable. New BSMs (based on SRB separation motor design) would require additional DDT&E funding. However, their performance could be optimized to LRB separation force and impulse requirements. Also, new BSMs might possibly utilize a different propellant formulation (with a lower content of burn rate and stability additives) which would reduce concerns about impingement on the Orbiter or ET TPS.

3.12.2 ABORT CONSIDERATIONS FOR SEPARATION SYSTEM DESIGN. To develop design criteria for the separation system, abort considerations were examined. Two abort modes are enhanced if LRBs can be separated prior to nominal staging. These are the intact Return-To-Launch-Site (RTLS) abort, and the contingency Downrange Crew Escape abort.

With SRBs, an intact RTLS abort is initiated after booster burnout. Preliminary results for LRBs indicate that it should be possible to initiate a RTLS abort prior to booster burnout. Trajectory simulations show that enough energy and altitude can be gained during a typical LRB ascent, such that if the boosters are shutdown and jettisoned prematurely, the Orbiter (using SSME thrust) can still attain RTLS pitch around conditions; see Figure 3.12.2-1. If the LRBs develop problems late during their boost phase, by careful design, it should be possible to separate the boosters and still conduct an RTLS abort. This enhancement provides the crew with an option not currently available with SRBs.

Likewise, the contingency Downrange Crew Escape abort can be improved. The current Downrange Crew Escape abort utilizes a "Fast Separation" of the Orbiter from the ET/SRB stack. The Downrange Crew Escape contingency abort (using LRBs) would start with booster separation, continue with SSME burn and reorient to a heads-up position (for ET disposal), and then include a controlled glide and crew escape. In contrast, the current "Fast Separation" version of this abort nearly precludes possibility of reorienting of the Orbiter to a heads-up position. Orbiter heads-up flight results in more controlled and stable glide after ET disposal, which in turn improves the chances for crew escape.

To examine the feasibility of separating LRBs for aborts, and to develop a weight trend which shows the penalty involved with designing the separation system for off-nominal conditions, the following scenarios were investigated:

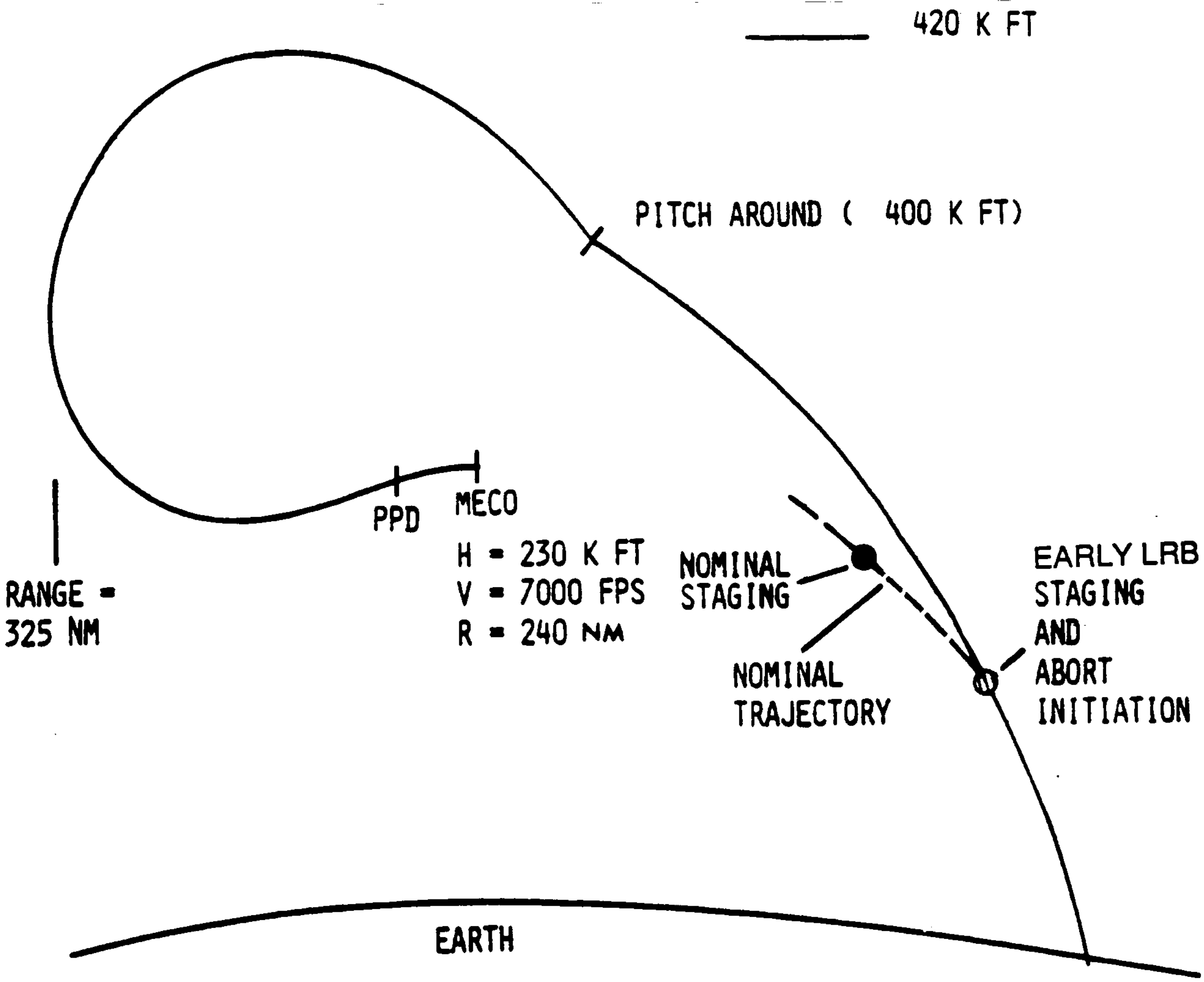


Figure 3.12.2-1. Press To RTLS Abort (Using LRBs)

1) Nominal Separation,
2) Separation For An Early Intact RTLS Abort, and
3) Separation For An Early Downrange Crew Escape Contingency Abort.

Analyses were conducted with a representative LRB design. The booster examined was the Interim Project Review (IPR) version of the LOX/LH2 booster powered by SSME derivative engines. This booster was 188 feet long by 15.3 feet in diameter and its burnout weight was 113,400 lbs.

For the IPR LO2/LH2 pump-fed configuration, nominal staging occurred at 119 seconds. Trajectory simulations tend to indicate that 100 seconds is the approximate earliest staging time for this booster after which the Orbiter can reach specified RTLS pitch around conditions using SSME thrust. The conditions to be reached were assumed to be: an altitude of 400,000 ft, downrange travel of no more than 250 nmi, a velocity of 4800 ft/sec, and a climb rate less than 1800 ft/sec (Reference: NASA Training Manual, GNC Abort 2202 - Return To Launch Site). The earliest contingency Downrange Crew Escape abort was estimated to be possible after 75 seconds of flight when the Orbiter had achieved Mach 2 and 40,000 feet altitude. The separation conditions corresponding to these staging times are shown in Figure 3.12.2-2.

Scenario / Condition	Nominal Separation	Press To RTLS	Crew Escape Abort
Time (sec)	119	100	75
Altitude (Ft)	132,250	91,420	49,850
Mach	4.5	3.35	2.0
Dynamic Pressure (PSF)	81	273	671
LRB Weight (LBs)	113,400	198,900	311,435

Data For LRB SSME-35 Option 5J (Dec. IPR Version)

Figure 3.12.2-2. Separation Conditions For Various Mission Times

Analyzing the staging conditions shown in Figure 3.12.2-2, a LRB separation weight trend was developed. Examine Figure 3.12.2-3.

For each separation case examined, various combinations of Booster Separation Motors (BSMs) were evaluated until separation with safe clearances was achieved. The orientation and layout for the LRB BSMs was the same as that on the SRBs. The NASA six degree of freedom, 3 rigid body, simulation program, 'Space Vehicle Dynamic Simulation' (SVDS), was used to evaluate separation dynamics and clearances. LRB dimensions, aerodynamic data, and mass properties were input. Power-on BSM effects were also included. Current SRB separation body rates limits (5 deg/sec pitch, 2 deg/sec yaw, and 2 deg/sec roll) were used. Reasonably conservative angles of attack (10 degrees), and sideslip (10 degrees) were also used. In lieu of LRB elemental wind tunnel data, free stream LRB element aerodynamic coefficients were predicted with the program, 'USAF Automated Missile DATCOM * Rev 11/85'.

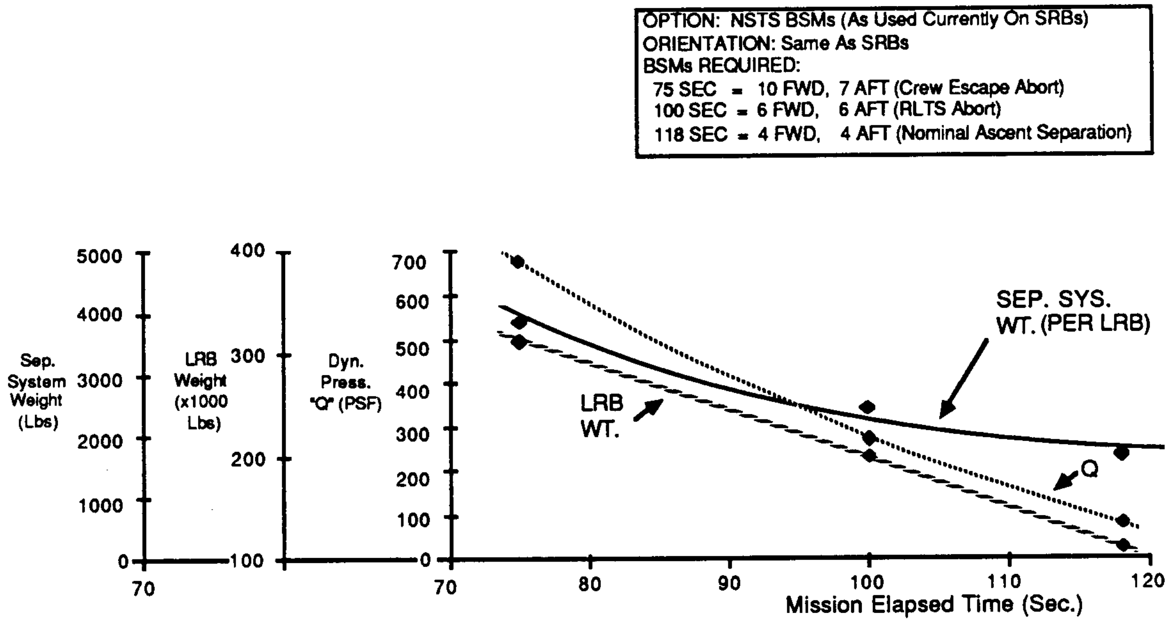


Figure 3.12.2-3. Separation System Weight Trend

Figure 3.12.2-3 indicates that the number of BSMs needed to achieve early booster separation increases significantly. However, because the system weight per BSM is small (on the order of 200 lbs) the overall system weight gain represents only a small (less than 5%) change in the booster inert weight. It was concluded, that when the separation system is designed to satisfy abort requirements, the resulting weight penalty is acceptable.

However, there are other feasibility issues associated with designing the booster separation system to accommodate aborts -- in particular, vehicle control during separation, and the amount of time needed to initiate LRB separation. Preliminary analyses indicate that it will be difficult to provide control for an early contingency abort separation of the LRBs. As indicated in Figure 3.12.2-2, the boosters may weigh as much as three times their nominal separation weight at the initiation of a Downrange Crew Escape abort. When the boosters are jettisoned, a large shift in the mated vehicle center of gravity (CG) results. The SSMEs must provide control moments during and after LRB separation, and the CG shift taxes the capabilities of the control system. The SSME pitch plane response needed is at the limits of the Orbiter control capabilities. The second concern is the amount of time which is available to begin LRB separation for an abort. Time is needed for failure detection, evaluation, and to perform engine shutdown. Failures resulting in commanded aborts are significant system failures, and there may not be enough time to complete LRB separation; a detailed failure effects analysis is required to evaluate this issue further.

Due to above discussed feasibility concerns, and because NSTS 07700 Vol. X, paragraph 3.2.1.5.2.1, states, "... (a) Contingency aborts will not be used to determine hardware design criteria. ", it was decided not to design the LRB separation system for contingency aborts. However, it was concluded that it is reasonable to design the booster separation system to accommodate staging for RTLS aborts -- if it can be verified by more detailed analysis that RTLS abort coverage does indeed begin prior to nominal staging.

3.12.3 SEPARATION SEQUENCE AND CUE. As part of the Separation System Trade Study, options were examined for signaling booster separation. The options included: vehicle state vector (as is used for the Atlas booster engine cutoff signal); mission elapsed time; low liquid level sensed in propellant tanks; and "on command" from Orbiter computers or ground control. A new separation signal is required because SRB separation is initiated on chamber pressure decay. The separation cue selected for LRBs is the sensing of 'Low Liquid Levels' in the propellant tanks of either booster; sensors used will be triply redundant for safety. This approach assures that the engines will not be run dry. If booster engines are run dry there is the possibility for engine damage and failure. In addition, because propellants would run out for each LRB engine at different times, unsymmetrical thrust decays might produce significant thrust differentials (with attendant control problems).

The LRB separation sequence was also examined, and a preliminary logic flow (simplified) is shown in Figure 3.12.3-1. Additional sequence details, such as the commands to null the LRB engines, switch to second stage control, and initiate attitude hold mode, are not shown on the figure. In Addition, the separation sequence is designed to maintain the current requirements for signal interlock (to eliminate the possibility of stray signals causing booster separation), and to provide automatic separation inhibit with manual override capability.

For nominal separation, monitoring for "Low Liquid Levels" in the propellant tanks begins after liftoff; in the event of a commanded RTLS abort, it will be possible to call-up the separation software. After the LRB "Low Liquid Levels" signal has been received, an "ARM" command will be sent to the booster from the Orbiter. This command will be used to trigger separation software contained in the LRB. The "ARM" command will start the engine shutdown sequence, and (after a predetermined time delay) charge the single-channel capacitors used to detonate the separation system pyrotechnics. The need to supply a time delay constant for

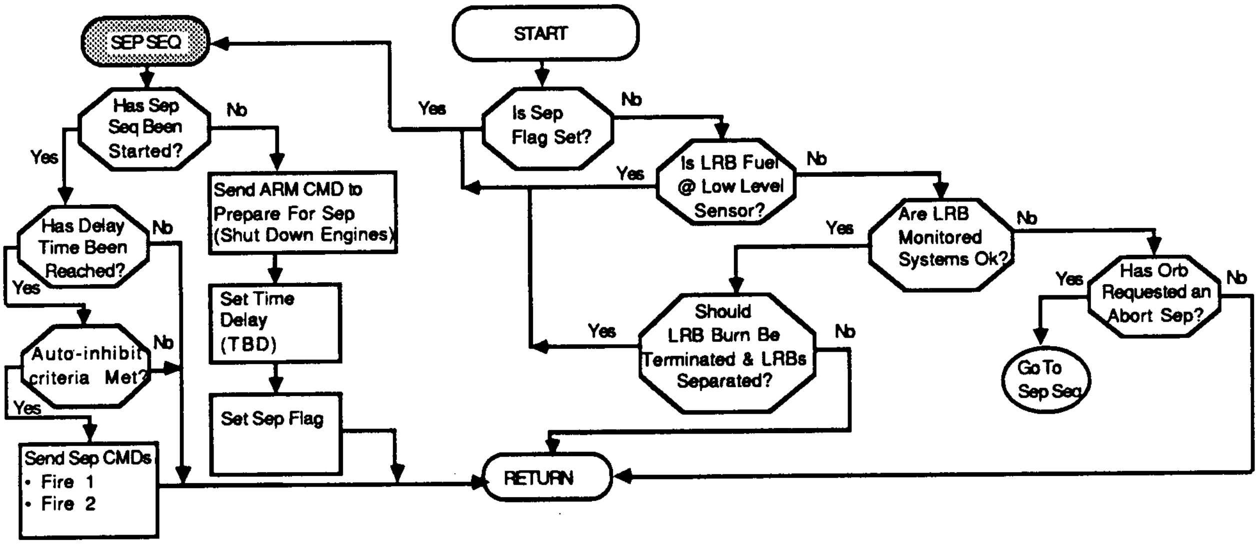


Figure 3.12.3-1. Preliminary Separation Sequence (Simplified)

engine shutdown prior to arming the separation pyrotechnics is a major difference compared to the SRBs. Once the time delay has passed, the Shuttle's dynamic state will then be compared to auto-inhibit criteria (which are limits on vehicle body rates and dynamic pressure). If the criteria are met, FIRE 1 and FIRE 2 hardwire commands from the Orbiter will be sent to the LRB pyrotechnic initiator controllers (PICs). The LRB PICs will in turn fire the BSMs and attachment pyrotechnics. If the auto-inhibit criteria are exceeded, separation will be averted until the criteria are met or manually overridden. In the event an RTLS abort is initiated, either new auto-inhibit criteria will be used, or the auto-inhibit criteria will be ignored altogether.

3.13 LRB CONCEPT DOWNSELECTION

Concept downselection considered all the data developed during the study, especially Propellant selection, Costs, and Evolution and Growth. It had been determined by June 1988 that there were three leading LRB candidates as shown in Figure 3.13-1. From the beginning of the study, there had been concern that LO2/LH2 concepts were "too big". Finally it was clear that an 18 ft diameter with clean exterior could be flown with slightly reduced max alpha-q without exceeding Orbiter wing loads. The smaller thrust section diameter made tower and ET LH2 arm clearance less of a problem than the wide Tailed pressure-fed LO2/RP concept.

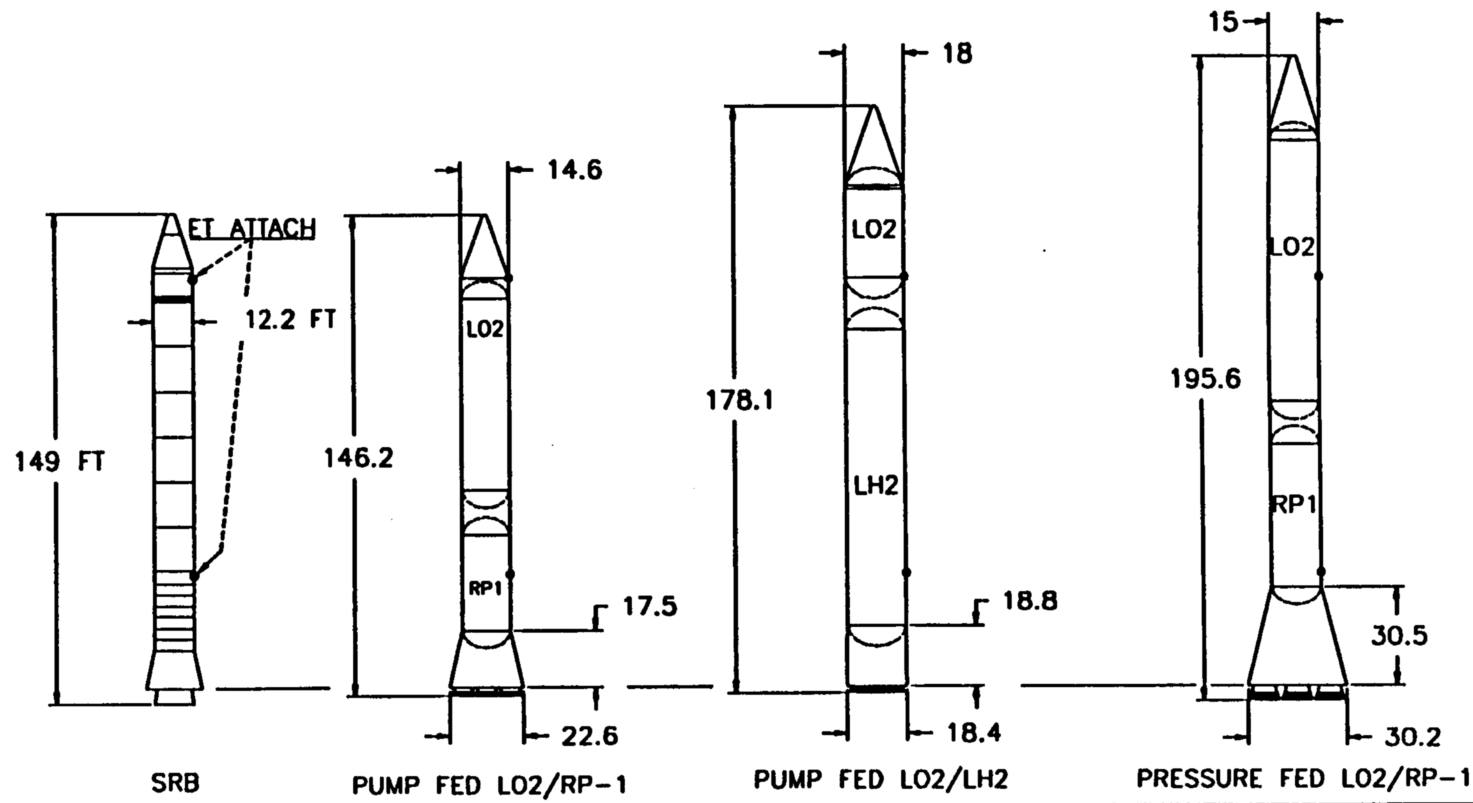
Figure 3.13-2 is a summary of the technical features of the three candidate concepts. All three are acceptable from a safety/environmental impact viewpoint although RP is less flammable than LH2, LH2 has the most benign exhaust, and high pressure is a hazard in the pressure fed concept. In the area of reliability/simplicity, RP is better than LH2, and pressure-fed LO2/RP holds the promise of being the best. Compatibility with the shuttle is about even - pump-fed LO2/RP is the smallest but the light weight and thrust of the LO2/LH2 concept simplify trajectory design. KSC feels that LO2/LH2 would be the easiest to integrate because they are used to operating with these propellants now. The above basic selection criteria do not show a clear winner - all three concepts are viable.

Figure 3.13-3 is a list of estimates on technical risk for various elements of the concepts. The pressure fed concept was judged to have the most technical risk since no pressure liquid engine has ever been built in the million pound thrust class. We endorse the Technology test program planned at MSFC under the "Booster Propulsion" part of the Civil Space Technology Initiative (CSTI). The program would reduce the uncertainty in combustion efficiency, instability, POGO, and pressurization. The risk in the two pump fed concepts was about the same: the complexity of the two cryogens on one is offset by the worry about combustion instability with LO2/RP.

Figure 3.13-4 summarizes the final comparison emphasizing risk, evolution/growth, and costs. We judge the LO2/LH2 concept to have the least technical risk and therefore the lowest probability of exceeding cost and schedule estimates. We believe LO2/RP-1 contamination of the upper atmosphere may someday be recognized as a serious drawback. Costs are within 10% of the same. We estimate the pressure fed LO2/RP concept to have the lowest DDT&E, but about the same life cycle costs as the LO2/LH2 concept. The pump-fed LO2/RP-1 concept has the lowest life cycle cost unless the USAF shares engine development of the STEP with NASA. Our recommendation of the LO2/LH2 concept, then, is based primarily on potential commonalty and cost sharing of the engine and the entire LRB with the USAF ALS program. For further discussion of Evolution and Growth, including commonality with ALS, see Appendix 9.

LRB DOWN-SELECT CONFIGURATIONS

GENERAL DYNAMICS
Space Systems Division



DATA (ONE BOOSTER)	SOLID ROCKET BOOSTER	LO2/RP-1 PUMP FED	LO2/LH2 PUMP FED	LO2/RP-1 PRESSURE FED
DRY WEIGHT (Klbs)	146	116	122	237
LRB GLOW (Klbs)	1,250	1,092	821	1,598
THRUST (sea level)(Klbs) (nominal)	2,912	565 (2260)	515 (2060)	841 (3364)
INITIAL T/W	1.5	1.37	1.46	1.53

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Figure 3.13-1 LRB Down-Slect Configurations

CONCEPT EVALUATION SUMMARY

CRITERIA	LO2/LH2 PUMP-FED	LO2/RP-1 PUMP-FED	LO2/RP-1 PRESS-FED
Safety / Environmental Acceptability	• Acceptable safety • Hydrogen more likely to leak than RP-1 • Environmentally benign • LH2 flammable at 4% in air	• Acceptable safety • Exhaust includes CO and CO2 • RP-1 less explosive than hydrogen • No cryo pumping	• Acceptable safety • Exhaust includes CO and CO2 • RP-1 less explosive than hydrogen • No cryo pumping
Reliability / Simplicity	• Complex insulation requirements- 2 cryogens • Hydrogen leakage detection	• Reduced insulation requirements • Less complex prop. loading system • Fuel storable at ambient conditions	• Reduced insulation requirements • Less complex prop. loading system • Fuel storable at ambient conditions
Compatibility (STS)	• Large size - (no problems identified)	• Smaller size - easier integration	• Large size - (no problems identified)
Compatibility (Facilities)	• Length changes ET vents/arms • Greater impact on RSS,FSS,VAB	• All new propellant logistics • Smaller impact on RSS,FSS,VAB	• All new propellant logistics • Greater impact on RSS,FSS,VAB
Performance	• Lightest: GLOW = 3585 klb • Thrust (Vacuum) = 558 klb	• GLOW = 4128 klb • Thrust (Vacuum) = 635 klb	• Heaviest: GLOW = 5140 klb • Thrust (Vacuum) = 976 klb

GENERAL DYNAMICS
Space Systems Division

QUANTIFICATION OF LRB TECHNICAL RISK

LRB As of Dec. '88 without benefit of Pressure Fed Propulsion Tech. Demo.

WBS ELEMENT	PUMP-FED LO2/LH2 DDT&E	PUMP-FED LO2/LH2 PRODUCTION	PUMP-FED LO2/RP-1 DDT&E	PUMP-FED LO2/RP-1 PRODUCTION	PRESS-FED LO2/RP-1 DDT&E	PRESS-FED LO2/RP-1 PRODUCTION
LRB HARDWARE						
STRUCTURES	10%	10	10	10	15	15
SEPARATION SYSTEM	5	5	5	5	5	5
THERMAL PROTECTION	10	10	5	5	5	5
MAIN PROPULSION	..	..	..	..	..	..
ENGINES	25 ①	25	35	25	50 ②	25
PROPELLANT FEED SYS.	20	20	10	10	33	20
PRESSURIZATION SYS.	5	5	10	10	50 ④	25
ACTUATORS (EMA)	50 ③	25	50	25	50	25
AVIONICS (INC. ORBITER I/F)	20	20	20	20	20	20
ELECTRICAL POWER	10	10	10	10	15	15
SYSTEM TEST	25	10	10	10	15	10
SOFTWARE	100 ⑤	100	100	100	100	100
GSE	10	10	5	5	10	10
INITIAL TOOLING	10	10	5	5	15	15
STS-RELATED MODIFICATIONS						
ORBITER MODS (SOFTWARE)	100 ⑥	100	100	100	100	100
ET MODS (STRUCTURE)	50 ⑦	25	25	25	50	25
FACILITIES (LAUNCH)	25	N/A	25	N/A	25	N/A
STS SE&I	100 ⑥	100	100	100	100	100
LRB LAUNCH OPERATIONS	25	25	25	25	25	25

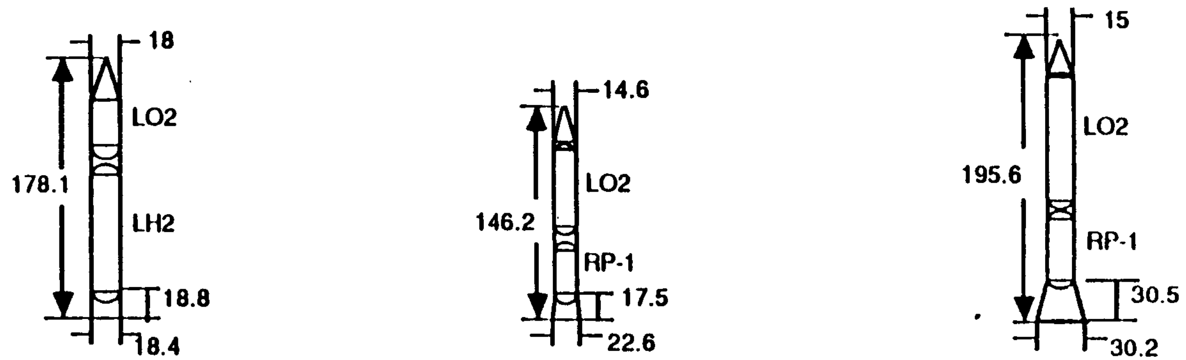
① Any big LH2 engine can produce surprises: hydrogen is complex/expensive; with RP worried about combust. stability
② No big pressure fed engines ever made: worried about combustion stability, injector, cooling, throttling
③ No big electromechanical actuators ever made
④ No big, light pressurization systems ever made: catalyst operation, snow, or large RP GG/Heat Exchanger
⑤ Software always uncertain, particularly interfacing with JSC and KSC
⑥ Shuttle/Centaur showed Orbiter Integration (even just software) is very complex, time comsuming
⑦ ET mods: working with MMC on existing design will be a challenge; even worse at 18' diam, 190' long

Figure 3.13-3 Quantification of LRB Technical Risk

CONCEPT EVALUATION SUMMARY

LRB DOWNSELECTION 12-08-88

EXPENDABLE LRB's FOR STS



VEHICLE / CRITERIA	LO2/LH2 PUMP FED	LO2/RP-1 PUMP FED	LO2/RP-1 PRESS FED
RISK TECHNICAL & SCHEDULE	BEST	MEDIUM : Combustion Instability with Throttling	POOR : New technology engine and pressurization system
EVOLUTION & GROWTH	BEST : 1 common engine for Shuttle-C LRB and SSME, Standalone and ALS LRB and Core.	GOOD : But limited reusability, development of two engines	FAIR : Limited reusability, less standalone performance, and two engine development
COST	GOOD : Share engine with USAF	GOOD	FAIR : Lowest DDT&E, but large size equals high unit cost
OTHER	No environmental impacts	Upper atmosphere contamination	Upper atmosphere contamination plus pressure vessel hazard

SECTION 4
PUMP FED CONCEPT - LOX/RP-1

One of the three selected pump-fed LRB concepts is an expendable engine using LO2/RP1 propellants and a gas-generator (GG) cycle. The technology and reliability of the LO2/RP1 GG engine system have been demonstrated through previous and current vehicles, such as the Saturn V with F-1 engines, the Atlas with MA-5 engines, and the Delta with RS-27 engines. For the current STS application, the LRB with an expendable LO2/RP1 GG engine concept offers several major advantages: (1) low development and schedule risks, (2) low risk of STS integration due to the smaller LRB size, (3) high operational flexibility and low hardware/software complexity on both ground and vehicle systems, and (4) low overall system cost.

4.1 STRUCTURES AND MECHANISMS

4.1.1 VEHICLE CONFIGURATION. The LO2-RP-1 Pump Fed Liquid Rocket Booster is shown in Figure 4.1.1-1. It's total length is 146.2 feet. The LO2 tank total length is 68.1 feet having a cylindrical section 57.5 feet long capped by two elliptical bulkheads. The LO2 tank is attached to the forward adapter as its forward end and is connected to an intertank adapter aft. The RP-1 tank total length is 40.0 feet having a cylindrical section 29.4 feet long capped by two elliptical bulkheads. The RP-1 tank is attached to the intertank adapter at its forward end and is connected to the aft skirt at its other end. Both tanks are 14.6 feet in diameter.

The intertank adapter is a total length of 13.4 feet. This length is established by the clearance required between the two propellent tanks bulkhead domes to allow packaging of the LO2 propellant feed line.

The aft skirt is 17.5 feet long having a forward diameter sized to interface with the RP-1 tank. The aft diameter is 22.6 feet sized to protect a gimbaled engine from the aerodynamic loads.

The nose cap has a fineness ratio of 1.33 which is similar to that of the solid rocket motor. Since it interfaces with the forward adapter it's length is geometry dependent. There are no packaging constraints since no recovery system packaging is required. The nose cone is 19.4 feet long. The exit diameter of the engine nozzles protrude 2.8 feet below the aft skirt. This provides the same reference station for both the solid rocket motor and Liquid Rocket Booster (LRB).

The locations for the External Tank (ET) to LRB attachments are at the same stations as that of the ET to SRM. The forward attachment is at the LRBs forward adapter. This adapter has a total length of 6.2 feet.

Tanks. Both the RP-1 and the LO2 tanks are similar in structural design. A representative tank design for these propellant tanks is shown in Figure 4.1.1-2. The tanks are made entirely of 2219 Aluminum Alloy and use Variable Polarity Plasma Arc (VPPA) welding to join all major structural components.

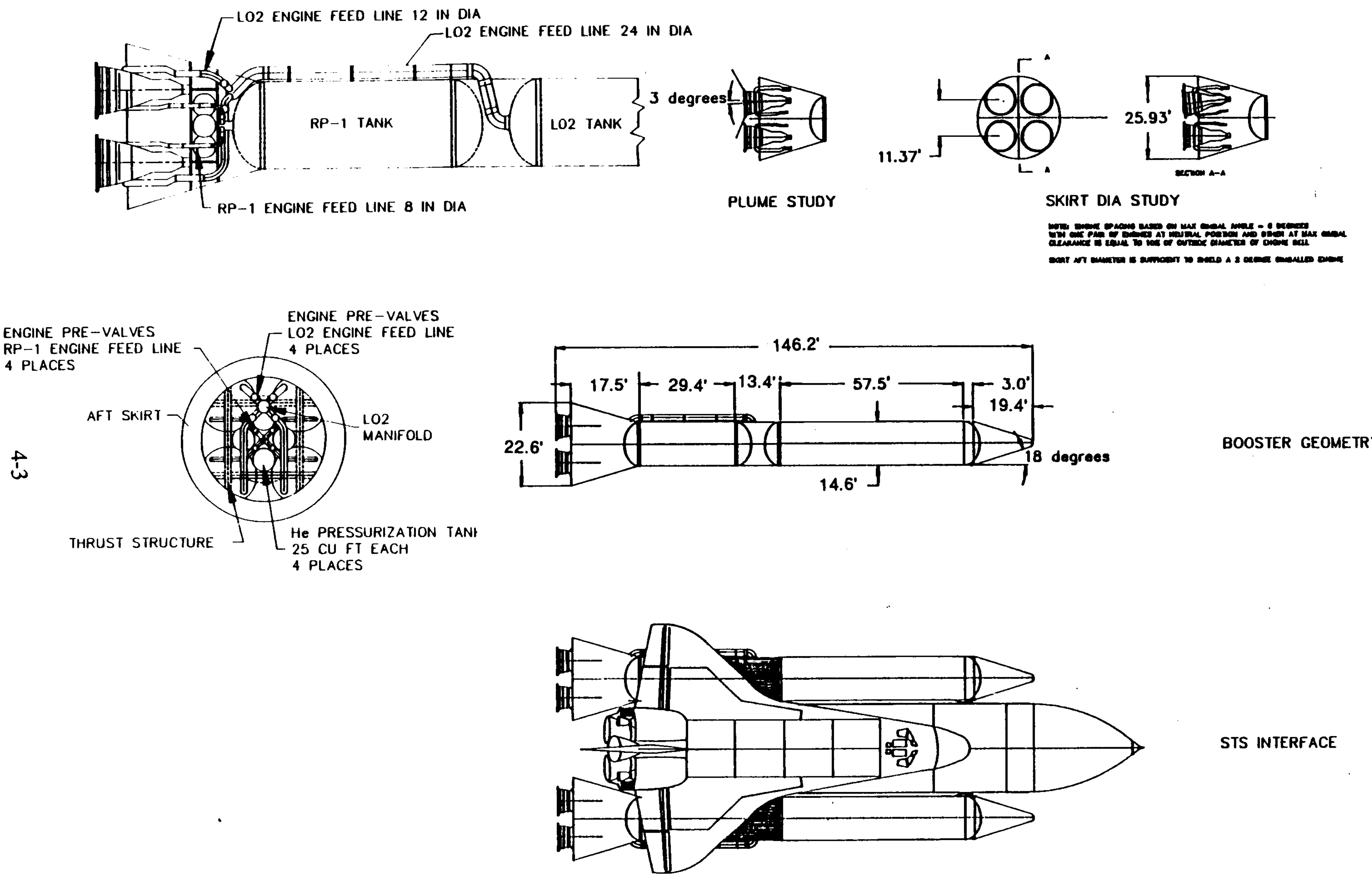


Figure 4.1.1-1 LO2/RP-1 Liquid Rocket Booster

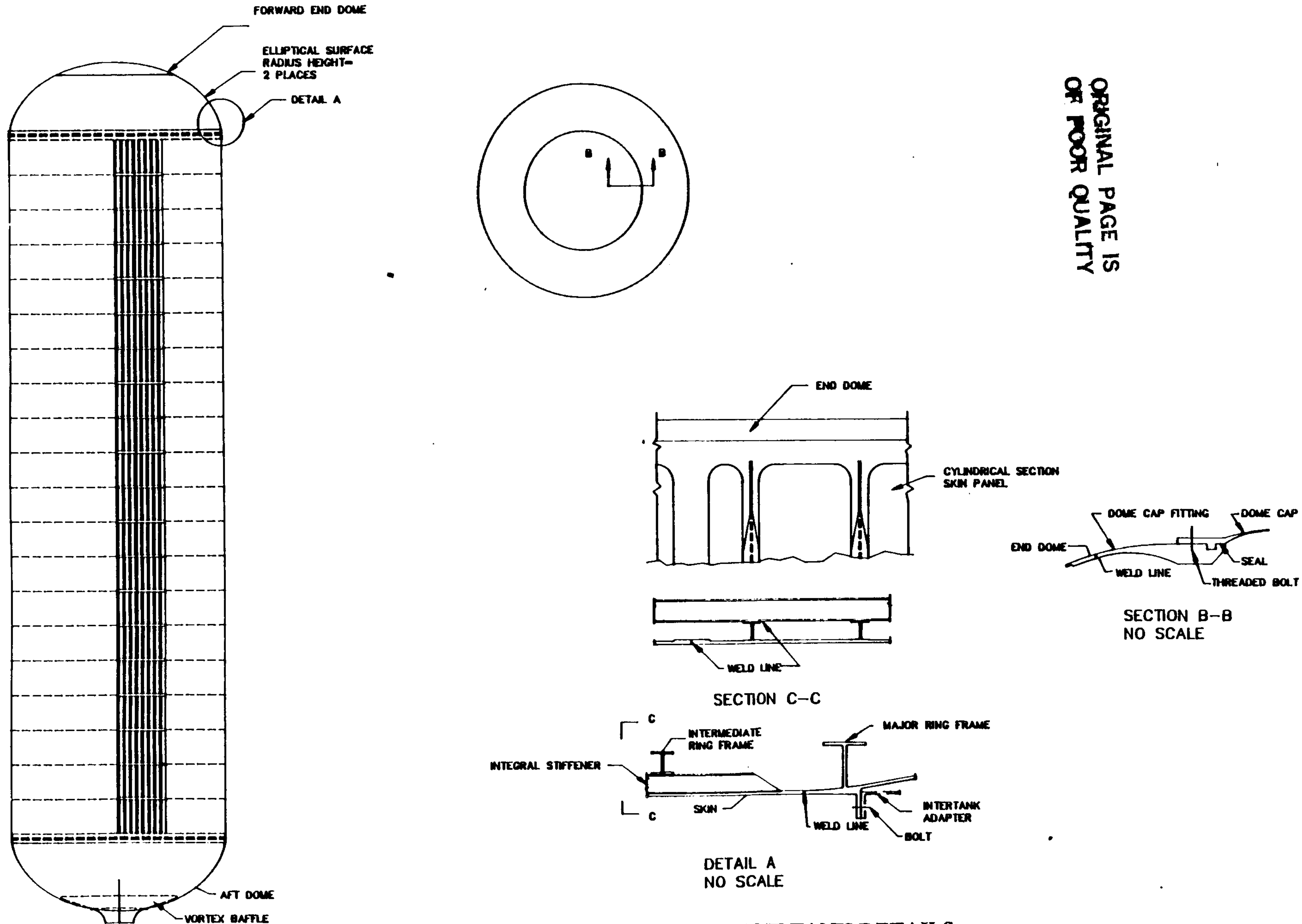


FIGURE 4.1.1.2 PROPELLANT TANK DETAILS

The tank cylindrical section skin panels are 2219-T8510 extrusions, with integral T (tee) stringers. Extruded panels are machined to the required thickness. Locally at the splices the skin remains thicker for welding. The T stringer flanges provide a mating surface for attaching the ring frames with mechanical fasteners. The ring frames are equally spaced along the cylindrical section to provide stability to the skin/stringer.

The end domes are 2219-T62 Aluminum alloy and are a single piece spun form part. Access provisions, propellant feed lines, pressurization lines, and vent lines will be located in the crest (or valley) area of the domes. The domes are machined to reduce weight, but are thicker in the edge weld land and for the above provisions.

A 2219 Aluminum alloy roll ring forging is machined to a Y shape to join the dome to the cylindrical tank section, and provides a member to attach the intertank or forward adaptors, or the aft skirt. An integral internal upstanding flange is machined into the forging. This provides a flange to attach a ring frame, to react loads normal to the tank skin. Mechanical fasteners attach the ring frame to the flange, but no fasteners are in or through the tank skin.

Skin/stringer/frame construction provides a fail safe tank structure. Should a failure occur in the skin due to internal tank pressure, the tank will leak and not explode. A tank structure with redundant load paths provide this safety, and it is not provided with a monocoque structure.

Intertank Adapter. The internal adaptor skin is 2024 Aluminum alloy corrugation, with internal ring frames spaced at 26.8 inches. The construction layout is shown in Figure 4.1.1-3. The ring frames are 2024-T3510 Aluminum alloy extruded I sections and are roll formed to the adaptor diameter. The frames are riveted to the corrugated skin at the corrugation valleys. The corrugations run the length of the adaptor and the longitudinal joints are lap spliced and riveted to the adjacent corrugated sheets. A U-shaped extruded 2024-T3510 Aluminum alloy section is used at the upper and lower ends of the adaptor for terminating the corrugation. The U-shaped ring frame slides over the ends of the corrugation and mechanical fasteners attach it to the hill and valley of the corrugation. the "U" ring frames are the splice joint to the LO2 and RP-1 tank adaptors.

The centroid of the corrugation is lined up with the centroid of the tank adaptors, so there is no eccentricity or moment introduced into the corrugation or adaptors.

NOTES:
1. SEVEN PANELS PER ADAPTER
2. ALLUMINUM ALLOY 2090-T8E41

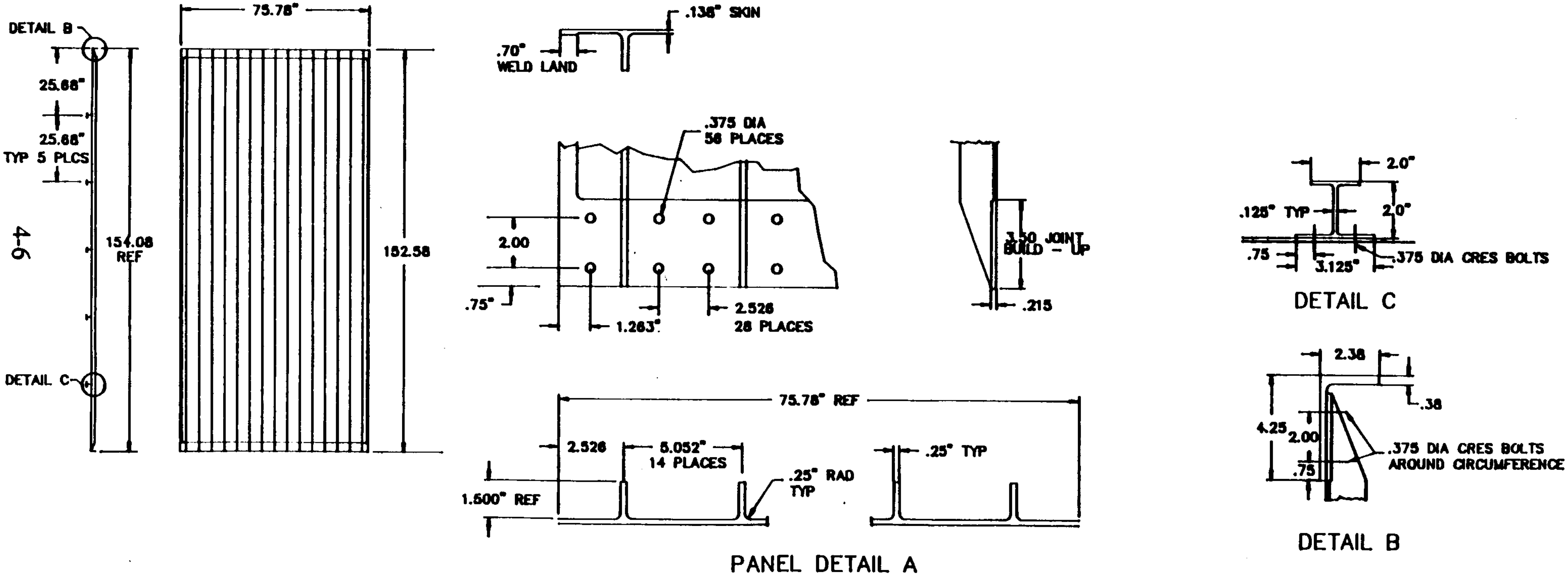


Figure 4.1.1-3 Intertank Adapter Details

Nose Cone. The nose cone is a conic shape mechanical assembly consisting of a nose cap and a stiffened truncated cone. The nose cone is fabricated from 2024-T3 aluminum. The inverted tee longerons and "I" shaped ring frames are mechanically fastened to the skin as shown in Figure 4.1.1-4. The nose cone is mechanically fastened to the forward adapter at a major ring frame.

Aft Skirt. The aft skirt, shown in Figure 4.1.1-5 consists of a semi-monocoque fustrum of a cone section and the engine thrust structure. The skin is 0.437 inch thick 2024-T85 Aluminum alloy with internal "I" section extruded 2024-T81 Aluminum alloy ring frames. The engine thrust structure consists of built up "I" beams arranged in a box pattern. Between engine thrust attach fittings the beam is a truss, and outboard to the skirt skin, web and stiffeners are used. The beams load the skin longerons for distribution into the skirt skin. There are eight (8) longerons, one at each beam end and they run the length of the skirt. The four (4) hold down fittings are located at the aft end of the skirt and extend the length of the skirt on the outside of the skin.

4.1.2 SEPARATION SYSTEM. Efforts to define the separation system were conducted during the initial LRB study phase. During the follow-on extension, separation analyses were not updated to reflect resizing of the LOX/RP-1 pump-fed booster. Thus, the results which follow require update and are presented primarily to show trends and typical designs.

The separation system for the LOX/RP-1 pump-fed booster consists of:

I. Booster Separation Motors (BSMs) With NASA Standard Initiators
II. Four Separation Bolts Per LRB with Pyrotechnics
III. BSM Mounting Structure, Insulation, And
IV. Electronics/Software To Initiate Separation

The separation system was designed to meet the basic requirements for providing separation without damage to, or recontact of, separating elements. The booster separation system provides fail-safe capability. In addition, automatic separation inhibit with manual override capability is provided.

The LOX/RP-1 pump-fed booster separation system has been initially sized for nominal ascent staging. As the LOX/RP-1 pump-fed booster design matures, RTLS abort coverage

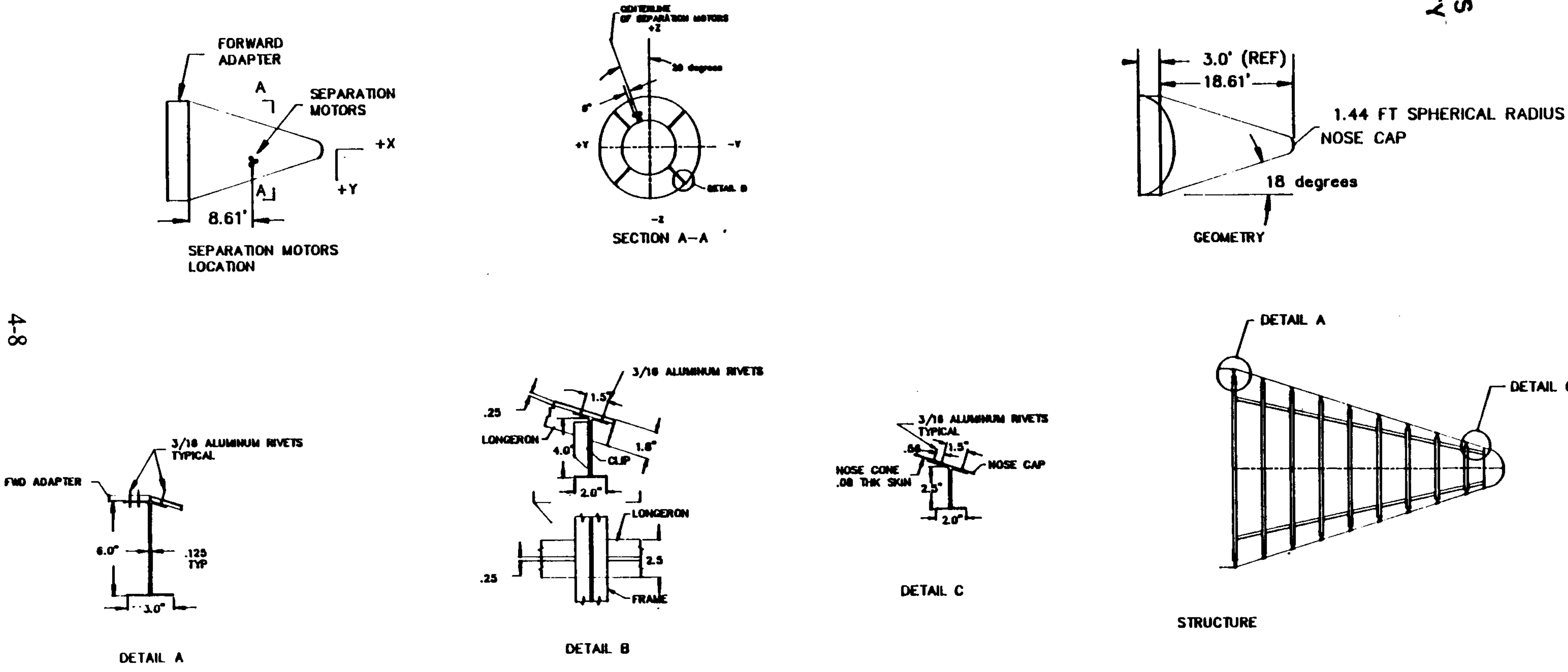


Figure 4.1.1-4 Nose Cone Details

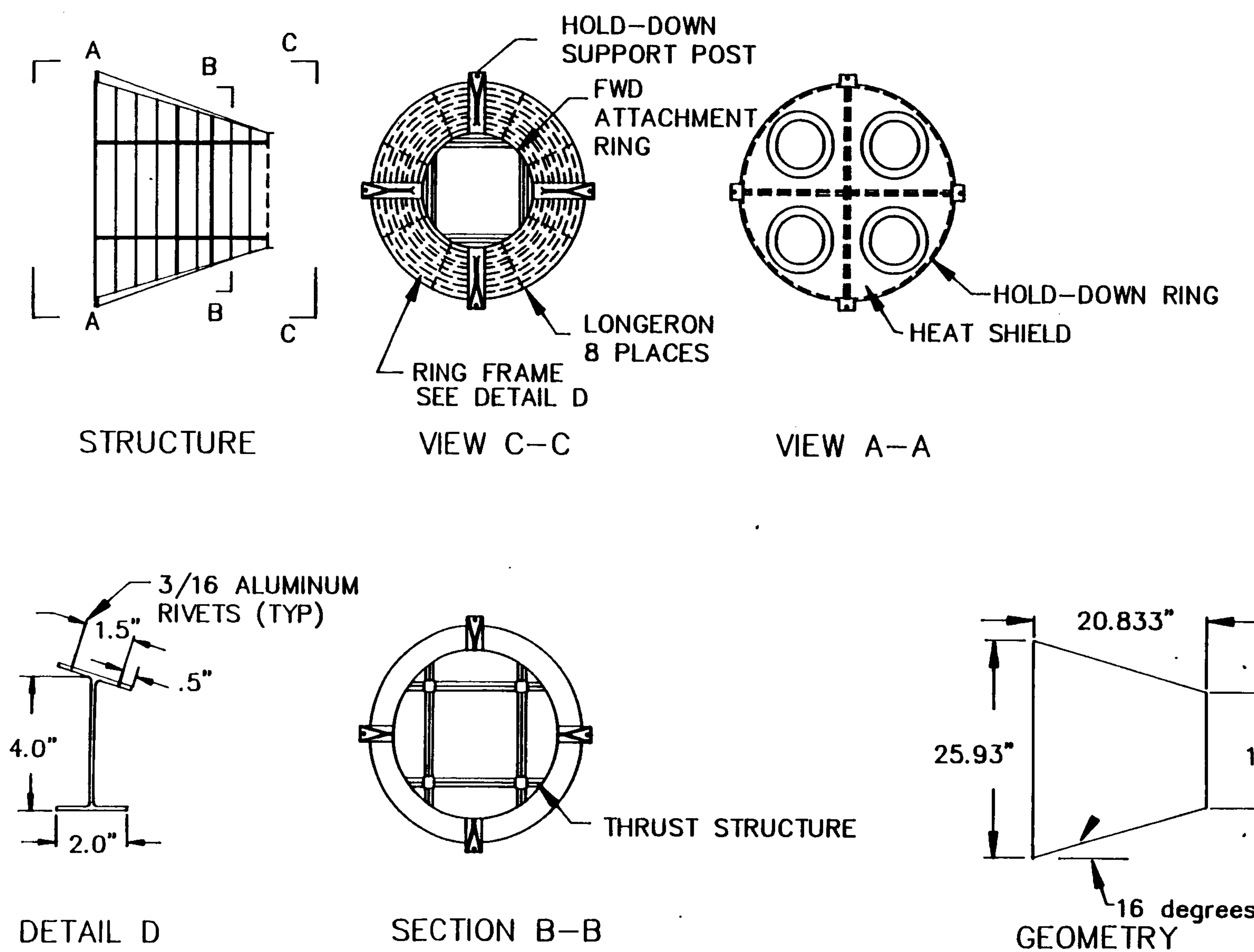


Figure 4.1.1-5 Aft Skirt Details

capabilities will be more thoroughly examined; if results show that it is possible to conduct a RTLS abort prior to nominal staging, simulations will be conducted to determine if the BSM quantity used for nominal ascent staging is sufficient for RTLS abort needs. If not, the number of BSMs will be increased accordingly.

The basic separation sequence for the LOX/RP-1 pump-fed booster is depicted in Figure 3.12.3-1 of section 3.12.3. The booster separation sequence is initiated on "Low Liquid Levels" sensed in the booster propellant tanks. To perform separation, the LRBs engines are nulled, and shutdown. Then separation pyrotechnics, and the BSMs, are armed and fired. Section 3.12.3, contains further details on the booster separation sequence.

In accordance with results from the Separation System Trade Study 1.16, all booster configurations shall use National Space Transportation System (NSTS) Booster Separation Motors (BSMs). These motors are manufactured by United Technologies, Chemical Systems Division. The motors provide an average thrust of 18,500 lbs, and have a total minimum impulse of 15,000 Lb-sec. The length of a BSM is 31 inches, and the case diameter is 12.88 inches. Each motor is fired by a NASA standard Initiator (NSI). An additional NSI for each motor will be provided for redundancy. The motors weigh 162 lbs each, but when attendant supporting structure, initiators and insulation are included, the system weight per motor is on the order of 200 lbs.

The conditions for nominal ascent staging of the LOX/RP-1 pump-fed LRB configuration are listed below:

Mission Elapsed Time	= 125.6 Seconds
Altitude	= 138,000 Ft
Mach	= 4.84
Dynamic Pressure	= 75 PSF
Inert Weight	= 126,000 Lbs

For all downselected boosters, separation simulations were performed using the NASA program SVDS (which was used to model/verify SRB staging). The LRB design case separation conditions include: body rates of 5 deg/sec pitch, 2 deg/sec yaw, and 2 deg/sec roll; alpha = 10 degrees; and beta = 10 degrees. For all downselected boosters, element aerodynamic coefficients were calculated, and mutual interference effects were considered.

The LRBs were assumed to be rigid bodies; further analyses is required to determine propellant slosh and aeroelastic effects upon booster separation.

For the LOX/RP-1 pump-fed booster, computer simulation results indicate that 3 BSMs forward, and 4 BSMs aft are required for nominal ascent (design case) staging conditions. The total system weight is on the order of 1,400 Lbs.

The 7 BSMs are distributed with 3 packaged in the nose cone and 4 placed on the aft skirt. The current SRB BSM orientation is maintained. The forward motors are aligned in a plane rotated 20 degrees away from the positive booster Z_B axis, and their thrust is directed 40 degrees forward. The aft separation motors are oriented in a similar manner, except their thrust is directed 40 degrees aft.

Separation plots for nominal ascent (design case) staging (Figures 4.1.2-1 through 4.1.2-3) indicate a clean separation. Because the inert weight of the LOX/RP-1 pump-fed booster is about 50,000 lbs less than a SRB at separation, and because the center of gravity is further aft, the booster nose pitches away from the Orbiter rapidly. For this reason it was possible to use only 3 BSMs in the LOX/RP-1 booster nose cone, whereas the SRBs rely on 4.

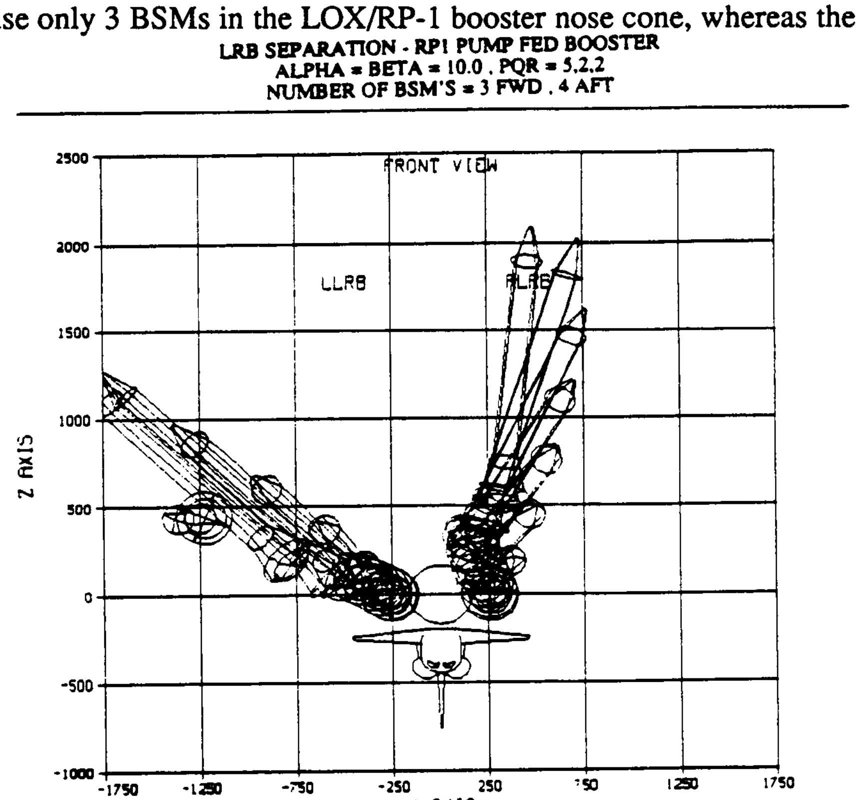


Figure 4.1.2-1. LOX/RP-1 Pump-fed Nominal Ascent (Design Case) Separation, Front View

View

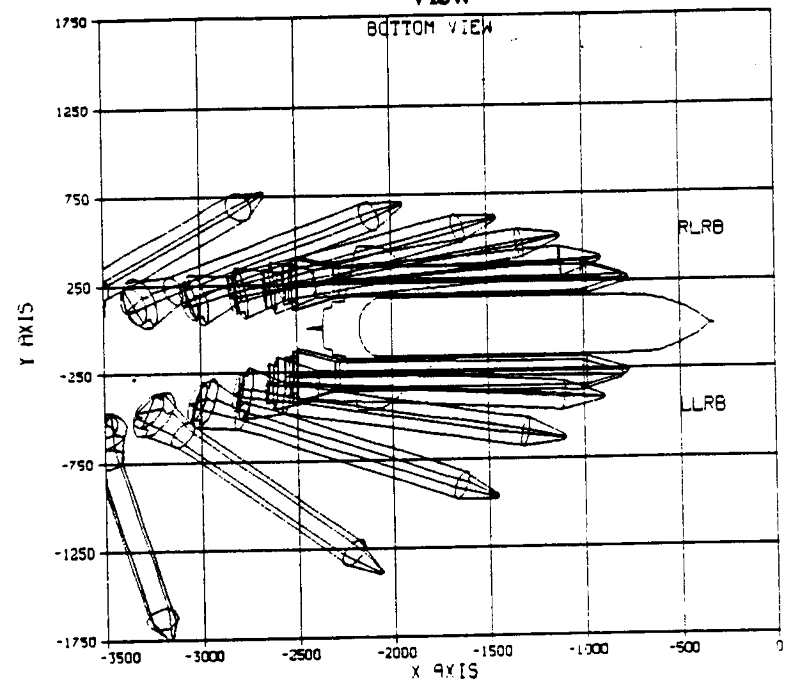


Figure 4.1.2-2. LOX/RP-1 Pump-fed Nominal Ascent (Design Case) Separation, Bottom View

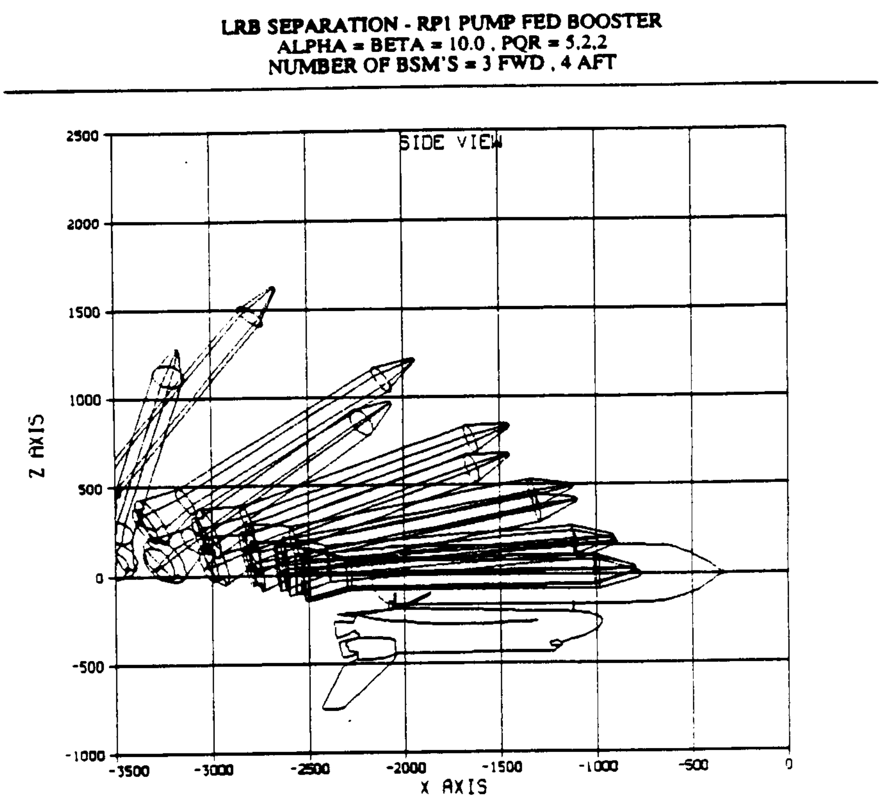


Figure 4.1.2-3. LOX/RP-1 Pump-fed Nominal Ascent (Design Case) Separation, Side View

4.1.3 <u>THERMAL PROTECTION AND INSULATION</u>. The LOX/RP-1 thermal protection and insulation requirements differ from the SRBs primarily because of insulation considerations for the cryogenic oxidizer used. The major thermal protection and insulation requirements are:

1) Protect structure exposed to severe aerodynamic heating by maintaining structural temperatures below acceptable maximums.
2) Insulate the liquid oxygen (LOX) supply in order to reduce losses due to boiloff, and to maintain good quality oxidizer to the engines.
3) Prevent formation of ice on the LOX tank (or ET) which might impact the Orbiter if it should come loose during flight.
4) Maintain thermal limits for booster subsystems (such as avionics)

4.1.3.1 <u>Aerodynamic Heating</u>. Aerodynamic heating during ascent was examined by comparing the LOX/RP-1 trajectory to a SRB trajectory for which heating rates have been determined. Because aerodynamic heating at the nose cone tip is a function of the square of atmospheric density and velocity cubed, it is possible to obtain relative nose tip heating values. Figure 4.1.3.1-1 shows a comparison between the LOX/RP-1 LRB trajectory and the aeroheating ascent design

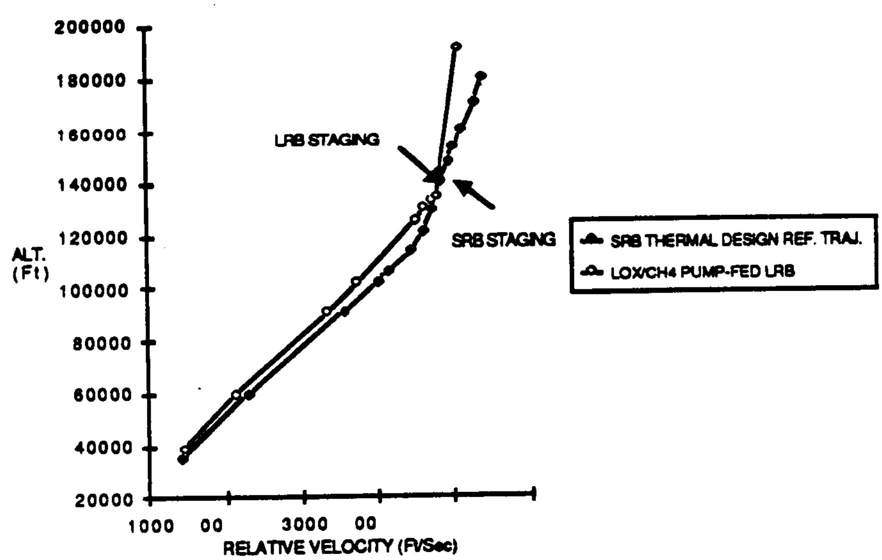


Figure 4.1.3.1-1. LOX/RP-1 Pump-fed booster And SRB Altitude Vs. Velocity Profiles Comparison.

trajectory used for the Space Shuttle elements. Both trajectories are very similar until approximately 2000 ft/sec velocity when the Shuttle Design Trajectory begins to deviate. At the time of peak heating, approximately 4000 ft/sec, the LOX/RP-1 trajectory is nearly 11,000 feet higher in altitude. This corresponds to a reduction in atmospheric density of 60-65%, which in turn produces lower aerodynamic heating than a SRB at the LRB nose cone tip; this lower nose cone tip heating trend is expected to apply for the rest of the booster.

For LRB areas subjected to vehicle shock interferences, the local heat transfer rates are strongly influenced by shock strength, local flow condition (laminar or turbulent) and boundary layer thickness; therefore heating to such areas is difficult to evaluate.

4.1.3.2 Thermal Protection System Definition. At present, the Thermal Protection System (TPS) design for the LOX/RP-1 pump-fed booster configuration is a combination of the TPS approaches used for the ET and SRB. The primary LRB TPS materials to be used are: ET Spray-On Foam Insulation CPR-488 (SOFI); ET Super Light Ablator (SLA-561); ET urethane foam BX-250; and SRB sprayable ablator MSA-1. However, formal trade studies are required to select the optimum LRB TPS materials. Figure 4.1.3.2-1 presents the booster TPS layout.

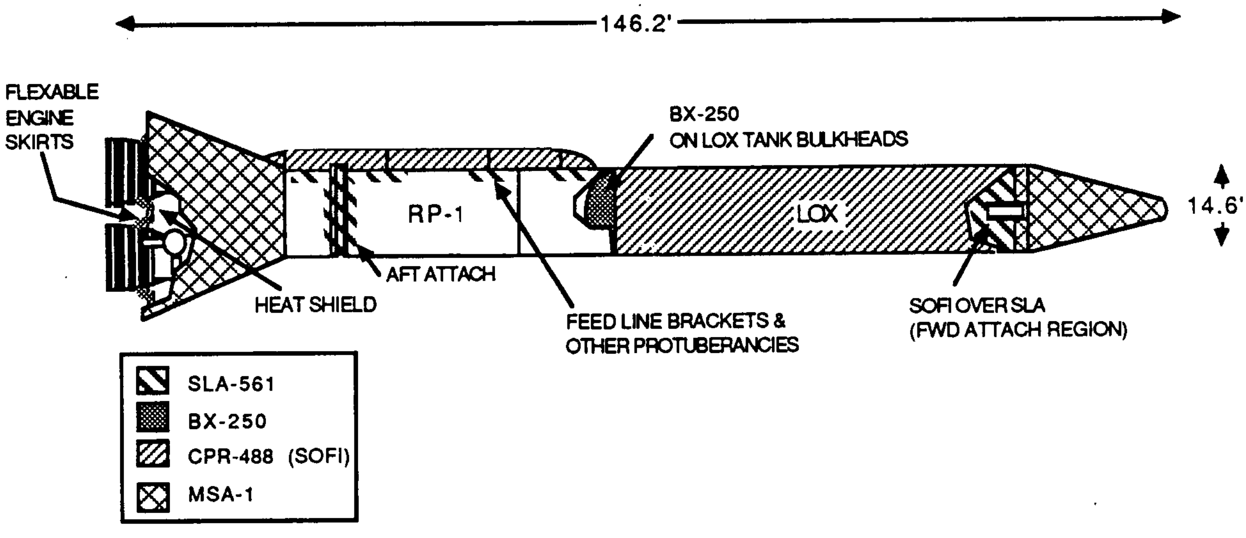


Figure 4.1.3.2-1. LOX/RP-1 Pump-fed Booster TPS Layout

Prelaunch requirements basically define the foam insulation thicknesses for the booster cryogenic LOX tank. The tank barrel sections will be coated with nominally 1" of SOFI (in a manner similar to the ET LOX tank). SOFI will serve to minimize ice formation following on-pad loading of propellants, and to reduce boiloff. The LOX tank end domes will be covered with urethane foam BX-250 after manufacture. This urethane foam will be used due to its more liberal application constraints.

Ablative material will cover high heating areas of the LRB, such as the nose cone, aft skirt, around vehicle interface attachments, feedline brackets, and other protuberancies. Ablative material thicknesses will be adjusted to maintain avionics environments below 150° F, and the aluminum structure below 300° F.

The ablator to be used for the nose cone and aft skirt is Marshall Sprayable Ablator (MSA-1). MSA-1 has a heating rate capability of 10-15 Btu/Ft^2-sec and furnishes adequate thermal protection for the SRB nose cap, frustum, forward skirt, and a significant portion of the systems tunnel. MSA-1 is considered appropriate for the LOX/RP-1 pump-fed booster's nose cone acreage because its exposure to aeroheating is not expected to be any more severe than the SRBs (see section 4.1.3.1).

Unlike the nose cone, the aft skirt on the SRB is covered with a cork ablator. Cork is used because of high heating, and large airstream shear forces, which arise during SRB reentry after staging (refer to SRB Thermal Design Data Book, SE-019-068-ZE). However, all LRB configurations are expendable, and because reentry heating will not be used for design criteria, MSA-1 will suffice for the aft skirt.

SLA-561 (which is more expensive than MSA-1) has a heat rate limit of 30 Btu/Ft^2-sec and will be used more sparingly underneath SOFI on the LOX tank at the forward attachment, and on other exposed high heating rate (>15 Btu/Ft^2-sec) protuberancies.

To protect booster engines and internal aft skirt components from LRB and SSME plume heating during ascent, protective flexible skirts (around engine structure for gimbal motion) and a rigid firewall system will be provided.

4.2 MAIN PROPULSION SYSTEMS

The main booster propulsion system consists of two LO2/RP1 LRBs which provide the velocity increment needed to meet the payload requirement and the ATO goal with one engine out in conjunction with the fixed mainstage propulsion. Each booster main propulsion system consists of the following major subsystems: 4 engines (engines and controllers), feed and pressurization systems (tanks, propellant management systems, pressurization systems), and the thrust vector controller (controller and electro-mechanical actuators). The interaction of various systems is schematically depicted in Figure 4.2.1, and described in the following subsections.

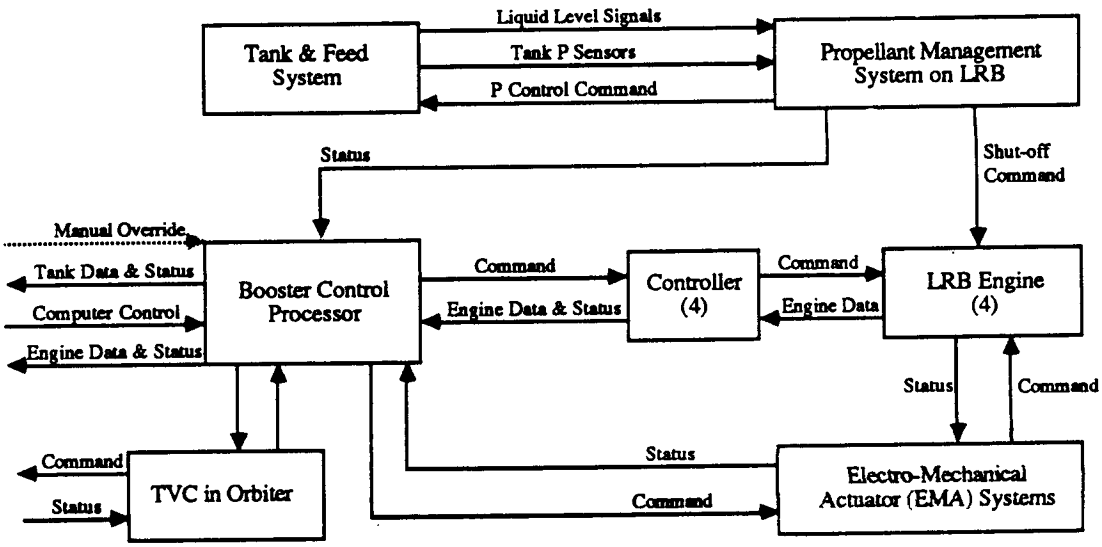


Figure 4.2-1 Interaction of Major Propulsion Subsystems

4.2.1 <u>ENGINE SYSTEMS</u>. For the pump-fed LRB concept employing a conventional engine cycle, we have selected the gas-generator engine over the stage combustion cycle because of its low cost and simplicity as shown through past and current experience and detailed engine studies. We have developed a conceptual definition of a new LO2/RP1 GG cycle engine that is tailored to the LRB constraints and requirements. Rocketdyne has provided engine performance and cost data and assisted in the engine system selection and optimization.

Boost pumps vs. NPSP Requirements. Baseline engine inlet pressure is 65 psia for the oxidizer and 45 psia for the fuel. Propellant tanks designed from structural load and manufacturing considerations can accommodate pressures corresponding to these engine inlet pressure requirements without added weight penalty. Boost pumps are not necessary at these pressure levels, and their elimination results in a reduction of engine weight and fewer components. Impacts of LO2 pump inlet pressure on engine weight and performance are depicted in Figure 4.2.1.1-1, in which the design point of 65 psia without a boost pump is most favorable. Similar trends are observed for RP1, in which 45 psia without boost pump is the optimum design point.

4.2.1.1 Engine Features Selection. Main engine features are shown in Table 4.2.1.1-1.

Table 4.2.1.1-1 Main Features of LO2/RP1 Pump-Fed Engine for LRB

Cycle	Gas Generator
Boost pumps	None
Throttling capability	Continuous, 75 to 110%
Control System	Closed loop
Turbine exhaust disposal	Injected into nozzle
Turbine start	Solid propellant gas generator
Inlet ducts	Scissors type bellows ducts
Ignition	Hypergolic Slug
Nozzle	80% Bell nozzle
Gimbal	Head end gimbal; ±6° square pattern
Delivered life	5 starts
Burn duration	150 sec.
Engine inlet Pressure requirement	LO2 65 psia
	RP1 45 psia

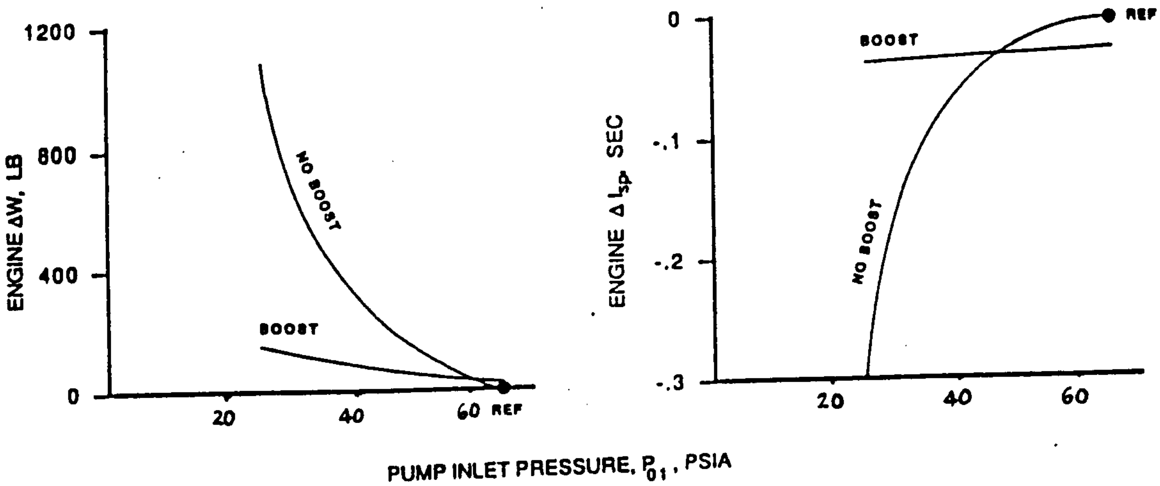


Figure 4.2.1.1-1 LO2 Pump Inlet Pressure Effects in LO2/RP1 GG Engine

Selection of Engine Parameters. The chamber pressure for the LO2/RP1 GG cycle is recommended not to exceed 1600 psia because of a potential coking problem in thrust chamber cooling passages associated with RP1 at high heat flux conditions. The maximum chamber pressure selected for our engine at EPL is 1400 psia to allow some development risk margin. The nozzle area ratio (AR) of 16.5:1, is optimized based on life cycle cost of the vehicle. The resultant configuration with nozzle exit diameter based on these design considerations can be accommodated by the current flame trench, without any modifications. Engine mixture ratio (MR) and thrust chamber MR are performance optimized. Sensitivity analysis shows that engine MR variation has a negligible effect on booster weight and size.

Control. A closed loop control system is required for the pump-fed LRB engines to accommodate throttling requirements. Engine control is discussed in detail in Section 4.2.1.5.

Gas-Generator Exhaust. The GG gases exiting from the turbines are discharged symmetrically into the nozzle, as depicted in Figure 4.2.1.1-2. This concept was selected mainly for packaging purposes. A secondary benefit is the reduction of RP1 cooling ΔP because the GG gases cool part of the nozzle.

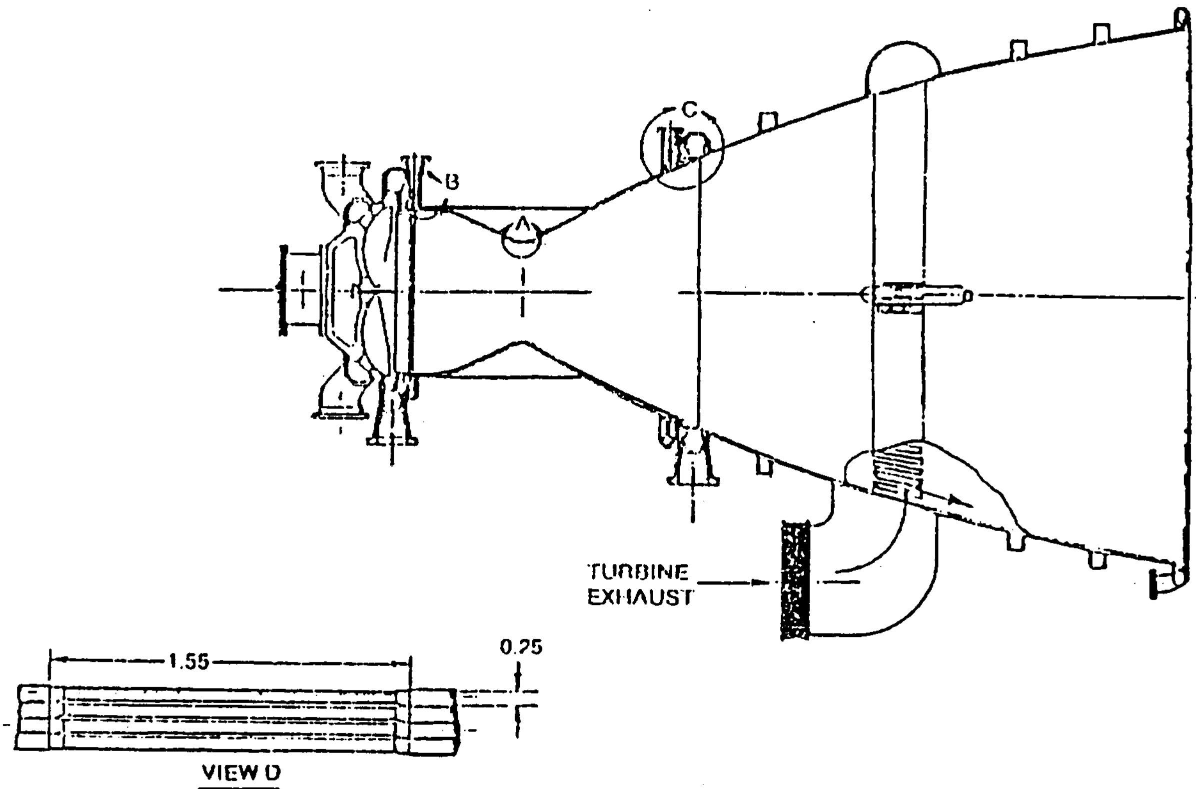


Figure 4.2.1.1-2 GG/Turbine Exhaust Injected through Nozzle Wall as Sonic Flow

The selection of the area ratio at which to inject the GG gases is based on minimum acceptable AR to maintain required turbine pressure ratio for high performance while providing minimum engine weight and packaging dimensions.

<u>Ignition</u>. Main injector propellant ignition employs multi-element 85/15 injecting TEA/TEB hypergolic propellants The hypergols are injected and introduced to the injector by an RP1 stream at the start of ignition.

4.2.1.2 <u>Selected Engine Characteristics</u>. The engine configuration selection was based on previous engine studies, experience, and trades and analyses performed during this study for STS application. Engine performance and a pressure balance were generated for the selected configuration. The resulting parameters were used to establish the pertinent combustion chamber, injector, nozzle, and turbopump characteristics leading to the current configuration and physical design. Table 4.2.1.2-1. summarizes performance data and characteristics of the LO2/RP1 GG cycle engine optimized for LRB trajectory and configuration requirements. The engine is expendable after one flight, and its delivered life of five starts allows a margin for testing prior to flight without increase in cost.

Table 4.2.1.2-1 Performance & Characteristics of LO2/RP1 GG Engine for LRB

Parameter	EPL (110 %)	NPL (100%)	Minimum Thrust
Thrust vac (k lbs)	692.9	629.9	472.4
Thrust SL (k lbs)	627.9	564.9	407.4
Chamber Pressure (psia)	1400	1273	955
C* Efficiency	96%		
Nozzle Expansion Ratio	16.5		
Mixture Ratio, Engine/TC	2.53/2.8		
Isp vac (sec)	309.6	310.1	311.4
Isp SL (sec)	280.6	278.1	268.6
TC Oxidizer flow (lb/sec)	1576		
TC Fuel flow (lb/sec)	558.4		
GG Oxidizer flow (lb/sec)	22.1		
GG Fuel flow (lb/sec)	73.1		
Engine Length (inch)	130.3		
Nozzle Exit Diameter (inch)	75.0		
Engine Dry Weight (lbs)	6216		

The turbo-pump characteristics of the LO2/RP1 engine are shown in Table 4.2.1.2-2.

Table 4.2.1.2-2 Turbo-pump Characteristics of LO2/RP1 Engine @ NPL

Component	LO2	RP1
Turbine		
Stages	2	2
Efficiency	.7655	.7689
Horsepower	11215	10717
Tip Speed (ft/sec)	746	823
Inlet Temperature (°R)	1464	1600
Outlet Temperature (°R)	1316	1464
Inlet pressure (psia)	352	1273
Outlet pressure (psia)	63	352
Pumps		
Stages	1	1
Efficiency	.819	.8046
Inlet pressure (psia)	65.0	45
Outlet pressure (psia)	1782	2922
Inducer		
Tip diameter (in)	10.7	6.8
Tip speed (ft/sec)	376	522
Impeller		
Tip diameter (in)	14.0	9.7
Tip speed (ft/sec)	492.0	743
Stage specific speed (RPM*GPM**0.5/Ft**0.75)	1710.0	1454

4.2.1.3 Engine Schematic and Operation.

Engine Cycle Description. A flow schematic of the LO2/RP1 GG cycle engine is shown in Figure 4.2.1.3-1.

Propellants from the main tanks are introduced into the pumps through the inlet scissors-type flex ducts. The high pressure fuel discharge from the main fuel pump is used as coolant for the thrust chamber and nozzle before being injected into the main combustion chamber (MCC). The oxidizer pump discharge is fed directly to the MCC. A small percentage of the oxidizer and fuel flows from the pumps are drawn-off and burned in the gas-generator (GG) to provide the high-pressure gaseous products which drive the fuel and the oxidizer turbines, and each turbine in turn drives its own pump. Mixture ratio for the GG is kept very low, fuel-rich, to keep its exhaust temperature within limits of the turbine blades. The exhaust from the oxidizer turbine is then used as heat source for the pressurization heat exchangers prior to being dumped into the nozzle.

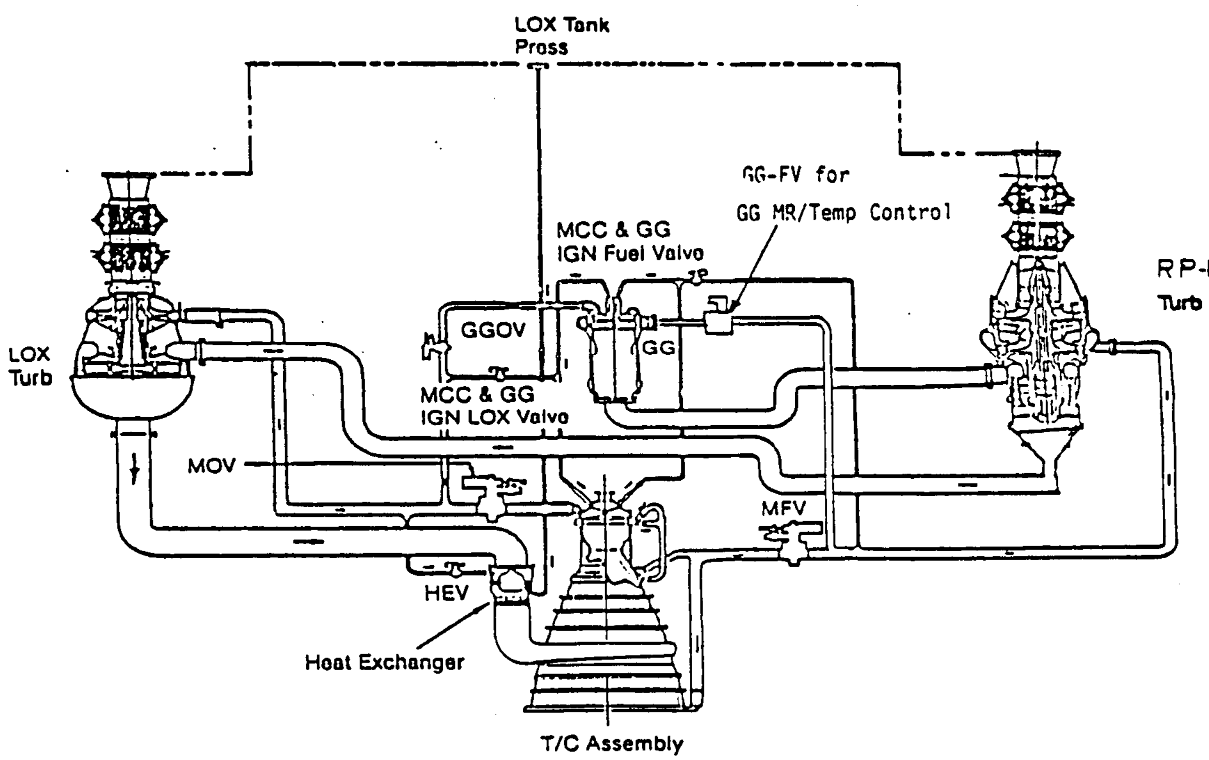


Figure 4.2.1.3-1 Flow Schematic of LO2/RP1 Pump-Fed Engine for LRB

A design modification, which can potentially reduce the cost of LO2/RP1 pump fed engine compared to above is single shaft instead of dual shaft turbo-pump. This alternative has been recently presented in STME/STBE Dec. 1988, quarterly meeting. Cost reductions are primarily due to two reasons: (1) simplification of turbopump, (2) possibility of simple open loop control because during throttling of the engines, two propellant flow tend to track since their pumps are always running at an identical speed. This engine has not been incorporated as the baseline because of lack of time.

Engine Start and Shutdown. The engine start and shutdown sequence for the LRB LO2/RP1 GG engine are similar to that of previous LO2/RP1 engines. A typical start and shutdown sequence of events is shown in Table 4.2.1.3-2. The engine start propellant consumption noted is from engine start signal to nominal operating level, and does not require any engine prechill.

All previous LO2/RP1 engines used a turbine spin-start with the exception of the F-1 which used a tank head start. Both types of start-up were reviewed, and turbine spin-start using a solid propellant gas generator (SPGG) was selected.

A SPGG turbine spin system substantially reduces run-to-run variations. The spin power is relatively repeatable and is large enough that variations in turbopump drag will have minor effects. A start with a spin system also requires less start transient development time.

The basic start sequence begins with opening of the main oxidizer valve (MOV). At the same time an igniter fuel valve (IFV) is opened, allowing RP1 to flow from tank to the hypergol cartridge and break its diaphragm. The hypergol then follows the RP1 flow through igniter fuel line to MCC and promotes ignition in the main chamber. Ignition is confirmed by burn through of a wire stretched across the chamber nozzle exit.

Once ignition has been confirmed, the helium is introduced to provide power for turbine spinning. The main fuel valve (MFV) is then opened to allow the fuel to fill the main fuel injector within one second, resulting in main propellant ignition. The GG propellant valves are then signalled to open, and combustion is generated in the GG which further increases the pumps speed. The start time from signal to spin to NPL is in the order of 1.2 seconds.

Table 4.2.1.3-2 Engine Start and Shutdown Summary for LRB LO2/RP1 GG Engine

Start time (sec)	Event
0.0	Open Main Oxidizer Valve
0.2	Open Igniter Fuel Valve
1.0	Detect Main Chamber Ignition
1.2	1. Signal Spin System Start
	2. Ramp Open Main Fuel Valve
2.1	Fuel Prime System to Main Chamber Generating Main Chamber Prime and Engine Boost Stage
2.2	Open GG Valves
2.4	Close Spin System Valve
2.6	Engine Reaches Full Thrust

Shutdown time (sec)	Event
0.0	Close GG Valves (0.1 to 0.2 second)
	Ramp Main Oxidizer Valve Closed (assume 0.5 sec travel)
0.1	Ramp Main Fuel Valve Closed (assume 0.5 sec travel)

Estimated Propellant Consumption During Transients

	Start	Shutdown
LO2	2100 lb	500 lb
RP1	470 lb	300 lb

For engine shut-down, the MOV, GG valves and MFV start closing at engine cut-off signal. This results in chamber pressure ramping down to zero within 0.5 - 1.5 seconds.

4.2.1.4 Engine Design Discussion & Preliminary Drawing. The engine is designed for a chamber pressure of 1400 psia and thrust chamber mixture ratio of 2.8. The engine employs a gas-generator cycle that produces 1800°R gases to drive the RP1 and the LO2 turbopumps, which are in series. The series turbines concept was selected to minimize secondary flow performance losses (GG gases). The turbine exhausts (GG gases) are then dumped into the thrust chamber nozzle.

The nozzle has an exit-to-throat area ratio of 16.5, which represents an optimum nozzle from a overall vehicle cost approach; see section 3.7.3. The nozzle contour is an 80% bell with a 4-degree exit wall angle.

Thrust Chamber Cooling Selection. The thrust chamber consists of an injector, a main combustion chamber (MCC), and a nozzle. The RP1 fuel is used to cool the surfaces of these components which are exposed to a 6500°R combustion gas environment. To adequately cool the thrust chamber components and maintain a minimum component weight, each component will use appropriate fabrication techniques and materials. Lightweight tubular construction using stainless steel for the MCC and nozzle is satisfactory only for the low heat flux nozzle. However, with high heat flux in MCC at 1400 psia, a copper base alloy (NARloy-Z) milled channel configuration, typical of the SSME, will be required.

At the MCC-to-nozzle attachment, located at an area ratio (AR) of 5:1, 50% of the RP1 coolant inlet flow is used to cool the MCC and the other 50% to cool the nozzle. This 50/50 flow split at the AR of 5:1 attachment provides minimum engine weight with lowest pump discharge pressure. An up-pass cooling circuit is used for both MCC and nozzle. A fraction of the nozzle cooling exit stream is diverted to the GG for combustion, and the remainder is mixed with MCC cooling exit stream and discharged into the main injector. The nozzle coolant has low ΔP compared to the MCC, and provides fuel with maximum energy to GG without penalizing pump-fed feed system. Fuel cooling was selected over oxidizer cooling from aspects of materials compatibility, engine start/cutoff sequencing, and safety.

Injector. The main injector will be of ring-type with self-impinging oxidizer and fuel doublet orifice pattern similar to previous LO2/RP1 injectors. These rings will be fabricated out of OFHC copper (same as for the F-1) to provide adequate injector face cooling at 1400 psia chamber pressure.

The injection pattern will be similar to that of the dependable RS-27 engine, but more closely packaged. Combustion stability will be aided by RP1 cooled baffles and by MCC injector-end acoustic absorbers.

Engine drawings. Figure 4.2.1.4-1 shows the physical arrangement and packaging of turbo-machinery and ducts, etc. They are packaged to minimize engine envelope size and to minimize engine skirt diameter.

Engine Weight. A preliminary engine weight summary is presented in Table 4.2.1.4-1 by component grouping. Engine design operating conditions and characteristics are also included. Total engine dry weight does not include engine accessories as noted. The necessity of these accessories will be determined later on, and they are subject to weight changes depending on vehicle requirements.

<u>Pogo Suppression</u>. Pogo instabilities associated with the coupling of the feed system, propulsion system, and vehicle structure during the boost phase can result in high amplitude vibrations, which in turn can cause structural failure. The proposed Pogo suppression system for the LRB is similar to the SSME system in function, and is located between the prevalve and engine/vehicle interface.

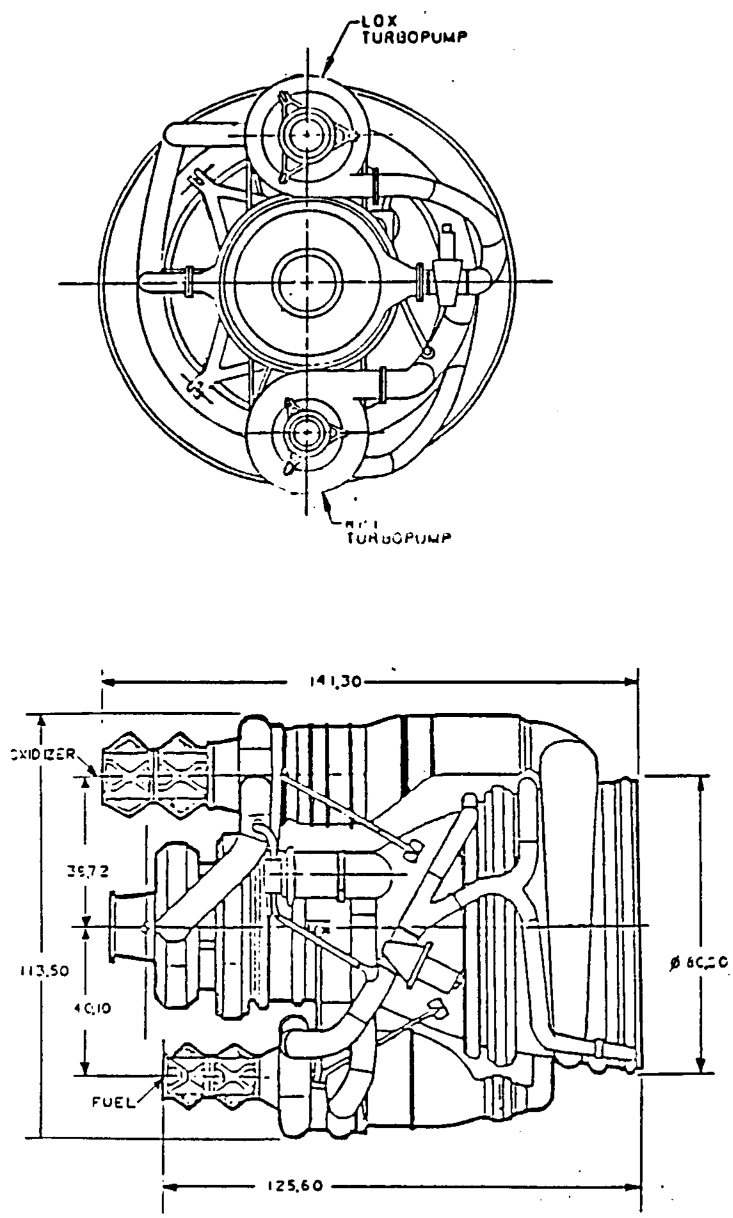


Figure 4.2.1.4-1 Drawing of LO2/RP1 Pump-Fed Engine for LRB

Table 4.2.1.4-1 Engine Weight Summary for LRB LO2/RP1 GG Engine

T/C THRUST	(KLB)		622.
CHAMBER PRESSURE	(PSIA)		1272.70
ATTACHED AREA RATIO	(NONE)		5.0
FIXED AREA RATIO	(NONE)		16.5
EXTENDIBLE AREA RATIO	(NONE)		16.5
T/C THRUST COEFFICIENT	(NONE)		1.8246
COMB. CHARACTERISTIC LENGTH	(IN)		39.13
CONTRACTION RATIO	(NONE)		2.8
ENGINE MIXTURE RATIO	(NONE)		2.53
NOZZLE PERCENT LENGTH	(PERCENT)		80.00
GIMBAL ANGLE	(DEG)		6.
TURBOMACHINERY :			
FUEL TURBOPUMP	443.9		
OXID MAIN TURBOPUMP	727.3		
SUB-TOTAL		1171.2	
GAS GENERATOR :		124.0	
EXHAUST GAS MANIFOLD :		69.5	
THRUST CHAMBER :			
GIMBAL BEARING	69.3		
INJECTOR	1171.1		
COMBUSTOR	1196.1		
FIXED NOZZLE	791.8		
SUB-TOTAL		3228.4	
VALVES AND CONTROLS :			
PROPELLANT VALVES	293.4		
CONTROL VALVES	54.6		
HARNESS AND SENSORS	183.4		
PNEUMATIC CONTROLS	122.6		
HYDRAULIC CONTROLS	48.8		
ATTACH PARTS	222.3		
SUB-TOTAL		925.1	
ENGINE SYSTEMS :			
PROPELLANT DUCTS	421.5		
ATTACH PARTS	60.0		
DRAIN LINES	40.8		
I.F. OXID. BLEED LINE	4.8		
I.F. FUEL BLEED LINE	16.3		
I.F. HYDRAULIC LINES	10.0		
I.F. GN2/HE LINES	26.3		
IGNITION LINES AND IGNI.RS	43.2		
PRESSURIZATION SYSTEM	74.8		
SUB-TOTAL		697.7	
ENGINE ACCESSORIES :			
FIXED NOZZLE THERMAL PROTECTION	46.3		
CONTROLLER AND MOUNT	85.0		
POGO SYSTEM	109.6		
SUB-TOTAL		241.0	
TOTAL ENGINE DRY WEIGHT W/O ACCESSORIES :			6215.9

Pogo suppressor design must be considered in the early stage of pump design to avoid significant impacts to either component.

Pressurization System. Two heat exchangers located downstream of the LO2 turbine provide GO2 and hot helium pressurants for the tanks. Oxidizer pressurant is bled off the main LO2 pump discharge stream, and is sent through the heat exchanger with a choked orifice at its exit. Helium, to be used for the fuel tank pressurization, is taken from vehicle storage bottles from vehicle stored bottles, and sent through a separate heat exchanger.

4.2.1.5 Engine Control.

Mixture Ratio Control & Throttling. The engine is required to have a continuous throttling capability of 25% down and 10% up from the NPL. This is a fairly large throttling range for LO2/RP1, and it will slightly penalize the pump discharge pressure requirements to provide adequate dynamic and combustion stability of the main injector. A $\Delta P/Pc$ design value of 20% is used for the gas generator and main injectors.

During throttling, the GG mixture ratio will be maintained constant to provide a constant combustion gas temperature of 1800°R to drive the hot gas turbines. This approach is taken for three reasons: First, 1800°R is the maximum design temperature without further development of turbomachinery materials and elaborate cooling concepts of the GG. Second, 1800°R has been shown by past experimental testing to be the temperature/mixture ratio limit for minimum carbon deposition on turbomachinery, a very desirable condition. And third, maintaining a constant GG combustion gas temperature requires a minimum secondary flow (GG gases) and thus results maximum engine performance during throttling. A preliminary list of required instrumentation is included in the Engine Appendix 5.2 to Volume II of the final report.

Since the GG gas flowrates are a small fraction of the engine flowrates, the thrust chamber and overall engine MR is not significantly impacted, and the engine operates at maximum efficiency over the entire mission trajectory.

Engine Instrumentation and Control. The LRB engine control and health monitoring system utilizes both performance and in-situ condition monitoring instrumentation to determine the overall health of the engine system to the extent required for acceptance testing. The health monitoring system will be integrated into the control system functions. The performance instrumentation is used by the controller to modulate the valve actuators to regulate both a constant GG mixture ratio and proper thrust level throttling.

4.2.2 VEHICLE PROPELLANT SYSTEMS. This section briefly describes the vehicle propellant feed, purge, fill/drain, pressurization, and vent systems for the LO2/RP1 pump-fed LRB. Some details applicable to all pump-fed systems are given in Section 6.2.2. For ground operation related systems, refer to Section 9.2 for more extensive description.

Preliminary design and operating requirements for the LO2/RP1 pump fed LRB propellant systems are summarized in Table 4.2.2-1.

4.2.2.1 Feed and Fill/Drain Systems. The feed system for the LO2 tank, schematically shown in Figure 4.2.2.1-1, includes a single 24-in external main line that leads from the center of tank aft bulkhead to a manifold at aft-end of the fuel tank. The manifold then divides into four 12-in ducts which connect to the oxidizer engine inlets. A single line was chosen over dual lines for minimum operational complexity, although dual lines for the LO2 side may be more cost effective as the ET's 17-in line toolings can be utilized. Also as shown, the feed system for RP-1 tank has a 16-in tank outlet that splits into four 8-in ducts leading to engine fuel inlets.

A prevalve, located immediately downstream of the manifold and in each engine inlet duct, regulates propellant flow to each engine during prelaunch, engine start and shutdown. The RP1 prevalves will be closed throughout tank fill operations, and opened only at engine start. The LO2 prevalves will be partially opened to allow some engine chill flows during the later phase of prelaunch operations.

The fill/drain lines for LO2 and RP1 are 6-in and 4-in, respectively, and each has a disconnect to facilities at one end and tee into the feed system manifold at the other end. The fill/drain lines provide a vehicle-facilities interface for tank purge, fill and drain. The fill/drain valve, located near the manifolds in the aft skirt region, are to be shut-off at the end of each tank fill operation (including replenish).

RP1 tanks are loaded long before LO2 tankfills for simplicity of operation. After purging the lines and tanks with GN2, LO2 tank chill and fill operations take place sequentially. All tank purges and loadings go through the fill/drain port disconnects which are vehicle interfaces with facilities propellant transfer lines.

In case of an abort on launch pad, propellants are drained through the fill/drain ports and back to facility storage tanks; RP1 tanks will need draining only in emergency cases. The tanks will need to be pressurized during draining to avoid subatmospheric ullage pressures and also to provide quick draining.

Table 4.2.2-1 Requirements for LO2/RP1 Pump-Fed LRB Propulsion System

Table 4.2.2-1 Requirements for LO2/RP1 Pump-Fed LRB Propulsion System

SYSTEMS	LO2	RP1	REMARKS
Engine Inlet			
Minimum P (psia)	65	45	
T (°R)	164	520	
Feed System			
Main Line D (in)	24	16	
Manifold to Engine D (in)	12	8	
Max Propellant Flowrate (lb/s)	6392	2528	
Max line P @ engine inlet (psia)	270	220	
Fill/Drain			
Duct D (in)	6	4	
Tank Operating			
Ullage P, Pre-press & Flight	18 psig	52 psia	constant from pre-start to BECO
Bulk T min-max (°R)	164-600	520-660	liquid - pressurant inlet @ diffuser
Max Tank Bottom P (psig)	70	60	occurs @ lift-off
Pressurization System	Autogenous	Heated He	
Medium	GO2	GHe	
Heating Source	turbine disch	turbine disch	
Supply Line T (°R)	164-1000	520-1000	
Supply Line P (psia)	ambient-600	ambient-600	
Main Supply Line D (in)	3	1.5	
Engine to Manifold Line D (in)	1.5	0.8	
Total Pressurant Wt (lb)	1180	360*	* 4 GHe bottles * each 25ft3 @ 4000psia, 520°R
GHe Pre-Press Line D (in)	1	1	
Vent System			
Valve D (in)	4	2	
Valve Operating P Range (psig)	0-20	0-54	
Purge System			
Engine Purge Supply Line D	TBD	TBD	GN2 ground supply @TBD
Total Liquid Residuals (lb)**	10800	2530	** Vehicle sizing assumes 1% of ascent propellant

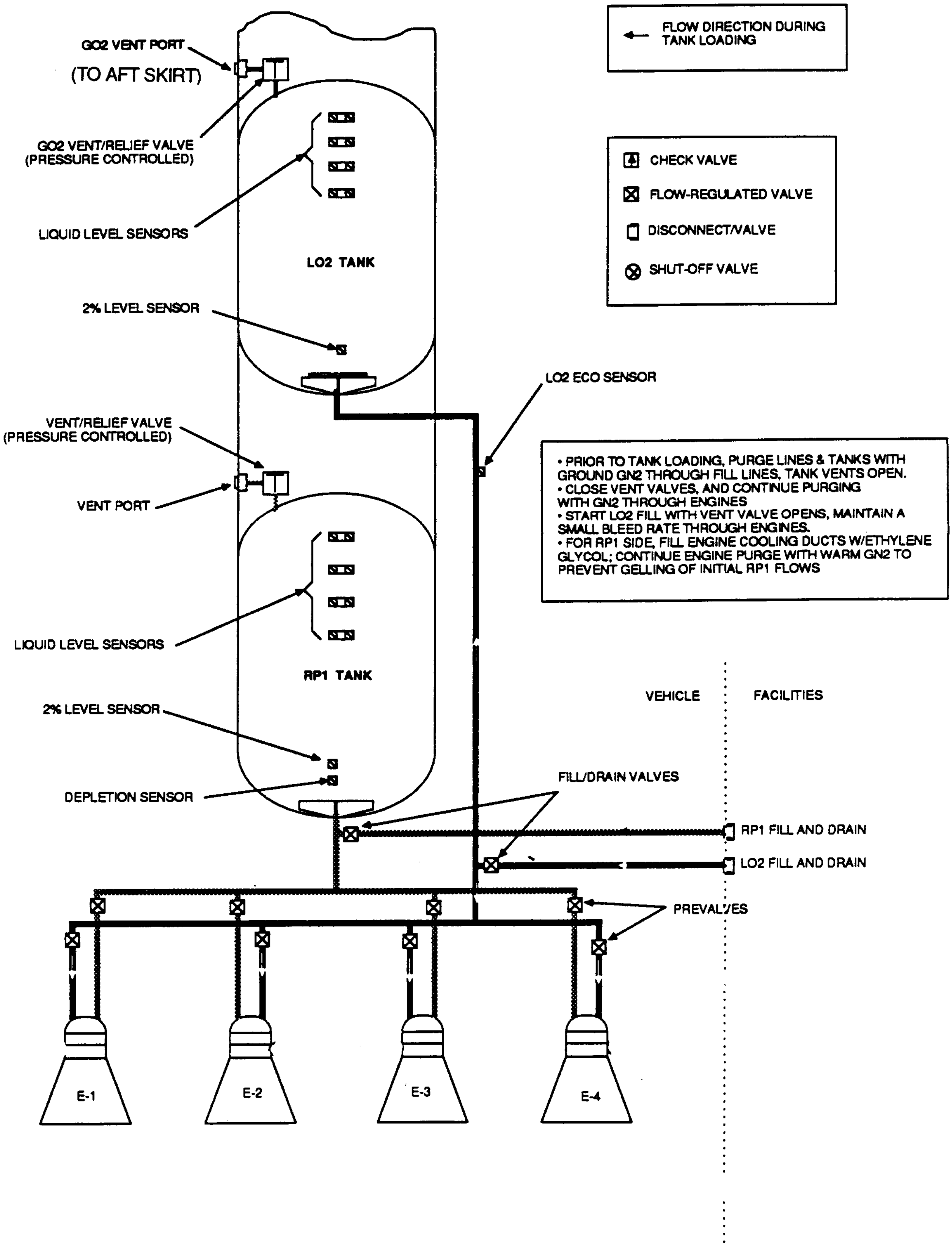


Figure 4.2.2.1-1 Feed & Fill/Drain Systems Schematic of LO2/RP1 Pump-Fed LRB

Anti-Geysering. In high density cryogenics like LO2, occurrence of geysering is very destructive because of large forces associated with it. This phenomena is dependent on the heat leaks, diameter and vertical length of the lines. Figure 4.2.2.1-2 shows a correlation of the data; heat leak parameter, B vs. geometrical parameter, A. For our LO2/RP1 booster, and for heat leaks scaled from ET, A = 0.36 and B= 5.5. As seen on the Figure 4.2.2.1-2, the conditions in the booster feedlines are far from the critical dividing line. Hence, at this time no anti-geyser system is provided for the booster. It should also be noted that He bubbling in the ET also causes de-stratification in the LO2 tank. The stratification in our LO2 tank, with replenish flow and heat leaks but no He bubbling, needs further analysis.

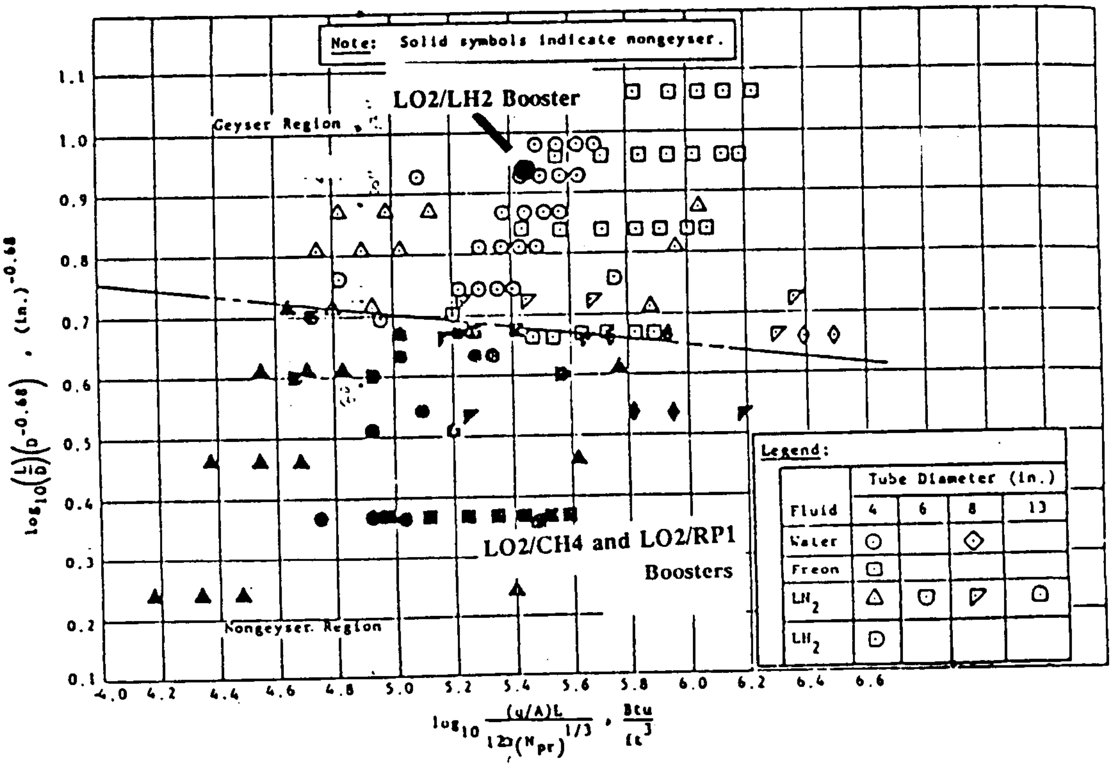


Figure 4.2.2.1-2 Geyser-Nongeyser Correlation

Engine Conditioning Systems. These systems are also shown in Figure 4.2.2.1-1. Engine conditioning is achieved by keeping The LO2 prevalve opened during tank loading. The vapor generated due to heat leaks leaves the system through the feedlines.

Engine cooling ducts are to be filled with ethylene glycol to prevent frost formation and provide conditioning for duct walls. Hypergols for engine ignition system should also be filled ahead of time. Fuel side turbomachinery should have a warm GN2 purge as discussed below.

4.2.2.2 Purge Systems. Schematic for purge systems is shown in Figure 4.2.2.2-1. Typically, purging of the tanks and feedlines with GN2 through fill/drain lines are required prior to tankfills.

Engine turbomachinery is also purged using GN2, through a separate engine purge connected to the warm high pressure ground supplied GN2. The purge on LO2 engine system is terminated for engine chill. The GN2 purge should still be maintained on the engine fuel side up to engine start to prevent frost formation and gelling of RP1.

The aft adapter, intertank adapter and nose cone compartments are purged with GN2 to maintain a low positive delta pressure from start of LO2 loading operation until lift-off All purge gases are ground supplied.

4.2.2.3 Pressurization Systems. Flow schematic for the tank pressurization systems is in Figure 4.2.2.3-1. The line sizes and operating conditions are included in Table 4.2.2.3-1. Prior to engine start, tank pre-pressurizations is done utilizing ground GHe. Separate GHe supply line/disconnects are provided for the fuel and oxidizer pressurization lines. Beginning at engine start and throughout flight, autogenous pressurization system takes over for the LO2 tank. GHe system acts as a back-up until to lift-off.

The RP1 tank switches to vehicle GHe pressurization system at lift-off. The GHe pressurant is provided by 4 helium bottles, stored at ambient temperature about 3.7 ft in diameter, located just below the fuel tank. All four helium bottle outlets are merged into a single path with pressure regulator. The line is then split into four individual paths, each leading to an engine heat exchanger where the helium is heated to about 700-1000°R.

In each engine, both oxygen and helium pressurant streams are heated through engine heat exchangers located at downstream of the LO2 turbine exhaust.

Pressurization systems downstream of the engines are identical for LO2 and RP1, the lines exiting the heat exchangers are merged into a single line that leads up to the top of each

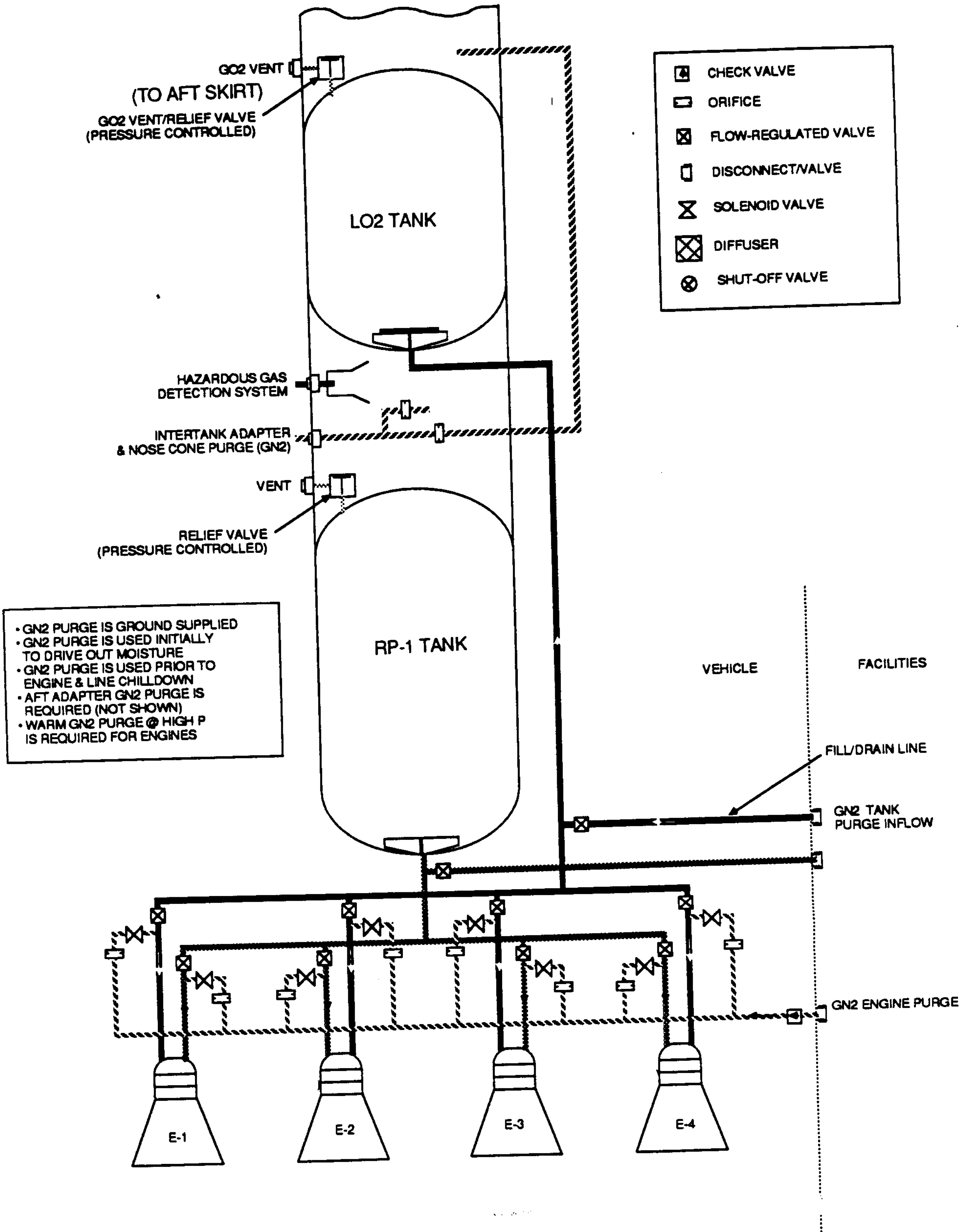


Figure 4.2.2.3-1 Purge Systems Schematic of LO2/RP1 Pump-Fed LRB

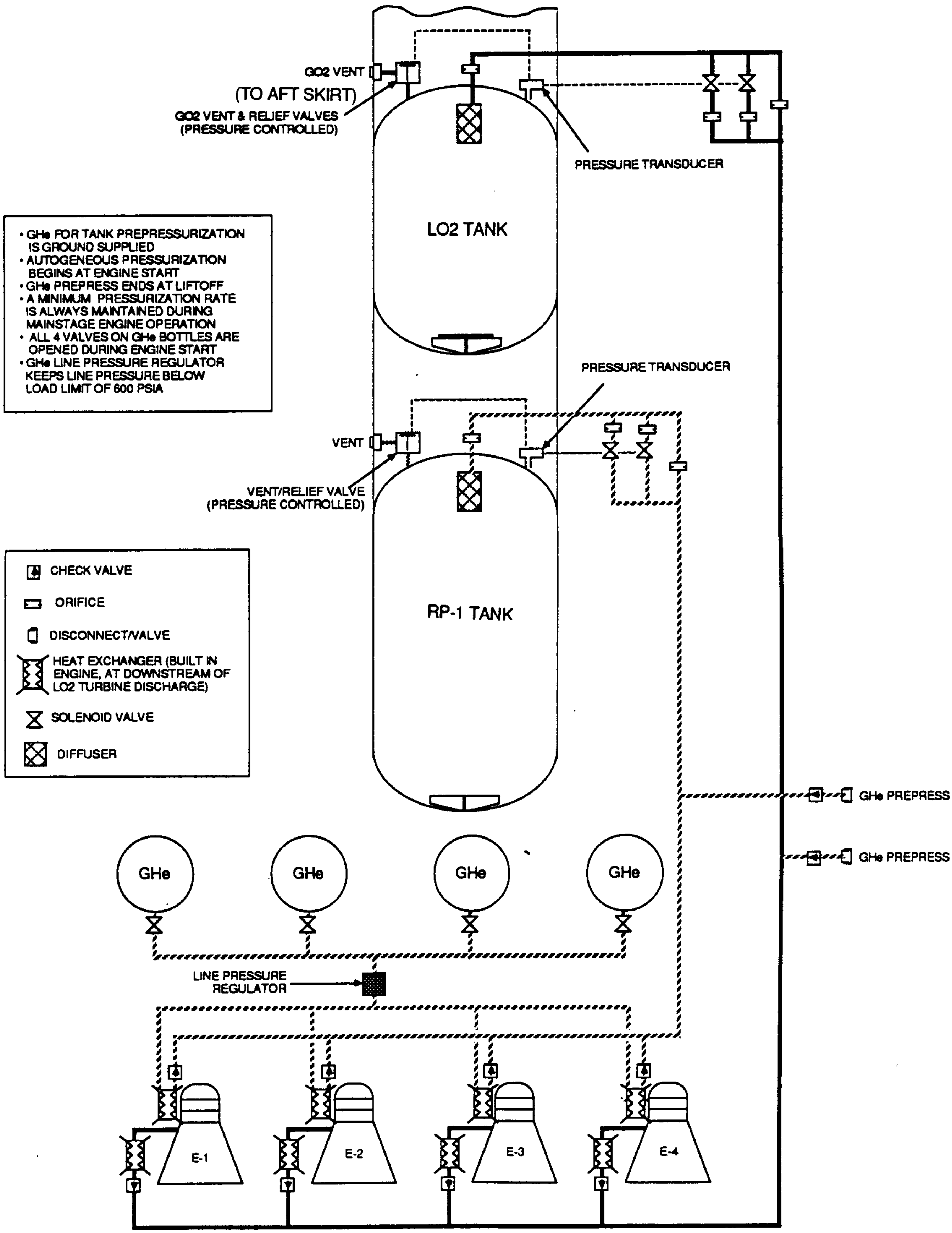


Figure 4.2.2.3-2 Pressurization & Vent Systems Schematic of LO2/RP1 Pump-Fed LRB

tank. For both the LO2 and RP1 pressurization systems, a two-level flow control with a bypass, that provides a minimum pressurization rate throughout flight, is utilized. This is discussed in Section 4.2.2.4. A diffuser is also required at the top of each tank to disperse the incoming pressurant stream.

Tank Operations. Throughout loading and replenish periods, a small positive delta pressure of about 2-6 psig is maintained in the LO2 tank. The RP1 tank only needs to keep a helium or nitrogen atmosphere in the ullage and pressurization lines at about 2-6 psig to lock out moisture prior to pre-pressurization.

Operating tank pressure requirements are driven by NPSP, line losses and hydrostatic pressure. Preliminary ullage pressure control requirements during flight are shown in Figure 4.2.2-5a and 5b for LO2 and RP1 tank, respectively. These pressure profiles are to minimize tank bottom pressure requirements and pressurization rate requirements, which can impose an increase in tank and pressurization system weight, while taking flight conditions and pressurization requirements into consideration. Further analysis and trades are needed to optimize the vehicle cost.

Pressure control logic and operation are discussed in Section 4.2.2.4.

Vent and Relief Systems. A vent/relief system is located at the top of each tank, providing outlet for purges and boil-offs during prelaunch operations and relief for over pressures during flight. LO2 vent gases are routed through a pipe external to the LRB Tank skins to the aft skirt where GSE connections are located. RP1 tank does not need a flare stack for vapor disposal, because RP1 is non-toxic and non-volatile.

Vent/relief valve operating pressure profiles are also depicted in Figure 4.2.2.3-3 and 4.2.2.3-4.

4.2.2.4 Propellant Management System and Propellant Inventory. The propellant management system of the booster consists of liquid propellant management and the gaseous propellant management systems. In this section, the baseline instrumentation used for the propellant management systems and the propellant inventory are also included.

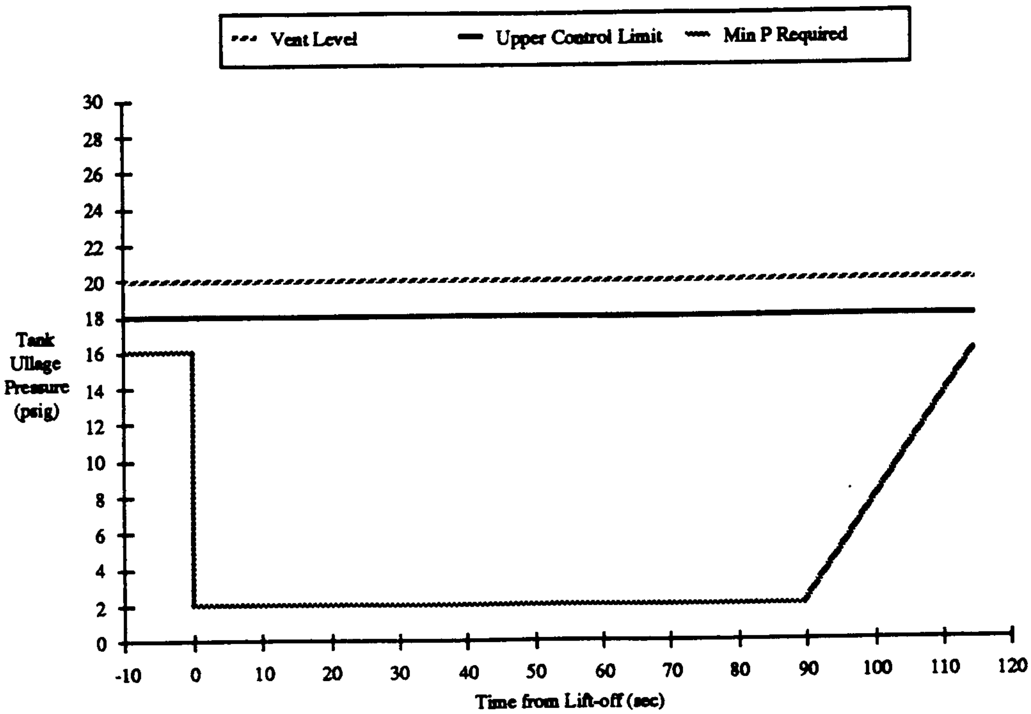


Figure 4.2.2.3-3 LO2 Tank Ullage Pressure Control Profile for LO2/RP1 Pump-Fed LRB

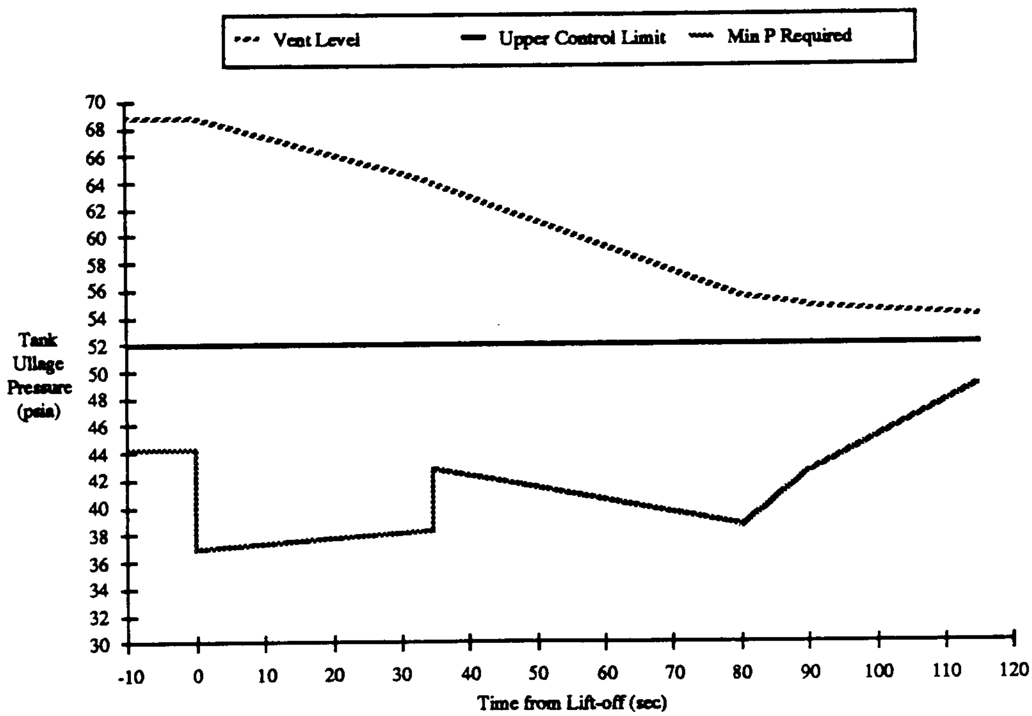


Figure 4.2.2.3-4 RP1 Tank Ullage Pressure Control Profile for LO2/RP1 Pump-Fed LRB

Liquid Propellant Management Systems. The liquid propellant management system starts with loading of the propellants. This is done by utilizing two 100% fill point type sensors. For proper tank fill operations and for high liquid level warning, one 100.5% and one 98% sensor are also needed.

An active propellant utilization system in the tank decreases the dispersion in the propellant outage. It replaces the rotating parts of the flowmeter by a passive system. However, P/U system does require more complex software and does require the engine to work over a range of mixture ratio. This has been done on the Atlas and Centaur without any problems, and at present we do not anticipate any problem with this engine. However cost reliability trade has not been completed, between a P/U system and no P/U system. Most of cost in the P/U system lies in operations before and after the flight.

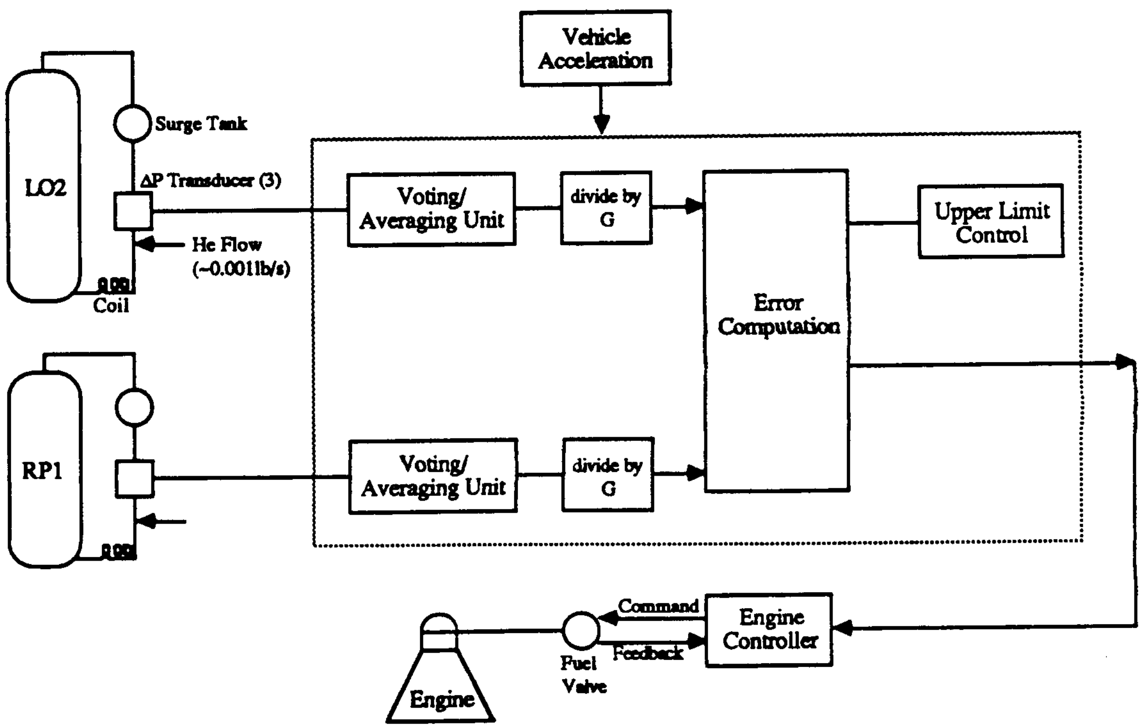


Figure 4.2.2.3-5 Propellant Utilization System

A propellant utilization sensing system envisioned for our concepts involves measuring the differential pressure in both the tanks and the vehicle acceleration as schematically shown in Figure 4.2.2.3-5. The propellant utilization control unit measures the ratio error existing between the LO2 and the RP1 tanks, and then adjusts the fuel valves simultaneously of all the engines for equal propellant consumption. The output signal is integrated over a period of time to prevent spurious signals caused by noise and propellant sloshing. Both the oxidizer and fuel valves are controlled simultaneously for keeping the thrust level desired by the orbiter. Cost and reliability trades have not been completed between an active P/U system and no P/U system. Our baseline at this time shows no P/U system.

Gaseous Propellant Management System. The gaseous propellant management system utilizes output of 3 sensors (with one backup) mounted on top of the tanks. The output of these sensors goes to the voting/averaging unit which compares the error between these sensors, and either takes the mean of all the sensors, or the mean of the best working ones. This signal is then used to operate the pressurization control system.

The pressurant GO2 from all the engines goes to a single pressurization line. The pressurization control system needed for this booster is very simple because of large structural and NPSH margin most of the time, and is depicted in Figure 4.2.2.3-6 It consists of three parallel pressurant paths, one a by-pass path which is always open and the other two which are actively controlled by the pressurant control system. The mass distribution between these paths are controlled by the orifice size. This is discussed in greater detail in Section 5. The logic control circuit operates the two actively controlled valves depending upon the variation from the desired control band, each staggered by 0.1 psi. That is, if the pressure is 0.1 psi below the desired control band value, one valve is opened; if 0.2 psi two valves are opened. This logic decreases the overshoot and provides good control by the system.

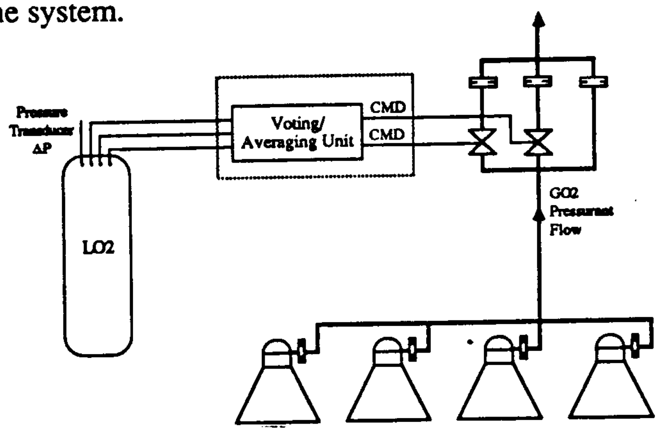


Figure 4.2.2.3-6 Pressurization Control System

Propellant Inventory. The ECO sensors on the LO2 system are located in the feedline at 43 ft. from the tank/engine interface. RP1 tank has a sump which prevents large dropout.

The current breakdown of the propellants weight as used in vehicle sizing is given in Table 4.2.2.3-3 below.

Table 4.2.2.3-3 Preliminary Propellant Inventory

Item	LO2	RP1
Ascent Propellants (lbs)	692,390	273,670
Start up Propellants (lbs)	33,020	18,060
Residuals including fuel bias (lbs)	6,920	2,740

Tank Instrumentation. A preliminary estimate of on the tank flight instrumentation required is given below:

- 4 tank pressure transducers
- 6 tank fill indicators (2 at 100%, 1 at 100.5%, 1 at 98%. 2 at 2%)
- 3 LO2 ECO sensors
- 2 RP1 low level fuel cut-offs
- 2 Temperature transducers in tank
- 4 pressure transducers in pressurization line
- 2 temperature transducer in pressurization line
- 1 propellant management unit
- 3 temperature transducers for intertank, nose cone and engine aft-adapter
- 2 ambient pressure measurement

4.2.2.6 Major Tank Interfaces and Interface Requirements.

- Hot N2 purge line to nose cone, intertank, and aft compartment
- Helium pressurization and anti-icing purge
- Pressurization line to each engine
- LO2 and RP1 feedlines to each engine
- Input from P/U system to engine controller
- Tank status to orbiter from propellant management system
- Hazardous gas detection to GSE

4.2.3 THRUST VECTOR CONTROL (TVC). Thrust vector control shall provide for engine excursions of six degrees, from the centerline, in any direction at a maximum angular slewing rate of 10 degrees per second and a maximum angular acceleration of one radian per second squared. The engine will gimbal about a ball pivot joint located at the top of the liquid oxygen manifold above the injector. The ball turns within the lubricated socket when the engine is gimballed in each direction by electromechanical gimbal actuators. In order to accept the active end of the gimbal actuators, the engine is furnished with outrigger struts spaced 90 degrees apart and projecting outward from the engine body. The actuators are positioned parallel to the engine with the stationary end anchored to the engine thrust structure and the active end anchored to the engine outrigger struts mentioned above. Control commands to the TVC will come from the Orbiter as shown in Figure 4.2.3-1. The power to drive and control the TVC shall be provided by the LRB.

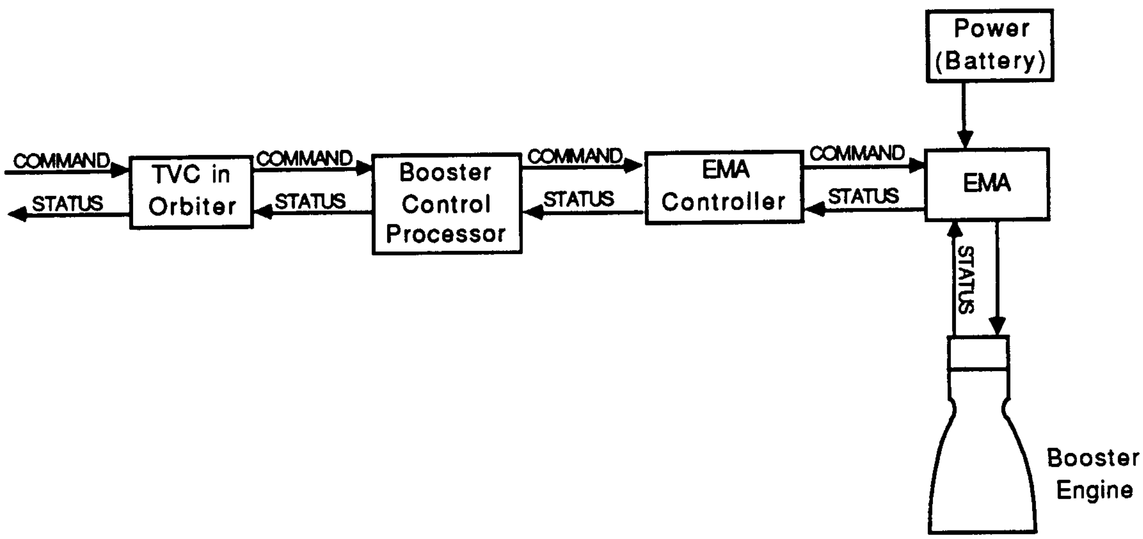


Figure 4.2.3-1. TVC Schematic

<u>LRB TVC EMA Requirements</u>. Recent advances in electronic power switching technology have made the concept of an electro-mechanical actuator (EMA) a viable alternative for thrust vector control (TVC) of the Liquid Rocket Booster. Primary benefits (see section 3.9) of the approach when compared with the more traditional hydraulics TVC concept are:

1. The ground test time now expended on hydraulics checkout will be reduced significantly with EMAs, which can be tested by computer in a manner similar to other redundant Avionics systems.

2. The single point failure associated with the auxiliary power unit normally used to provide hydraulic fluid pressure and energy does not exist in an EMA system.

3. A loss of pressure due to APU failure could be avoided by adding a redundant APU unit and extensive check valving, but the redundancy management problem is more complex since real time computerized valve control is a significant problem, especially when its launch site testing is considered.

Given these three significant benefits, the study of EMAs for LRB was undertaken to synthesize a set of EMA requirements for an electrical approach.

The goals of the EMA approach were to define a system with:

1. Single failure tolerance for the system, including its power source and gearing/leadscrew.
2. Total computer supervised ground testability with a minimum of operator interactions.
3. To be capable of slewing the engine at 10° per second.
4. To be designed to meet the requirements of each of the LRB engine types under consideration.
5. Use of available lead screw and motor technology with significant legacy wherever these are available and appropriate.

Torque Calculation Parameter Definition. The following parameters are needed to define the set of torque requirements for thrust vector control of a rocket booster. A diagram is provided in Figure 4.2.3-2, which may aid in visualizing what some of the parameters represent.

d^2Z/dt^2	=	Maximum LRB Longitudinal Acceleration
d^2Y/dt^2	=	Maximum LRB Lateral Acceleration
$d^2\theta/dt^2$	=	Maximum LRB Angular Acceleration
δ	=	Engine Gimbal Angle
$d\delta/dt$	=	Engine Gimbal Rate (slew rate)
$d^2\delta/dt^2$	=	Engine Gimbal Acceleration
M_E	=	Engine Mass
l_r	=	Distance from Engine C.G. to Gimbal Point
Y_{CGE}	=	Engine C.G. Trim Eccentricity
I_E	=	Engine Moment of Inertia

l_c	=	Distance from Vehicle C.G. to Gimbal Point
F	=	Engine Thrust
ε	=	Engine Thrust Misalignment
μ_F	=	Gimbal Block Coefficient of Friction
R_B	=	Gimbal Block Pin Radius

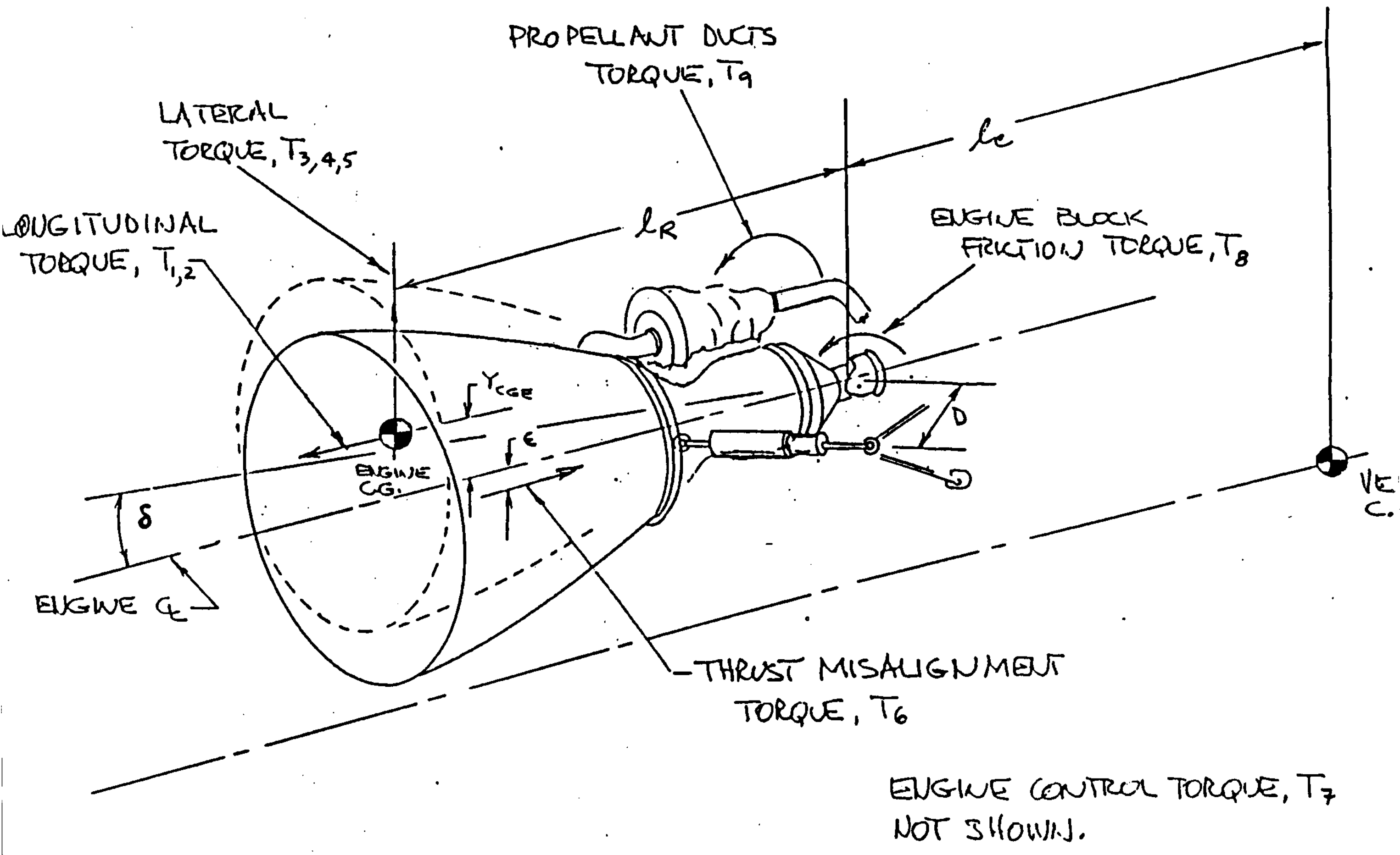


Figure 4.2.3-2 Diagram of TVC torque requirements.

<u>Equations Defining Requirements</u>. The following set of equations represent various components of the torque requirements placed on the thrust vector control system. These equations correspond to the equations used in the summary sheet shown in Table 4.2.3-1 for torque requirements of each of the engines under consideration.

Eq. 1)	T_1	=	Longitudinal Acceleration Torque due to Engine C.G. Offset
		=	$M_E*(d^2Z/dt^2)*Y_{CGE}$
Eq. 2)	T_2	=	Longitudinal Acceleration Torque due to Engine Gimbal Angle
		=	$M_E*(d^2Z/dt^2)*l_r*SIN(\delta)$
Eq. 3)	T_3	=	Lateral Acceleration Torque due to Vehicle Lateral Acceleration
		=	$M_E*(d^2Y/dt^2)*l_r$
Eq. 4)	T_4	=	Lateral Acceleration Torque due to Engine Moment of Inertia
		=	$d^2\theta/dt^2*I_E$
Eq. 5)	T_5	=	Lateral Acceleration Torque due to Vehicle Angular Acceleration
		=	$M_E*(d^2\theta/dt^2)*l_r*l_c$
Eq. 6)	T_6	=	Thrust Misalignment Torque
		=	$F*\varepsilon$
Eq. 7)	T_7	=	Engine Control Torque
		=	$I_E*(d^2\delta/dt^2)$
Eq. 8)	T8	=	Engine Block Friction Torque
		=	$F*\mu_F*R_B$
Eq. 9)	T9	=	Propellant Duct Torque (Provided by Rocketdyne)

Torque calculations and the values of the engine parameters they are based on, are given for the LO2/RP1 engine in Table 4.2.3-1.

EMA System Definition. Given the broad framework of the goals for an EMA TVC system, several concepts trades were conducted for development of a preliminary design. The issues traded were:

1) The use of two or four actuators per engine, including: Consideration of whether a strategy can be developed to return the actuator to null in the event of a failure.

2) The use of high frequency induction motors (400-1000 Hz) as an alternative to rare earth permanent magnet DC motors, including: The addition of brakes for the DC brushless approach, so that motor power requirements are reduced.

3) The use of silver zinc, lithium thionyl chloride, or thermal batteries.

4) The use of resonant power converters vs. standard DC brushless conversion.

5) The use of a redundant AC inverters and a redundant AC bus instead of redundant DC busses.

6) The use of cold plate vs. phase change materials for storing the waste heat from the motor controller electronics.

7) The use of rotary actuators instead of lead screw linear actuators.

Quantity of Actuators. This trade study addresses the total quantity of actuators needed, as well as how the overall system can be single failure tolerant. Two alternatives were considered: the use of two actuators at right angles, and the use of four actuators, where there is a pair of actuators in series for pitch control and two in series for yaw control (See figure 4.2.3-3). In the four actuator configuration, one of each pair is a de-activated backup with twice the stroke of the other prime unit. It is held at null unless the prime actuator fails.

Although further study is required to ensure that engine bell clearances and overall shuttle dynamics can be controlled properly, the two actuator approach appears viable. A total system control dynamics study is required which includes the shuttle orbiter and external tank so that dynamics of the entire vehicle is modeled. It must also consider plume impingement from an actuator hung up in an unnulled position.

As part of this trade, the question of whether the engine would return to null if it were free to move was considered. It was assumed that the leadscrew linkage would be pyrotechnically sheared in the event an actuator hangup is detected. There are two forces which would return the engine to a null (see Table 4.2.3-1). They are: 1. Flex line Table

4.2.3-1 LRB TVC Requirements for LOX/RP1 Gas Generator Engine

Input Parameters		**Source**
Thrust per Engine	629871 -lbs.**	Rocketdyne
Number of engines on LRB	4	LRB Baseline
Thrust Vector Offset	0.25 -inches	Rocketdyne
Gimbal Block Coeficient of Friction	0.06	Rocketdyne
Gimbal Block Pin Radius	4.44 -inches	Back-calc from R data
Maximum Gimbal Angle	6.00 -Degrees	.Specification
Gimbal Rate Required	10.00 -Deg/Sec	Specification
Gimbal Acceleration Required	57.30 -Deg/Sec^2	Specification
Engine Weight	6216 -lbs.	Rocketdyne
Engine Inertia	2821 -lb-ft^2	Back-calc from R data
Distance from Engine C.G. to Gimbal	43.00 -inches	Rocketdyne
Distance from Vehicle C.G. to Gimbal	100.00 -ft	Estimated
Actuator Moment Arm	32.00 -inches	Rocketdyne
C.G. Offset from Centerline of Eng.	0.00 -inches	Assumed
Max Veh Longitudinal Acceleration	3.00 -g's	STS limit
Max Veh Lateral Acceleration	0.30 -g's	STS/LRB Traj Sim
Max Veh Angular Acceleration	3.00 -Deg/Sec^2	STS/LRB Traj Sim
LOX Line Torque	3409 -ft-lbs	Rocketdyne
Fuel Line Torque	2246 -ft-lbs	Rocketdyne
Total Flex Line Stiffness Torque	5655 -ft-lbs	Rocketdyne

Torque Calculations		
Longitudinal Acceleration Torque	6985 ft-lbs	83816 in-lbs
T1 - Due to Engine C.G. Offset	0 ft-lbs	0 in-lbs
T2 - Due to Max Gimbal Angle Offset	6985 ft-lbs	83816 in-lbs
Lateral Acceleration Torque	10452 ft-lbs	125421 in-lbs
T3 - Due to Vehicle Lateral Acc	6682 ft-lbs	80186 in-lbs
T4 - Due to Engine Inertia	148 ft-lbs	1772 in-lbs
T5 - Due to Vehicle Angular Accel	3622 ft-lbs	43463 in-lbs
T6 Thrust Misalignment Torque	13122 ft-lbs	157468 in-lbs
T7 Engine Control Torque	2820 ft-lbs	33845 in-lbs
T8 Engine Block Friction Torque	13967 ft-lbs	167609 in-lbs
T9 Propellant Duct Torque (Given)	5655 ft-lbs	67858 in-lbs
Total Static Torque	36214 ft-lbs	434563 in-lbs
Total Dynamic Torque	16788 ft-lbs	201454 in-lbs
Total Required Torque	**53001 ft-lbs**	**636017 in-lbs**

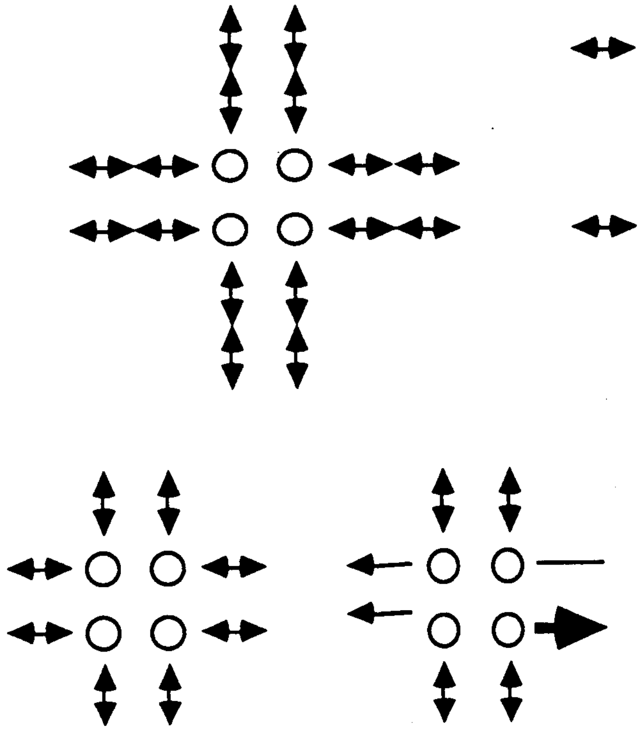
Peak Power Requirements
(Using Torque X Gimbal Rate)

Peak HP Output req'd per Actuator	16.8 -hp	12.5 -kW
Peak HP/Act (using sys eff of 50%)	33.6 -hp	25.1 -kW
System Peak HP Requirement	67.3 -hp	50.2 -kW
Total Peak Required for LRB	**538.2 -hp**	**401.3 -kW**

Actuator Sizing

Peak Operating Output Force	19876 -lbs
Stall Force	29813 -lbs

** Head-end gimbal point



Prime actuator moves engine in the direction shown, ± 6° from null. If it ever hangs up, a brake holds it in its stuck position.

Secondary actuator, which is in series mechanically with the prime unit, normally has its brake on. If prime fails,
secondary brake is released, and the secondary unit relatively slowly
drives the engine to its center position.

Prime actuator moves engine in the direction shown, ± 6° from null. If it ever hangs up, a brake holds it in its stuck position. Other actuators provide control
Assume bottom right actuator stuck--left side engine actuators counteract its force. Actuators shown in the vertical direction provide roll control,
may require disabling of stops at 6 ° to provide sufficient dynamic range.
Also requires countermotion of engines.

Figure 4.2.3-3 Sixteen and eight actuator configurations were considered, with the eight actuator concept being preferred.

stiffness; and 2. Gravity and acceleration at the minimum acceleration of 1.3 g. Together at their maximum values (67858 in lbs+83816 in lbs) they total less than the nominal offset

torque of 157468 in lbs. Unfortunately this torque would continue to force the engine to a further offset position if the actuator connection was sheared and the engine would not return to null position if it were free to move. Shearing of the actuator attachment was therefore deemed to be not viable.

<u>Induction Motor vs. Rare Earth Motors</u>. Normal 400 Hz induction motors rated at 30 horsepower have an estimated rotor inertia of 50 in-lb-sec and friction torque loss of 50 in-oz at their rated speed. These numbers increase the torque required for an LRB sized actuator to 600 million in-lbs. This is a unreasonable requirement, and eliminates the use of existing induction motors.

Using lower inertia permanent magnet motors, only 32 hp are required. It should be noted that the above conclusion is based on induction motor technology as it exists today. Advances in induction motors have the potential to make them competitive. The rotor of an induction motor can be fabricated using aluminum rather that steel, making it three times lighter. It could also be hollow, reducing its weight again by a factor of two to three. Because of these issues, further work is planned and should be done to re-evaluate the induction motor approach.

When DC brushless motors are used, they must be sized so they will not overheat while they react continuously to the constant loads. Their manufacturers proposed that a brake could be included to resist continuous loads when the actuator motor is not turning and is static. This has been rejected because it is another component which could cause a single point failure, reducing reliability and safety. Also the loads are predominantly static, so braking to reduce duty cycle would seriously impact the dynamic response of the system. Therefore, the motors were sized for continuous duty.

The maximum rated continuous operating temperature of the motor is 225C, which is limited by insulation capability and permanent magnet degradation above this temperature. For the 150 second flight period, the 8 lb mass of each motor, and its thermal path to its actuator flange which acts as a cold plate both, provide a heat sink. At a continuous 8 HP per motor, temperature rise with a 75% efficient motor is an acceptable 150C. Therefore, motor size specifications have been established so that heating will not be a problem.

Batteries. Proven Silver Zinc technology has been used on launch vehicles such as Centaur for years. On Centaur, a specific energy of 41 Watt-hrs/lb is achieved. Recently, our lithium thionyl chloride technology battery demonstrated 100 Watt-hrs/lb. We expect to implement it on Centaur during the next decade. Thermal batteries which have a density of 200 Watt-hrs/lb fly on our Tomahawk cruise missile during its 30 sec boost phase. These however, could not be tested as a system prior to launch in the same way that primary batteries can.

For our LRB baseline we recommend the silver/zinc system, since it is a proven, safe, well understood system, and testable before launch. The lithium thionyl chloride design should be retained as an alternative, pending further development.

Resonant Power Conversion. Recent NASA sponsored work on motor controllers at the University of Wisconsin has demonstrated the use of a bi-directional 20khz motor controller (its power stage was designed by GDSSD) with a two horsepower 400 Hz induction motor. The approach demonstrates that an induction motor can be used as a generator to absorb "braking" energy so that the multi-kilowatt regenerative power which the system would produce can be used by other actuators or dissipated under electronic control. Since there are always constant thrust vector offset loads as well as g and spring loads whenever the engines are off null, it is highly probable that any regenerative energy will be used and will not be wasted. A full dynamics simulation of the system is required to estimate the savings in battery weight which will accrue due to this capability.

Existing DC brushless motor controllers also have a regenerative capability. Final selection of the conversion technology requires more detailed trades involving the specific quality requirements of the switches used to drive the motors, the control dynamics assessment of aiding loads, and final motor type selection.

Type of Power Bus. Significant funding is being expended on developing the Space Station 20 kHz dual bus, a fault tolerant architecture for power levels in the 25 kW range. For example, the space station main inverters are rated at 25 kW. However, the guaranteed transient recovery time for this bus is 50 milliseconds, and this would not be acceptable for the LRB. Accordingly, individual inverter/converter pairs are recommended for each 8 kw motor.

Cold Plate Approach. Two types of cold plates were considered to absorb electronics waste heat during engine start and powered flight. They were:

1. A pure beryllium plate.
2. A cold plate which uses a phase change material.

For the beryllium plate, our design groundrules allow a 110° switch junction temperature. This allows a 50C delta T rise for the cold plate. The phase change approach uses the melting heat of fusion of one of several phase change materials which when they melt have a 200 J/gm heat of fusion. This option would save 7 lb per engine. However, since the total mass saving involved is small (28 lbs total), beryllium is recommended for its simplicity.

Actuator Type. Rotary actuators were considered for TVC application. However, since linear actuators have supported engines in this thrust class successfully in the past (SSME), and since Rocketdyne recommends a linear actuator for the engine, it was used in the baseline. Further study in this area is also recommended.

Mass Properties. Table 4.2.3-2 lists the mass properties of the baseline configuration, as well as the deltas associated with some of the trade options.

Sources for all the mass properties data are also listed. Mass estimates for the baseline motor configuration are based discussions with two manufacturers whose speciality is high speed rare earth motors. These discussions included the development of specifications and a peer review. Speed ratio was selected to match required output slew rate to motor speed capability. In a similar manner, leadscrew masses are based on minor design modifications of existing designs, with attention paid to load force, transfer function identification for the required speed ratio, and viscous friction.

The weight estimate for the electronics required to control power to the baselined rare earth DC motors are based on our space station study experience. General Dynamics is designing the Space Station 20 kHz 25kw Main Inverter Units and has also worked on motor controllers for high speed induction type motors using 20 kHz technology. We now have a significant body of predesign data which supports the weight estimate given. On Space Station, we are contractually committed to delivering inverter hardware with this quoted specific power of 0.3 kw/kg.

Table 4.2.3-2 Mass properties estimates and deltas for the alternative configurations considered.

COMPONENT	WEIGHT (LBS)	DATA SOURCE
MOTORS (4X 8 HP=32 HP)	32	MPC and PARKER Catalog and spec d:
ACTUATOR / GEARS(120,000 FT LB)	35	MPC data submittal
SINGLE CHANNEL ELECTRONICS	75	Space Station studies and predesigns
BATTERY STRING (24 KW FOR 150 SEC)	48	Centaur main battery
BATTERY POWER SWITCHING	25	Atlas Power Distribution Unit
SUBTOTAL PER AXIS (EMA)	215	
SUBTOTAL PER ENGINE (X2)	430	
TOTAL PER VEHICLE (X4)	1720	
TOTAL (4 PAIRS OF NULLING REDUNDANT EMAS)	+584	PER VEHICLE
TOTAL (LITHIUM BATTERYS)	-96	PER VEHICLE
TOTAL (PC HEAT SINK)	-28	PER VEHICLE

4.3 AVIONICS

4.3.1 SYSTEM ARCHITECTURE. The LRB avionics architecture is an autonomous 3-string single fault tolerant configuration (see Figure 4.3.1-1). The Booster Control Processor (BCP) is the central controller and data handler of the avionic system. Each BCP will command functions redundantly over three system buses independently. The failure tolerant Remote Voter (RVU) takes inputs from the BCP for system commands. From the RVU, the discrete commands are sent to the proper loads such as tank pressure control, pyrotechnic control, TVC, engine control, etc. In the present configuration with the SRB , the Orbiter avionics commands ARM, FIRE1, FIRE2 are required for SRB separation. This command sequence is based on "SRB CHAMBER PRESSURE" which is used to determine if the SRBs have completed burning. Therefore, it is proposed that the Orbiter avionics retain the separation command to minimize the changes to the Orbiter interface and software.

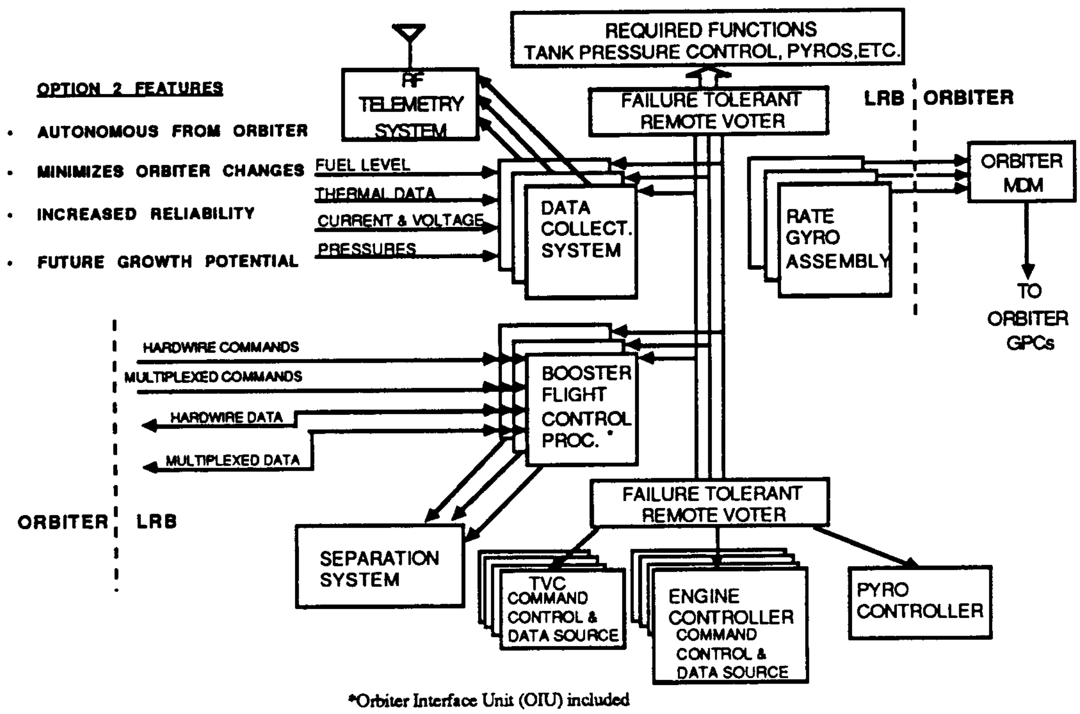


Figure 4.3.1-1 LRB avionics architecture (3 string single fault tolerant configuration).

Concerns which accompany the separation command from Orbiter avionics are that normally the LRB engines may be providing full thrust at that time, tank pressures may cause venting, and so on.

Such concerns may be alternated by the following procedure. A command from the Orbiter avionics must be sent to advise LRB avionics that the separation will occur shortly (at a precise time). Engines will be shut down and LRB subsystem will be readied for separation. Then the Orbiter avionics will command the pyrotechnic separation of LRBs. Since the avionics shut-down period is very short compared to fluid and propulsion system shut-down, these sequences will drive total time between command to prepare for separation and separation command.

Problems which may occur are dispersion of engine shut down times, if a time sequence is used. The Orbiter/ET may have to drag the LRB for the time after nominal shut-down to worst case shut down prediction.

The Data Collection System (DCS) collects data for both control feedback and for telemetry information. The information required for control is provided to the BCP over the system bus, therefore maintaining the failure tolerant system. The telemetry data is sent to an on board RF telemetry system and transmitted to the ground. The Orbiter Interface Unit (OIU) included in each BCP minimizes the electrical interface changes with the Orbiter avionics. It does this by accepting the Orbiter hardwired and multiplexed commands and reformats them for the LRB system. All data from the LRB to the Orbiter is likewise reformatted to make the data compatible with the Orbiter formats. The Rate Gyro Assembly (RGA) provides rate data during powered flight. The rate data (pitch, roll, yaw) will be transferred to the Orbiter avionics through a direct connection to an Orbiter MDM.

4.3.2 ORBITER INTERFACES.

Flight Control. The Orbiter GPC's will retain flight control over the entire vehicle stack. The booster Rate Gyro Units (RGU's) will supply data to the GPC's over the same lines as were used by the SRBs. The GPC's will generate the Thrust Vector Control commands for each Liquid Booster exactly as it is done for the SRBs. This would seem to make a large impact on the number of TVC drivers that need to be incorporated on the Orbiter. The orbiter TVC drivers "view" the cluster of liquid nozzles as one large nozzle like an SRB. The orbiter TVC drivers provide the same current that is necessary to drive the thrust vector control. A dedicated module within the Booster Control Processor receives the current signal and translates it into individual commands for each LRB engine. These commands are sent to the EMA TVC actuator controllers and complete the TVC command.

Commands/Data. Commands will be sent in a redundant method to meet the required failure tolerance. The Orbiter avionics will interface directly with an Orbiter Interface Unit (OIU) in each of the three Booster Control Processors (BCP), to communicate with and command the LRB Avionics system. All commands that are flight critical (e.g., engine start, booster separation) will be hardwired to the LRBs. These hardwires will utilize the existing SRB to ET lines. All non flight critical commands will utilize existing multiplex command channels.

Telemetry. There are several methods which may be utilized to transmit the multiple engine, TVC, pressurization, and other data associated with the LRB to the ground for processing. One method would be to send all the LRB data through the Orbiter TLM system, interleave it with Orbiter and spacecraft data, and transmit it to the ground. This method is not very feasible since the present system is filled to capacity. The LRB data requirements are much higher than the SRB data requirements hence overloading the present Orbiter data/TLM system and forcing major modifications. A second method would incorporate separate TLM systems on each LRB. This would result in three separate data streams being transmitted to the ground for real time monitoring or to be recorded for post flight analysis. The second method seems to be the best choice because it minimizes effects on the Orbiter TLM system. It is believed that there are adequate RF tracking stations available in the launch area to receive this additional telemetry.

Flight Crew. The flight crew will receive key engine parameters for engine status. These parameters can be displayed on the current CRTs by modifying the present software (the preferred method) or by incorporating the use of new CRTs. Further study will be necessary to determine the best option. The flight crew will also have the capability to send an engine shutdown command manually to any of the LRB engines.

Power. The power supply for each LRB will be separate from the orbiter power. This means that no power interfaces will be necessary. The LRB power source for the avionics will be supplied by batteries, either Lithium based or Silver Zinc. The EMA power supply will also be batteries, and will be either a thermal battery or a Silver Zinc battery. Further investigation as to the battery type needs to be conducted, as that technology continues to advance from year to year.

4.3.3 <u>SUBSYSTEM INTERFACES</u>. The booster control processor will provide LRB specific commands to the LRB subsystems. These commands will be voted on in the Remote Voter Units (RVU). The RVU will turn on/off the appropriate end function (valves, engine ignite, etc.).

<u>Thrust Vector Control</u>. Each engine on the LRB will be gimbaled separately. A typical TVC system is shown in Figure 4.3.3-1. The TVC 'command' from the Orbiter will be translated into four separate TVC commands, one for each LRB engine, via a dedicated TVC processor within the BCP. The TVC processor will utilize input of engine thrust, engine out, and other parameters to command the LRB engines to yield the equivalent thrust vector as required by the GPC.

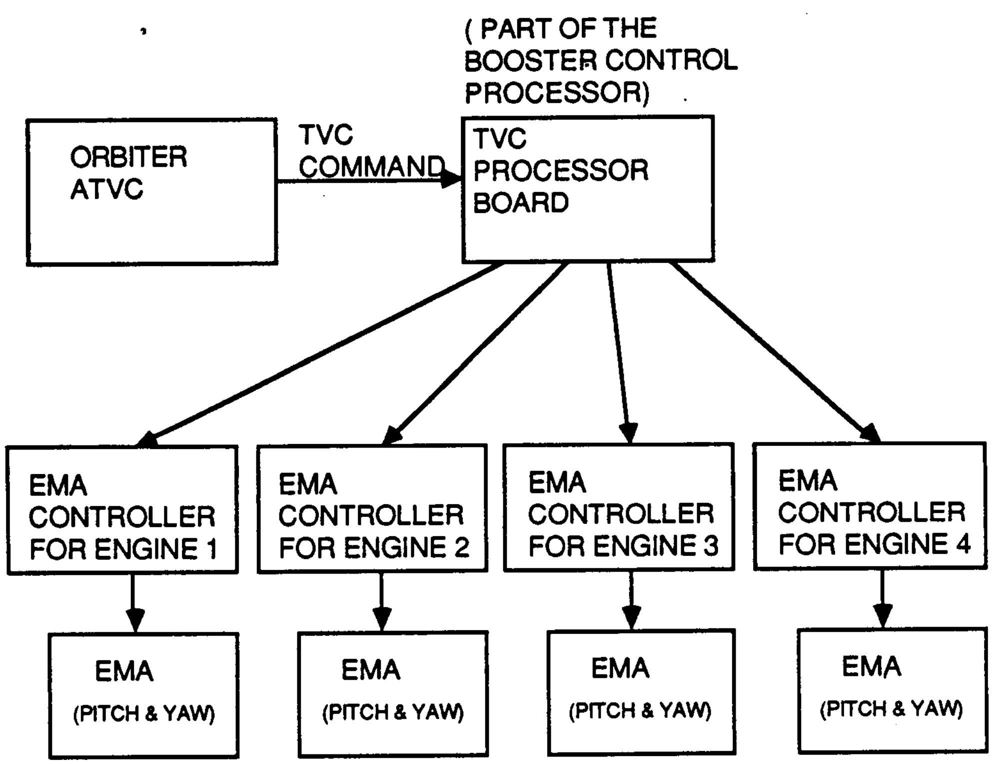


Figure 4.3.3-1 Typical TVC System

<u>Engine Control (Propulsion)</u>. The engine controller will be an autonomous processor which will monitor each LRB engine and change the parameters appropriately. The engine

controller will send the appropriate engine status data to the BCP and to the Orbiter crew. A complete data stream from the engines will be made available through a downlink scheme.

Pressurization. The pressurization system is made up of the Booster Control Processors, Redundant Pressure Sensors and Control Switches. The system will be autonomous from the Orbiter and only relay caution and warning information when pressurization limits are exceeded.

Data Acquisition. The data acquisition system will be a combination of redundancies. For measurements which will be used for control and/or decision purposes the sensors will be redundant. For those measurements used for monitoring purposes only the sensors need no redundancy. Table 4.3.3-1 contains a preliminary list of the types of sensors, the subsystems using the sensors and the number of sensors used.

Table 4.3.3-1. Preliminary LRB instrumentation

A preliminary breakdown of the instrumentation needed by the vehicle is shown below. The various subsystems, such as propulsion, structures, avionics, etc., supplied inputs with the assumption that enough instrumentation was to be included to provide automatic checkout, health monitoring, and fault detection/isolation.

	TEMP	PRESSURE	CURRENT	VOLTAGE	VIBRATION	ACCEL	DISPL	POSITION	RATE	DISCRETE	ANG SPEED	MISC	TOTAL
AIRFRAME	21	18			6					27		6	78
RANGE SAFTEY				8						65			73
ELECTRICAL (RVU)*			30	20						139			189
PNEUMATICS	15	28						4					47
TVC		4		8				8		8	4		32
GUIDANCE	6		3			9	12			3		24	57
TELEMETRY	1			2									3
MAIN ENGINES	72	96	24					48		12	8		260
PROPELLANT FEED	12	12						32					56
MISC													
TOTAL	127	158	57	38	6	9	12	92	0	254	4	30	791

* REMOTE VOTER UNIT (RVU)

4.3.4 GROUND INTERFACE. The ground interface for the LRBs will be as simple as possible. The umbilicals will consist of three data busses, an instrumentation bus and a power bus. The preferred location for the connection is on the aft end of each LRB.

4.4 RP-1 PUMP-FED PERFORMANCE AND TRAJECTORIES

4.4.1 NOMINAL MISSION. A description of the nominal mission trajectory simulation can be found in section 8.1.3. Since the ATO mission determined the required size of the RP-1 pump-fed LRB configuration, the nominal trajectory simulations were run simply to determine nominal performance.

The following table is a summary of the nominal performance for the RP-1 pump-fed configuration.

```
Lift off conditions:

   Weight (lb)                                   =   4,128,467.6000
   Payload (lb)                                  =   70,500.000000
   Thrust (lb)                                   =   5,646,246.6260
   Thrust to weight                              =   1.3676373834
   Initial inertial velocity (ft/sec)            =   1,342.4324022
   Launch site latitude                          =   28.307566153
   Launch site longitude                         =  -80.540959056


Max Q conditions:

   Max dynamic pressure (lb/ft**2)               =   727.93565042
      Time (sec)                                 =   62.048529322
      Angle of attack (deg)                      =   4.1212426304
      Altitude (ft)                              =   30,675.957171
      Mach number                                =   1.3042327125
      Q * ALPHA (deg-lb/ft**2)                   =   2,999.9994347


LRB separation:

   Staging time (sec)                            =   125.60030949
   Altitude (ft)                                 =   137,984.36662
   Dynamic pressure (lb/ft**2)                   =   74.998857866
   Angle of attack (deg)                         =  -1.9999997282
   Mach number                                   =   4.8462468298
   Inertial velocity (ft/sec)                    =   6,334.2340348
   Inertial flight path angle (deg)              =   21.302478395
   Relative velocity (ft/sec)                    =   5,100.1972575
   Relative flight path angle (deg)              =   26.820254934
   Delta V (ft/sec)                              =   9,054.9971859
   Weight after separation (lb)                  =   1,538,198.0990
   Remaining ET propellent (lb)                  =   1,181,913.1172
   SSME throttle at separation                   =   1.0400000000
   LRB throttle at separation                    =  0.78501228412
   Thrust (lb)                                   =   1,464,653.9808
   Thrust-to-weight after separation             =  0.95218813605
   Acceleration after separation                 =  0.93230790568
   LRB propellent used (lb)                      =   1,932,118.6182
```

Geodetic latitude (deg)	=	28.486273247
Longitude (deg)	=	-80.034805888
Average back pressure (psi)	=	5.8427847129

MECO conditions:

Time (sec)	=	495.29381562
Altitude (ft)	=	360,647.93463
Inertial velocity (ft/sec)	=	25,871.052177
Inertial flight path angle (deg)	=	0.76676527368
Delta V (ft/sec)	=	30,147.104973
Shuttle & payload perigee (nm)	=	35.178217832
Shuttle & payload apogee (nm)	=	159.91459258
MECO weight (lb)	=	361,320.92128
SSME throttle @ MECO	=	0.76938186376
SSME propellent weight used (lb)	=	1,582,914.0605
ET remaining propellent weight (lb)	=	5,035.9394585
Average back pressure (psi)	=	1.4825316475

Throttle schedules:

Max q throttle down time (sec)	=	49.997531407
LRB throttle setting	=	0.78819331610
Post max q throttle up time (sec)	=	67.313515565
Start 1620 kips attach load throttling	=	95.803320435
Start LRB 3g throttling (sec)	=	125.02651348
LRB throttle setting	=	0.78501228412
SSME throttle @ separation	=	1.0400000000
Start SSME only 3g throttling (sec)	=	449.82675656
SSME final throttle setting	=	0.76938186376

Losses to LRB separation

Total delta V	=	9,054.9971859
Steering losses	=	1,839.8237635
Drag losses	=	430.76863999
Gravity losses	=	1,399.6569473
Pressure losses	=	387.74674163

Losses to MECO

Total delta V	=	30,147.104973
Steering losses	=	2,483.8847469
Drag losses	=	440.14788517
Gravity losses	=	2,302.7839359
Pressure losses	=	387.90211342

Min/Max conditions:

Max (+) angle of attack (deg)	=	8.6710046035
Time (sec)	=	4.8618900363

```
Max (-) angle of attack (deg)                     =  -9.0831580276
   Time (sec)                                     =   16.108638460
Max (+) Q * Alpha (lbf-deg/ft**2)                 =   2,999.9995923
   Time (sec)                                     =   69.542264065
Max (-) Q * Alpha (lbf-deg/ft**2)                 =  -1,078.3574776
   Time (sec)                                     =   87.431028650
Max acceleration (g's)                            =   3.0002073618
   Time (sec)                                     =   125.02651348
```

Figures 4.4.1-1 thru 4.4.1-9 show various performance parameters obtained from the RP-1 pump-fed LRB configuration's nominal trajectory simulation.

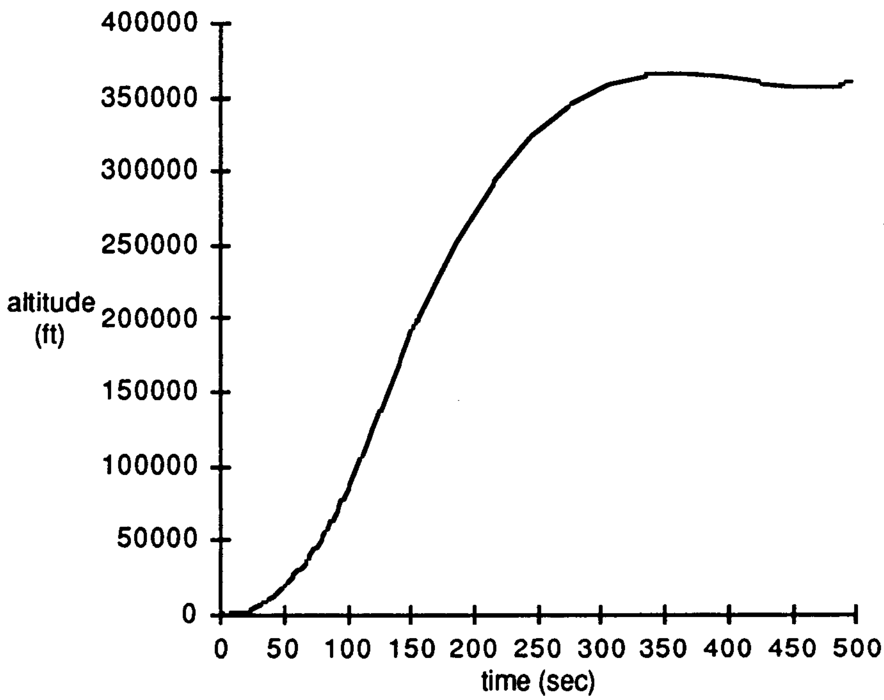


Figure 4.4.1-1 Altitude vs Time

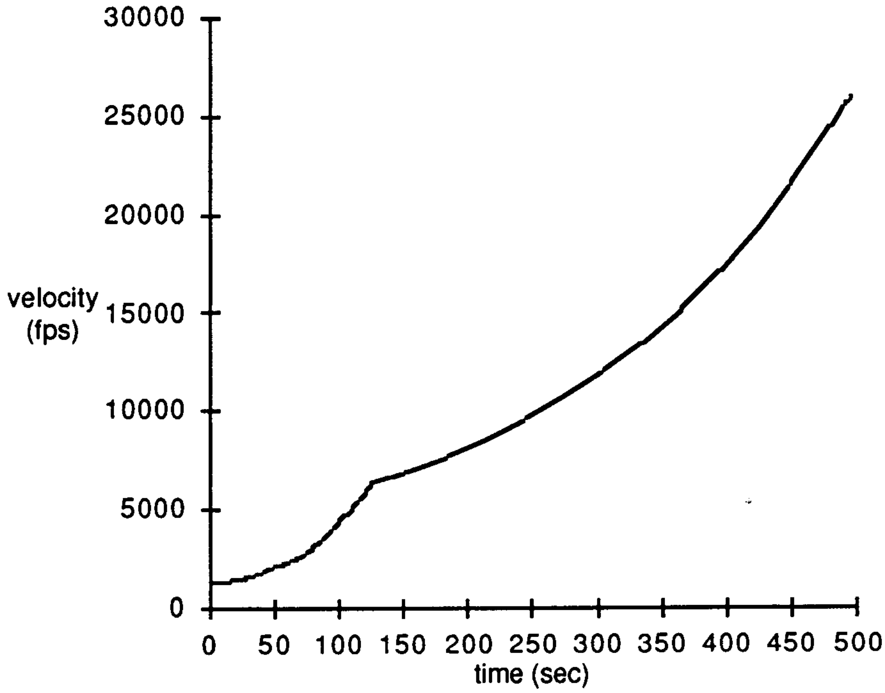


Figure 4.4.1-2 Velocity vs Time

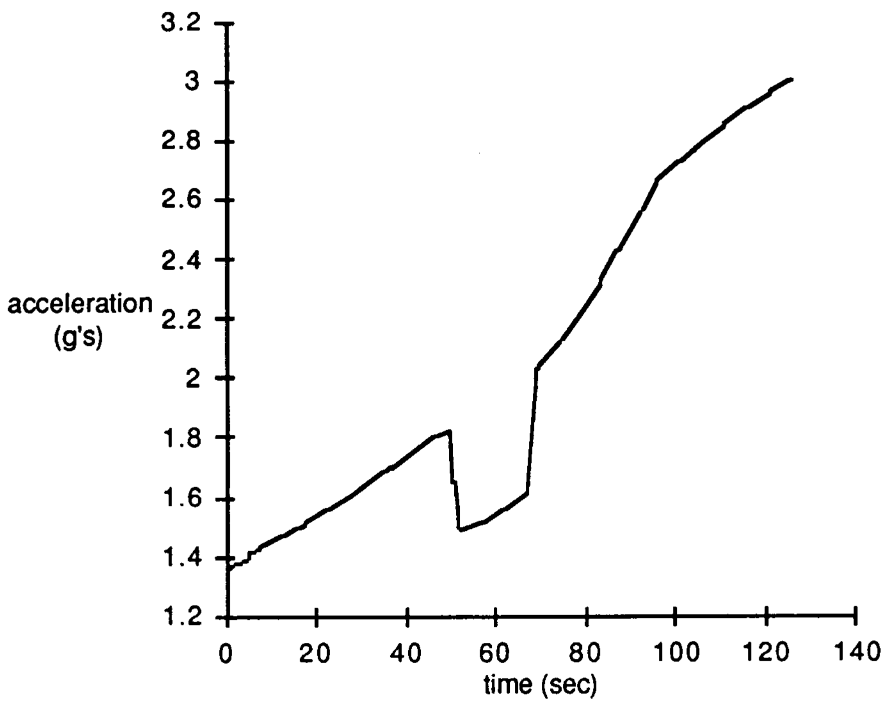


Figure 4.4.1-3 First Stage Acceleration vs Time

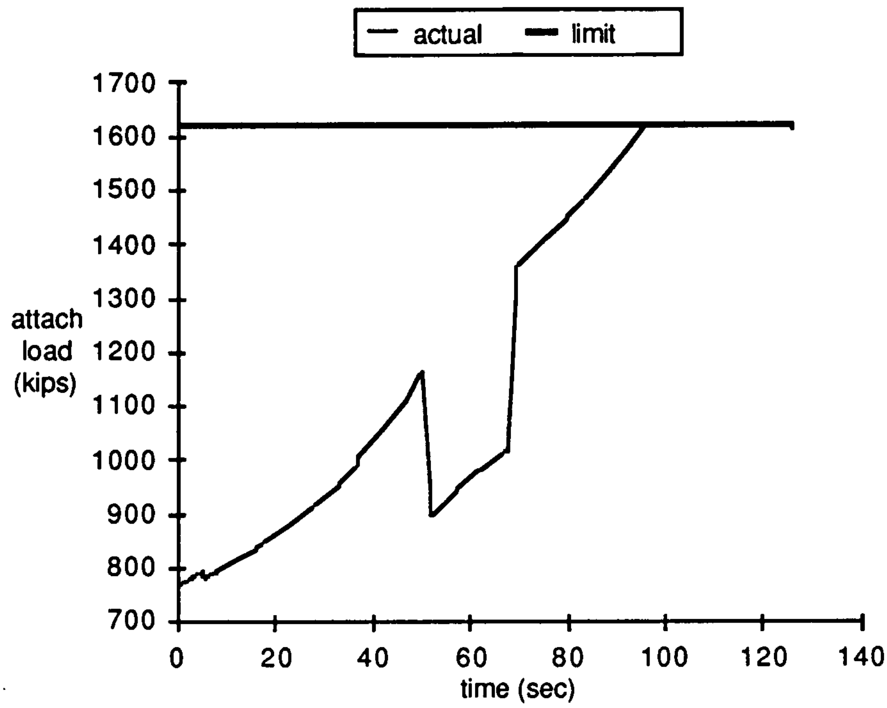


Figure 4.4.1-4 ET Attach load vs Time

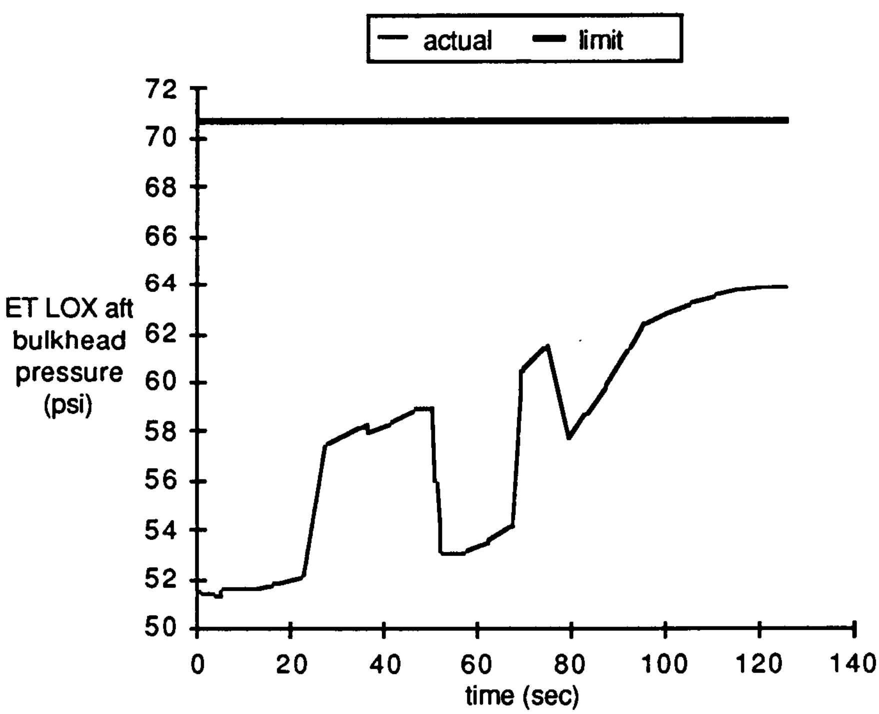


Figure 4.4.1-5 ET LOX aft bulkhead Pressure vs Time

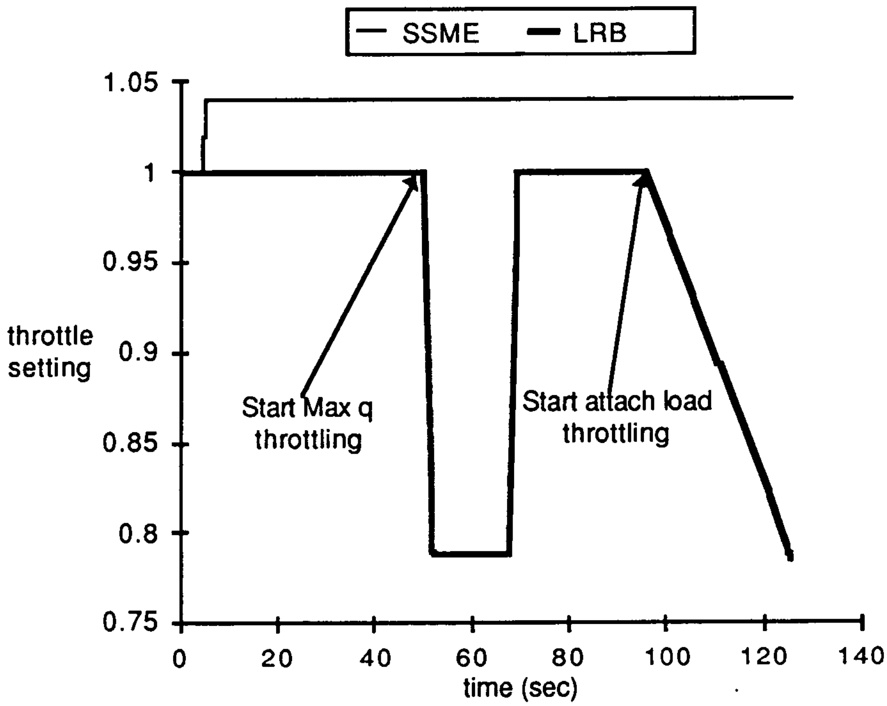


Figure 4.4.1-6 Throttle setting vs Time

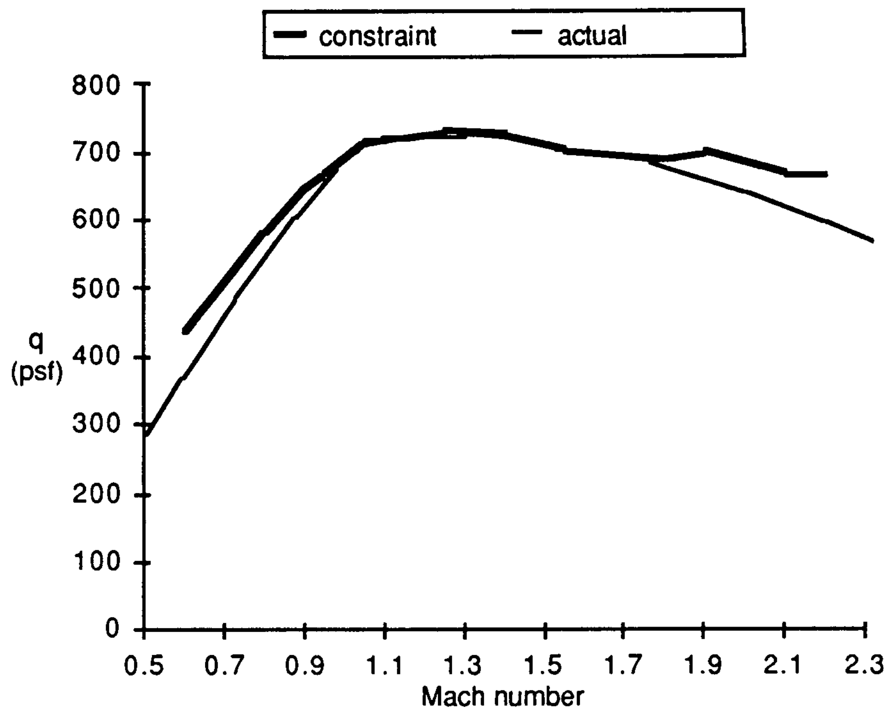


Figure 4.4.1-7 Dynamic pressure vs Mach number

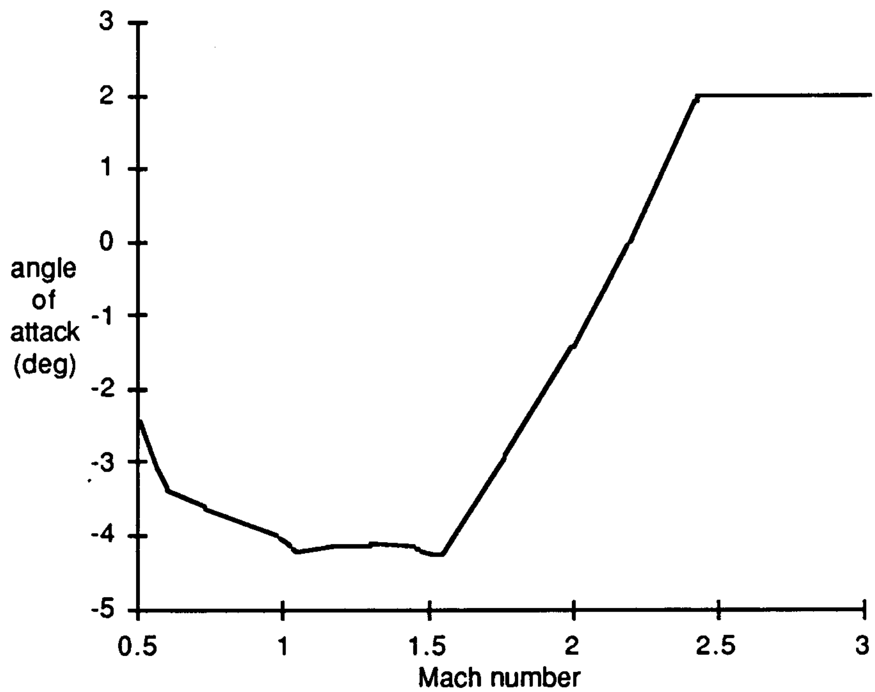


Figure 4.4.1-8 First Stage Angle of Attack vs Mach number

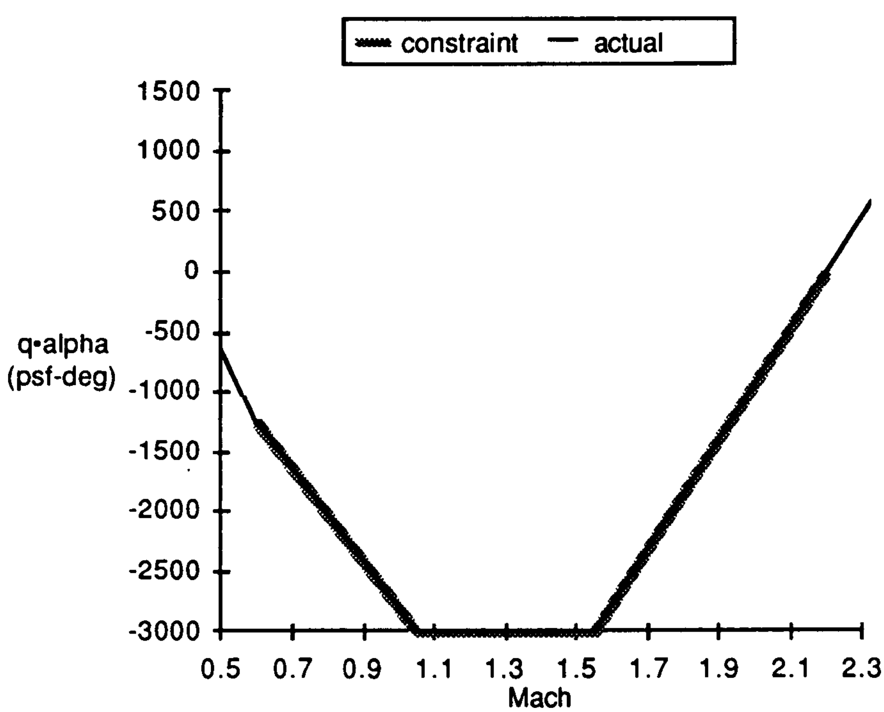


Figure 4.4.1-9 First Stage qα vs Mach

4.4.2 <u>ATO MISSION</u>. A description of the ATO mission trajectory simulation can be found in section 8.1.3. The ATO mission determined the required size of the RP-1 pump-fed LRB configuration. The RP-1 pump-fed LRB configuration's propellent, thrust, and structure were adjusted by the FASTPASS program until the desired performance was obtained.

The following table is a summary of the ATO performance for the RP-1 pump-fed configuration.

```
Lift off conditions:

   Weight (lb)                                     =   4,128,467.6085
   Payload (lb)                                    =   70,500.000000
   Thrust (lb)                                     =   5,081,365.6683
   Thrust to weight                                =   1.2308115626
   Initial inertial velocity (ft/sec)              =   1,342.4324022
   Launch site latitude                            =   28.307566153
   Launch site longitude                           =  -80.540959056


Max Q conditions:

   Max dynamic pressure (lb/ft**2)                 =   691.31565164
      Time (sec)                                   =   78.103153962
      Angle of attack (deg)                        =   4.3395521728
      Altitude (ft)                                =   33,464.143886
      Mach number                                  =   1.3557668344
      Q * ALPHA (deg-lb/ft**2)                     =   3,000.0003382

 LRB separation:

   Staging time (sec)                              =   144.12390364
   Altitude (ft)                                   =   139,645.96096
   Dynamic pressure (lb/ft**2)                     =   71.881788986
   Angle of attack (deg)                           =  -1.9999997282
   Mach number                                     =   4.8997274192
   Inertial velocity (ft/sec)                      =   6,427.3608541
   Inertial flight path angle (deg)                =   19.089253409
   Relative velocity (ft/sec)                      =   5,170.5380991
   Relative flight path angle (deg)                =   23.987397622
   Delta V (ft/sec)                                =   9,644.8976379
   Weight after separation (lb)                    =   1,457,995.4772
   Remaining ET propellent (lb)                    =   1,101,710.4850
   SSME throttle at separation                     =  0.99615164855
   Engine out LRB throttle                         =   1.0000004732
   Good LRB throttle                               =  0.75000035493
   Thrust (lb)                                     =   1,402,913.8586
   Thrust-to-weight after separation               =  0.96222099491
   Acceleration after separation                   =  0.94211890215
   LRB propellent used (lb)                        =   1,930,568.4570
   Engine out remaining prop. (lb)                 =   1,550.1491560
   Geodetic latitude (deg)                         =   28.485819223
```

Longitude (deg)	=	-79.996361638
Average back pressure (psi)	=	6.2034329859

MECO conditions:

Time (sec)	=	477.63116590
Altitude (ft)	=	358,023.39629
Inertial velocity (ft/sec)	=	25,626.149142
Inertial flight path angle (deg)	=	0.67723014703
Delta V (ft/sec)	=	30,161.674041
Shuttle & payload perigee (nm)	=	-19.420227758
Shuttle & payload apogee (nm)	=	78.729857459
MECO weight (lb)	=	356,284.99157
SSME throttle @ MECO	=	0.75866227807
SSME propellent weight used (lb)	=	1,587,950.0006
ET remaining propellent weight (lb)	=	-5.81878004596E-04
Average back pressure (psi)	=	1.8727502355

Throttle schedules:

Max q throttle down time (sec)	=	68.996645236
LRB throttle setting	=	1.0993491274
Post max q throttle up time (sec)	=	69.003153962
Start LRB 3g throttling (sec)	=	134.37649175
LRB throttle setting (engine out)	=	1.0000004732
LRB throttle setting (good LRB)	=	0.75000035493
SSME throttle @ separation	=	0.99615164855
Start SSME only 3g throttling (sec)	=	422.96253128
SSME final throttle setting	=	0.75866227807

Losses to LRB separation

Total delta V	=	9,644.8976379
Steering losses	=	2,250.8887097
Drag losses	=	421.95715164
Gravity losses	=	1,450.3365746
Pressure losses	=	436.42414354

Losses to MECO

Total delta V	=	30,161.674041
Steering losses	=	2,751.5209614
Drag losses	=	433.15952527
Gravity losses	=	2,257.5863754
Pressure losses	=	436.59568165

Min/Max conditions:

Max (+) angle of attack (deg)	=	16.215100719
Time (sec)	=	6.3000000000
Max (-) angle of attack (deg)	=	-9.2143173095

```
   Time (sec)                              =   25.093828170
Max (+) Q * Alpha (lbf-deg/ft**2)          =   3,000.0015122
   Time (sec)                              =   68.997226902
Max (-) Q * Alpha (lbf-deg/ft**2)          =  -930.16355463
   Time (sec)                              =   103.38925300
Max acceleration (g's)                     =   3.0000000000
   Time (sec)                              =   143.08734520
Max attach load (kips)                     =   1,565.6592762
   Time (sec)                              =   136.37649175
```

The following table is a summary of the RP-1 pump-fed configuration's mass properties obtained when sizing to the ATO mission.

LO2/RP1 PUMP-FED LRB (2)	SUBSYS	SYSTEM	GROUP	VEHICLE
STRUCTURE			58,996.3	
LH2 TANK (2219 skin stiffner)		12,692.2		
Cylinder section	9,518.8			
Bulk head	1,448.5			
ET Attach frame	1,724.9			
LO2 TANK (2219 skin stiffner)		22,386.1		
Cylinder section	18,640.1			
Bulk head	1,448.5			
ET Attach frame	2,297.5			
LO2 TANK SLOSH BAFFLES		227.5		
RP1 TANK INSULATION		0.0		
LO2 TANK INSULATION		638.5		
NOSE CAP		1,642.5		
FRWD ADAPTER		2,354.6		
INTERTANK ADAPTER		1,720.4		
AFT ADAPTER		11,532.5		
Aft adapter skin	6,673.8			
Aft adapter stringers	412.6			
Aft adapter frame	3,140.2			
Hold down posts	1,305.8			
THRUST STRUCTURE		5,691.9		
4 thrust beams	4,290.4			
4 longerons	533.3			
Engine mount bulk head	454.8			
Skirt aft frame	413.4			
LAUNCH GEAR		110.0		
PROPULSION SYSTEM			43,075.6	
MAIN ENGINES		27,708.2		
ENGINE GIMBAL SYSTEM		3,436.7		
ENGINE PURGE SYSTEM		922.0		
ENGINE MOUNTS		670.6		
MAIN PROPELLANT SYSTEM		10,338.0		
SUB-SYSTEMS			3,743.0	
SEPARATION SYSTEM		1,400.0		
AVIONICS		806.0		
POWER		1,537.0		
ELECTRICAL SYSTEM		0.0		
RECOVERY SYSTEM		0.0		
CONTINGENCY			10,581.5	
DRY WEIGHT				116,396.4
MAIN RESIDUALS			9,660.6	
RP1 FUEL		2,736.7		
LO2 FUEL		6,923.9		
INERT WEIGHT				126,057.0
ASCENT PROPELLANTS			966,059.3	
RP1 FUEL		273,671.2		
LO2 FUEL		692,388.1		

LRB LIFT OFF WEIGHT				1,092,116.3
MAIN START-UP FUEL			51,082.4	
RP1 FUEL		18,058.6		
LO2 FUEL		33,023.8		
STEP WEIGHT				1,143,198.7

TDDP NO: LRBRM-2, REF SODS NO: J-789 ADD 11

STS Weight Summary	SUBSYS	SYSTEM	GROUP	VEHICLE
Orbiter inert				193,693.0
OV 103 (7)			150,811.0	
SSME x 3 inert			20,958.0	
Buoyancy			80.0	
Crew Module			4,361.0	
Non-Prop. Consumables			5,397.0	
RCS Propellent			6,920.0	
Vented after SSME valve close			230.0	
MPS Propellent @ Ignition			4,936.0	
Orbiter lines - usable		2,782.0		
Orbiter lines - unusable		771.0		
SSME x 3 - unusable		1,383.0		
ET inert				76,892.0
ET dry weight			66,623.0	
ET Buoyancy			175.0	
MPS Pressurant			423.0	
Flight Press. Gas			3,730.0	
Usable propellent			5,046.0	
ET FPR		2,219.0		
BIAS		949.0		
Shutdown Propellent		1,878.0		
LH2	609.0			
LOX	1,269.0			
Unusable Propellent			895.0	
ET wet walls		175.0		
LH2 lines & tank, LOX lines		720.0		
Ascent propellent				1,587,950.0
LH2			225,590.0	
LOX			1,362,360.0	
OMS propellent				15,200.0
OMS Fuel			5,708.0	
OMS Oxidizer			9,492.0	
Payload weight				70,500.0

ENGINE PARAMETERS	NOMINAL	ABORT	MINIMUM
NUMBER	4.0		
WEIGHT	6,927.1		
THROTTLE	100.0	110.0	75.0
OXIDIZER FLOW RATE	1,457.4	1,603.1	1,093.0
FUEL FLOW RATE	576.0	633.6	432.0
VACUUM THRUST	635,015.0	698,516.5	476,261.2
SEA LEVEL THRUST	564,880.7	628,382.2	406,126.9
CHAMBER PRESSURE (psi)	1,272.7	1,400.0	954.55
VACUUM ISP (sec)	312.29	311.58	313.84
SEA LEVEL ISP (sec)	277.80	280.30	267.62
MIXTURE RATIO	2.5300		
NOZZLE AREA RATIO	16.000		
X-AREA (in^2)	4,772.3		
THROAT RADIUS (in)	9.7439		
EXIT DIAMETER (in)	77.951		
OVERALL LENGTH (in)	135.92		

VEHICLE PARAMETERS	
GLOW	4,128,467.6
T/W LIFTOFF (nominal)	1.3676
BOOSTER SL TOTAL (nominal)	4,519,045.5

Note: performance runs use parametric engine data. See the propulsion section for the engine point design.

ORB SL TOTAL (nominal)	1,127,200.9	
T/W LIFTOFF (1 LRB engine-out)	1.2000	
BOOSTER SL TOTAL (1 LRB engine-out)	3,700,158.8	
ORB SL TOTAL (1 LRB engine-out)	1,254,002.3	

DIMENSIONS/SHUTTLE COORDINATES	LNG.(FT)	STA.(IN)
FUEL TANK SPACING	2.9	
ENGINE CLEARANCE	3.8	
EXIT PLANE	2.8	2,511.6
AFT ADAPTER	17.5	2,477.6
AFT FUEL TANK	29.4	2,267.6
INTERTANK ADAPTER	13.4	1,915.1
FORWARD FUEL TANK	57.5	1,754.0
FORWARD ADAPTER	6.2	1,063.6
NOSE CAP	19.4	989.6
NOSE TIP	0.0	757.2
TOTAL LENGTH	146.21	
VEHICLE DIAMETER	14.567	
Length/Diameter	10.037	

NOSE-CAP GEOMETRY				
Nose fineness ratio	1.3300			
Nose bluntness ratio	0.20000			
Conic angle (deg)	17.654			
Nose length (ft)	19.374			
Nose cap spherical radius (ft)	1.5287			
Description	Radius	Height	Area	Weight
Nose cap	1.4567	1.0651	9.7481	
Conic section	7.2833	18.309	527.55	
Totals	0.00000	19.374	537.30	1,642.5

Aft Skirt Diameter Inputs/Results	
Nozzle Exit Plane Thickness (in)	5.0000
Nozzle Outsize Diameter (in)	87.951
Engine Gimbaling Length (in)	122.33
Maximum Gimbal Angle (deg)	6.0000
Gimbaling distance pad (in)	5.0000
Gimbaling distance (in)	17.305
Aft diameter (in)	271.41

Propellant tanks (Skin Stiffener)	Oxidizer	Fuel
Tank diameter	14.5	14.5
Material Density	0.10300	0.10300
Bulkhead		
Radius/Height	1.3784	1.3784
Wall thickness (in)	0.18000	0.18000
Length	5.3	5.3
Eccentricity	0.68825	0.68825
Surface area	271.3	271.3
Volume	577.4	577.4
Cylinder section		
Wall thickness (in)	0.48000	0.48000
Inside diameter (in)	173.84	173.84
Length	57.5	29.4
Surface area	2,618.2	1,337.0
Volume	9,482.3	4,842.3
Totals		
Total tank volume	10,637.2	5,997.1
Total surface area	3,160.8	1,879.6
Occupied volume	10,318.1	5,817.2
Propellent density	70.976	50.620
Total propellent	732,335.8	294,466.5
Ullage %	3.0	3.0

Figures 4.4.2-1 thru 4.4.2-9 show various performance parameters obtained from the RP-1 pump-fed LRB configuration's ATO trajectory simulation.

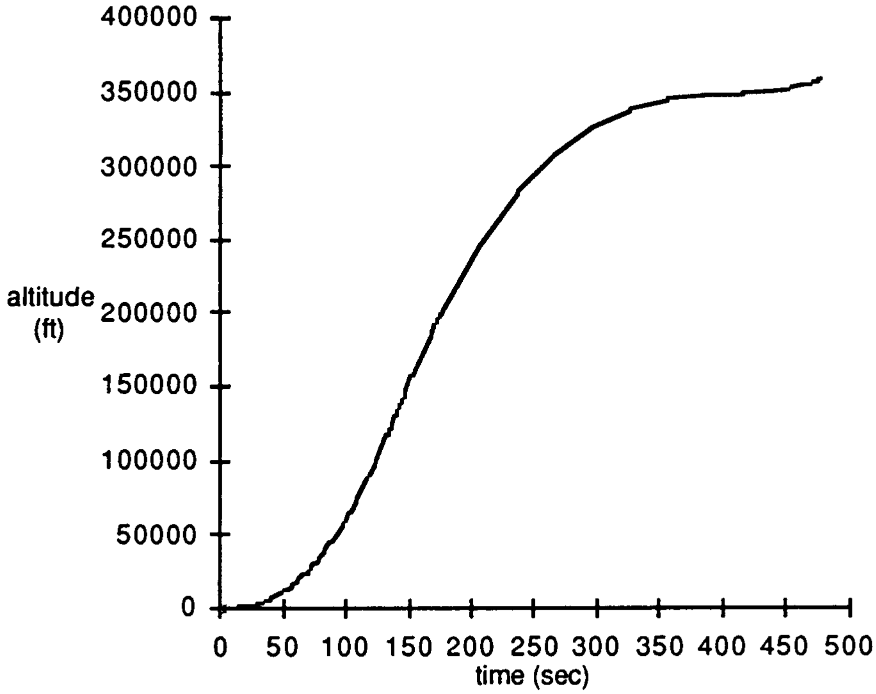


Figure 4.4.2-1 Altitude vs Time

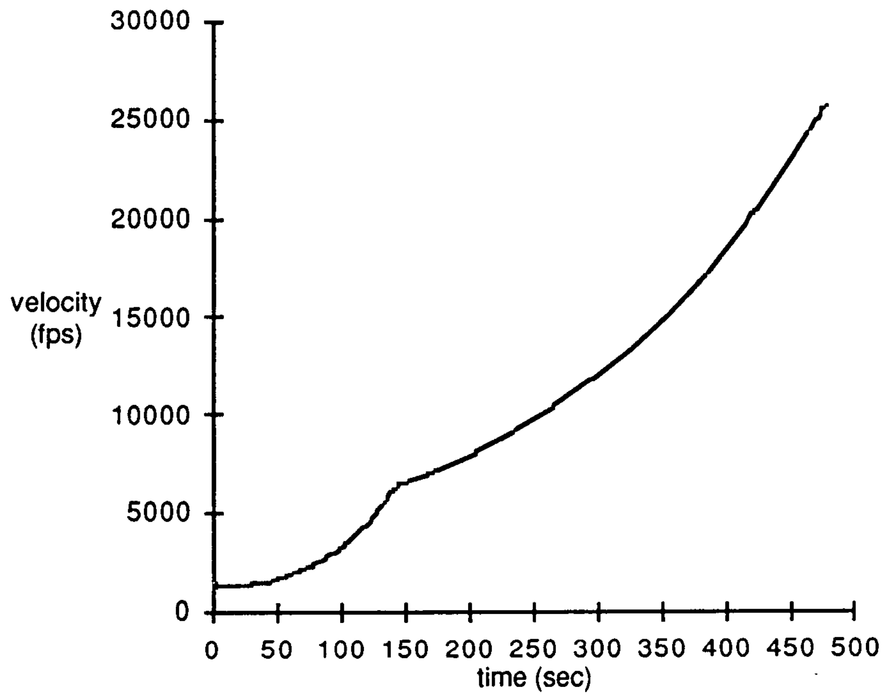


Figure 4.4.2-2 Velocity vs Time

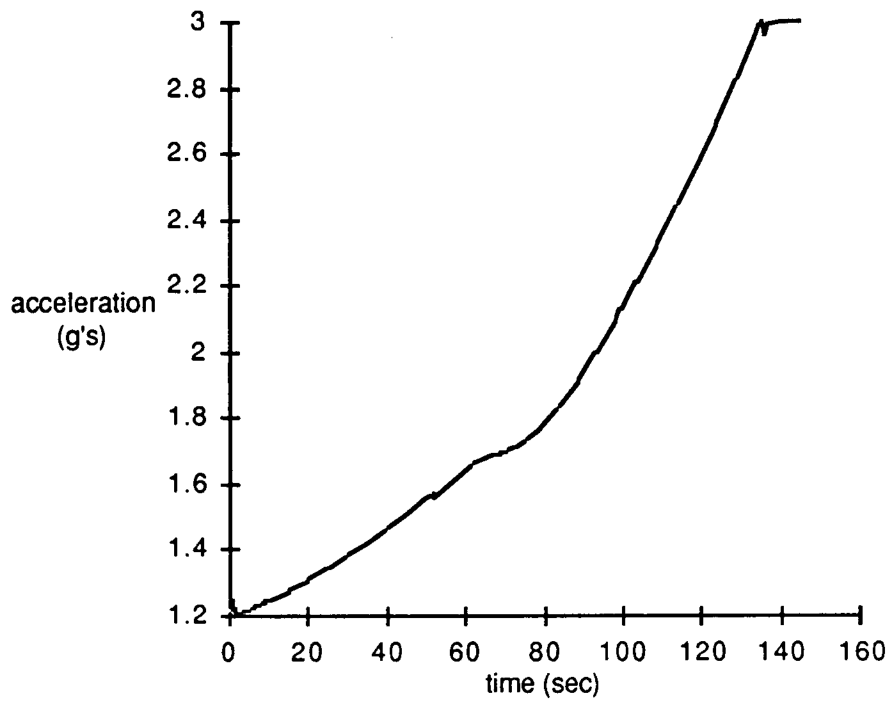


Figure 4.4.2-3 First Stage Acceleration vs Time

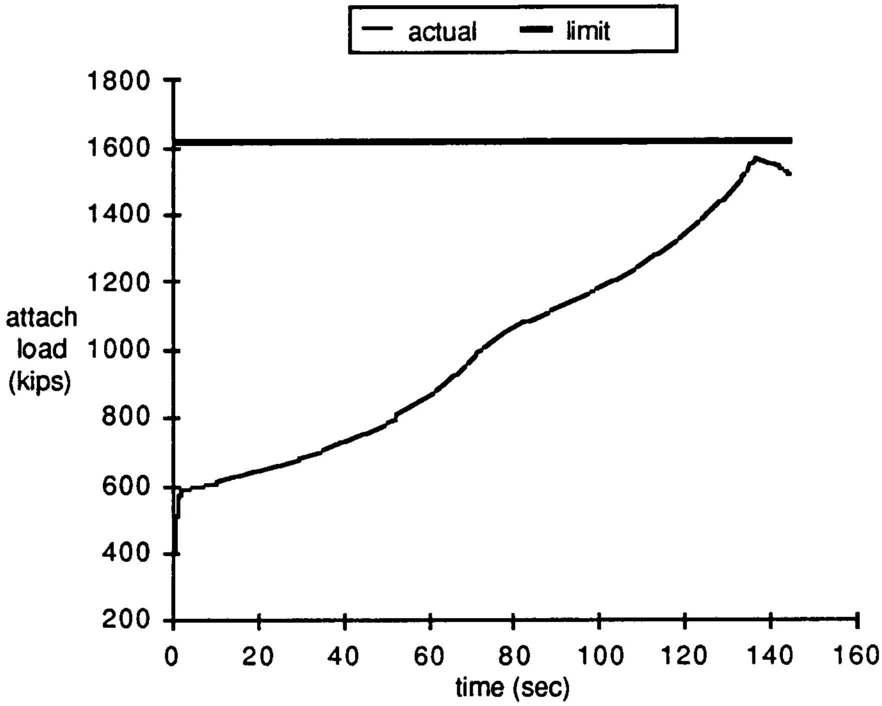


Figure 4.4.2-4 ET Attach load vs Time

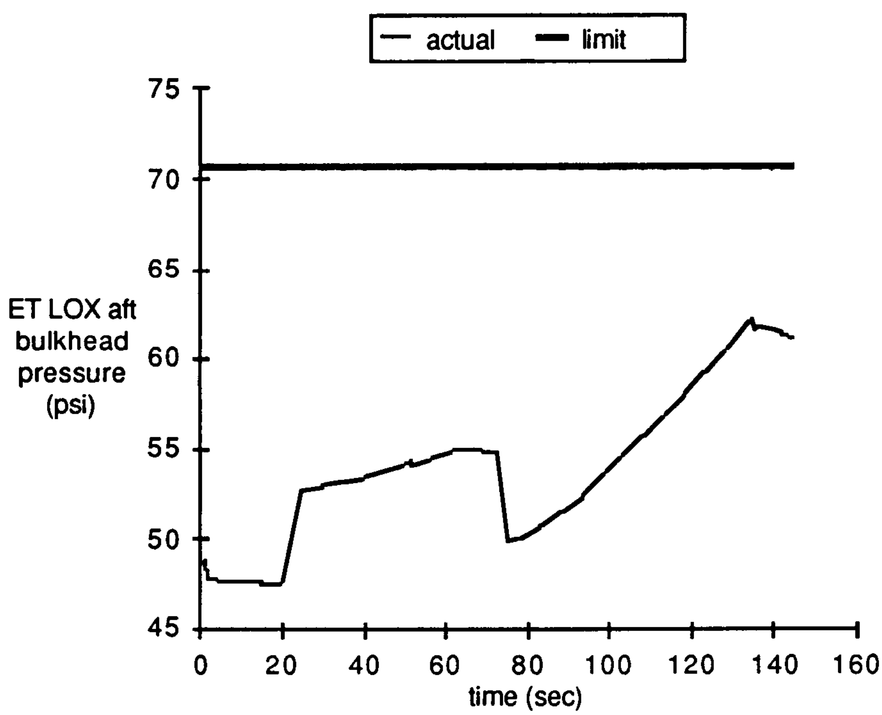


Figure 4.4.2-5 ET LOX aft bulkhead Pressure vs Time

C-3

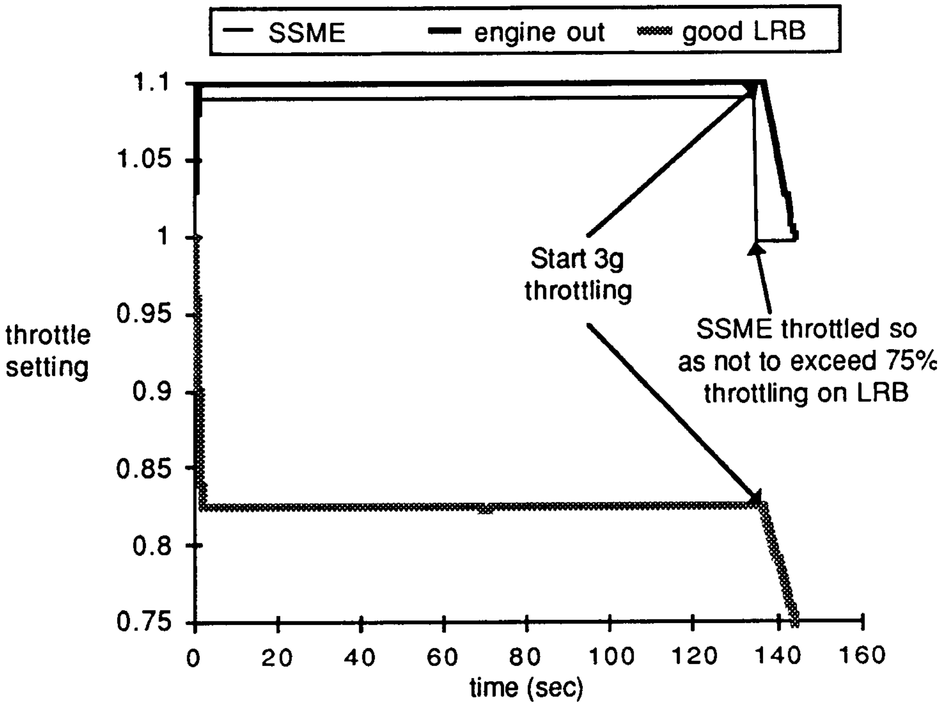


Figure 4.4.2-6 Throttle setting vs Time

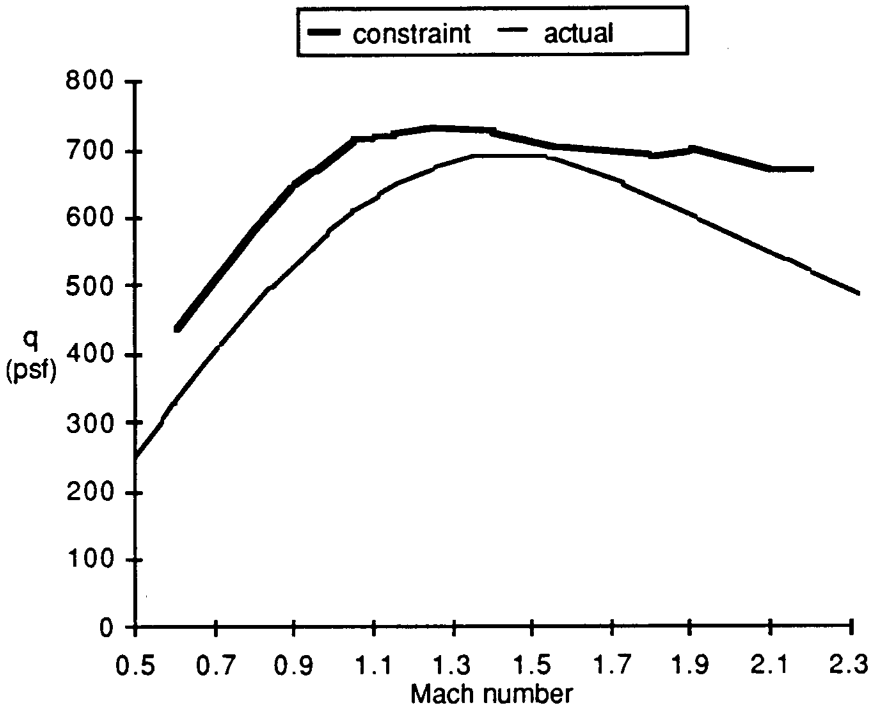


Figure 4.4.2-7 Dynamic pressure vs Mach number

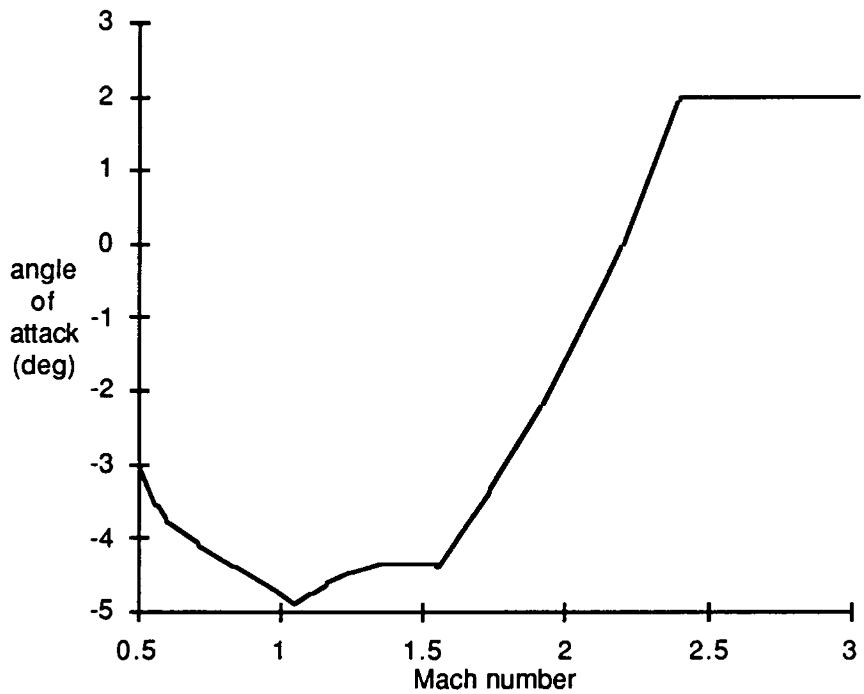


Figure 4.4.2-8 First Stage Angle of Attack vs Mach number

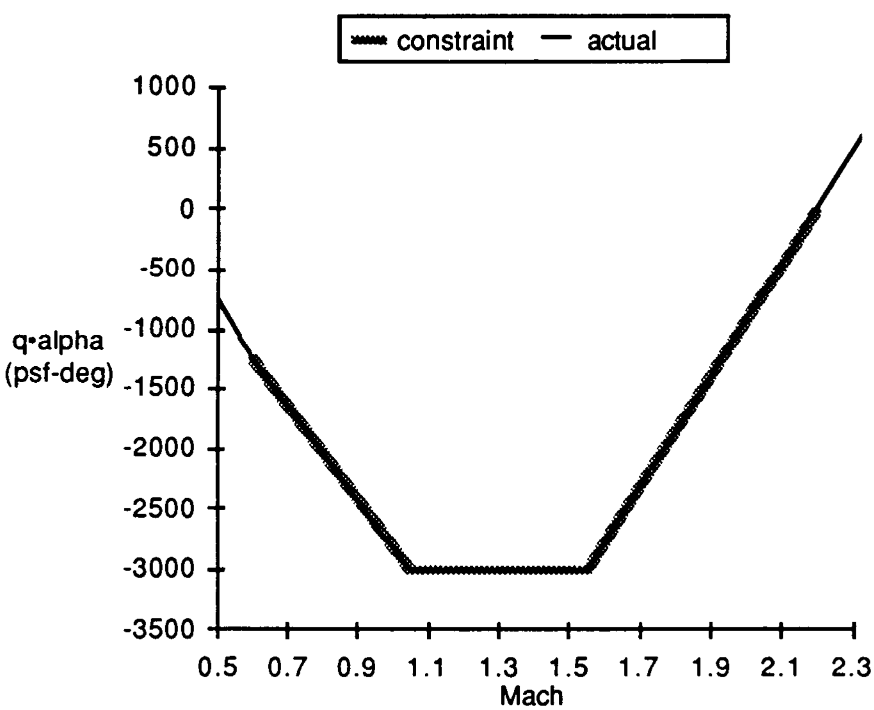


Figure 4.4.2-9 First Stage qα vs Mach

Sizing the RP1 pump fed LRB to meet the minimum ATO conditions with a single LRB engine out at lift off is not without its penalties. The penalties involved in sizing to meet ATO mission requirements are shown in the following table.

	ATO sizing	Nominal Sizing	Δ	Δ%
LRB Length (ft)	146.2	145.8	0.4	0.2
LRB Diameter (ft)	14.57	14.29	0.28	2.0
Dry weight (Klbs)	116.4	113.9	2.5	2.2
Ascent Propellent (Klbs)	966.1	930.1	36.0	3.9
LRB GLOW (Klbs)	1,092	1,053	39	3.7
LRB Vacuum Thrust (Klbs)	635.1	618.6	16.5	2.7

SECTION 5

LO2/LH2 PUMP-FED LRB CONCEPT

The LO2/LH2 pump-fed configuration was recommended as the best final concept for the LRB. This concept offers low technical risks, minimal environmental impacts, propellant commonality with the current STS, and more importantly, commonality with ALS concepts. The LO2/LH2 LRB Life Cycle Cost can significantly be reduced by the ALS sharing of DDT&E costs in engine or booster development, and production rate effects.

The engine selected as baseline for this vehicle concept is the LO2/LH2 gas-generator cycle engine. As an alternative which seems promising in reducing costs and improving reliability, the split expander cycle engine, was studied in parallel with the gas-generator engine. However, this engine cycle needs further technology demonstration. Both the gas generator and split-expander engine concepts result in the same size vehicles, and have the same interface conditions.

5.1 STRUCTURES AND MECHANISMS

5.1.1 VEHICLE CONFIGURATION. The geometry and STS interface of the pump fed LO2/LH2 Liquid Rocket Booster as shown in Figure 5.1.1-1. It has an overall length of 178.1 feet and an 18 foot diameter. The LO2 tank total length is 40.1 feet, having a cylindrical section 27.1 feet long capped by two elliptical bulkheads. The LO2 tank is attached to the forward adapter at is forward end and is connected to an intertank adapter aft. The LH2 tank total length is 102.1 feet having a cylindrical section 89.1 feet long capped by two elliptical bulkheads. The LH2 tank is attached to the intertank adapter at its forward end and is connected to the aft skirt at its other end. The configuration was sized such that the upper ET attachment is located in the intertank adapter thus eliminating the need for an in-tank attachment. The lower ET attachment is inside the LH2 tank.

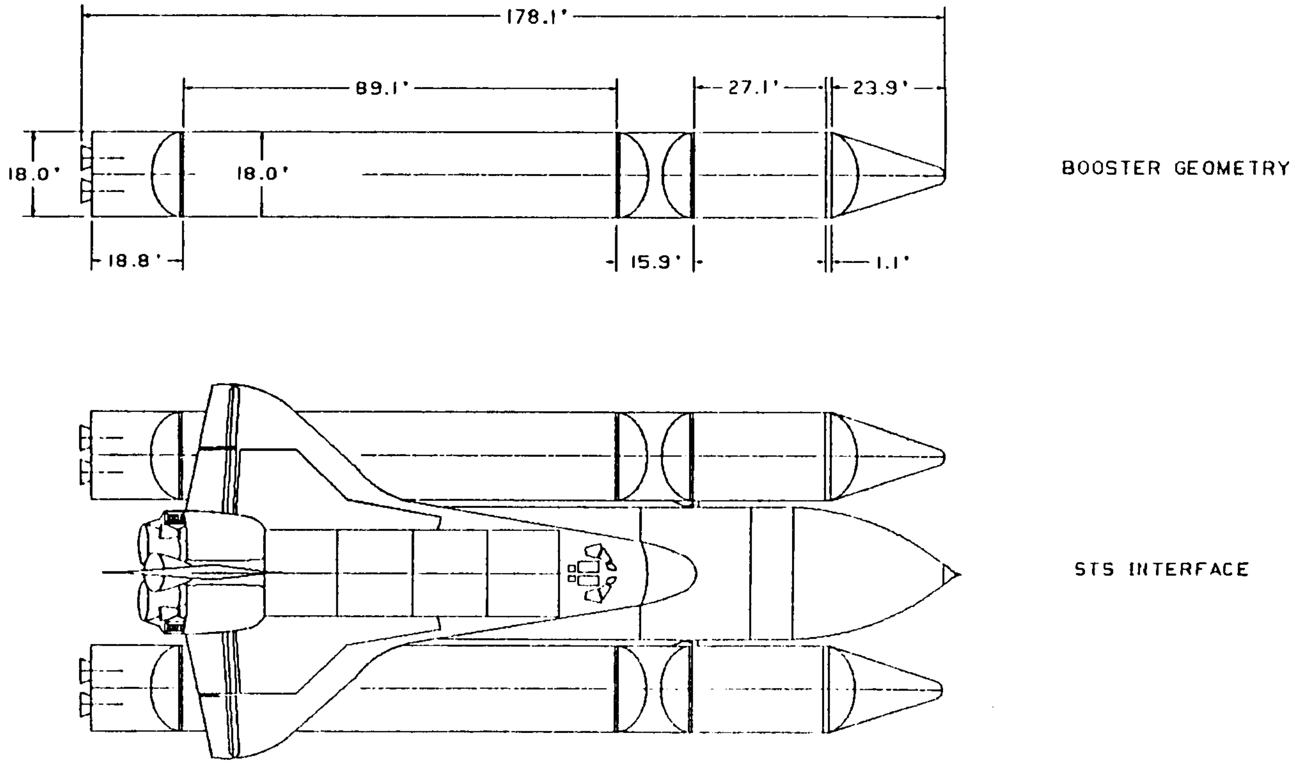


Figure 5.1.1-1 LO2/LH2 Pump Fed LRB with Shuttle and External Tank

Location of the main propellant feedline was selected such that the 180° surface of the LRB that is adjacent to the Orbiter wing is smooth and has no protrusions. This will minimize aerodynamic loadings on the Orbiter wings. Ground interface attachments at the end of the aft skirt, and the general engine arrangement are shown in Figure 5.1.1-2. The LO2 main feedline penetrates the intertank adapter and remaining external to the LH2 tank until it penetrates the aft skirt. Once internal to the aft skirt, the LO2 main propellant line culminates into a two legged manifold as shown in Figure 5.1.1-3. Each manifold supplies two engines via direct engine feed lines. Fuel is provided to each engine directly from the aft LH2 tank via four feedlines. The engines mounted flex bellows for each propellant line allow for the thermal expansion and gimbaling.

Tanks. Both the LO2 and LH2 tanks are similar in the design and construction. The tanks are made entirely of 2219 aluminum alloy and use variable polarity plasma arc (VPPA) welding to join the structural components. The skin panels of the tank cylindrical portion are 2219-T8510 extrusions, with integral T (TEE) stringers. The extruded skin panel are machined to the required thickness. Locally at the splices the skin is thicker for welding. Figure 5.1.1-4 shows typical tank construction that applies for both the LO2/LH2 and LO2/RP-1 pump fed vehicles. The T stringer flanges provide a mating surface for attaching the ring frames with mechanical fasteners. The ring frames are equally spaced along the cylindrical section to stabilize the skin/stringer.

The end domes are 2219-T62 aluminum alloy and are a single piece spun formed part. Access provisions, propellant feed lines, pressurization lines and vent lines are located in the domes. The domes are machined to reduce weight, but are thicker in the edge weld land and for the above provisions.

A 2219 aluminum alloy roll ring forging is machined to a Y shape to join the dome to the cylindrical tank section. One leg provides a member to attach the intertank forward adaptors, or the aft skirt. An integral internal upstanding flange is machined into the forging. This provides a flange to attach a ring frame, to react loads normal to the tank skin. Mechanical fasteners attach the ring frame to the flange, but no fastener are in or through the tank skin.

Skin/stringer/frame construction provides a fail safe tank structure. Should a failure occur in the skin due to internal tank pressure, the tank will leak and not explode. A tank structure with redundant load paths provides this safety, and it is not provided with a monocoque structure.

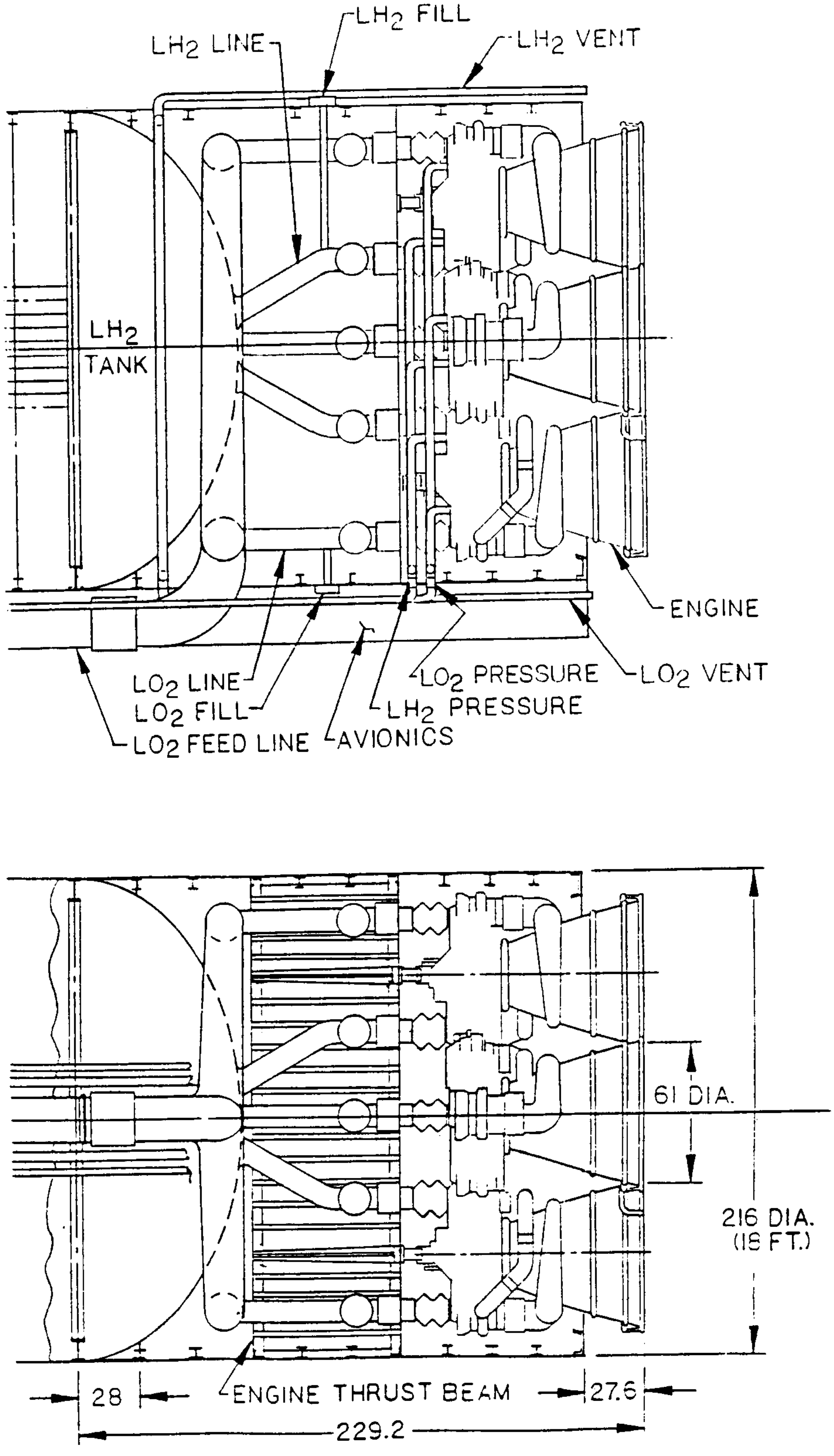


Figure 5.1.1-2 Engine Arrangement

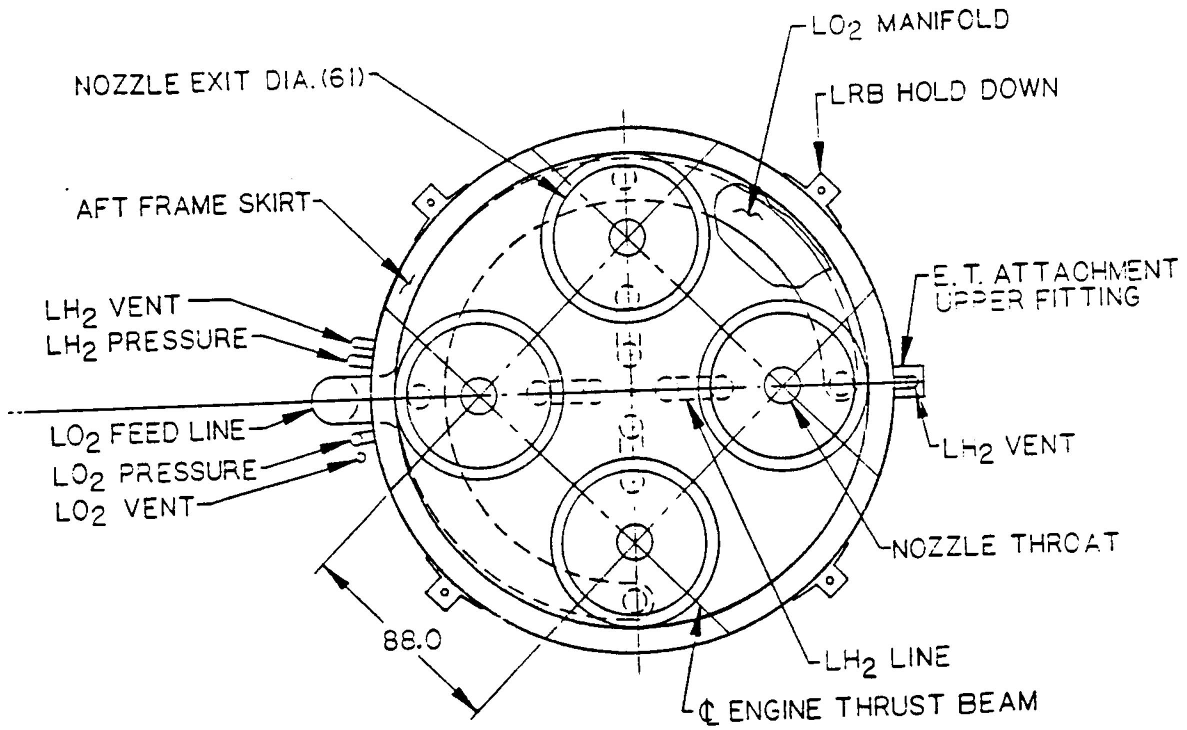
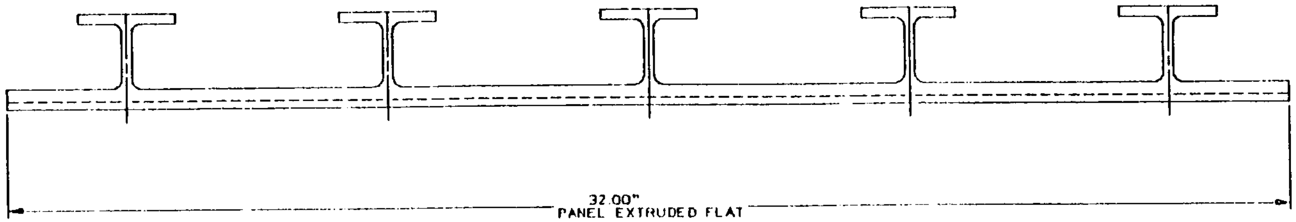


Figure 5.1.1-3 Propellant Feed Line Distribution to Each Engine

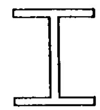
Forward ET Attachment. The forward attachment location with respect to the external tank is approximately mid length of the intertank adaptor. The attachment fitting, Figure 5.1.1-5a and 5.1.1-5b, is machined from forged 2219-T852 aluminum alloy. The LRB forward fitting picks up the ET fitting and is welded to the internal LRB, intertank adaptor, and frame web. The internal frame web and stiffeners are mechanically fastened to an integral stiffeners in the skin, as shown in Figure 5.1.1-6. The skin is thicker locally at the integral stiffener location, and is the outer cap material of the frame, as shown in the frame cross section view Figure 5.1.1-7. The web and the stiffeners of the frame use mechanical fasteners as shown in the web cross section view Figure 5.1.1-8. The internal web decreases in thickness outside of the 100.8 degree arc as shear load stresses decrease.

Nose Cone. The nose cone is a semi-monocoque structure consisting of skin, longerons, frames and a nose cap. Figure 5.1.1-9 shows the nose cone structure. The skin is 0.080 inch thick and rolled to a cone shape. There are eleven circular frames made from extruded I sections. The four longerons are extruded T sections and are machined to vary the cross section as required along the length. The nose cap is a single piece and is spun to the half sphere shape.

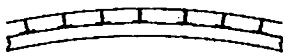
The separation motors are located in the are supported by the nose cone structure. The nose cone and nose cap are attached by screws. This allows removal and reinstallation.



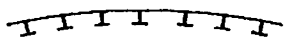
DETAIL C



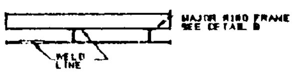
RING FRAME
DETAIL B



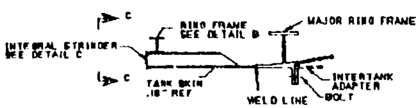
SECTION AT
RING FRAME



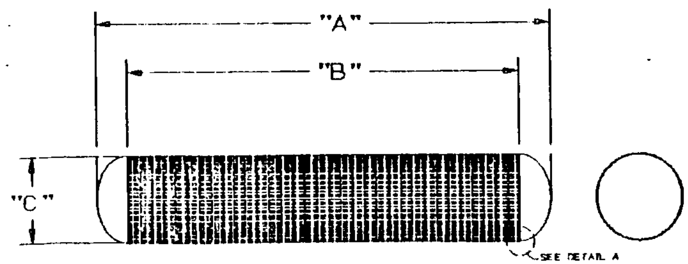
SECTION MIDWAY
BETWEEN RING FRAMES



SECTION C-C



DETAIL A



LRB CONFIGURATION	FUEL TANK "A"	FUEL TANK "B"	FUEL TANK "C"	LO2 TANK "A"	LO2 TANK "B"	LO2 TANK "C"
LO2 / RP-1	39.98'	29.4'	14.6'	68.08'	57.5'	14.6'
LO2 / LH2	102.1'	89.1'	18.0'	40.1'	27.1'	18.0'

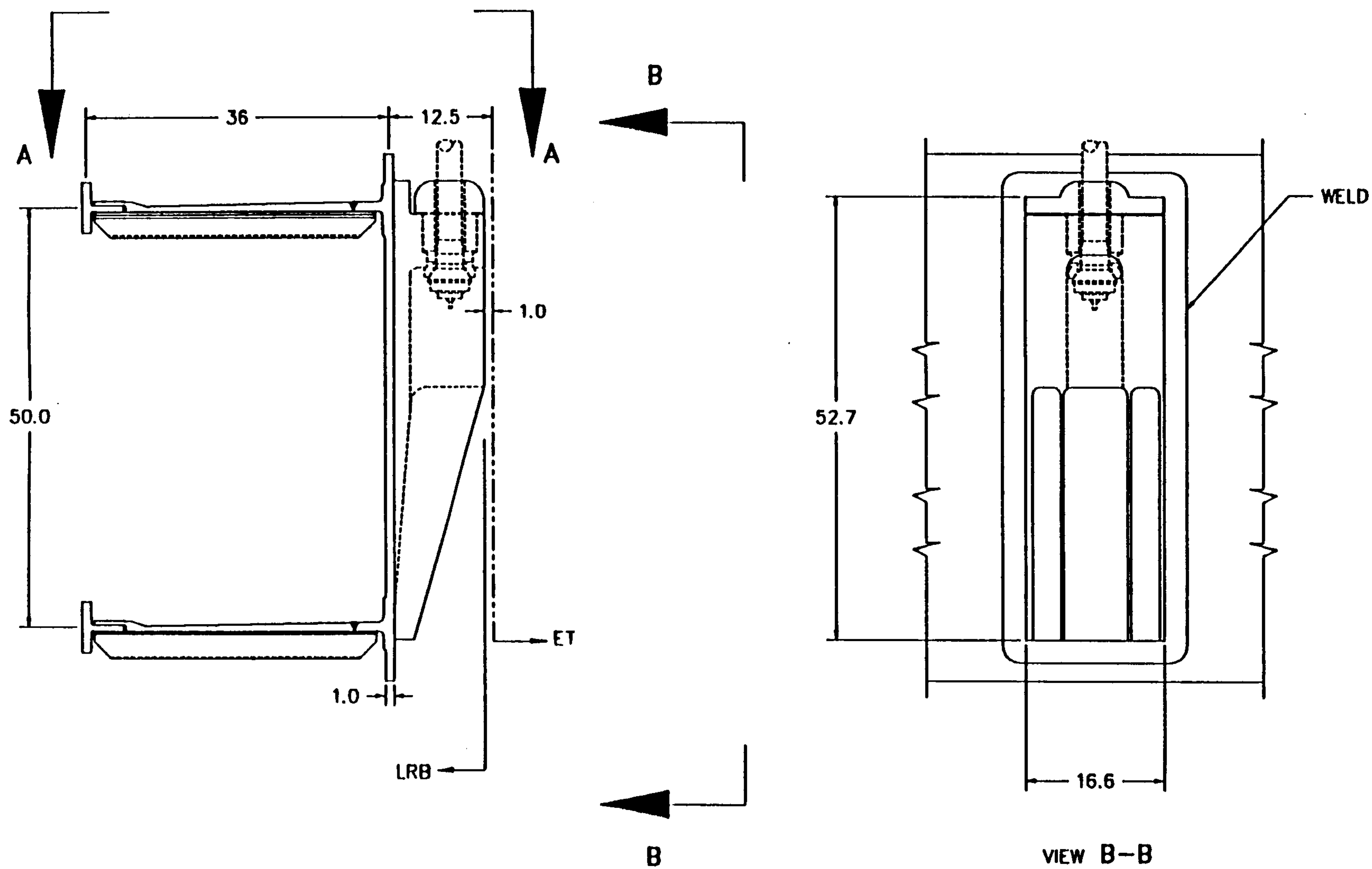


Figure 5.1.1-5a Forward ET Attachment Assembly (Side View)

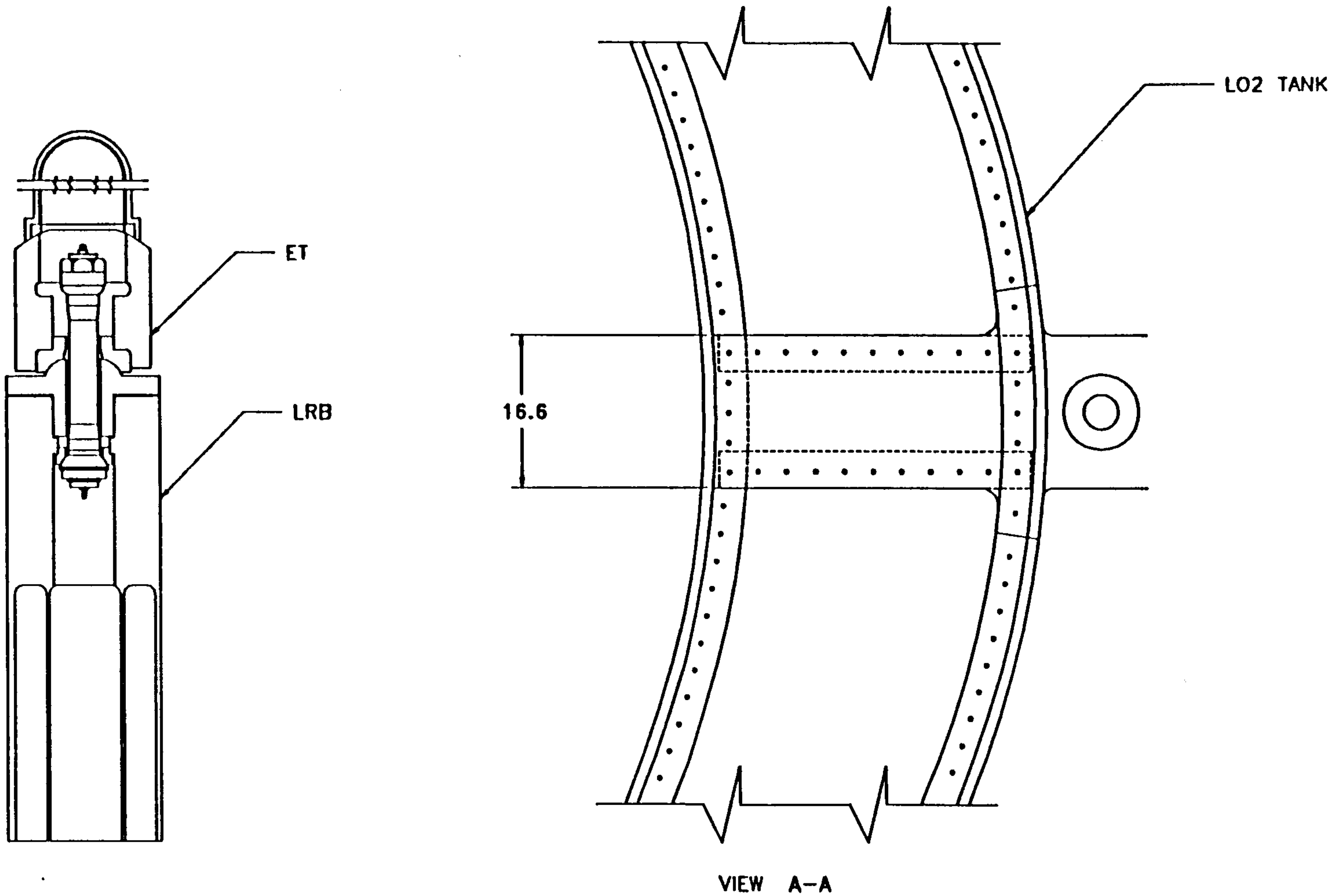


Figure 5.1.1-5b Forward ET Attachment Assembly (Front View)

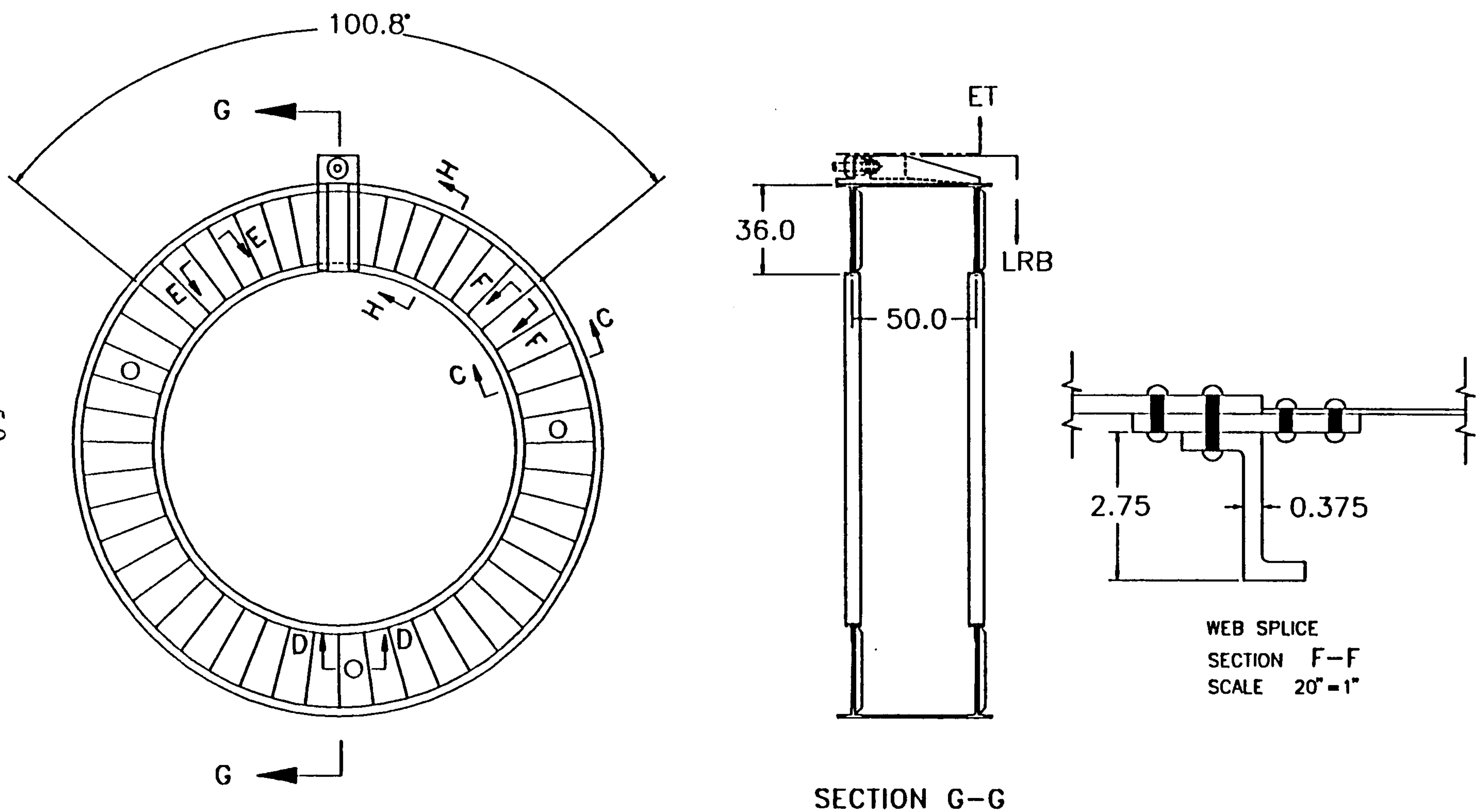


Figure 5.1.1-6 Forward ET/LRB Attachment Internal Frame

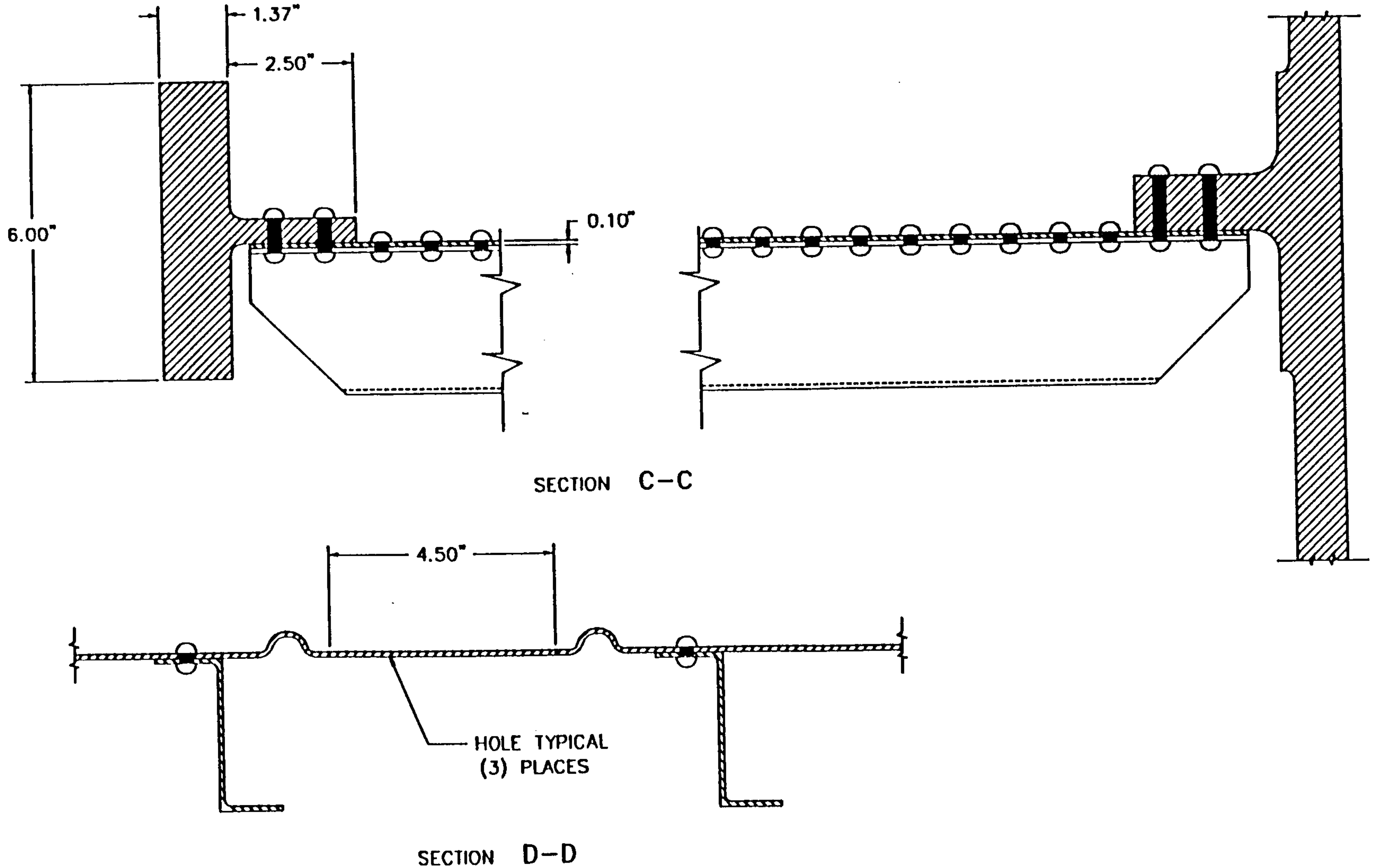


Figure 5.1.1-7. Forward ET/LRB Attachment Internal Frame

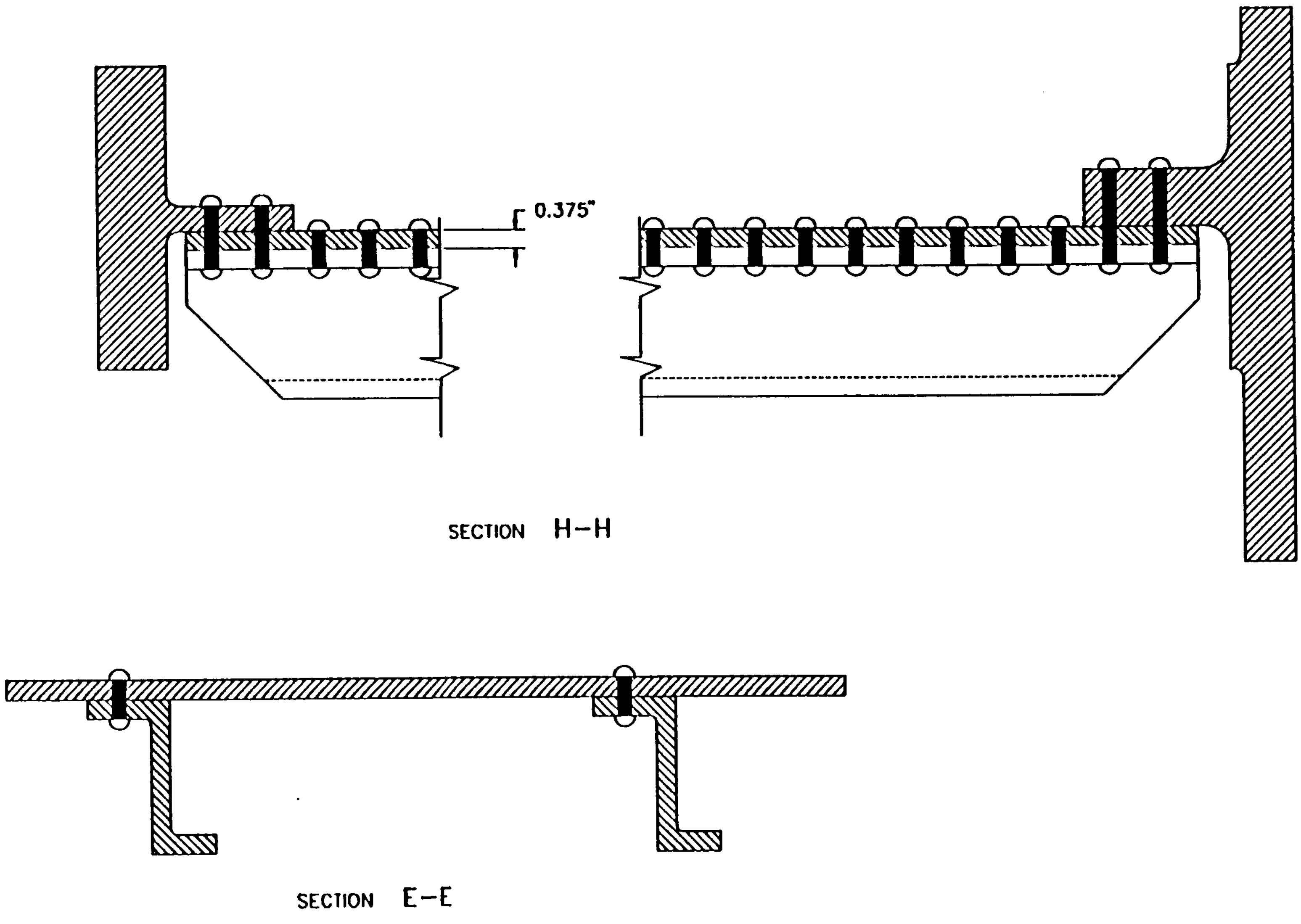


Figure 5.1.1-8 Forward ET/LRB Attachment Internal Frame Construction (Cross Section of Web)

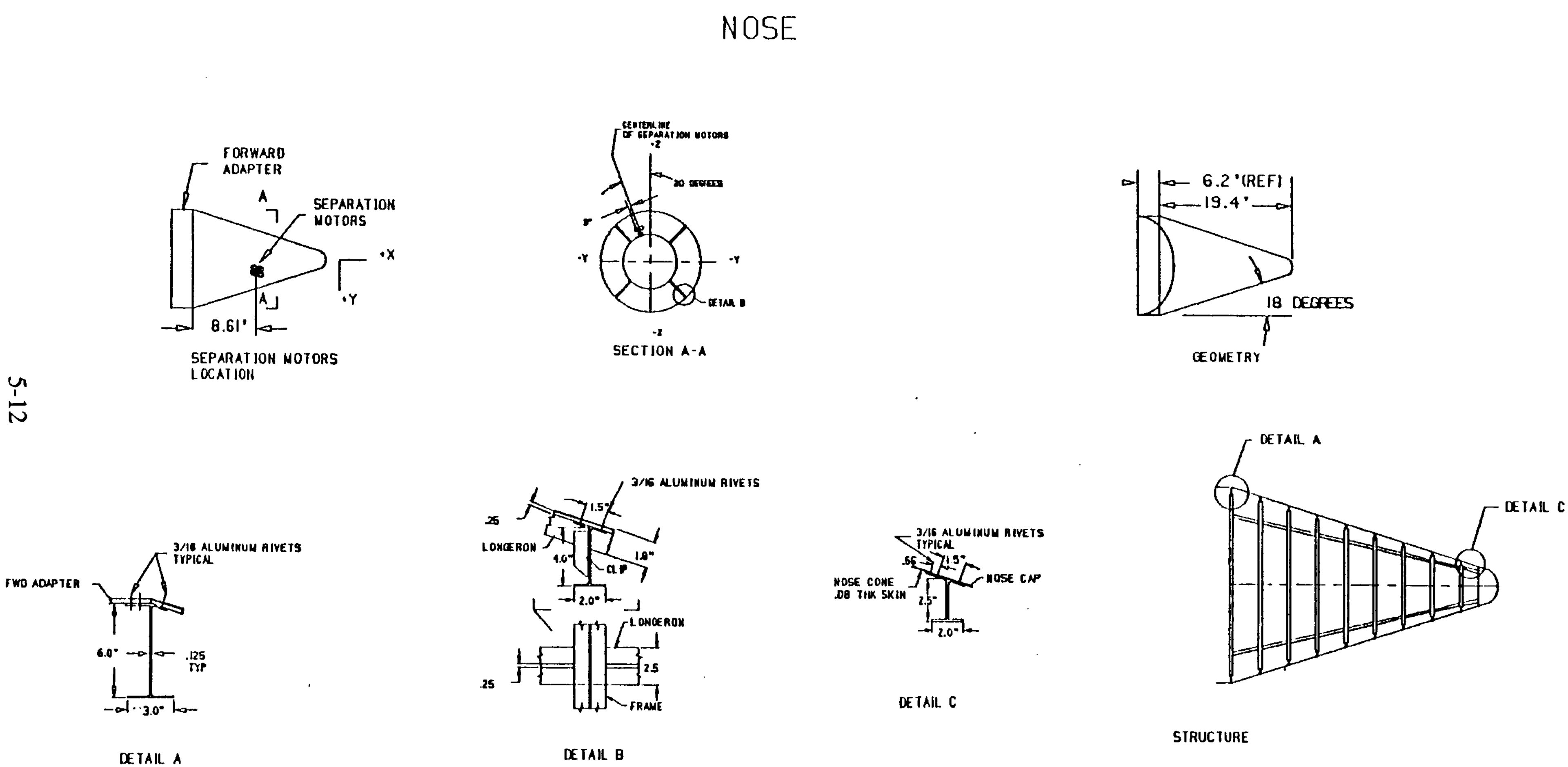


Figure 5.1.1-9 Nose Cone Construction Details

Intertank Adaptor. The intertank adapter skin is 2024-T851 aluminum alloy, rolled to a cylindrical shape. The 2024 aluminum alloy stringers are attached to the skin with mechanical fasteners. The external tank upper attachment fitting is located in the intertank adaptor section. Figure 5.1.1-10 shows the attachment fitting and the two (2) frames that react the kick loads from the fitting. The above frames have extruded T (TEE) caps and conventional web and stiffners. Mechanical fasteners are used to join the frame parts and attach it to the skin. The frame caps, web and stiffners are 2024 aluminum alloy. The intertank adaptor is 142.8 inches long and is bolted (spliced) to the tank skirts. An access door is provided for entering the intertank adaptor area. This is provided for maintenance and installing equipment.

Aft Skirt. The aft skirt is a constant 18 foot diameter, the same as the vehicle. The skirt is 173.6 inches long and is spliced to the LH2 tank skirt. The skirt is shown in Figures 5.1.1-2 and 5.1.1-3. The skirt skin is 2024-T851 aluminum alloy plate rolled to a cylindrical shape. Extruded 2024-T8510 aluminum alloy I section ring frames are used to stabilize the skin. There are four (4) hold down fittings and they are located so two (2) fittings will be in tension, due to bend over loads when firing the Orbiter SSME engines. Each hold down fitting extends the length of the aft skirt, similar to an external longeron. The hold bolt in each fitting is located externally at the lower edge of skirt and is clocked around the skirt in 90° increments. Orientation of the vehicle on the MLP is shown in Figure 5.1.1-11.

There are four (4) engine thrust beams to transmit the engine loads to the skirt skin. The beams form a square pattern, as shown in Figure 5.1.1-2. The beam depth is large in relation to its length, so engine deflections will be small. The engine attach thrust fitting runs the depth of the beam and there are two (2) per beam. The upper and lower beam caps are extruded T members. The beam has web and stiffners between the thrust fitting and the aft skirt skin. Between thrust fittings, truss members are used to stabilize the beam caps and carry shear load. At each beam end, at the skirt skin, a member runs the length of the skirt structure. This is a longeron which distributes the beam loads (engine thrust) to the skirt skin. The skirt has twelve (12) longerons, eight (8) at beam ends and four (4) at the hold down fittings.

5.1.2 SEPARATION SYSTEM. Efforts to define the separation system were conducted during the initial LRB study phase. During the follow-on extension, separation analyses were not updated to reflect resizing of the LOX/LO2 pump-fed booster. Thus, the results

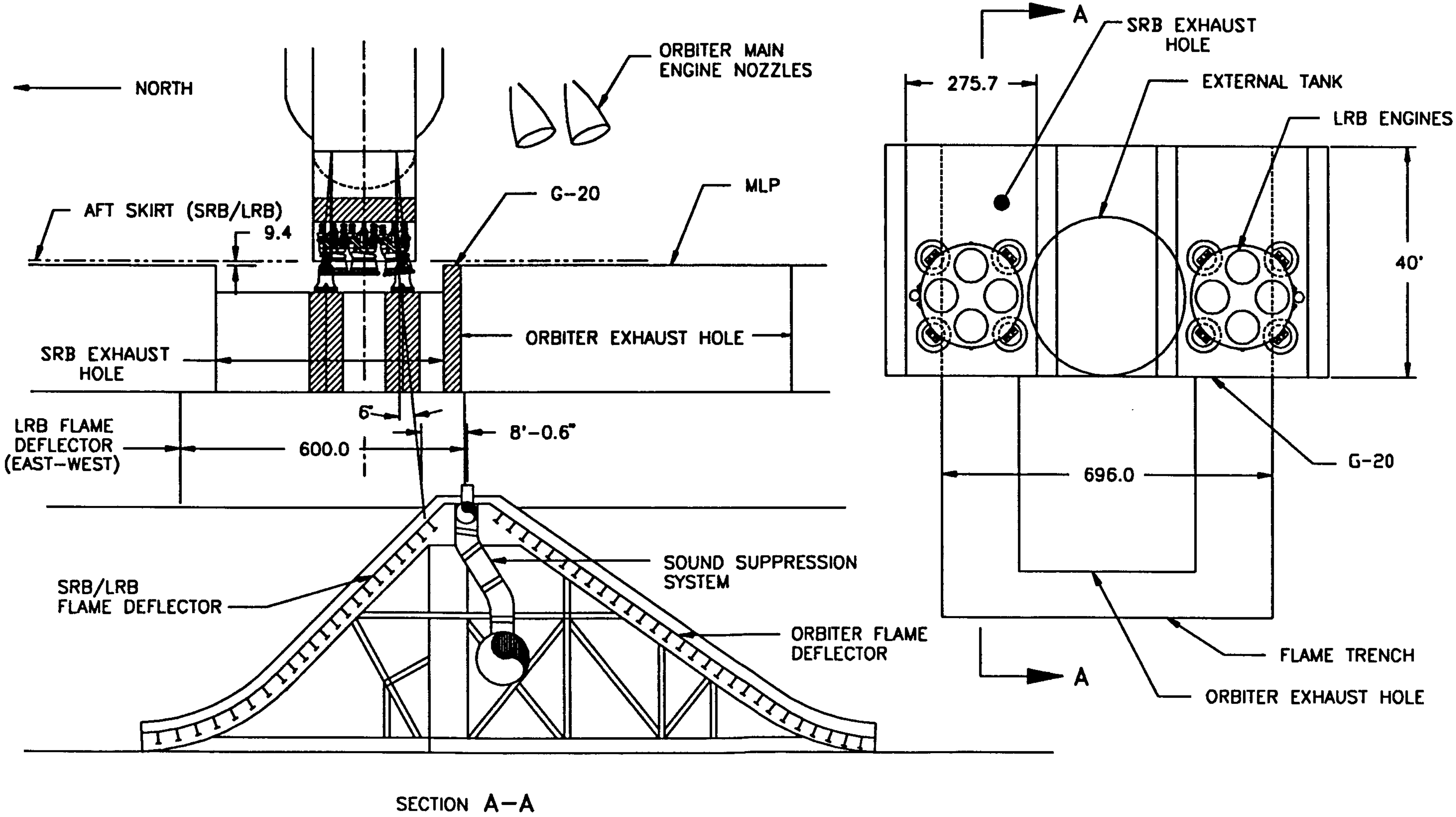


Figure 5.1.1-11 LRB Orientation on MLP

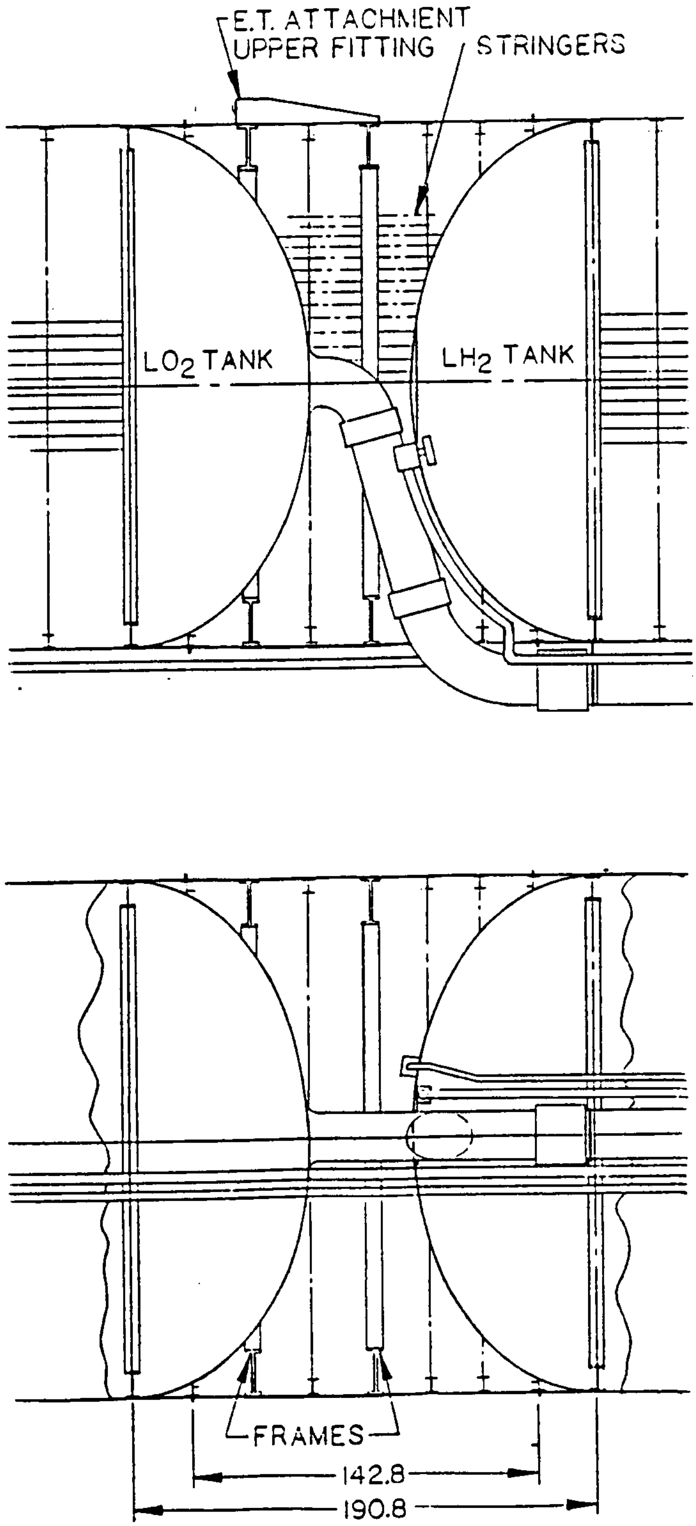


Figure 5.1.1-10 Intertank Adapter Construction Details

which follow require update and are presented primarily to show trends and typical designs.

The separation system for the LOX/LH2 pump-fed configuration employs the same basic BSM system as the LOX/RP-1 pump-fed booster (see section 4.1.2). The LO2/LH2 booster approximately the same weight at separation as the LOX/RP-1 pump-fed booster, but has geometry and aerodynamics more akin to the pressure-fed booster. For nominal ascent separation, the lightness of the booster tends to offset increased separation aerodynamic forces.

The LOX/LH2 pump-fed booster separation system has been initially sized for nominal ascent staging. As the LOX/LH2 pump-fed booster design matures, RTLS abort coverage capabilities will be more thoroughly examined; if results show that it is possible to conduct a RTLS abort prior to nominal staging, simulations will be conducted to determine if the BSM quantity used for nominal ascent staging is sufficient for RTLS abort needs. If not, the number of BSMs will be increased accordingly.

Nominal ascent separation of the LOX/LH2 pump-fed booster is designed to occur at initial conditions of:

Mission Elapsed Time	= 153.4 Seconds
Altitude	= 181,000 Ft
Mach	= 5.35
Dynamic Pressure	= 17.42 PSF
Inert Weight	= 128,900 Lbs

The LOX/LH2 separation system design requires for 8 BSMs. This is based on computer simulation results which indicate that a placement of 4 BSMs forward and 4 aft will produce safe separation for nominal ascent (design case) staging. Design case staging conditions include: body rates of 5 deg/sec pitch, 2 deg/sec yaw, and 2 deg/sec roll; alpha = 10 degrees; and beta = 10 degrees. The corresponding booster separation system weight is on the order of 1,600 lbs.

The 8 BSMs are distributed with 4 packaged in the nose cone and 4 placed on the aft skirt. The same orientation of BSMs for the LOX/RP-1 pump-fed booster is also used for the LOX/LH2 booster configuration.

Separation plots for nominal ascent staging (Figures 5.1.2-1 through 5.1.2-3) indicate a clean separation. Because the booster is longer than a SRB, the forward separation motors are forward of their normal location, and plume impingement on the Orbiter TPS is reduced. Thus, it may be possible to reorient the forward BSMs so that they fire more laterally, which makes separation more efficient. If the forward BSMs are redirected, there is the possibility fewer would be needed for nominal separation.

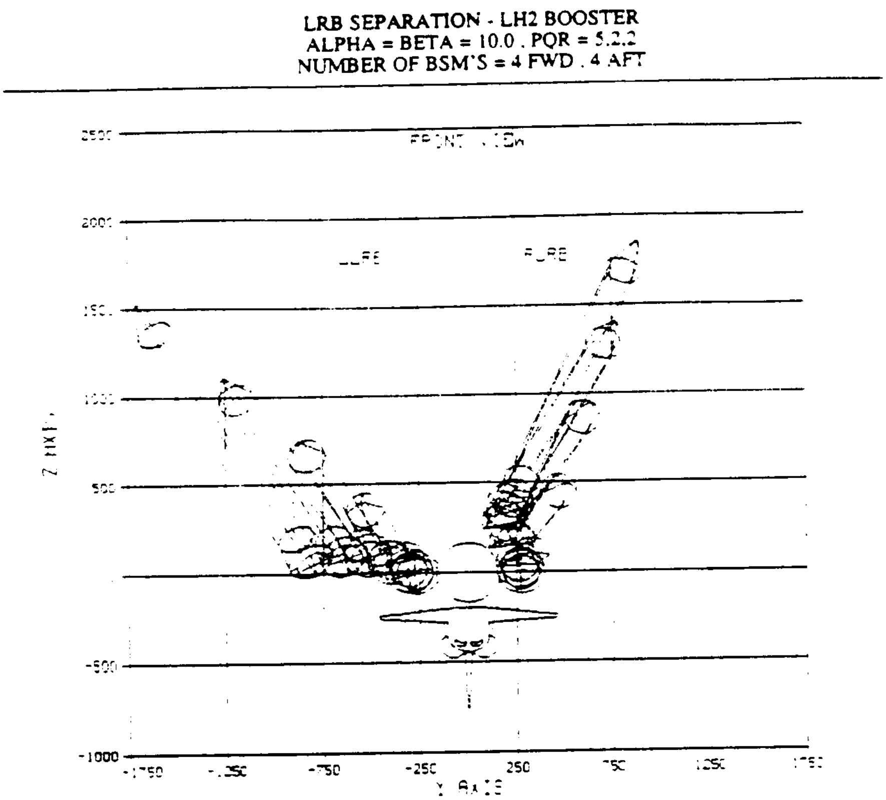


Figure 5.1.2-1. LOX/LH2 Pump-fed Nominal Ascent (Design Case) Separation, Front View

5.1.3 <u>THERMAL PROTECTION SYSTEM</u>. The LO2/LH2 booster configuration requires the most complicated TPS design of the downselected LRB configurations -- two cryogenic fluids require insulation. However, the overall TPS approach is similar to that used on the LOX/RP-1 pump-fed booster (see section 4.1.3).

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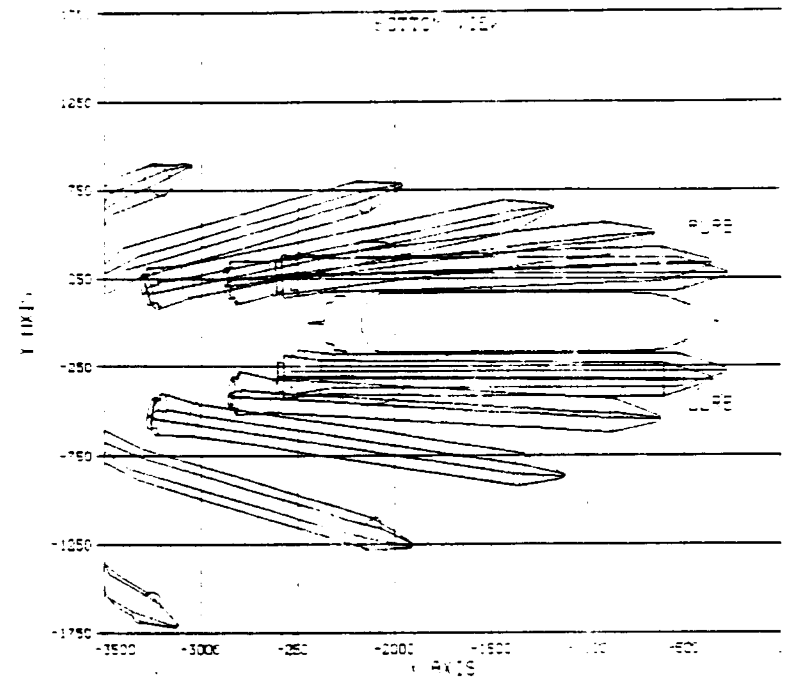


Figure 5.1.2-2. LOX/LH2 Pump-fed Nominal Ascent (Design Case) Separation, Bottom View

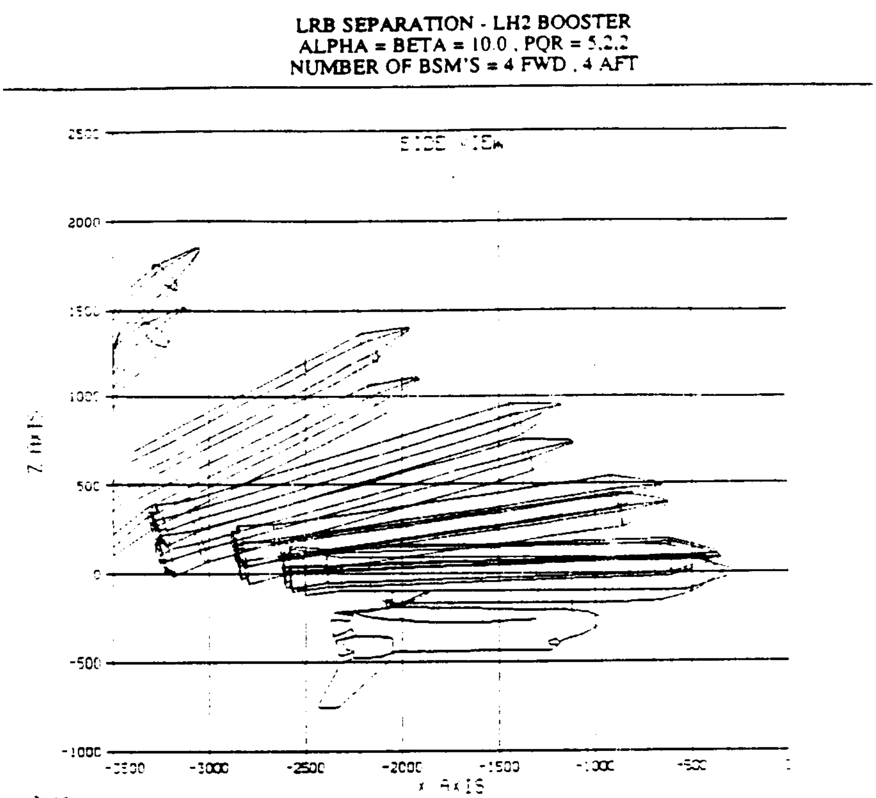


Figure 5.1.2-3. LOX/LH2 Pump-fed Nominal Ascent (Design Case) Separation, Side View

5.1.3.1 Aerodynamic Heating. An examination of the LO2/LH2 booster's 'Altitude vs. Velocity' ascent profile indicates that the booster will experience generally lower heating than a SRB because of a more lofted trajectory, thus lower air density is encountered during high velocity portions of flight (examine Figure 5.1.3.1-1 below, and refer to section 4.1.3.1).

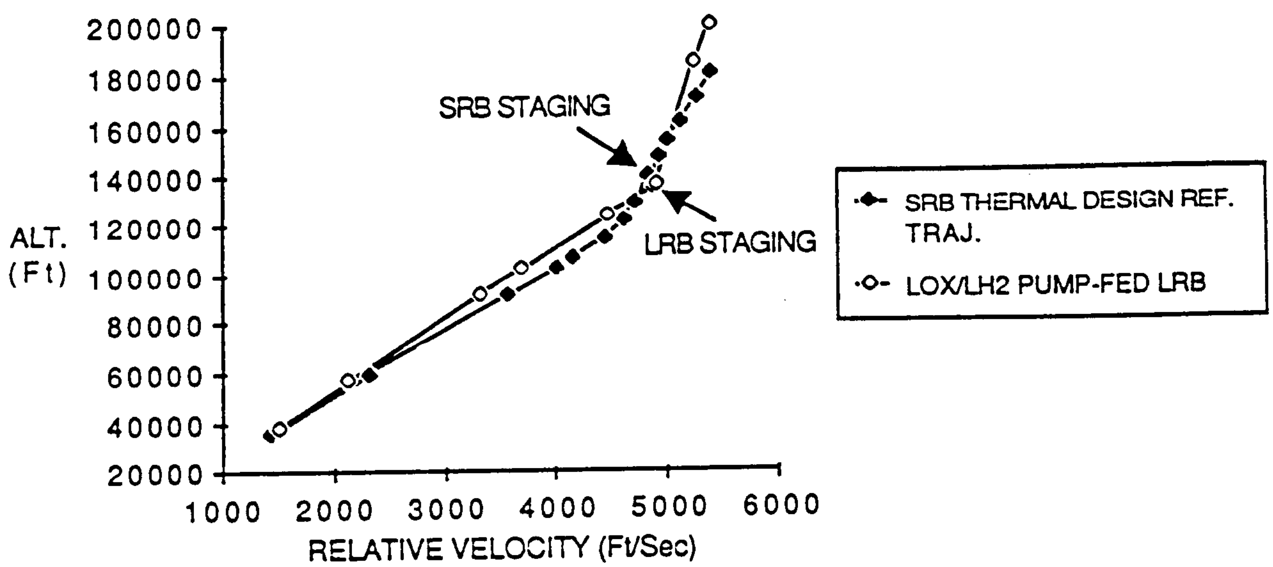


Figure 5.1.3.1-1 LOX/LH2 Pump-fed Booster And SRB Altitude Vs. Velocity Profiles Comparison.

5.1.3.2 Thermal Protection System. The LOX/LH2 pump-fed booster's thermal protection system is similar to that for the LOX/RP-1 Pump-fed booster (see section 4.1.3.2). The booster TPS system will be comprised of SOFI (CRS-488) applied in nominally 1" thickness to barrel sections of the oxidizer and fuel tanks. MSA-1 will cover portions of the nose cone and aft skirt. SLA-561 will be applied more sparingly to other high heating areas of booster such as interface attachment structure, feedline brackets, and other protuberancies. On tank bulkheads, urethane foam will be applied after manufacture. Flexible skirts and a heat shield will be used to protect booster engines and aft skirt components from plume heating. Refer to Figure 5.1.3.2-1 .

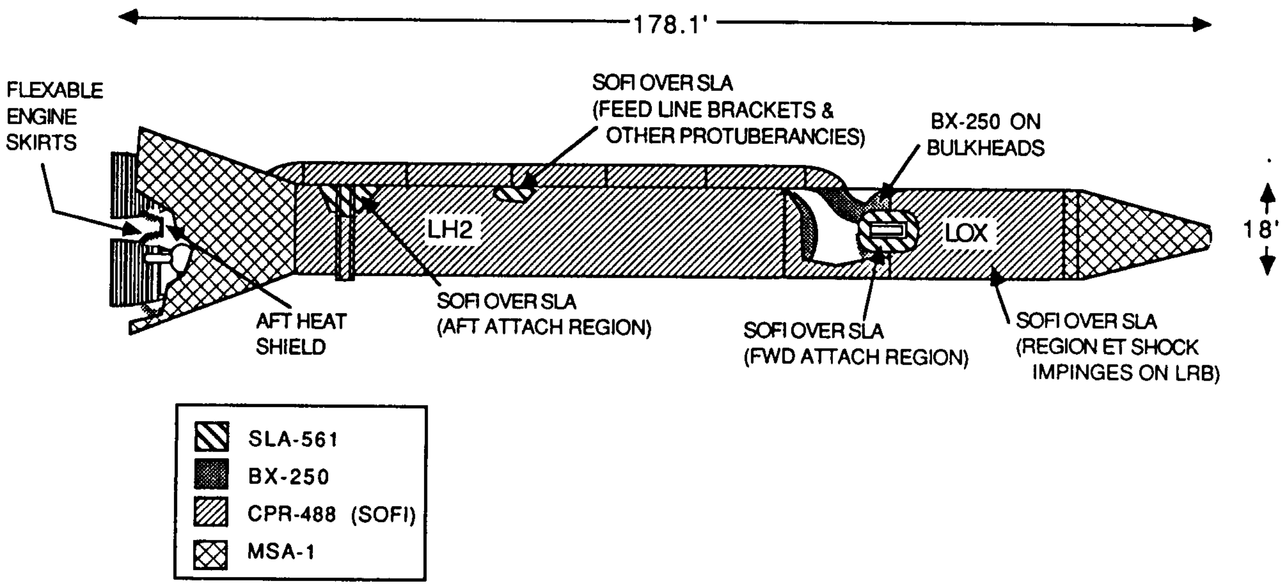


Figure 5.1.3.2-1. LOX/LH2 Pump-fed Booster TPS Layout

5.2 MAIN PROPULSION SYSTEM

5.2.1 LO2/LH2 GAS GENERATOR ENGINE SYSTEM. This section presents the characteristics of the chosen LRB engine configuration consisting of four $LO2/LH_2$ gas generator pump-fed engines virtually identical to the present Space Transportation Main Engine (STME) configuration now being studied by Rocketdyne under a separate STME/STBE contract.

The specific baseline engine concept was selected based on previous studies and experience along with trade studies for the STS application. This engine system allows the following main advantages: 1) low technical risk, 2) no environmental concerns, 3) commonality with current shuttle ET propellants, 4) reduced POGO stability compensation hardware size and complexity, and 5) reduced exit diameter diminishing the need to make major launch platform alternations.

5.2.1.1 Performance and Characteristics. An engine performance and cycle balance was generated for the selected configuration and the resultant parameters were used to establish the pertinent combustion chamber, injector, nozzle, and turbopump characteristics leading to the recommended configuration and physical design. The engine characteristics are tabulated in Table 5.2.1-1.

The engine selected is of an expendable type with step throttling capability of 100% to 75% of the nominal thrust level. Engine thrust, chamber pressure, expansion ratio, and engine throttling range were determined by GDSS, based on minimum vehicle life cycle cost analyses and engine data provided by Rocketdyne. The propulsion system described here is based on an overall mixture ratio of 6.0 and expansion ratio of 20.

The engine is baselined with no boost pumps, and minimum inlet pressures of 65 psia for LO2 and 45 psia for LH_2. Boost pump trades conducted in the STME studies showed an increase in engine weight, cost and complexity when boost pumps are included, and the STME is baselined without boost pumps. Rationales for the selected pump inlet pressures is described in Propulsion Trade Section 3.7.4. Various options for disposing of the engine gas genertator (GG) exhaust were studied previously and are given in the engine data Appendix 6 (Phase II Report), to Volume II of the LRB Final report. A solid propellant gas generator (SPGG) assisted start method is selected over the tank head start method because it

Table 5.2.1-1 Baseline LRB LO2/LH2 Gas Generator Engine Characteristics

ENGINE PARAMETERS	NOMINAL	MINIMUM
Number of Engines per LRB	4	
Weight (lb)	6100	
Throttle (percent)	100	75.0
Oxidizer Flow Rate (lb/sec)	1162.7	893.3
Fuel Flow Rate (lb/sec)	193.8	148.9
Vacuum Thrust (lb)	558,000	418,500
Sea Level Thrust (lb)	518,574	388,930
Chamber Pressure (psia)	2250	1701
Vacuum Isp (sec)	411.4	412.3
Sea Level Isp (sec)	382.3	373.2
Mixture Ratio	6.0	6.0
Nozzle Area Ratio	20.0	
Throat Radius (in)	6.543	
Exit Diameter (in)	58.44	
Overal Length (in)	112.1	
Inlet Pressure: LO2 (psia)	65	
Inlet Pressure: LH_2 (psia)	45	
Throttling Type	Step-Open Loop	
Mission Life	1	
No. of Starts	5	
Boost Pump	None	
Bleed Required	None	
Engine Start	SPGG	
Thrust Vector Control Actuator Type	Electromagnetic	
Valve Actuator Type	Electromagnetic	
Inlet Temperature	Below 16 PSIA Saturation Temperature	
Inlet Line Diam. (both oxid. & fuel)	10 (in.)	
Reliability	99% @ 90% confidence level	
No. of Pump stages		
LO2	Single Stage	
LH_2	Two Stage	

provides more repeatable starts. In addition, the tank head start is comparatively slow compared to other types of starts, and this may complicate optimization of ignition sequencing for the STS vehicle.

5.2.1.2 Schematic and Operations. A schematic diagram of the engine is shown in Figure 5.2.1-1 and a side view and top view of the engine are shown in Figure 5.2.1-2.

This engine is similar to the present STME engine configuration except for the expansion ratio and the propellant inlet pressures. The engine has separate LO2 and LH2 turbopumps. The two turbines are driven in series by the same gas generator. The GG exhaust gases first drive the fuel turbine and then the LO2 turbine. The LO2 heat exchanger is located downstream of the LO2 turbine and supplies LO2 for use in pressurizing the LO2 propellant tank. The GG exhaust gas is then utilized to cool the nozzle and is dumped at the nozzle exit around the periphery of the nozzle. Vaporized hydrogen required to pressurize the hydrogen propellant tank is supplied from the combustion chamber coolant. The STME has a SPGG assisted start. Steady state operation is reached in approximately 3.5 seconds. The valve start and shutdown sequences and the moment of ignition of the SPGG are shown in Figure 5.2.1-3. The transient flows during startup (and during shutdown) are shown in Figure 5.2.1-4 and Figure 5.2.1-5. The corresponding main chamber pressure and GG chamber pressure are shown in Figure 5.2.1-6.

The LO2 heat exchanger valve is then opened allowing a small amount of LO2 to be vaporized and utilized to pressurize the LO2 tank.

5.2.1.3 Design. A side and top view of the LO2/LH_2 engine are shown in Figure 5.2.1-2. The selected expansion ratio of 20 has resulted in a relatively short nozzle. A protective insulation type exhaust covering, (not shown) will likely be required to protect the engine from excessive heat transfer from the plume. The reduced exit diameter and length of the engine are a distinct advantage since the overall plume diameter and gimballing space required are both substantially reduced.

The regneratively cooled combustion chamber has an expansion ratio of 7. A GG exhaust gas cooled nozzle extends the expansion ratio from 7 to 20. The LRB nozzle design will have an optimized 80% bell nozzle from the throat to an expansion ratio of 20 at the nozzle exit.

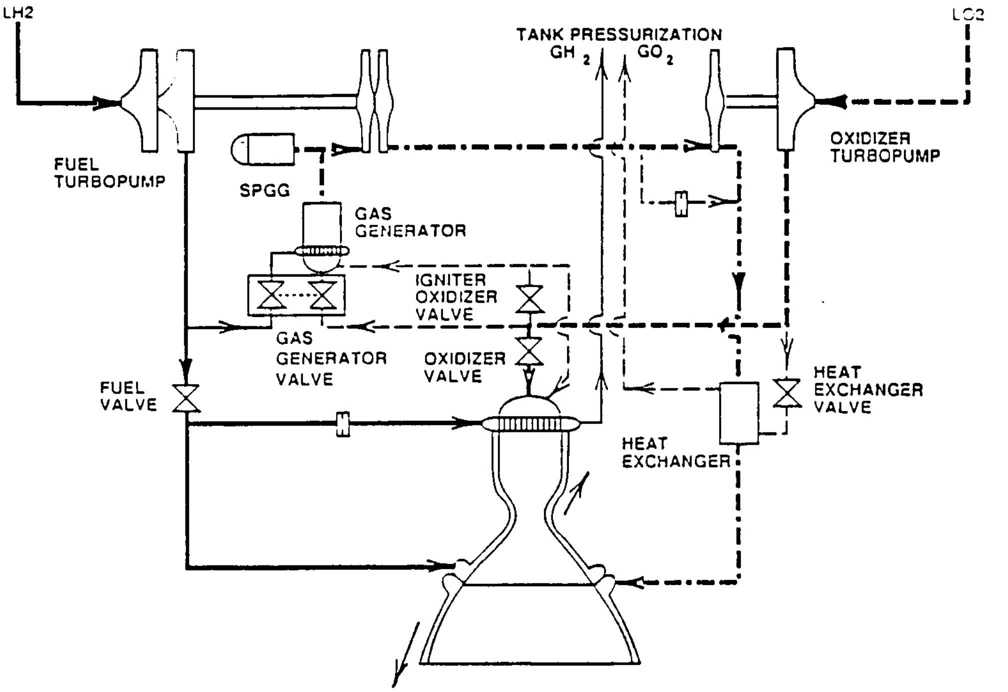
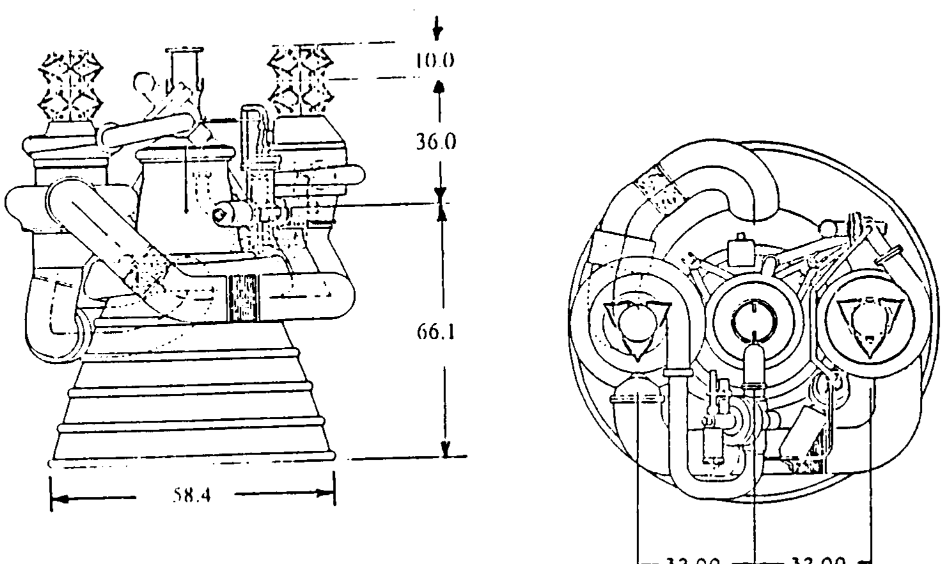


Figure 5.2.1-1 LO2/LH_2 Gas Generator (GG) Cycle With Gas Cooled Nozzle (Schematic)



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Rocketdyne Division

Figure 5.2.1-2 LRB LO2/LH2 Engine (Top and Side Views)

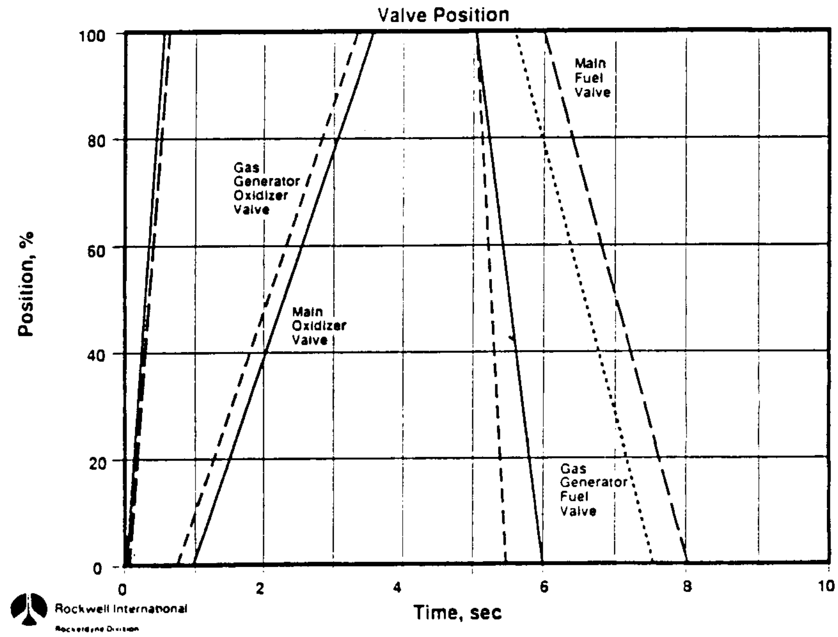

Figure 5.2.1-3 Start-up/Shut-down Sequence and Valve Movement for LO2/LH2 GG Engine.

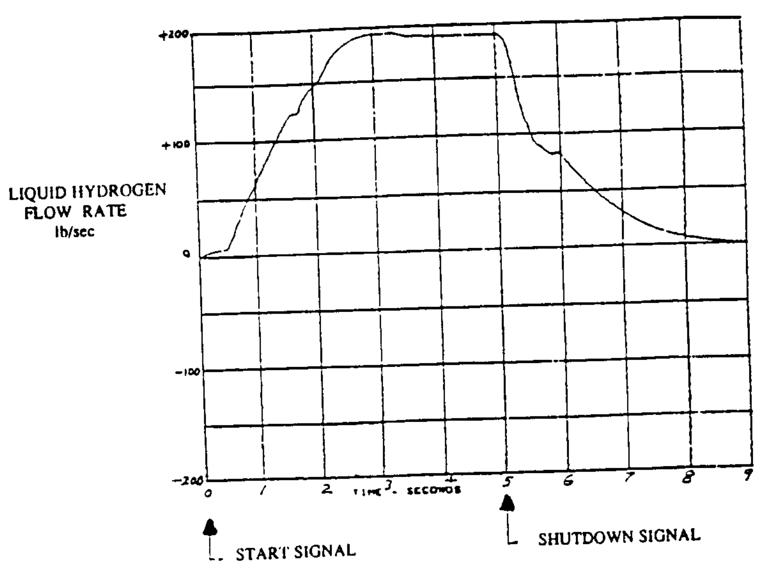

Figure 5.2.1-4 Start and Shutdown Transient LH_2 flow

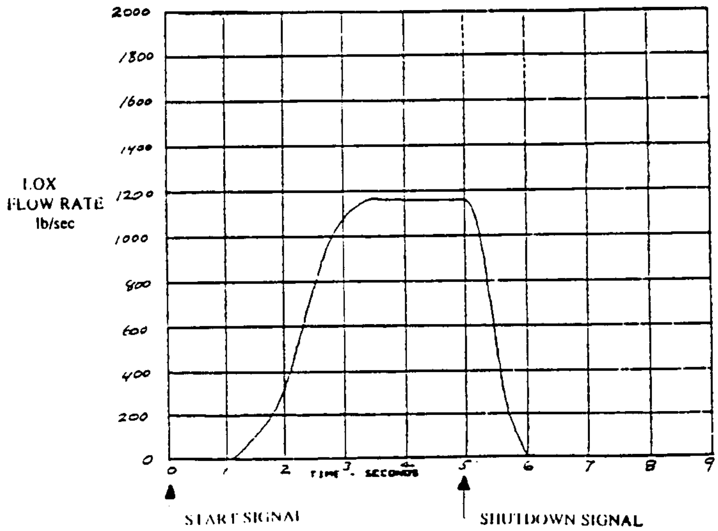

Figure 5.2.1-5 Start and Shutdown LO2 Transient Flow

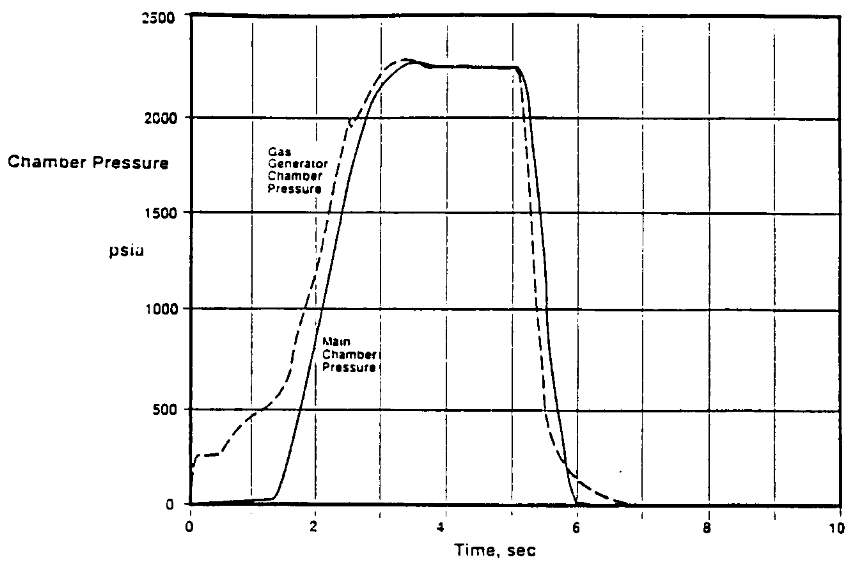

Figure 5.2.1-6 GG Cycle Transient Performance - Main Chamber Pressure and GG Chamber Pressure

The nozzle has a tubular wall construction. The nozzle tubes, from an expansion ratio of 7 to 20, are cooled with the exhaust gas coming from the LO2 turbine exhaust duct. At the point where the gas enters the tubes, part of the gas flow is used as film coolant. The remainder conductively cools the tubes. The coolant gases (GG exhaust gas) flow in the same direction as the primary nozzle flow and are dumped out of the tubes at the nozzle exit plane.

An injector design cross-section is shown in Figure 5.2.1-7. It is a typical gas/liquid coaxial injector of conventional design used in LO2/LH2 rocket engines. The detail design has been specially constructed to reduce fabrication costs.

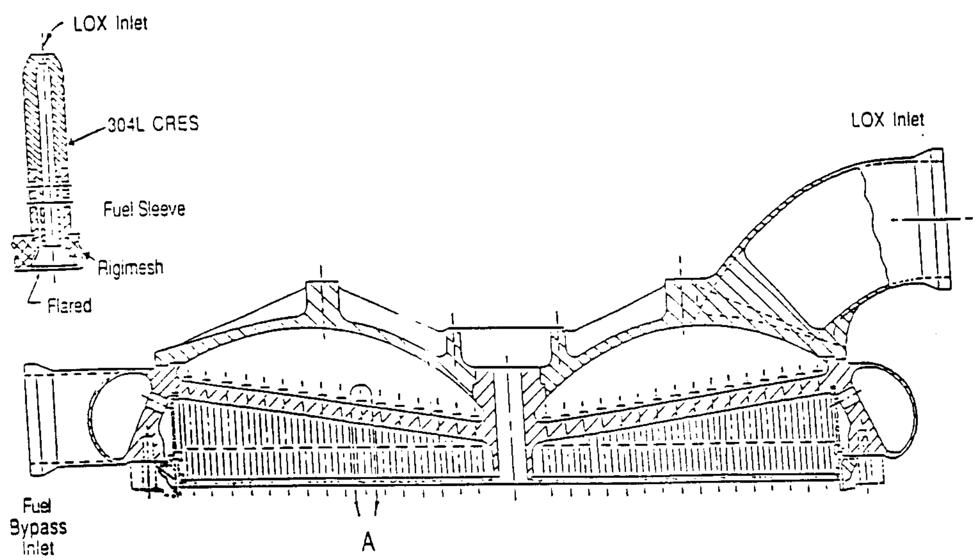

Figure 5.2.1-7 Injector Design

Cross-sections of the fuel and LO2/LH2 turbopumps are shown in Figures 5.2.1-8 and 5.2.1-9, respectively. The single stage LO2 turbopump is driven by a single stage turbine. The fuel pump has two stages driven by a two stage turbine. Again, the designs minimize fabrication costs.

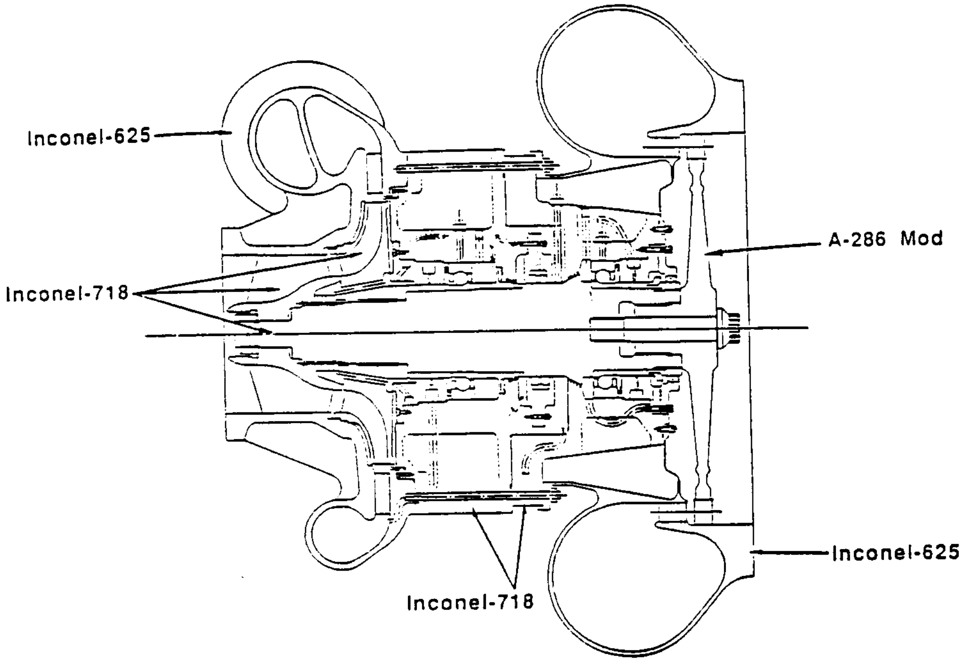

Figure 5.2.1-8 Gas Generator Cycle LO2 Turbopump

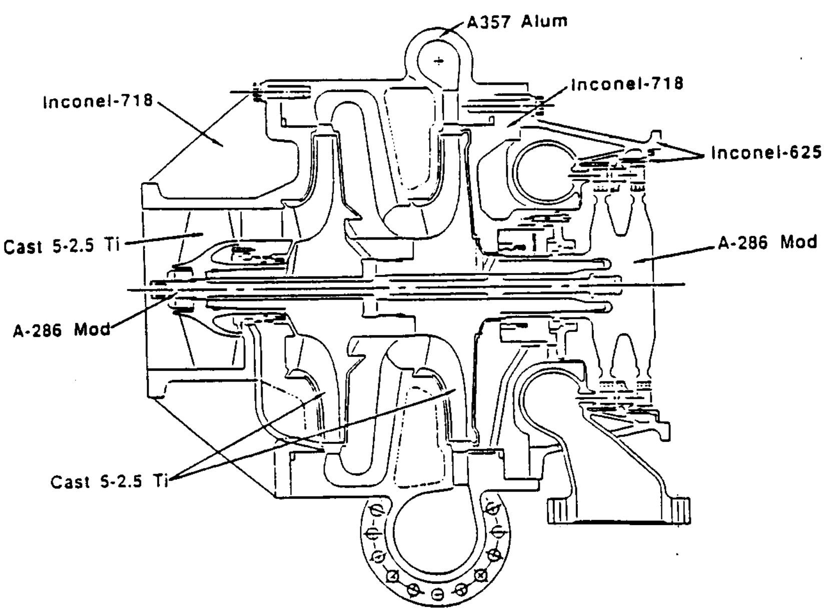

Figure 5.2.1-9 Gas Generator Cycle Fuel Turbopump

A heat exchanger design sketch for vaporizing LO2 and pressurizing the LO2 tank is shown in Figure 5.2.1-10.

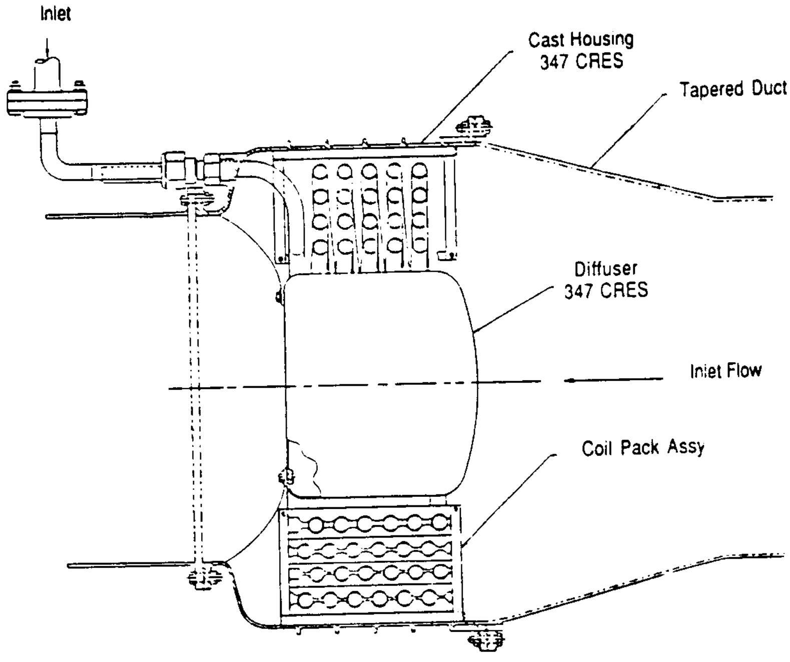

Figure 5.2.1-10 GO2 Heat Exchanger Design

5.2.1.4 POGO System. A preliminary estimate was made of the size of a POGO compensator. It was estimated that a unit of about 1 cubic. ft. volume placed just above each of the four LO2 prevalves will be adequate. A single helium supply line branching to each of the four compensators can be provided. Three slightly different POGO suppressor device concepts are shown in Figure 5.2.1-11. In each case a spherical or cylindrical volume surrounds the LO2 feed line with connecting ports at the bottom to allow rapid propellant flow in and out of the volume, thus suppressing feed line flow oscillations to the engine. The action is similar to that of a piston accumulator. Concept 1 uses very little helium, since it is filled with helium only once just before lift-off. However, as vehicle acceleration is increased, the gas volume will decrease due to an increase in static head pressure. Counteracting this is a gradually decreased static head due to lowering of the level in the propellant tank. Concept 2 maintains the gas volume independent of the static pressure, but requires a small helium bleed flow throughout the boost period. Concept 3 has an active liquid level control to ensure that the static volume remains relatively constant. Trade-offs can be made during more detailed design efforts. Meanwhile, Concept 2 is considered the suppressor of choice at this time.

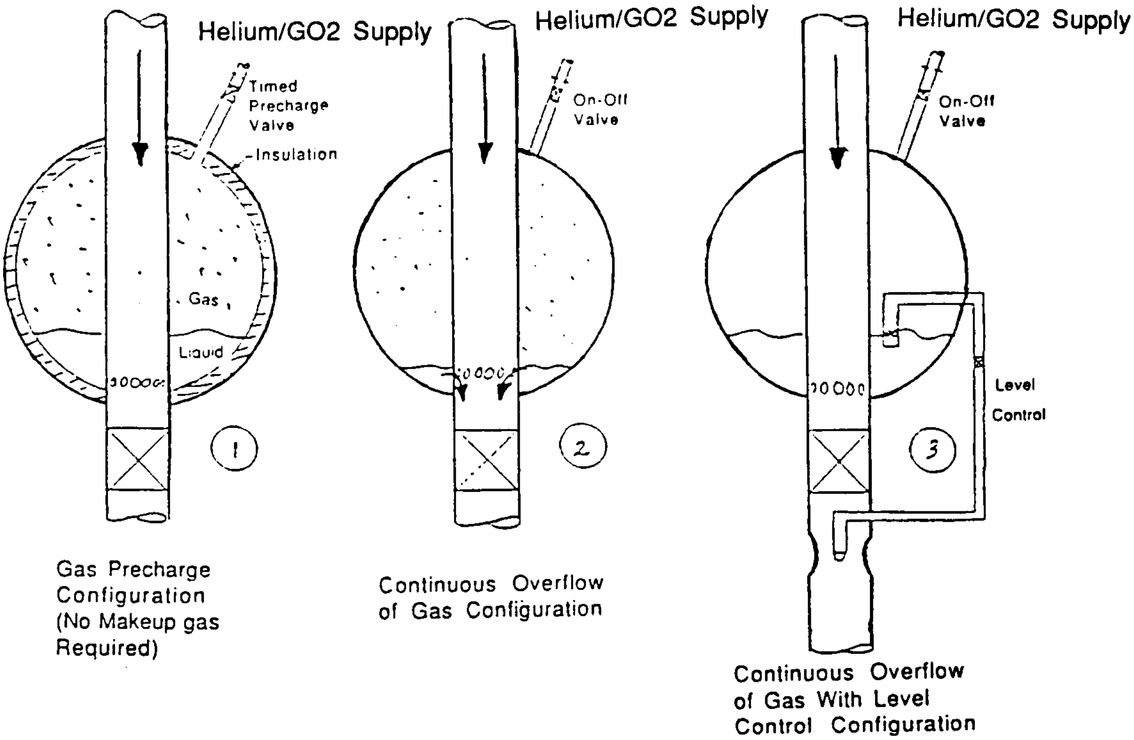

Figure 5.2.1-11 POGO Suppressor Concepts with Concept 2 (Baseline)

5.2.1.5 Engine Control. The control system of choice is an open loop, step throttled type. The system for controlling the state of the engine and for engine condition monitoring is shown Figure 5.2.1-12. Changing the thrust in steps is accomplished by changing the gas generator propellant flow in steps by means of the GG propellant valve. When GG flow output is reduced, the power to the turbopumps is reduced and the main propellant flows are decreased. For example, referring to Figure 5.2.1-12, a signal to reduce thrust coming from the Vehicle Command Bus is received by the State Controller which in turn signals the Control Module to energize the appropriate valve actuator. Except for the ignition and shutdown operation, the balance of the operations by the controller are of the condition monitoring type. Signals from the instrumentation shown in Figure 5.2.1-13 are compared with preset high/low limits. If these limits are not exceeded, no action is taken. If they are exceeded, warnings to vehicle command and/or automatic engine shutdown are initiated.

The engine control system also furnishes the signals required to carry out the engine start and shutdown sequences.

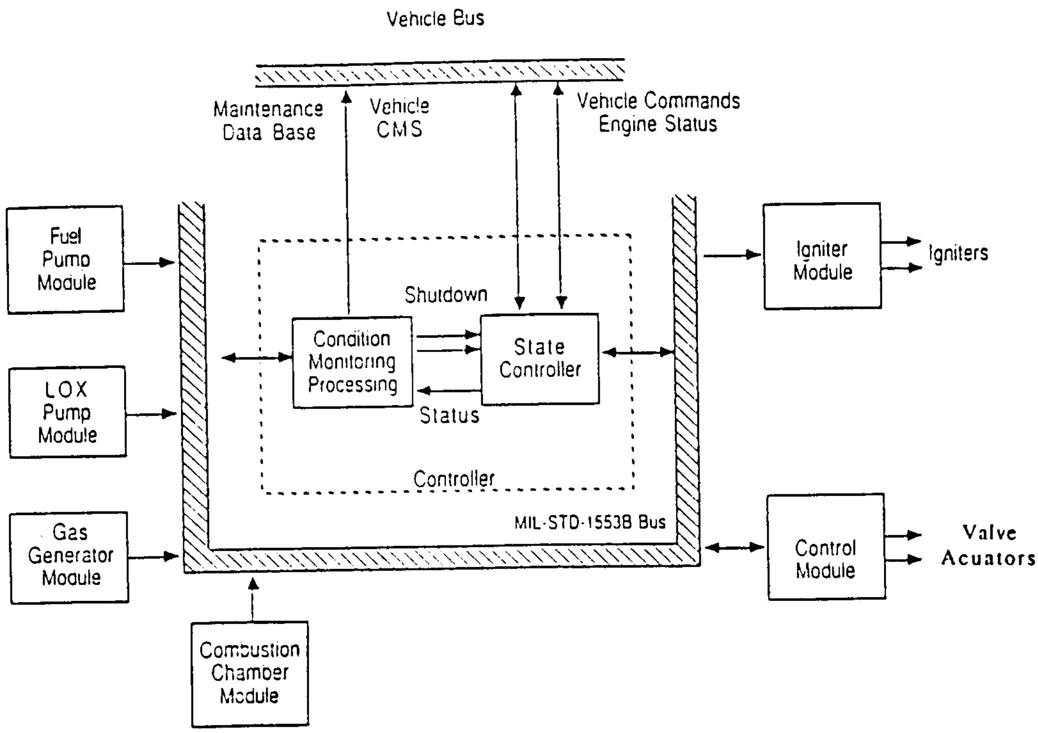

Figure 5.2.1-12 Control System Diagram

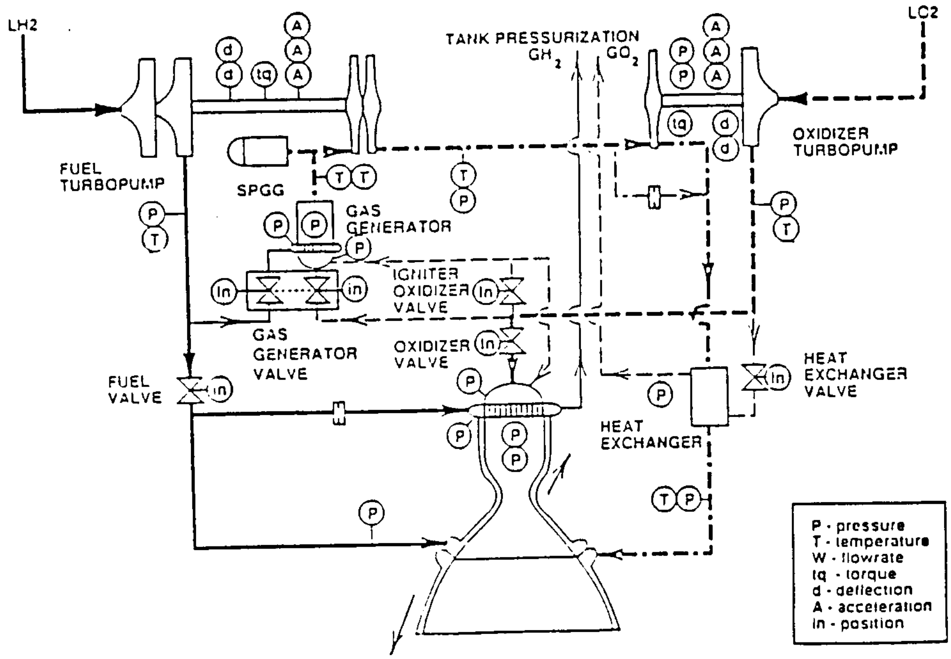

Figure 5.2.1-13. Flight Instrumentation Schematic

The number and type of instruments utilized to carry out the engine condition health monitoring function is a trade-off between 1) the cost, weight and reliability of instrumentation hardware, computer hardware, and software, and 2) the engine reliability requirements needed to meet the overall vehicle reliability requirements. Subsequent Engine Phase B studies will define the health monitoring functions and system design.

5.2.1.6 Engine Interface Requirements. The following interface requirements have been defined from overall LRB studies (Table 5.2.1-2). More detailed interface requirements will be derived in Phase B studies.

Table 5.2.1-2. LRB LO2/LH2 Interface Conditions

Gimbal requirement	±6°
Minimum Inlet pressure (psia)	
Hydrogen	45
Oxygen	65
Minimum Inlet Temperature (°R)	
Hydrogen	37.5
Oxygen	164
Inlet Diameter (in) both LO2 & LH2	10
Maximum Pressurization Flowrate (lb/s)	
Hydrogen	1.5
Oxygen	7.5
Mixture ratio tolerance(1)	±3%
Thrust tolerance(1)	±3%

(1) at standard propellant inlet conditions

5.2.1.7 Engine Check-out on the Pad. The engine condition monitoring system and its associated measuring system can be used for the engine checkout operation. A fault detection algorithm can then be used to aid in locating the source of any anomalous operating condition.

For in-flight operation, however, only the decision of whether or not to initiate an engine shut down signal and to continue the flight under a one-engine-out condition or not is of importance. The fault diagnosis is only of secondary importance and any hardware and software required is considered ground support equipment. The detailed analysis to determine the characteristics of abort procedures be determined in phase B from a vehicle standpoint with consideration for engine condition monitoring, shutdown and throttling capabilities and limitations.

5.2.1.8 Engine Schedule and Programmatics. The overall development program schedule for the LO2/LH2 pump fed engine (and applicable to the LO2/RP-1 pump fed engine), is shown in Figure 5.2.1-14. The 63 months (5 1/4 years) development program is designed to support a first vehicle launch in the third quarter of 1995 and therefore would benefit from a Phase B effort and a modest technology program in terms of reduced risk. (For further details see also the engine report, Appendix 6 to Volume II of the final report, RI/RD88-180 of June 1988, page 102, ff.)

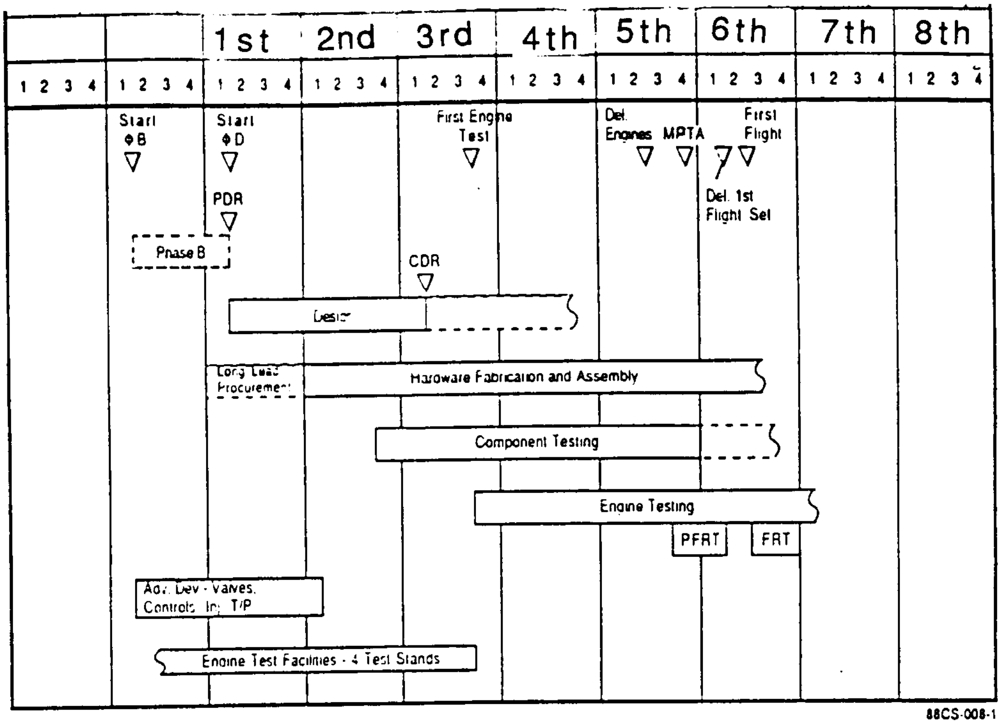

Figure 5.2.1-14 LO2/LH2 Pump Fed Engine Development Program

First, a benefit of the Phase B design effort would be to allow early long lead procurement of casting tooling for some of the major components such as the pump housings. Secondly, significant benefits in terms of reduced risk would be derived from a technology program that is started in parallel with the Phase B design effort and completed in time to provide data for the development program design phase. The specific technology that would provide the most benefit is in the area of injector design for stability and for turbo pump bearings and seals and rotating elements. Thirdly, as indicated in Figure 5.2.1-14 engine test facilities are required by the fourth quarter of 1992. These test facilities are assumed to be provided by the government or the vehicle contractor. Formal Pre-Flight Rating Tests (PFRT) are planned prior to the first flight and Flight Rating Tests (FRT) to certify readiness for production; full operational status which are planned after the first flight.

The development program has been estimated to cost \$987M and is spread out in time as shown in Figure 5.2.1-15.

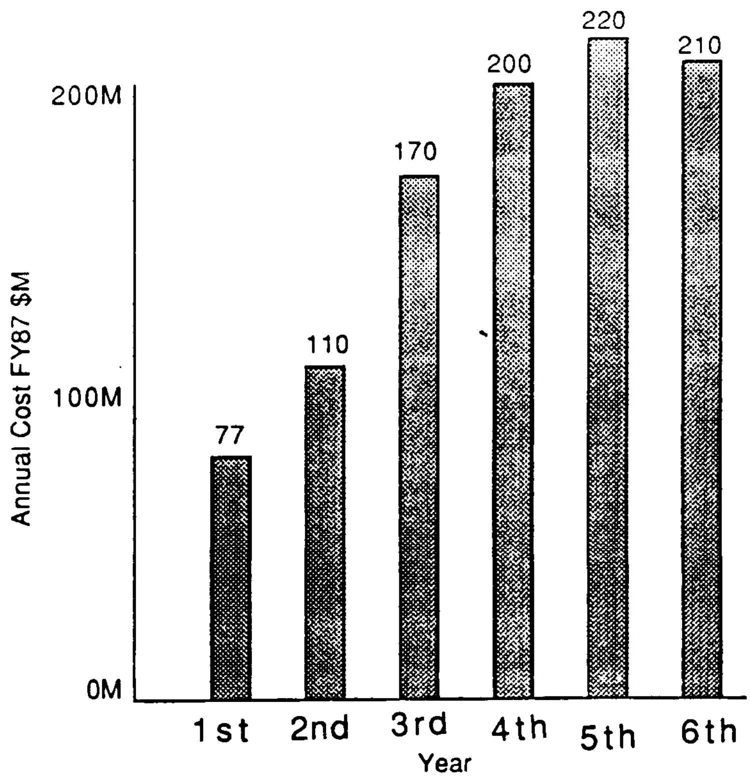

Figure 5.2.1-15 LO2/LH2 LRB Pump Fed Full Scale Engine Development Cost

5.2.2 <u>VEHICLE PROPELLANT SYSTEMS</u>. Preliminary requirements and groundrules for the LO2/LH2 pump-fed LRB vehicle propellant systems are summarized in Table 5.2.2-1. For systems that are related to ground operations, refer to Section 9.2 for more detailed descriptions.

5.2.2.1 Feed and Fill/Drain Systems. Flow schematic of the feed and fill/drain systems is shown in Figure 5.2.2-1. The feed system for the LO2 tank includes a single 20-in external main line that leads from the center of tank aft bulkhead to a two legged manifold at aft-end of the fuel tank. Each manifold then divides into two 10-in ducts, each in turn is connected to an engine oxidizer inlet. A Single main oxidizer line was selected over dual line because of lower design and operational complexity. The LH2 tank has four 10-in ducts, each directly feeding an engine fuel inlet.

A prevalve is located in each of the 10-in ducts immediately downstream of the manifold, and it regulates propellant flow to each engine during prelaunch operations and at engine start and shutdown. Both the LH2 and LO2 prevalves will be closed only during engine purge and open throughout tank fill operations. In case of a turbopump failure in an engine, its prevalves will close to isolate that engine from affecting the rest of the system.

For either LO2 or LH2, the fill/drain system.includes a 6-in line with facilities disconnect at one end and tee into feed system manifold at the other end. The fill/drain line provides a vehicle-facilities interface for tank purge, loading and drain. A fill/drain valve, located close to each manifold, regulates propellant flowrates during prelaunch operations and closes prior to tank pre-pressurization.

In case of an abort on launch pad, LO2 and LH2 are drained through the same fill paths and back to facility storage tanks. Both fuel and oxidizer tanks will need to be pressurized with ground GHe during draining to avoid subatmospheric ullage pressures and also to provide quick draining.

Anti-Geyser. The LO2 liquid height in this vehicle is over 110ft above the engines, and as shown in Figure 4.2.2-2, geysering is quite probable. Several geysering suppression methods were considered, and GHe injection method as depicted in Figure 5.2.2-1 is chosen. A ground supply GHe at a flowrate on the order of that for the ET (~ 0.01 lb/sec) will be injected at the bottom of the LO2 feedline from the beginning of slowfill until start of pre-pressurization.

Table 5.2.2-1 Requirements/Assumptions for LO2/LH2 Pump-Fed Propulsion System

SYSTEMS	LO2	LH2	REMARKS
Engine Inlet			
Minimum P (psia)	65	45	
T (°R)	164	38	
Diameter (in)	10	10	
Feed System			
Main Line Diameter (in)	20	20	
Manifold to Engine Diameter (in)	10	10	
Max Propellant Flowrate (lb/s)	4651	776	
Max line P @ engine inlet (psia)	170	55	
GHe Injection for Anti-Geysering in LO2 Feedline			
Line Diameter (in)	TBD		Inject at LO2 Feedline manifold
GHe Injection Rate (lb/s)	TBD		
Fill/Drain			
Duct/Disconnect Diameter (in)	6	6	
Tank Operating			
Ullage Control Level (psia)	35	55	constant from pre-start to BECO
Bulk T min-max (°R)	164-600	38-660	liquid - pressurant inlet
Max Tank Bottom P Limit (psig)	70	70	
Pressurization System	Autogenous	Autogenous	
Medium	GO2	GH2	
Heating Source	turbine disch	coolant disch	
Line Operating T (°R)	164-1000	38-570	
Line Operating P (psia)	ambient-600	ambient-600	
Main Line D (in)	3.5	3.5	
Engine to Manifold Line D (in)	1.7	1.7	
Max Autog Pressurant Flow (lb/s	30	6	@ Nominal with 4 engines
Total Pressurant Wt (lb)	3600	925	
GHe Pre-Press Line D (in)	1.0	1.0	
GHe Pre-Press Supply	2000 psia & 520°R		
Vent System			
Valve Diameter (in)	4	4	GH2 vent line exits at aft skirt
Vent Level (psig)	40	60	
Purge System			
Engine Purge Line Diameter (in)	TBD	TBD	GN2 & GNe ground supplies @ TBD
Engine Purge Supply Condition	TBD	TBD	
Tank Purge Supply Condition	TBD	TBD	
Total Liquid Residuals (lb)**	8200	200	** Vehicle Sizing Assumes 1% of Ascent Propellant

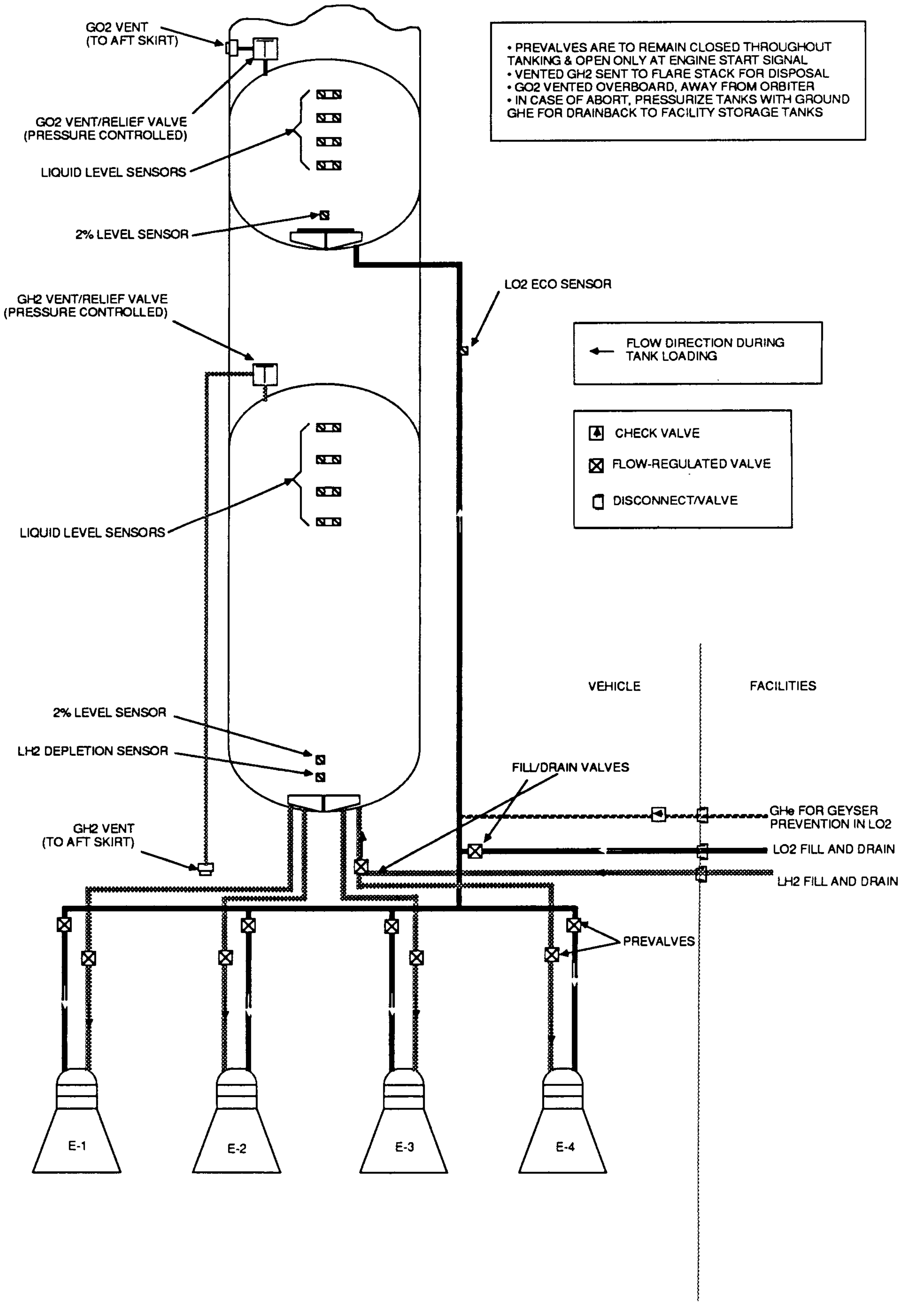

Figure 5.2.2-1 Schematic of Feed & Fill/Drain Systems for LO2/LH2 Pump-Fed LRB

Engine Conditioning Systems. Based on new STME design approaches, the LO2/LH2 engine will no longer require bleed systems for chilldown prior to start. Pre-pressurization in tanks added to the static pressure will raise saturation temperatures significantly at the engine inlets, condense all vapors formed due to external heating, and provide sufficient NPSP to avoid pump cavitiation.

5.2.2.2 Purge Systems. Schematic for the purge systems is shown in Figure 5.2.2-2. The tanks are to be purged initially with nitrogen gas to remove moisture before tanking operations. In addition, the LH2 tank will also then be purged with helium to prevent frosting of nitrogen gas residuals. Tank purges are supplied through the main fill/drain lines from facilities at moderately low pressures.

With prevalves closed, the engines are to be purged with warm GN2 from start of tank purge at a low flowrate. At start of tank fill, the engine purge supply is switched to GHe and then terminated at engine start signal. An orifice is placed immediately upstream of purge line engine interface, for LO2 as well as LH2 sides, to assure adequate flowrate through each engine system. All engine purge gases are ground supplied at high pressures provided through a single disconnect.

5.2.2.3 Pressurization Systems. A flow schematic of the tank pressurization systems is shown in Figure 5.2.2-3. Line sizes and operating conditions are included in Table 5.2.2-1.

Prior to engine start and up to lift-off, the tanks will be pressurized with facility supplied helium. During engine start, tank pressurization will transition to autogenous pressurization. With the Helium pressurization terminating at lift-off, the propellant tanks will be solely pressurized by autogenous pressurant from lift-off and throughout entire boost phase operation. The autogenous pressurant is supplied from each of the four main engines. GO2 pressurant is bled-off the LO2 pump discharge and heated through a heat exchanger, which uses the LO2 turbine exhaust stream as heat source; while the GH2 pressurant stream is bled-off directly from the thrust chamber cooling exit stream, where the LH2 coolant flow has been superheated. The pressurant flow rate requirements for heat exchanger sizing are given in Table 5.2.2-1 or be dictated by vehicle imposed requirements.

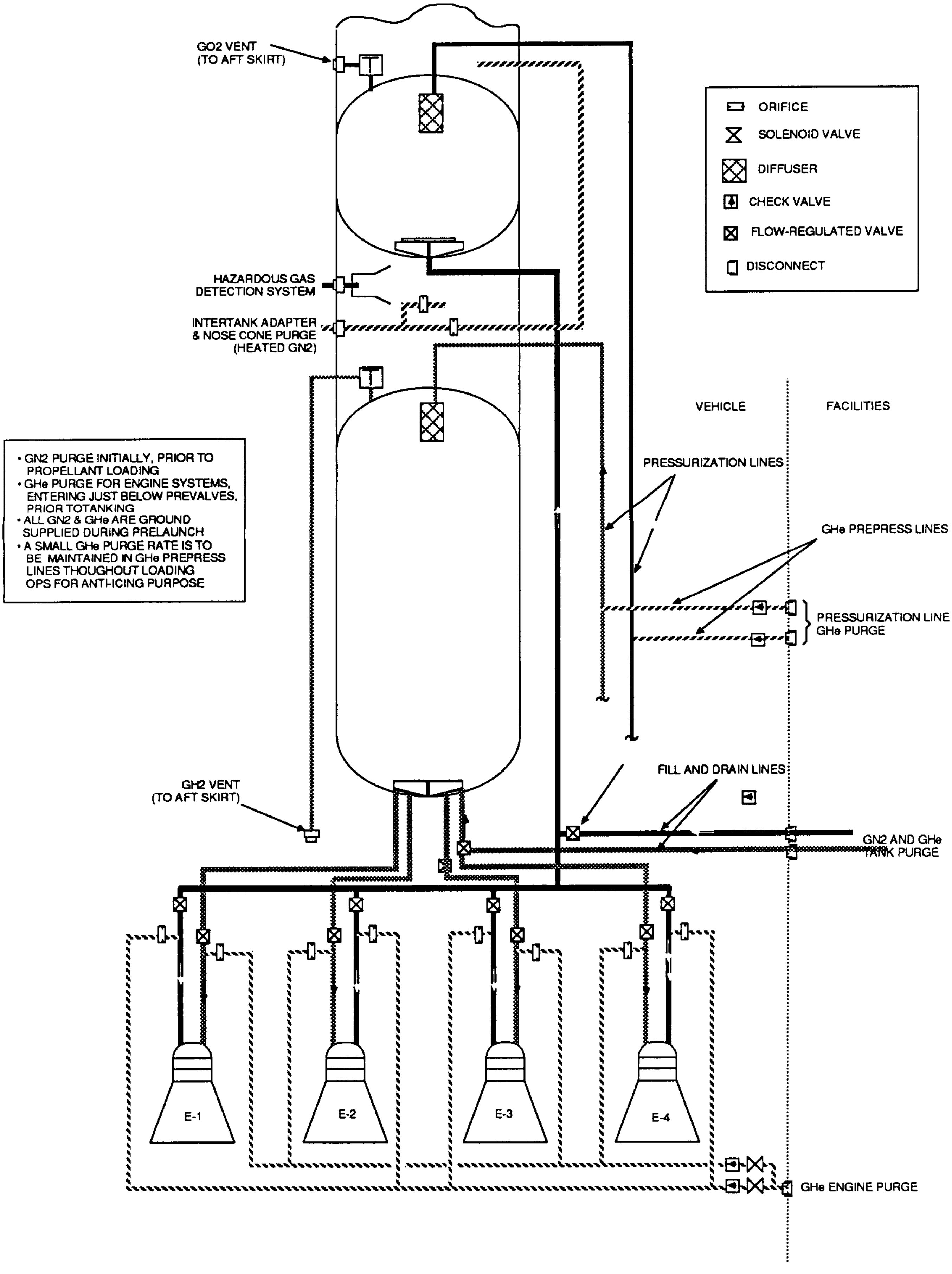

Figure 5.2.2-2 Schematic of Purge Systems for LO2/LH2 Pum-Fed LRB

Prior to lift off the tanks are pressurized with GSE Helium introduced upstream of the vehicle pressurization valve/orifice network. The GSE pressurization module consists of two parallel regulated branches, primary branch AA and secondary branch BB, and the module is located on the ground side of the disconnect. The primary branch operates prior to engine start and maintains tank pressure within the nominal control settings. However, in order to accommodate the pressure decays during start transients, this pressurization will be augmented through the secondary branch BB that will be commanded open during the transient if pressures fall below the nominal operating control band.

As shown in Figure 5.2.2-3, the pressurization lines from the engines are manifolded to a single main line that leads to the top of each tank. Inside each tank, a diffuser located at the top will disperse the pressurant stream entering the tank ullage.

Preliminary tank pressure control, vent/relief, minimum required and expected inlet pressure levels for LO2 and LH2 tanks are shown in Figure 5.2.2-4a and 4b, respectively. The pressurization system for both GSE and vehicle is depicted in more details in Figure 5.2.2-5, same for either LO2 or LH2 tank. Preliminary sizing and operating levels of orifices/valves are given in Table 5.2.2-2. Note that all control valves shown are normally closed and of solenoid type.

Autogenous pressurant flow control network is located on the main line, and it composes of an open branch and two regulated branches. The primary branch A is open to provide a minimum pressurant flowrate at all times, and it is sized based on lowest flowrate requirement expected at minimum power level. The other two regulated branches, secondary branch B and back-up branch C, with On-Off valves, are to provide higher pressurant flowrates that are needed during engine start or flight at 100% power level. These branches will operate synchronously, with one acting as a back-up branch.

As indicated, flow is initiated through the supplemental branches during the start transients for both tanks. Preliminary failure mode analysis results are shown in The Figures 5.2.2-6 and 5.2.2-7 for the LO2 and LH2 tanks, respectively. And as shown, the pressurization system is atleast single failure tolerant (dual failure cases have not been examined yet). Figure 5.2.2-6a, b and c indicate the LO2 tank pressure profiles for normal operation, a

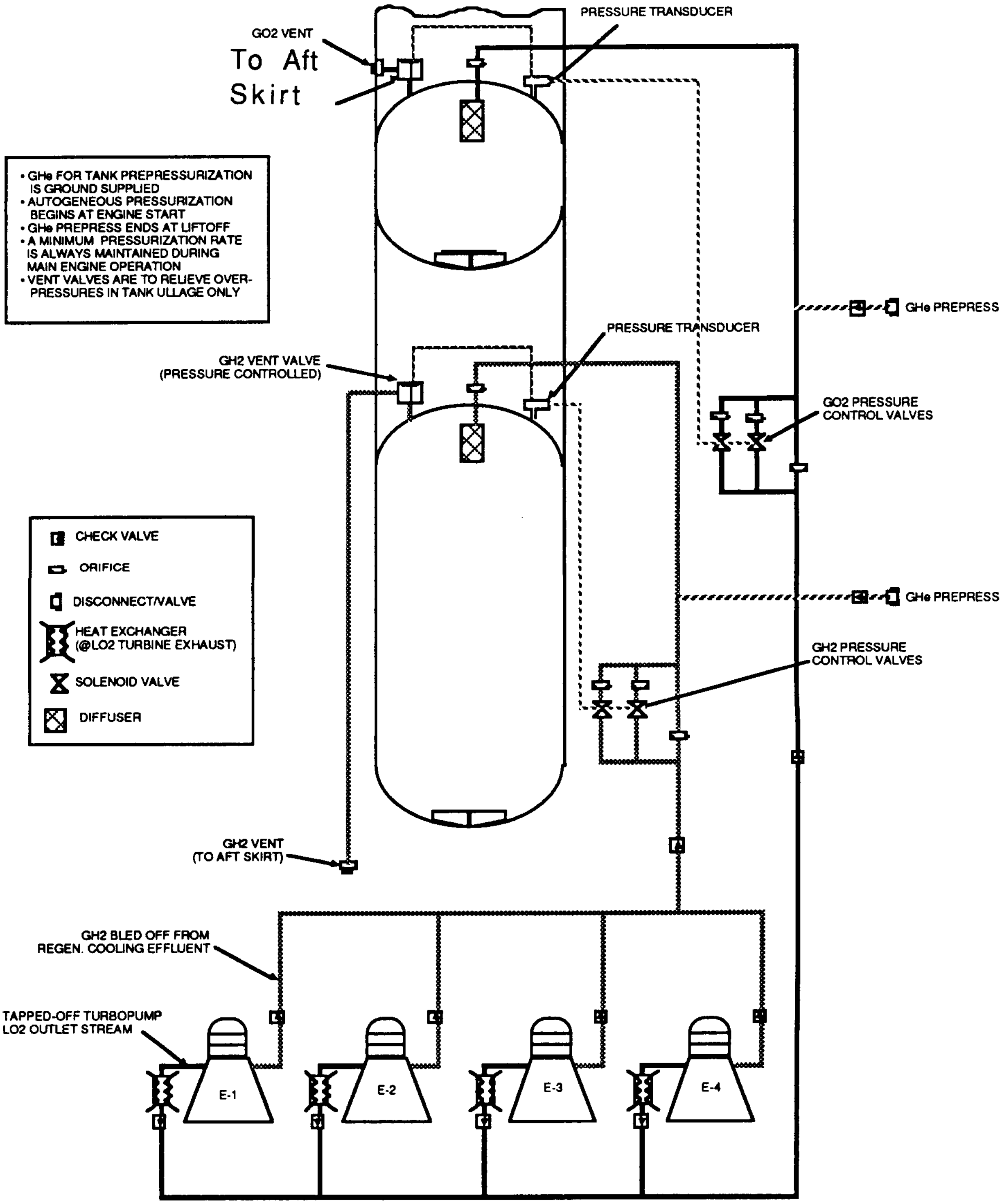

Figure 5.2.2-3 Pressurization & Vent Systems Schematic of LO2/LH2 Pump-Fed LRB

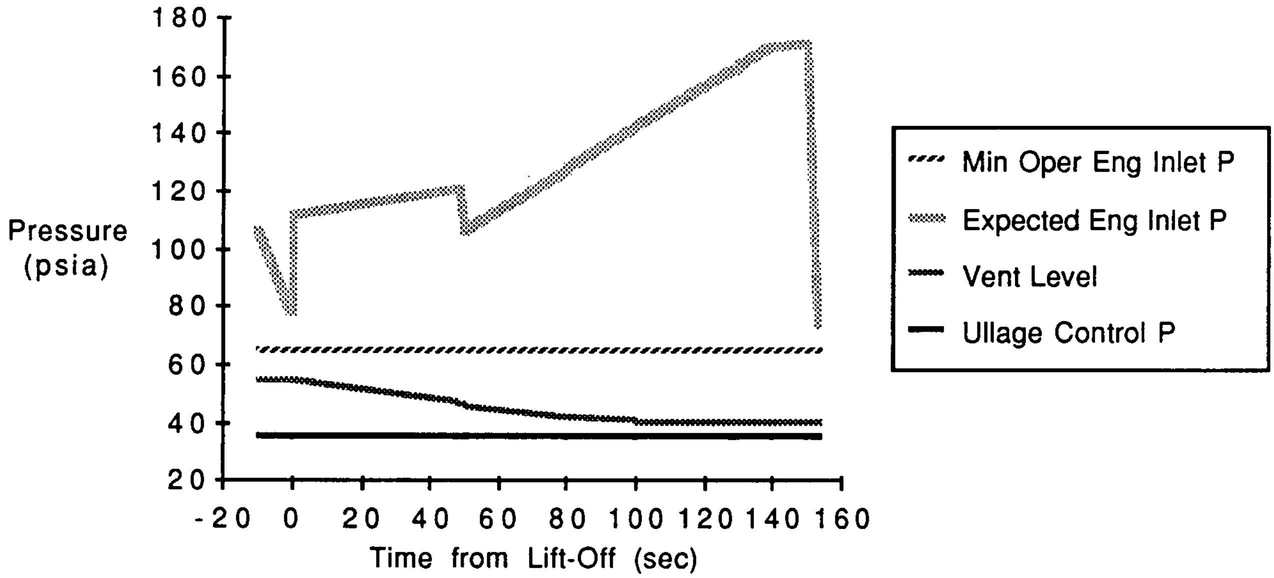

Figure 5.2.2-4a LO2 Tank Ullage Pressure Control Profile for LO2/LH2 LRB

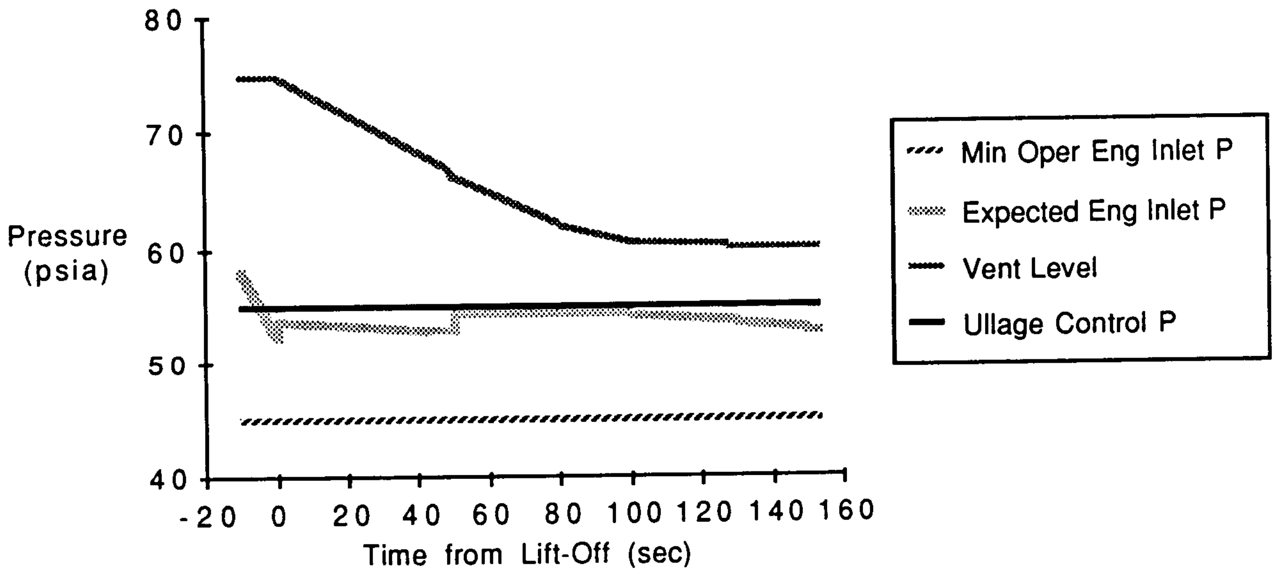

Figure 5.2.2-4b LH2 Tank Ullage Pressure Control Profile for LO2/LH2 LRB

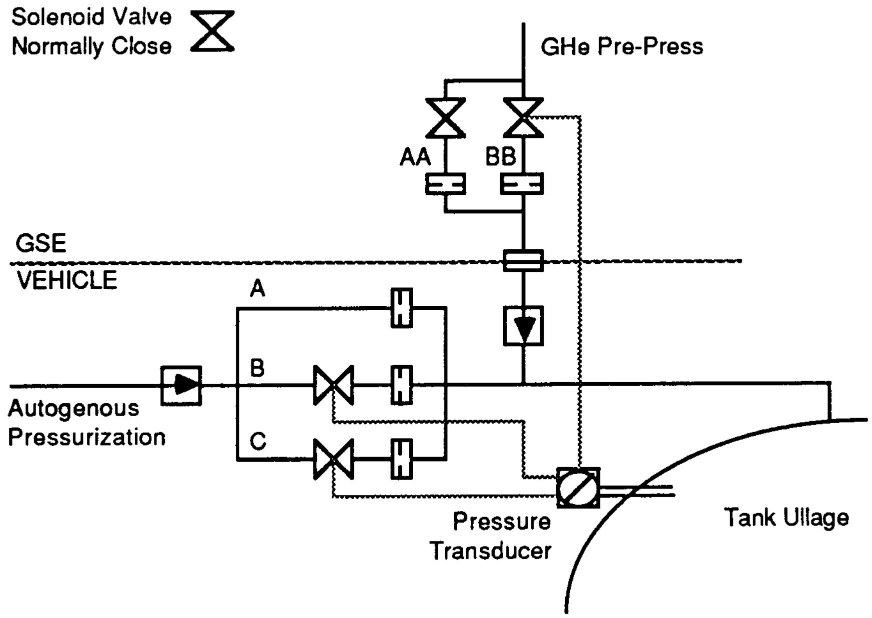

Figure 5.2.2-5 GSE and Vehicle Pressurization System Network for LO2/LH2 LRB

Table 5.2.2-2 Pressurization Orifice Sizing and Valve Operating Levels

Branch/Orifice		LO2 Tank D (in)	LO2 Tank Operating P (psia) Open - Close	LH2 Tank D (in)	LH2 Tank Operating P (psia) Open - Close
Autog. Primary	A	1.3	n/a	0.7	n/a
Autog. Secondary	B	0.7	34 - 35	0.4	54 - 55
Autog. Back-up	C	0.7	33 - 35	0.4	53 - 55
He Primary	AA	0.2	34 - 35	0.4	54 - 55
He Secondary	BB	0.3	33 - 35	0.7	53 - 55

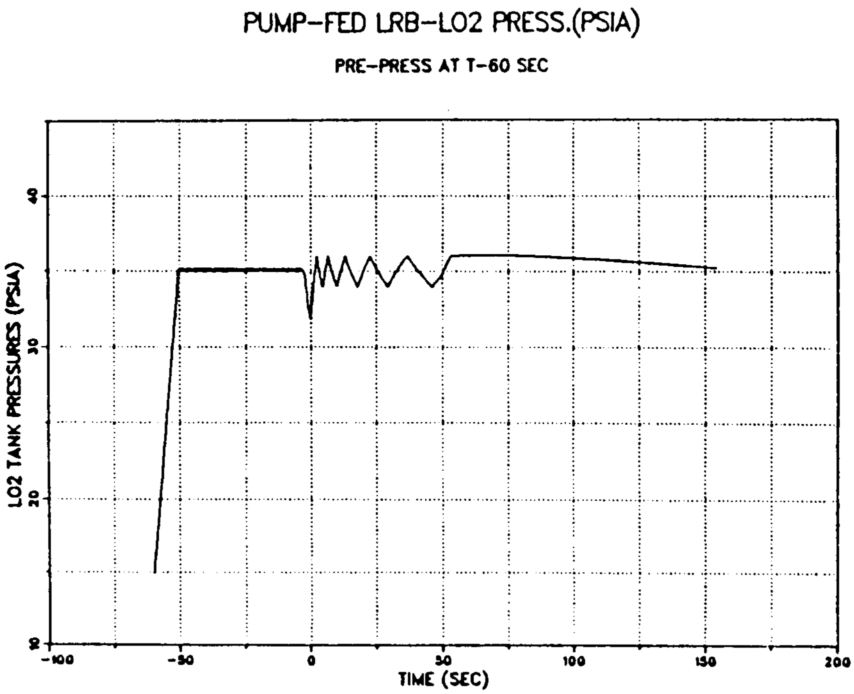

Figure 5.2.2-6a Expected LO2 Tank Ullage Pressure Profile for Nominal Case

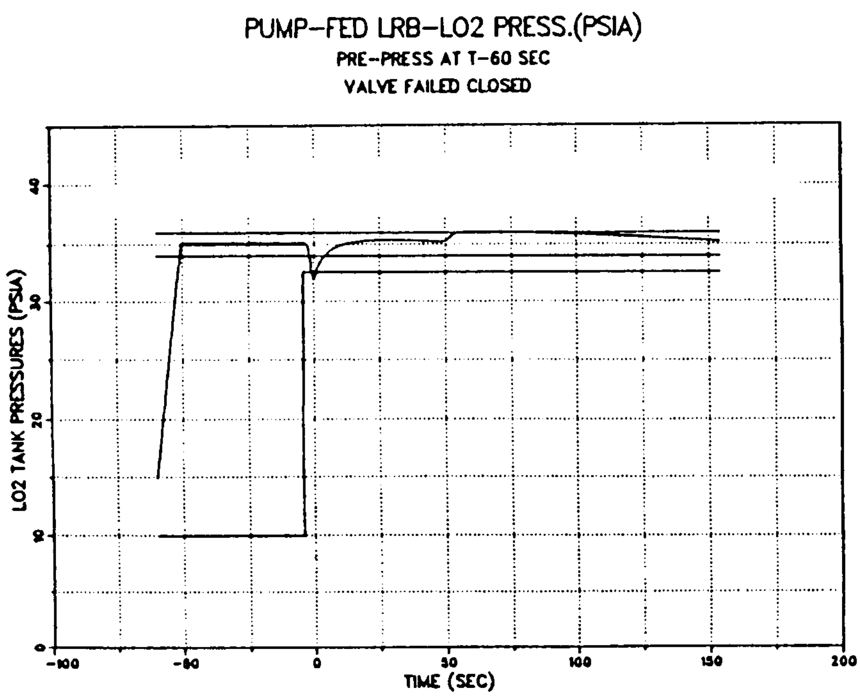

Figure 5.2.2-6b Expected LO2 Tank Ullage Pressure Profile for Valve Failed Close Case

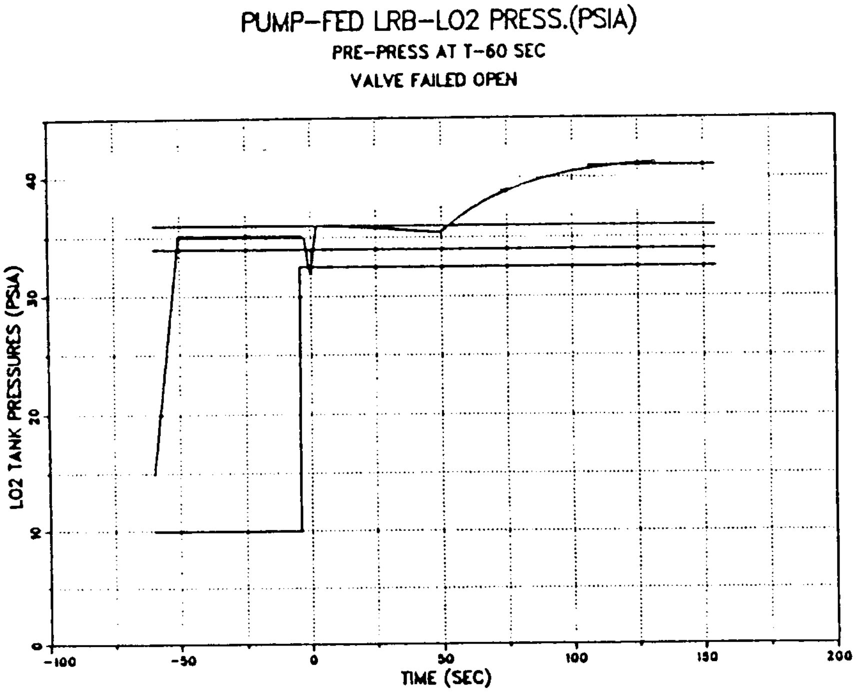

Figure 5.2.2-6c Expected LO2 Tank Ullage Pressure Profile for Valve Failed Open Case

pressurization valve failed closed and open, respectively. Similar profiles for the LH2 tank are shown in Figure 5.2.2-7a (normal), b (failed close) and c (failed open).

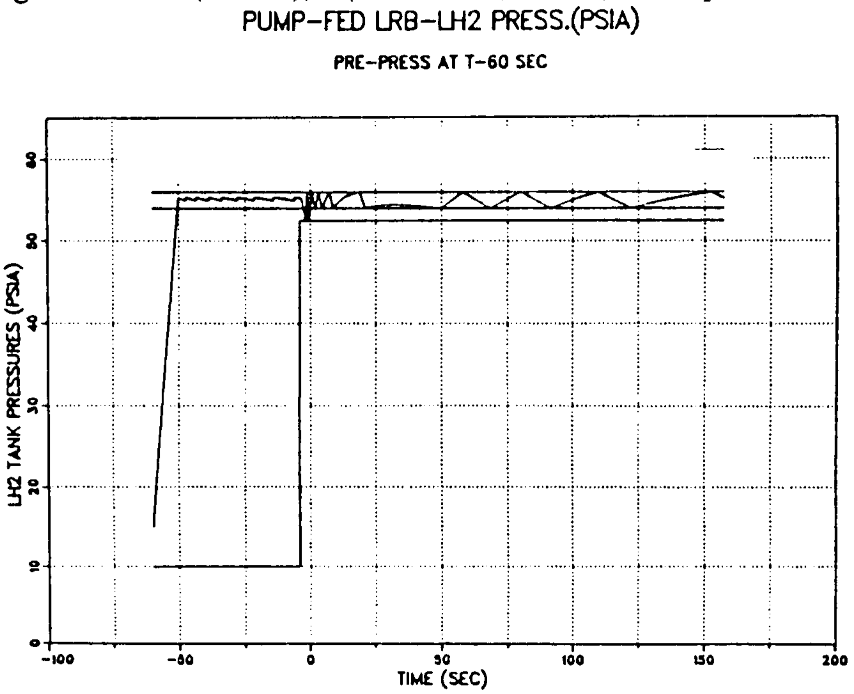

Figure 5.2.2-7a Expected LH2 Tank Ullage Pressure Profile for Nominal Case

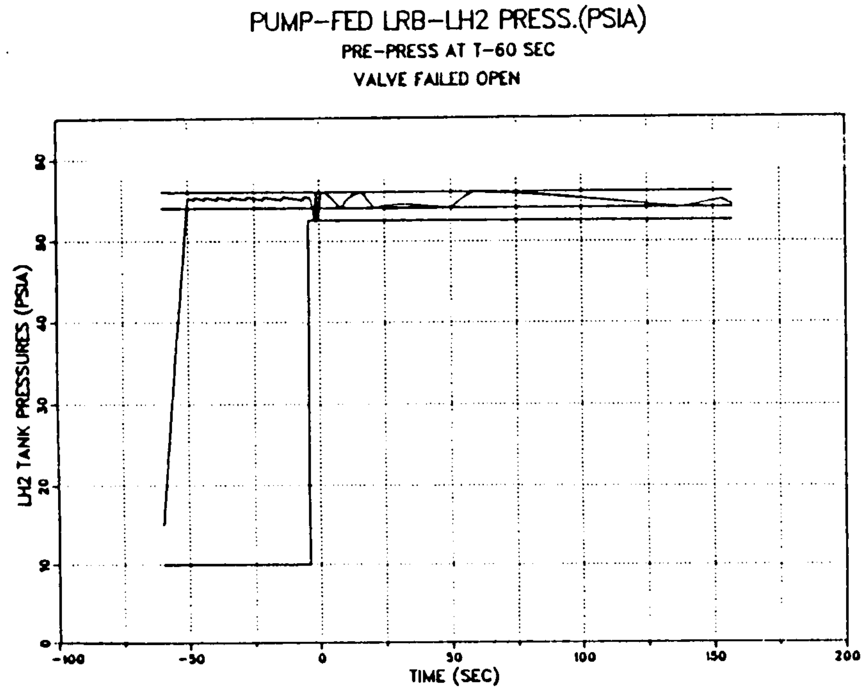

Figure 5.2.2-7b Expected LH2 Tank Ullage Pressure Profile for Valve Failed Close Case

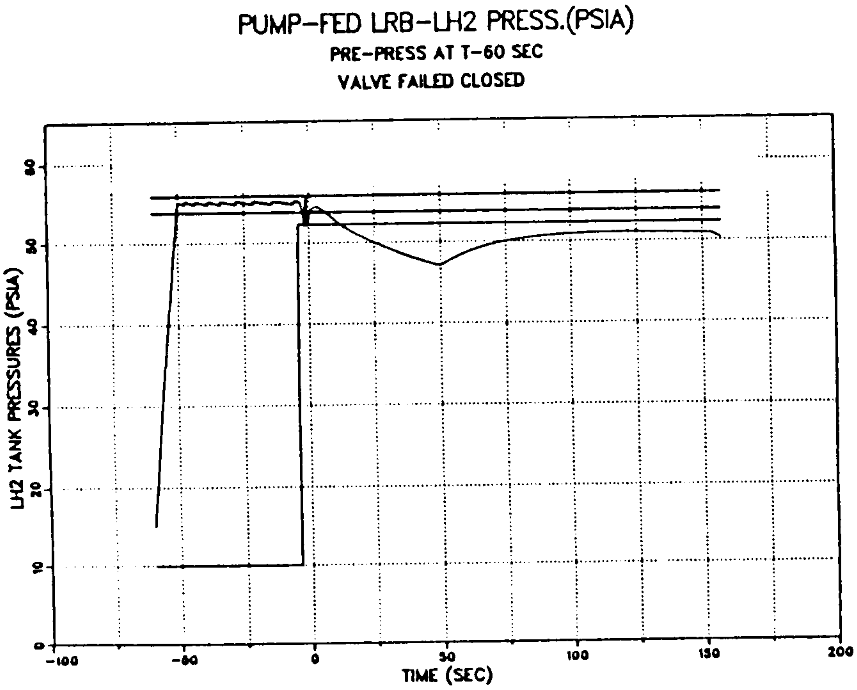

Figure 5.2.2-7c Expected LH2 Tank Ullage Pressure Profile for Valve Failed Open Case

5.2.2.4 Tank Operations. Structurally, the tanks are capable of withstanding a maximum pressure of 70 psig, and they are not pressure stabilized. Throughout pre-launch operations, i.e., tank purging and loading, the tanks will be maintained at a positive gage pressure by helium gas purges and propellant boil-offs.

The prelaunch operations are similar to, and may be parallel with, ET operations. The tanks are initially purged with GN2 to remove moistures. The LH2 tank is then purged with GHe to avoid GN2 condensation. Chilldown of facility transfer line and vehicle line with propellant vapors initiate the tank fill process. The tanks are then slowfill to 2% liquid level, then fastfill to 95% level, then topping slowly to 97-98% level, and the propelllant flow is then limited to replenishing boil-offs and maintaining full liquid level until just prior to engine start.

Shortly before engine start, the vent and fill/drain valves will be closed, and the tanks will then be pre-pressurized with ground helium to the control pressure levels, 35 psia for LO2 tank and 55 psia for LH2. This helium pressurant supply is augmented with autogenous pressurant during engine start and terminated at lift-off from umbilical disconnect. Throughout flight operation, the tank ullages are kepts at the same pressure levels with autogenous pressurization. In case of an overpressure at any time, the vent valves are to provide relief at 40 psig for LO2 tank and 60 psig for LH2.

For a normal flight, depletion of oxidizer, signaled by an ECO liquid level sensor in external feedline, will trigger the booster separation sequence and engine cut-off operations. Similar to engine start, the engines will also be shut-off sequentially, symmetrically in pair with opposite booster's engine, with 120-150 ms delay.

5.2.2.5 Vent and Relief Systems. A vent/relief system is located at the top of each tank, providing relief in case of overpressures in tank during prelaunch and flight operations, as mentioned earlier.

The LO2 tank vent port is located at the aft end to prevent icing, other options of venting need further study. GH2 vent stream will be sent to a flare stack for disposal while the vehicle is at the launch pad. Current facilities require the GH2 vent port to be located at aft end instead of intertank to avoid having swing vent arms on MLP. Having the vent port at aft-end of LRB requires a vent line from the top of LH2 tank to the aft end servicing interface, to be carried on the vehicle.

The vent valves will be open throughout tank purging and loading, and commanded closed for pre-pressurization. The vent/relief valve settings during pre-pressurization and flight are shown in Figure 5.2.2-4a and 4b.

5.2.2.6 Propellant Inventory.

Table 5.2.2-3 Preliminary Propellant Inventory Estimates

ITEM	Oxygen (lb)	Hydrogen (lb)
On Pad Consumption	8,000	1600
Ascent Propellant	592,868	98,811
Shutdown Consumption	2400	960
Liquid Residuals (minimum)	8,200	200
Total In-Flight Pressurant	3,600	925
Fuel Bias	-	2000

5.2.3 ALTERNATE ENGINE SYSTEM - LO2/LH2 SPLIT EXPANDER CYCLE ENGINE. The LO2/LH2 Split Expander cycle engine, an innovative variation of the RL-10 basic expander cycle engine, has been considered for use as an alternate to the LO2/LH2 GG cycle engine. Major reasons for its consideration over other conventional engines are: lower projected cost by Pratt & Whitney, simplicity of the cycle (and hence its higher inherent reliability), and comparatively benign failure modes.

The engines used are expendable with two position step throttle capability at 75% and 100% of the nominal power level. The engine thrust was determined based on GDSS vehicle synthesis runs with engine parametric data provided by Pratt & Whitney.

Pratt & Whitney has investigated engine sensitivity to mixture ratio and engine inlet pressures in terms of cost, weight and chamber pressure, and these impacts are taken into account during vehicle sizing. It should be noted that the performance of a Split Expander cycle is dependent upon the heat transfer areas and flow rates, and hence, in principle, the operating chamber pressure is sensitive to both the mixture ratio and inlet pressures.

5.2.3.1 Engine Feature Selection. Main features of the LO2/LH2 split expander engine are shown in Table 5.2.3-1, and are discussed below.

Table 5.2.3-1 Main Features of LO2/LH2 Split Expander Engine

Cycle	Split Expander cycle	
Boost Pumps	None	
Throttling Capability	Step; 75% and 100%	
Control system	Open loop	
Turbine Start	Boost-strap	
Inlet Ducts	Scissors	
Ignition	Spark Ignition	
Nozzle	80% Bell	
Thrust Chamber Material	Haynes 230	
Gimbal	Head end gimbal; ±6° square pattern	
Delivered life	5 starts	
Burn time	150 sec	
Engine inlet requirements:	LO2	65 psia
	LH2	45 psia

The engine assumes no boost pumps with engine inlet pressure of 65 psia for LO2 and 45 psia for LH2, same as for the baseline gas-generator engine.

An open loop control system is selected corresponding to two-step thrust levels. Haynes 230 is selected as the baseline material for the thrust chamber over 347 stainless steel and the shrouded fuel turbopump impellers design is used over the current RL-10 design as these two modifications provide about 10% higher Pc without increasing the risk.

5.2.3.2 Selected Engine and Pump Characteristics. A cycle heat/power balance was done on the point design resulting from vehicle sizing program. The main engine characteristics are given in Table 5.2.3-2 below.

The pump and turbine characteristics at nominal rated power level are shown in Table 5.2.3-3.

5.2.3.3 Engine Schematic and Operation. Figure 5.2.3-1a and b show the engine flow schematic with propellant flow rates and conditions at various parts of the cycle at nominal

Table 5.2.3-2 LO2/LH2 Split Expander Engine Characteristics

Parameter	RPL
Thrust vac (klbf)	563.9
Thrust SL (klbf)	511.8
Chamber pressure (psia)	968
C* Efficiency	0.996
Nozzle Expansion Ratio	10.6
Engine Mixture Ratio	6.0
Isp vac (sec)	409.5
Oxidizer flow (lb/sec)	1185.9
Fuel flow (lb/sec)	197.4
Engine Length (inch)	114
Nozzle Exit Diameter (inch)	67.2
Dry Weight (lbs)	4140

and minimum thrust levels, respectively. The LO2 pump and the LH2 pump are driven by separate turbines. The LO2 pump consists of a single stage centrifugal pump while the fuel pump is a two stage centrifugal pump.

Here MOV is the main oxidizer valve, JBV the jacket bypass valve, TBV the turbine bypass valve, NSOV the nozzle shutoff valve, and FSOV the fuel shutoff valve. During acceptance testing, the JBV is set to provide proper fuel jacket bypass flow split, the MOV set to provide the proper mixture ratio, and the TBV set to provide proper thrust setting.

Chilldown. During the chilldown period, FSOV, MOV, JBV, NSOV and TBV are in their normally closed position. Cooldown is accomplished by opening the prevalves at the engine inlets, and letting the vapor escape through the feedline into the propellant tanks.

Start. Engine start is accomplished by opening the NSOV, FSOV, and MOV bypass to provide a small LO2 flow to the injector. The JBV is still in its normally closed position. LH2 flows through the combustion chamber coolant passages, in the process changing into gaseous LH2 and initiating turbopump rotation. The MOV bypass actuation is set faster

Table 5.2.3-3 LO2/LH2 Turbo-pump Characteristics

Component	LO2	LH2
Turbine		
Stages	2	2
Efficiency	0.851/0.809	0.868/0.846
Horsepower	3174	15536
Tips speed (ft/sec)	581/582	1530/1545
Shaft speed (rpm)	7026	27841
Inlet temperature (°R)	545	640
Outlet temperature (°R)	525	545
Inlet pressure (psia)	1338	2817
Outlet pressure (psia)	1139	1378
Pumps		
Stages	1	2
Efficiency	0.803	0.789/0.619
Inlet pressure (psia)	65	45
Outlet pressure (psia)	1240	1363/3473
Impeller		
Diameter (in)	13.48	13.79/15.62/15.59
Tip speed (ft/s)	411	1650/1906
Specific speed	1790	1396/538

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than the FSOV so that there is initially oxidizer rich atmosphere in the igniter and the thrust chamber. The ignition occurs when the mixture composition reaches the flammable conditions. Combustion in the chamber causes increase in the wall temperature, and as the turbopumps speed increase, propellants flow rates and the system pressures increase.

When turbopump speed reaches approximately 50% of the steady state level, the JBV is opened allowing bypass around the jacket and the turbines, and providing some turbine backpressure to slow the acceleration. At 60 to 80% of the steady state turbopump speed, the MOV is opened, providing high oxidizer flowrates which results in rapid increase in the

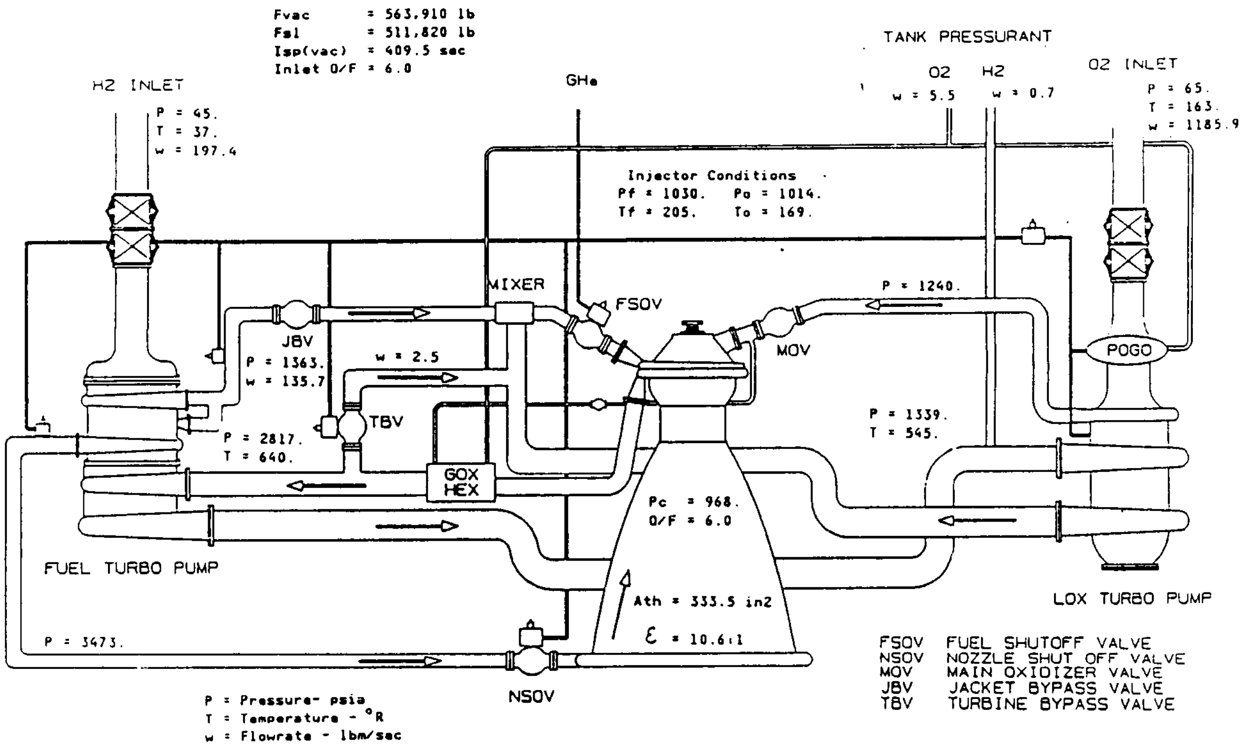


Figure 5.2.3-1a LO2/LH2 Split Expander Flow Conditions at NPL Normal Power Level (100% Throttled)

chamber pressure and mixture ratio. When engine reaches approximately 85% of the rated thrust level, the TBV is opened bypassing fuel flow around the turbines and preventing a thrust overshoot.

A two to three second acceleration from start signal to 95% thrust is expected with a 0.3 second variation.

Steady State. In the steady-state, engine thrust and mixture ratio are controlled by an open loop control system. The valves are actuated by the electromechanical actuators.

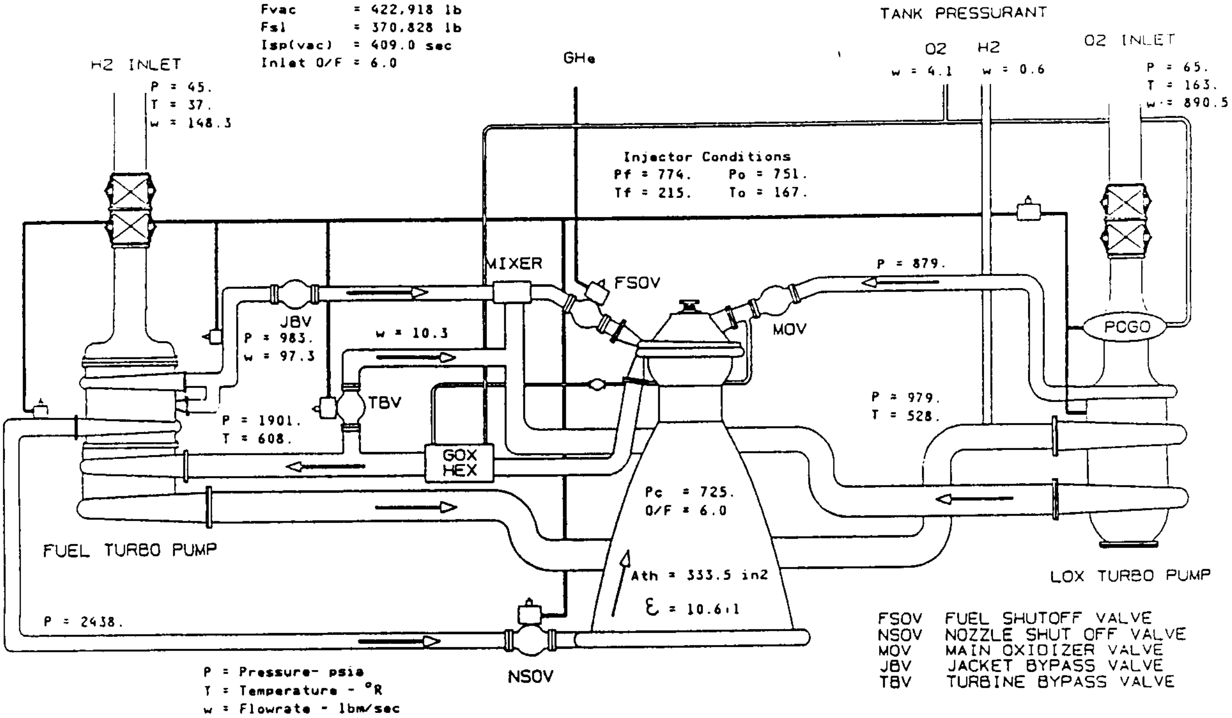


Figure 5.2.3-1b LO2/LH2 Split Expander Flow Conditions at MPL Minimum Power Level (75% Throttle)

Shutdown. Shutdown is accomplished by closing the FSOV. The shutdown time can be extended by controlling the rate of valve closure. Shutting of the FSOV causes a rapid deceleration of the engine.

Abort. If the safety monitoring system indicates a problem, the engine can be shutdown in about 150 milli-seconds. Safe shutdown is possible if ground monitoring indicates a problem during the start sequence such as slow speed buildup, etc.

5.2.3.4 Engine Control. All the LO2/LH2 LRB engines can be step throttled at 75% and 100% of their nominal power level. All the engines receive the same throttle command at the same time. These come automatically from the general purpose computers through the engine controllers. The only manual control of the engine provided is the engine shut-off command and up-throttle command.

Engine thrust level is controlled by utilizing the TBV. To throttle the engine up to 100% thrust level, the TBV will close to the set position and increase the amount of turbine flow and available horsepower. The pumps spin up and pressures increase throughout the engine system until desired chamber pressure (thrust level) is attained. It should be noted that all control is open loop and mixture ratio is maintained to within 3%. All prestart and engine start activities are scheduled by the vehicle and sequenced by the controller.

5.2.3.5 Engine Design Discussion and Engine Layout.

Turbopump. The turbopumps of the Split expander Engine use relatively low cost materials and low cost manufacturing techniques to provide a low cost, reliable engine. The low temperature of the turbines allow the use of forged one piece aluminium disk and blades known as a blisk. Fuel pump impellers are machined from aluminum, although studies are planned to produce cast aluminum impellers to further reduce costs. Pump housings are made from cast aluminum. The oxidizer pump impeller is made of forged 347 stainless steel, and the integral turbine is made of forged aluminum. Figures 5.2.3-2a and b show cross sectional view of the LO2 and LH2 turbopumps, respectively.

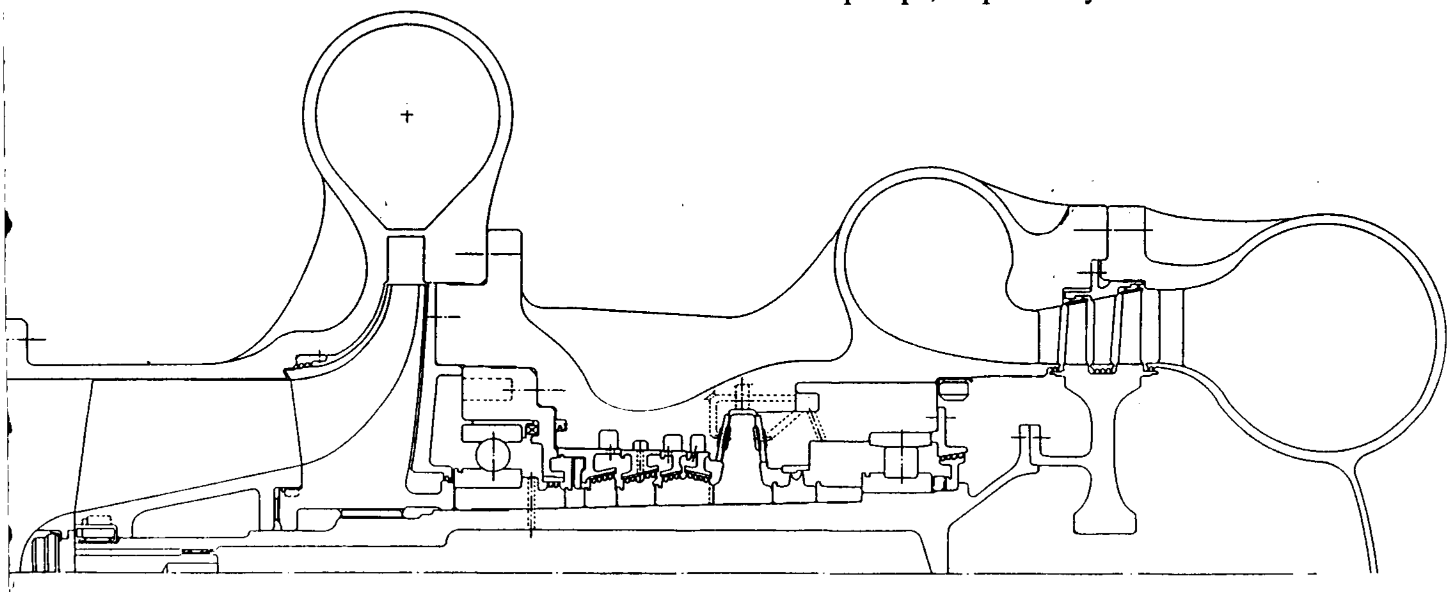


Figure 5.2.3-2a LO2 Turbopump Design

Injector. The main injector, as depicted in Figure 5.2.3-3, is configured of multiple tangential entry oxidizer elements with a concentric annulus of LH2. The injector face plate is a porous material that allows transpiration cooling of the face. This design provides a hollow cone spray of liquid oxygen and is then exposed to high velocity fuel for better atomization.

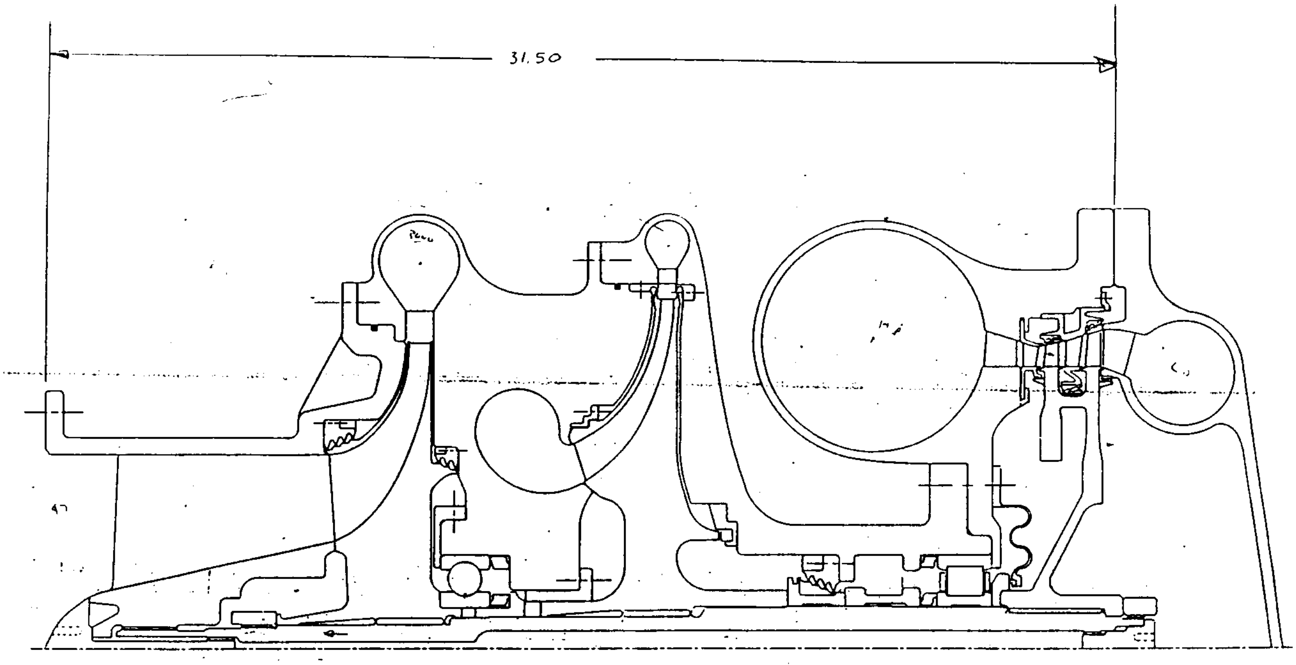


Figure 5.2.3-2b LH2 Turbopump Design

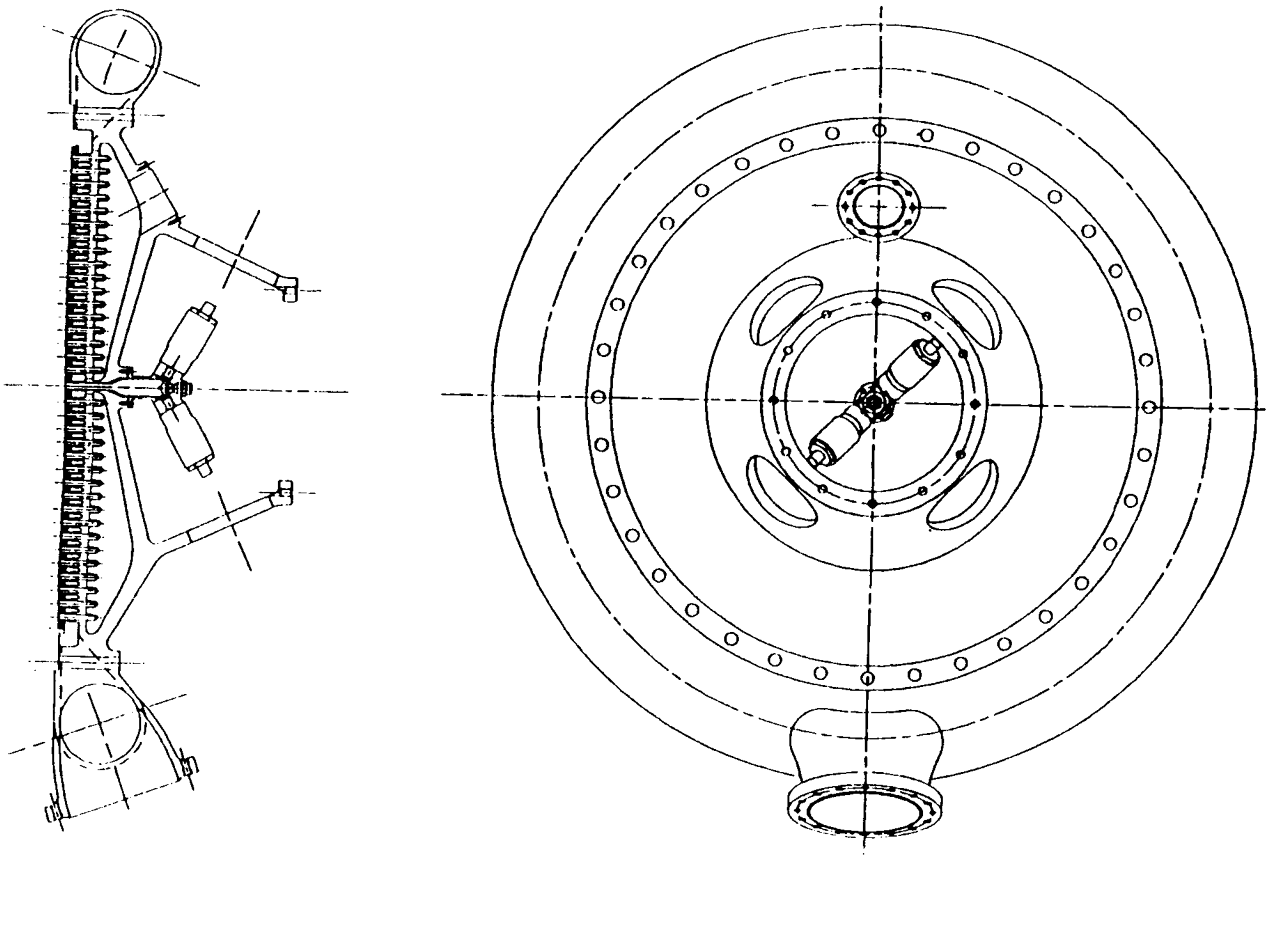


Figure 5.2.3-3 Main Injector Design

Thrust Chambers and Nozzle. The thrust chamber is fabricated from Haynes 230 tubes brazed together. A Haynes 230 jacket will be used to provide structural support to the tubes in combustion chamber area. This support is brazed simultaneously with the tube. The engine utilizes a dual circuit cooling scheme. Both cooling circuits are single pass with the thrust chamber employing counterflow and the nozzle employing parallel flow. The second stage LH2 pump discharge flow enters the thrust chamber at its base which is downstream of the throat. After cooling the chamber the exiting coolant is routed to the top skirt manifold and passes to the end of the nozzle, is collected in a manifold, and then directed to the pump turbine inlet.

Mixer. Figure 5.2.3-4 depicts a mixer concept that will be utilized for mixing hot H2 gas and cold liquid hydrogen in the Split Expander cycle engine. This mixer concept provides efficient, turbulent mixing between the hot and cold fuel flows with a simple, compact configuration and an acceptable pressure drop. The concept has previously been used by P&W on the XLR-129 test stand to mix hot and cold hydrogen, and it is similar to a mixer used on the SSME.

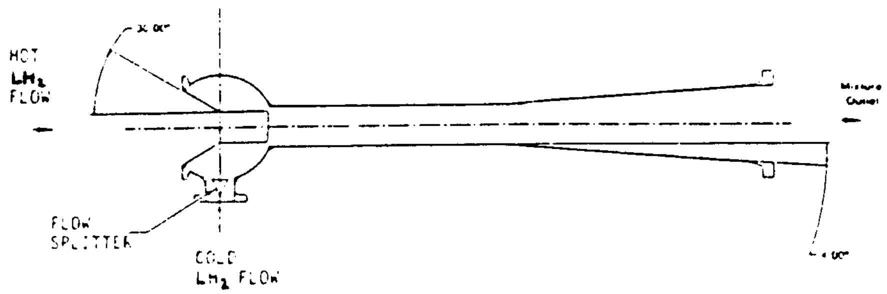


Figure 5.2.3-4 Schematic of Split Expander Mixer

Ignition System. An augumented spark igniter (torch) type is baselined as it can be easily maintained and can be checked out prior to flight.

Pressurization System. The engine is designed to provide gaseous hydrogen at maximum flowrate of 1.5 lbs/sec at about 1300 psia and 520°R for H2 tank pressurization and gaseous oxygen at a maximum flowrate of 7.5 lbs/sec at approximately 1100 psia and 600°R for LO2 tank pressurization. Gaseous hydrogen is bled off the engine between the turbine discharge and the mixer. The gaseous oxygen is produced in a GO2 heat exchanger which utilizes the hot GH2 to vaporize the oxidizer.

The GO2 heat exchanger, Figure 5.2.3-5, consists of an aluminium duct wall that has trip-strips on the turbine exhaust side wall for improved convective heat transfer film coefficients. The oxygen passages are constructed of offset fins that are bonded to the high strength outer wall and the inner aluminum plate. The offset fins enhance the oxygen side convection heat transfer film coefficients which reduces the size of the heat exchanger. The aluminum plate is separated from the duct wall by a highly conductive layer of either dead soft copper or copper powder in colloidal suspension. The copper layer has been incorporated into the design to stop crack propagation from the inner plate to the duct wall.

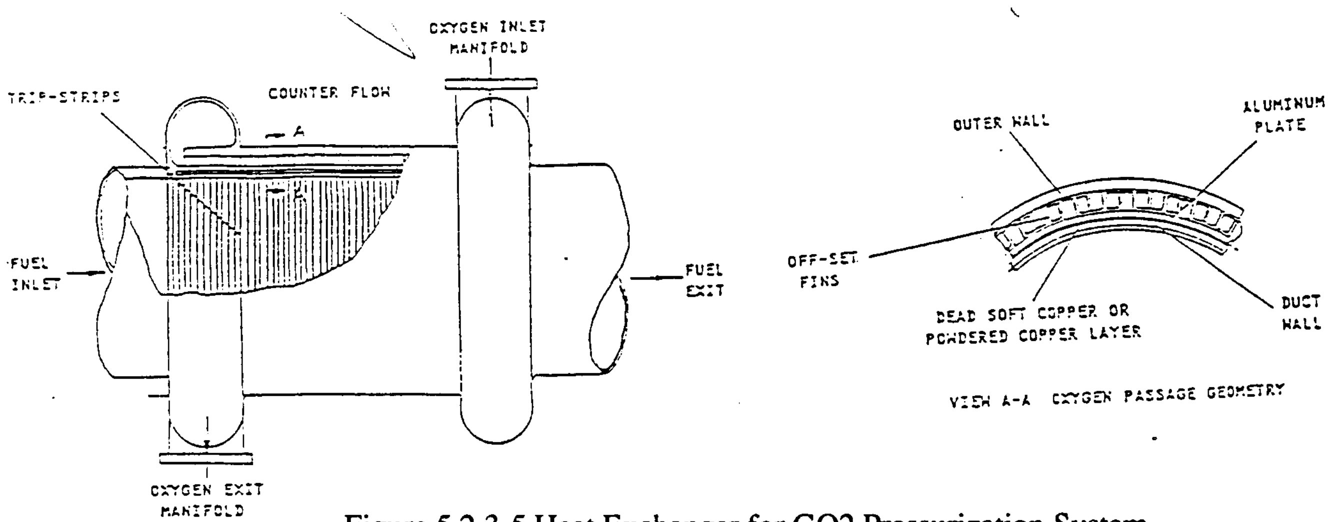


Figure 5.2.3-5 Heat Exchanger for GO2 Pressurization System

Pogo System. The Pogo system baselined is similar the one used currently in the SSME Pogo system which uses a gas filled plenum to isolate engine feedline oscillations from the engine. The pressurized GO2 is supplied by the GO2 heat exchanger, and is used to energize the Pogo suppressor.

Split Expander Cycle Power Margin. All engines independent of cycle face the challenge of reaching rated thrust during their development program. In the development phase, the components rarely meet all of their performance goals in the first engine build. Some modifications and/or minor redesigns are normally needed to achieve rated engine operational capability by the end of the development program. While gas generator and stage combustion chamber cycle engines are plagued with having turbine temperature too high to meet rated thrust, the expander cycle engine could possibly have too low a turbine temperature.

The attainment of rated thrust in expander cycle engines which depends upon the regenerative heat in the nozzle for turbine power, is impacted by both the heat picked up as well as the pressure loss in the nozzle tubes and manifolds. The design point for the expander cycle is currently set with a turbine by-pass margin of about 10% (excess available horsepower). This excess power capability can be expressed in terms of excess chamber pressure or thrust, and is approximately equal to 75 psi margin. This margin is deemed sufficient to meet extreme design uncertainties.

Preliminary Engine Drawing and Layout. A preliminary drawing of the LO2/LH2 engine is shown in Figures 5.2.3-6a and 5.2.3-6b. The arrangement of engines takes into account engine gimbal capability. A quick check was made to see that the plume of one engine does not impinge on the other engine even with failure of one engine gimbal system, and with the expected gimbal angles. It was assumed that the engine with the faulty gimbal system can be brought to neutral position by the back-up system; as described in the TVC system description. These checks were made near BECO where maximum plume expansion would take place, and near max-Q region where maximum gimbal angle can occur.

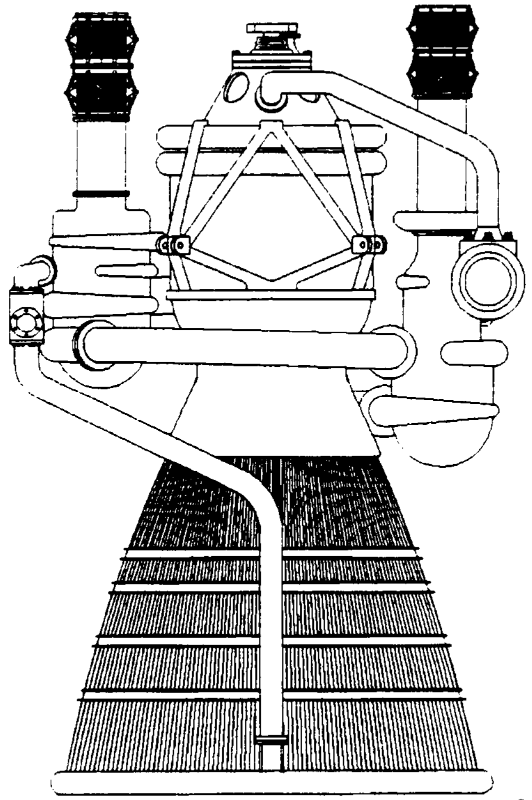


Figure 5.2.3-6a Preliminary Drawing of LO2/LH2 Split Expander Engine

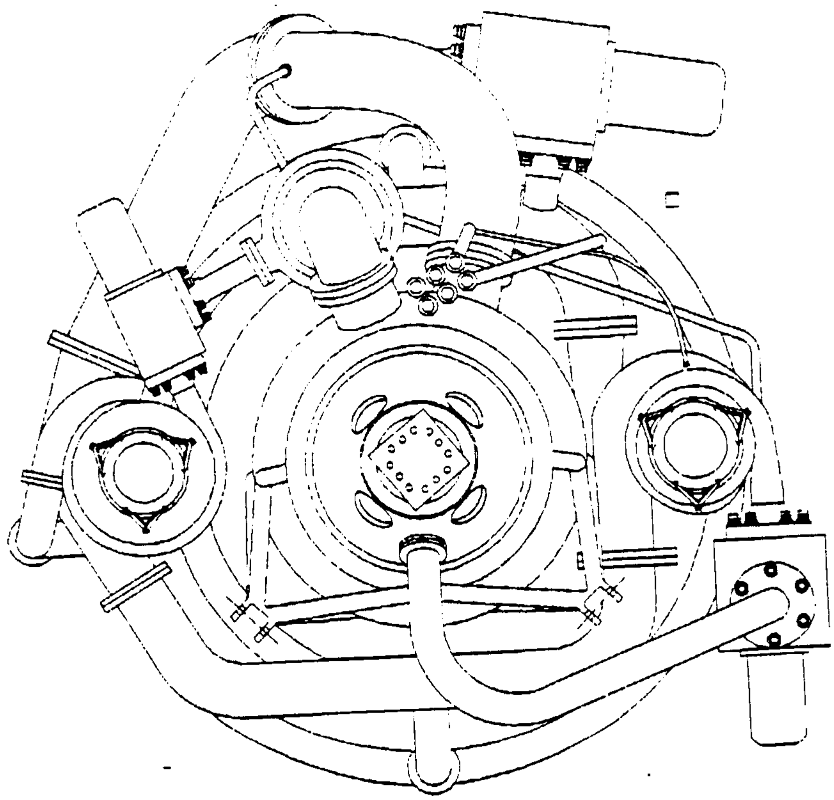


Figure 5.2.3-6b Preliminary Drawing of LO2/LH2 Split Expander Engine

5.2.3.6 Flight Engine Instrumentation. The following flight engine instrumentation for engine health monitoring and for engine control has been identified:

- Chamber pressure
- Fuel pump inlet pressure
- Fuel pump inlet temperature
- Fuel pump housing temperature
- Fuel pump vibration
- Fuel pump speed
- Fuel turbine inlet pressure
- Fuel turbine inlet temperature

Similar instrumentation on the oxidizer side is needed.

5.2.3.7 Engine Interface Requirements

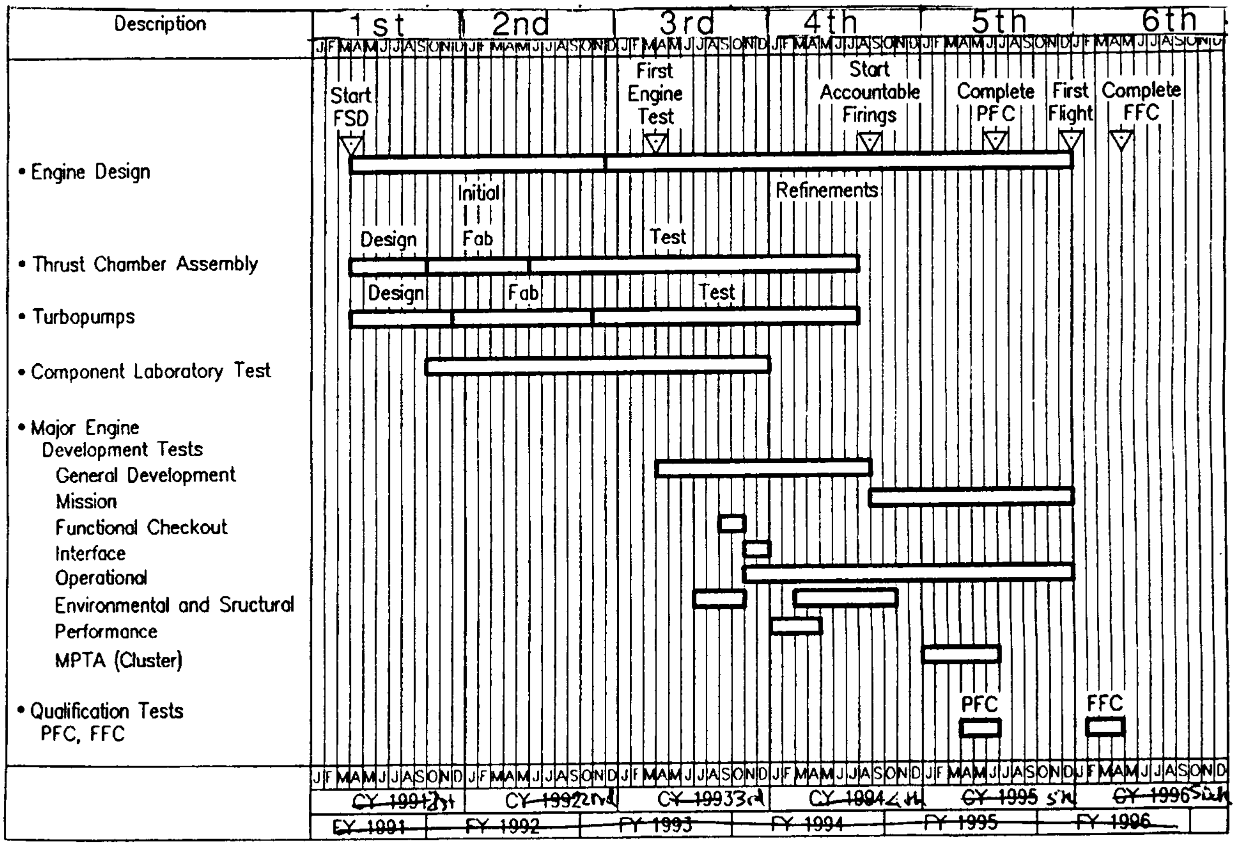
LO2 pump inlet pressure:	65 psia
LH2 pump inlet pressure:	45 psia

LO2 inlet temperature:	164°R
LH2 inlet temperature:	38°R
LO2 inlet diameter:	10 in.
LH2 inlet diameter:	10 in.
GO2 engine pressurization line:	1 in.
GH2 engine pressurization line:	1 in.
Straight line requirement:	TBD
Pump chilldown time:	Approx. 1 hr.
Purge gas:	N2 (flowrate TBD)
Electrical power:	TBD

5.2.3.8 Engine Schedule and Programmatics. The attached development schedule Table 5.2.3-4, is dependent on facilities being available at specified dates. Total development time from full scale development (FSD) start through completion of FCC is 61 months. Included in this schedule are 960 engine firings. To meet first flight goal of end of 1995, following schedule on the the facilities should be met. A comparison is made for these requirements versus the current NASA planning guidelines for the STME program. All test facilities are located at Stennis Space Center (SSC).

Facility	LRB Assumption	NASA Guidelines for STME
Component Test Facility at Stennis Space Center (SSC) for Thrust Chambers and Turbopumps	June 1992	October 1993
First Engine Test Stand (two positions) at SSC	April 1993	June 1994
Two Additional Engine Test Stands at SSC (two positions each)	July 1993	October 1994
MPTA Test Stand	January 1995	Unspecified. Assumed October
Launch Facilities at Kennedy Space Center	January 1996	March 1998
Development Period (Thru FFC)	61 months	90 months

Table 5.2.3-4 LRB Split Expander Schedule



5.2.4 <u>THRUST VECTOR CONTROL (TVC)</u>. For system description and schematic see Section 4.2.3.

A summary of the TVC requirements for the LOX/LH2 pump-fed engine is shown in Table 5.2.4-1.

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Table 5.2.4-1 LRB TVC Requirements for LOX/LH2 Gas Generator Engine

Input Parameters		**Source**
Thrust per Engine	558000 -lbs.**	Rocketdyne
Number of engines on LRB	4	LRB Baseline
Thrust Vector Offset	0.25 -inches	Rocketdyne
Gimbal Block Coeficient of Friction	0.06	Rocketdyne
Gimbal Block Pin Radius	4.44 -inches	Back-calc from R data
Maximum Gimbal Angle	6.00 -Degrees	Specification
Gimbal Rate Required	10.00 -Deg/Sec	Specification
Gimbal Acceleration Required	57.30 -Deg/Sec^2	Specification
Engine Weight	5480 -lbs.	Rocketdyne
Engine Inertia	2230 -lb-ft^2	Back-calc from R data
Distance from Engine C.G. to Gimbal	55.00 -inches	Rocketdyne
Distance from Vehicle C.G. to Gimbal	100.00 -ft	Estimated
Actuator Moment Arm	32.00 -inches	Rocketdyne
C.G. Offset from Centerline of Eng.	0.00 -inches	Assumed
Max Veh Longitudinal Acceleration	3.00 -g's	STS limit
Max Veh Lateral Acceleration	0.30 -g's	STS/LRB Traj Sim
Max Veh Angular Acceleration	3.00 -Deg/Sec^2	STS/LRB Traj Sim
LOX Line Torque	3409 -ft-lbs	Rocketdyne
Fuel Line Torque	2246 -ft-lbs	Rocketdyne
Total Flex Line Stiffness Torque	5655 -ft-lbs	Rocketdyne

Torque Calculations		
Longitudinal Acceleration Torque	7876 ft-lbs	94513 in-lbs
T1 - Due to Engine C.G. Offset	0 ft-lbs	0 in-lbs
T2 - Due to Max Gimbal Angle Offset	7876 ft-lbs	94513 in-lbs
Lateral Acceleration Torque	11736 ft-lbs	140831 in-lbs
T3 - Due to Vehicle Lateral Acc	7535 ft-lbs	90420 in-lbs
T4 - Due to Engine Inertia	117 ft-lbs	1401 in-lbs
T5 - Due to Vehicle Angular Accel	4084 ft-lbs	49009 in-lbs
T6 Thrust Misalignment Torque	11625 ft-lbs	139500 in-lbs
T7 Engine Control Torque	2230 ft-lbs	26764 in-lbs
T8 Engine Block Friction Torque	12374 ft-lbs	148484 in-lbs
T9 Propellant Duct Torque (Given)	5655 ft-lbs	67858 in-lbs
Total Static Torque	36892 ft-lbs	442702 in-lbs
Total Dynamic Torque	14604 ft-lbs	175248 in-lbs
Total Required Torque	**51496 ft-lbs**	**617949 in-lbs**

Peak Power Requirements		
(Using Torque X Gimbal Rate)		
Peak Output power req'd per Actuator	16.3 -hp	12.2 -kW
Peak input Power/Act (sys eff = 53%)	30.8 -hp	23.0 -kW
Peak Power Required per Engine	61.7 -hp	46.0 -kW
Total Peak Required for LRB	**246.7 -hp**	**183.9 -kW**

Actuator Sizing	
Peak Operating Output Force	19311 -lbs
Stall Force	28966 -lbs

** Head-end gimbal point

5.3 AVIONICS

Avionics system architecture is same as previously discussed for LOX/RP-1 Pump-Fed concept. (Refer to Section 4.3)

5.4 H2 PUMP-FED PERFORMANCE AND TRAJECTORIES

5.4.1 NOMINAL MISSION. A description of the nominal mission trajectory simulation can be found in section 8.1.3. Since the ATO mission determined the required size of the LH2 pump-fed LRB configuration, the nominal trajectory simulations were run simply to determine nominal performance.

The following table is a summary of the nominal performance for the LH2 pump-fed configuration.

```
Lift off conditions:

   Weight (lb)                                =   3,585,296.8000
   Payload (lb)                               =   70,500.000000
   Thrust (lb)                                =   5,248,812.9453
   Thrust to weight                           =   1.4639828271
   Initial inertial velocity (ft/sec)         =   1,342.4324022
   Launch site latitude                       =   28.307566153
   Launch site longitude                      =  -80.540959056


Max Q conditions:

   Max dynamic pressure (lb/ft**2)            =   718.38953893
      Time (sec)                              =   50.185833686
      Angle of attack (deg)                   =   4.1760110874
      Altitude (ft)                           =   22,136.587259
      Mach number                             =   1.0736953059
      Q * ALPHA (deg-lb/ft**2)                =   3,000.0026796


LRB separation:

   Staging time (sec)                         =   153.38848370
   Altitude (ft)                              =   180,994.28532
   Dynamic pressure (lb/ft**2)                =   17.428524505
   Angle of attack (deg)                      =  -1.9999997282
   Mach number                                =   5.3496641176
   Inertial velocity (ft/sec)                 =   6,959.7839062
   Inertial flight path angle (deg)           =   16.601665837
   Relative velocity (ft/sec)                 =   5,676.7825208
   Relative flight path angle (deg)           =   20.505086902
   Delta V (ft/sec)                           =   10,091.914840
   Weight after separation (lb)               =   1,448,109.7057
   Remaining ET propellent (lb)               =   1,091,824.7057
   SSME throttle at separation                =   1.0400000000
   LRB throttle at separation                 =  0.75000000000
   Thrust (lb)                                =   1,465,145.3709
   Thrust-to-weight after separation          =   1.0117640709
   Acceleration after separation              =   1.0068414478
   LRB propellent used (lb)                   =   1,383,358.2000
   Geodetic latitude (deg)                    =   28.535679873
   Longitude (deg)                            =  -79.778104626
```

Average back pressure (psi)	=	4.5601766930

MECO conditions:

Time (sec)	=	494.77120641
Altitude (ft)	=	360,687.30195
Inertial velocity (ft/sec)	=	25,871.002694
Inertial flight path angle (deg)	=	0.76698758709
Delta V (ft/sec)	=	30,258.277465
Shuttle & payload perigee (nm)	=	35.167051341
Shuttle & payload apogee (nm)	=	159.91016206
MECO weight (lb)	=	362,488.89798
SSME throttle @ MECO	=	0.77186891532
SSME propellent weight used (lb)	=	1,581,746.1020
ET remaining propellent weight (lb)	=	6,203.8979823
Average back pressure (psi)	=	1.4139359566

Throttle schedules:

Max q throttle down time (sec)	=	48.216722908
LRB throttle setting	=	0.75000000000
SSME throttle @ separation	=	1.0400000000
Start SSME only 3g throttling (sec)	=	449.78555698
SSME final throttle setting	=	0.77186891532

Losses to LRB separation

Total delta V	=	10,091.914840
Steering losses	=	1,893.6790495
Drag losses	=	521.90556410
Gravity losses	=	1,753.6818052
Pressure losses	=	303.73763831

Losses to MECO

Total delta V	=	30,258.277465
Steering losses	=	2,497.7163298
Drag losses	=	525.13496777
Gravity losses	=	2,402.8885558
Pressure losses	=	303.77332326

Min/Max conditions:

Max (+) angle of attack (deg)	=	8.0814399416
Time (sec)	=	6.9141119380
Max (-) angle of attack (deg)	=	-8.6994222051
Time (sec)	=	16.914111938
Max (+) Q * Alpha (lbf-deg/ft**2)	=	3,000.0028332
Time (sec)	=	49.185833686
Max (-) Q * Alpha (lbf-deg/ft**2)	=	-775.37969754
Time (sec)	=	16.914111938

```
Max acceleration (g's)                    =    3.0000000000
   Time (sec)                             =    494.77029594
```

Figures 5.4.1-1 thru 5.4.1-9 show various performance parameters obtained from the LH2 pump-fed LRB configuration's nominal trajectory simulation.

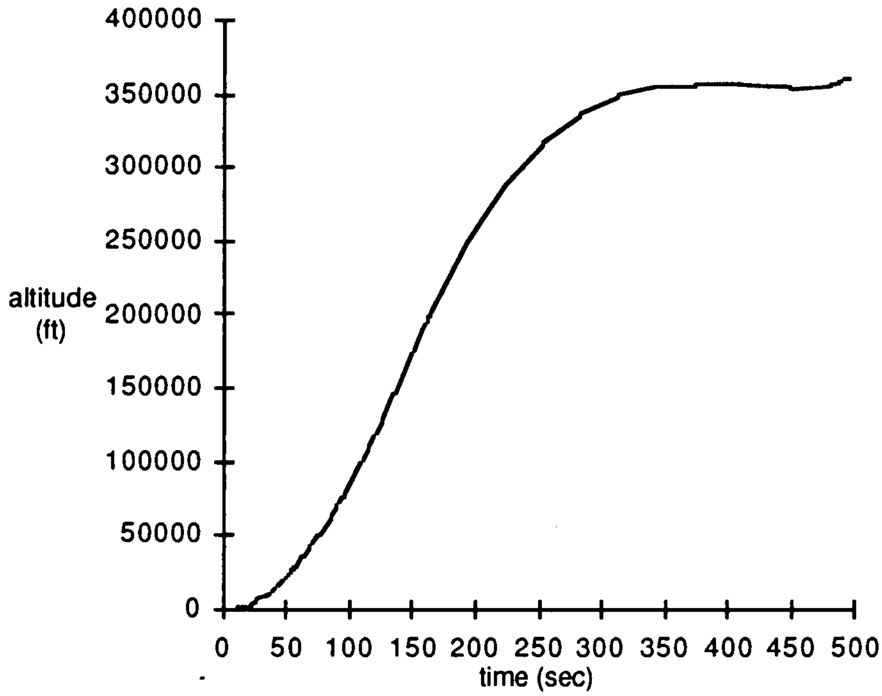


Figure 5.4.1-1 Altitude vs Time

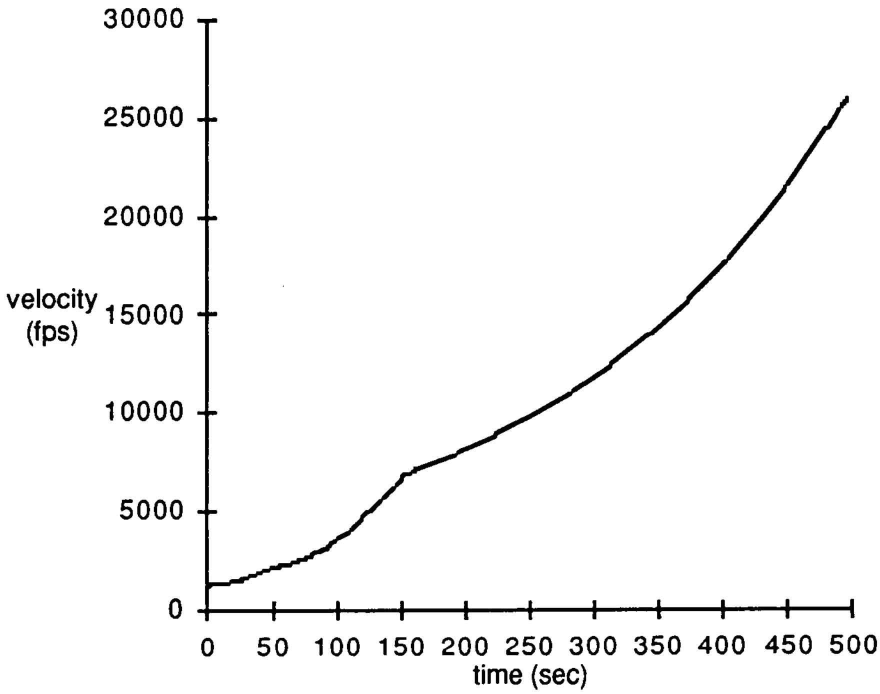


Figure 5.4.1-2 Velocity vs Time

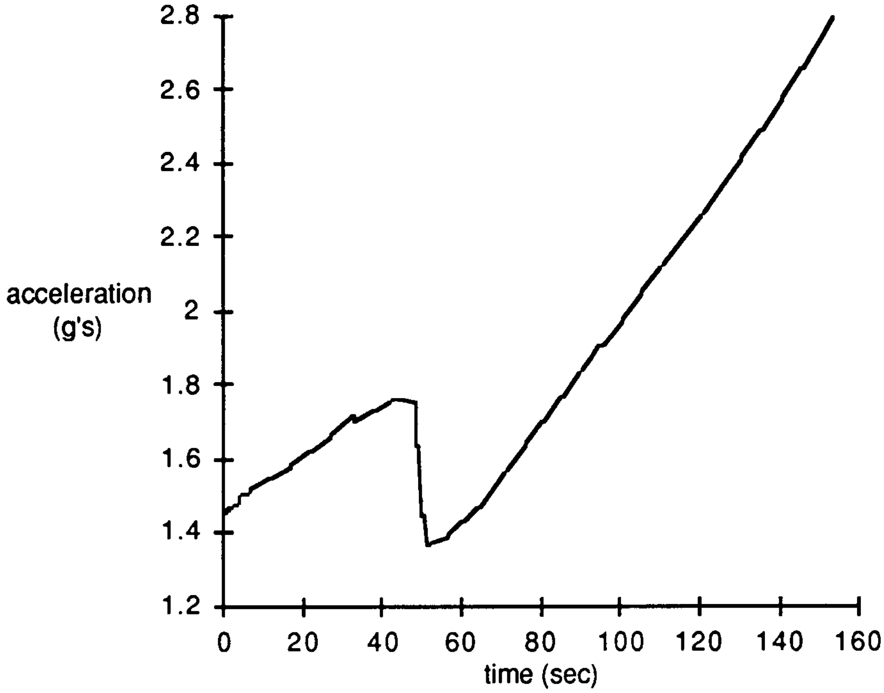


Figure 5.4.1-3 First Stage Acceleration vs Time

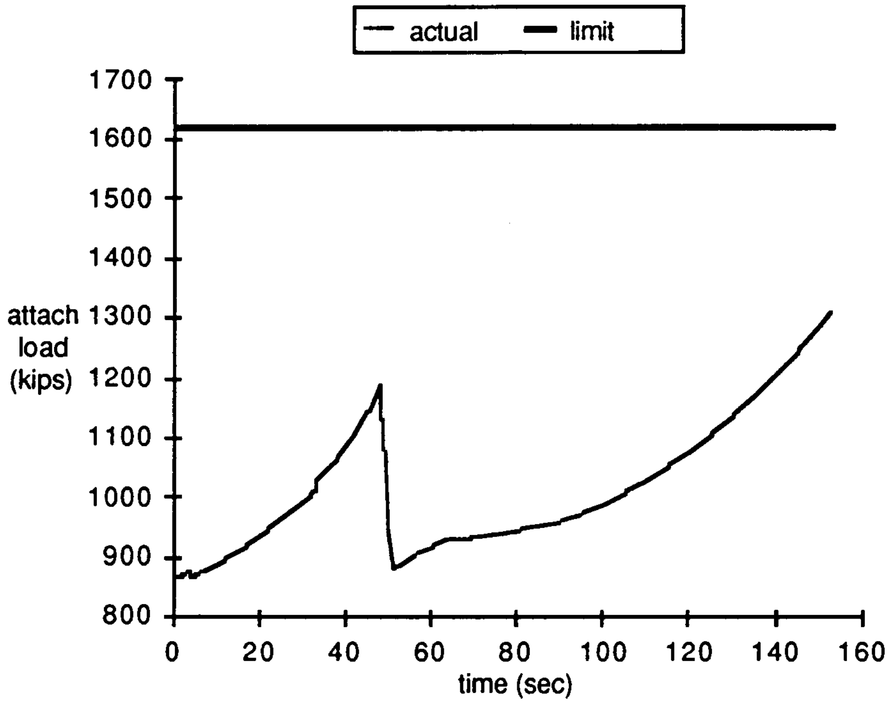


Figure 5.4.1-4 ET Attach load vs Time

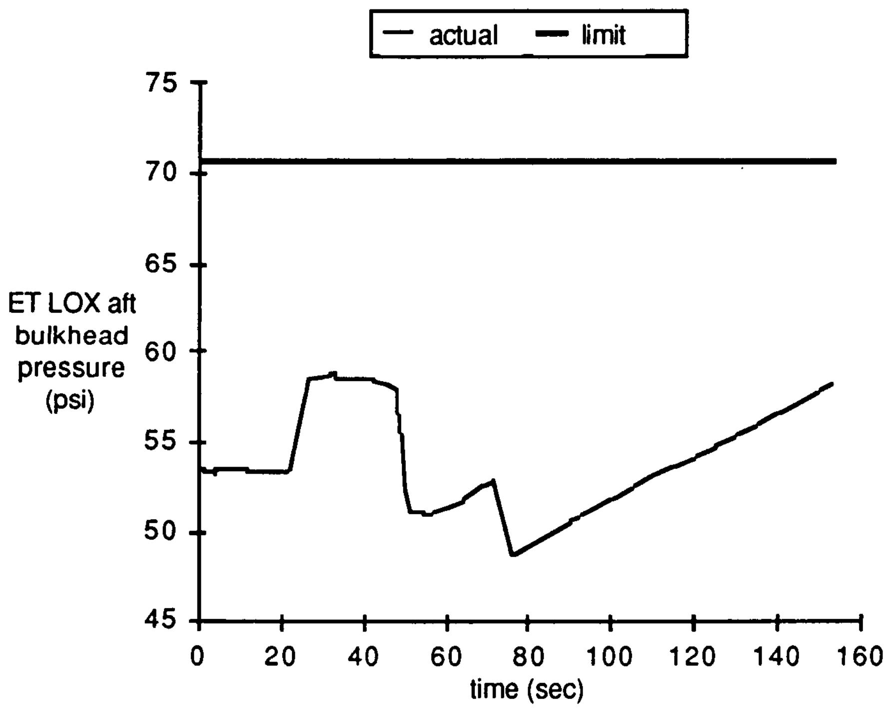


Figure 5.4.1-5 ET LOX aft bulkhead Pressure vs Time

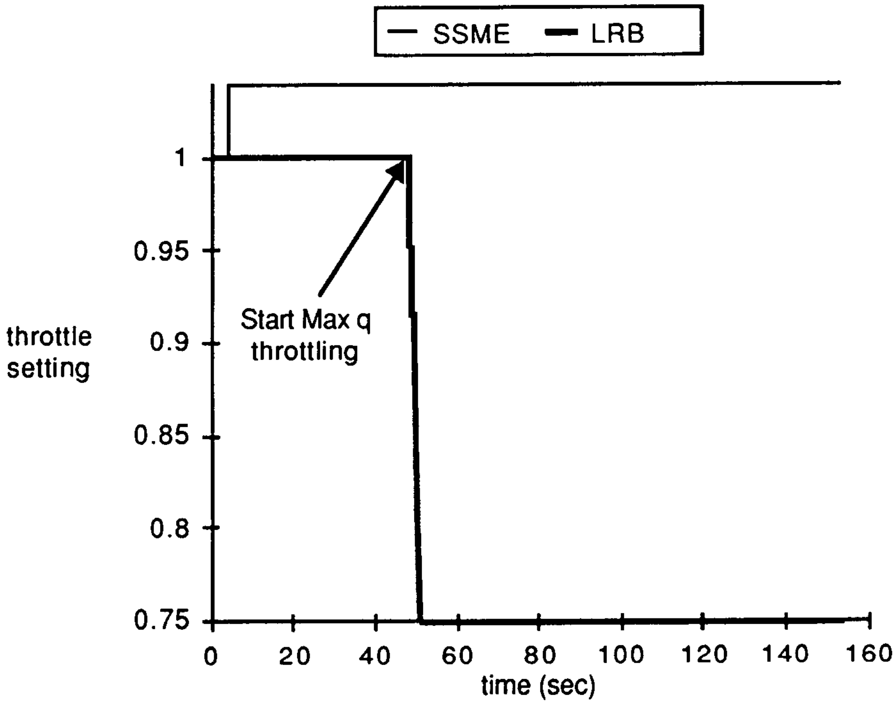


Figure 5.4.1-6 Throttle setting vs Time

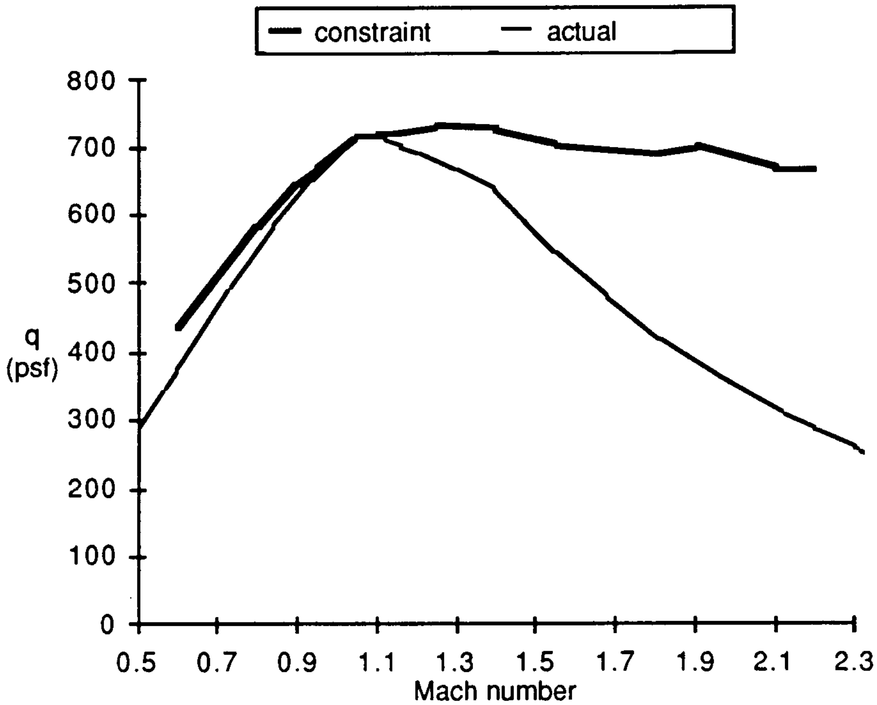


Figure 5.4.1-7 Dynamic pressure vs Mach number

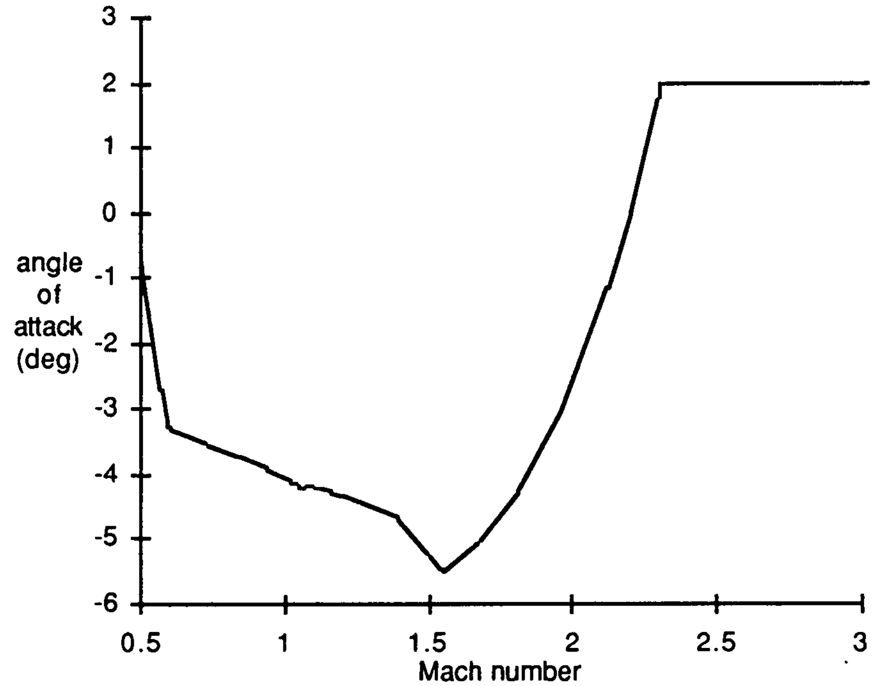


Figure 5.4.1-8 First Stage Angle of Attack vs Mach number

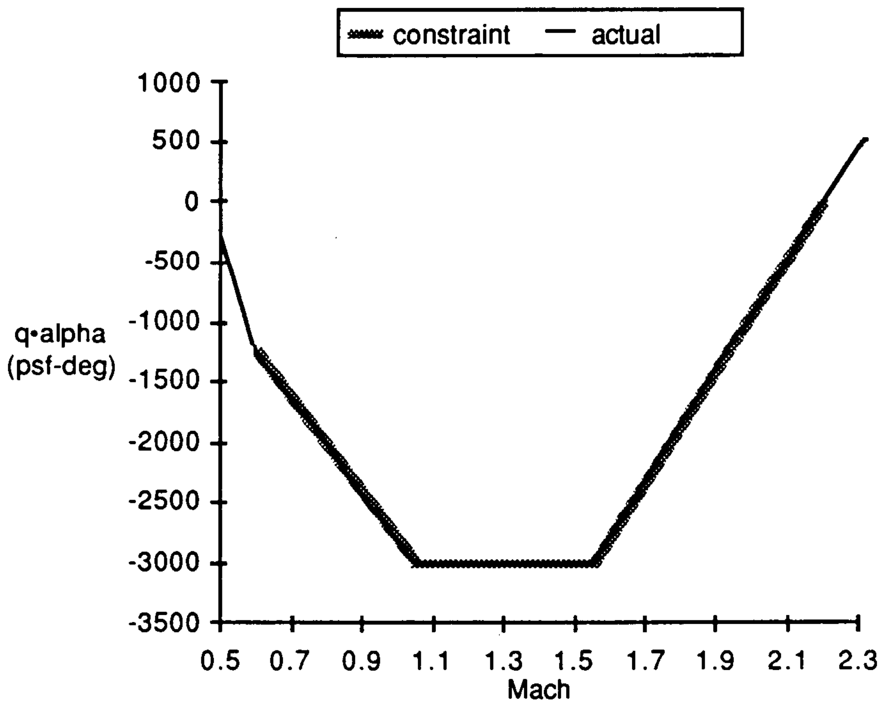


Figure 5.4.1-9 First Stage $q\alpha$ vs Mach

5.4.2 <u>ATO MISSION</u>. A description of the ATO mission trajectory simulation can be found in section 8.1.3. The ATO mission determined the required size of the LH2 pump-fed LRB configuration. The LH2 pump-fed LRB configuration's propellent, thrust, and structure were adjusted by the FASTPASS program until the desired performance was obtained.

The following table is a summary of the ATO performance for the LH2 pump-fed configuration.

```
Lift off conditions:

   Weight (lb)                              =   3,585,296.7627
   Payload (lb)                             =   70,500.000000
   Thrust (lb)                              =   4,733,611.6989
   Thrust to weight                         =   1.3202844875
   Initial inertial velocity (ft/sec)       =   1,342.4324022
   Launch site latitude                     =   28.307566153
   Launch site longitude                    =  -80.540959056


Max Q conditions:

   Max dynamic pressure (lb/ft**2)          =   578.19725398
      Time (sec)                            =   83.590060068
      Angle of attack (deg)                 =   5.1885377979
      Altitude (ft)                         =   35,457.094884
      Mach number                           =   1.2995100208
      Q * ALPHA (deg-lb/ft**2)              =   2,999.9983069


LRB separation:

   Staging time (sec)                       =   169.45669869
   Altitude (ft)                            =   168,117.25208
   Dynamic pressure (lb/ft**2)              =   28.753859988
   Angle of attack (deg)                    =  -1.9999997282
   Mach number                              =   5.3590262981
   Inertial velocity (ft/sec)               =   7,089.1920367
   Inertial flight path angle (deg)         =   15.579271879
   Relative velocity (ft/sec)               =   5,798.5575580
   Relative flight path angle (deg)         =   19.168632070
   Delta V (ft/sec)                         =   10,747.083137
   Weight after separation (lb)             =   1,369,318.0085
   Remaining ET propellent (lb)             =   1,013,033.0085
   SSME throttle at separation              =   1.0900000000
   Engine out LRB throttle                  =   1.0000000000
   Good LRB throttle                        =  0.75000000000
   Thrust (lb)                              =   1,535,516.0532
   Thrust-to-weight after separation        =   1.1213728613
   Acceleration after separation            =   1.1127854916
   LRB propellent used (lb)                 =   1,381,661.5780
   Engine out remaining prop. (lb)          =   1,696.5556137
```

Geodetic latitude (deg)	=	28.515924569
Longitude (deg)	=	-79.787420914
Average back pressure (psi)	=	5.4783550525

MECO conditions:

Time (sec)	=	476.79154717
Altitude (ft)	=	358,107.38005
Inertial velocity (ft/sec)	=	25,626.009050
Inertial flight path angle (deg)	=	0.67795669141
Delta V (ft/sec)	=	30,350.202357
Shuttle & payload perigee (nm)	=	-19.492273301
Shuttle & payload apogee (nm)	=	78.762530695
MECO weight (lb)	=	356,284.99942
SSME throttle @ MECO	=	0.75866249024
SSME propellent weight used (lb)	=	1,587,950.0006
ET remaining propellent weight (lb)	=	-5.81837259233E-04
Average back pressure (psi)	=	1.9474125636

Throttle schedules:

LRB throttle setting (engine out)	=	1.0000000000
LRB throttle setting (good LRB)	=	0.75000000000
SSME throttle @ separation	=	1.0900000000
Start SSME only 3g throttling (sec)	=	422.12465114
SSME final throttle setting	=	0.75866249024

Losses to LRB separation

Total delta V	=	10,747.083137
Steering losses	=	2,418.0154705
Drag losses	=	524.23838517
Gravity losses	=	1,678.1751575
Pressure losses	=	379.53269439

Losses to MECO

Total delta V	=	30,350.202357
Steering losses	=	2,850.1708714
Drag losses	=	530.44685804
Gravity losses	=	2,307.2432090
Pressure losses	=	379.60034851

Min/Max conditions:

Max (+) angle of attack (deg)	=	17.933325691
Time (sec)	=	9.7063945411
Max (-) angle of attack (deg)	=	-9.9607861617
Time (sec)	=	476.79154717
Max (+) Q * Alpha (lbf-deg/ft**2)	=	2,999.9995923
Time (sec)	=	92.716152874

```
Max (-) Q * Alpha (lbf-deg/ft**2)        =  -610.65692834
   Time (sec)                            =   114.39416844
Max acceleration (g's)                   =   3.0000000000
   Time (sec)                            =   473.72358570
Max attach load (kips)                   =   1,287.3759381
   Time (sec)                            =   169.45669869
```

The following table is a summary of the LH2 pump-fed configuration's mass properties obtained when sizing to the ATO mission.

LO2/LH2 PUMP-FED LRB (2)	SUBSYS	SYSTEM	GROUP	VEHICLE
STRUCTURE			71,677.9	
LH2 TANK (2219 skin stiffner)		34,863.3		
Cylinder section	30,512.4			
Bulk head	2,219.4			
ET Attach frame	2,131.5			
LO2 TANK (2219 skin stiffner)		11,836.9		
Cylinder section	7,018.5			
Bulk head	2,223.5			
ET Attach frame	2,594.9			
LO2 TANK SLOSH BAFFLES		290.5		
LH2 TANK INSULATION		1,181.5		
LO2 TANK INSULATION		476.6		
NOSE CAP		2,508.1		
FORWARD ADAPTER		171.8		
INTERTANK ADAPTER		5,324.4		
AFT ADAPTER		10,617.5		
Aft adapter skin	6,858.3			
Aft adapter stringers	433.1			
Aft adapter frame	1,955.5			
Hold down posts	1,370.6			
THRUST STRUCTURE		4,297.3		
4 thrust beams	3,133.1			
4 longerons	533.3			
Engine mount bulk head	294.5			
Skirt aft frame	336.4			
LAUNCH GEAR		110.0		
PROPULSION SYSTEM			35,229.2	
MAIN ENGINES		22,951.9		
ENGINE GIMBAL SYSTEM		2,745.6		
ENGINE PURGE SYSTEM		736.6		
ENGINE MOUNTS		535.7		
MAIN PROPELLANT SYSTEM		8,259.3		
SUB-SYSTEMS			3,943.0	
SEPARATION SYSTEM		1,600.0		
AVIONICS		806.0		
POWER		1,537.0		
ELECTRICAL SYSTEM		0.0		
RECOVERY SYSTEM		0.0		
CONTINGENCY			11,085.0	
DRY WEIGHT				121,935.0
MAIN RESIDUALS			6,916.8	
LH2 FUEL		988.1		
LO2 FUEL		5,928.7		
INERT WEIGHT				128,851.8
ASCENT PROPELLANTS			691,679.1	
LH2 FUEL		98,811.3		
LO2 OXIDIZER		592,867.8		
LRB LIFT OFF WEIGHT				820,530.9
MAIN START-UP FUEL			29,491.0	
LH2 FUEL		5,525.9		
LO2 FUEL		23,965.1		
STEP WEIGHT				850,021.9

TDDP NO: LRBRM-2, REF SODS NO: J-789 ADD 11

STS Weight Summary	SUBSYS	SYSTEM	GROUP	VEHICLE
Orbiter inert				193,693.0
OV 103 (7)			150,811.0	
SSME x 3 inert			20,958.0	

Buoyancy			80.0	
Crew Module			4,361.0	
Non-Prop. Consumables			5,397.0	
RCS Propellent			6,920.0	
Vented after SSME valve close			230.0	
MPS Propellent @ Ignition			4,936.0	
Orbiter lines - usable		2,782.0		
Orbiter lines - unusable		771.0		
SSME x 3 - unusable		1,383.0		
ET inert				76,892.0
ET dry weight			66,623.0	
ET Buoyancy			175.0	
MPS Pressurant			423.0	
Flight Press. Gas			3,730.0	
Usable propellent			5,046.0	
ET FPR		2,219.0		
BIAS		949.0		
Shutdown Propellent		1,878.0		
LH2	609.0			
LOX	1,269.0			
Unusable Propellent			895.0	
ET wet walls		175.0		
LH2 lines & tank, LOX lines		720.0		
Ascent propellent				1,587,950.0
LH2			225,590.0	
LOX			1,362,360.0	
OMS propellent				15,200.0
OMS Fuel			5,708.0	
OMS Oxidizer			9,492.0	
Payload weight				70,500.0

ENGINE PARAMETERS	NOMINAL	ABORT	MINIMUM
NUMBER	4.0		
WEIGHT	5,738.0		
THROTTLE	100.0	100.0	75.0
OXIDIZER FLOW RATE	1,163.4	1,163.4	872.5
FUEL FLOW RATE	193.9	193.9	145.4
VACUUM THRUST	558,058.6	558,058.6	418,544.0
SEA LEVEL THRUST	515,201.5	515,201.5	375,686.9
CHAMBER PRESSURE (psi)	2,250.0	2,250.0	1,687.5
VACUUM ISP (sec)	411.17	411.17	413.31
SEA LEVEL ISP (sec)	379.59	379.59	370.99
MIXTURE RATIO	6.0000		
NOZZLE AREA RATIO	20.000		
X-AREA (in^2)	2,916.2		
THROAT RADIUS (in)	6.8127		
EXIT DIAMETER (in)	60.935		
OVERALL LENGTH (in)	105.47		

VEHICLE PARAMETERS	
GLOW	3,585,296.8
T/W LIFTOFF (nominal)	1.4640
BOOSTER SL TOTAL (nominal)	4,121,612.4
Orbiter SL TOTAL (nominal)	1,127,200.9
T/W LIFTOFF (1 LRB engine-out)	1.2000
BOOSTER SL TOTAL (engine-out)	3,048,352.2
Orbiter SL TOTAL (engine-out)	1,254,002.3

DIMENSIONS/SHUTTLE COORDINATES	LNG.(FT)	STA.(IN)
FUEL TANK SPACING	2.9167	
ENGINE CLEARANCE	5.7500	
EXIT PLANE	2.2	2,504.0
AFT ADAPTER	18.8	2,477.6
AFT FUEL TANK	89.1	2,251.5

Note: performance runs use parametric engine data. See the propulsion section for the engine point design.

INTERTANK ADAPTER	15.9	1,182.7
FORWARD FUEL TANK	27.1	991.5
FORWARD ADAPTER	1.1	666.6
NOSE CAP	23.9	653.6
NOSE TIP	0.0	366.3
TOTAL LENGTH	178.14	
VEHICLE DIAMETER	18.000	
Length/Diameter	9.8967	

NOSE-CAP GEOMETRY				
Nose fineness ratio	1.3300			
Nose bluntness ratio	0.20000			
Conic angle (deg)	17.654			
Nose length (ft)	23.940			
Nose cap spherical radius (ft)	1.8890			
Description	Radius	Height	Area	Weight
Nose cap	1.8000	1.3161	14.885	
Conic section	9.0000	22.624	805.55	
Totals	0.00000	23.940	820.43	2,508.1

Aft Skirt Diameter Inputs/Results	
Nozzle Exit Plane Thickness (in)	5.0000
Nozzle Outsize Diameter (in)	70.935
Engine Gimbaling Length (in)	94.926
Maximum Gimbal Angle (deg)	6.0000
Gimbaling distance pad (in)	5.0000
Gimbaling distance (in)	14.534
Aft diameter (in)	220.87

Propellant tanks (skin stiffner)	Oxidizer	Fuel
Tank diameter	17.9	17.9
Material Density	0.10300	0.10300
Bulkhead		
Radius/Height	1.3784	1.3784
Wall thickness (in)	0.18000	0.18000
Length (ft)	6.5	6.5
Eccentricity	0.68825	0.68825
Surface area (ft^2)	416.4	415.7
Volume (ft^3)	1,098.2	1,095.1
Cylinder section		
Wall thickness (in)	0.31000	0.41000
Inside diameter (in)	215.38	215.18
Length (ft)	27.1	89.1
Surface area (ft^2)	1,526.5	5,017.6
Volume (ft^3)	6,849.3	22,493.3
Totals		
Total tank volume	9,045.6	24,683.5
Total surface area	2,359.3	5,848.9
Occupied volume	8,774.3	23,943.0
Propellent density	70.976	4.3990
Total propellent	622,761.5	105,325.3
Ullage %	3.0	3.0

Figures 5.4.2-1 thru 5.4.2-9 show various performance parameters obtained from the LH2 pump-fed LRB configuration's ATO trajectory simulation.

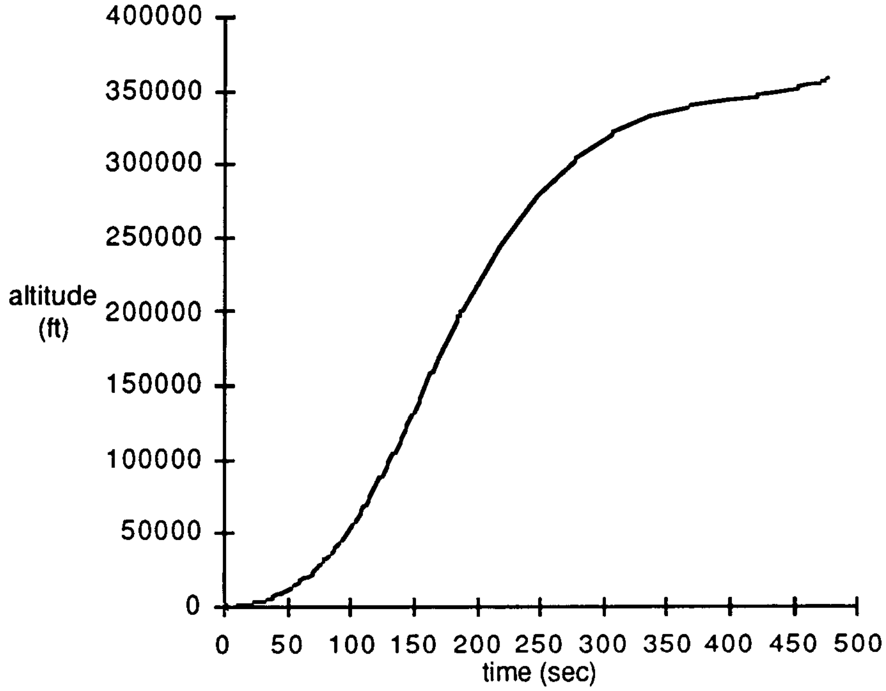


Figure 5.4.2-1 Altitude vs Time

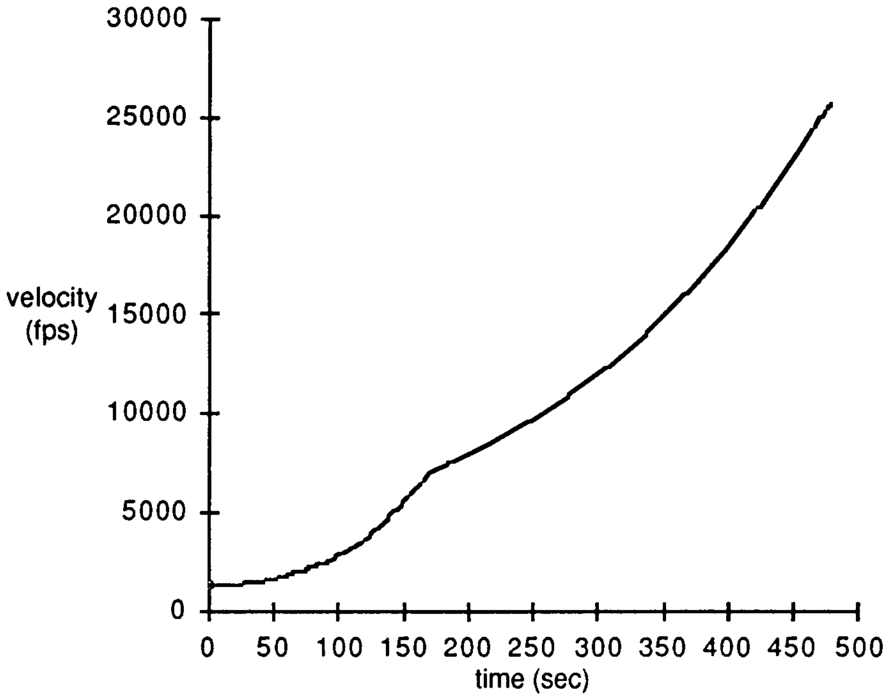


Figure 5.4.2-2 Velocity vs Time

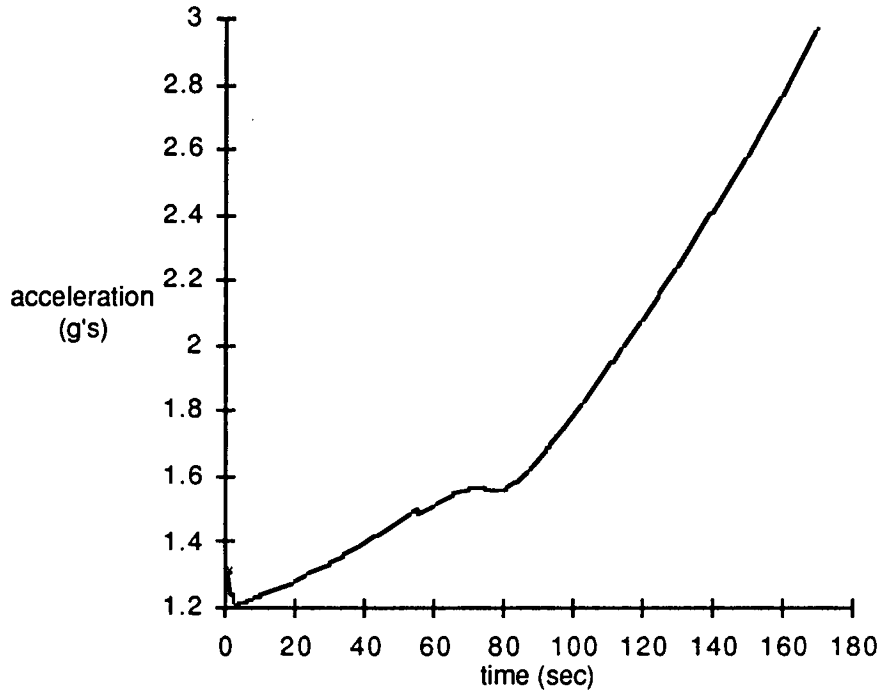


Figure 5.4.2-3 First Stage Acceleration vs Time

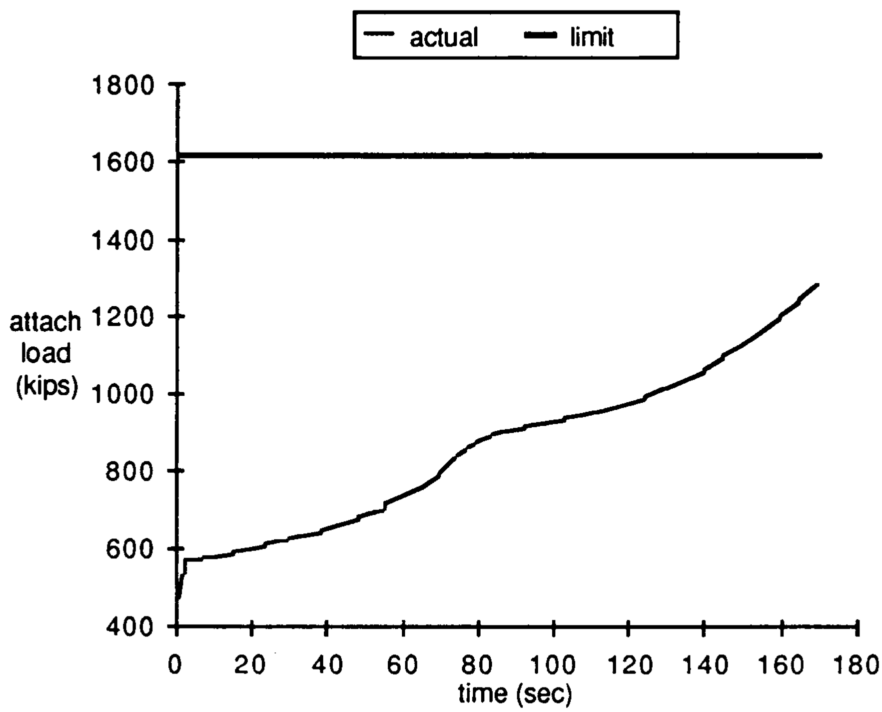


Figure 5.4.2-4 ET Attach load vs Time

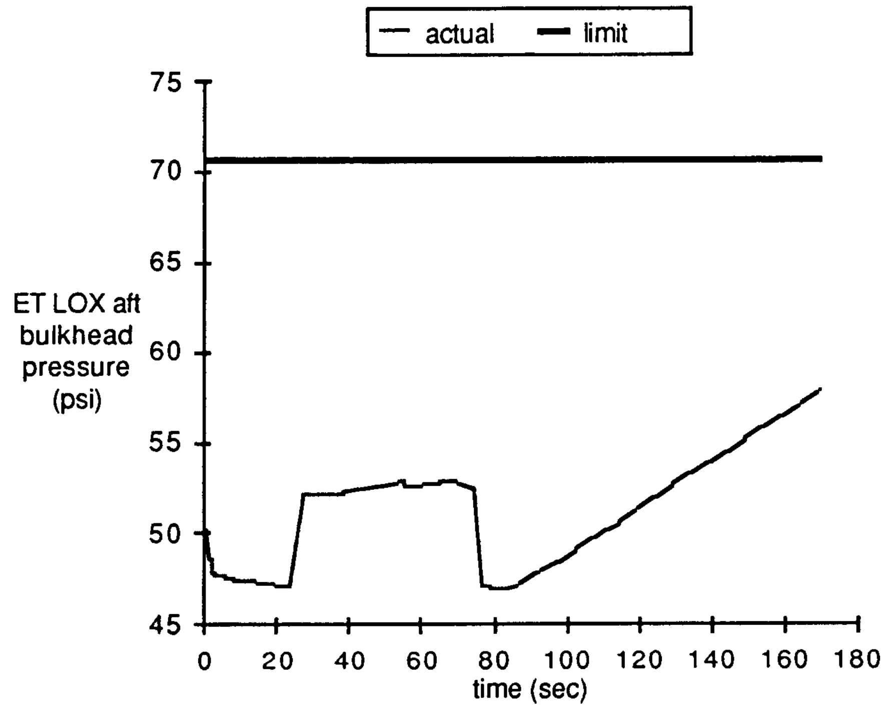


Figure 5.4.2-5 ET LOX aft bulkhead Pressure vs Time

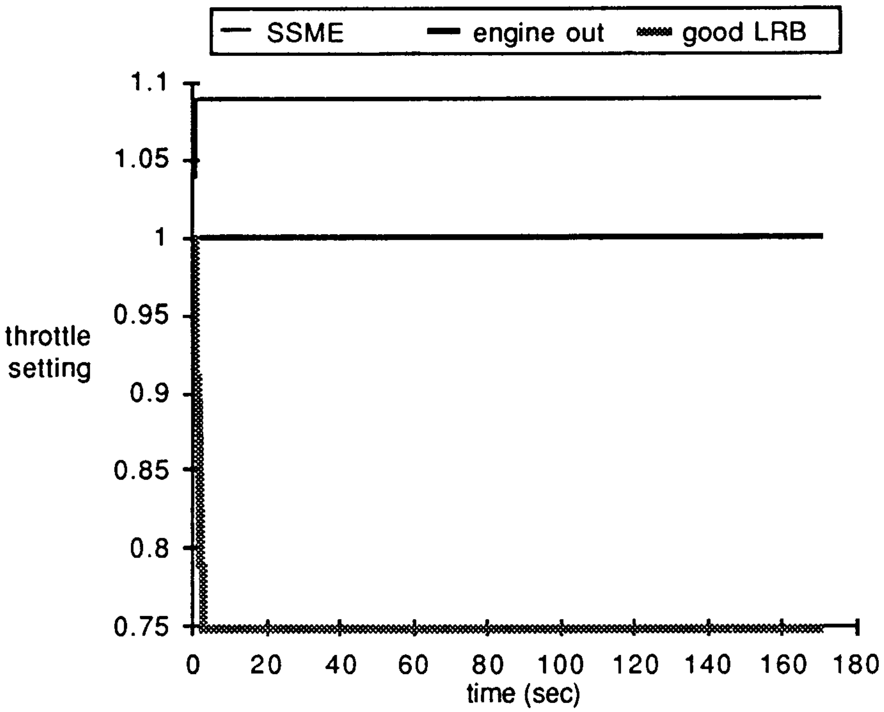


Figure 5.4.2-6 Throttle setting vs Time

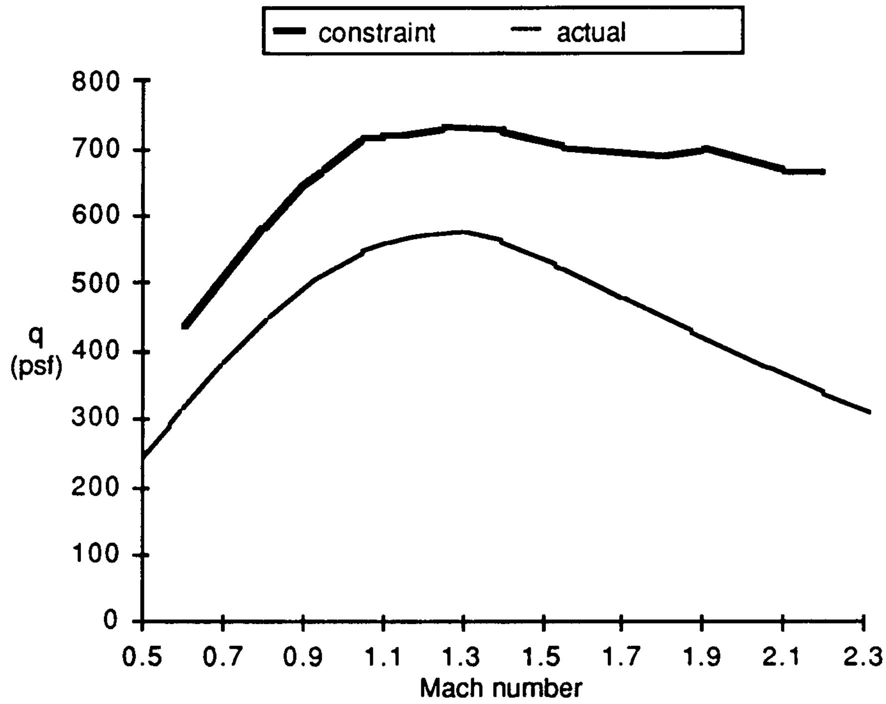


Figure 5.4.2-7 Dynamic pressure vs Mach number

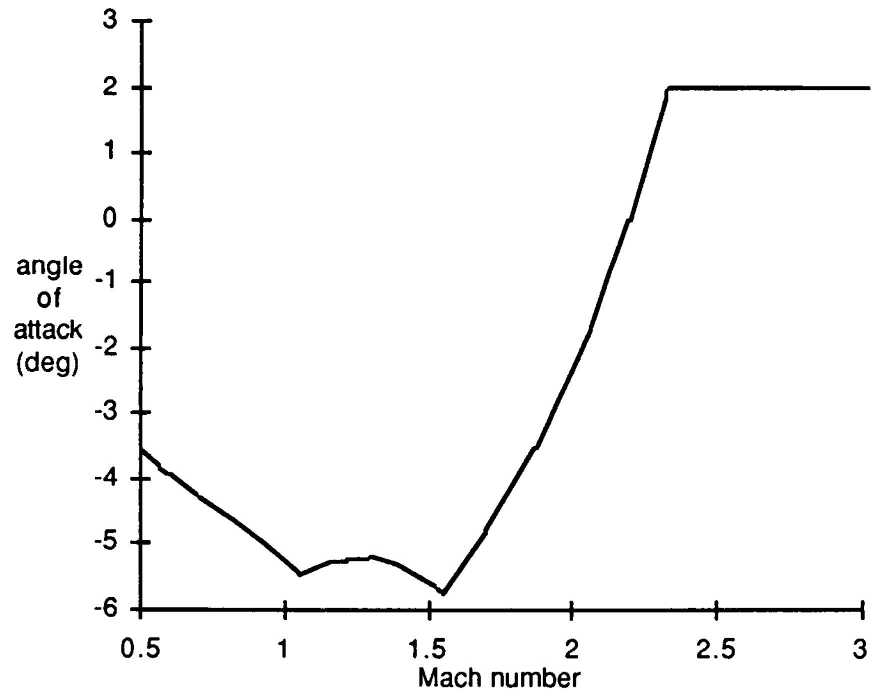


Figure 5.4.2-8 First Stage Angle of Attack vs Mach number

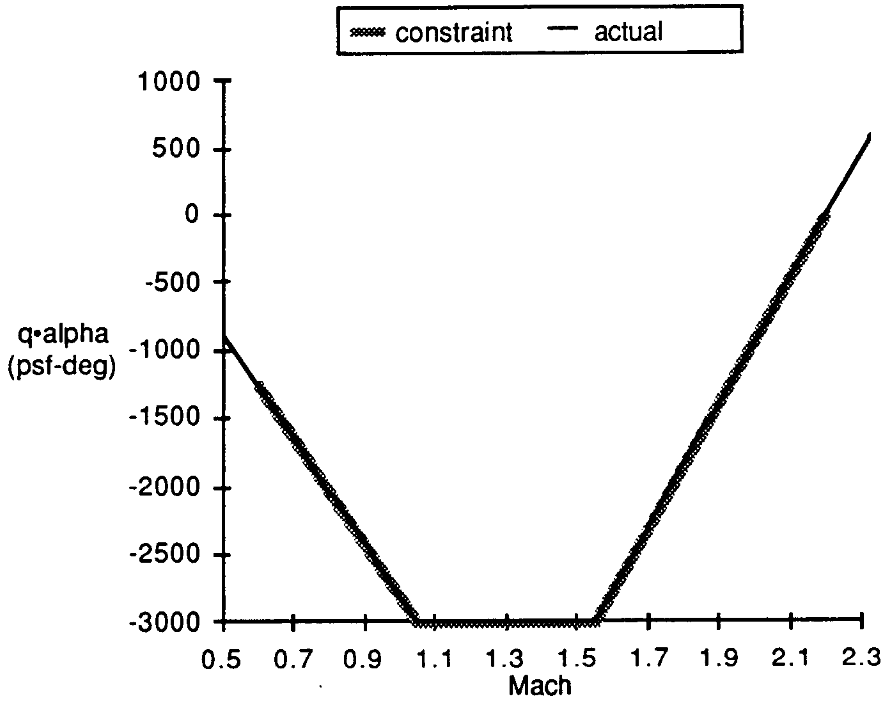


Figure 5.4.2-9 First Stage qα vs Mach

Sizing the LH2 pump fed LRB to meet the minimum ATO conditions with a single LRB engine out at lift off is not without its penalties. The penalties involved in sizing to meet ATO mission requirements are shown in the following table.

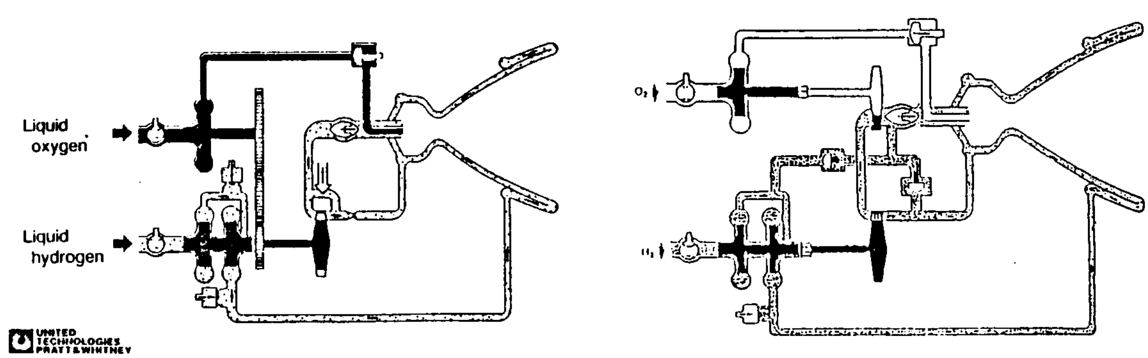
	ATO sizing	Nominal Sizing	Δ	Δ%
LRB Length (ft)	178.1	172.5	5.6	3.2
Dry weight (Klbs)	121.9	118.7	3.2	2.7
Ascent Propellent (Klbs)	691.7	662.7	29.0	4.4
LRB GLOW (Klbs)	820.5	788.0	32.5	4.1
LRB Vacuum Thrust (Klbs)	558.1	543.3	14.8	2.7

SECTION 6

LO2/CH4 SPLIT EXPANDER CYCLE PUMP-FED LRB CONCEPT

The LO2/CH4 Split Expander cycle engine is the last of the three pump-fed LRB engine concepts that was considered in greater detail during this phase of the study. The Split Expander cycle is an innovative variation of the expander cycle used on the RL-10. Major reasons for its consideration over conventional engines are: its lower projected cost by Pratt & Whitney, simplicity of the cycle (and hence its higher inherent reliability), and comparatively benign emergency shutdown.

The basic expander cycle has a thrust limitation for practical chamber pressures at about 100 klbf. This is because the chamber pressure in an expander cycle is determined by the balance between the turbine power available and the power consumed by the propellant pumps; the power available increases slower (proportional to the squareroot of thrust) than the power required (proportional to thrust) with increase of thrust. As shown in Figure 6-1, in a Split Expander cycle the power required to pump the fuel is reduced as only a part of the fuel passes through the upper stages of the pump and the cooling jacket. The power generated, which is a function of heat available, is essentially held constant resulting in practical chamber pressures at higher thrust. Figure 6-2 shows the chamber pressure vs. the thrust for LO2/LH2 and LO2/CH4 engines.



Expander Cycle   Split Expander Cycle

Figure 6-1 Basic Expander Cycle (RL-10) and Split Expander Cycle

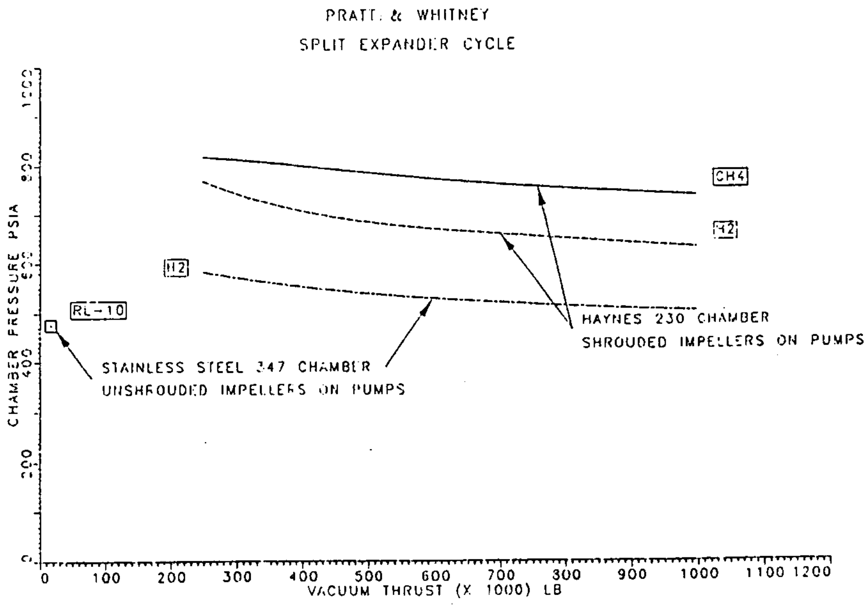


Figure 6-2 Thrust vs.Chamber Pressure for Split Expander Cycle Engine

A Split Expander cycle needs a low boiling point and high heat capacity fuel for its operation. Hydrogen and methane have been identified as the most viable fuels for this engine cycle. These two systems, LO2/LH2 and LO2/CH4, were initially sized using basic 347 stainless steel for thrust chamber material, and LO2/CH4 system was selected as the baseline split expander cycle concept due to its smaller booster size while LO2/LH2 was kept as an alternate to gas generator cycle engine (see Section 5).

6.1 STRUCTURES AND MECHANISMS

6.1.1 VEHICLE CONFIGURATION. The LOX-CH4 Pump Fed Liquid Rocket Booster is shown in Figure 6.1.1-1. Its total length is 150.47 feet. The LOX tank total length is 61.55 feet having a cylindrical section 50.75 feet long capped by two elliptical bulkheads. The LOX tank is attached to the forward adapter as its forward end and is connected to an intertank adapter aft. The CH4 tank total length is 49.05 feet having a cylindrical section 38.25 feet long capped by two elliptical bulkheads. The CH4 tank is attached to the intertank adapter at its forward end and is connected to the aft skirt at its other end. Both tanks are 15.0 feet in diameter.

The intertank adapter is a total length of 15.1 feet. This length is established by the clearance required between the two propellent tanks bulkhead domes to allow packaging of the LOX propellant feed line.

The aft skirt is 18.66 feet long having a forward diameter sized to interface with the CH4 tank. The aft diameter is sized to protect a gimbaled engine from the aerodynamic loads.

The nose cap has a fineness ratio of 1.33 which is similar to that of the solid rocket motor. Since it interfaces with the forward adapter it's length is geometry dependent. There are no packaging constraints since no recovery system packaging is required. The nose cone is 19.8 feet long.

The exit diameter of the engine nozzle protrudes 4.92 feet below the aft skirt. This provides the same reference station for both the solid rocket motor (SRM) and the Liquid Rocket Booster (LRB).

The locations for the external tank (E.T.) to LRB attachments are the same stations as that of the ET to SRM. The forward attachment is at the LRBs forward adapter. This adapter has a total length of 3.0 feet.

The structural design details of the LOX-CH4 Pump Fed Liquid Rocket Booster are identical to that of the LOX-RP-1 LRB.

A weight summary for the LOX-CH4 Pump Fed Liquid Rocket Booster is provided in Section 6.4.1.

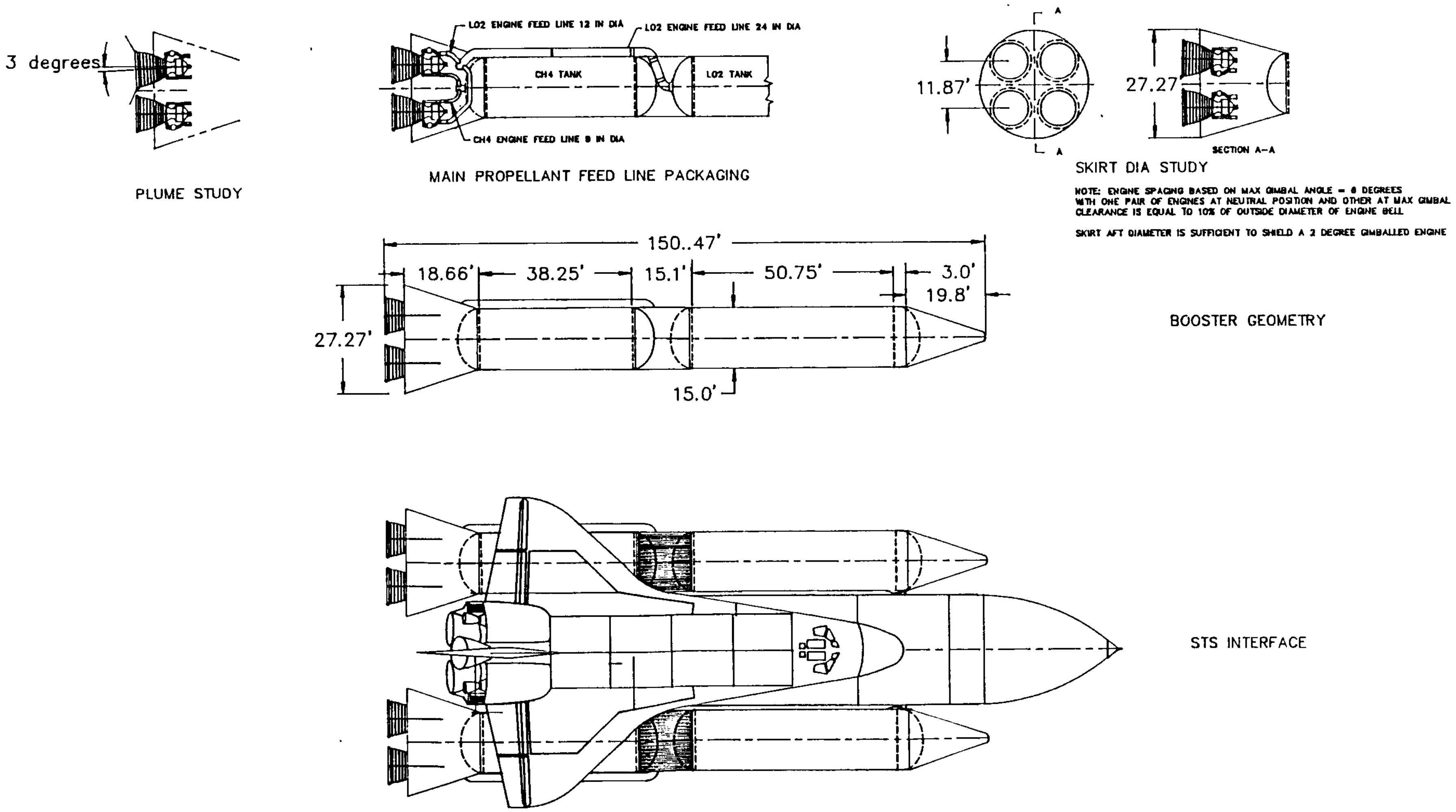


FIGURE 6.1.1-1. LO2 / CH4 LIQUID ROCKET BOOSTER

6.1.2 SEPARATION SYSTEM. Efforts to define the separation system were conducted during the initial LRB study phase. During the follow-on extension, separation analyses were not updated to reflect resizing of the LOX/CH4 pump-fed booster. Thus, the results which follow require update and are presented primarily to show trends and typical designs.

The separation system definition for the LOX/Methane pump-fed configuration is identical to the system designed for the LOX/RP-1 pump-fed booster (see section 4.1.2). The two configurations have nearly the same mass properties and aerodynamic characteristics.

The LOX/Methane pump-fed booster separation system has been initially sized for nominal ascent staging. As the LOX/Methane pump-fed booster design matures, RTLS abort coverage capabilities will be more thoroughly examined; if results show that it is possible to conduct a RTLS abort prior to nominal staging, simulations will be conducted to determine if the BSM quantity used for nominal ascent staging is sufficient for RTLS abort needs. If not, the number of BSMs will be increased accordingly.

Nominal ascent staging of the LOX/Methane pump-fed booster is designed to occur at initial conditions of:

Mission Elapsed Time	= 132.8 Seconds
Altitude	= 147,000 Ft
Mach	= 5.02
Dynamic Pressure	= 75 PSF
Inert Weight	= 123,000 Lbs

The LOX/Methane pump-fed booster separation system design requires 7 BSMs. This is based on computer simulation results for nominal ascent (design case) booster separation. Design case staging conditions include: body rates of 5 deg/sec pitch, 2 deg/sec yaw, and 2 deg/sec roll; alpha = 10 degrees; and beta = 10 degrees. The corresponding booster separation system weight is on the order of 1,400 lbs.

The 7 BSMs used are distributed with 3 packaged in the nose cone and 4 placed on the aft skirt. The same BSM orientation for the LOX/RP-1 pump-fed booster is also used for the LOX/Methane booster configuration.

Separation plots for nominal ascent (design case) staging (Figures 6.1.2-1 through 6.1.2-3) indicate clean separation. Conclusions presented about separation of the LOX/RP-1 pump-fed booster (see section 4.1.2) also apply to the LOX/Methane pump-fed booster.

LRB SEPARATION - CH4 BOOSTER
ALPHA = BETA = 10.0 , PQR = 5,2,2
NUMBER OF BSM'S = 3 FWD , 4 AFT

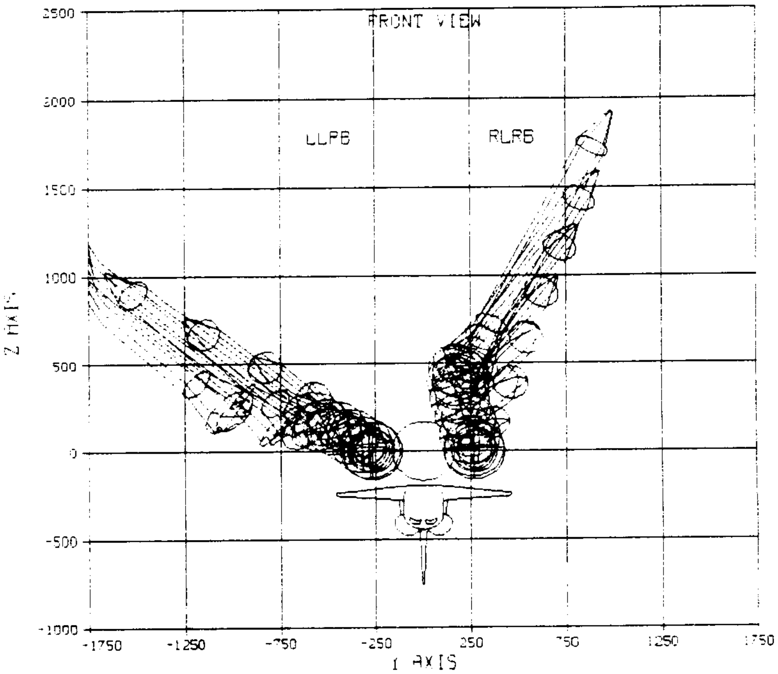


Figure 6.1.2-1. LOX/CH4 Pump-fed Nominal Ascent (Design Case) Separation, Front View

6.1.3 <u>THERMAL PROTECTION SYSTEM</u>. The thermal protection system for the LOX/Methane booster configuration is more extensive than that which is used on the RP-1 fueled boosters. Both the LOX oxidizer, and the Methane fuel tanks will require insulation. Otherwise, the TPS design for the methane pump-fed booster is similar to the LOX/RP-1 pump-fed LRB (see section 4.1.3).

6.1.3.1 <u>Aerodynamic Heating</u>. An examination of the LOX/Methane booster's 'Altitude vs. Velocity' ascent profile indicates that the booster will experience generally lower heating than a SRB because the trajectory is more lofted, and thus lower air density is encountered during high velocity portions of flight.

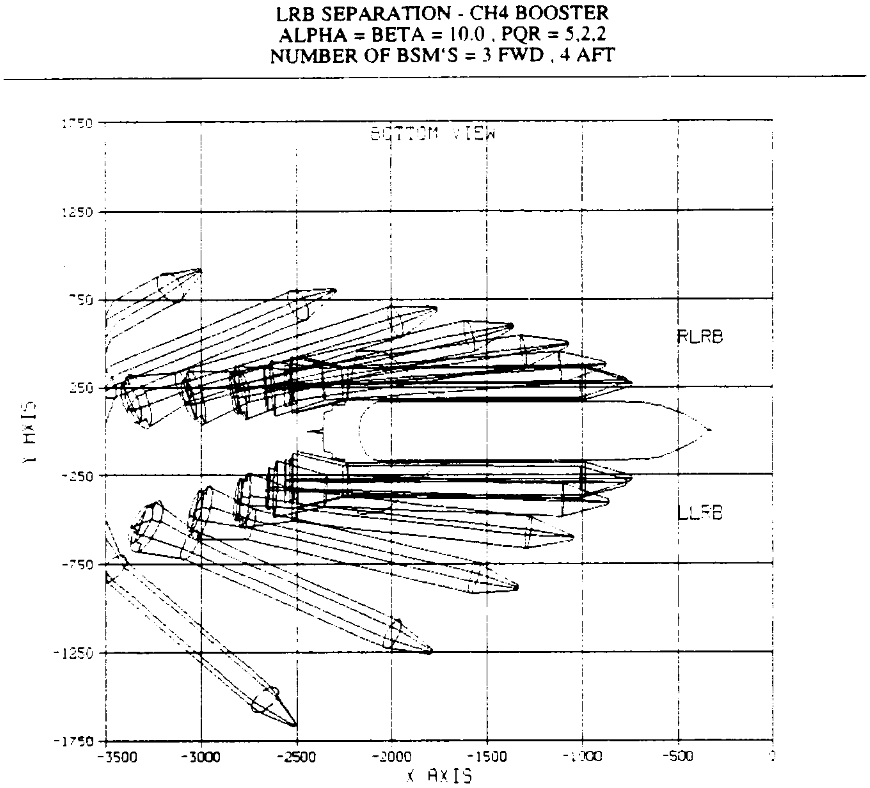


Figure 6.1.2-2. LOX/CH4 Pump-fed Nominal Ascent (Design Case) Separation, Bottom View

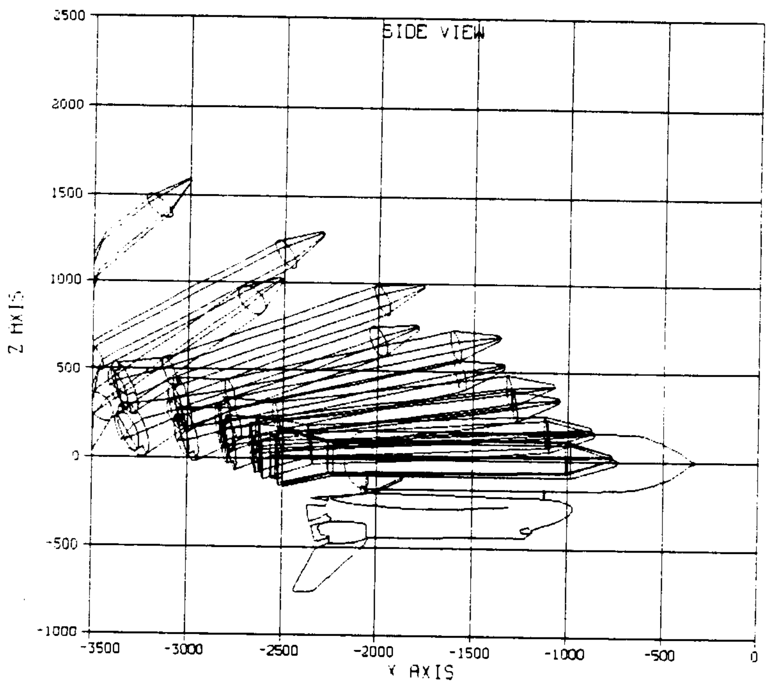


Figure 6.1.2-3. LOX/CH4 Pump-fed Nominal Ascent (Design Case) Separation, Side View

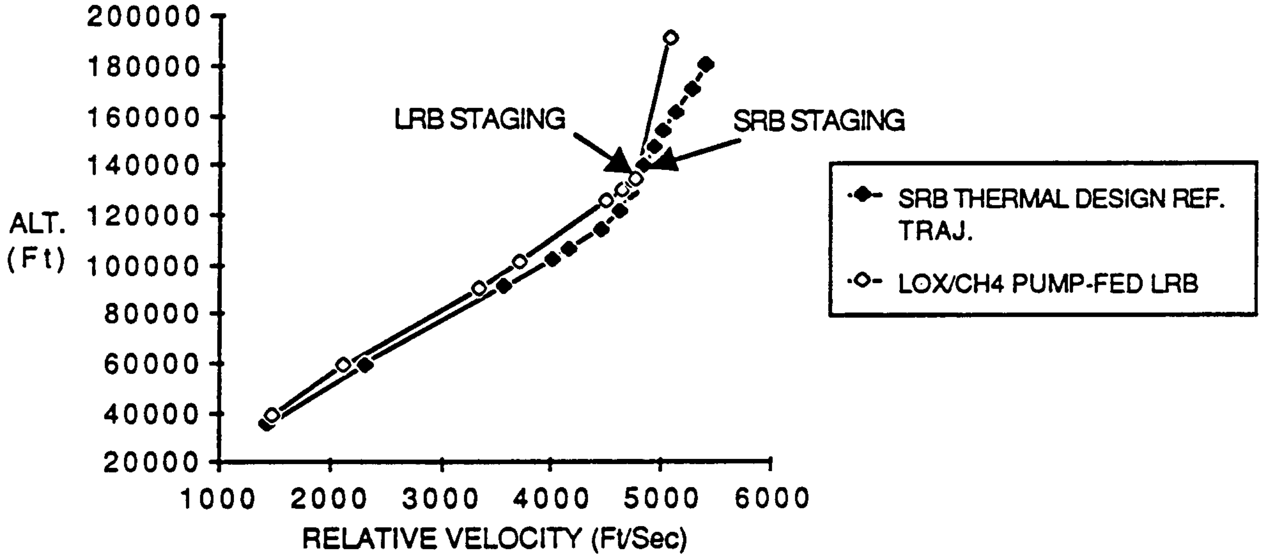


Figure 6.1.3.1-1 LOX/LH2 Pump-fed booster LRB And SRB Altitude Vs. Velocity Profile Comparison.

6.1.3.2 Thermal Protection System Design. The LOX/Methane pump-fed booster's TPS system is similar to the LOX/RP-1 pump-fed booster's (see section 4.1.3.2). SOFI (CRS-488) is applied in nominally 1" thickness to barrel sections of the cryogenic oxidizer and fuel tanks to prevent ice buildup and to minimize boiloff losses. MSA-1 will cover portions of the nose cone and aft skirt. SLA-561 will be applied to other high heating areas of booster such as interface attachment structure, feedline brackets, and other protuberancies. On propellant tank bulkheads, urethane foam will be applied after manufacture. Flexible skirts, and a heat shield will be used to protect booster engines and aft skirt components. Refer to Figure 6.1.3.2-1 below.

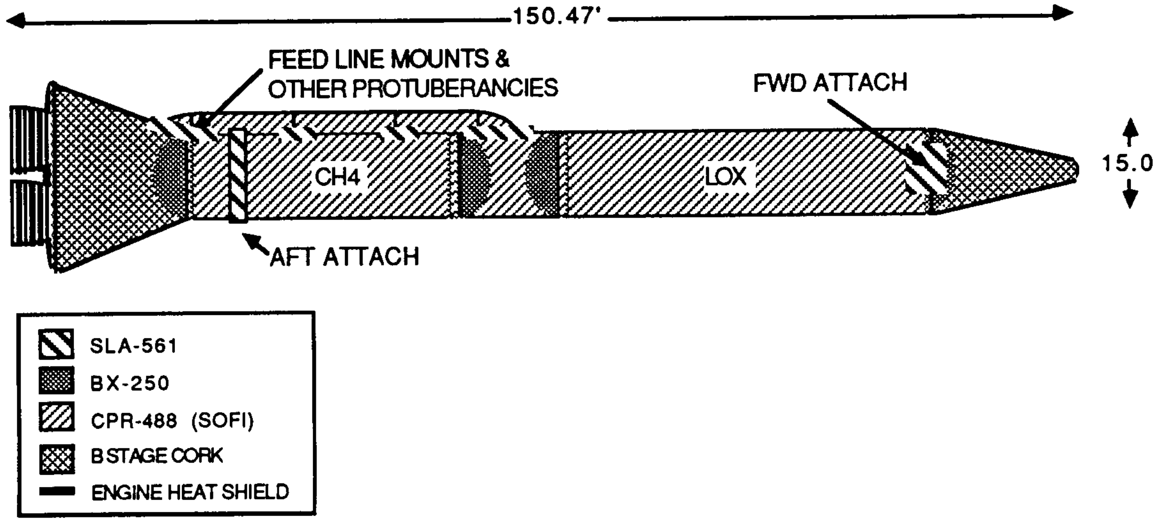


Figure 6.1.3.2-1. LOX/Methane Pump-fed Booster TPS Layout.

6.2 MAIN PROPULSION

6.2.1 ENGINE SYSTEM. The engines used are of expendable type with continuous variable thrust capability from 65% to 100% of the normal power level. The engine thrust was based on the vehicle sizing runs, and used the parametric data provided by Pratt & Whitney. A sensitivity analysis was made to determine the impact of mixture ratio and expansion ratio, and the results are shown in Figure 6.2.1-1 and 6.2.1-2.

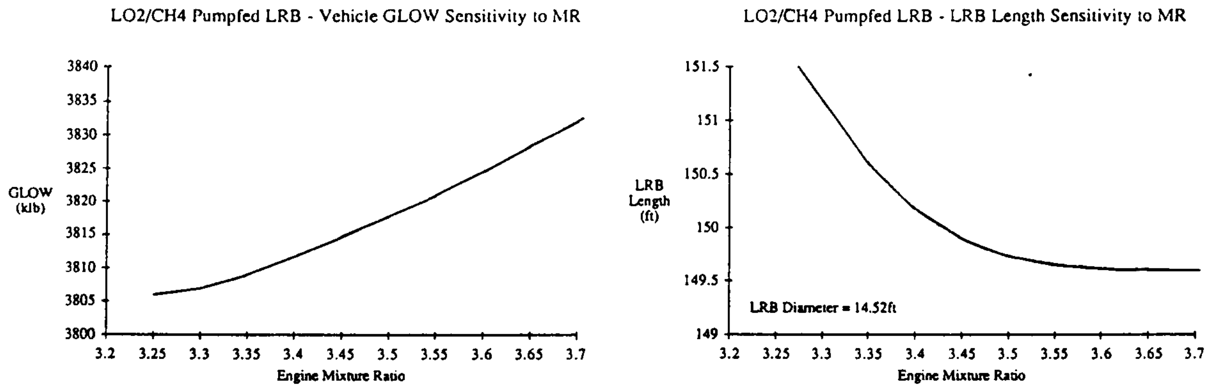


Figure 6.2.1-1 Mixture Ratio Sensitivity

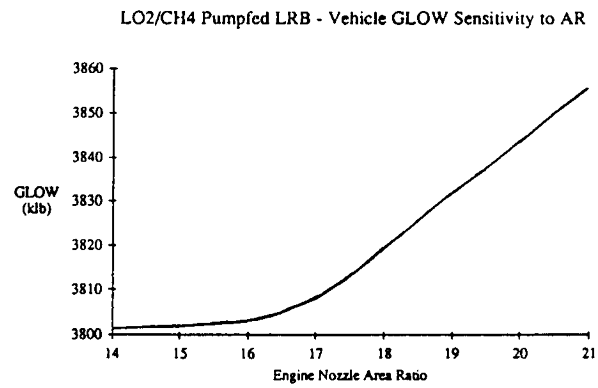


Figure 6.2.1-2 Expansion Ratio Sensitivity

As shown in Figure 6.2.1-1, the gross lift-off weight (GLOW) is minimum at approximately the same mixture ratio (3.3) at which the mean Isp is maximum, while the size of the LRB is minimum at mixture ratio of about 3.7, i.e., GLOW is function of Isp alone while size is in addition a weak function of density. For our preliminary design, a mixture ratio of 3.5 is used as it gives size and GLOW which lie close to the minimum.

The impact of area ratio on the vehicle weight and size is very small in the range of area ratio considered as shown in Figure 6.2.1-2. A area ratio of 16.5 was chosen based on a one dimensional equilibrium (ODE) run that gave nozzle exit pressure approximately the same as the average ambient pressure during flight (and hence maximum mean Isp). This vehicle concept was dropped before cost optimization step. Cost optimization on other vehicles indicates that area ratio, and hence exit area, should be significantly lower than for performance optimized vehicle.

6.2.1.1 Engine Feature Selection. Main features of the LO2/CH4 engine are shown in Table 6.2 1, and are discussed below.

Table 6.2.1 Main Features of LO2/CH4 Pump-Fed Engine

Cycle	Split Expander cycle
Boost Pumps	None
Throttling Capability	Continuous; 65% to 100%
Control system	Closed loop
Turbine Start	Boot-strap
Inlet Ducts	Scissors
Ignition	Spark Ignition
Nozzle	80% Bell
Thrust Chamber Material	Haynes 230
Gimbal	Head end gimbal; ±6° square pattern
Delivered life	5 starts
Burn time	150 sec
Engine inlet requirements:	LO2 60 psia
	CH4 40 psia

The baseline engine assumes no boost pumps with engine inlet pressure of 60 psia for LO2 and 40 psia for CH4. Pratt & Whitney has recently generated data on the impact of engine inlet pressures on the weight and cost of the engine, under STME/STBE contract (December 1988), which shows that higher inlet pressures should result in engine/vehicle cost savings.

A closed loop control system is selected because of the need of minimum three, possibly more, thrust levels. This requirement is imposed on the engine because of engine out capability for the booster. The cost saving between fixed point throttling for more than two thrust levels compared to continuous throttling is small. A single channel (SC) controller concept is selected over a dual channel (DC) controller concept as the reliability gain from SC to DC is only 0.99984 to 0.99990 while the increase in cost is $300k per engine. The actual control of the engine is described in the engine control.

Haynes 230 is selected as the baseline material for the thrust chamber over 347 stainless steel and the shrouded fuel turbopump impellers design is used over the current RL-10 design as these two modifications provide about 10% higher Pc without increasing the risk.

6.2.1.2 Selected Engine and Pump Characteristics. A cycle heat/power balance was done on the point design arrived at using the sizing program. The main engine characteristics are given in Table 6.2.2 below.

Table 6.2.2 Split Expander Engine Characteristics

Parameter	RPL
Thrust vac (k lbs)	756.3
Thrust SL (k lbs)	624.3
Chamber pressure	758.2
C* Efficiency	0.99
Nozzle Expansion Ratio	16.46
Engine Mixture Ratio	3.5
Isp vac (sec)	337.5
Isp SL (sec)	277.9
Oxidizer flow TC (lb/sec)	1754.0
Fuel flow TC (lb/sec)	499.6
Coolant fuel flow (lb/sec)	218.9
Engine Length (inch)	165.4
Nozzle Exit Diameter (inch)	106.9
Dry Weight (lbs)	5640

Flow characteristics at the nozzle exit plane are given in the Appendix 6 of Volume II of the final report. The pump and turbine characteristics at nominal rated power level are shown in Table 6.2.3.

6.2.1.3 Engine Schematic and Operation. Figure 6.2.1-3 shows the engine flow schematic with propellant flow rates and conditions at various parts of the cycle at 100% thrust level. The LO2 pump and the CH4 pump are driven by separate turbines. The LO2 pump consists of a single stage centrifugal pump while the fuel pump is a three stage centrifugal

Table 6.2.3 LO2/CH4 Turbo-pump Characteristics

	Component	LO2	CH4
Turbine			
	Stages	1	1
	Efficiency	0.739	0.743
	Horsepower	8096	16322
	Tips speed (ft/sec)	394	590
	Shaft speed (rpm)	7386	11071
	Inlet temperature (°R)	760	870
	Outlet temperature (°R)	705	760
	Inlet pressure (psia)	1536	3286
	Outlet pressure (psia)	1013	1536
Pumps			
	Stages	1	3
	Efficiency	.856	.836/.722/.722
	Inlet pressure (psia)	60	40
	Outlet pressure	1140	1134/2806/4480
Impeller			
	Diameter	13.48	13.79/15.62/15.59
	Tip speed	435	667/755/754
	Specific speed	2431	1505/727/731

pump. The schematic shown here is for a fixed thrust engine. The engine control will be discussed in a separate section.

Here MOV is the main oxidizer valve, JBV the jacket bypass valve, TBV the turbine bypass valve, FCV the fuel cooldown valve, OCV the oxidizer cooldown valve, and FSOV the fuel shutoff valve. During acceptance testing, the JBV is set to provide proper fuel jacket bypass flow split, the MOV set to provide the proper mixture ratio, and the TBV set to provide proper thrust setting. The OCV when closed meters the starting oxidizer flow to the igniter and chamber.

Chilldown. The current configuration requires no bleeds and cooldown is achieved by having the prevalves in open position. During the chilldown period, FSOV, MOV, JBV, and TBV are in their normally closed position, and FCV and OCV are in their normally open position. Cooldown is accomplished by opening the prevalves at the engine inlets.

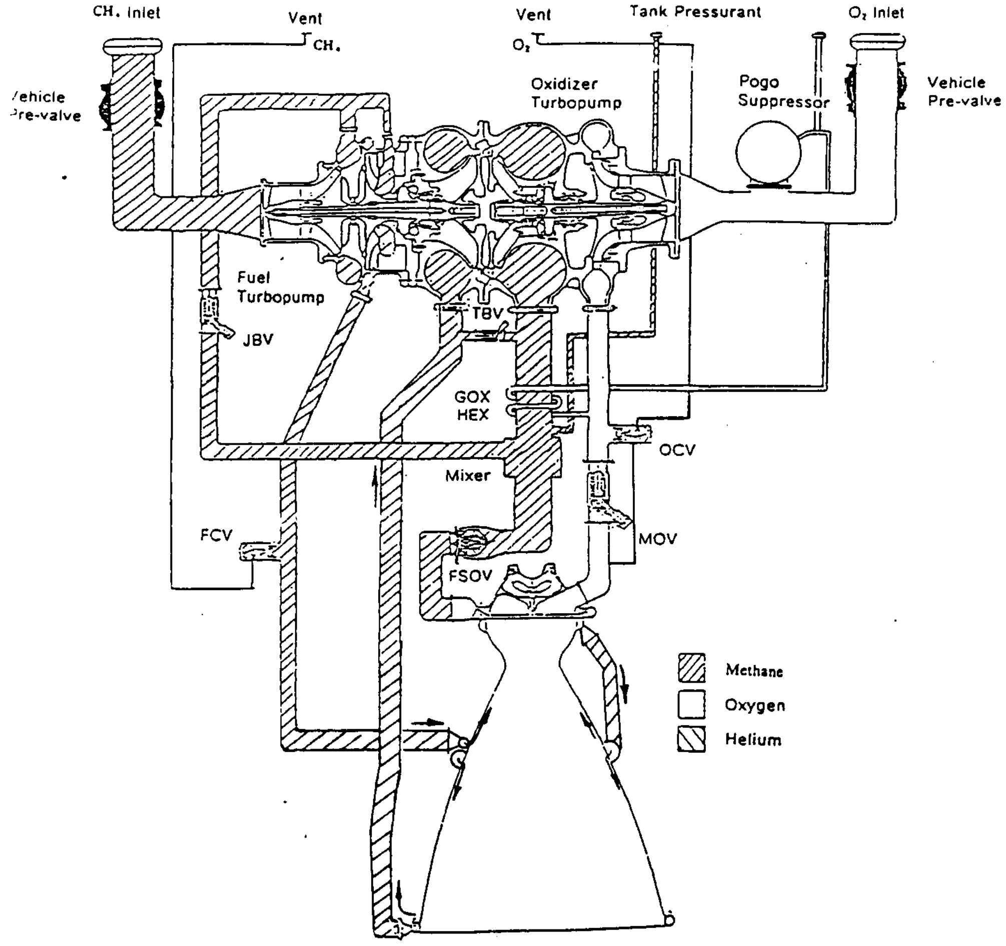


Figure 6.2.1-3a Flow Schematic of LO2/CH4 Split Expander Engine

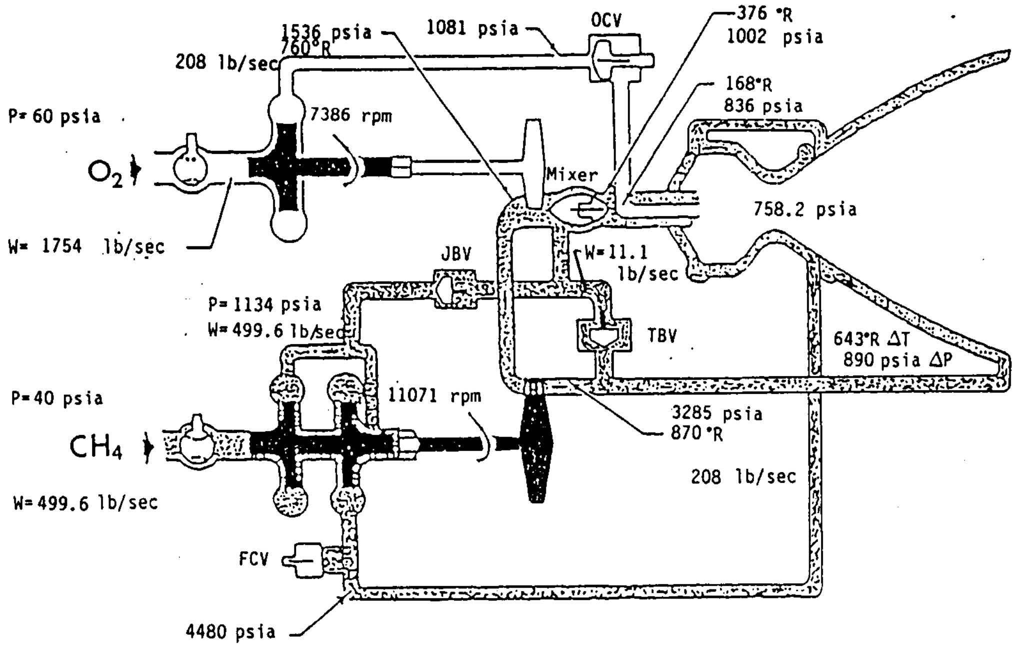


Figure 6.2.1-3b L02/CH4 Split Expander Flow Conditions at NPL

Start. Engine start is accomplished by shutting the FCV and the OCV, and opening the FSOV. Shutting the OCV provides small LO2 flow to the injector. The JBV is still in its normally closed position. CH4 flows through the combustion chamber coolant passages, in the process changing into gaseous CH4 and initiating turbopump rotation. The OCV actuation is set faster than the FCOV so that there is initially oxidizer rich atmosphere in the igniter and the thrust chamber. The ignition occurs when the mixture composition reaches the flammable conditions. Combustion in the chamber causes increase in the wall temperature. And the turbopumps speed increase, propellants flow rates and the system pressures increase.

When turbopump speed reaches approximately 50% of the steady state level, the JBV is opened allowing bypass around the jacket and the turbines, and providing some turbine backpressure to slow the acceleration. At 60 to 80% of the steady state turbopump speed, the MOV is opened, providing high oxidizer flowrates which results in rapid increase in the chamber pressure and mixture ratio. When engine reaches approximately 85% of the rated thrust level, the TBV is opened bypassing fuel flow around the turbines and preventing a thrust overshoot.

A two to three second acceleration from start signal to 95% thrust is expected with a 0.3 second variation.

Steady State. In the steady-state, engine thrust and mixture ratio are controlled by a closed loop control system. The valves are actuated by the electromechanical actuators. This will be further discussed in the engine control and propellant management sections.

Shutdown. Shutdown is accomplished by closing the FSOV, and this results in a rapid thrust decay (less than 0.15 sec to 1% thrust). The shutdown time can be extended by controlling the rate of valve closure. The FCV and OCV are opened to vent the high pressure propellants. Shutting of the FSOV causes a rapid deceleration of the engine, and opening of the FCV and OCV prevents any system overpressure from the sudden flow stoppage.

Abort. If the safety monitoring system indicates a problem, the engine can be shutdown in less than 0.15 seconds. Safe shutdown is possible if ground monitoring indicates a problem during the start sequence such as slow speed buildup, etc.

6.2.1.4 Engine Control. The engines of the LRB can be throttled over a range of 65% to 100% of rated power level. All engines receive the same throttle command at the same time. These come automatically from the general purpose computers through the engine controllers. Only manual control of the engine provided is the engine shut-off command for contingency situations. Throttling capability is needed to reduce the vehicle loads during maximum dynamic pressure region and to keep the vehicle acceleration below 3 g.

Engine thrust level is controlled by utilizing the TBV to maintain chamber pressure and therefore thrust. To throttle the engine down to a lower thrust level, the TBV will open up and reduce the amount of turbine flow and available horsepower. The pumps spin down and pressures decrease throughout the engine system until desired chamber pressure (thrust level) is attained.

The variable thrust control system concept used here is shown in Figure 6.2.1-4. It includes a controller with closed loop thrust and mixture ratio control and the necessary sensors and actuators to effect closed loop control. Dual sensors, dual actuator interface coils, and dual power supplies are used for higher reliability. All prestart activities and engine "ON" activities are scheduled by the controller. The controller system baselined for this concept is a single channel (SC) controller concept. In the event of channel becomes inoperative, failsafe shutdown is effected.

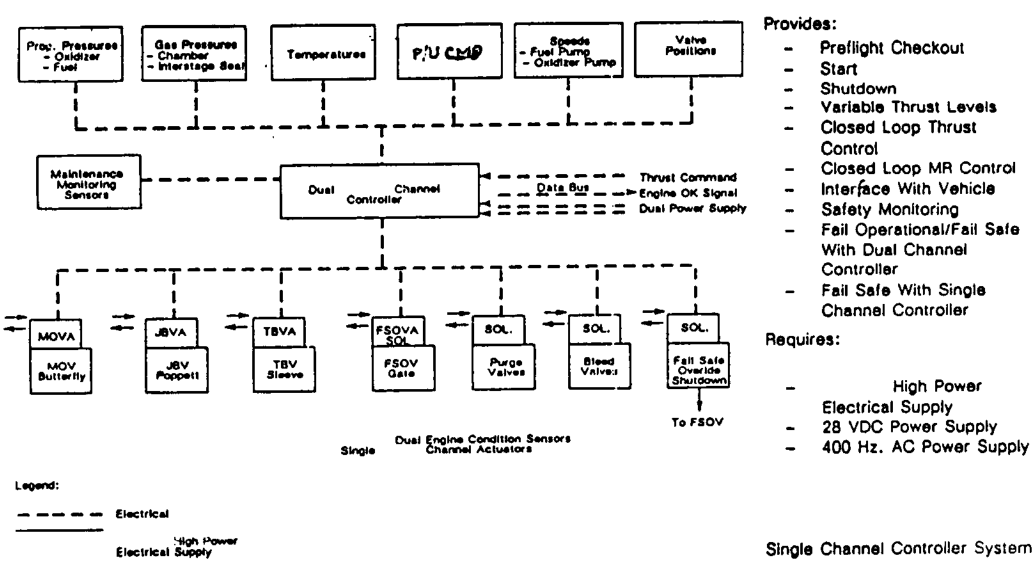


Figure 6.2.1-4 Variable Thrust Control System

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6.2.1.5 Engine Design Discussion and Engine Layout

Turbopump. The turbopumps of the Split expander Engine use relatively low cost materials and low cost manufacturing techniques to provide a low cost, reliable engine. The low temperature of the turbines allow the use of forged one piece aluminium disk and blades known as a blisk. Fuel pump impellers are machined from aluminum although studies are planned to produce cast aluminum impellers to further reduce cost. Pump housing are made from cast aluminum. The oxidizer pump impeller is made of forged 347 stainless steel, and the integral turbine is made of forged aluminum. The pumps will be mounted back-to-back counter rotating turbopumps contained (mounted) to a common turbine inlet and exit housing.

Injector. The main injector is configured of a multiple tangential entry oxidizer elements with a concentric annulus of CH4. The injector face plate is a porous material that allows transpiration cooling of the face. This design provides a hollow cone spray of liquid oxygen and is then exposed to high velocity fuel for better atomization.

Thrust Chambers and Nozzle. The thrust chamber is fabricated from Haynes 230 tubes brazed together. A Haynes 230 jacket will be used to provide structural support to the tubes in combustion chamber area. This support is brazed simultaneously with the tube. The engine utilize a dual circuit cooling scheme. Both cooling circuits are single pass with the thrust chamber employing counterflow and the nozzle employing parallel flow. The third stage CH4 pump discharge flow enters the thrust chamber at its base which is downstream of the throat. After cooling the chamber the exiting coolant is routed to the top skirt manifold and passes to the end of the nozzle, collected in a manifold and directed to the pump turbine inlet. Figure 6.2.1-5 shows typical features of this thrust chamber and nozzle.

Mixer. Figure 6.2.1-6 depicts a mixer concept that will be utilized for mixing hot gas and cold liquid methane in the Split Expander cycle engine. This mixer concept provides efficient, turbulent mixing between the hot and cold fuel flows with a simple, compact configuration and an acceptable pressure drop. The concept has previously been used by P&W on the XLR-129 test stand to mix hot and cold hydrogen, and it is similar to a mixer used on the SSME.

Ignition System. An augumented spark igniter (torch) type is baselined as it can be easily maintained and can be checked out prior to flight.

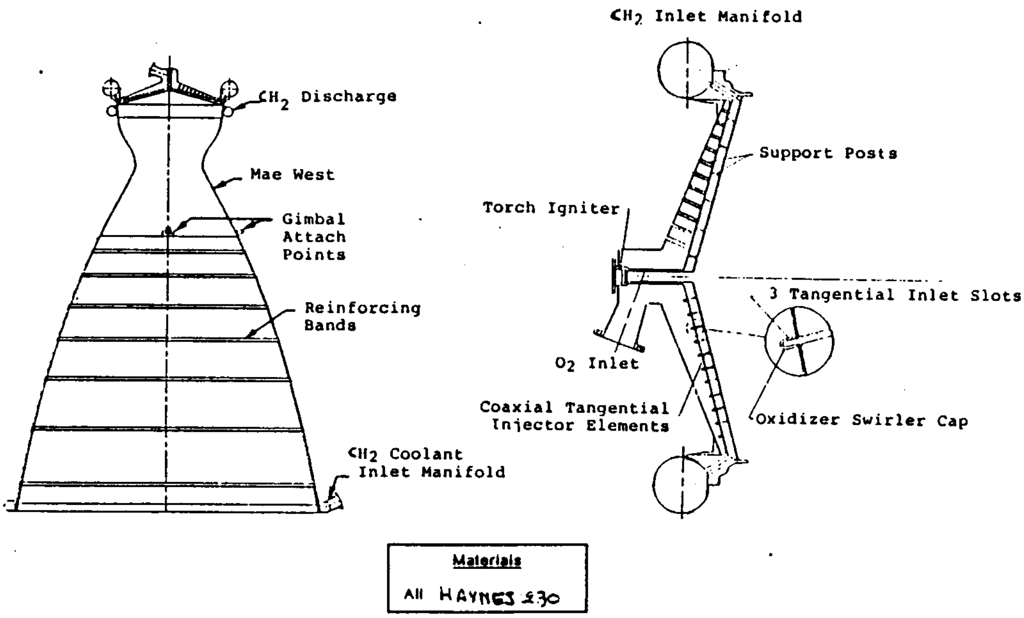

Figure 6.2.1-5 Thrust Chamber and Nozzle Construction

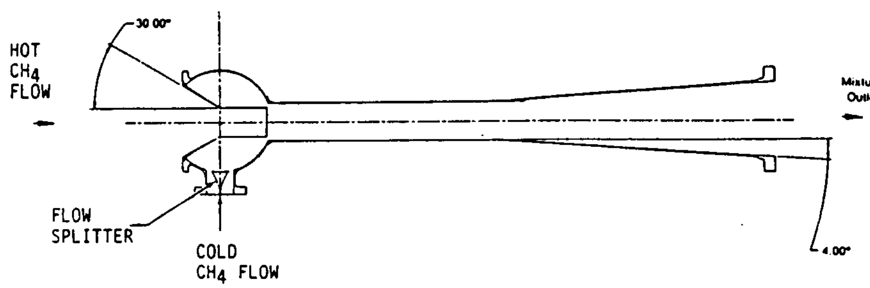

Figure 6.2.1-6 Schematic of Split Expander Mixer

<u>Pressurization System</u>. The engine is designed to provide gaseous methane at maximum flowrate of 3 lbs/sec at about 1100 psia and 705°R, and gaseous oxygen at a maximum flowrate of 3.5 lbs/sec at approximately 1100 psia and 400°R. Gaseous methane is bled off the engine between the turbine discharge and the mixer. The gaseous oxygen is produced in a GO2 heat exchanger which utilizes the hot gaseous methane to vaporize the oxidizer.

The GO2 heat exchanger, Figure 6.2.1-7, consists of an aluminium duct wall that has trip-strips on the turbine exhaust side wall for improved convective heat transfer film coefficients. The oxygen passages are constructed of offset fins that are bonded to the high strength outer wall and the inner aluminum plate. The offset fins enhance the oxygen side convection heat transfer film coefficients which will reduce the size of the heat exchanger. The aluminum plate is separated from the duct wall by a highly conductive layer of either dead soft copper or copper powder in colloidal suspension. The copper layer has been incorporated into the design to stop crack propagation from the inner plate to the duct wall.

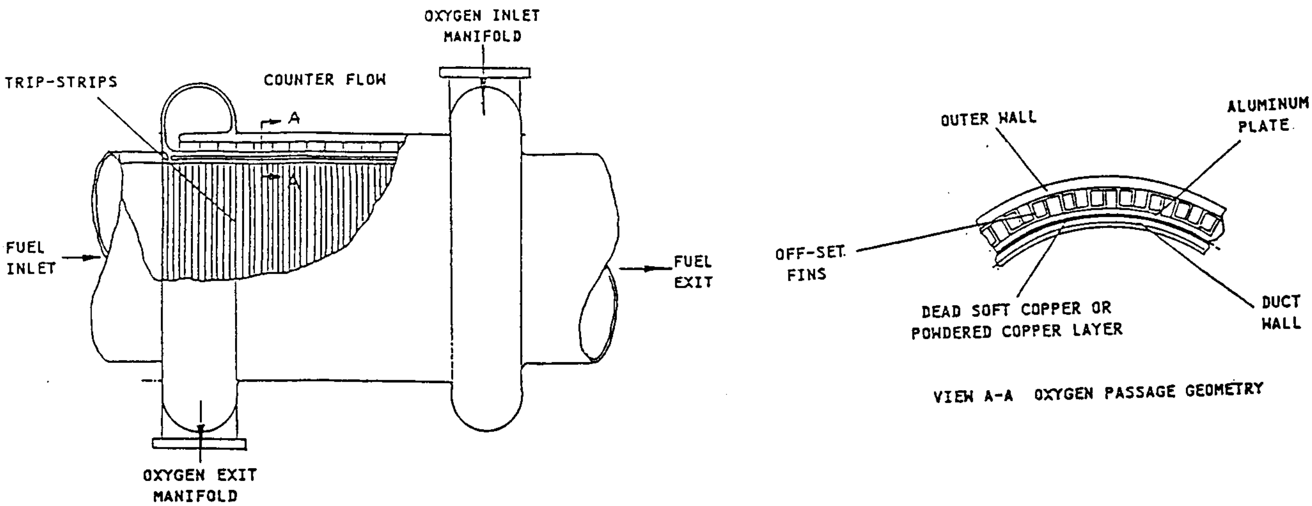

Figure 6.2.1-7 Heat Exchanger for GO2 Pressurization System

<u>Pogo System</u>. The POGO system baselined is similar the one used currently in the SSME POGO system which uses a gas filled plenum to isolate engine feedline oscillations from the engine. The pressurized GO2 is supplied by the GO2 heat exchanger, and is used to energize the POGO suppressor.

<u>Split Expander Cycle Power Margin</u>. All engines independent of cycle face the challenge of reaching rated thrust during their development program. In the development phase, the components rarely meet all of their performance goals in the first engine build. Some modifications and/or minor redesigns are normally needed to achieve rated engine operational capability by the end of the development program. While gas generator and stage combustion chamber cycle engines are plagued with having turbine temperature too high to meet rated thrust, the expander cycle engine could possibly have too low a turbine temperature.

The attainment of rated thrust in expander cycle engines which depend upon the regenerative heat in the nozzle for turbine power, is impacted by both the heat picked up as well as the pressure loss in the nozzle tubes and manifolds. The design point for the expander cycle is currently set with a turbine by-pass margin of about 10% (excess available horsepower). This excess power capability can be expressed in terms of excess chamber pressure or thrust, and is approximately equal to 75 psi margin. This margin is deemed sufficient to meet extreme design uncertainties.

Preliminary Engine Drawing and Layout. A preliminary drawing of the LO2/CH4 engine is shown in Figures 6.2.1-8. The engine arrangement takes into account the engine gimbal capability. A quick check was made to see that the plume of one engine does not impinge on the other engine even with failure of one engine gimbal system, and with the expected gimbal angles. It is assumed that the engine with the faulty gimbal system can be brought to neutral position by the back-up system. These checks were made near BECO where maximum plume expansion would take place, and near max-Q region where maximum gimbal angle can occur. (NOTE: Recent STBE studies made after completion of this task indicate that two separate turbopump mounted usually on the engine may be a better configuration.)

6.2.1.6 Flight Engine Instrumentation. The following flight engine instrumentation for engine health monitoring and for engine control has been identified:

- Chamber pressure
- Fuel pump inlet pressure
- Fuel pump inlet temperature
- Fuel pump housing temperature
- Fuel pump vibration
- Fuel pump speed
- Fuel turbine inlet pressure
- Fuel turbine inlet temperature

Similar instrumentation on the oxidizer side is needed

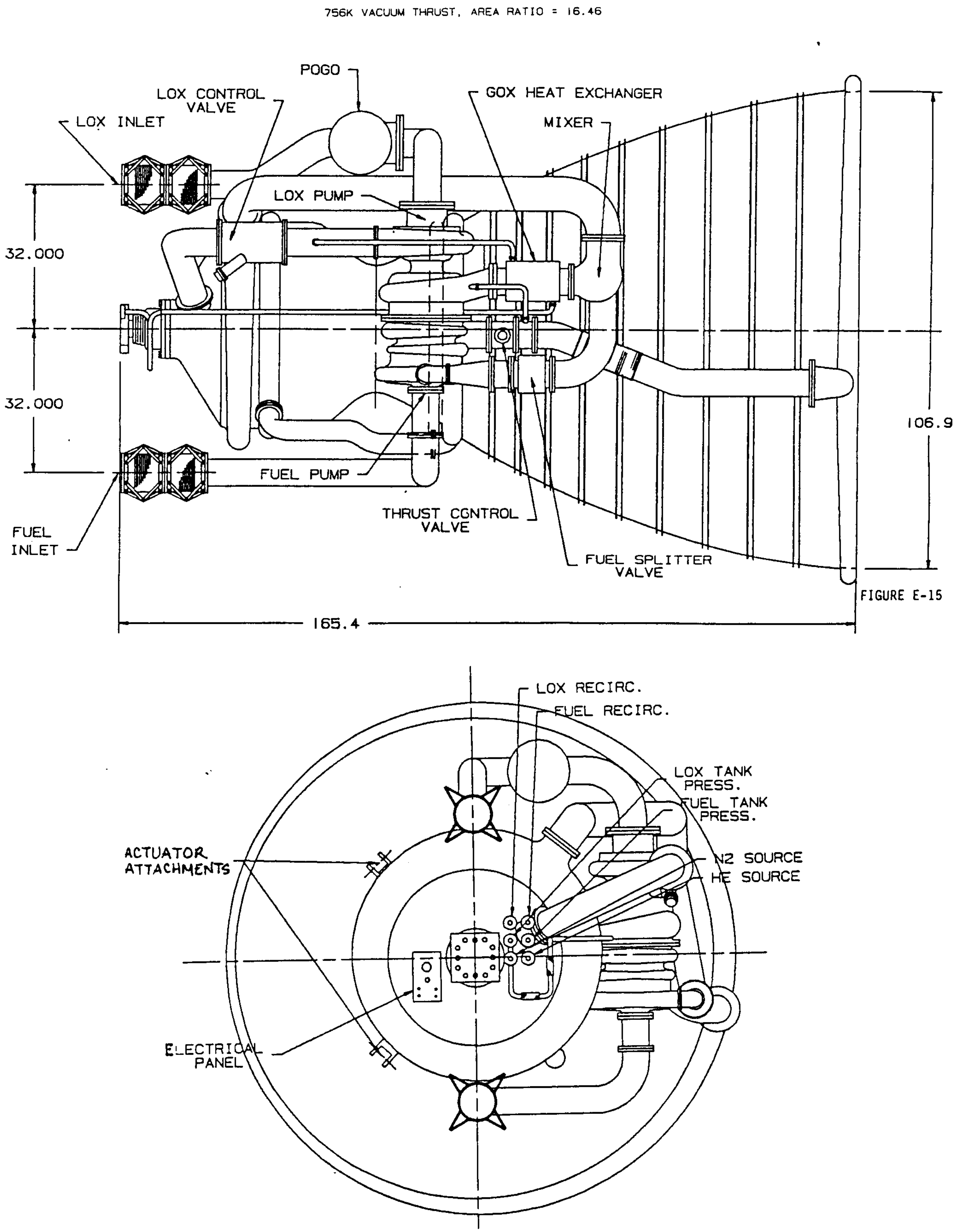

Figure 6.2.1-8 Preliminary Drawing of LO2/CH4 Split Expander Engine

6.2.1.7 Engine Interface Requirements

LO2 pump NPSP:	44 psi
CH4 pump NPSP:	24 psi
LO2 inlet temperature	164°R
CH4 inlet temperature	202°R
LO2 inlet diameter:	12 in.
CH4 inlet diameter:	9 in.
GO2 pressurization line:	3 in.
CH4 pressurization line:	3in.
Straight line requirement:	None
Purge gas:	N2 (flowrate TBD)
Electrical power:	TBD

6.2.2 VEHICLE SYSTEM. Since both CH4 and LH2 are cryogenic fuels, the LO2/CH4 vehicle propulsion systems are very similar to the LO2/LH2 systems, which are described in Section 5.2.2.

6.2.2.1 Feed and Fill/Drain Systems. The basic features of the LO2 and CH4 feedlines are depicted in Figure 6.2.2-1. The baseline for the LO2 system is a single external feedline. This arrangement was selected because of minimum complexity and higher reliability (compare to dual feedlines), as recommended by the manufacturing personnel. The flow velocity used in the line sizing is based on the current operating systems and the guidelines suggested by rule-of-thumb equations. From these considerations, the line diameter of LO2 is 24-in and of CH4 is 18-in with line velocities of approximately 31ft/sec and 43ft/sec, respectively. The lines are designed to withstand pressures of 300 psid for LO2 and 180 psid for CH4. This takes into consideration the water-hammer effects during opening and shut-off operations. The outlets for both the LO2 and CH4 tanks are located at the bottom of the tank and along the tank axis. They are contoured to minimize drop-out. At each outlet, a vortex baffle is mounted to reduce the propellant swirl. On the bottom of this baffle is attached a fine mesh screen.

The LO2 feedline runs external to the tank from the booster intertank assembly. Both the LO2 and CH4 lines are insulated mainly to prevent ice formation. In addition, this reduces the heat leaks, boil-off, and provides good quality propellants to the engines. Figure

6.2.2-1 show the schematic of the feedlines. The LO2 feedline has six flexible joints in the mainline with specifications for the flex joints and flanged joints similar to that for the ET

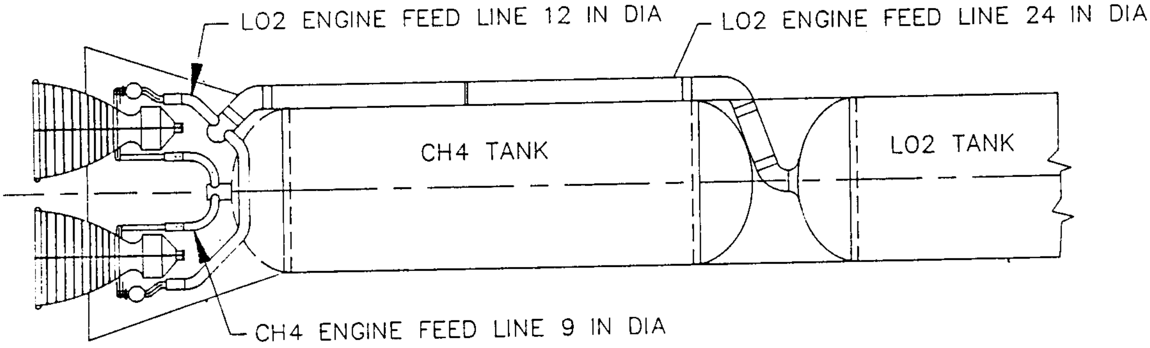

Figure 6.2.2-1 LO2 and CH4 Feedlines Lines

LO2 feedline. It ends in a manifold with four 12-in feedlines which connect to the four engine inlet. Each of the smaller feedlines has a open/shut prevalve. The CH4 feed system has 2 flex joints in the mainline, and has four 9-in equal length feedlines with a prevalve in each of the smaller lines. There are 2 flex joints in each of the lines connected to the engine disconnect. It should be noted that there is no straight line requirement for LO2/CH4 engines since the pumps are located far downstream of the engine inlet.

Similar to LO2/RP1 system, geysering is not expected in the case of LO2/CH4 LRB.

Fill/drain lines are 6-in dia and 4-in dia with maximum flowrates of 450 lbs/sec and 100 lbs/sec for LO2 and CH4 systems, respectively. A schematic of the feed and fill/drain systems are illustrated in Figure 6.2.2-2. Fill and drain operations are more extensively discussed in the ground operations section.

6.2.2.2 Pressurization System. Both the LO2 and CH4 tank pressurization systems are autogeneous in flight. These tanks are pressurized with GSE supplied ambient temperature He before the engine start. The helium system also acts as a back-up system after the engine start while the shuttle is on the ground. In autogenuous systems, CH4 gas is bled off the engine at 1100 psia and 705 R, while GO2 is obtained via a heat exchanger at 1100 psia and 400°R.

Oxygen System. Tank pressure profiles for the LO2 tank during flight are shown in Figure 6.2.2-3. The engine inlet pressure requirement for the LO2 pump is 60 psia (44 psi

NPSP). The minimum required ullage pressure and the corresponding tank bottom pressure are calculated based on feedline system losses and for a representative trajectory. It is assumed that the minimum ullage pressure in the tank is always 2 psi greater than the ambient pressure, so that a positive pressure difference is assured at all times. Highest tank ullage pressure is required prior to lift-off, which include pre-start and during start. Because at this time G-level is the lowest, therefore hydraulic head exerted by the LO2 in the feedline and tank is minimum. During this period, the minimum pressure at the bottom of the tank is 60 psia and the minimum ullage pressure needed is 30 psia. It should be noted that just at lift-off, the ullage pressure required decreases from 60 psia to 16.7 psia because of increase in head due to lift-off acceleration of 1.6G. In practice, the ullage pressure cannot be dropped instantly; therefore the tank bottom should be able to withstand 77 psia (30 psia ullage + 47 psia tank head). Our LO2 tank is designed to withstand 70 psig (84.7 psia on the ground) based on manufacturing considerations. Hence there is no extra strength requirement imposed by the engine NPSH at any time, either in flight or on ground.

The final ullage pressure in the tank is approximated on the basis of the unusable residuals (pressurant and liquid) left in the system. As long as there is propellant left in the tank and in the horizontal portion of the LO2 feedline to meet the NPSH requirement, a substantial reduction in residual propellants can be obtained by raising the tank pressure. Pressure required to meet NPSH requirement with no propellant in the tank is about 12 psia which is approximately equal to 12 psig at this time of flight. The net decrease in residual propellants becomes very small once the liquid enters the vertical portion of the line because of the drop in the head exerted by the LO2 in the feedline and increase in pressurant requirement. For simplicity of the pressurization control system, a constant 17 psig upper setting of the control band is selected. It is assumed that the NPSH requirement is waived during engine shutdown because of lower flowrates during this period. With this assumption the ECO sensors in the LO2 system should be located at about 30ft above the engine inlet for best utilization of the tank ullage pressure.

The maximum flow rates required to achieve a constant 17 psid pressurization profile in the LO2 tank is about 23 lbs/sec. This flowrate is required during the initial period of the flight when there is actually almost no ullage pressure requirement to satisfy the NPSP. Average pressurant requirement is only 12.6 lb/sec, half the amount. Hence to minimize heat exchanger and line sizes, a maximum flow rate of 14 lb/sec (about 10% margin over the minimum) is baselined.

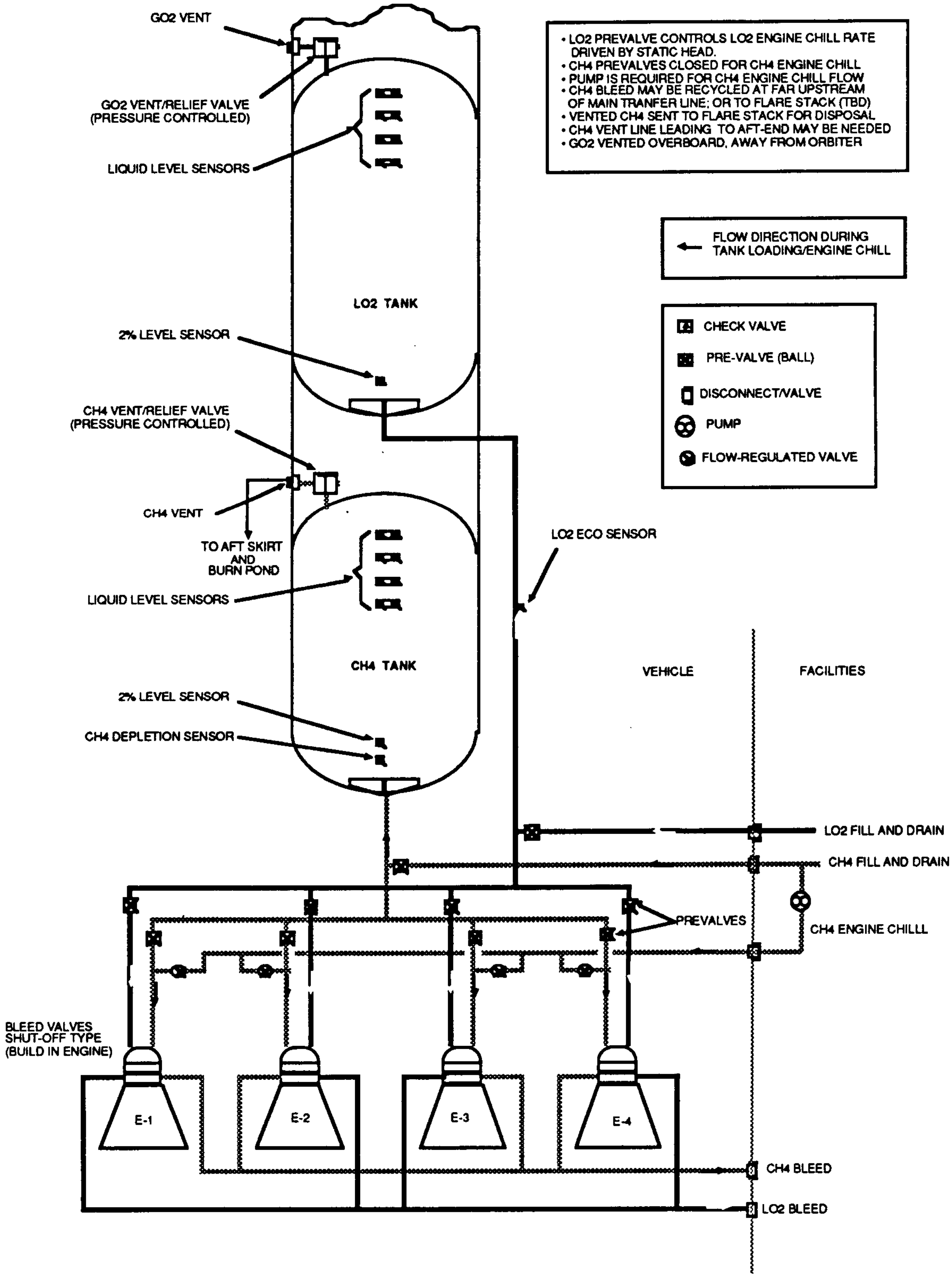

Figure 6.2.2-2 Schematic of LO2/CH4 Feed and Fill/Drain Systems

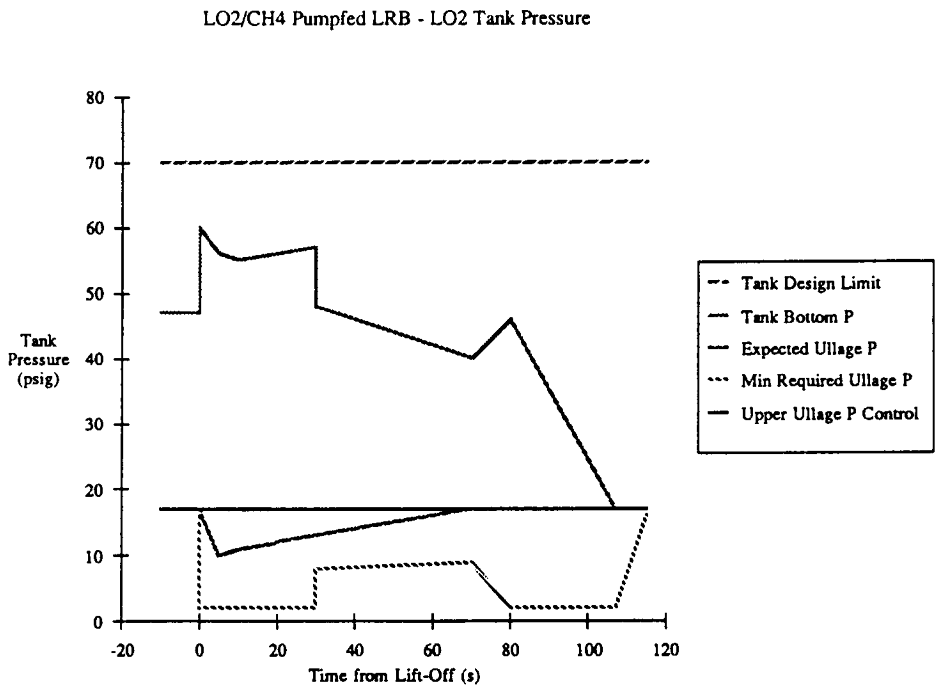

Figure 6.2.2-3 LO2 Tank Pressure Profiles

The main pressurization line size is assumed 3-in dia based on Mach number and pressure losses considerations. This line is designed to withstand a pressure of 600 psig at ambient temperature. Gases enter the tank through a multi-hole diffuser, and construction of the ET diffuser is baselined for this system. A 2-in orifice is used at the entrance to the diffuser. This helps in minimizing diffuser vibrations and also reduce pressure losses.

The pressurant mass flow control required is not very critical because of large safety margin, and is accomplished through opening and closing of the valves located in the pressurization line. The actual control logic is discussed in greater detail in the propellant management system of Section 4.2.2, LO2/RP1 system. Schematic of the pressurization and purge system, and valve operation is similar to that for the LO2/LH2 system, Section 5.2.2.

CH4 System. The NPSH requirement for the CH4 engine is 40 psia. As shown in Figure 6.2.2-4 , maximum ullage pressure is required at BECO and at start to satisfy the NPSH requirements, and is 50 psia. For simplicity of tank pressure control, upper setting of the control band is taken as 50 psia. CH4 tank is again designed to withstand 70 psid (84.7

psia on ground), and hence no extra strength requirement is imposed by the current baselined NPSH requirement.

The average flow rate required to meet the NPSH requirements at BECO is 9.9 lb/sec. The pressurization system is designed with a margin of 10% to provide an average flow rate of 11 lb/sec. The expected pressure signature in the CH4 tank, the tank bottom pressure and the pressure margin for the tank are shown in Figure 6.2.2-4. This flow rate gives a line diameter of 3 in. The lines are designed to withstand a pressure of 600 psid at maximum temperature of 800 R. The diffuser design is similar to one described for the oxygen system. A 2-in orifice is located upstream of the diffuser.

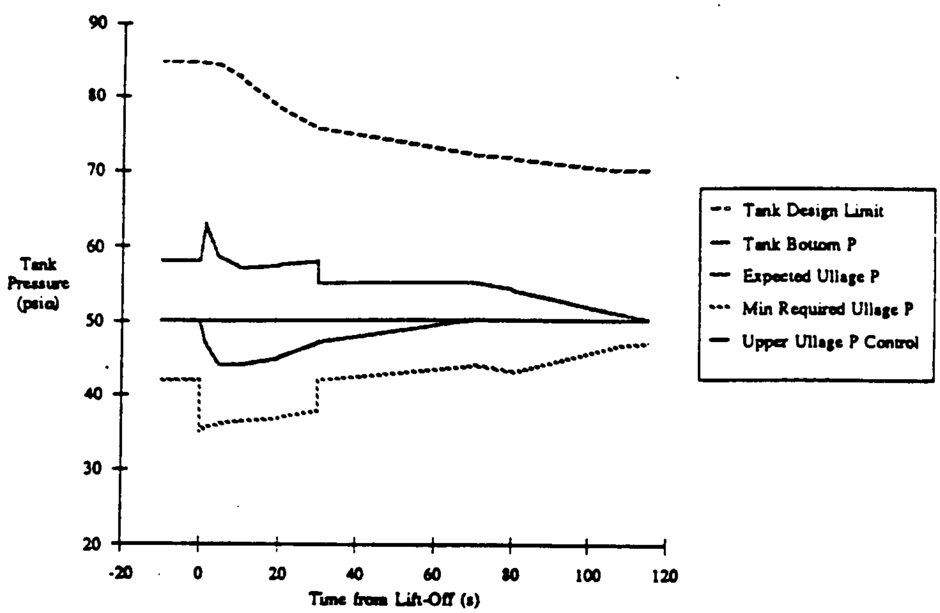

Figure 6.2.2-4 CH4 Tank Pressure Profiles

6.2.2.3 <u>Vent and Relief System</u>. Each booster tank is provided with a vent and relief system at the top of the tank. A valve similar to that used on ET is currently baselined. The valve is a dual function valve, which is in normally closed position.

On ground, it can be kept in the open position by applying ground supplied Helium (1/4-in line). This position is applicable while purging the tank, loading the tank, conditioning of the propellants, and in case of abnormal conditions. Vent valve for LO2 tank is 4-in in diameter, and for CH4 tank is 3-in. The sizing of the vent valve is usually based on the heat leak considerations and the propellant properties desired in the tank. Steady state boil-off rates are 0.75 lb/sec for LO2 tank and 0.25 lb/sec for CH4 tank. Here the rates are scaled from the ET tank. The vent valves will also open in the relief mode on the ground if the pressure in the LO2 tank increases greater than 22 psig, and more than 55 psig in the CH4 tank. GO2 is vented directly to the atmosphere while vented CH4 is sent to flare stack when the vehicle is on ground. Both lines are 5-in in diameter. The size of the lines are determined again on the basis of heat leak rates (which governs the boil-off rate) and the minimum tank pressure required for propellant conditioning.

Once the vehicle is in flight, the valves can operate only in the relief mode, and functions on the basis of pressure difference between the ambient sensing port and the tank pressure. The vent gases are directly vented to the atmosphere. The setting of the valve is the same as on the ground, that is, 22 psid for the LO2 tank and 55 psid for the CH4 tank.

6.2.2.4 Purge Systems. The basic functions of the purge systems are (1) to provide inert and moisture free atmosphere in the tank before loading, (2) to keep hazardous gas concentrations below the safe level, and (3) to provide thermal conditioning in order to prevent condensation of moisture and ice formation for ensuring proper functioning of the instrumentation and valve operation. The purge system for LO2/CH4 can be schematically represented by LO2/LH2 purge system in Section 4.2.2.

Tank purging will use dry ambient nitrogen with dew point of -40°F. The purge gases will flow from the fill/drain system and exit through the vent system. The water vapor concentration in the exit gases will be monitored to see that no water vapor/pool is left in the tank. The timelines and flowrates at present are TBD.

The engine compartment area below the booster tank, the intertank area and the nose cone area will be purged utilizing GSE supplied hot nitrogen. The temperature in these compartments will be maintained between 32 and 112°F. Sufficient nitrogen flow rates will be allowed in the intertank so as to keep GO2 concentration in the intertank area less than TBD% with 65 knots of wind. The CH4 concentration levels will be monitored in both the engine compartment and the intertank compartment using hazardous gas detection system, with 3% as the maximum allowable concentration level. The flow rates in these will be maintained through control orifices and at present are TBD.

A continuous helium anti-icing purge through the pressurization lines is provided throughout the loading period. This keeps positive pressure and flow in the lines, and also prevents ice formation on the pressurization lines. A flowrate of 0.01 lb/sec should be adequate.

6.2.2.5 Propellant Management Systems. The propellant management system is similar to that discussed in Section 4.2.2 of LO2/RP1 Pump-Fed Concept.

6.2.2.6 Overview of Tank Operations and Tank Summary. Here a summary of the booster tank operations is given based on various topics discussed in the previous sections and in other parts of the report.

The tank operation at the launch pad starts with purging of the tanks with nitrogen at -40°F dew point. This is to assure that there is no moisture or air in the tank. Our current thought is that helium purge is unnecessary, because the amount of nitrogen that would be dissolved is very small and this would not have any significant impact on the performance. After the tanks have been checked for moisture and air concentrations, the tank chilldown/fill process is started. This is controlled by the ground launch processing (GLP) system. The vent valves are cycled closed and open based on the tank pressure (between 2 and 8 psig) for quick chilldown. The chilldown of the tank is followed by slow fill to 2% sensor level, and fast fill from 2% to 98% sensor levels. During this period, the vent valve is kept open, again by application of high pressure helium. The tank pressure is monitored to see that it is above 2 psig so that no air can enter the system. Both booster tanks fill can be simultaneously done with ET loading and would require about 1 hour loading time from chilldown to fastfill, and presently is scheduled between T-TBD minutes to T-TBD minutes. Topping of the tanks to 100% starts after 98% fill, and is then followed by the replenish mode during which the propellant conditioning, with vent valve in open position, is accomplished. The time required for propellant conditioning is about 30 minutes. The steady state replenish rate/boil-off rate expected is 0.75 lb/sec for LO2 and 0.25 lb/sec for the CH4 while the transient rates can be about 10 times higher than these rates. The steady state pressures in the tanks are about 16 psia.

During the tank chilldown and fill operations, purges in the nose cone, intertank, and aft compartments purge are maintained. These purge flowrates are controlled so that temperatures in these compartments temperatures lie within the specifications (32°F to 112°F). Anti-icing purges carried through the pressurization line are also turned on to prevent any ice formation on the pressurization line and the vent lines. The relief operation of the valve is set at 22 psig for the O2 system and for 45 psig the CH4 system.

LO2 and CH4 vent valves are closed about 1.5 minutes prior to the launch, followed by tank pressurization using the ground He at room temperature. The tank pressure of LO2 tank is maintained between 15 and 17 psig, while the CH4 tank pressure is maintained between 48 and 50 psia. The time allocated for pre-pressurization period is about 60 secs, and the maximum He mass flowrate needed is 0.4 lb/sec for LO2 tank and 0.8 lb/sec for the CH4 tank. The initial rise period is between 10 to 15 secs; thereafter, the valves in the ground helium control system are cycled to maintain the pressure within the band. The pressure in the tank falls because of the heat losses to the tank and the liquid, and due to small increase in the ullage volume because of bleeds.

At about T-5 secs the engines are fired and the autogeneous gases from the the engines are supplied to the tanks. He system acts as a back-up to the autogeneous system while the vehicle is on ground. At T-0 He system is disconnected. Pressurant flowrate from the engines initially is not sufficient to maintain tank pressure within the band (it is still higher than 8 psig) even though both the control passages are open. However, because of vehicle acceleration, NPSP requirements are met at this time of flight. At about 60 secs, ullage pressure falls within the control band, and is actively controlled by the pressurant management system.

The engine shut-off is normally initiated by the ECO sensor located in the LO2 feedline. In case improper functioning of the P/U system or other abnormal situations, engine shut-off sequence can also be initiated by low level fuel cut-off.

In case of a problem on the launch pad, the tanks can be drained at any time by closing the vent/relief valve, pressurizing the tanks using pad helium, and opening the fill/drain lines. The tanks of the booster are also provided with two linear shaped charge assemblies (similar to that on ET), initiation of which is controlled by the Range Safety officer.

A summary of the design parameters for the feed and pressurization system is given in Table 6.2.2.3.

Table 6.2.2.3 Summary of Tank Design and Operating Parameters

SYSTEMS	LO2	CH4	REMARKS
Engine Inlet			
Minimum P (psia)	60	40	
T (°R)	164	202	
Feed System			
Main Line D (in)	24	18	
Manifold to Engine D (in)	12	9	
Max Propellant Flowrate (lb/s)	6972	1992	
Max line P @ engine inlet (psia)	300	180	
Fill/Drain			
Duct D (in)	6	4	
Tank Operating			
Ullage P, Pre-press & Flight	17 psig	50 psia	set constant from pre-start to BECO
Bulk T min-max (°R)	164-600	202-700	liquid to pressurant inlet T
Max Tank Bottom P (psig)	63	50	occurs @ lift-off
Pressurization System	Autogenous	Autogenous	
Medium	GO2	GCH4	
Heating Source	turbine disch	coolant disch	
Line Operating T (°R)	164-600	202-800	
Line Operating P (psia)	ambient-600	ambient-600	
Main Line D (in)	3	3	
Engine to Manifold Line D (in)	1.5	1.5	
Total Pressurant Wt (lb)	1400	1100	
GHe Pre-Press Line D (in)	1	1	
Vent System			
Valve D (in)	4	3	
Valve Operating P Range (psid)	0-22	0-55	
Purge System			
Engine Purge Supply Line D	TBD	TBD	GN2 grnd supplies @ TBD
Total Liquid Residuals (lb)	8740	1860	

6.2.3 THRUST VECTOR CONTROL (TVC). For system description and schematic see Section 4.2.3.

A summary of the TVC requirements for the LOX/CH4 pump-fed engine is shown in Table 6.2.3-1.

Table 6.2.3-1 LRB TVC Requirements for LOX/CH4 Pump Fed Engine

Input Parameters		**Source**
Thrust per Engine	624300 -lbs.**	Pratt & Whitney
Number of engines on LRB	4	LRB Baseline
Thrust Vector Offset	0.25 -inches	Estimate
Gimbal Block Coeficient of Friction	0.06	Estimate
Gimbal Block Pin Radius	5.50 -inches	Estimate
Maximum Gimbal Angle	6.00 -Degrees	Specification
Gimbal Rate Required	10.00 -Deg/Sec	Specification
Gimbal Acceleration Required	57.30 -Deg/Sec^2	Specification
Engine Weight	5640 -lbs.	Pratt & Whitney
Engine Inertia	2821 -lb-ft^2	Estimate
Distance from Engine C.G. to Gimbal	68.00 -inches	Pratt & Whitney
Distance from Vehicle C.G. to Gimbal	100.00 -ft	Estimated
Actuator Moment Arm	69.10 -inches	Pratt & Whitney
C.G. Offset from Centerline of Eng.	0.00 -inches	Assumed
Max Veh Longitudinal Acceleration	3.00 -g's	STS limit
Max Veh Lateral Acceleration	0.30 -g's	STS/LRB Traj Sim
Max Veh Angular Acceleration	3.00 -Deg/Sec^2	STS/LRB Traj Sim
LOX Line Torque	3409 -ft-lbs	Estimate
Fuel Line Torque	2246 -ft-lbs	Estimate
Total Flex Line Stiffness Torque	5655 -ft-lbs	Estimate

Torque Calculations		
Longitudinal Acceleration Torque	10022 ft-lbs	120264 in-lbs
T1 - Due to Engine C.G. Offset	0 ft-lbs	0 in-lbs
T2 - Due to Max Gimbal Angle Offset	10022 ft-lbs	120264 in-lbs
Lateral Acceleration Torque	14933 ft-lbs	179191 in-lbs
T3 - Due to Vehicle Lateral Acc	9588 ft-lbs	115056 in-lbs
T4 - Due to Engine Inertia	148 ft-lbs	1772 in-lbs
T5 - Due to Vehicle Angular Accel	5197 ft-lbs	62362 in-lbs
T6 Thrust Misalignment Torque	13006 ft-lbs	156075 in-lbs
T7 Engine Control Torque	2820 ft-lbs	33845 in-lbs
T8 Engine Block Friction Torque	17168 ft-lbs	206019 in-lbs
T9 Propellant Duct Torque (Given)	5655 ft-lbs	67858 in-lbs
Total Static Torque	43616 ft-lbs	523388 in-lbs
Total Dynamic Torque	19989 ft-lbs	239864 in-lbs
Total Required Torque	**63604 ft-lbs**	**763252 in-lbs**

Peak Power Requirements (Using Torque X Gimbal Rate)		
Peak Output power req'd per Actuator	20.2 -hp	15.1 -kW
Peak input Power/Act (sys eff = 53%)	38.1 -hp	28.4 -kW
Peak Power Required per Engine	76.2 -hp	56.8 -kW
Total Peak Required for LRB	**304.7 -hp**	**227.2 -kW**

Actuator Sizing	
Peak Operating Output Force	11046 -lbs
Stall Force	16568 -lbs

** Head-end gimbal point

6.3 AVIONICS (Refer to Section 4.3)

Avionics systems architecture same as previously discussed for Pump-fed LOX/RP-1 concept.

6.4 METHANE PUMP-FED PERFORMANCE AND TRAJECTORIES

6.4.1 NOMINAL MISSION. A description of the nominal mission trajectory simulation can be found in section 8.1.3. Since the ATO mission determined the required size of the methane pump-fed LRB configuration, the nominal trajectory simulations were run simply to determine nominal performance.

The following table is a summary of the nominal performance for the methane pump-fed configuration.

```
Lift off conditions:

   Weight (lb)                              =   3,956,786.7000
   Payload (lb)                             =   70,500.000000
   Thrust (lb)                              =   5,383,129.5811
   Thrust to weight                         =   1.3604801040
   Initial inertial velocity (ft/sec)       =   1,342.4324022
   Launch site latitude                     =   28.307566153
   Launch site longitude                    =  -80.540959056


Max Q conditions:

   Max dynamic pressure (lb/ft**2)          =   722.75329283
      Time (sec)                            =   61.227165633
      Angle of attack (deg)                 =   4.1507949324
      Altitude (ft)                         =   28,539.958166
      Mach number                           =   1.2383677019
      Q * ALPHA (deg-lb/ft**2)              =   3,000.0007053


LRB separation:

   Staging time (sec)                       =   132.82638138
   Altitude (ft)                            =   147,099.49469
   Dynamic pressure (lb/ft**2)              =   56.263146768
   Angle of attack (deg)                    =  -1.9999997282
   Mach number                              =   5.0238707087
   Inertial velocity (ft/sec)               =   6,614.6843952
   Inertial flight path angle (deg)         =   20.017833593
   Relative velocity (ft/sec)               =   5,365.7702407
   Relative flight path angle (deg)         =   24.960152951
   Delta V (ft/sec)                         =   9,370.8802619
   Weight after separation (lb)             =   1,514,815.0588
   Remaining ET propellent (lb)             =   1,158,530.0709
   SSME throttle at separation              =   1.0400000000
   LRB throttle at separation               =  0.84567539975
   Thrust (lb)                              =   1,464,837.3684
   Thrust-to-weight after separation        =  0.96700739792
   Acceleration after separation            =  0.95185487795
   LRB propellent used (lb)                 =   1,766,268.2121
   Geodetic latitude (deg)                  =   28.496366105
   Longitude (deg)                          =  -79.972443885
   Average back pressure (psi)              =   5.6342937496
```

MECO conditions:

Time (sec)	= 493.50543034
Altitude (ft)	= 360,604.81892
Inertial velocity (ft/sec)	= 25,871.153798
Inertial flight path angle (deg)	= 0.76623644106
Delta V (ft/sec)	= 30,066.920377
Shuttle & payload perigee (nm)	= 35.203141715
Shuttle & payload apogee (nm)	= 159.91748145
MECO weight (lb)	= 365,640.30044
SSME throttle @ MECO	= 0.77857927055
SSME propellent weight used (lb)	= 1,578,594.6874
ET remaining propellent weight (lb)	= 9,355.3125605
Average back pressure (psi)	= 1.5170928227

Throttle schedules:

Max q throttle down time (sec)	= 54.327043957
LRB throttle setting	= 0.85654527459
Post max q throttle up time (sec)	= 68.341548762
Start 1620 kips attach load throttling	= 115.28336050
Start LRB 3g throttling (sec)	= 126.03603999
LRB throttle setting	= 0.84567539975
SSME throttle @ separation	= 1.0400000000
Start SSME only 3g throttling (sec)	= 449.82830042
SSME final throttle setting	= 0.77857927055

Losses to LRB separation

Total delta V	= 9,370.8802619
Steering losses	= 1,884.4304462
Drag losses	= 444.51096914
Gravity losses	= 1,464.4936988
Pressure losses	= 300.02121498

Losses to MECO

Total delta V	= 30,066.920377
Steering losses	= 2,485.8575325
Drag losses	= 452.14700743
Gravity losses	= 2,296.3027085
Pressure losses	= 300.13409962

Min/Max conditions:

Max (+) angle of attack (deg)	= 9.2483645913
Time (sec)	= 4.9698673203
Max (-) angle of attack (deg)	= -8.6043158784
Time (sec)	= 17.801979117
Max (+) Q * Alpha (lbf-deg/ft**2)	= 3,000.0013780
Time (sec)	= 54.327165633

```
Max (-) Q * Alpha (lbf-deg/ft**2)           =  -972.84129109
   Time (sec)                               =   90.132130307
Max acceleration (g's)                      =   3.0001859694
   Time (sec)                               =   126.03603999
```

Figures 6.4.1-1 thru 6.4.1-9 show various performance parameters obtained from the methane pump-fed LRB configuration's nominal trajectory simulation.

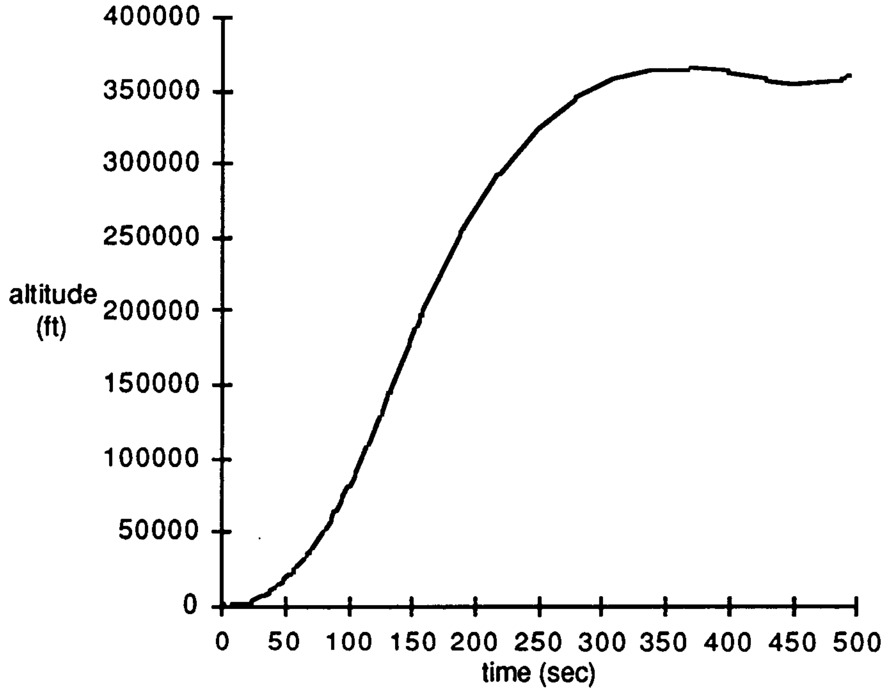

Figure 6.4.1-1 Altitude vs Time

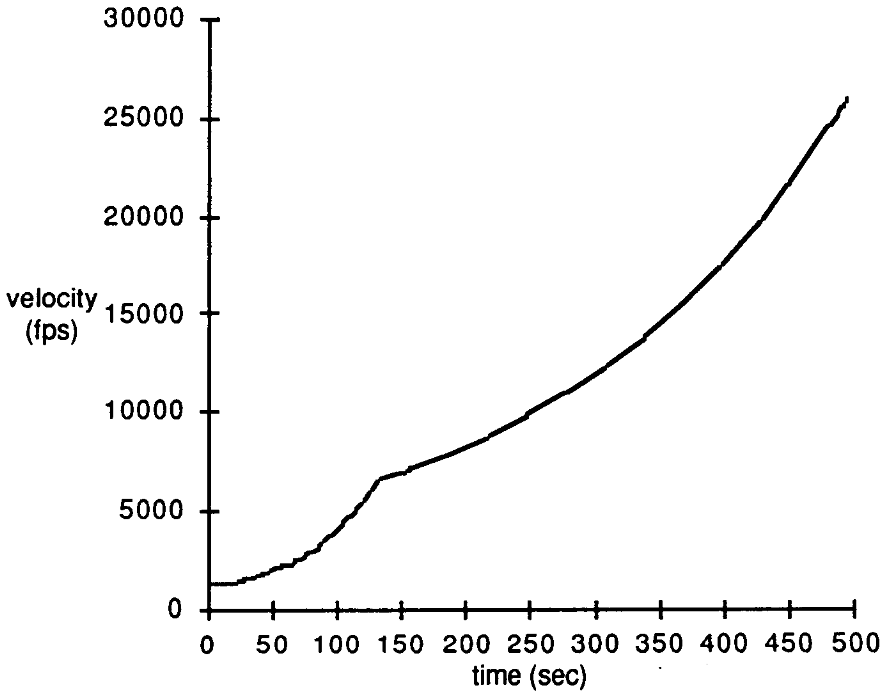

Figure 6.4.1-2 Velocity vs Time

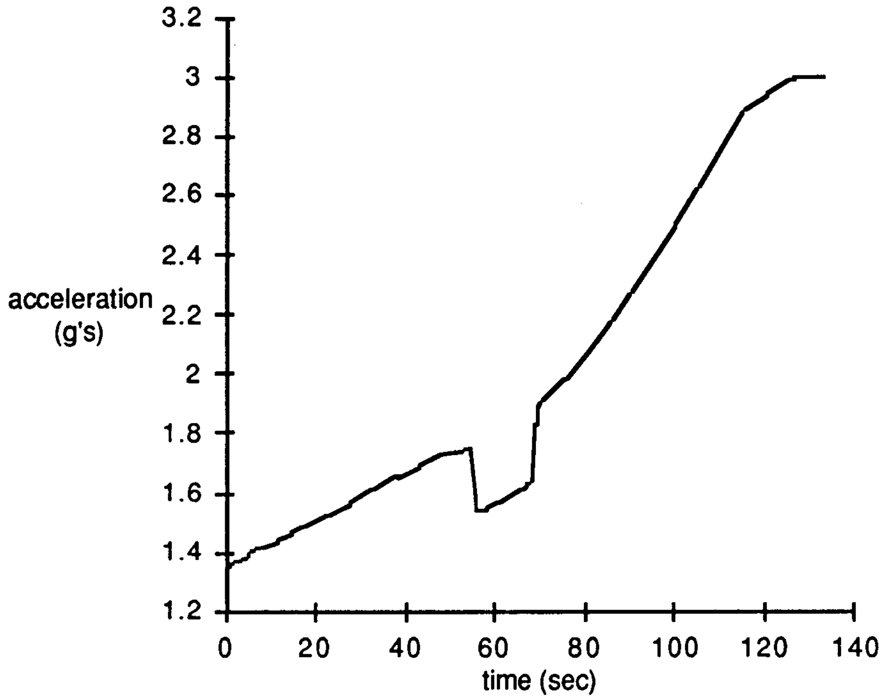

Figure 6.4.1-3 First Stage Acceleration vs Time

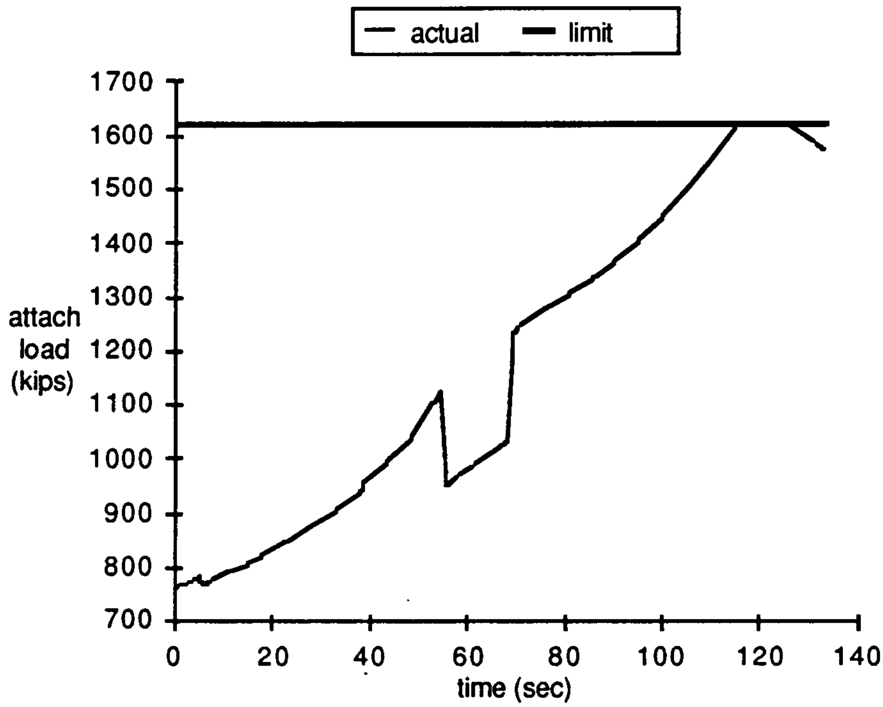

Figure 6.4.1-4 ET Attach load vs Time

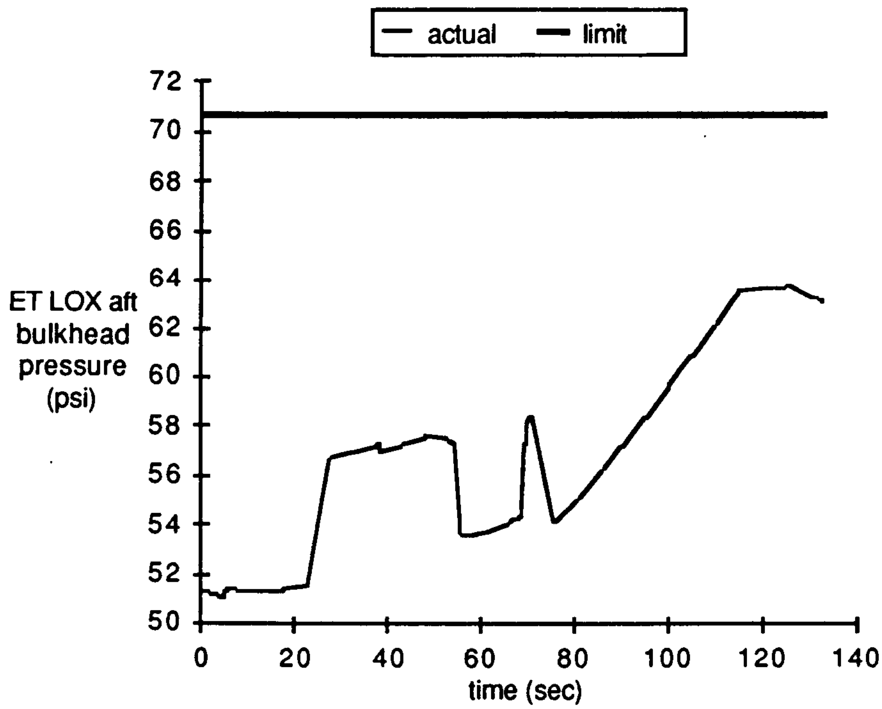

Figure 6.4.1-5 ET LOX aft bulkhead Pressure vs Time

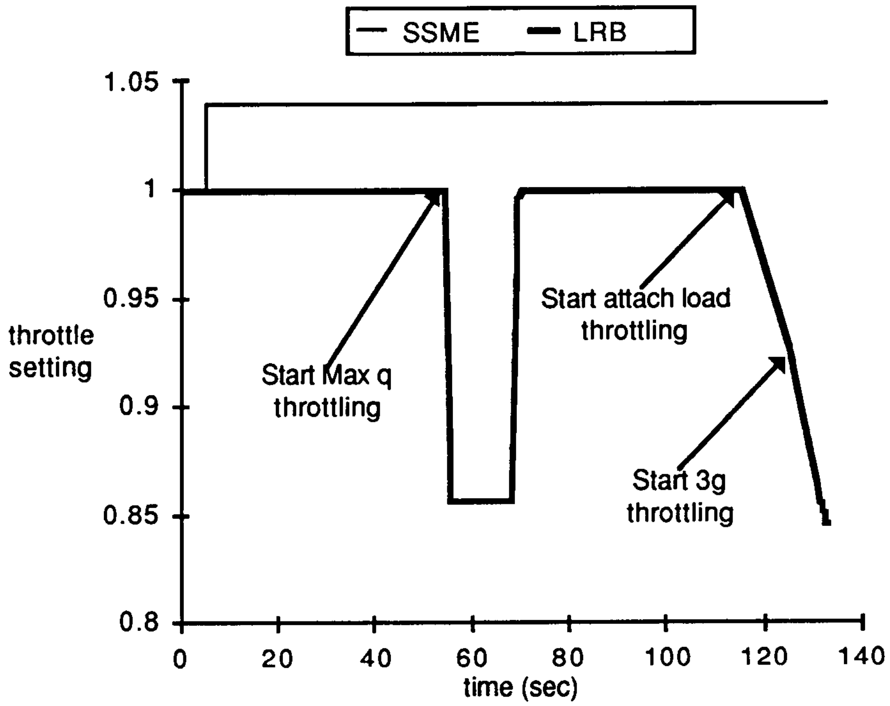

Figure 6.4.1-6 Throttle setting vs Time

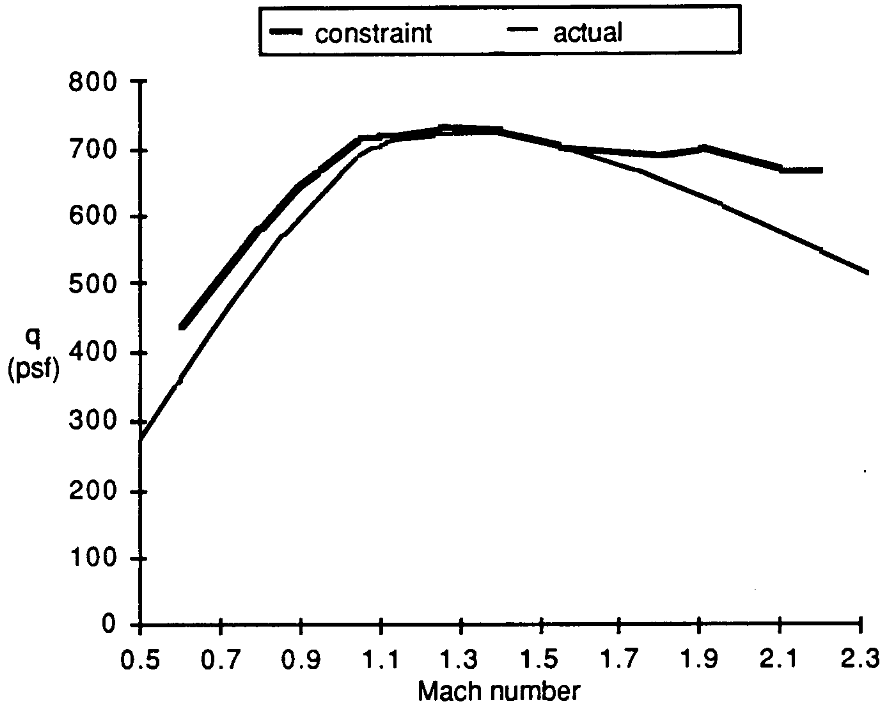

Figure 6.4.1-7 Dynamic pressure vs Mach number

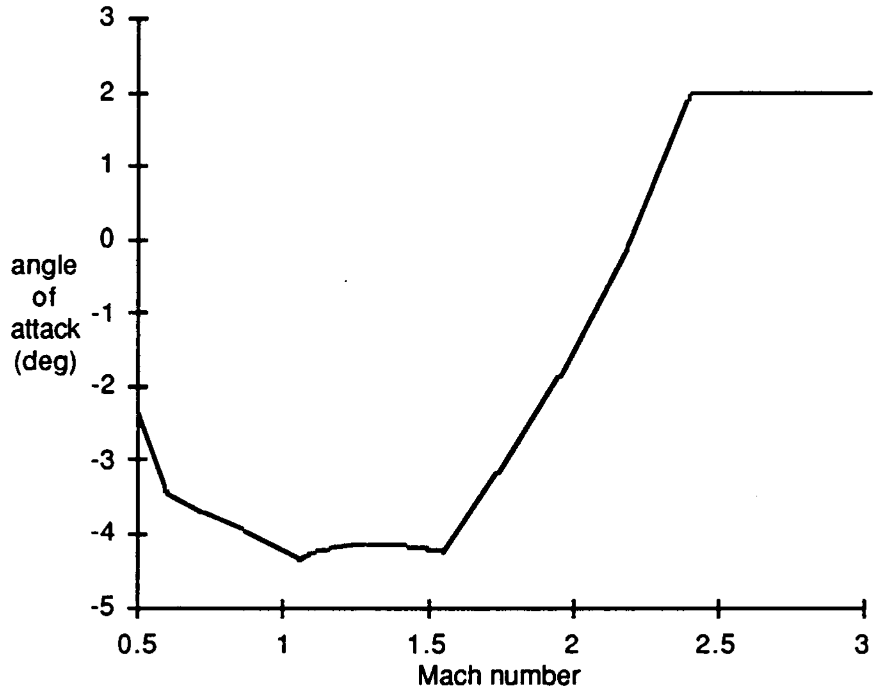

Figure 6.4.1-8 First Stage Angle of Attack vs Mach number

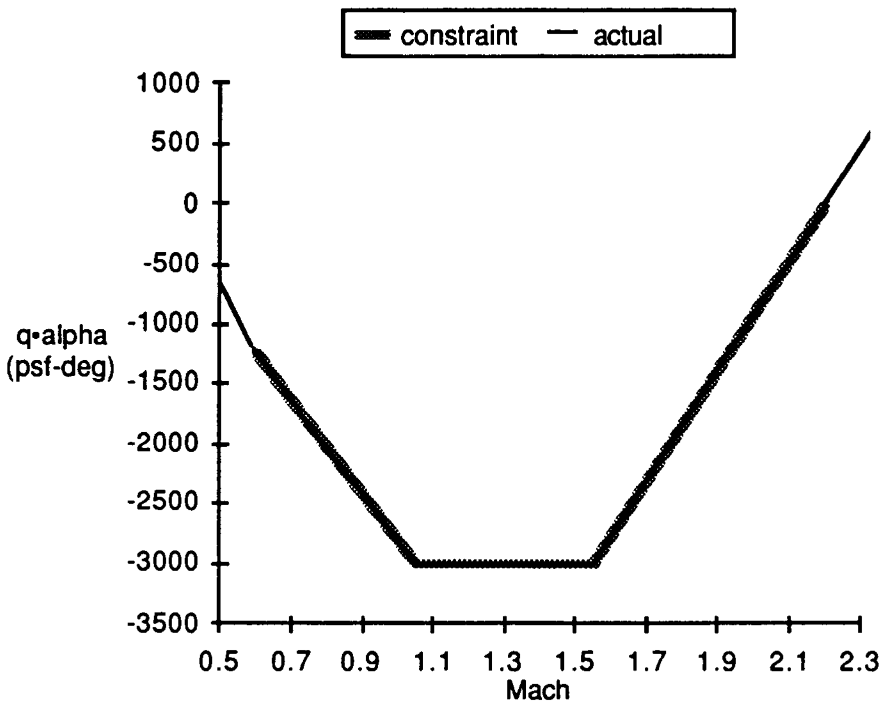

Figure 6.4.1-9 First Stage qa vs Mach

6.4.2 ATO MISSION. A description of the ATO mission trajectory simulation can be found in section 8.1.3. The ATO mission determined the required size of the methane pump-fed LRB configuration. The methane pump-fed LRB configuration's propellent, thrust, and structure were adjusted by the FASTPASS program until the desired performance was obtained.

The following table is a summary of the ATO performance for the methane pump-fed configuration.

Lift off conditions:

Weight (lb)	=	3,956,786.6343
Payload (lb)	=	70,500.000000
Thrust (lb)	=	4,851,138.8061
Thrust to weight	=	1.2260299214
Initial inertial velocity (ft/sec)	=	1,342.4324022
Launch site latitude	=	28.307566153
Launch site longitude	=	-80.540959056

Max Q conditions:

Max dynamic pressure (lb/ft**2)	=	568.37749520
Time (sec)	=	80.652908866
Angle of attack (deg)	=	5.2781815000
Altitude (ft)	=	33,584.505053
Mach number	=	1.2327106403
Q * ALPHA (deg-lb/ft**2)	=	2,999.9995802

LRB separation:

Staging time (sec)	=	161.78002771
Altitude (ft)	=	158,768.77050
Dynamic pressure (lb/ft**2)	=	38.119230894
Angle of attack (deg)	=	-1.9999997282
Mach number	=	5.1688113143
Inertial velocity (ft/sec)	=	6,869.4351860
Inertial flight path angle (deg)	=	17.164297833
Relative velocity (ft/sec)	=	5,592.7417119
Relative flight path angle (deg)	=	21.252588648
Delta V (ft/sec)	=	10,309.073677
Weight after separation (lb)	=	1,395,368.4103
Remaining ET propellent (lb)	=	1,039,083.4187
SSME throttle at separation	=	1.0900000000
Engine out LRB throttle	=	0.99999992859
Good LRB throttle	=	0.74999994644
Thrust (lb)	=	1,535,435.1198
Thrust-to-weight after separation	=	1.1003797338
Acceleration after separation	=	1.0892212806
LRB propellent used (lb)	=	1,764,968.3500
Engine out remaining prop. (lb)	=	1,299.7607378
Geodetic latitude (deg)	=	28.507103704
Longitude (deg)	=	-79.876282443

Average back pressure (psi)	=	5.6839283734

MECO conditions:

Time (sec)	=	476.79157462
Altitude (ft)	=	358,110.35571
Inertial velocity (ft/sec)	=	25,626.022743
Inertial flight path angle (deg)	=	0.67749034252
Delta V (ft/sec)	=	30,186.591555
Shuttle & payload perigee (nm)	=	-19.460893580
Shuttle & payload apogee (nm)	=	78.741811434
MECO weight (lb)	=	356,284.99108
SSME throttle @ MECO	=	0.75866210066
SSME propellent weight used (lb)	=	1,587,950.0006
ET remaining propellent weight (lb)	=	-5.81789761782E-04
Average back pressure (psi)	=	1.9290743630

Throttle schedules:

Max q throttle down time (sec)	=	60.333605304
LRB throttle setting	=	1.0000000000
Post max q throttle up time (sec)	=	110.32174436
Start LRB 3g throttling (sec)	=	153.98226345
LRB throttle setting (engine out)	=	0.99999992859
LRB throttle setting (good LRB)	=	0.74999994644
SSME throttle @ separation	=	1.0900000000
Start SSME only 3g throttling (sec)	=	422.12671064
SSME final throttle setting	=	0.75866210066

Losses to LRB separation

Total delta V	=	10,309.073677
Steering losses	=	2,396.3657805
Drag losses	=	413.90204502
Gravity losses	=	1,625.0940102
Pressure losses	=	346.37033956

Losses to MECO

Total delta V	=	30,186.591555
Steering losses	=	2,811.8173108
Drag losses	=	420.90477010
Gravity losses	=	2,324.6946468
Pressure losses	=	346.45868139

Min/Max conditions:

Max (+) angle of attack (deg)	=	17.054911020
Time (sec)	=	7.3000000000
Max (-) angle of attack (deg)	=	-7.9848444048
Time (sec)	=	29.557606256

Max (+) Q * Alpha (lbf-deg/ft**2)	=	2,999.9995923
Time (sec)	=	91.917059988
Max (-) Q * Alpha (lbf-deg/ft**2)	=	-621.19423591
Time (sec)	=	113.29013557
Max acceleration (g's)	=	3.0000000000
Time (sec)	=	161.69263956
Max attach load (kips)	=	1,397.7961439
Time (sec)	=	153.98226345

The following table is a summary of the methane pump-fed configuration's mass properties obtained when sizing to the ATO mission.

LO2/CH4 PUMP-FED LRB (2)	SUBSYS	SYSTEM	GROUP	VEHICLE
STRUCTURE			61,181.4	
LH2 TANK (2219 skin stiffner)		18,902.7		
Cylinder section	15,351.8			
Bulk head	1,689.0			
ET Attach frame	1,861.9			
LO2 TANK (2219 skin stiffner)		19,484.4		
Cylinder section	15,397.7			
Bulk head	1,689.0			
ET Attach frame	2,397.7			
LO2 TANK SLOSH BAFFLES		248.7		
CH4 TANK INSULATION		563.4		
LO2 TANK INSULATION		564.7		
NOSE CAP		1,913.6		
FRWD ADAPTER		2,462.1		
INTERTANK ADAPTER		1,973.0		
AFT ADAPTER		9,699.7		
Aft adapter skin	5,335.0			
Aft adapter stringers	361.5			
Aft adapter frame	2,859.4			
Hold down posts	1,143.8			
THRUST STRUCTURE		5,259.1		
4 thrust beams	3,938.1			
4 longerons	533.3			
Engine mount bulk head	454.8			
Skirt aft frame	332.8			
LAUNCH GEAR		110.0		
PROPULSION SYSTEM			38,994.2	
MAIN ENGINES		25,091.5		
ENGINE GIMBAL SYSTEM		3,109.1		
ENGINE PURGE SYSTEM		834.2		
ENGINE MOUNTS		606.7		
MAIN PROPELLANT SYSTEM		9,352.7		
SUB-SYSTEMS			3,743.0	
SEPARATION SYSTEM		1,400.0		
AVIONICS		806.0		
POWER		1,537.0		
ELECTRICAL SYSTEM		0.0		
RECOVERY SYSTEM		0.0		
CONTINGENCY			10,391.9	
DRY WEIGHT				114,310.4
MAIN RESIDUALS			8,831.3	
CH4 FUEL		2,355.0		
LO2 FUEL		6,476.3		
INERT WEIGHT				123,141.8
ASCENT PROPELLANTS			883,134.1	
CH4 FUEL		235,502.4		
LO2 FUEL		647,631.6		
LRB LIFT OFF WEIGHT				1,006,275.8
MAIN START-UP FUEL			42,558.5	

CH4 FUEL		14,244.5		
LO2 FUEL		28,314.0		
STEP WEIGHT				1,048,834.3

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STS Weight Summary	SUBSYS	SYSTEM	GROUP	VEHICLE
Orbiter inert				193,693.0
OV 103 (7)			150,811.0	
SSME x 3 inert			20,958.0	
Buoyancy			80.0	
Crew Module			4,361.0	
Non-Prop. Consumables			5,397.0	
RCS Propellent			6,920.0	
Vented after SSME valve close			230.0	
MPS Propellent @ Ignition			4,936.0	
Orbiter lines - usable		2,782.0		
Orbiter lines - unusable		771.0		
SSME x 3 - unusable		1,383.0		
ET inert				76,892.0
ET dry weight			66,623.0	
ET Buoyancy			175.0	
MPS Pressurant			423.0	
Flight Press. Gas			3,730.0	
Usable propellent			5,046.0	
ET FPR		2,219.0		
BIAS		949.0		
Shutdown Propellent		1,878.0		
LH2	609.0			
LOX	1,269.0			
Unusable Propellent			895.0	
ET wet walls		175.0		
LH2 lines & tank, LOX lines		720.0		
Ascent propellent				1,587,950.0
LH2			225,590.0	
LOX			1,362,360.0	
OMS propellent				15,200.0
OMS Fuel			5,708.0	
OMS Oxidizer			9,492.0	
Payload weight				70,500.0

ENGINE PARAMETERS	NOMINAL	ABORT	MINIMUM
NUMBER	4.0		
WEIGHT	6,272.9		
THROTTLE	100.0	110.0	75.0
OXIDIZER FLOW RATE	1,249.5	1,374.5	937.1
FUEL FLOW RATE	454.4	499.8	340.8
VACUUM THRUST	574,490.7	631,939.8	430,868.1
SEA LEVEL THRUST	531,991.1	589,440.2	388,368.4
CHAMBER PRESSURE (psi)	2,954.5	3,250.0	2,215.9
VACUUM ISP (sec)	337.17	338.60	333.58
SEA LEVEL ISP (sec)	312.22	315.83	300.68
MIXTURE RATIO	2.7500		
NOZZLE AREA RATIO	30.000		
X-AREA (in^2)	2,891.9		
THROAT RADIUS (in)	5.5393		
EXIT DIAMETER (in)	60.680		
OVERALL LENGTH (in)	100.09		

VEHICLE PARAMETERS	
GLOW	3,956,786.6
T/W LIFTOFF (nominal)	1.3605
BOOSTER SL TOTAL (nominal)	4,255,929.0
ORB SL TOTAL (nominal)	1,127,200.9
T/W LIFTOFF (1 LRB engine-out)	1.2000

Note: performance runs use parametric engine data. See the propulsion section for the engine point design.

BOOSTER SL TOTAL (1 LRB engine-out)	3,494,141.6			
ORB SL TOTAL (1 LRB engine-out)	1,254,002.3			
DIMENSIONS/SHUTTLE COORDINATES	LNG.(FT)	STA.(IN)		
FUEL TANK SPACING	2.9			
ENGINE CLEARANCE	3.8			
EXIT PLANE	2.1	2,502.7		
AFT ADAPTER	15.7	2,477.6		
AFT FUEL TANK	43.9	2,289.5		
INTERTANK ADAPTER	14.3	1,762.9		
FORWARD FUEL TANK	44.0	1,591.8		
FORWARD ADAPTER	6.2	1,063.6		
NOSE CAP	20.9	989.6		
NOSE TIP	0.0	738.7		
TOTAL LENGTH	147.00			
VEHICLE DIAMETER	15.723			
Length/Diameter	9.3492			
NOSE-CAP GEOMETRY				
Nose fineness ratio	1.3300			
Nose bluntness ratio	0.20000			
Conic angle (deg)	17.654			
Nose length (ft)	20.912			
Nose cap spherical radius (ft)	1.6500			
Description	Radius	Height	Area	Weight
Nose cap	1.5723	1.1496	11.357	
Conic section	7.8615	19.762	614.63	
Totals	0.00000	20.912	625.99	1,913.6
Aft Skirt Diameter Inputs/Results				
Nozzle Exit Plane Thickness (in)	5.0000			
Nozzle Outsize Diameter (in)	70.680			
Engine Gimbaling Length (in)	90.084			
Maximum Gimbal Angle (deg)	6.0000			
Gimbaling distance pad (in)	5.0000			
Gimbaling distance (in)	14.029			
Aft diameter (in)	218.54			
Propellant tanks (Skin Stiffener	Oxidizer	Fuel		
Tank diameter	15.6	15.6		
Material Density	0.10300	0.10300		
Bulkhead				
Radius/Height	1.3784	1.3784		
Wall thickness (in)	0.18000	0.18000		
Length	5.7	5.7		
Eccentricity	0.68825	0.68825		
Surface area	316.3	316.3		
Volume	727.0	727.0		
Cylinder section				
Wall thickness (in)	0.48000	0.48000		
Inside diameter (in)	187.72	187.72		
Length	44.0	43.9		
Surface area	2,162.8	2,156.4		
Volume	8,458.1	8,432.9		
Totals				
Total tank volume	9,912.2	9,887.0		
Total surface area	2,795.4	2,789.0		
Occupied volume	9,614.8	9,590.4		
Propellent density	70.976	26.287		
Total propellent	682,422.0	252,101.9		
Ullage %	3.0	3.0		

Figures 6.4.2-1 thru 6.4.2-9 show various performance parameters obtained from the methane pump-fed LRB configuration's ATO trajectory simulation.

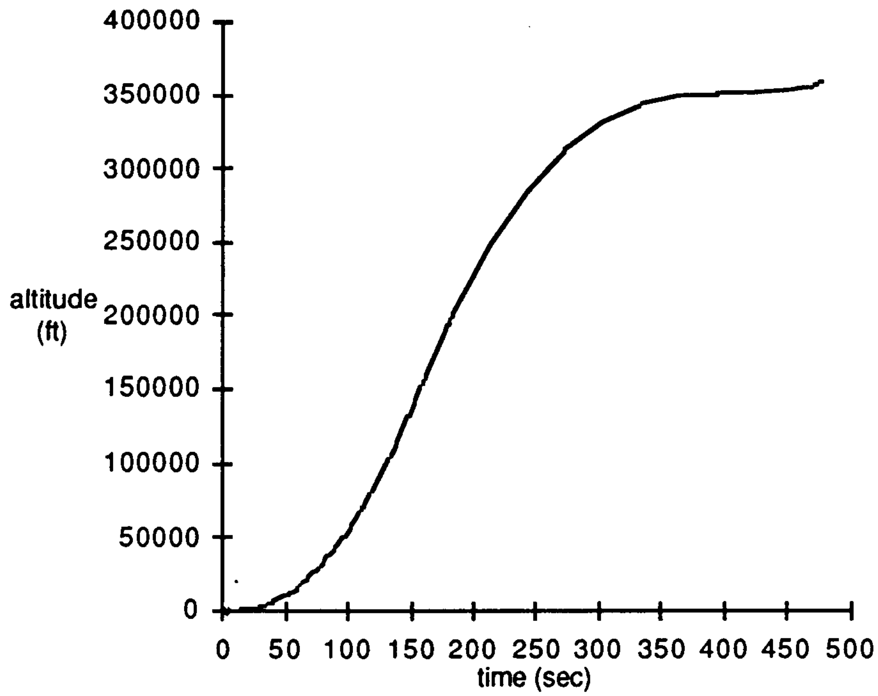


Figure 6.4.2-1 Altitude vs Time

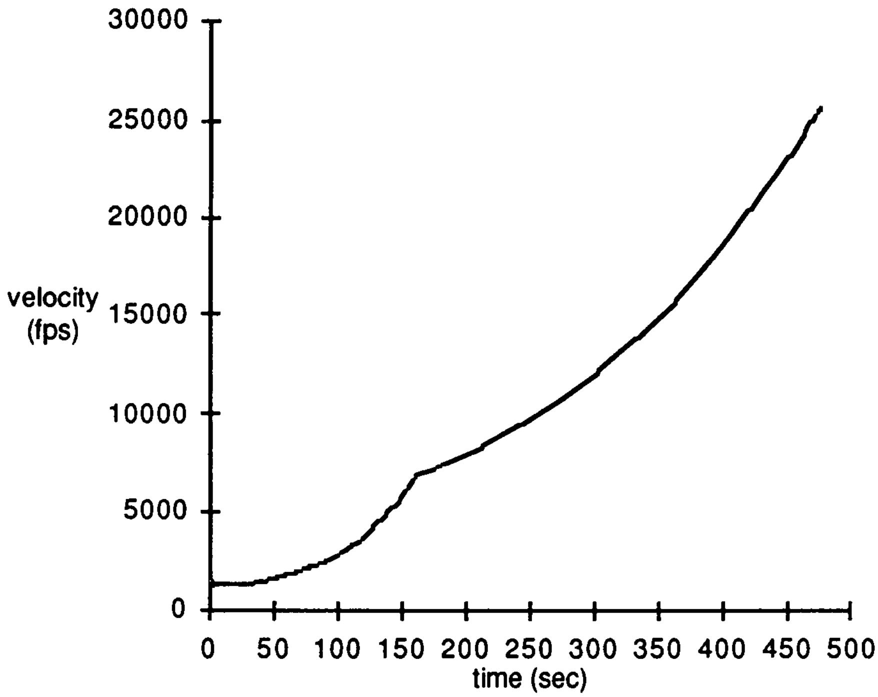


Figure 6.4.2-2 Velocity vs Time

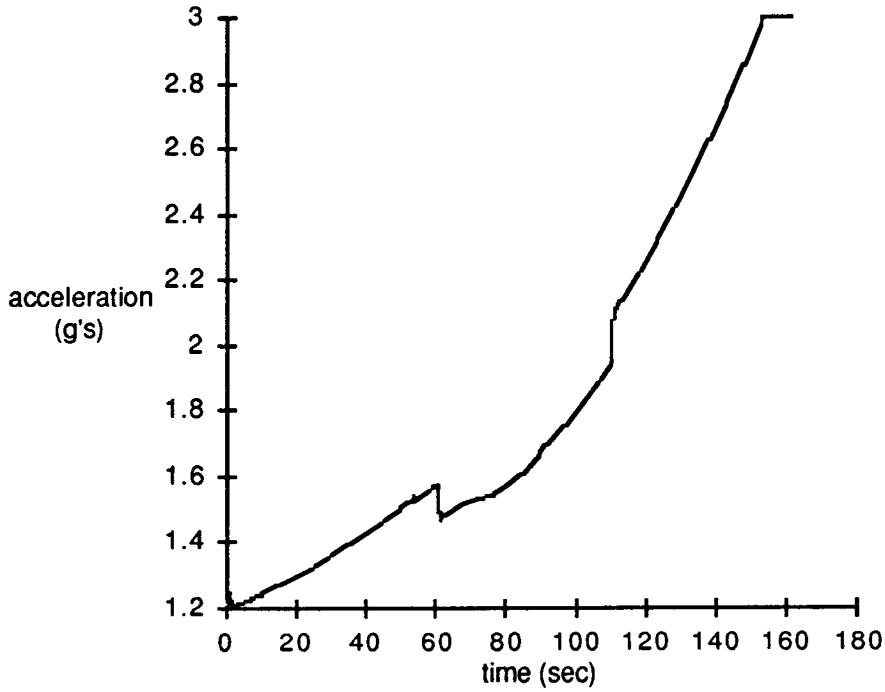


Figure 6.4.2-3 First Stage Acceleration vs Time

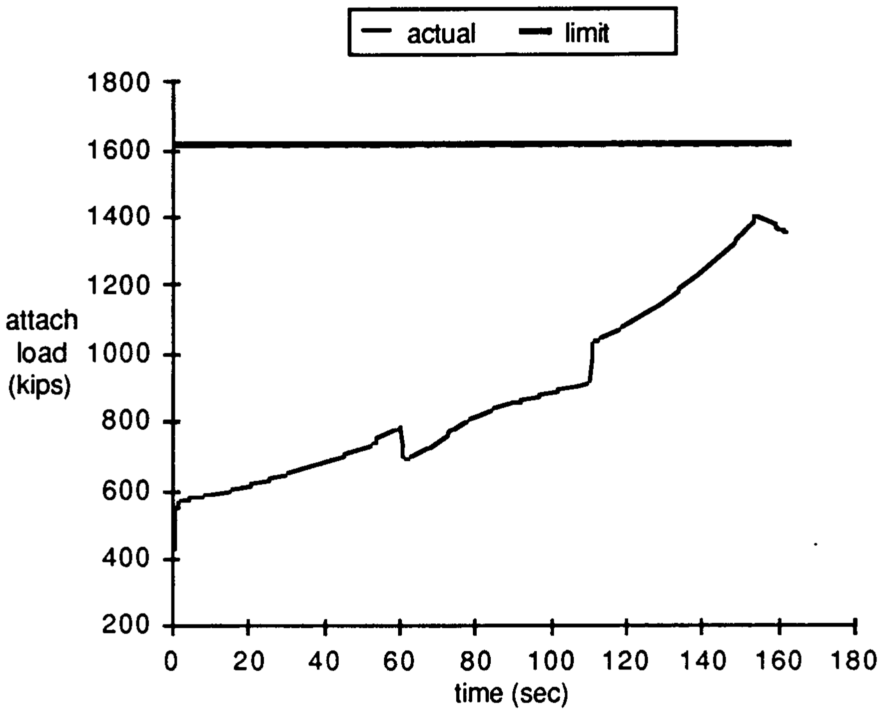


Figure 6.4.2-4 ET Attach load vs Time

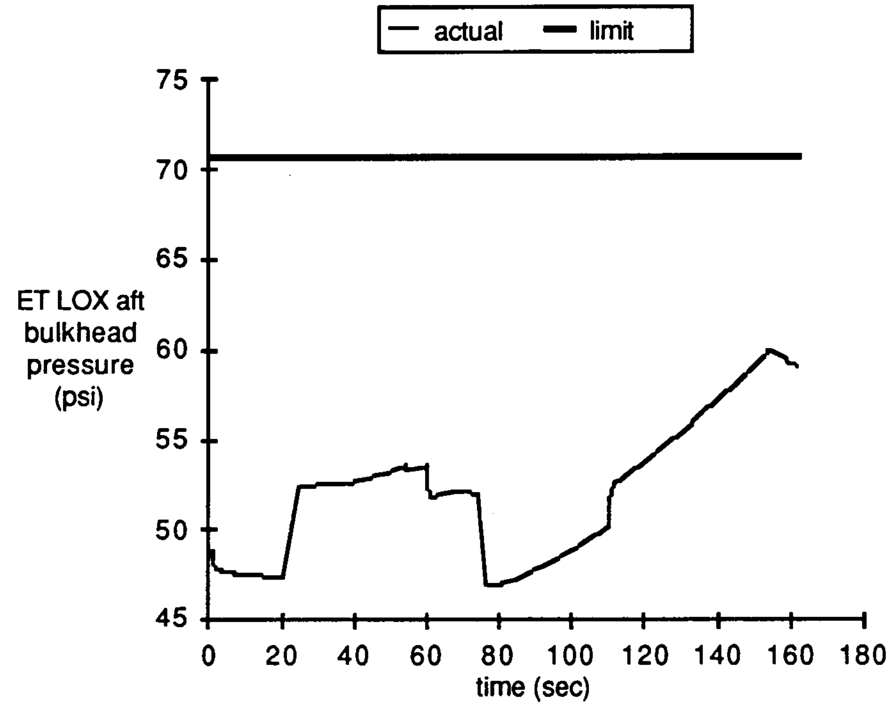


Figure 6.4.2-5 ET LOX aft bulkhead Pressure vs Time

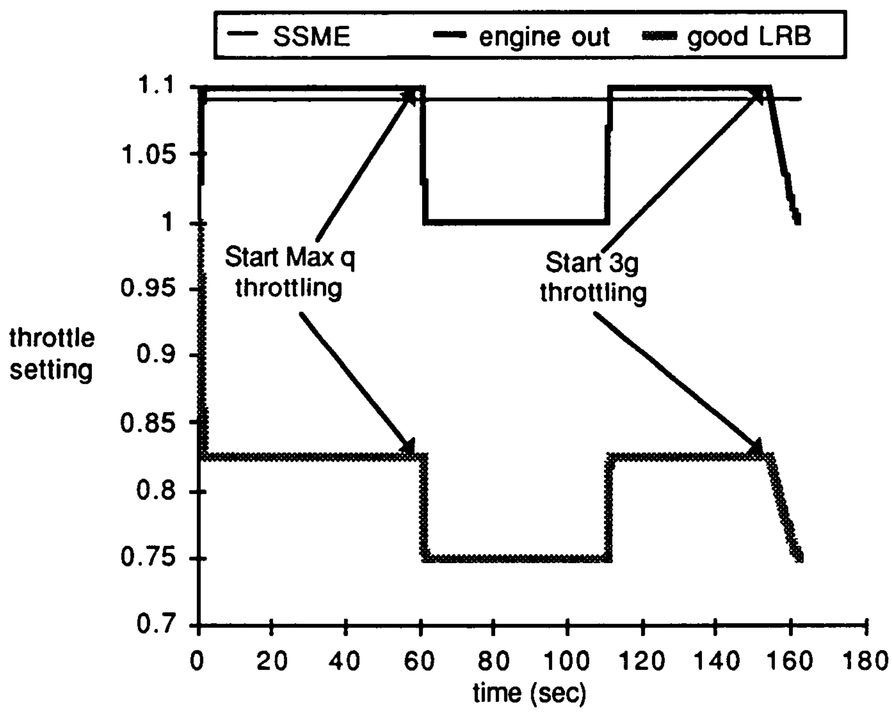


Figure 6.4.2-6 Throttle setting vs Time

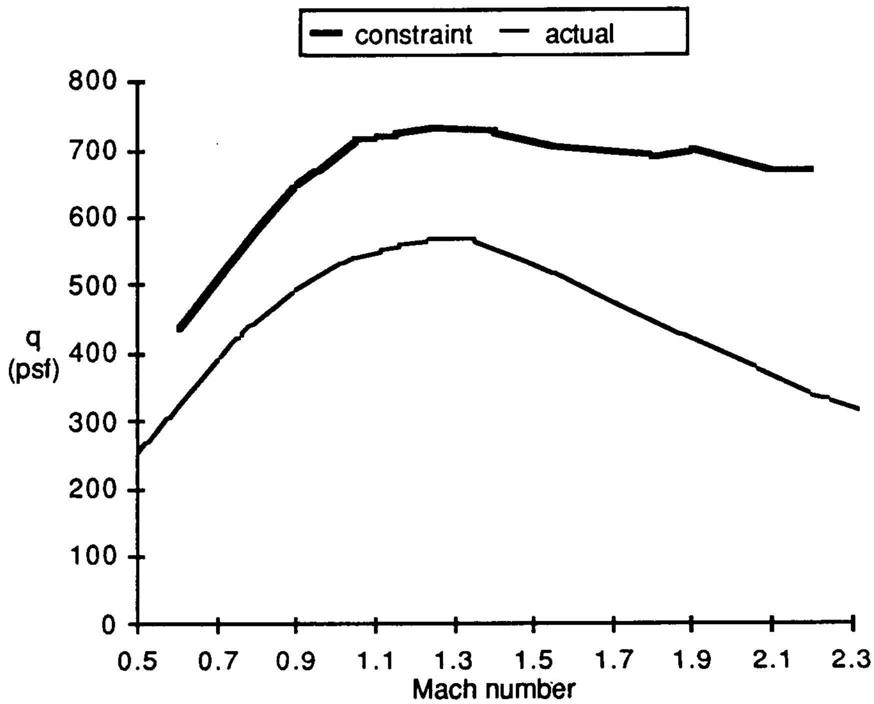


Figure 6.4.2-7 Dynamic pressure vs Mach number

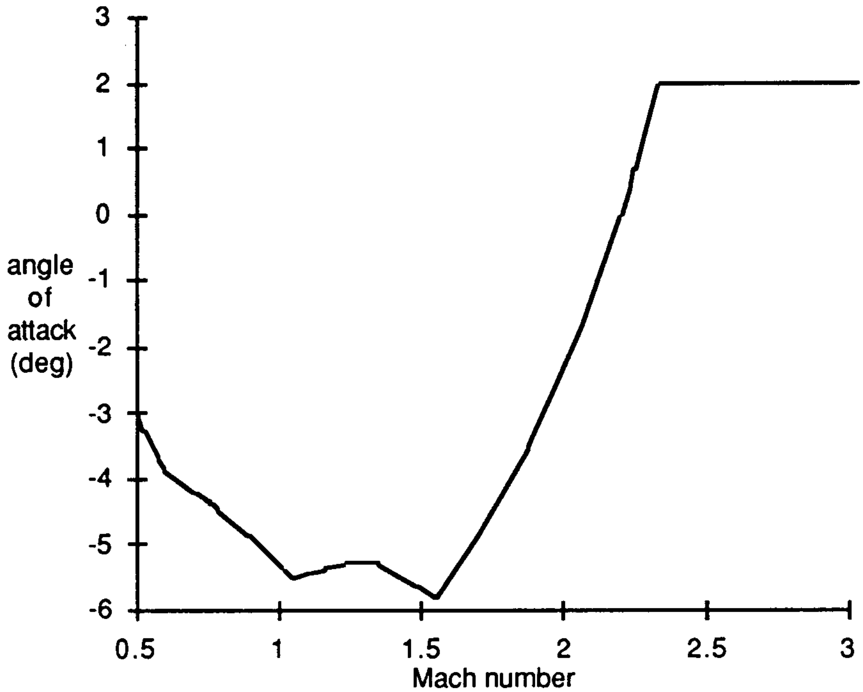


Figure 6.4.2-8 First Stage Angle of Attack vs Mach number

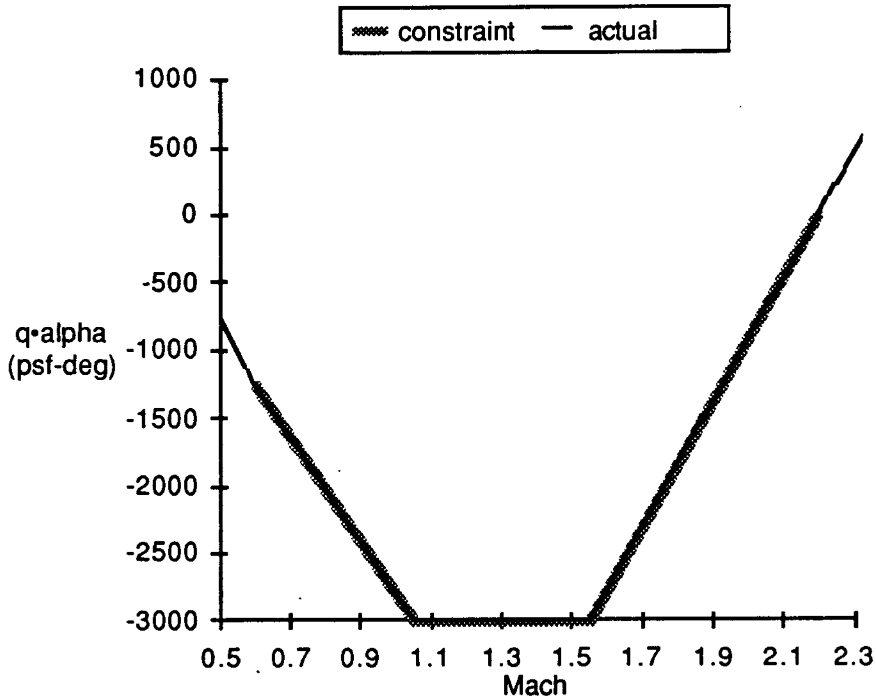


Figure 6.4.2-9 First Stage qa vs Mach

Sizing the methane pump fed LRB to meet the minimum ATO conditions with a single LRB engine out at lift off is not without its penalties. The penalties involved in sizing to meet ATO mission requirements are shown in the following table.

	ATO sizing	Nominal Sizing	Δ	Δ%
LRB Length (ft)	147.0	146.2	0.8	0.5
LRB Diameter (ft)	15.72	15.18	0.54	3.6
Dry weight (Klbs)	114.3	110.5	3.8	3.4
Ascent Propellent (Klbs)	883.1	823.5	59.6	3.9
LRB GLOW (Klbs)	1,006	942.2	63.8	6.8
LRB Vacuum Thrust (Klbs)	574.5	549.2	25.3	4.6

SECTION 7
LOX/RP-1 PRESSURE-FED CONCEPT

The pressure fed LRB uses LO2/RP-1 propellants and the tanks are pressurized to feed the engines. The engines have a minimum Pc of 200 psia at the minimum throttle setting. The engines have regeneratively cooled combustors and nozzles and are head-end gimballed with electromechanical actuators. The engines are sized for nominal thrust and can be throttled down and have a 108 inch I.D. nozzle exit. The propellant tanks are made of 2219 aluminum alloy and are welded. The nose cone, intertank adaptor and apt skirt are made of 2024 aluminum alloy and use mechanical fasteners. The LO2/RP-1 pressure fed concept offers the major advantage of a low cost, reliable engine.

7.1 STRUCTURE AND MECHANISM MATERIALS

7.1.1 VEHICLE CONFIGURATION. The General Dynamics Pressure-Fed Liquid Rocket Booster uses LOX/RP-l fuel and is shown in Figure 7.1.1-1. The diameter is 15 feet and the overall length is 199.5 feet.

The nose cone is 22.2 feet long and has a fineness ratio of 1.48. The nose cone houses the primary and secondary helium spheres to pressurize the LOX tank, and the forward LRB separation motors.

The LOX tank has cylindrical section length of 82.4 feet and two elliptical end dome bulkheads, each being 5.4 feet, for an overall tank length of 93.2 feet. The LOX tank volume is 15487 cu. ft.

The intertank adaptor is 12.7 feet long. The length is established by the space requirements for the LOX feed line.

The RP-l tank has an overall length of 55.5 feet. The cylindrical section is 44.4 feet and has two elliptical end dome bulkheads. The tank volume is 8686 cu. ft.

The aft skirt is 29.6 feet long and the shape is a frustum of a cone. The forward diameter is 15 feet and the aft diameter of 26.8 feet is determined by the rocket motor nozzle size and providing clearance for a gimbal angle of 6 degrees. The rocket motor nozzle exit is 4.9 feet below the aft edge of the skirt.

The pressure-fed LRB attachment to the external tank is at the same locations as the solid rocket motor booster. Having a common datum plane at the aft edge of the skirt for the LRB and SRB hold down fittings, the aft attachment to the ET is 34.97 feet and the forward attachment is 124 feet.

The General Dynamics Pressure-fed LRB LOX and RP-l tanks are manufactured from 2219-T6 aluminum alloy. The tank sections are joined together using VPPA type welding. The nose cone, inter-tank adaptor and aft skirt are made from 7075 aluminum alloy. All structure fabricated from 7075 uses mechanical fasteners such as rivets, huck bolts screws, etc. for joining parts. The above alloys were selected because they have been used for

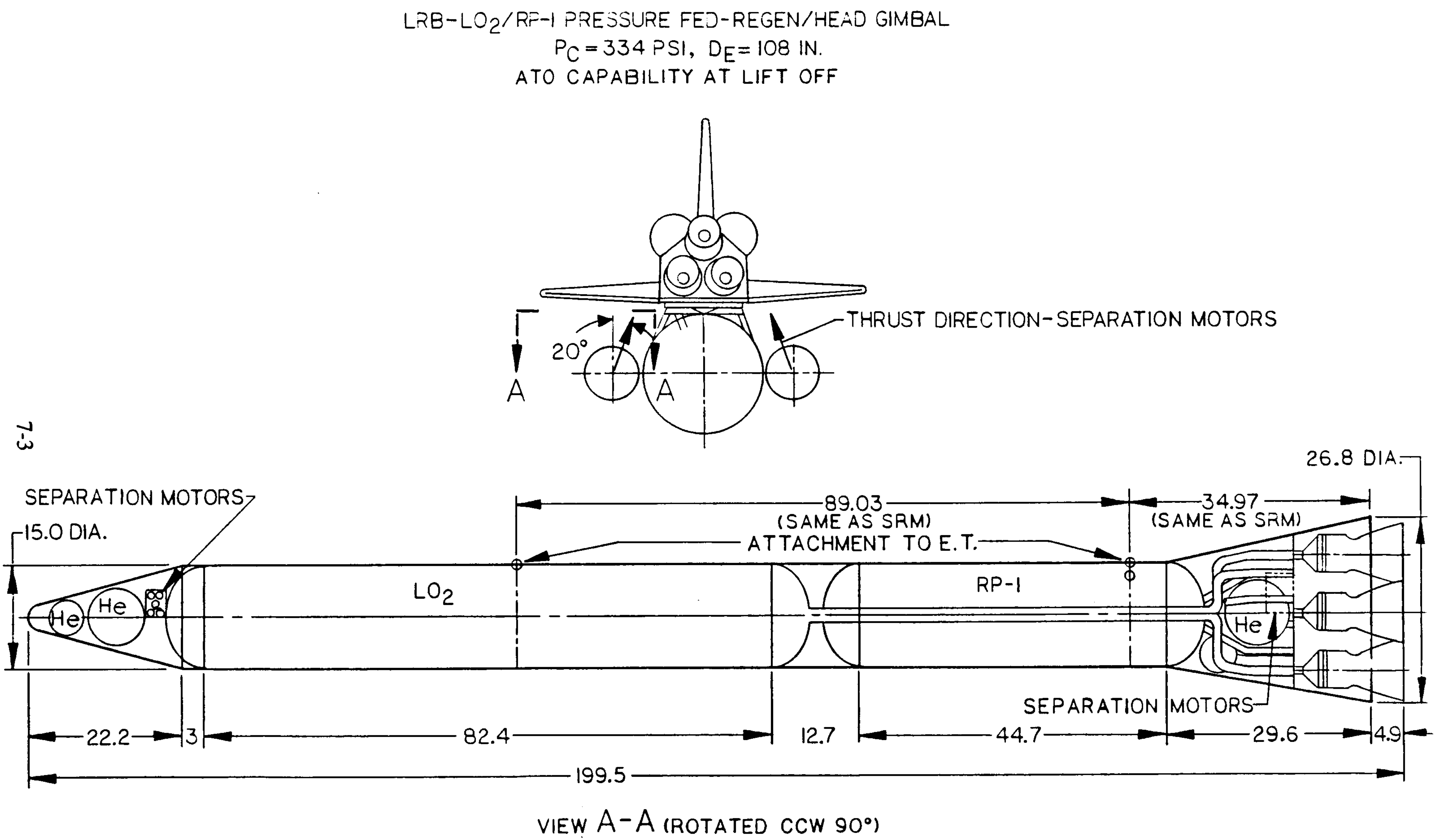


Figure 7.1.1-1 Pressure-Fed LO2/RP-1 Configuration

many years and there is a large data base on allowable stresses and manufacturing methods. This results in a safe, reliable, efficient and low cost booster.

An evaluation was made on the effects of increasing the diameter to 18 feet. The comparative effects are shown in Figure 7.1.1-2. It was not apparent that the increased weight and resulting thrust offered any advantage with the decreased height. Therefore, this LRB configuration was maintained as shown in Figure 7.1.1-1.

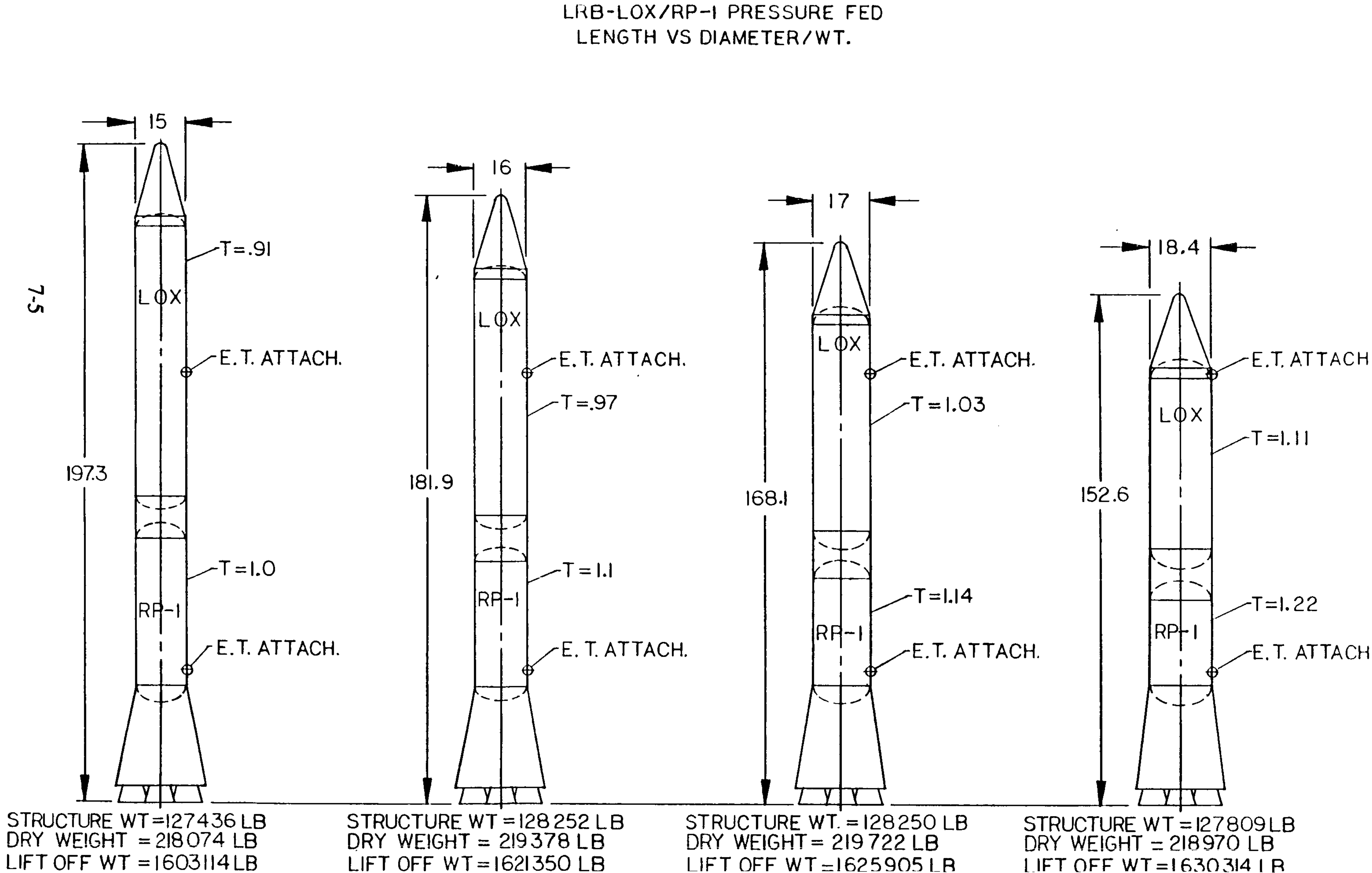
Tanks. The LOX and RP-l tanks are monocoque structures with internal frames at load introduction points and baffles. The cylindrical sections are roll ring forgings, that can be obtained in lengths up to 52 inches long. The cylindrical, 1.0 inch thick, ring sections are butt welded together by circumferential VPPA welding. There are no longitudinal weld seams. The load due to internal tank pressure is twice as high on longitudinal seams as it is for circumferential seams. By eliminating the longitudinal seams, very thick weld lands have been eliminated and safety/reliability improved.

The LOX tank requires 18 roll ring forgings to make up the cylindrical tank section. The RP-l tank uses 10 roll ring forgings in the cylindrical section as shown in Figure 7.1.1-3. The connection details of the RP-1 tank are illustrated in Figure 7.1.1-4. The elliptical end domes of the tanks are a large single plate spun formed. The only welding required is around the perimeter of the dome to the bulkhead to tank adapter, see Figure 7.1.1-5. The top dome of each tank has a bolted access cover to each tank. The access cover has metallic K type seals for the LOX tank and elastomer seals for the RP-l tank. The domes have provisions for fill, drain, vent and pressure lines. The domes are chem-milled to provide mounting lands for the above and to reduce dome weight. Surfaces requiring seals are machined to provide the required flatness and surface finish.

A roll ring forging that is machined, provides continuity of the structure to the dome, the tank cylinder section and the adapter skirt. The part is VPPA welded to the above parts, providing structural continuity.

The forward attachment point to the external tank (ET) is located approximately mid-length of the LOX tank, and is shown in Figures 7.1.1-6a, 7.1.1-6b, and 7.1.1-6c. The attachment fitting is machined

Figure 7.1.1-2 Pressure-Fed Diameter Comparison



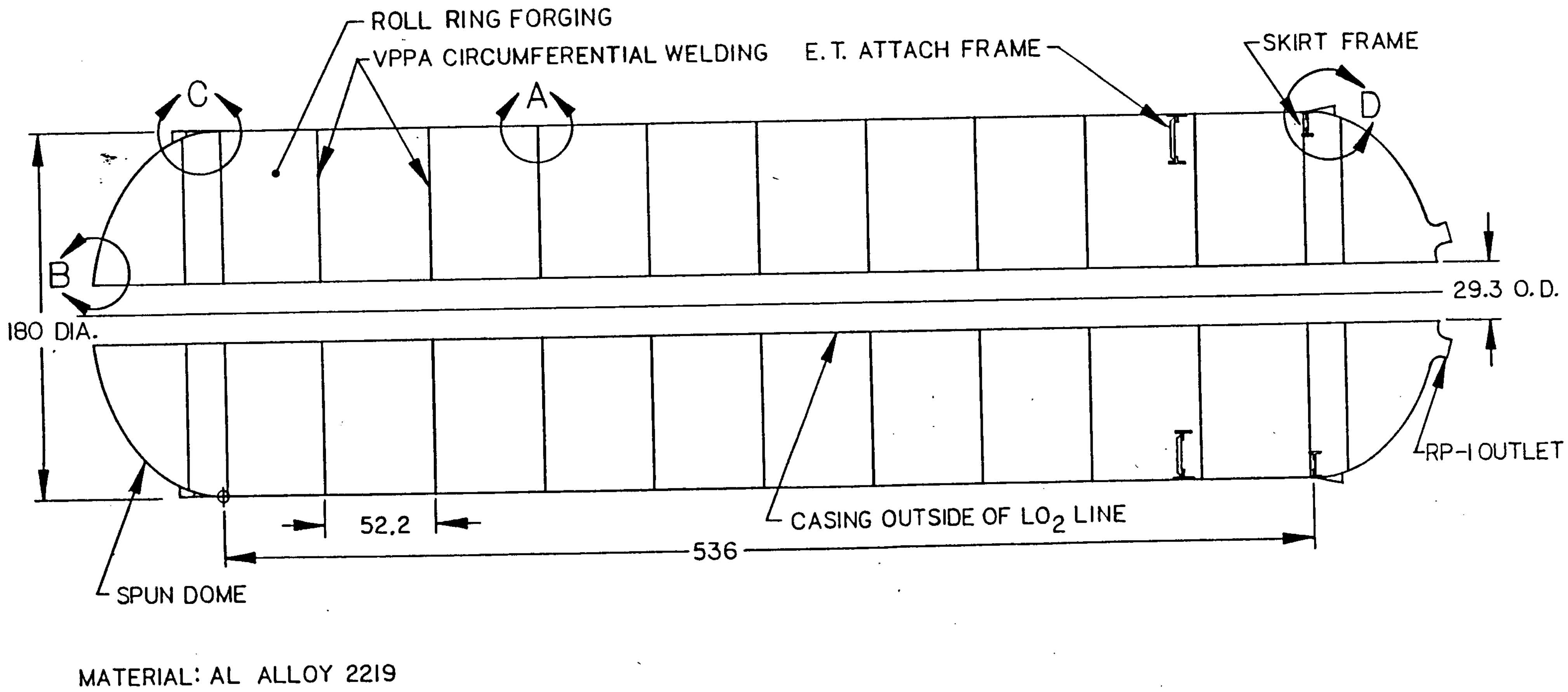


Figure 7.1.1-3 Pressure Fed LRB RP-1 Tank Structure Layout

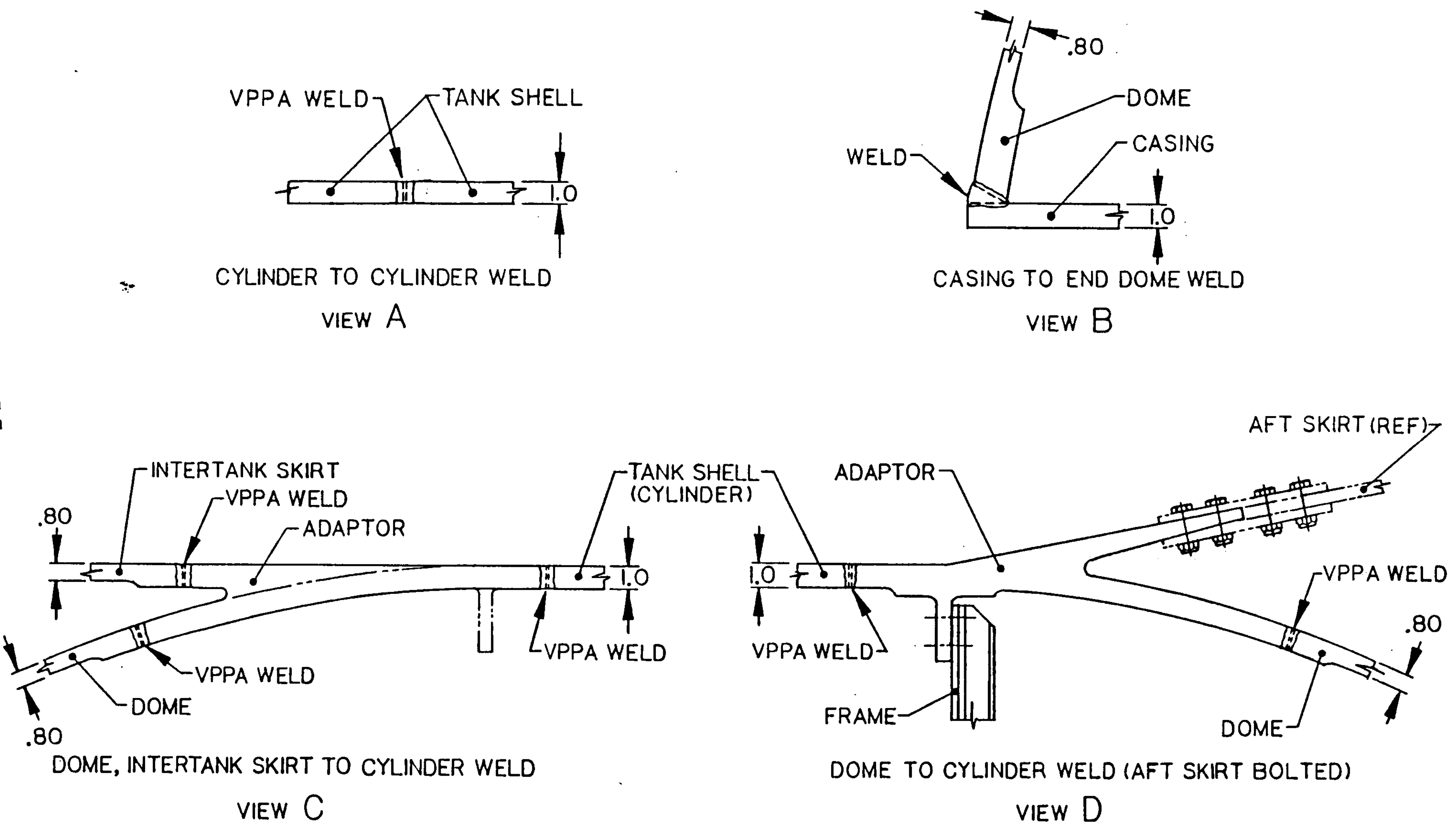
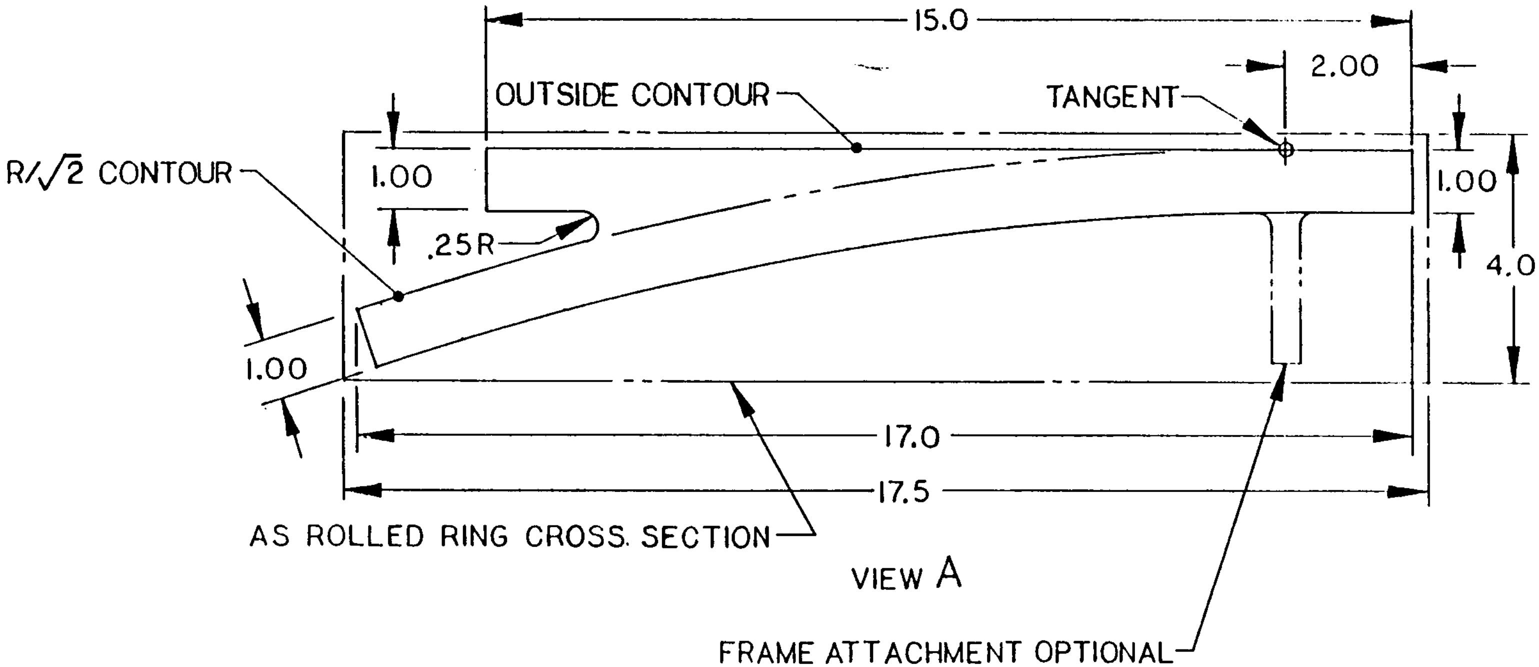
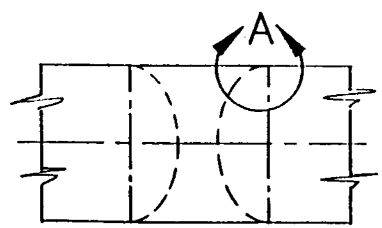


Figure 7.1.1-4 Pressure Fed LRB RP-1 Tank Structure Details



DIMENSIONS IN INCHES
MATERIAL: 2219 AL ALLOY
2519 AL REDUCES THICKNESS 10%
PART IS A ROLL RING FORGING



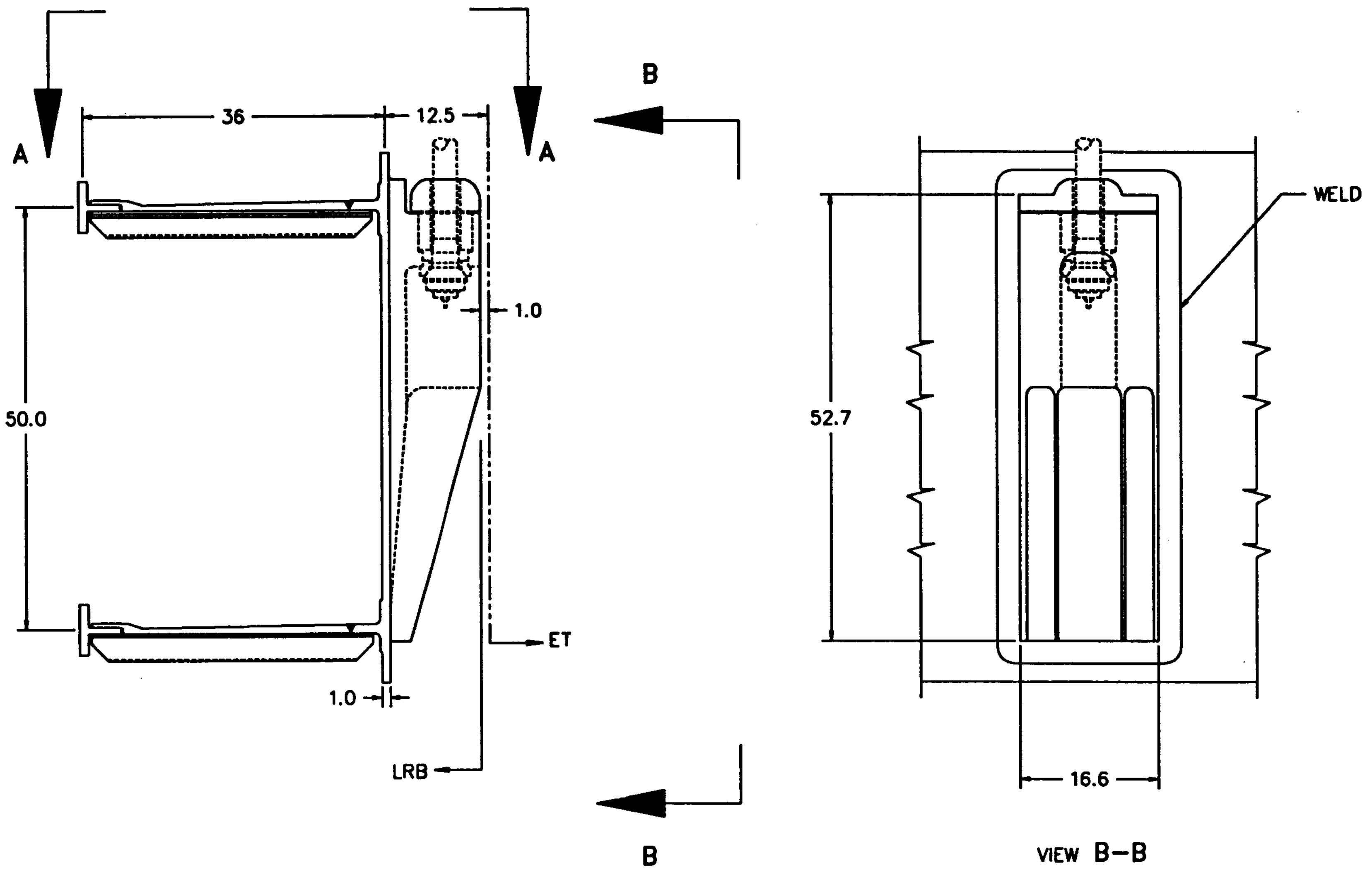


Figure 7.1.1-6a ET/LRB Forward Attachment Fitting

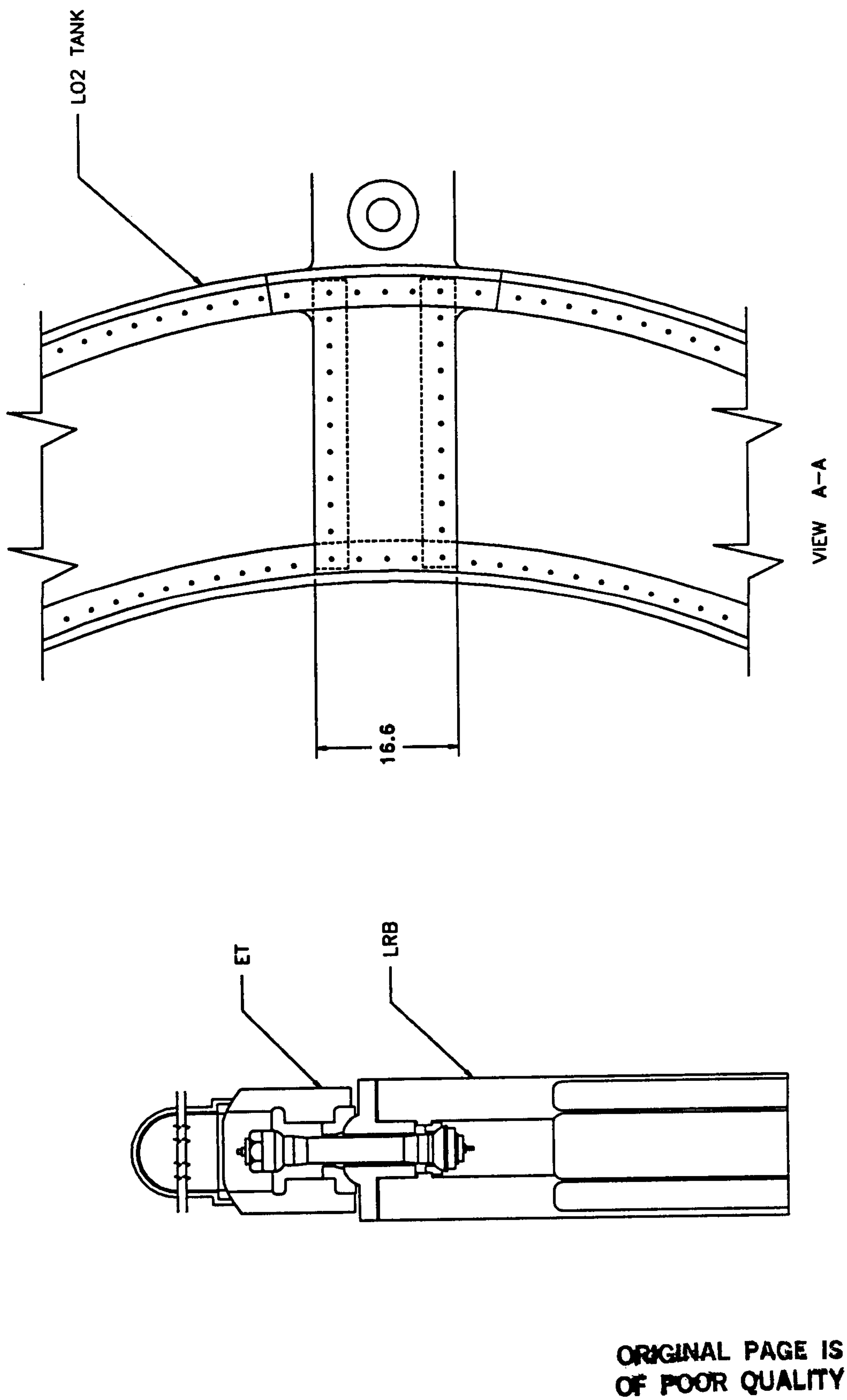


Figure 7.1.1.6b ET/LRB Forward Attachment Fitting

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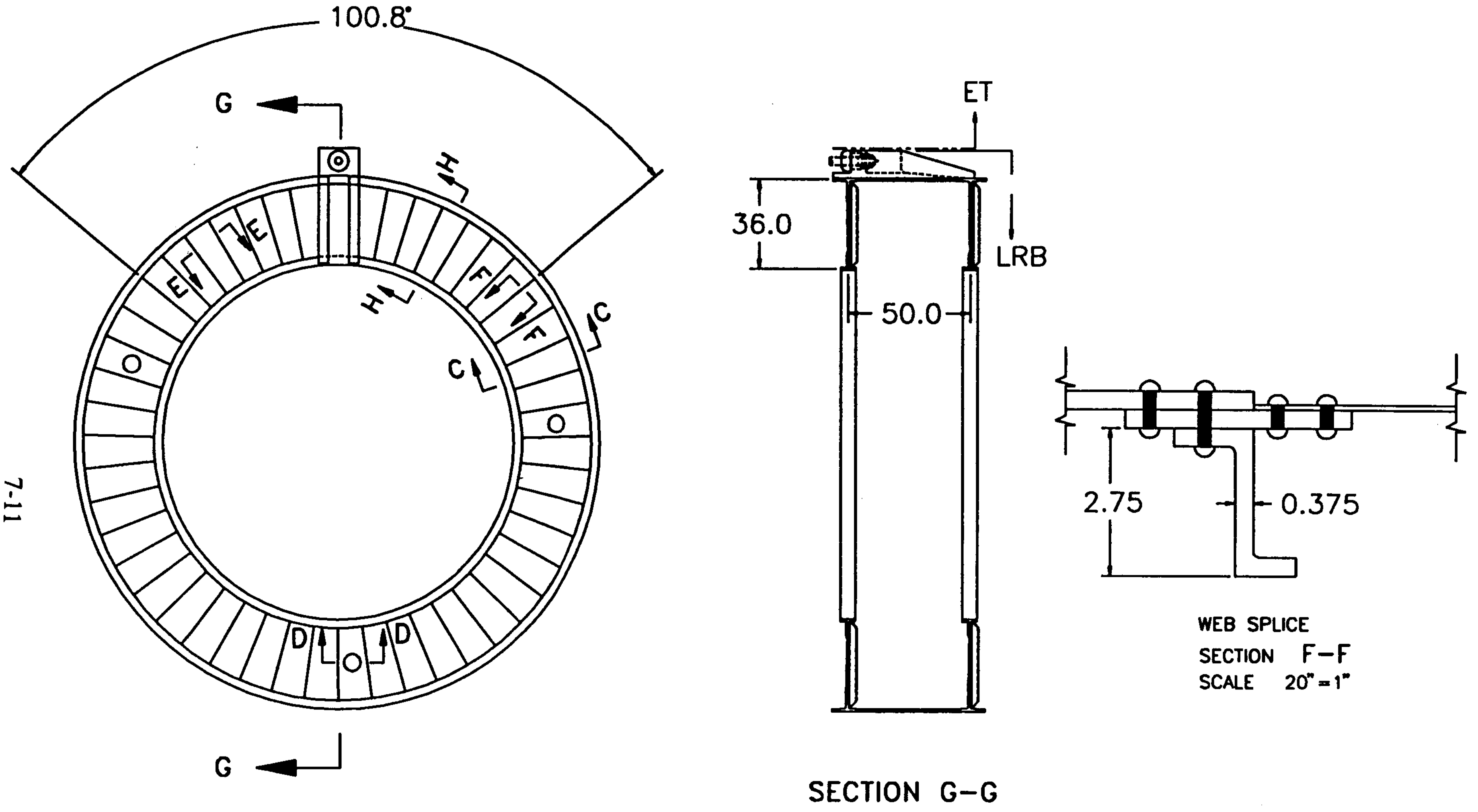


Figure 7.1.1-6c ET/LRB Forward Attachment Ring Frame
Support Cross Section View

from forged 2219-T852 aluminum alloy. The fitting picks up the ET fitting and is welded to the internal LRB tank frame and the tank skin. The internal frame web and stiffeners are mechanically fastened to an integral stiffener in the tank skin Figure 7.1.1-7a and 7.1.1-7b. The tank skin is thicker locally at the integral stiffener, and is the outer cap material of the frame. The inner frame cap in turn picks up the web and stiffeners. Mechanical fasteners are used internally, where there can be no leaks to the exterior of the tank. The ET aft attachment in the fuel tank, shown in Figure 7.1.1-8, is similar in construction as the forward attachment except for the external fittings. The internal web decreases in thickness, as the shear stresses are reduced. The RP-l tank has an internal cylinder casing, which runs the full length of the tank to accommodate the LO2 down comer feed line. The casing is welded to the upper and lower elliptical dome bulkheads of the RP-l tank. The casing outside diameter 29.3 inches and a wall thickness l.0 inch. The 23.5 inch outside diameter LOX feed line runs inside the above casing through the RP-l tank. The LOX line has external insulation and is supported at intervals along its length by glass fabric/epoxy supports (G-10 CR or equivalent). The supports provide low heat transfer from the room temperature RP-l tank to the cold LOX feed line.

Vortex baffles are installed in both tanks in the region of the fuel exit. Slosh type baffles are being investigated to determine if they are required.

Nose Cone. The nose cone is a semi-monocoque structure consisting of skin, longerons, frames and a nose cap. Figure 7.1.1-9 shows the nose cone structure. The skin is 0.080 inch thick and rolled to a cone shape. There are eleven circular frames made from extruded I sections. The four longerons are extruded T sections and are machined to vary the cross section as required along the length. The nose cap is a single piece and is spun to the half sphere shape.

The primary and secondary helium spheres are each supported by a frame. The support fitting on one side of the frame takes load in any direction (X, Y & Z). The fitting on the other side of the frame is a slip fit, and reacts load, except in the direction of the sliding motion. This motion accommodates expansion and contraction of the sphere and the structure.

The separation motors are located in and are supported by the nose cone structure. The motors are between the primary helium sphere and the LOX tank.

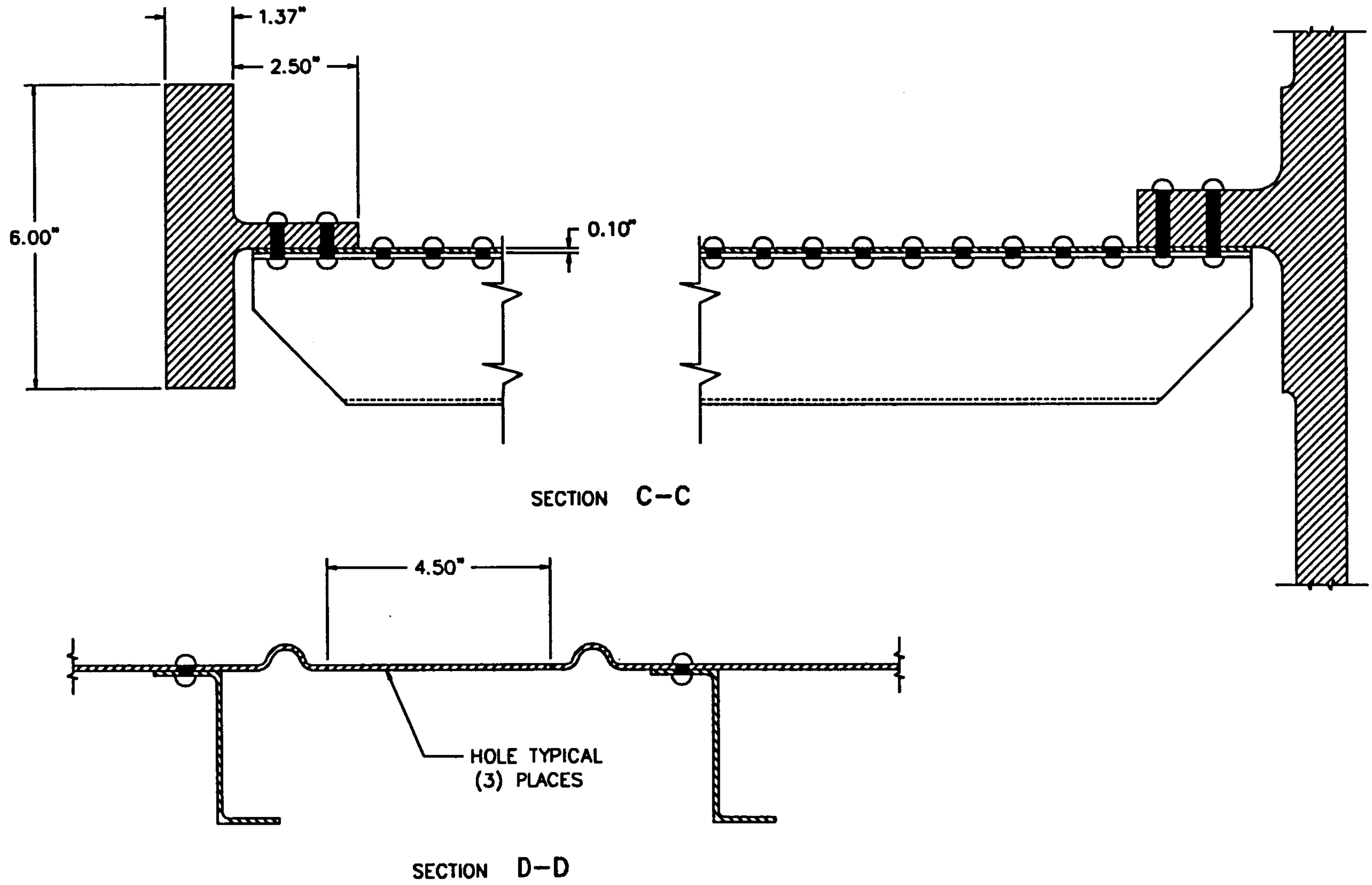
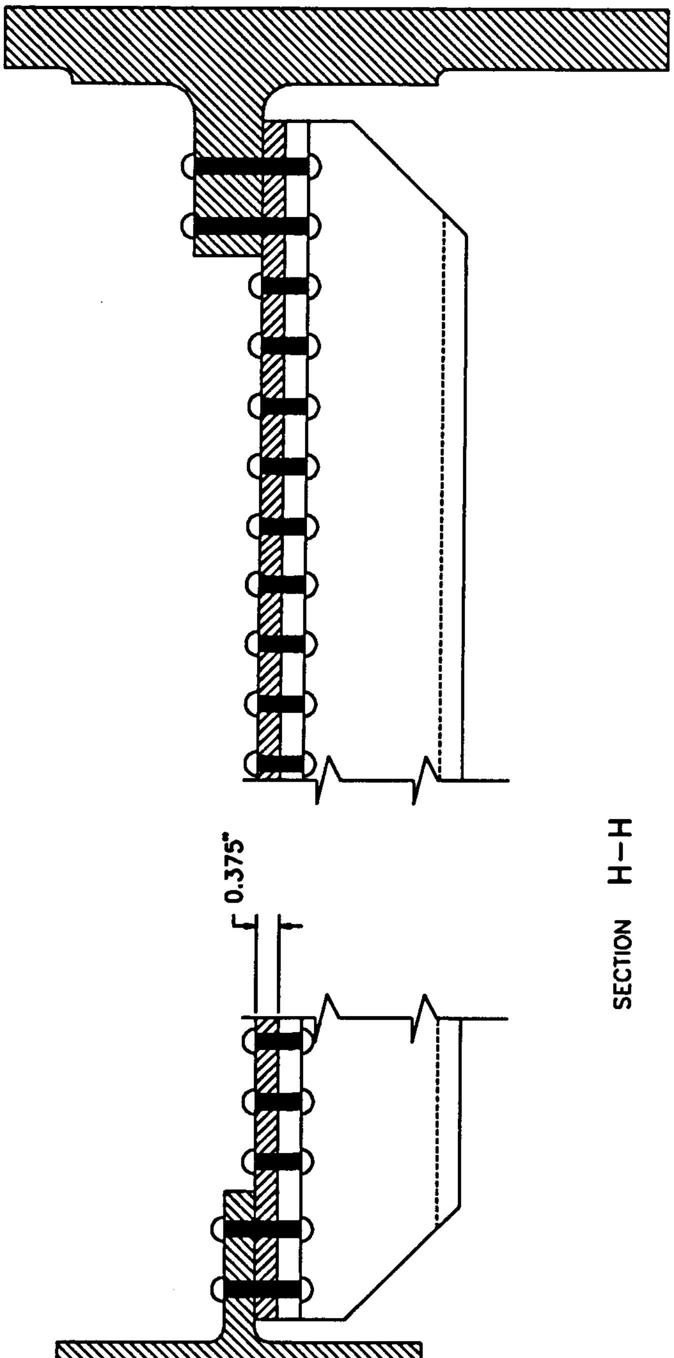
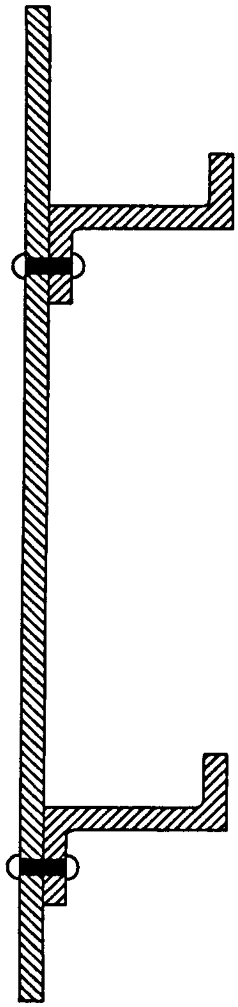


Figure 7.1.1-7a ET/LRB Forward Attachment Ring Frame Web Cross Section View



SECTION H–H



SECTION E–E

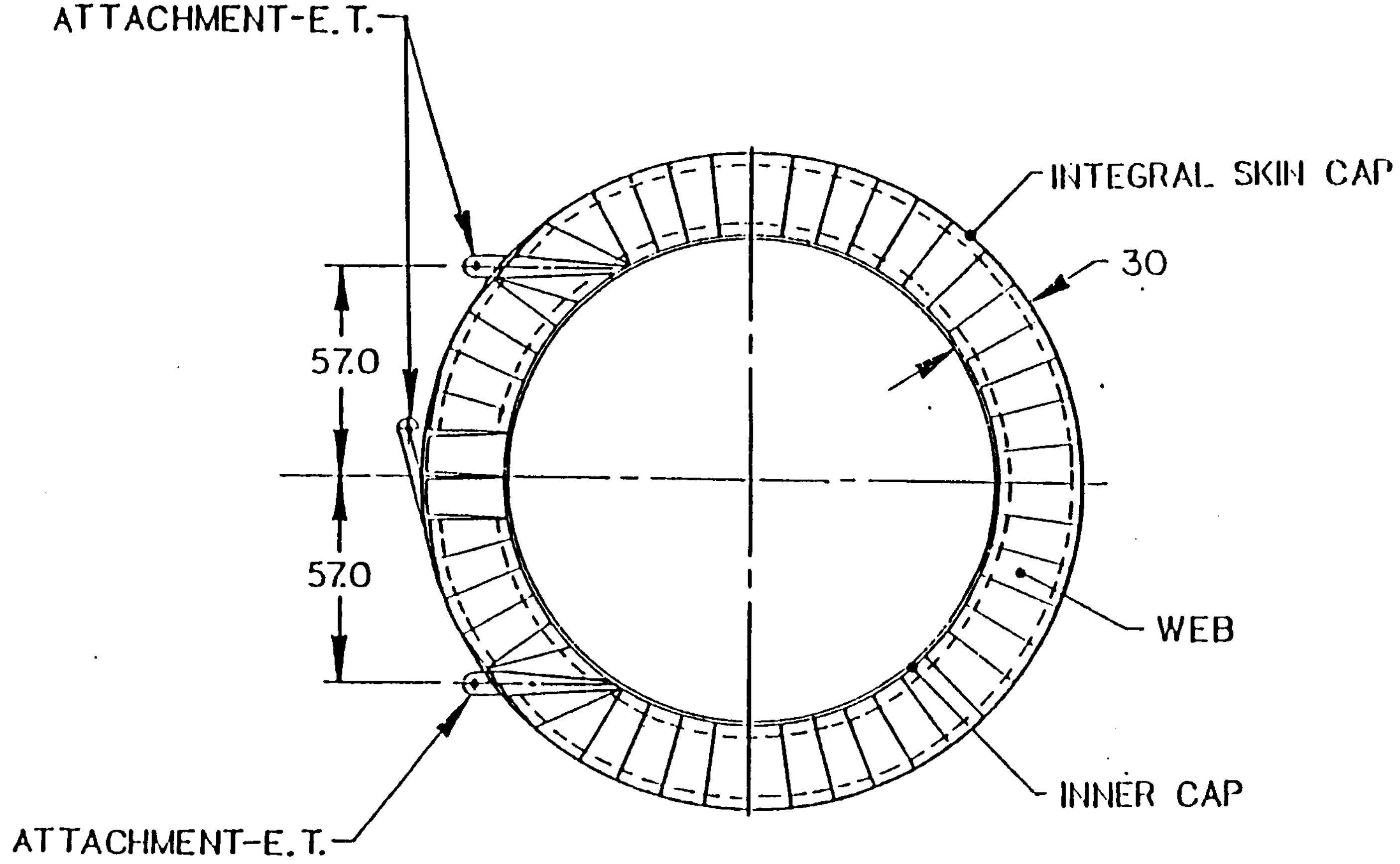
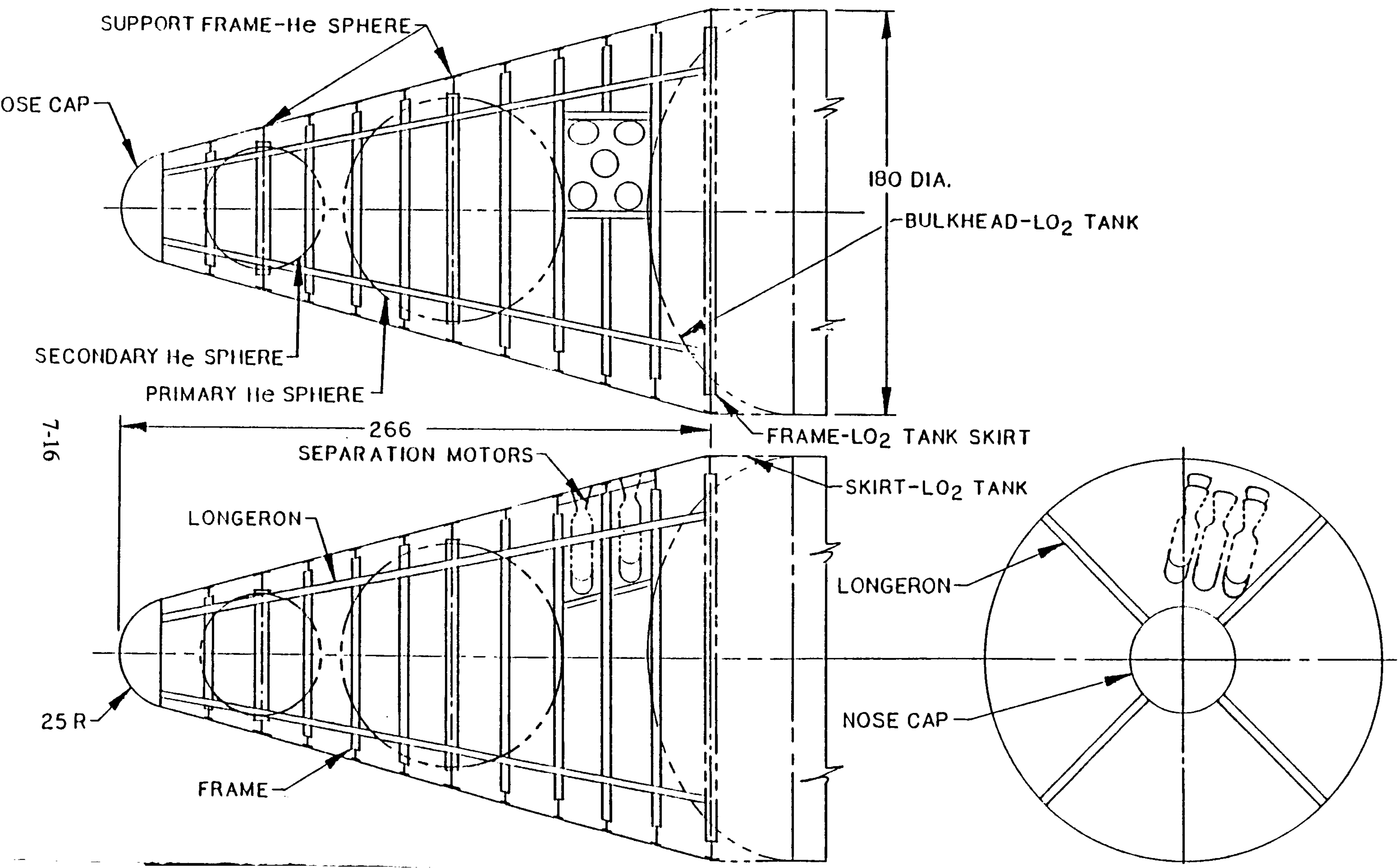


Figure 7.1.1-8 LRB Tank ET Aft Attachment

LO2/RP-1 PRESSURE FED
NOSE CONE ASSEMBLY



The nose cone and nose cap are attached by screws. This allows removal and reinstallation. Figure 7.1.1-10 shows the nose cone and nose cap attachment.

Intertank Adaptor. The intertank adaptor is a corrugated 7075-T6 aluminum alloy skin structure with internal frames at approximately 24 inch spacing. Figure 7.1.1-11 shows the intertank adapter structural arrangement. Figure 7.1.1-12 shows the intertank adaptor to tank skirt splice arrangement. A corrugated structure was selected since it is an efficient, low cost, low weight approach. Also, a corrugated structure readily conforms to expansion and contraction differences between the LOX and RP-l tank. The adaptor has a bolted splice to the LOX tank skirt and the RP-l tank skirt. The access panel into the intertank adapter is shown in Figure 7.1.1-13.

Aft Skirt. The aft skirt structure consists of the skin, frames, and longerons (hold down fittings and engine thrust structure). Figure 7.1.1-14 shows the skirt structure and engine installation. The skin is 0.438 inch thick and rolled to the required radii. The frames are extruded 7075-T6 aluminum alloy I sections and are spaced at approximately 24 inches. The LRB hold down fittings (four 4 per LRB) are on the exterior surface of the skin and run the length of the skirt. Figure 7.1.1-15 is a view looking at the base of the engines and also shows the holddown fittings and the aft frame of the skirt. The hold down fitting has the maximum cross-section area at the aft end of the skirt and tapers to a minimum area at the forward end. The engine thrust structure is a box pattern (Figure 7.1.1-16), supported by four beams that distribute the loads to the skirt skin. The beam depth is large, to minimize deflections and not affect the gimbal angle of the engines.

The primary and secondary helium pressure bottles for the RP-l tank are located in the skirt. The large primary helium bottle is located between the engine thrust beams and is supported by the beams. The fuel lines are supported by glass fabric/epoxy laminate structure. The structure becomes a truss type rigid support when riveted in place. Screws or bolts are used to splice the aft skirt to RP-l tank. Mechanical fasteners are used to join or splice parts in the skirt structure. The aft booster separation motors are located on the aft skirt in a manner similar to the SRBs.

LRB To STS Interfaces. The following sections will describe the efforts made to define the interfaces between the pressure-fed LRB configuration and the STS system. The interfaces describe are the ET fore and aft fittings, the LRB support and release system

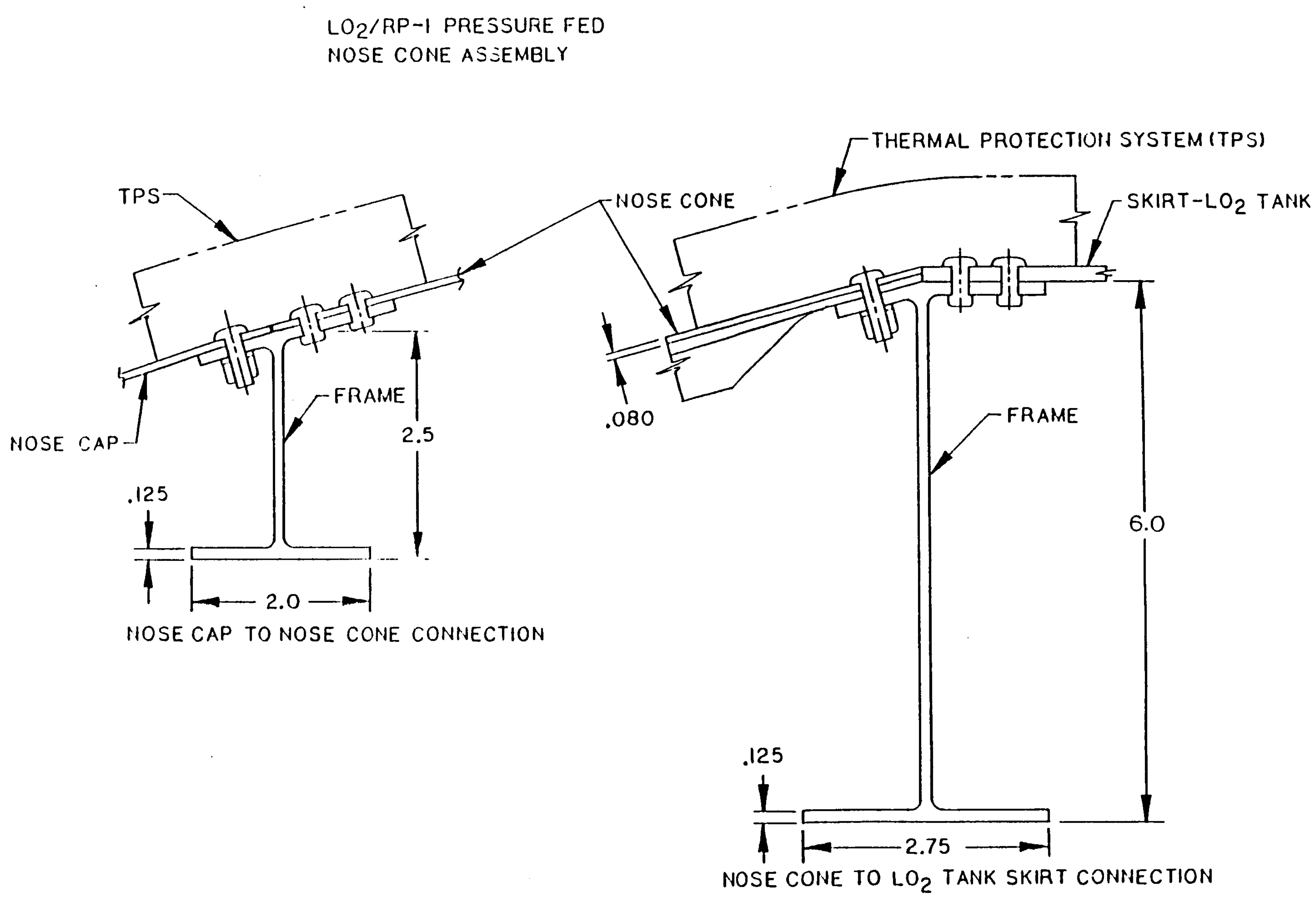


Figure 7.1.1-10 Nose Cone Assembly - Detail of Attach Point

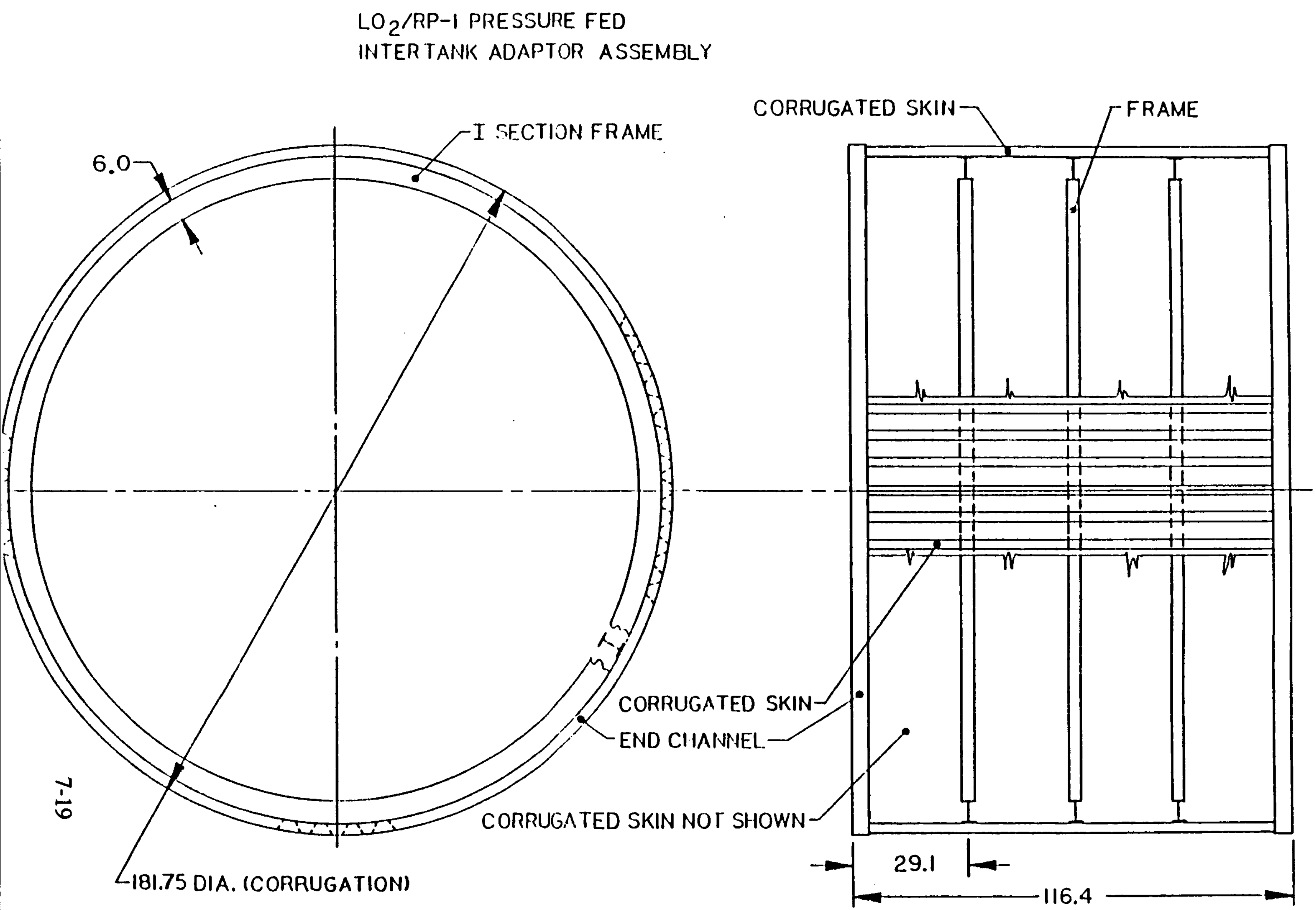
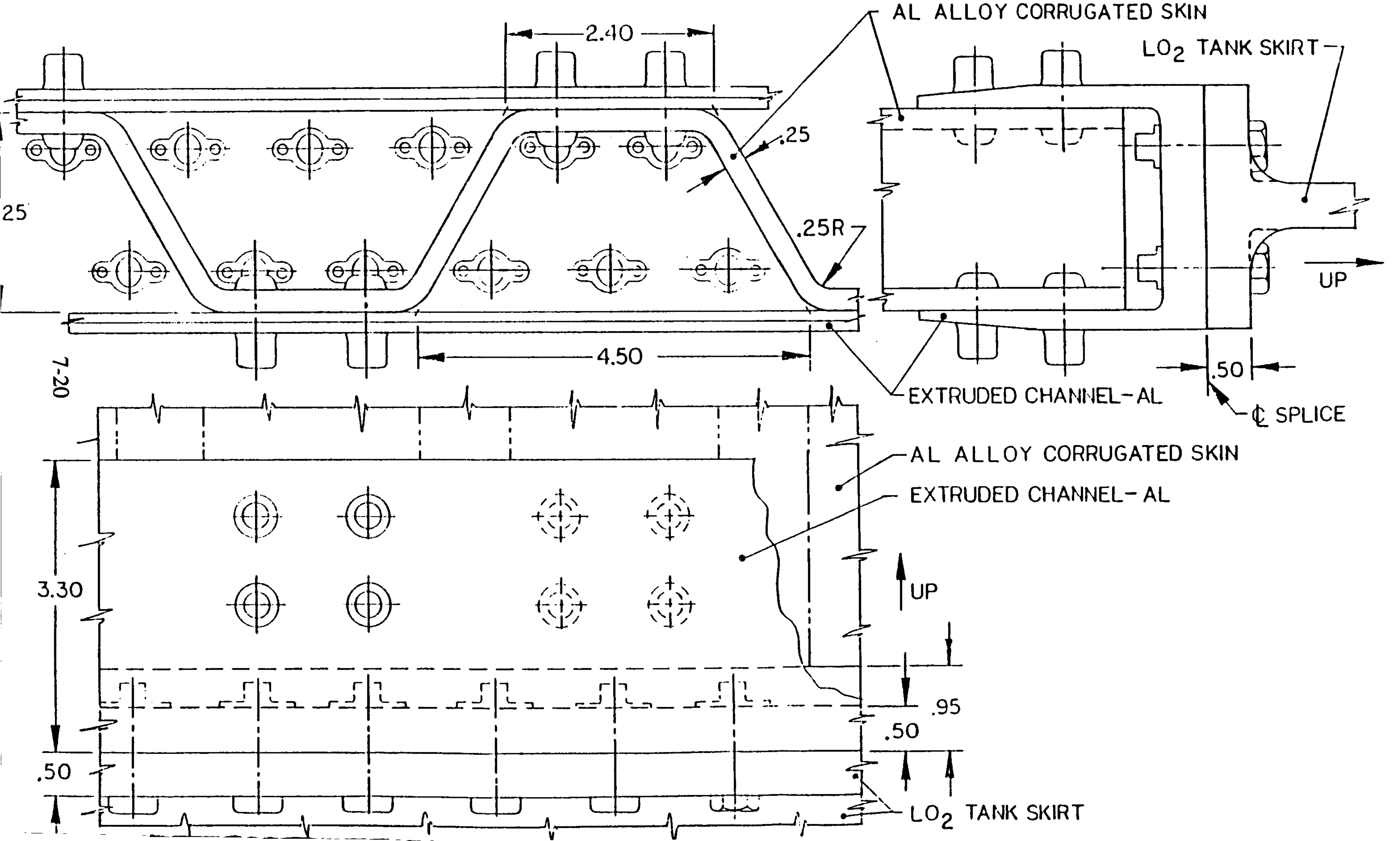


Figure 7.1.1-11 Intertank Adapter Structural Arragnement

LO$_2$/RP-1 PRESSURE FED
INTERTANK ADAPTOR ASSEMBLY
ADAPTOR TO TANK SPLICE



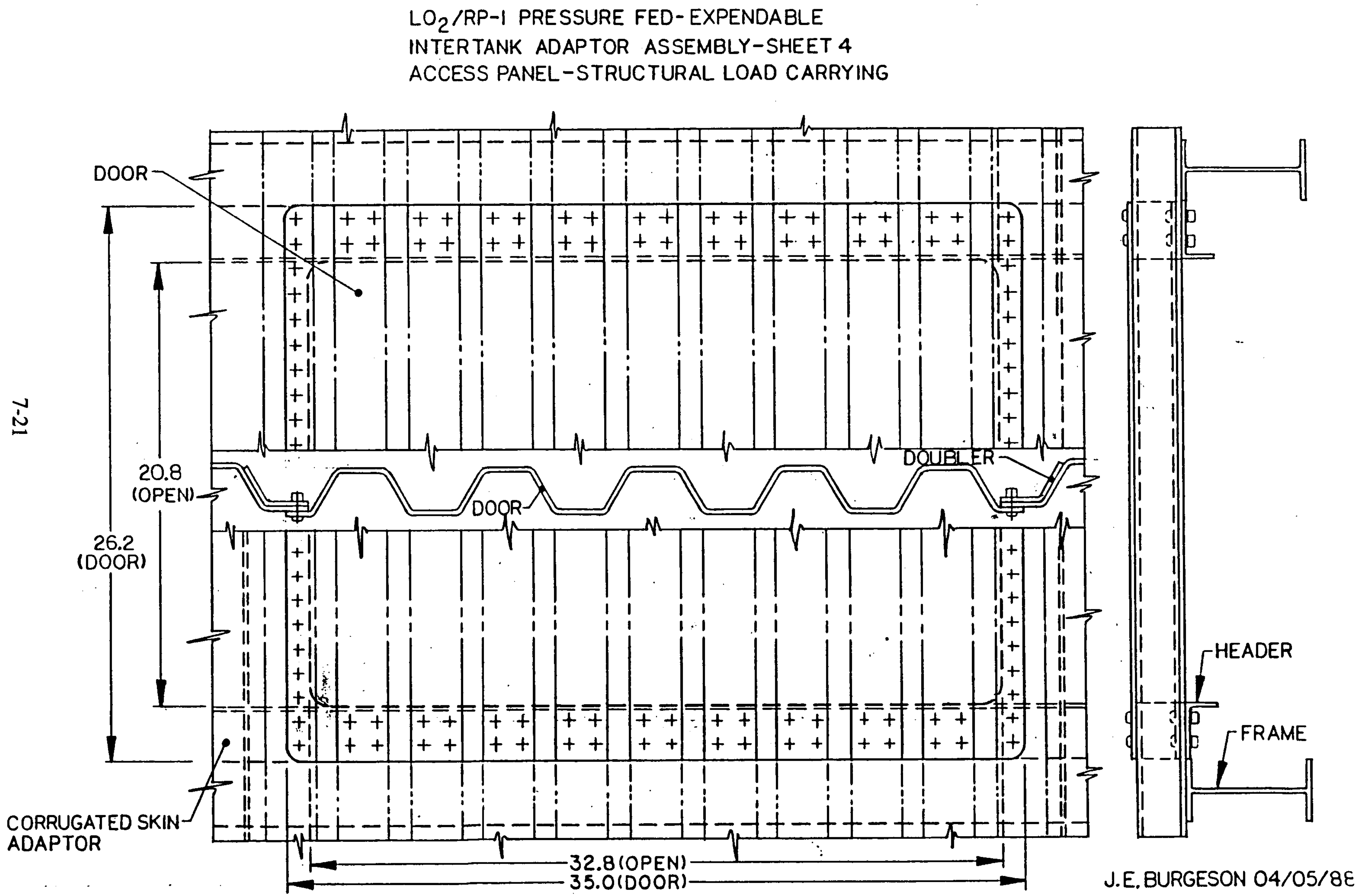


Figure 7.1.1-13 LO2/RP-1 Pressure Fed-Expendable Intertank Adaptor Assembly-Sheet 4 Access Panel - Structural Load Carrying

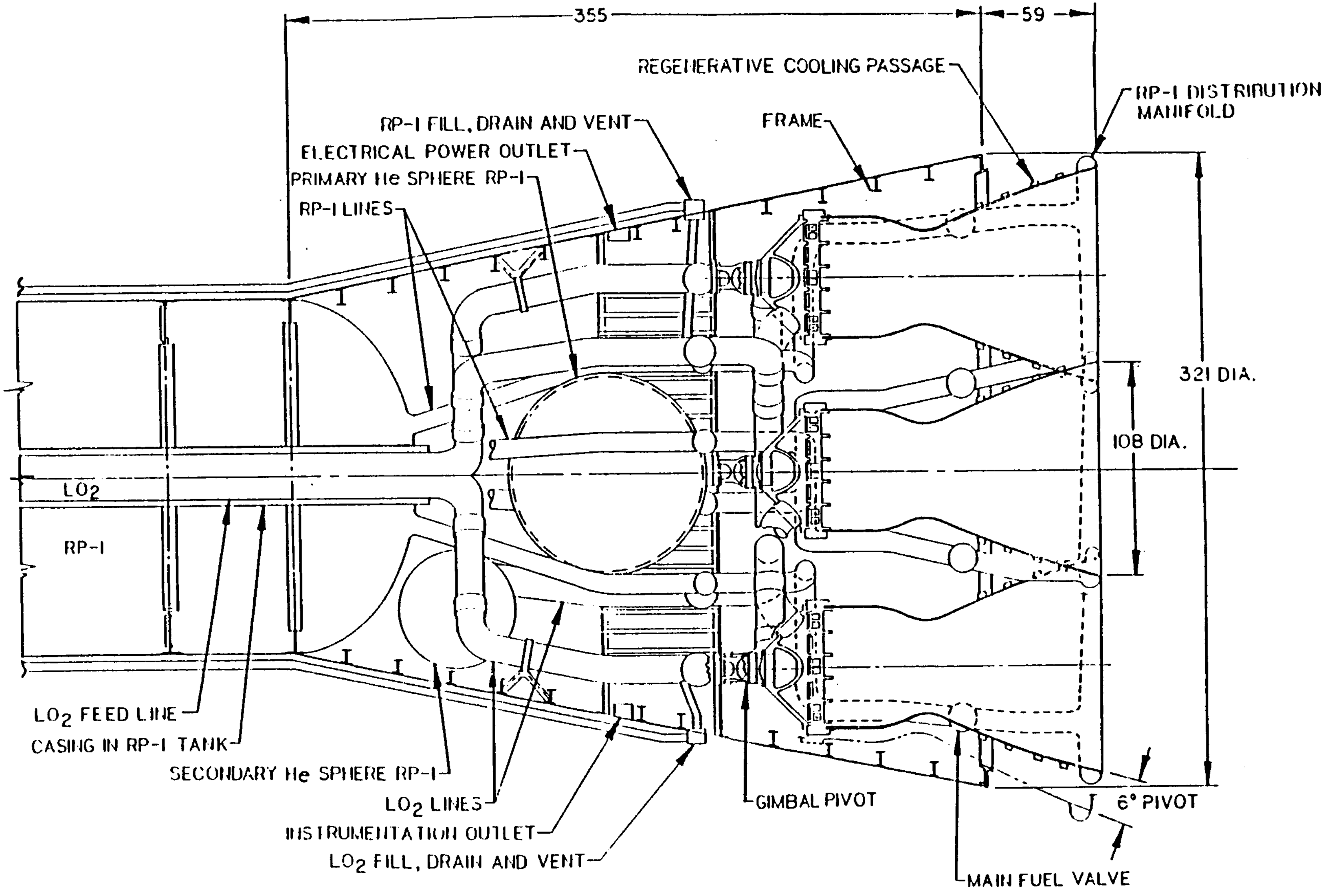


Figure 7.1.1.14 Aft Skirt Structure and Engine Installation

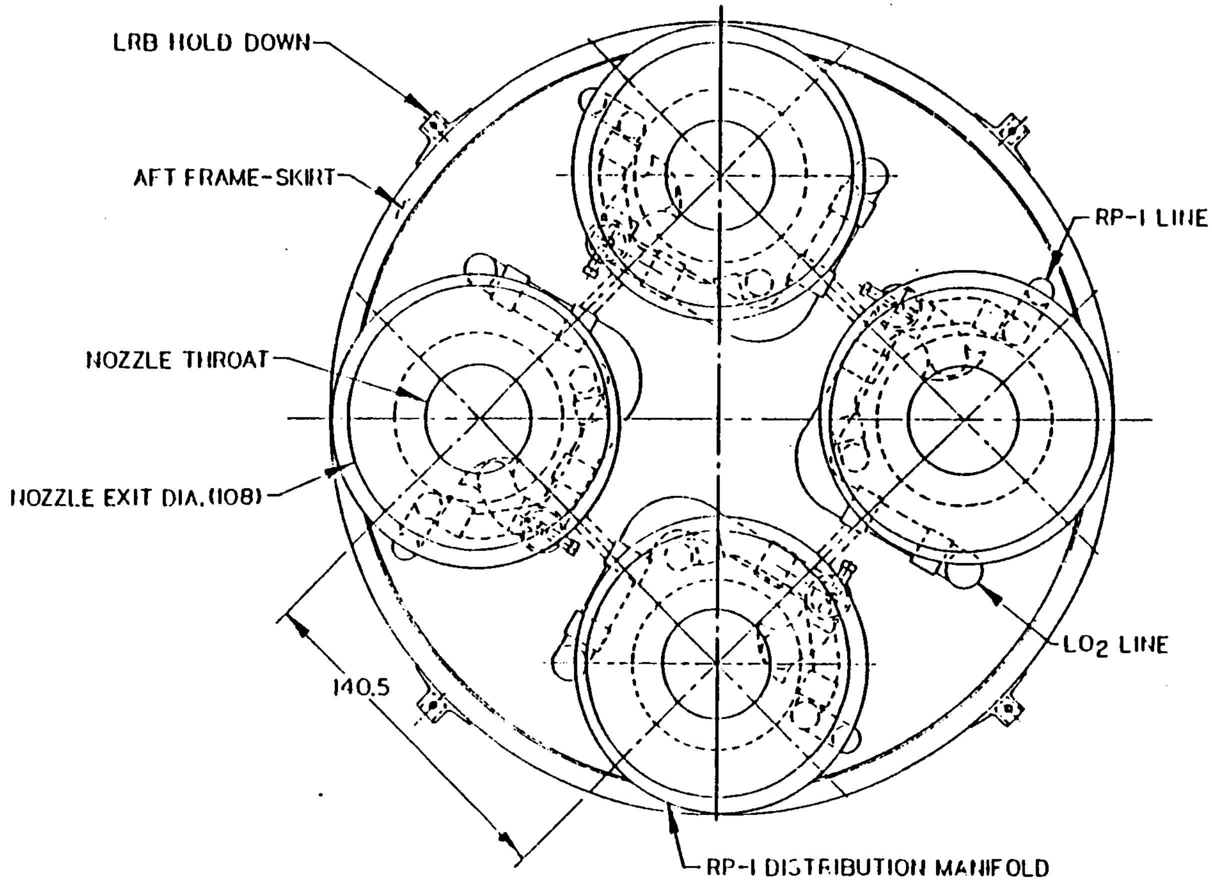
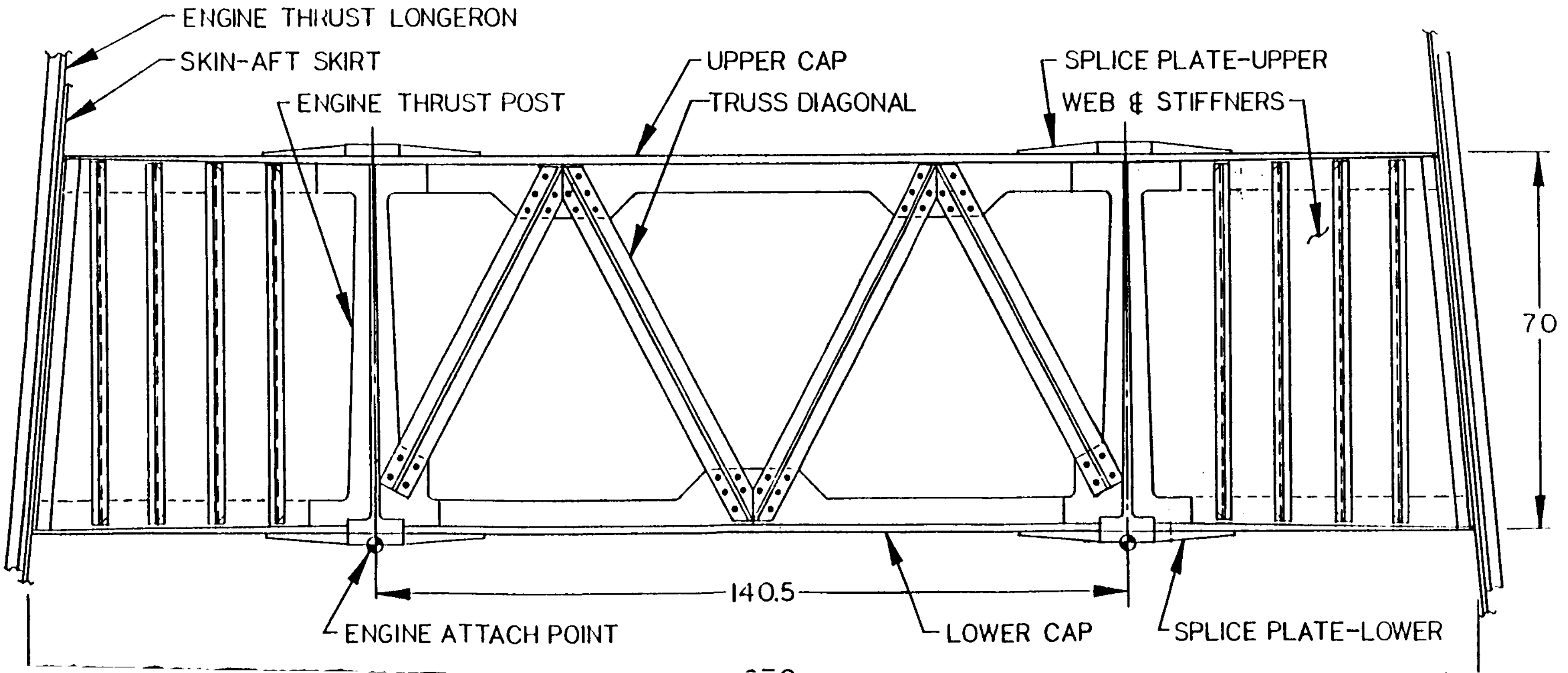


Figure 7.1.1-15 Aft Skirt Structure and Engine Installation-Rear View

PRESSURE FED LRB
ENGINE THRUST STRUCTURE



(LRB to MLP interface) and the relationship of the LRB to the exhaust system on the MLP and launch pad.

Aft Struts. Figure 7.1.1-17 shows the redesigned aft strut configuration for the LRB. The LRB 15 foot diameter will require the aft struts currently being used to decrease in length 4.75 inches. The top and lower struts will decrease to 30.98 inches from 35.73 inches, while the diagonal strut will remain consistent with current SRB design. The diagonal strut can be rotated to adjust to the increase in diameter of the LRB. The external tank attach fitting location for the aft struts will not be changed, therefore, the only required change in the current aft strut design will be the change in length of the upper and lower strut.

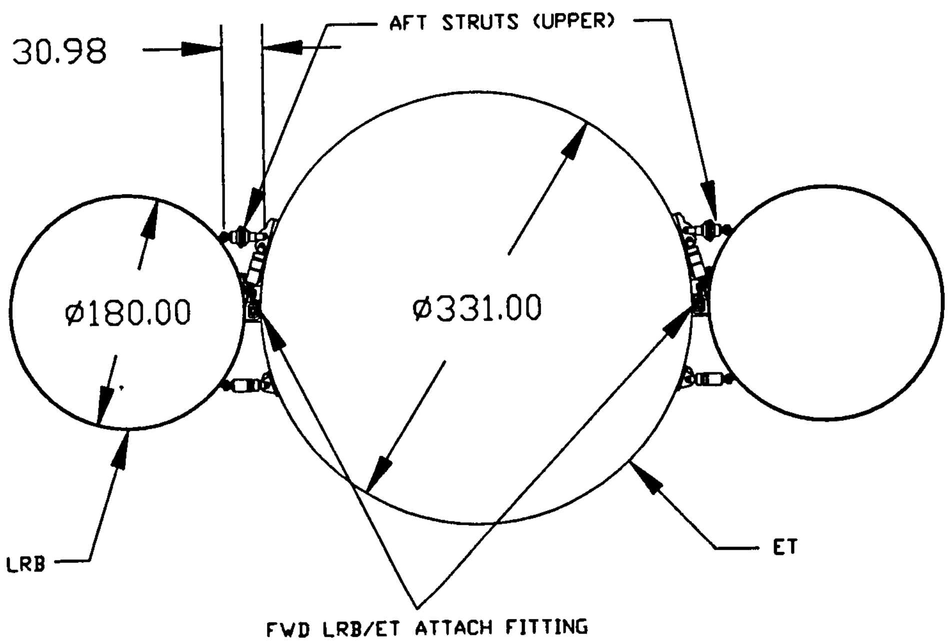


Figure 7.1.1-17. LRB Aft Strut Configuration

Forward LRB/ET Attach Fitting. The location of the LRB will be such that the current requirement of 12 inches between the ET OSL and the LRB OSL in an unloaded condition will be consistent with the current SRB design. At this point the current SRD attach fitting location is appropriate and any deviation in location will result from further loads analysis.

LRB Support and Release System. The LRB hold down support structure will be consistent with the SRB design except for the addition of a controlled release mechanism and any deviations due to load increases which may result from further analysis. The LRB will be supported by a 4 ft. by 6 ft. support and secured by explosive bolts (see Figure 7.1.1-18). The explosive bolts will secure the vehicle until a pre-set thrust has been achieved, and if necessary, withstand the rebound resulting from a launch abort.

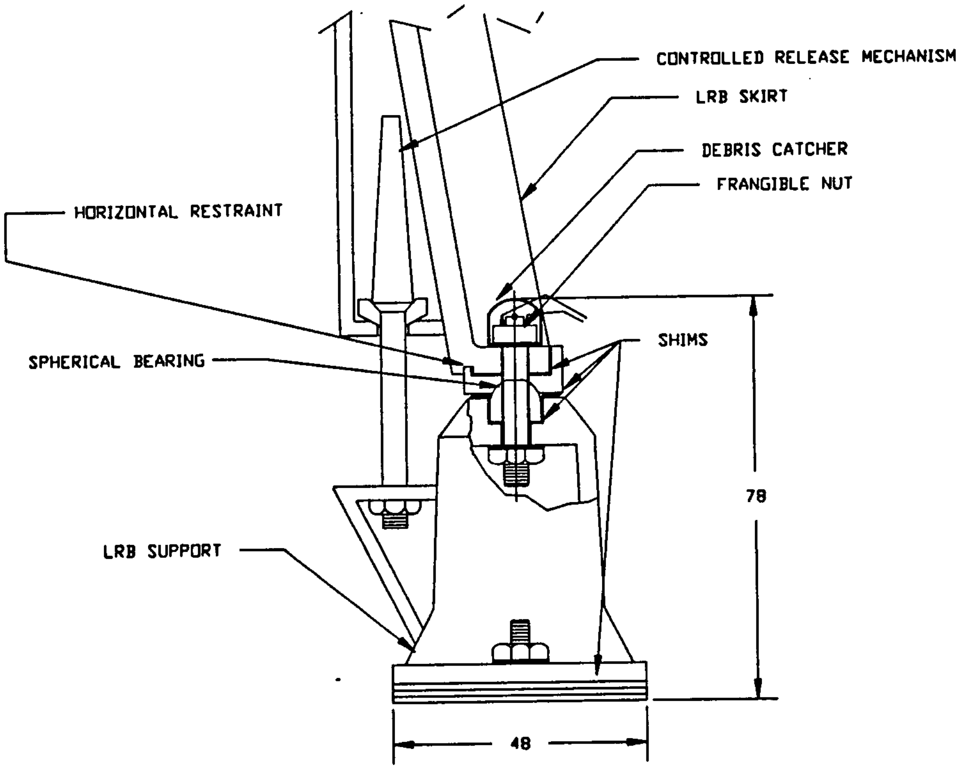


Figure 7.1.1-18 LRB Hold down and Release System

The controlled release mechanism consists of a die and cold drawn aluminum rod attached to the LRB support frame and collocated with the explosive bolts. At release, the bolt is blown and the die will shear the aluminum rod in the direction of decreasing cross-sectional area, permitting a smooth vertical release.

<u>LRB Stacking Interface</u>. The LRB will be stacked in the 90 deg. position (see Figure 7.1.1-19) as shown. This stacking approach will prove beneficial in the event of forward drift. With the 90 deg placement, a 29 inch spacing between the hold down post and the engines will be provided, with an 18 inch space between the engines and the well wall for lateral motion.

The current SRB well diameter will be increased to 342.6 inches to provide for the increased LRB diameter. The LRB supports will be placed as shown in the three (3) engine view. A 345.7 inch flame deflector will be added to guide the LRB flame into the exhaust trench. The true view of the flame deflector is shown in the two engine view.

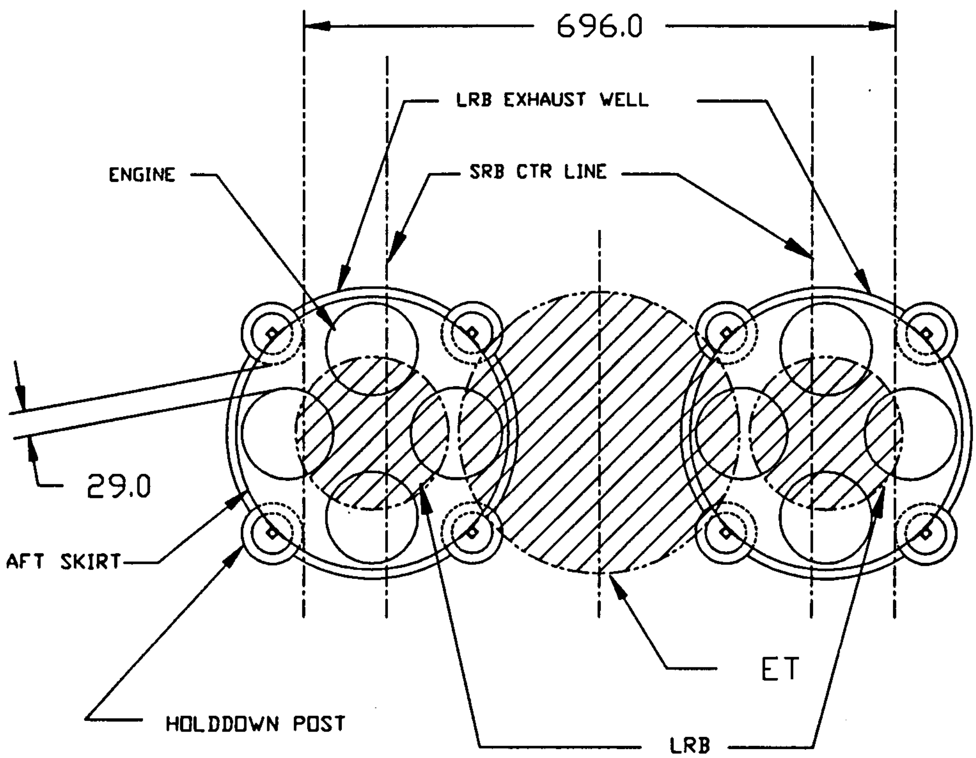


Figure 7.1.1-19 LRB Aft Skirt Configuration

7.1.2 <u>SEPARATION SYSTEM</u> Efforts to define the separation system were conducted during the initial LRB study phase. During the follow-on extension, separation analyses were not updated to reflect resizing of the LOX/RP-1 pressure-fed booster. Thus, the results which follow require update and are presented primarily to show trends and typical designs.

The separation system definition for the RP-1/LOX pressure-fed booster is largely the same as that designed for the LOX/RP-1 pump-fed booster (see section 4.1.2). However, the pressure-fed booster is twice as heavy as the LOX/RP-1 pump-fed booster at separation. Also, because the pressure-fed booster is much larger aerodynamic forces are increased at separation. This leads to a less "clean" separation, and an increase in the number of BSMs required compared to the LOX/RP-1 pump-fed configuration.

The LOX/RP-1 pressure-fed booster separation system has been initially sized for nominal ascent staging. As the LOX/RP-1 pressure-fed booster design matures, RTLS abort coverage capabilities will be more thoroughly examined, if results show that it is possible to conduct a RTLS abort prior to nominal staging, simulations will be conducted to determine if the BSM quantity used for nominal ascent staging is sufficient for RTLS abort needs. If not, the number of BSMs will be increased accordingly.

Nominal ascent staging of the LOX/RP-1 pressure-fed booster is designed to occur at initial conditions of:

Mission Elapsed Time	= 119.5 Seconds
Altitude	= 141,600 Ft
Mach	= 4.87
Dynamic Pressure	= 66 PSF
Inert Weight	= 2500,000 Lbs

The LOX/RP-1 pressure-fed booster separation system design uses 10 BSMs. This is based on computer simulation results which indicate that a placement of 5 BSMs forward and 5 aft will produce safe separation for nominal ascent (design case) staging conditions. Design case staging conditions include: body rates of 5 deg/sec pitch, 2 deg/sec yaw, and 2 deg/sec roll; alpha = 10 degrees; and beta = 10 degrees. The corresponding booster separation system weight is on the order of 2,000 lbs.

The 10 BSMs used are distributed with 5 packaged in the nose cone, and 5 placed on the aft skirt. The same BSM orientation for the LOX/RP-1 pump-fed booster is also used for the LOX/RP-1 pressure-fed configuration.

Separation plots for nominal ascent (design case) staging (Figures 7.1.2-1 through 7.1.2-3) indicate a clean separation. Because the pressure-fed booster is longer than a SRB (which places forward separation motors ahead of where they are normally) plume impingement on the Orbiter TPS is reduced. Thus, it may be possible to reorient the forward BSMs so that they fire more laterally, which makes separation more efficient. If the forward BSMs are redirected, there is the possibility fewer would be needed for nominal separation.

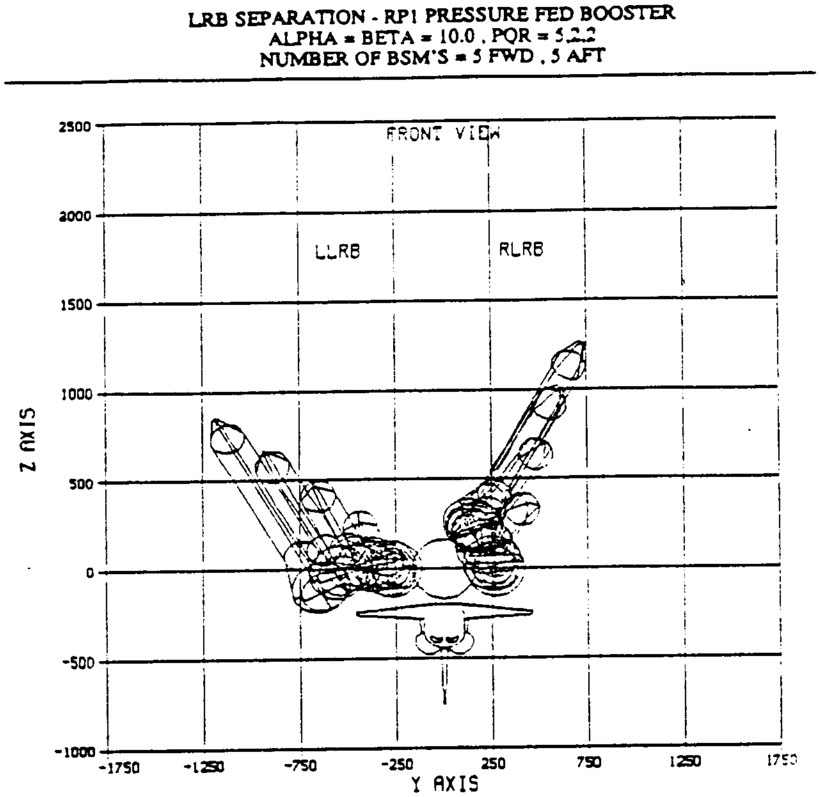


Figure 7.1.2-1. LOX/RP-1 Pressure-fed Nominal Ascent (Design Case) Separation, Front

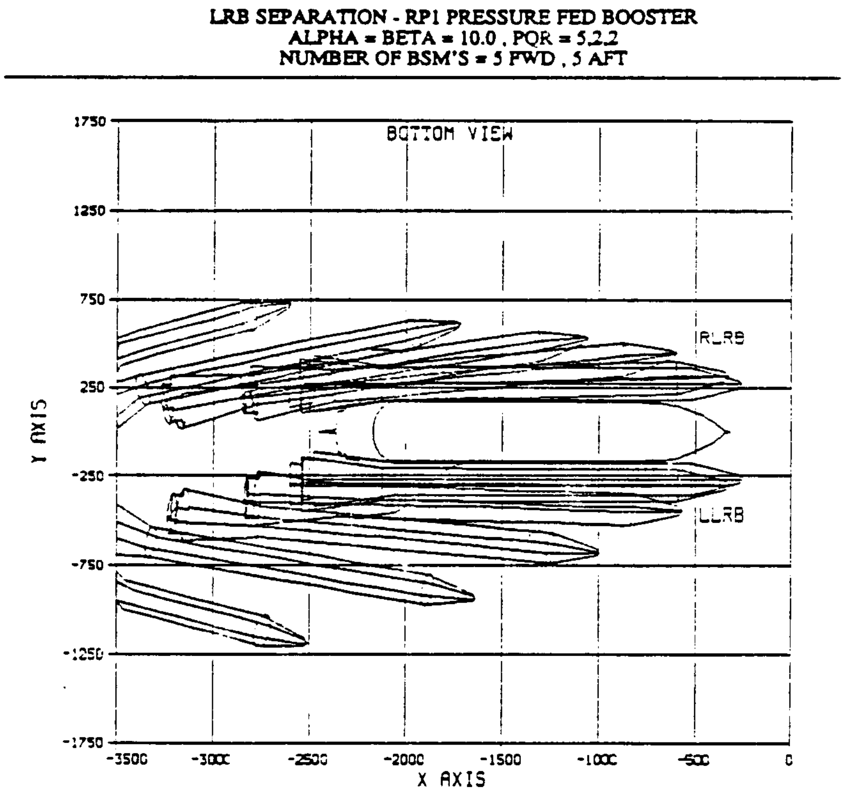


Figure 7.1.2-2. LOX/RP-1 Pressure-fed Nominal Ascent (Design Case) Separation, Bottom View

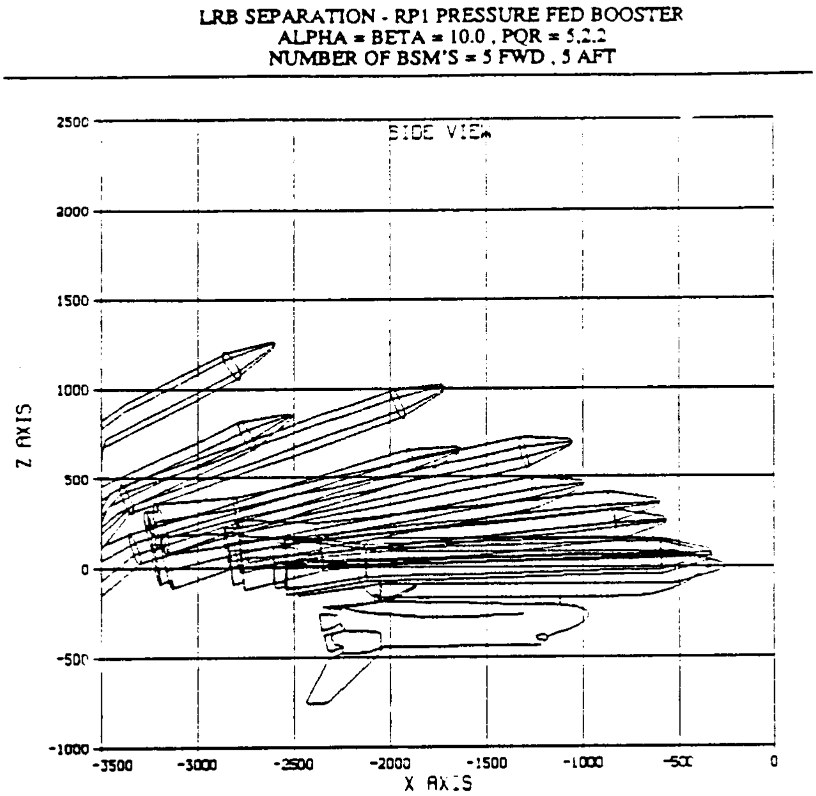


Figure 7.1.2-3. LOX/RP-1 Pressure-fed Nominal Ascent (Design Case) Separation, Side View

7.1.3 THERMAL PROTECTION SYSTEM. Because the pressure-fed booster burns the same propellants as the LOX/RP-1 pump-fed LRB, the overall thermal protection system is similar (see section 4.1.3).

Aerodynamic Heating. An examination of the LOX/RP-1 pressure-fed booster's 'Altitude vs.Velocity' ascent profile indicates that the booster will experience generally lower heating than a SRB because the ascent is more lofted, thus lower air density is encountered during high velocity portions of flight (examine Figure 7.1.3-1 below, and refer to section 4.1.3.1).

However, because the LOX/RP-1 pressure-fed booster's nose cone tip is approximately 15 feet ahead of the ET, it does not fall within the ET bow shock. Thus, the LOX/RP-1 pressure-fed booster nose cone is exposed to free-stream conditions, and the booster's bow shock will create localized stagnation point heating on its nose cone region which is higher than a SRB's. In addition, the ET's bow shock will strike the pressure-fed booster on its forward LOX tank. Depending upon the shock structure and flow conditions, this will produce high localized heating to portions of the LOX tank sidewall.

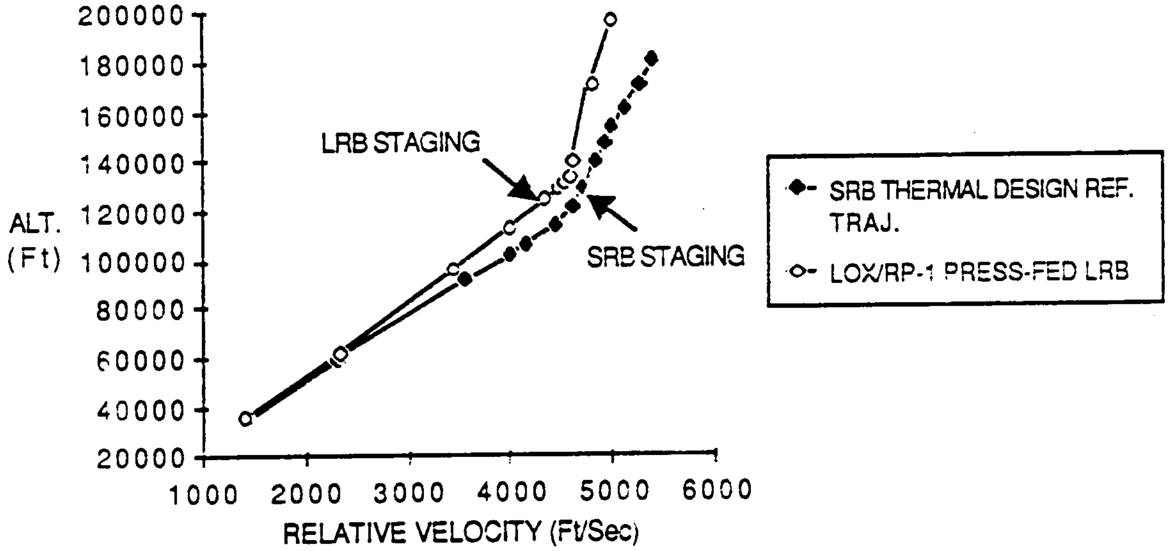


Figure 7.1.3-1. LOX/RP-1 Pressure-fed Booster And SRB Altitude Vs. Velocity Profiles Comparison.

Thermal Protection System Definition. The thermal protection system design for the LOX/RP-1 pressure-fed booster is similar to the LOX/RP-1 pump-fed booster (see section 4.1.3.2). SOFI (CPR-488) is applied to barrel sections of the cryogenic oxidizer tank in nominally 1" thickness, while urethane foam BX-250 covers tank bulkheads. Ablative materials are applied to high heating areas of the booster such as the nose cone, aft skirt, and protuberancies. Flexible skirts and a heat shield protect the booster engines and aft skirt components from plume heating. See Figure 7.1.3-2 below.

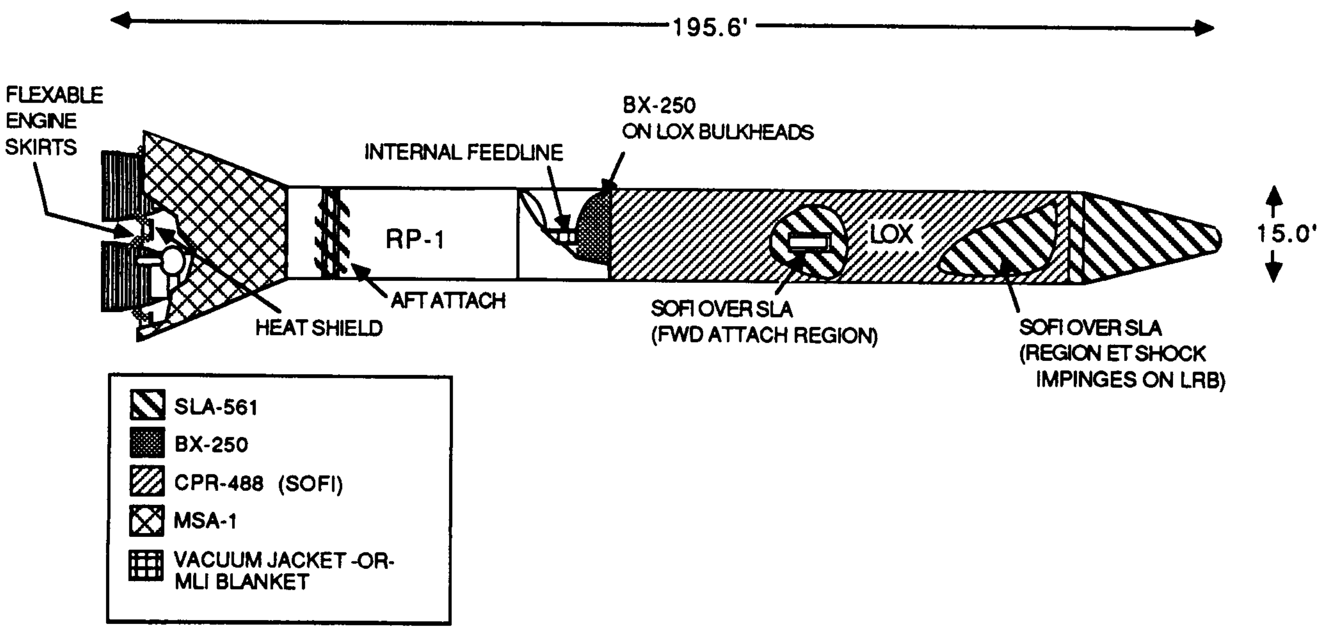


Figure 7.1.3-2. LOX/RP-1 Pressure-fed Booster TPS Layout

Because the pressure-fed booster will experience high heating due to flow stagnation, the ablative material used will be SLA-561. SLA-561 has a higher heat rate capacity than the MSA-1 which used on the SRB nose cone regions. In addition, because the bow shock off the ET will strike the booster LOX tank sidewall, creating a local heating increase, a patch of SLA-561 underlies SOFI on the upper region of the LOX tank; further analysis is required to verify that the SOFI/SLA combination can withstand high shear forces from the impinging ET bow shock.

The LOX/RP-1 booster employs an internal feedline to supply the cryogenic oxidizer to booster engines. This feedline passes through the booster intertank and RP-1 fuel tank. Depending upon the outcome of more detailed analyses, this line will be surrounded by either a vacuum jacket or a Multi-Layer Insulation (MLI) blanket system.

7.2 MAIN PROPULSION SYSTEM

The main LRB propulsion system consists of: four (4) engines burning RP-1 fuel with LOX as the oxidizer (generating 841,482 lbs sea level thrust each), propellant feed lines, propellant tanks, and a Tridyne pressurization subsystem. Current state-of-art technology is employed throughout; however, technology demonstration programs are required to demonstrate the propulsion system at the required thrust levels and sizes. The propellant tanks, the design of which is discussed in detail in Section 7.1, are fabricated from type 2219 aluminum using thicknesses compatible with proven welding techniques. The engine design is consistent with the established technology data base for fully regenerative cooled units, the type with by far the greatest number of applications. The feed line subsystem employs piping and expansion provisions in keeping with technology developed during the past thirty years of space vehicle and commercial applications. The Tridyne pressurization system is a tested, but not applied, system being explored competitively by TRW and Rocketdyne to reduce pressurant subsystem size and weight. Extensive testing has been accomplished with smaller scale catalytic chambers. The major technical problems remaining are those concerning upscaling to the flow levels involved with full scale, pressure-fed propellant, vehicles.

7.2.1 PROPULSION SYSTEM CONFIGURATION. The general configuration of the LO2/RP-1 pressure-fed booster is depicted in Figure 7.1-1. The LOX tank pressurization subsystem is located in the nose cone. Immediately below the nose cone, the cylindrical section of the LOX tank starts, with the forward elliptical dome protruding into the nose cone. Venting and pressure relief valves and connections to the LOX tank are mounted above the tank's upper end dome.

The intertank section below the LOX tank contains the vent and pressure relief valves and piping for the RP-1 tank. Below the RP-1 tank is the aft skirt containing the fuel tank pressurant subsystem, propellant tank, all flex joints including those for thermal expansion/contraction, the propellant piping to distribute propellant to all engines with minimal pressure differentials, and the engines themselves. Ground connections for vents, pressurization gas, propellant fill and drain are also located in/on the surface of the aft skirt.

Other significant features affecting propulsion which are located in the aft skirt include the gimbal mounts for TVC, the TVC actuators and power distribution systems, the launch release mechanisms to reduce/control launch release loads, etc.(Refer to Figure 7.1.1-19).

Engine System. The engine shown in Figures 7.2.1-1 and 7.2.1-2, is a full regenerative unit, designed from a data base dating directly back to the Navaho booster, and to the current Atlas and Delta boosters. Throttling however, is an additional requirement for pressure-fed boosters and has been demonstrated for higher pressure engines than the design now under consideration. Conservative analytical application of the current technology indicates the selected design to be within the range of feasible throttling regime and should be achievable at minimal cost. The characteristics of the engine selected are shown in Table 7.2.1-1 The nominal engine chamber pressure is 334 psia. This lower chamber pressure allows for reduced tank pressures resulting in a lighter weight and cheaper vehicle. The rational for selection of chamber pressure has been discussed in section 3.7.2. The Rocketdyne Engine Appendix 6, contains preliminary analysis on combustion stability for our pressure-fed engine. As shown in Figure 7.2.1-3, our chamber pressure of 300 psia falls well within the "safe-box", causing a stable engine. From the requirement for the capability to achieve ATO with one engine out at lift-off. This, combined with the structural limits of the ET, imposes the requirement for throttling to 60 percent of nominal thrust. Presently closed loop continuous throttling is baselined for this concept.

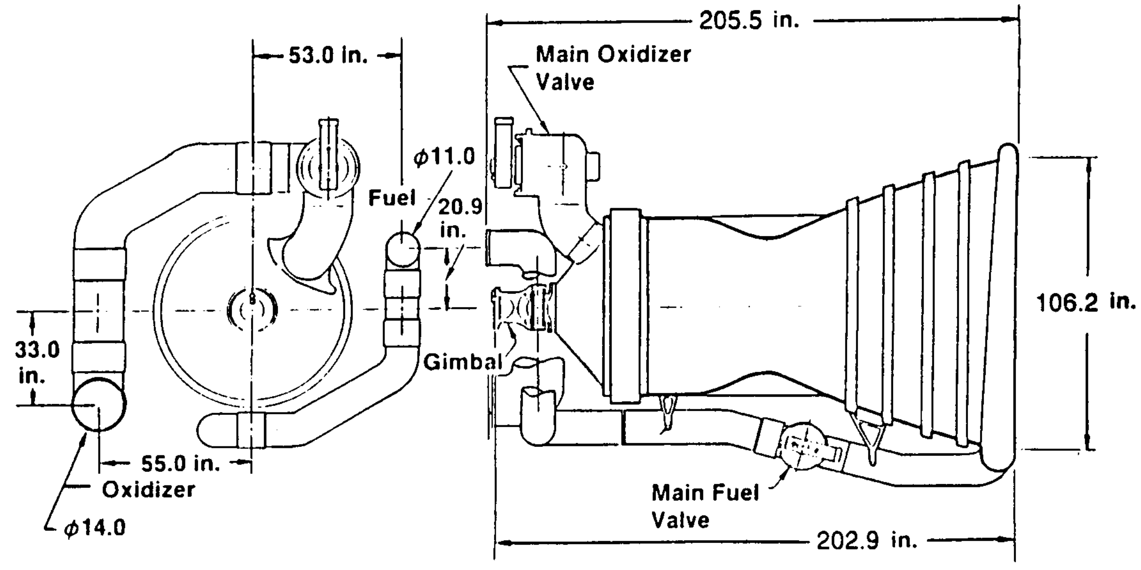


Figure 7.2.1-1 Pressure-fed Engine

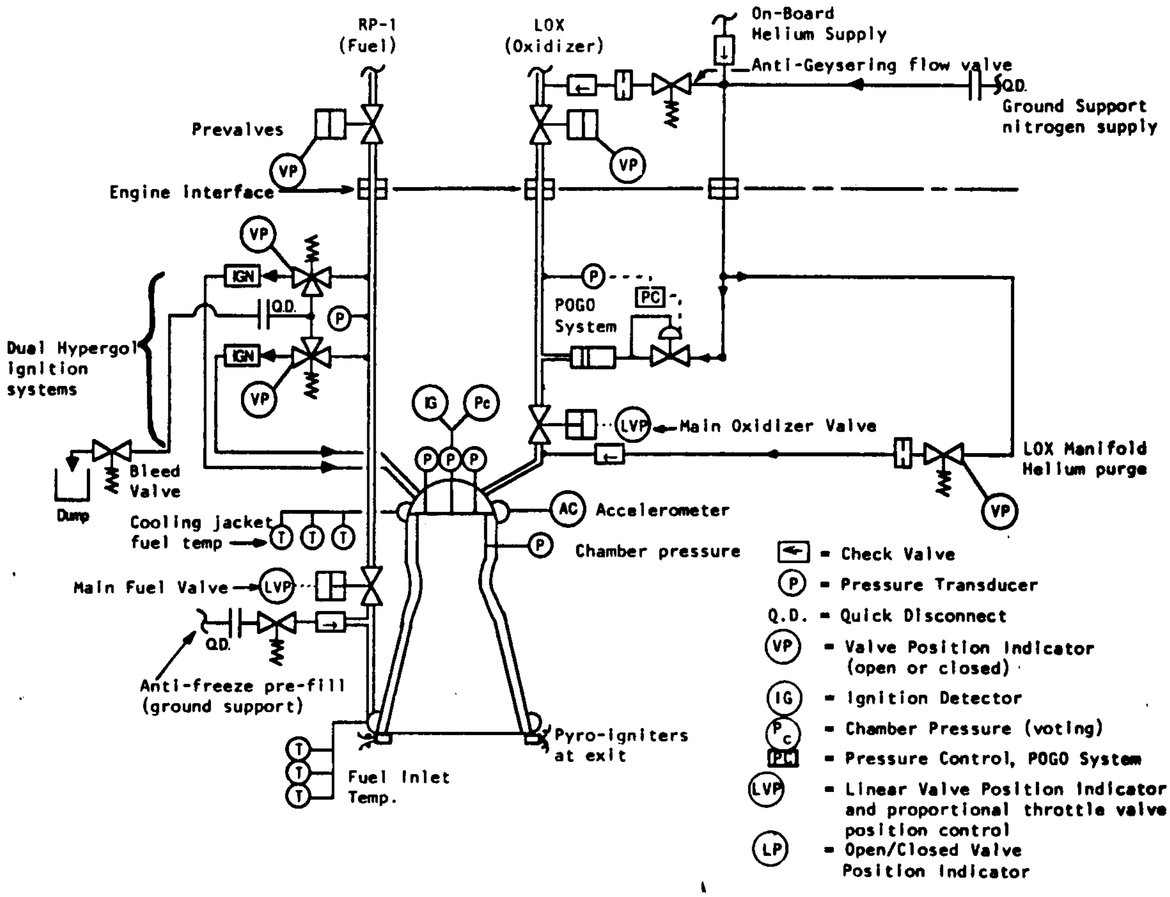


Figure 7.2.1-2 Engine Schematic

Table 7.2.1-1 LRB LOX-RP-1 Pressure Fed Engine Characteristics

ENGINE PARAMETERS	NOMINAL THRUST	MINIMUM THRUST
Weight (lb)	7017	
Throttle (percent)	100.0	60.0
Oxidizer flow rate (lb/sec)	2433.0	1469.3
Fuel flow rate (lb/sec)	973.2	587.7
Vacuum thrust (lb)	971595	582625
Sea level thrust (lb)	841482	452542
Chamber Pressure (psia)	334.0	200.4
Vacuum Isp (sec)	285.2	283.2
Sea level Isp (sec)	247.0	220.0
Mixture ratio	2.5	2.5
Nozzle area ratio	4.96	
Area (in2)	8854	
Throat radius (in)	23.84	
Exit diameter (in)	106.2	
Overall length (in)	205.5	

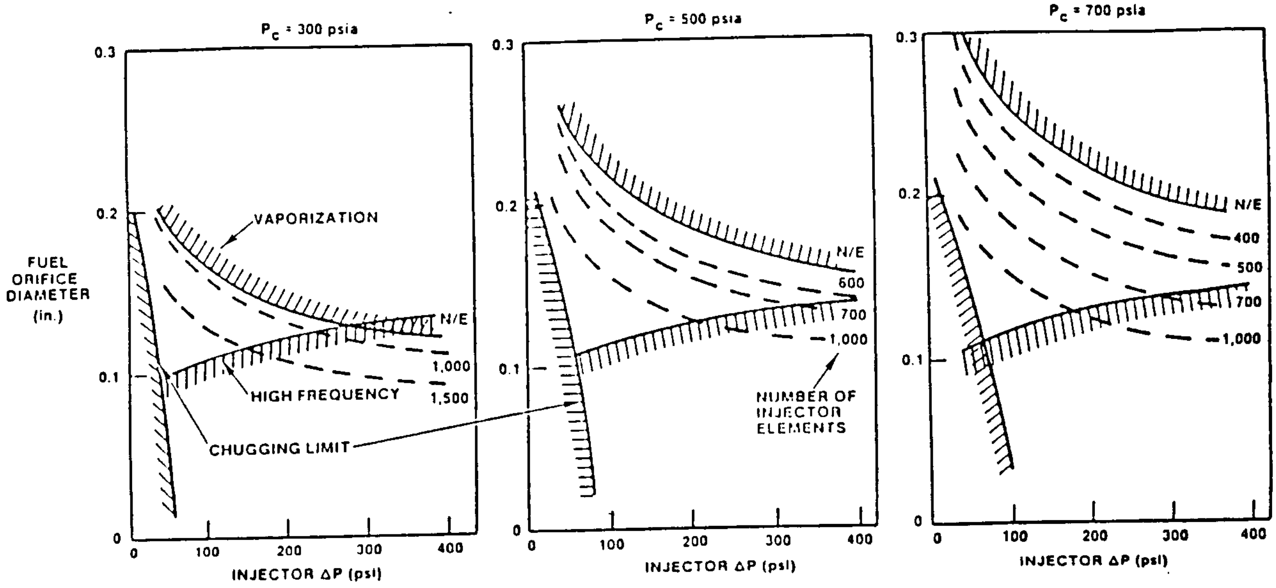


Figure 7.2.1-3 Combustion Stability Evaluation
Design ΔP = 334 psia
Design injector ΔP = 25%

Combustion Chamber/Nozzle Assembly. The combustion chamber/nozzle assembly is of the tubular wall regenerative type. The fuel is used as coolant and is admitted into the cooling passages at the nozzle exit end. It flows forward to the injector end where it is collected and injected into the combustion chamber. The energy imparted to the fuel is recovered by higher combustion temperature, thus losses occurring in other cooling schemes, like film or transpiration, are not incurred.

The current nozzle design is such that direct use can be made of construction materials used for, and techniques developed for the Navaho and Atlas.

Injector. The injector design selected is, conventional baffled like on like doublet having alternate fuel and oxidizer rings. The injector baffles and orifice placement and size are based on stability analyses at throttled conditions, in order to maintain a high performance level with sufficiently high stability margin.

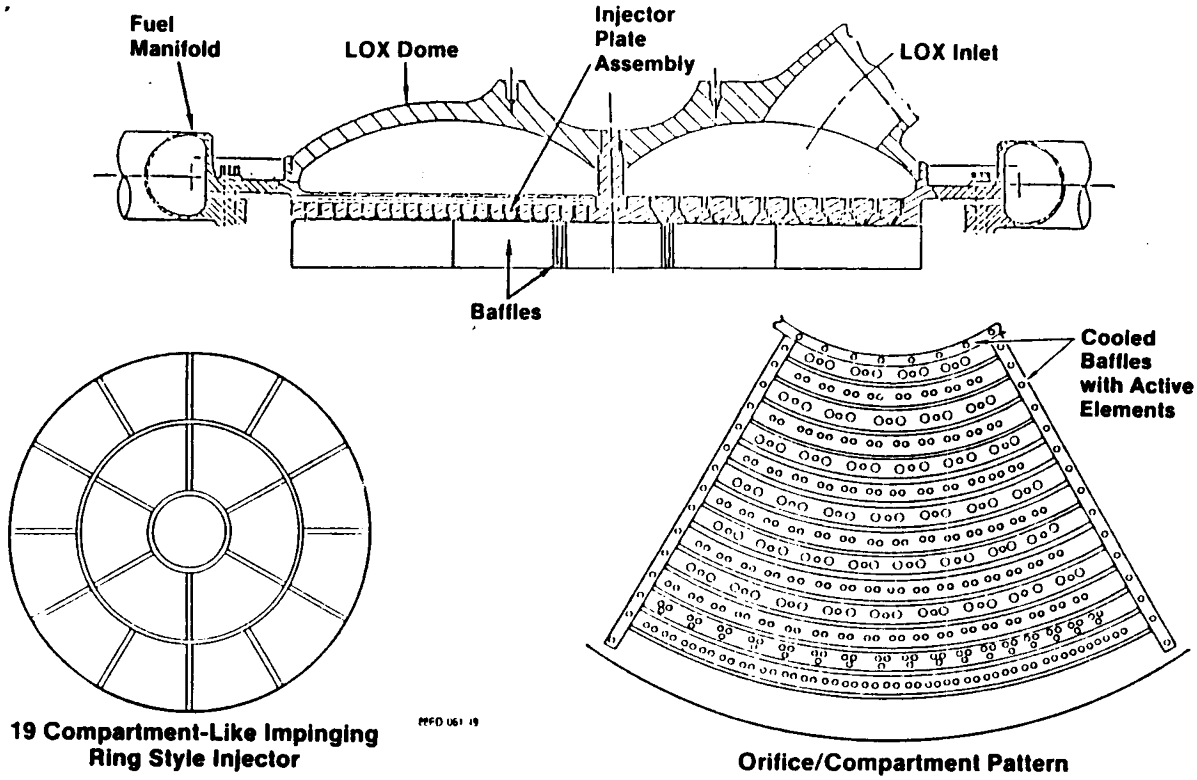


Figure 7.2.1-4 Injector

Gimbal. The head-end gimbal system proposed is again of the tried and proven design type. It is located above the injector on the engine centerline. High load capacity combined with low friction and extensive "proof of design" in flight applications make this a very attractive selection for safety and reliability reasons as well as lower cost. The gimbal is discussed as part of the TVC subsystem in Section 7.2.3 of this report.

Throttle Valve. The ball-type throttle valve (selected based on trade studies by Rocketdyne; Vol. II Appendix 6) is also of a design well proven in flight application.

Start and Shut-off Transients. The start and shutdown transient flow rates and chamber pressure build-up are shown in Figure 7.2.1-5a and 7.2.1-5b. These can be modified if detailed structural dynamics analysis in phase B indicate such a need.

7.2.2 PROPELLANT FEED SYSTEM.

Pressurant Subsystem. The pressurant subsystem is based upon the use of a helium - hydrogen-oxygen mixture which, in storage or in contact with the propellants used, is not flammable. However a exothermic reaction can be initiated and maintained in the presence of a catalyst. The nature of the appropriate catalytic materials are such (iridium and platinum with rhodium are the most active) that their presence as impurities in the construction materials used is highly unlikely, if not impossible. Thus unwanted catalytic impurities for

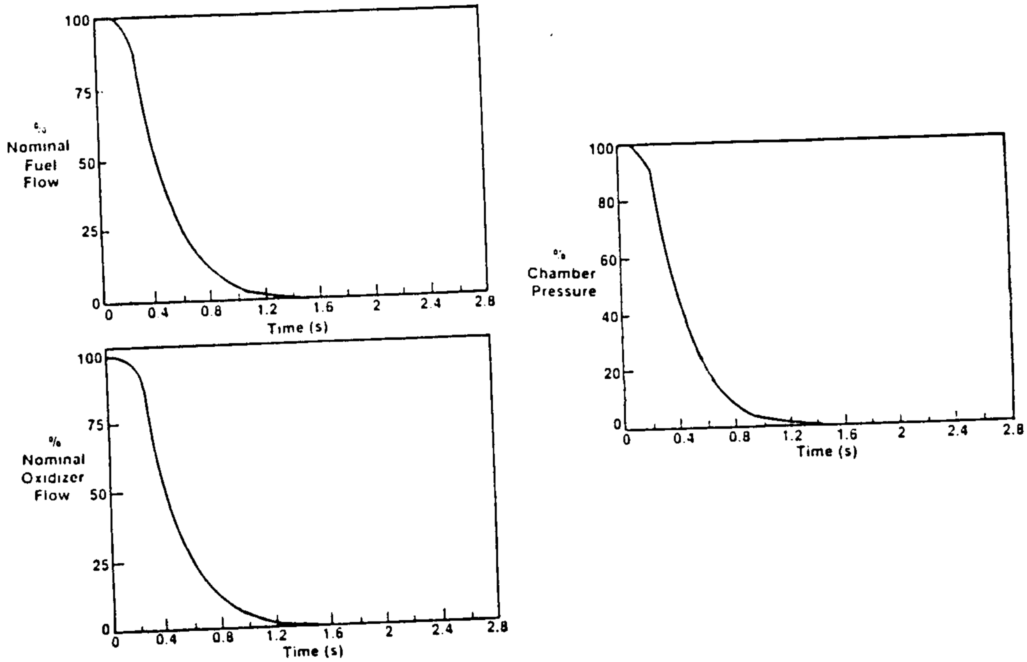


Figure 7.2.1-5a Pressure-fed LO2/RP-1 start transients

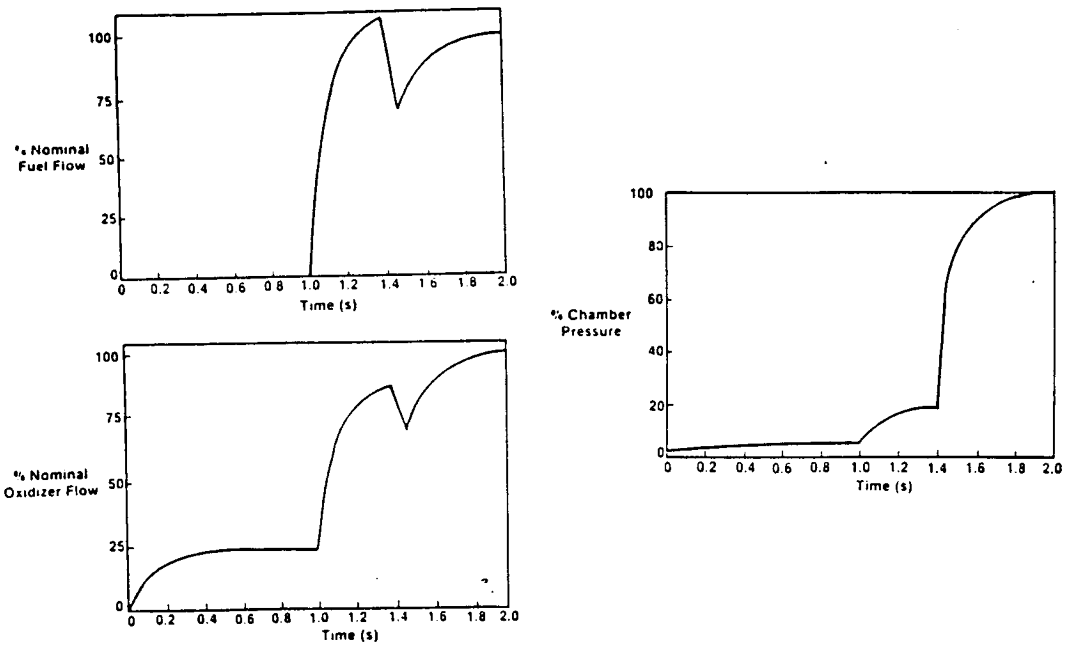


Figure 7.2.1-5b Pressure-Fed Engine Shut Down Transients

reaction are virtually impossible, and will be inspected for in all procurement and inspection procedures.

When catalyzed, the reaction produces sufficient heat to provide pressurant gas temperatures of 800 degrees Rankine (340 deg. F). This gas occupies more volume per pound than low temperature gas, providing a smaller, lighter system of the required capacity.

A dual cascade system, one subsystem for each propellant tank, is recommended. Each cascade consists of an ambient (secondary) storage sphere, a (secondary) catalytic reactor, a low temperature (primary) stage sphere, and a primary catalytic reactor with appropriate shutoff, check, servicing, and control valves as illustrated in figure 7.2.2-1. Initially a fixed bypass was considered for each secondary reactor. During optimization studies it was found that the ratio of sphere diameters together with pre-calculated initial hydrogen-oxygen concentration in primary and secondary spheres should provide accurate and simple temperature control.

To reduce system size and weight, the bulk of the pressurant is stored in a sphere maintained at 225 degrees Rankine. A second sphere at ambient temperature contains a similar gas mixture, and through its catalytic reactor provides energy to the primary sphere to increase its temperature 50 to 100 degrees during the mission (refer to Figure 7.2.2-1). In this manner condensation and solidification of the oxygen component is prevented, and proper system operation assured. The secondary sphere will not reach the 225 degree Rankine point as long as launch temperature of the sphere is maintained above 495 degrees Rankine (35 degrees F). The condensation temperature of oxygen is 164 degrees Rankine at normal atmospheric pressure.

Separation of the pressurant subsystem into two individual systems has two advantages. Each system can be operated in accordance with the different ullage pressure requirement of each propellant tank, and, with oversized catalytic chambers combined with dual, normally closed, cross connections which can be opened in the event of a malfunction, will provide at least enough pressurant for a safe abort for shuttle and crew. These cross connections are illustrated in Figure 7.2.2-1.

Predicted pressurization system weights and the sizes of the major components are presented in Tables 7.2.2-1 through 7.2.2-3. Predicted performance of the system

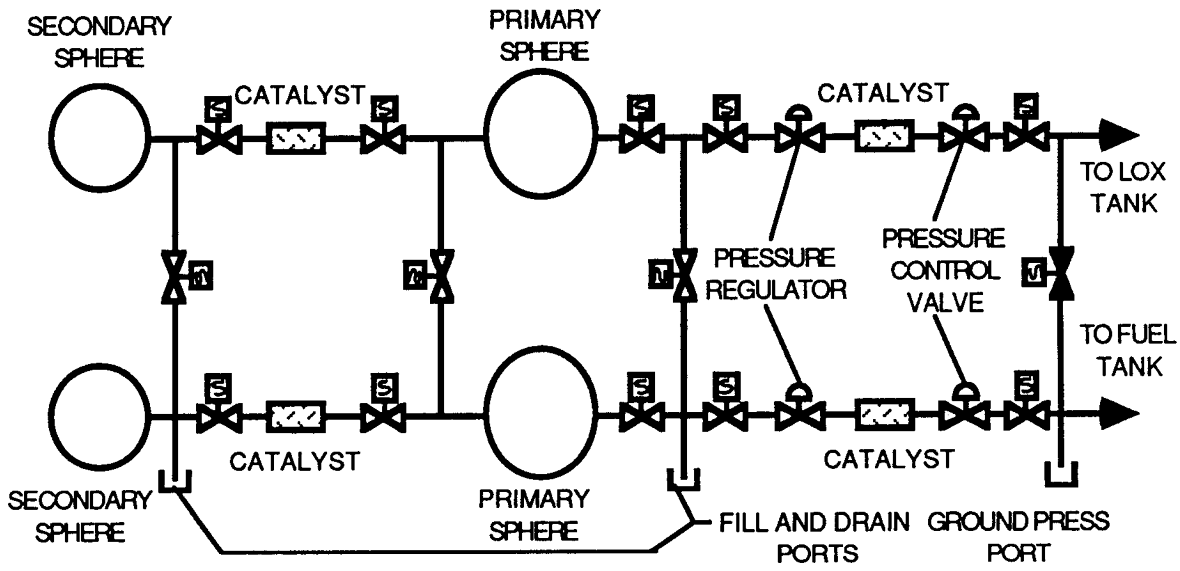


Figure 7.2.2-1 Pressurization System Schematic

combined with the tankage, feedline, and engine systems is presented in Figures 7.2.2-2 through 7.2.2-21. The predictions are based upon maintaining propellant tank pressures at levels required for a one LRB engine out ATO at all times.

Two sets (Nominal and ATO) of curves are provided to illustrate available margin in the design. These are the Engine Throttle Valve and Pressure Control Valve pressure drop curves, Figures 7.2.2-18 through 7.2.2-21. These valves are usually designed to have a maximum pressure loss, wide open, of 20 psia. Here the minimum pressure drops used for these valves is over three (3) times the value indicating a substantial operating margin.

Table 7.2.2-1 Sphere and Gas Weight Summary

LOX SUBSYSTEM		
PRIMARY SPHERE		
VOLUME	512 cu ft	
WEIGHT	8835 lbs	
HELIUM WEIGHT	2766 lbs	
TOTAL GAS WEIGHT	3129 lbs	
SECONDARY SPHERE		
VOLUME	182 cu ft	
WEIGHT	3137 lbs	
HELIUM WEIGHT	414 lbs	
TOTAL GAS WEIGHT	528 lbs	
TOTAL WEIGHTS		
SPHERES - DRY		11972 lbs
HELIUM		3179 lbs
TOTAL GAS		3657 lbs
FUEL SUBSYSTEM		
PRIMARY SPHERE		
VOLUME	308 cu ft	
WEIGHT	5315 lbs	
HELIUM WEIGHT	1663 lbs	
TOTAL GAS WEIGHT	1882 lbs	
SECONDARY SPHERE		
VOLUME	109 cu ft	
WEIGHT	1886 lbs	
HELIUM WEIGHT	249 lbs	
TOTAL GAS WEIGHT	318 lbs	
TOTAL WEIGHTS		
DRY		7202 lbs
HELIUM		1913 lbs
TOTAL GAS		2199 lbs

Table 7.2.2-2 Catalytic Reactor Design Estimates

OXYGEN SUBSYSTEM		
PRIMARY REACTOR		
BED VOLUME	0.726	cu ft
BED DIAMETER	12.737	in
BED LENGTH	9.852	in
CATALYST WEIGHT	50.9	lbs
CASING WEIGHT	67.5	lbs
TOTALWEIGHT	118.4	lbs
PRESSURE DROP	18	psi
SECONDARY REACTOR		
BED VOLUME	0.141	cu ft
BED DIAMETER	5.359	in
BED LENGTH	10.771	in
CATALYST WEIGHT	9.8	lbs
CASING WEIGHT	39.2	lbs
TOTAL WEIGHT	49.0	lbs
PRESSURE DROP	13	psi
FUEL SUBSYSTEM		
PRIMARY REACTOR		
BED VOLUME	0.441	cu ft
BED DIAMETER	10.125	in
BED LENGTH	9.465	in
CATALYST WEIGHT	30.9	lbs
CASING WEIGHT	33.4	lbs
TOTAL WEIGHT	64.3	lbs
SECONDARY REACTOR		
BED VOLUME	0.068	cu ft
BED DIAMETER	3.679	in
BED LENGTH	11.046	in
CATALYST WEIGHT	4.8	lbs
CASING WEIGHT	16.2	lbs
TOTAL WEIGHT	21.0	lbs
PRESSURE LOSS	14	psi

Table 7.2.2-3 Pressurization System Weight Summary

SPHERE DRY WEIGHTS	
LOX SUBSYSTEM	
PRIMARY	8835 lbs
SECONDARY	3137 lbs
TOTAL	11972 lbs
FUEL SUBSYSTEM	
PRIMARY	5315 lbs
SECONDARY	1886 lbs
TOTAL	7202 lbs
CATALYTIC REACTOR WEIGHTS	
LOX SUBSYSTEM	167.4 lbs
FUEL SUBSYSTEM	85.3 lbs
TOTAL	252.7 lbs
GAS WEIGHTS	
LOX SUBSYSTEM	
HELIUM	3180 lbs
O2 & H2	477 lbs
TOTAL	3657 lbs
FUEL SUBSYSTEM	
HELIUM	1913 lbs
O2 & H2	288 lbs
TOTAL	2201 lbs
PIPING, VALVES, FLANGES, SUPPORTS, ETC.	1313.3 lbs
SYSTEM TOTAL	26598 lbs

HELIUM FLOW RATE

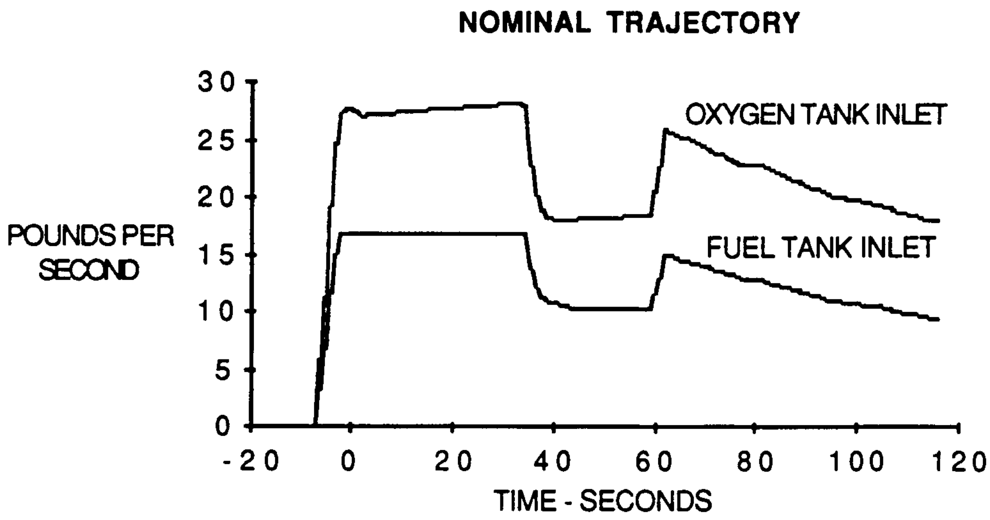


Figure 7.2.2-2

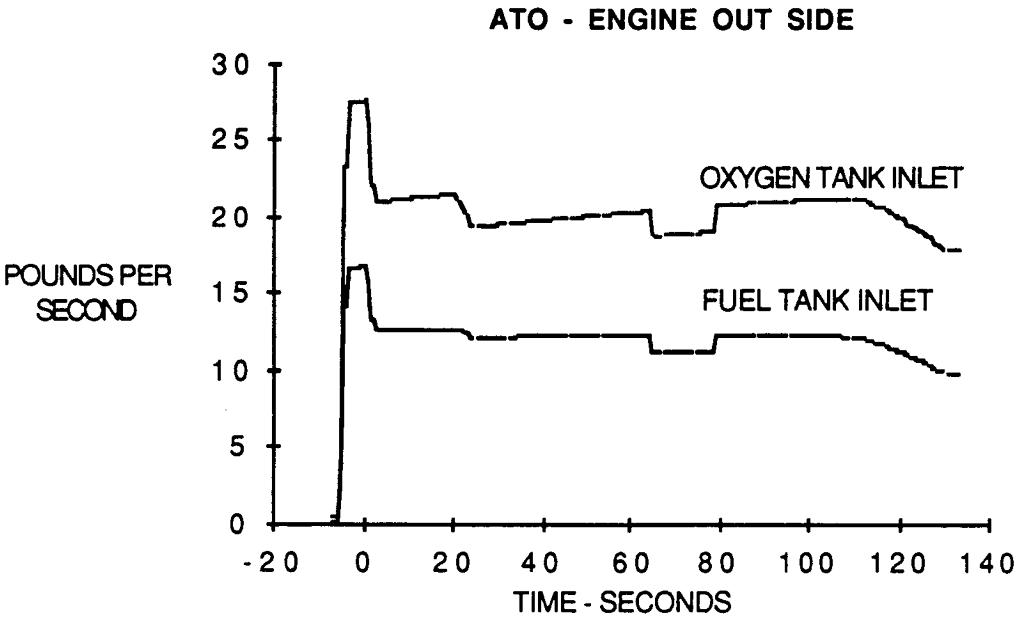


Figure 7.2.2-3

HELIUM FLOW RATE

SECONDARY SPHERE TO PRIMARY

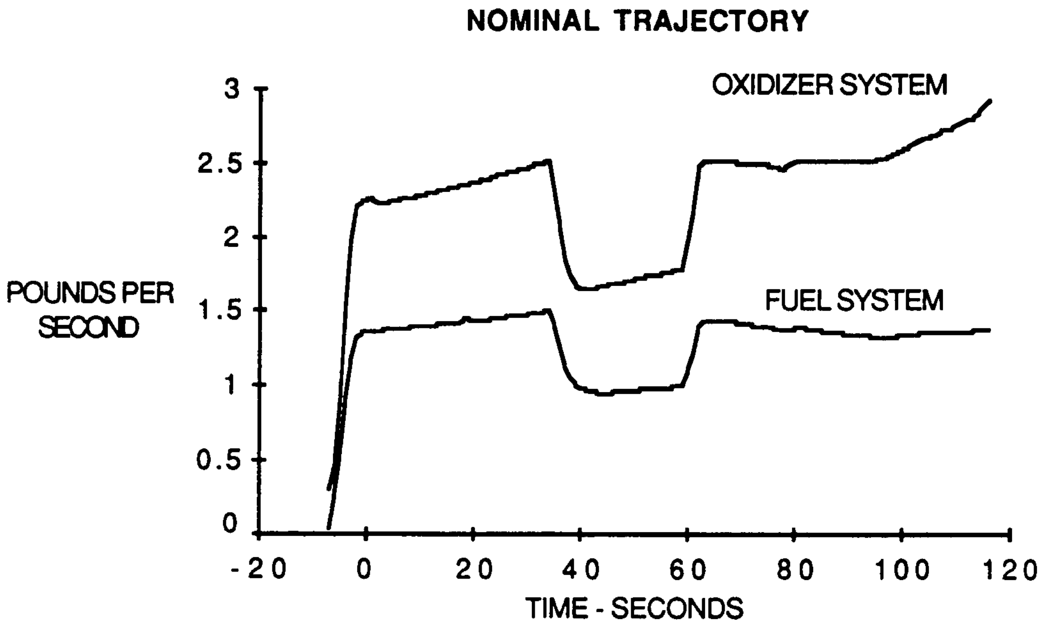


Figure 7.2.2-4

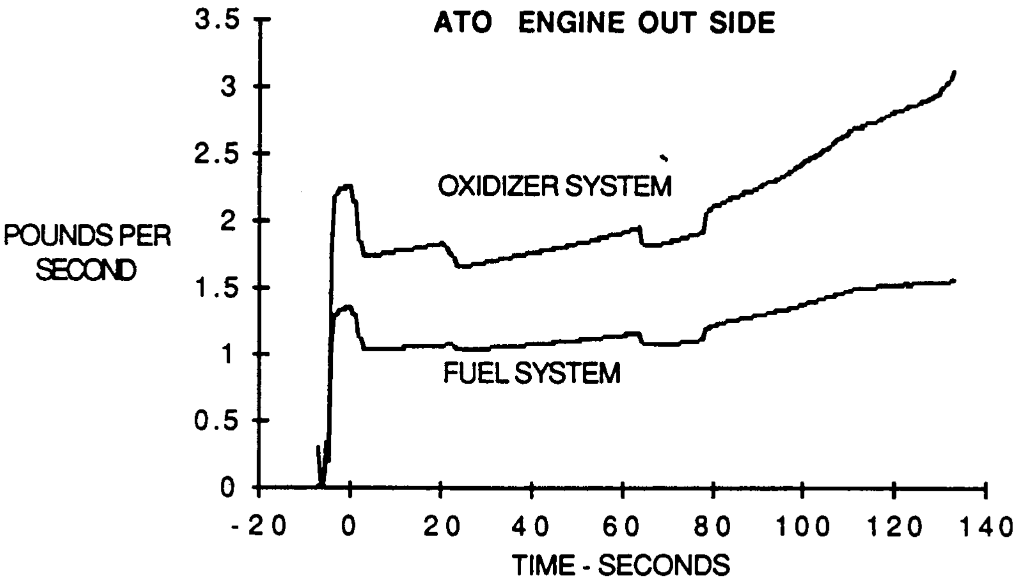


Figure 7.2.2-5

HELIUM in PROPELLANT TANKS

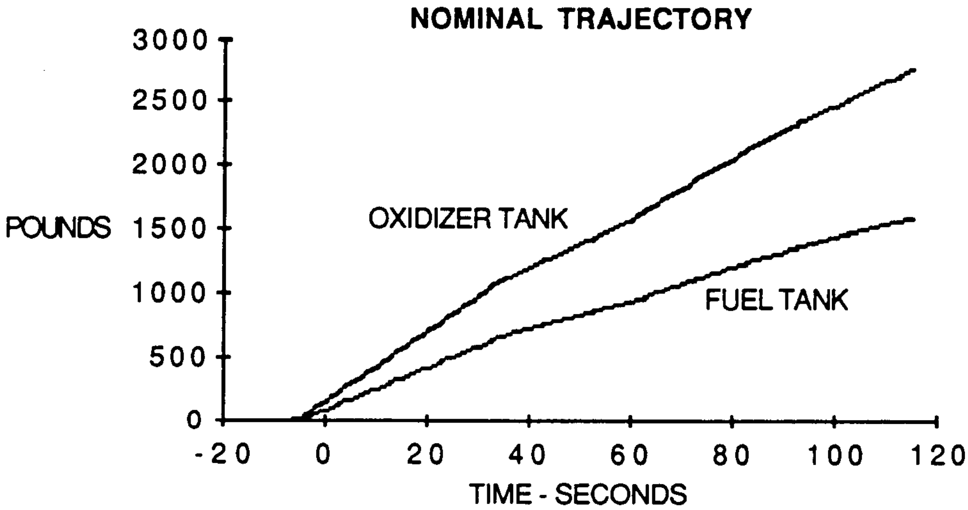


Figure 7.2.2-6

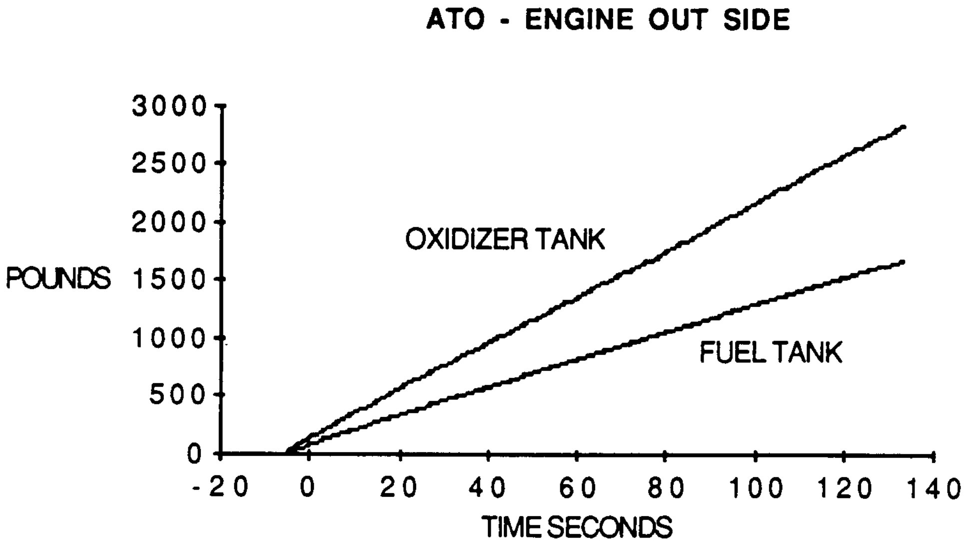


Figure 7.2.2-7

OXIDIZER TANK TEMPERATURE

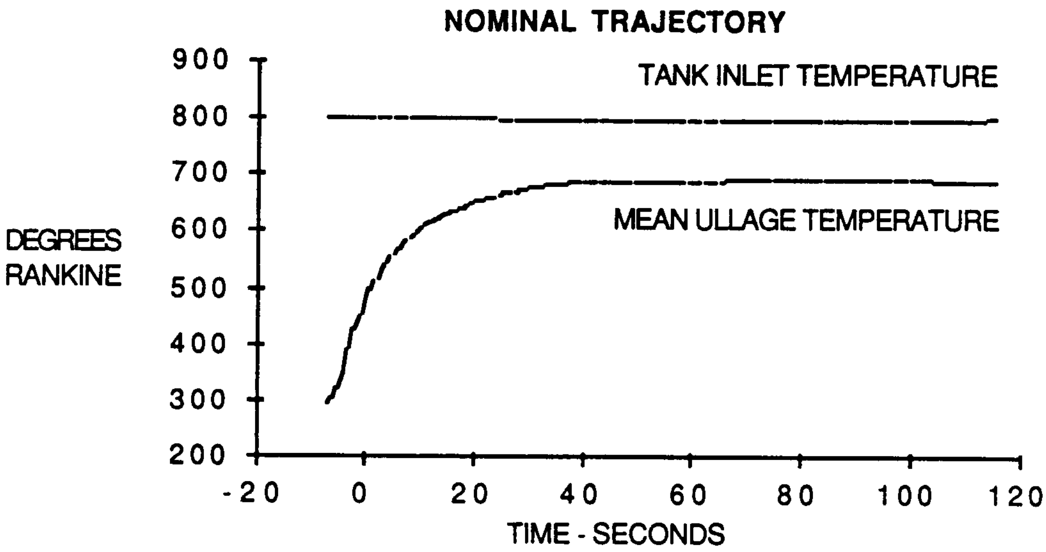


Figure 7.2.2-8

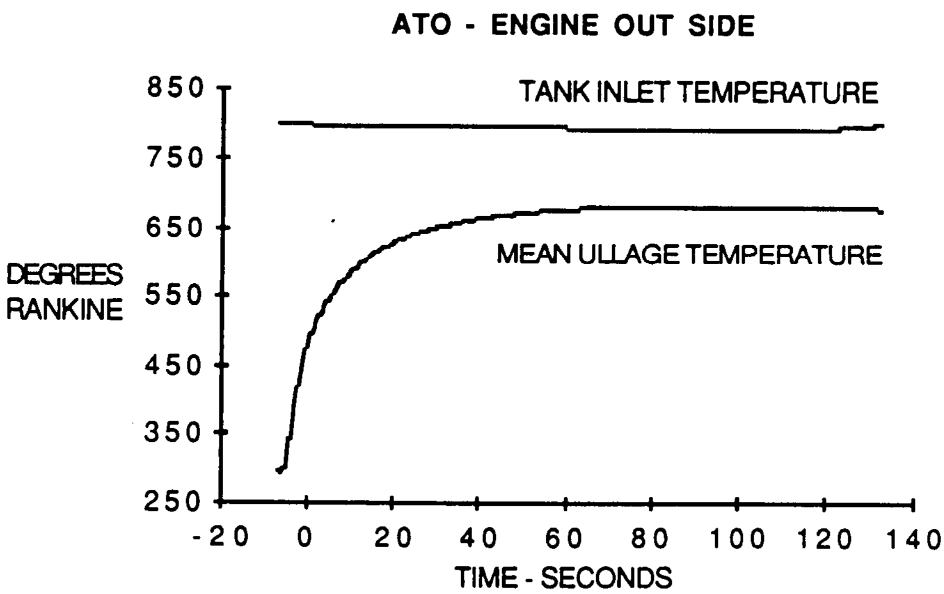


Figure 7.2.2-9

PROPELLANT TANK BOTTOM PRESSURE

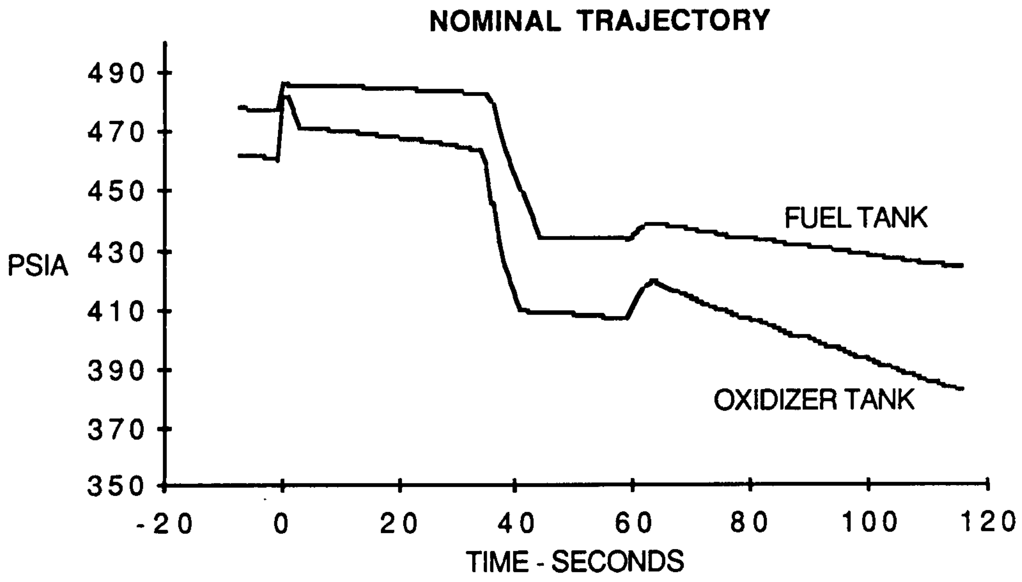


Figure 7.2.2-10

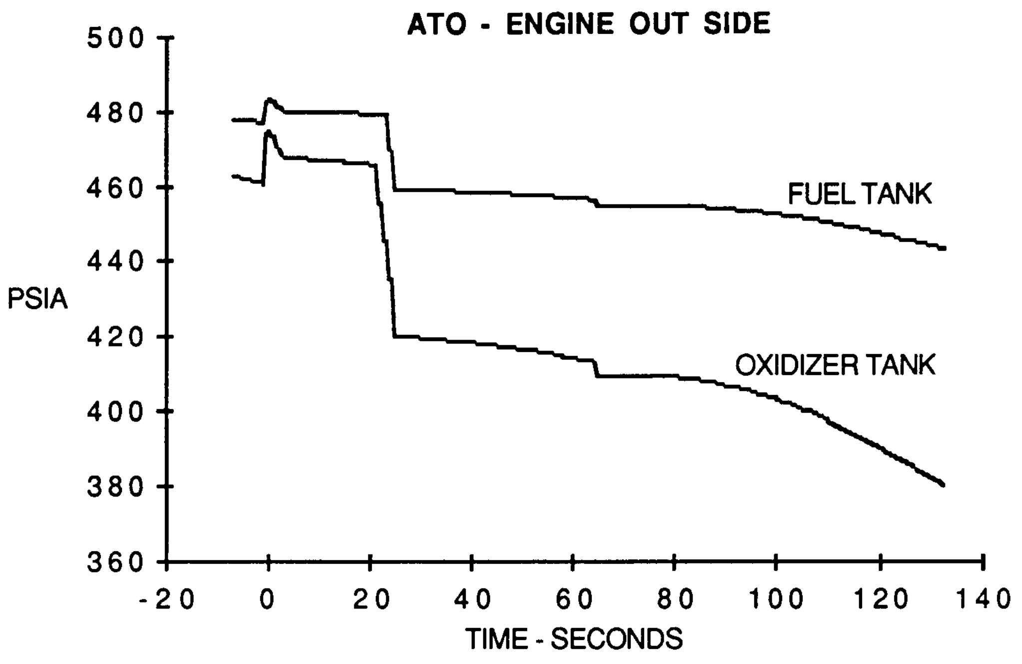


Figure 7.2.2-11

REGULATED ULLAGE PRESSURE

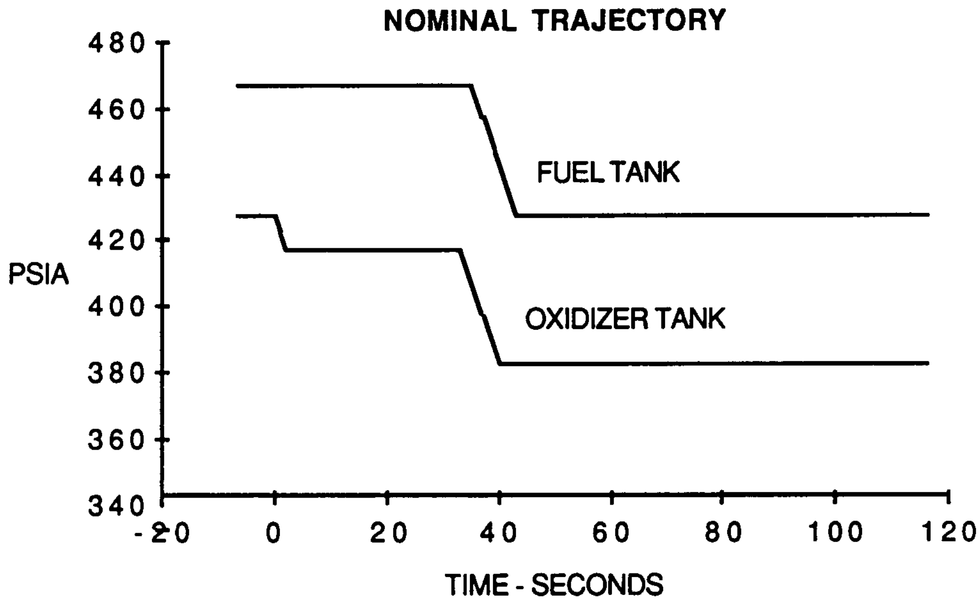


Figure 7.2.2-12

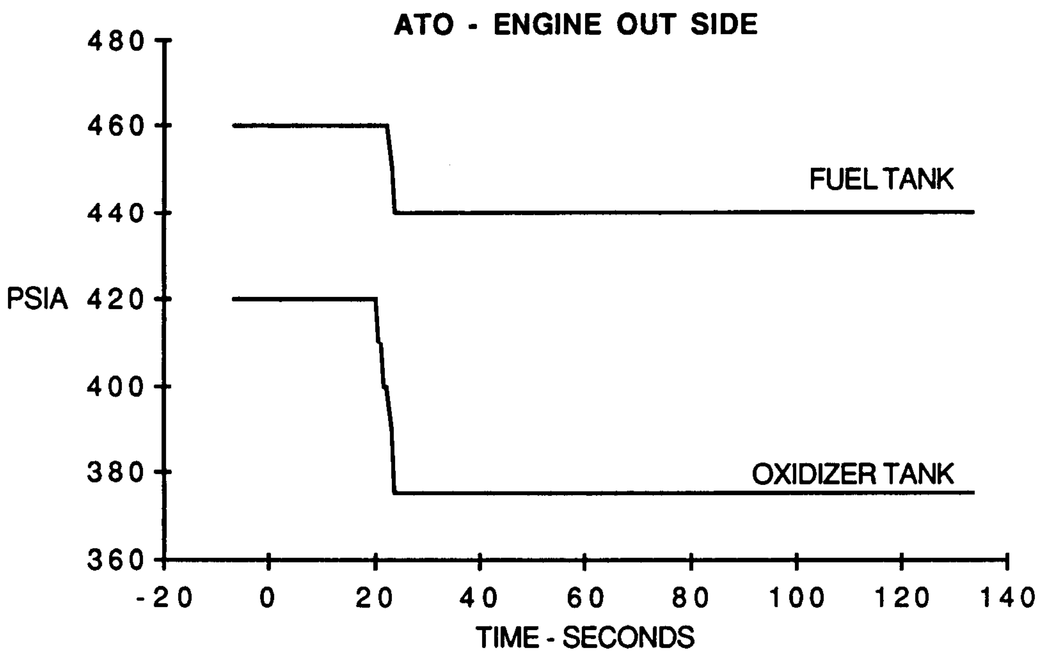


Figure 7.2.2-13

PRESSURE LOSS

SECONDARY SPHERE TO PRIMARY

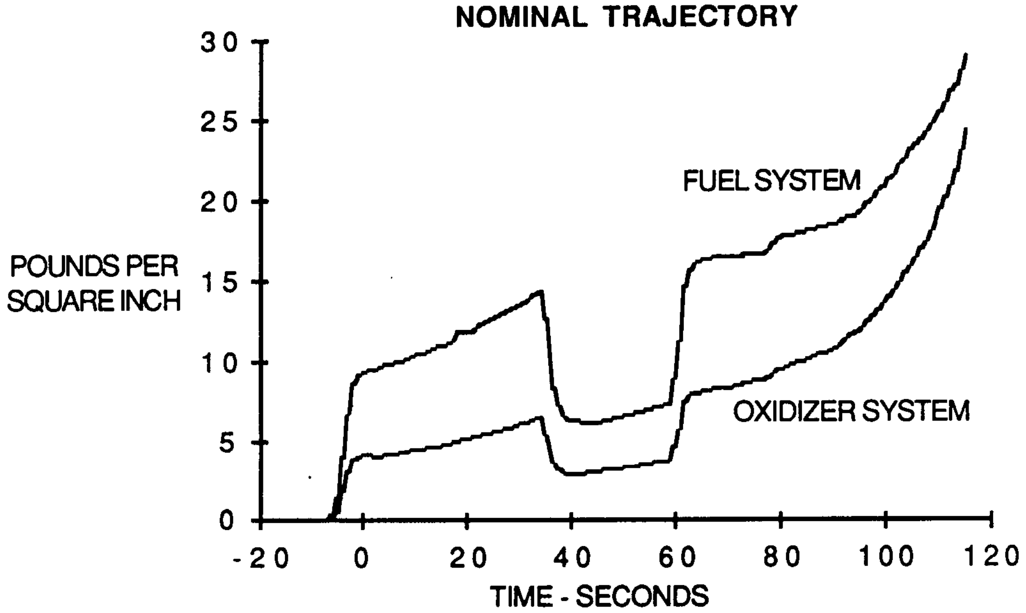


Figure 7.2.2-14

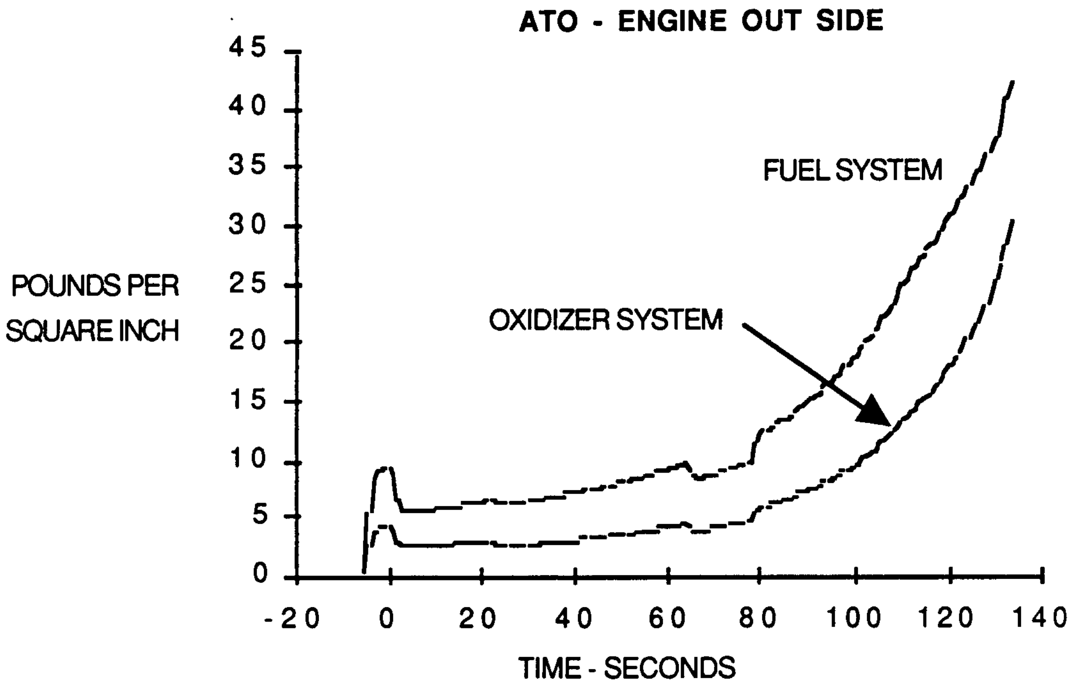


Figure 7.2.2-15

HELIUM PRESSURE LOSS
PRIMARY SPHERE TO PROPELLANT TANK
(exclusive of regulator and pressure control valve)

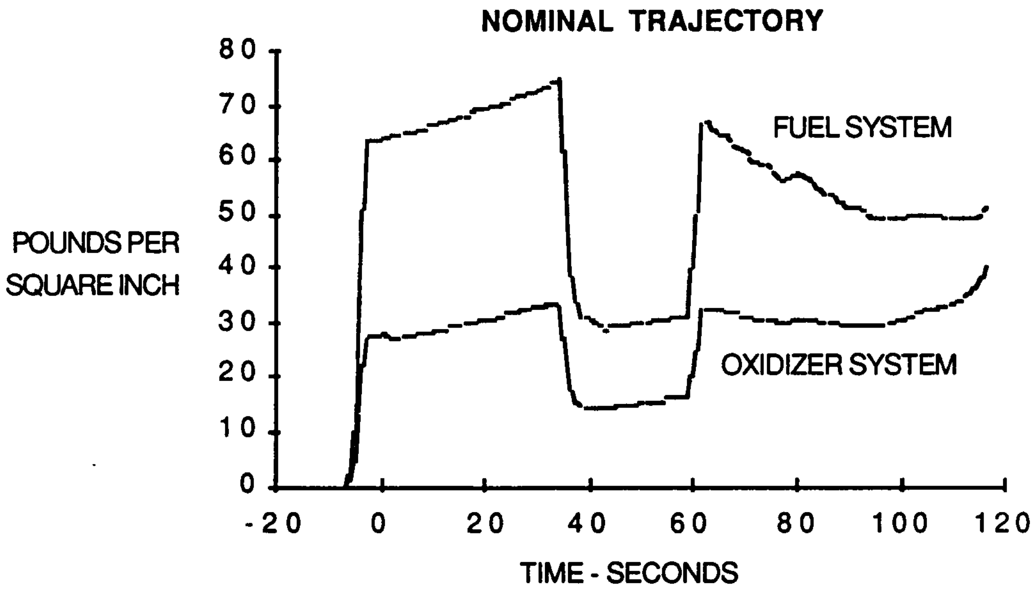


Figure 7.2.2-16

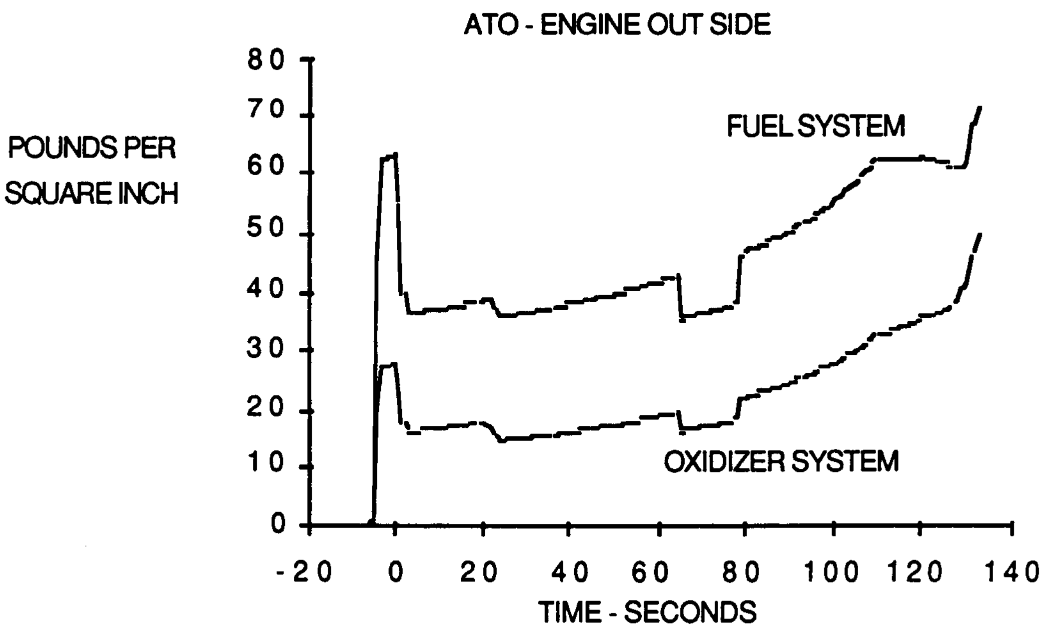


Figure 7.2.2-17

PRESSURE DROP
in
HELIUM PRESSURE CONTROL VALVE

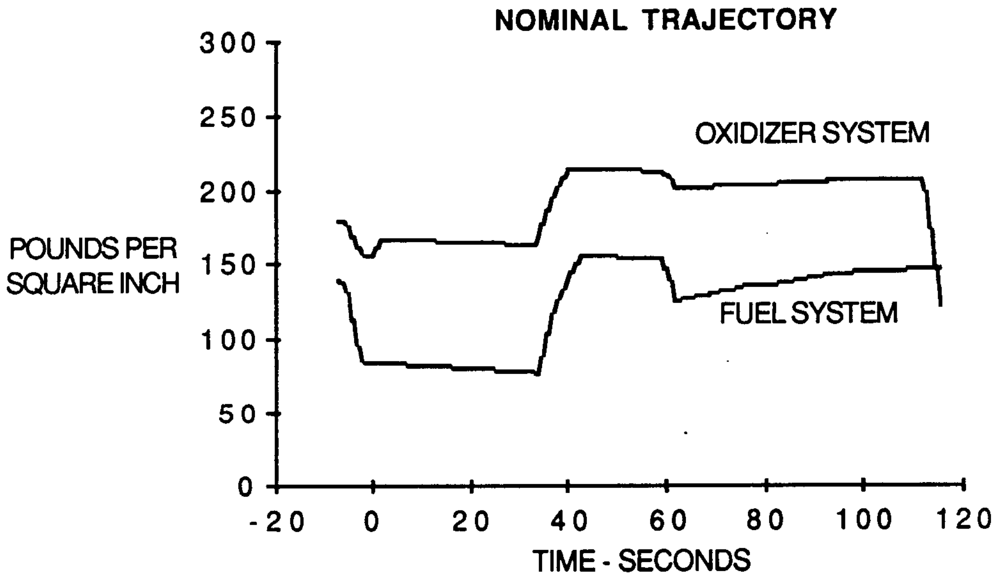


Figure 7.2.2-18

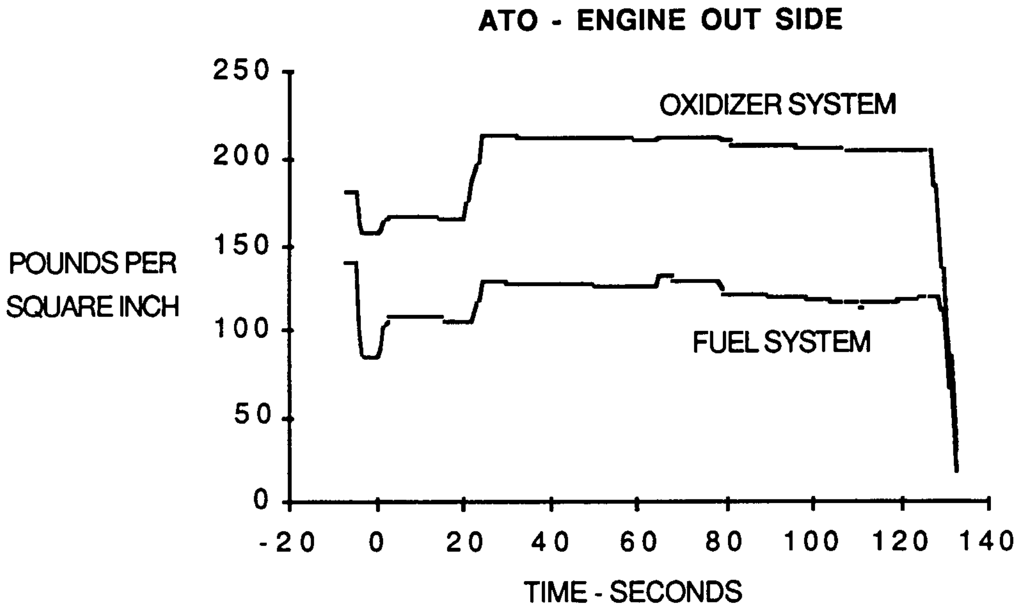


Figure 7.2.2-19

ENGINE THROTTLE PRESSURE DROP

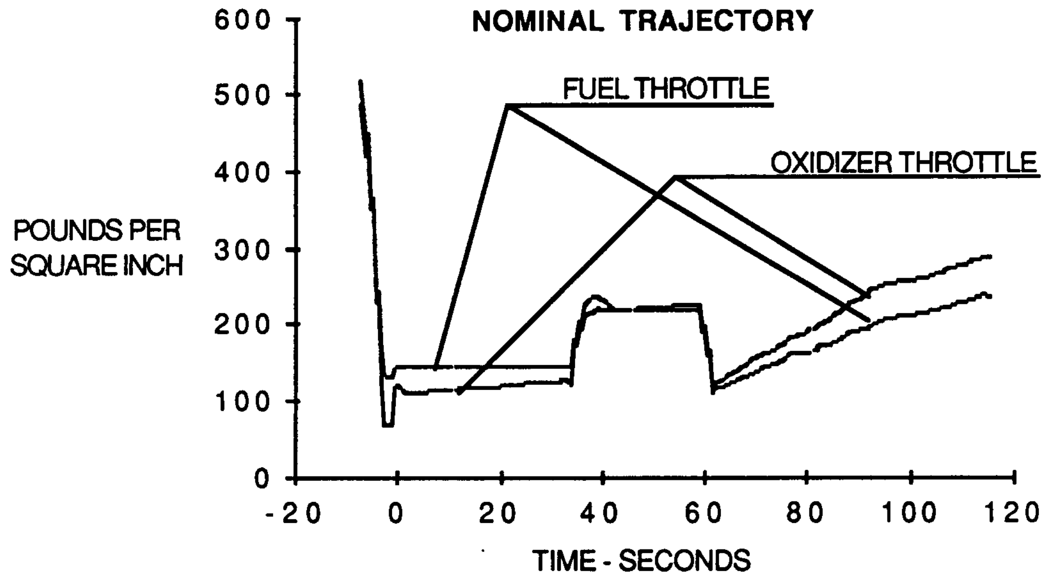


Figure 7.2.2-20

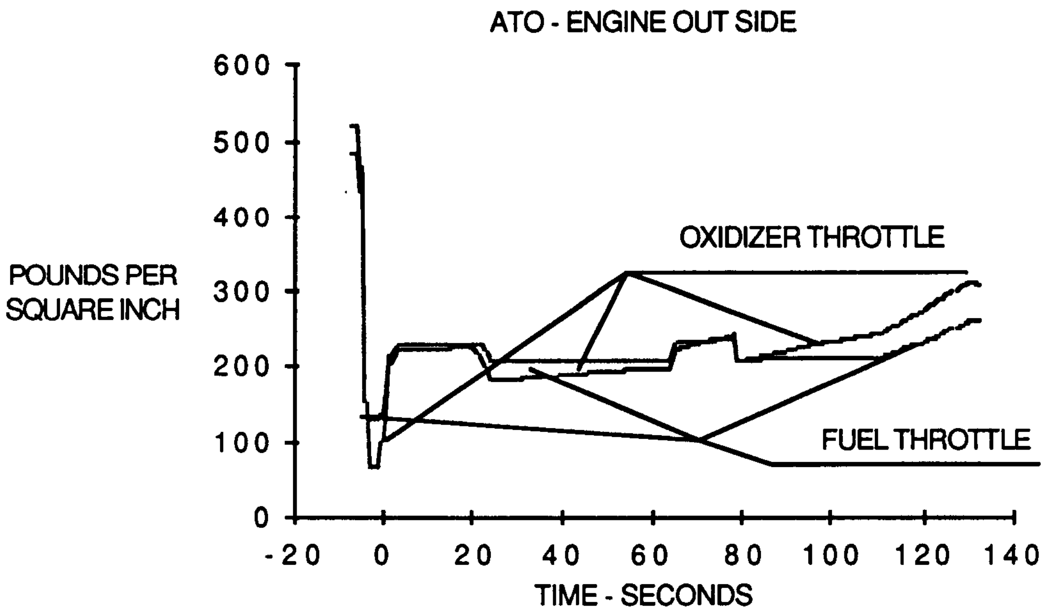


Figure 7.2.2-21

The predictions have been made using a preliminary, math model written for this purpose. Helium quantity per this model does not include the heat transfer to the pressurizing gas by the condensation/solidification of the steam generated during the catalytic combustion process. Until more accurate models are developed, this reduction potential is being considered as an unquantified, margin.

Oxidizer Pressurization Subsystem. The oxidizer pressurization subsystem is located in the Nose Cone immediately adjacent to the forward dome of the oxidizer tank. All system elements are adjacent to each other, leading to a low pressure loss design. Because of this close coupling of components the possibility of mutual self excitation and oscillation between the regulator and the pressure control valve exists. To date this has not been investigated, but if the problem does exist the addition of an accumulator will serve to decouple these components. No other potential problems due to close packaging are anticipated.

Fuel Pressurization Subsystem . The fuel pressurization subsystem is mainly located in the Aft Skirt of the vehicle, with the pressure control valve being the only major element located in the Intertank section. The volume of the hot gas pipe from the Aft Skirt to the Intertank is expected to be sufficient to damp out any tendency toward resonance between the regulator and the pressure control valve. It will, however, be investigated together with the oxidizer subsystem resonance investigation to assure adequate damping is present. As with the oxidizer subsystem, no other potentially serious problems, not previously identified in trade studies, are anticipated with this system.

Catalytic Reactor Design. The catalyst performance database used to generate the designs reported was laboratory generated at flow rates below one (1) percent of the maximum LRB design flow rate. The low temperature data limit of the database is approximately 340 degrees Rankine. For these reasons size and weight calculations were made conservatively with respect to the experimental performance test results. The potential size and weights achievable are less than half those reported in table 7.2.2-3.

Feedline Systems. All feedlines are constructed of aluminum. Flex joints, where needed, are made of propellant compatible, high strength, high fatigue life, weldable steels. These include Inconel, Armco, and CRES materials similar to those previously employed on the space shuttle and ET. The principal vendor for this type equipment is Arrowhead Products which has produced similar equipment for the Saturn and STS programs. Design will be according to the research and development sponsored by NASA and conducted by Southwest Research of San Antonio, TX. Full attention will be focused upon life and will emphasize flow induced vibration investigation, a phenomena not anticipated to be a problem in the high pressure bellows at flow rates under fifty (50) ft/sec. Flow velocity at the flex joints is approximately thirty five (35) feet per second (Ref Table 7.2.2-5).

LOX Feedline. The general configuration selected for the lox feedline is that the main downcomer be routed through a passage through the RP-1 tank on the LRB centerline (figure 7.1-1). No bellows or other expansion devices are incorporated into this line. It is planned to be a one piece extruded 23.5 in inner diameter pipe with no welds except those at each end joining it to the tank and the distribution piping of the aft skirt. The annular passage will contain sliding supports and will be sealed at each end with a very low pressure, zero flow, bellows. This annulus will be purged and filled with CO_2 at atmospheric pressure or slightly above. Upon admission of LOX to the feedline the CO_2 will solidify on the line providing an extremely high vacuum insulating barrier combined with a low emissivity "frost" of CO_2 prior to lift off. This assures minimal chilling of the RP-1 during and prior to flight, and low heat input to the LOX feedline to minimize any tendency to geyser.

After passage through the RP-1 tank, the feedline separates into four (4) distribution lines roughly 90 degrees apart (ref. figures 7.1-1 and -7). These lines initially are radial and perpendicular to the LRB centerline. These radial sections each contain two (2) flex joints of the BALSTRA type. The separation distance of these joints together with their angular deflection capacity provide a differential expansion capacity well in excess of that required with the main downcomer anchored at the LOX tank lower dome and the distribution lines anchored at the engine gimbal station.

At the separation point dividing the single main into the distribution lines, helium will be injected to prevent geysering, if tests and analytical predictions indicate such a requirement.

LOX Fill and Drain. One of the distribution lines will be utilized as part of the fill/drain pipe for the oxidizer, being coupled to the LOX fill/drain quick disconnect located on the aft skirt. This line, and the other engine feeds will also be provided with helium injection for anti-geyser purposes as required.

At the engine head end a conventional wrap around flex joint piping arrangement is employed. In the figures this is depicted as containing three (3) flex joints, but depending upon joint/line configuration may contain four (4) such joints. Similarly, the joint may be of either BLSTRA or a gimbal type (contains torsional restriction) depending upon final design requirements. The lower weight, lower cost BLSTRA is preferred if adequate.

RP-1 Feedlines. RP-1 will be ducted to each engine individually directly from the tank (Figures 7.1-7). Four feedlines will run directly from the tank, one to each engine. This not only provides the simplest approach to manifolding, but permits the RP-1 tank to act as an accumulator to decouple any tendency for engine thrust crosstalk, and to provide "pogo" suppression.

RP -1 Fill and Drain. As with the LOX system, one of the engine feedlines will double as the RP-1 fill and drain line being coupled to the fuel fill/drain quick disconnect mounted on the aft skirt. The RP-1 quick disconnect will be located between 90 and 180 degrees from the LOX QD's.

Feedline Sizing. All preliminary line sizing was performed based upon achieving the lowest practical pressure drop. When line routings were established and the location of BALSTRA's and gimbal joints were selected, fluid velocity was added as a second prime consideration in the line sizing process.

An existing pressure loss program based upon the Darcy-Weisbach equation, including an iterative solution for the Colebrook correlation of the vonKarman and Prandtl-vonKarman friction factors was used. The initial sizing utilized a program form permitting manual variation of the line diameter, holding other input factors constant. These calculations were followed by using another form of the program to automatically output flow - loss tables for the flow range specified. The LOX final configuration outputs are presented in Tables 7.2.2-4 and 7.2.2-5. Similar data for the RP-1 feedlines is given in Table 7.2.2-6.

Propellant Tankage. A geyser baffle plate is located with and supported by the anti vortex baffling in the LOX tank. Baffles are not shown in the tanks, and further control/stability analysis are required to verify that baffles are not needed.

Table 7.2.2-4 Oxidizer Main Pressure Loss-Flow Performance

LINE LENGTH	58.6 ft
INSIDE DIAMETER	23 in

FLOW lb.sec	VELOCITY ft/sec	PRESS.LOSS psi
NOMINAL THRUST FLOW		
9889.6	48.29	7.52
THROTTLING		
10000	48.83	7.69
9000	45.90	6.27
8000	39.07	4.99
7000	34.18	3.86
6000	29.30	2.87
5000	24.42	2.03

Table 7.2.2-5 Oxidizer Feeder Pressure-Flow Performance

LINE LENGTH	15 ft
EQUIPMENT EQUIVALENT LENGTH	115 ft
EFFECTIVE LINE LENGTH	120 ft
INSIDE DIAMETER	14 in

FLOW lb/sec	VELOCITY ft/sec	PRESS. LOSS psi
FLOW AT NOMINAL THRUST		
2472.4	32.59	12.16
THROTTLING		
2500	32.95	12.42
2250	29.65	9.73
2000	26.36	8.11
1759	23.06	6.29
1500	19.77	4.70
1250	16.47	3.33
1000	13.18	2.19

Table 7.2.2-6 Fuel Feedline Pressure-Flow Performance

LINE LENGTH	15.0 ft
EQUIPMENT EQUIVALENT LENGTH	33.3 ft
EFFECTIVE LINE LENGTH	48.3 ft
INSIDE DIAMETER	10.0 in

FLOW lb/sec	VELOCITY ft/sec	PRESS. LOSS psi
FLOW AT NOMINAL THRUST		
989.0	35.82	8.38
THROTTLING		
1000	36.22	6.41
900	32.60	5.23
800	28.98	4.17
700	25.35	3.23
600	21.73	2.41
500	18.11	1.70

The RP-1 tank will require only an anti-vortex device. This will be primarily supported by the casing around the LOX main downcomer, and will be suitably designed to prevent localized vortexing at each tank exit as well as the tendency for a full tank vortex.

Vent Relief Valve Locations and Lines. (The location of the items discussed in this section are shown in the vehicle sketch, Figure 7.1-1.)

RP-1 Tank. The vent and relief valves for the RP-1 tank are located in the intertank section on the top of the tank. Separate valves are proposed for each function as this seems to be a more cost effective and trouble free option. The valve discharges will be combined into one to duct the gasses overboard away from the orbiter.

LOX Tank. The LOX tank vent and relief valves will be located in the nose cone on the top of the tank. These valves will discharge into a common downcomer along which runs down the LOX tank, the intertank, and the RP-1 tank exterior and into the aft skirt to the vicinity of the LOX fill and drain quick disconnect. Hard mounting of this line will be near the aft end of the LOX tank, all other exterior mounts being the sliding type. Relative expansion will be provided for by using BALSTRA type joints at the top of the LOX tank and at the entry of the vent line into the aft skirt. In spite of being the least expensive joint, its proper application is capable of the greatest compensation for expansion.

Propellant Management System and Propellant Inventory. The liquid propellant management and inventory systems are identical in concept and most hardware to those described previously in Section 4.2.2.4. The only real differences are those directly associated with the absence of turbomachinery from the pressure fed propulsion system.

The gaseous propellant (pressurant) management is accomplished by the periodic reduction of propellant tank ullage pressure to the minimum consistent with current and subsequent thrust requirements. (A continually fluctuating pressure consumes more pressurant due to repeated compression-expansion losses.)

Gasseous propellant inventory is accomplished with pressure sensors for each storage sphere. Because of the continuous flow through the systems, and the relative closeness of the pressures between spheres in each cascade, sensor redundancy in each sphere is not deemed necessary for normal operations.

7.2.3 THRUST VECTOR CONTROL (TVC). For system description and schematic see Section 4.2.3.

A summary of the TVC requirements for the LOX/RP-1 pressure-fed engine is shown in Table 7.2.3-1.

Table 7.2.3-1 LRB TVC Requirements for LOX/RP-1 Pressure Fed Engine

Input Parameters		**Source**
Thrust per Engine	971595 -lbs.**	Rocketdyne
Number of engines on LRB	4	LRB Baseline
Thrust Vector Offset	0.25 -inches	Rocketdyne
Gimbal Block Coeficient of Friction	0.06	Rocketdyne
Gimbal Block Pin Radius	5.50 -inches	Back-calc from R data
Maximum Gimbal Angle	6.00 -Degrees	Specification
Gimbal Rate Required	10.00 -Deg/Sec	Specification
Gimbal Acceleration Required	57.30 -Deg/Sec^2	Specification
Engine Weight	7017 -lbs.	Rocketdyne
Engine Inertia	6912 -lb-ft^2	Back-calc from R data
Distance from Engine C.G. to Gimbal	62.62 -inches	Rocketdyne
Distance from Vehicle C.G. to Gimbal	100.00 -ft	Estimated
Actuator Moment Arm	70.73 -inches	Rocketdyne
C.G. Offset from Centerline of Eng.	0.00 -inches	Assumed
Max Veh Longitudinal Acceleration	3.00 -g's	STS limit
Max Veh Lateral Acceleration	0.30 -g's	STS/LRB Traj Sim
Max Veh Angular Acceleration	3.00 -Deg/Sec^2	STS/LRB Traj Sim
LOX Line Torque	7218 -ft-lbs	Rocketdyne
Fuel Line Torque	4876 -ft-lbs	Rocketdyne
Total Flex Line Stiffness Torque	12094 -ft-lbs	Rocketdyne

Torque Calculations		
Longitudinal Acceleration Torque	11482 ft-lbs	137789 in-lbs
T1 - Due to Engine C.G. Offset	0 ft-lbs	0 in-lbs
T2 - Due to Max Gimbal Angle Offset	11482 ft-lbs	137789 in-lbs
Lateral Acceleration Torque	17301 ft-lbs	207614 in-lbs
T3 - Due to Vehicle Lateral Acc	10985 ft-lbs	131821 in-lbs
T4 - Due to Engine Inertia	362 ft-lbs	4343 in-lbs
T5 - Due to Vehicle Angular Accel	5954 ft-lbs	71450 in-lbs
T6 Thrust Misalignment Torque	20242 ft-lbs	242899 in-lbs
T7 Engine Control Torque	6912 ft-lbs	82940 in-lbs
T8 Engine Block Friction Torque	26719 ft-lbs	320626 in-lbs
T9 Propellant Duct Torque (Given)	12094 ft-lbs	145128 in-lbs
Total Static Torque	61119 ft-lbs	733429 in-lbs
Total Dynamic Torque	33630 ft-lbs	403566 in-lbs
Total Required Torque	**94750 ft-lbs**	**1136995 in-lbs**

Peak Power Requirements (Using Torque X Gimbal Rate)		
Peak Output power req'd per Actuator	30.1 -hp	22.4 -kW
Peak input Power/Act (sys eff = 53%)	56.7 -hp	42.3 -kW
Peak Power Required per Engine	113.5 -hp	84.6 -kW
Total Peak Required for LRB	**453.8 -hp**	**338.4 -kW**

Actuator Sizing	
Peak Operating Output Force	16075 -lbs
Stall Force	24113 -lbs

** Head-end gimbal point

7.3 AVIONICS

The avionics for the pressure-fed configuration will be very similar in both architecture and function to the LOX/RP-1 pump fed configuration (see Section 4.3). Future analysis will undoubtedly reveal differences in the avionics; however, these differences are thought to be relatively minor changes to the basic design.

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7.4 RP-1 PRESSURE-FED PERFORMANCE AND TRAJECTORIES

7.4.1 NOMINAL MISSION. A description of the nominal mission trajectory simulation can be found in section 8.1.3. Since the ATO mission determined the required size of the RP-1 pressure-fed LRB configuration, the nominal trajectory simulations were run simply to determine nominal performance.

The following table is a summary of the nominal performance for the RP-1 pressure-fed configuration.

```
Lift off conditions:

   Weight (lb)                              =   5,140,224.5000
   Payload (lb)                             =   70,500.000000
   Thrust (lb)                              =   7,859,059.4788
   Thrust to weight                         =   1.5289331193
   Initial inertial velocity (ft/sec)       =   1,342.4324022
   Launch site latitude                     =   28.307566153
   Launch site longitude                    =  -80.540959056


Max Q conditions:

   Max dynamic pressure (lb/ft**2)          =   717.42405671
      Time (sec)                            =   59.837431445
      Angle of attack (deg)                 =   4.1816266933
      Altitude (ft)                         =   35,045.196851
      Mach number                           =   1.4332011615
      Q * ALPHA (deg-lb/ft**2)              =   2,999.9995860


LRB separation:

   Staging time (sec)                       =   119.48415690
   Altitude (ft)                            =   141,605.07758
   Dynamic pressure (lb/ft**2)              =   65.709505766
   Angle of attack (deg)                    =  -1.9999997282
   Mach number                              =   4.8739211434
   Inertial velocity (ft/sec)               =   6,391.5058060
   Inertial flight path angle (deg)         =   21.562839660
   Relative velocity (ft/sec)               =   5,159.7678711
   Relative flight path angle (deg)         =   27.081409402
   Delta V (ft/sec)                         =   8,949.7003737
   Weight after separation (lb)             =   1,557,822.5739
   Remaining ET propellent (lb)             =   1,201,537.5915
   SSME throttle at separation              =   1.0400000000
   LRB throttle at separation               =  0.60534899658
   Thrust (lb)                              =   1,464,735.2746
   Thrust-to-weight after separation        =  0.94024524945
   Acceleration after separation            =  0.92304315000
   LRB propellent used (lb)                 =   2,695,330.0176
   Geodetic latitude (deg)                  =   28.491599321
   Longitude (deg)                          =  -80.028972251
```