L. D. Wear
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CONTRACT NO. NAS8-37137

# VOLUME I
# EXECUTIVE SUMMARY


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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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# LIQUID ROCKET BOOSTER STUDY
# FINAL REPORT

## VOLUME I • EXECUTIVE SUMMARY

CONTRACT NO. NAS8-37137

March 1989

Submitted to
National Aeronautics and Space Administration
Marshall Space Flight Center

Prepared by

GENERAL DYNAMICS

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# ACRONYMS AND ABBREVIATIONS

**A**

| | |
|---|---|
| ALS | Advanced Launch System |
| AOA | Abort Once Around |
| AR | Area Ratio |
| ATO | Abort to Orbit |

**B**

| | |
|---|---|
| BECO | Booster Engine Cutoff |
| BLOW | Booster Liftoff Weight |
| BSM | Booster Separation Motor |

**C**

| | |
|---|---|
| $CH_4$ | Methane |
| CMD | Command |
| CDR | Critical Design Review |

**D**

| | |
|---|---|
| DC | Dual Channel |
| DDT&E | Design, Development, Test, and Engineering |

**E**

| | |
|---|---|
| ECO | Engine Cutoff |
| ECS | Environmental Control System |
| EMA | Electrical Mechanical Actuator |
| ERB | Engineering Review Board |
| ET | External Tank |

**F**

| | |
|---|---|
| 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, g | Acceleration of Gravity |
| GDSS | General Dynamics Space Systems |
| GLOW | Gross Liftoff Weight |
| GLP | Ground Launch Processing |
| GOX | Gaseous Oxygen |
| GSE | Ground Support Equipment |

**H**

| | |
|---|---|
| HYPRS | A Pressurization Model Computer Program |

**I**

| | |
|---|---|
| IRD | Interface Requirements Document |
| ICD | Interface Control Document |
| IRD | Interface Requirements Document |
| ISP | Specific Impulse |

**J**

| | |
|---|---|
| JBV | Jacket Bypass Valve |
| JSC | Johnson Space Center |

**K**

| | |
|---|---|
| Klb | Thousands of Pounds |
| KSC | Kennedy Space Center |

**L**

| | |
|---|---|
| LCC | Life Cycle Cost |
| L/D | Length to Diameter Ratio |
| LEMSCO | Lockheed Engineering and Management Services Company |
| LOX/$LO_2$ | Liquid Oxygen |
| LPS | Launch Processing System |
| LRB | Liquid Rocket Booster |
| LSOC | Lockheed Space Operations Company |

**M**

| | |
|---|---|
| MECO | Main Engine Cutoff |
| MLP | Mobile Launch Platform |
| MMC | Martin-Marietta Corporation |
| MMH | Monomethyl Hydrazine |
| MOV | Main Oxygen Valve |
| MR | Mixture Ratio |
| MSFC | Marshall Space Flight Center |

**N**

| | |
|---|---|
| $N_2$ | Nitrogen |
| NASA | National Aeronautics and Space Administration |
| nmi | 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 |

**O**

| | |
|---|---|
| OCV | Oxygen Cooldown Valve |
| ODC | One-Dimensional Equilibrium |
| OMS | Orbital Maneuvering System |
| ORB | Orbiter |

**P**

| | |
|---|---|
| P/A | Propulsion/Avionics |
| Pc | Engine Combustion Chamber Pressure |

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| | |
|---|---|
| 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 and Whitney Company |
| **Q** | |
| Q | Dynamic Pressure |
| Q-Alpha | Dynamic Pressure x Angle of Attack |
| **R** | |
| R | Rankine |
| RP-1 | Rocket Propellant |
| RSS | Rotating Service Structure |
| RTLS | Return to Launch Site |
| **S** | |
| SC | Single Channel |
| sec | Second |
| 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 | Superlight 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 |
| **T** | |
| TBV | Turbine Bypass Valve |
| TFU | Theoretical First Unit |
| TPS | Thermal Protection System |
| TVC | Thrust Vector Control |
| T/W | Thrust to Weight Ratio |
| **V** | |
| V | Velocity |
| VAB | Vertical Assembly Building |
| VAC | Vacuum |
| **W** | |
| WBS | Work Breakdown Structure |

# CONTENTS

# FOREWORD

This report presents the results of the first sixteen months of work under Contract NAS 8-37137, "Liquid Rocket Booster for STS Systems Study." This work was performed by General Dynamics Space Systems Division under the Program Office for NASA Advanced Space Transportation Systems, headed by Howard Bonesteel and Paul Bialla. The contract was directed at NASA Marshall Space Flight Center by Larry Wear and Ned Hughes. The principal study team contributors included:

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|---|---|
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| Olen Britnell | Flight Control and Avionics |
| Hal Britton | Propulsion Subsystems |
| John Burgeson | Structural Design |
| Jerry Davis | Ignition and Release Sequence |
| Dan Eimers | Economic Analysis |
| Greg Farmer | Engine Test Bed Requirements |
| Paul Ferrara | Structural Design and Pump-fed Configurations |
| Frank Hauser | Control Analysis |
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| Ken Miller | Avionics Systems |
| Tina Nguyen | Propulsion/Propellant Systems |
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| | |
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| Bill Pierce | Pressurization Systems Trades |
| Art Prast | Mass Properties |
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| Mike Simon | Program Implementation Plans Lead |
| Donna Smith | Geometry Trade |
| Joe Szedula | Performance Analysis |
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| Mike Vaccaro | Systems Engineering |
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**Subcontractors**

TRW

Pioneer

Eagle Engineering

Rocketdyne

PRC

Pratt & Whitney

# 1 ✦ STUDY BACKGROUND AND RESULTS

In a number of prior studies liquid propulsion was considered as an alternative to solid rocket motors for the Space Shuttle boosters. Most recommended liquid oxygen/liquid hydrogen ($LO_2/LH_2$) configurations with four Space Shuttle main engines (SSMEs). The tragic *Challenger* accident of January 1986 revived interest in LRBs for the Shuttle, particularly to improve safety.

This study and the companion Martin-Marietta Corporation (MMC) study treated Shuttle and KSC compatibility and safety in greater depth than previous work. NASA/MSFC led this study and provided essential wind tunnel data on the larger LRB configurations. NASA/JSC and their contractor Lockheed Engineering and Management Services Company (LEMSCO) evaluated STS compatibility, emphasizing Orbiter wing loads and trajectories constraints. NASA/KSC and their contractor, Lockheed Space Operations Company (LSOC), also participated in the LRB study by evaluating the impact of LRB concepts on KSC facilities and operations. Together this government-contractor team provided more depth than a normal Phase A study.

General Dynamics Space Systems Division performed this study with personnel located both in Huntsville, Alabama and San Diego, California. Our study team included Rocketdyne, TRW, and Pratt & Whitney for engine concepts; Eagle Engineering for Shuttle interfaces and operations; Planning Research Corporation (PRC) for KSC interfaces and operations; and Pioneer for recovery systems. The period of performance for this Phase A concept study was October 1987 to January 1989. This Executive Summary of the study final report provides a brief overview of the study methodology, results, and principal recommendations.

## 1.1 PROGRAM OBJECTIVES AND SIGNIFICANT FINDINGS

The purpose of this study was to determine the feasibility of liquid rocket boosters (LRBs) replacing solid rocket boosters (SRBs) on the Space Shuttle program. Table 1-1 lists major findings.

## 1.2 SIGNIFICANT STUDY ACHIEVEMENTS

The most significant conclusion of this study is that LRBs offer significant safety and performance advantages over the SRBs currently used by the Space Transportation System (STS) without major impact to the ongoing program.

Existing, proven technologies are sufficient to support development and operation of pump-fed LRBs with low risk. A propulsion system test program is needed to reduce risk in pressure-fed concepts. The transition from SRBs to LRBs, including modification of the STS and support facilities, is manageable though challenging.

*Table 1-1. LRBs offer significant advantages over SRBs.*

| Goals | Significant Findings |
|---|---|
| **Exceed SRB in:** | |
| – Safety | – Engine-out, intact abort capability |
| | – Engine shut down/throttling enabled boost phase aborts |
| | – No hazardous propellants in VAB |
| – Performance | – 20K performance increase |
| | – Enables improved aborts, mission flexibility |
| | – No boost phase SSME throttling |
| – Environmental impacts | – Less contaminating exhaust products |
| **Satisfy essential design criteria** | |
| – Minimal impact on Shuttle | – Minimum Orbiter, ET modifications |
| | – Down time of STS avoided during transition of SRB to LRB |
| | – Better operation with STS liftoff/ascent constraints |
| – Minimal facility impacts | – No significant launch pad modifications |
| – High reliability | – Mission accomplished with one LRB engine out |
| | – Reduced critical areas |
| | – Engine health verified prior to release |
| **Attractive programmatic features** | – Evolutionary capability |
| | – Common application of engine and booster in alternate applications (ALS, Shuttle C, Standalone) |

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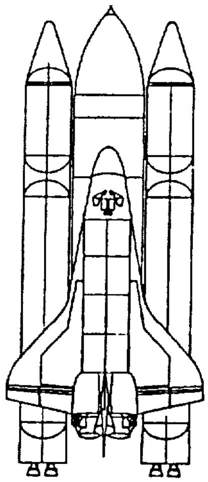
## 1.3 RECOMMENDED LRB CONFIGURATION

Figure 1-1 shows our recommended LRB concept using a new low-cost main engine that burns liquid oxygen and liquid hydrogen. A new $LO_2/LH_2$ engine in the half-million-pound thrust class can meet the needs of STS LRB and the Air Force's Advanced Launch System (ALS), and serve as an alternative for Shuttle-C (instead of SSMEs).

Some basic LRB features we recommend include:

- **Liquid oxygen and liquid hydrogen propellant combination.** $LO_2/LH_2$ has the least environmental impact and is the same propellant used with the current STS, and core vehicle of the ALS program. Our second choice is $LO_2$/RP-1, which has been used since the 1950s with the highly reliable Atlas, Delta, and Saturn launch vehicles. Both have significant safety and environmental advantages over storable propellants ($N_2O_4$/MMH). $LO_2/CH_4$ was evaluated but



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 (70.5 Klb to 160 nmi)
- 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

GSX0150-2

*Figure 1-1. Recommended LRB concept.*

not selected, because the benefits were not significant enough to offset the lack of experience with and cost of an all-new engine development program based on a new fuel.

- **A new low-cost, pump-fed engine.** This option is considerably more cost-effective than adapting an existing engine such as the SSME or the earlier, Saturn class first stage of engine to the LRB.

- **Four engines on each LRB.** This feature will provide the best combination of engine-out capability, reliability, and cost. Note that these are low complexity engines of the gas-generator-type cycle, which have been previously flown.

- **Initiate LRB program as expendable system.** Recoverable systems with engines designed for long life cannot compete effectively with less expensive expendable engines. A limited-life recoverable engine concept offers attractive cost benefits over expendables, although unresolved issues remain such as the uncertainties in engine recovery/refurbishment and verification of reuse of an expendable engine.

- **Diameters up to 18 feet.** Such configurations will still maintain STS compatibility. The latest aerodynamics and wing load study results indicate that vehicle diameters greater than 18 feet incur performance penalties and limit flight trajectory design options.

- **Common development/production with ALS.** The recommended LRB engine can be developed and produced in common with the USAF ALS program.

### 1.4 PROGRAM COST

The LRB vehicle can be developed for approximately $2 billion in DDT&E (FY1987 dollars) and could be in operation as an element of the Space Transportation System by the end of 1996. LRB recurring production and operations cost are estimated to be $65 million per STS mission.

The LRB program cost estimates are sensitive to a variety of programmatic assumptions. Cost reductions could be realized through higher LRB or engine production rates or by employing cost-saving production and management techniques that are being examined in the ongoing ALS study.

# 2 ✦ STUDY APPROACH

Our study approach was to review all previous LRB studies carefully and then generate requirements. With these requirements basic trades such as propellant selection were performed, from which concepts were sized, and then the best were selected using an approved list of criteria.

## 2.1 BASIC REQUIREMENTS, GROUND RULES, AND ASSUMPTIONS

- Each concept sized for 70.5-Klb payload to 160 nmi due east from KSC
- Intact abort with one LRB engine out
- General Dynamics goal: Full payload Abort to Orbit (105 nmi) with one LRB engine out
- High reliability or probability of mission success (approximately 0.99)
- Virtually no hardware changes to Orbiter
- STS trajectory constraints used on max Q, max G, etc.
- Orbiter wing loads limited to current levels
- Changes to external tank (ET) minimized
- Reasonable changes to KSC facilities and GSE (need new mobile launch platform [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, alternate recovery modes, and evolutionary options — led us to a concept selection approach in which selections were made in three stages.

In the first part of the study, engines and propellants were evaluated on the basis of safety, performance, and STS compatibility. Figure 2-1 shows the 15 combinations of engine type and propellant combinations that were initially considered. Separately, various recovery options for those selected LRBs were examined (including the consideration of not recovering the boosters). Cost comparisons became important at that time. Finally, each candidate was examined for evolution and growth approaches through the analysis of alternate growth paths.

Current engines included those candidates judged suitable for LRB application that 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 features desirable for LRB applications. New engine designs were based on NASA/MSFC Space Transportation

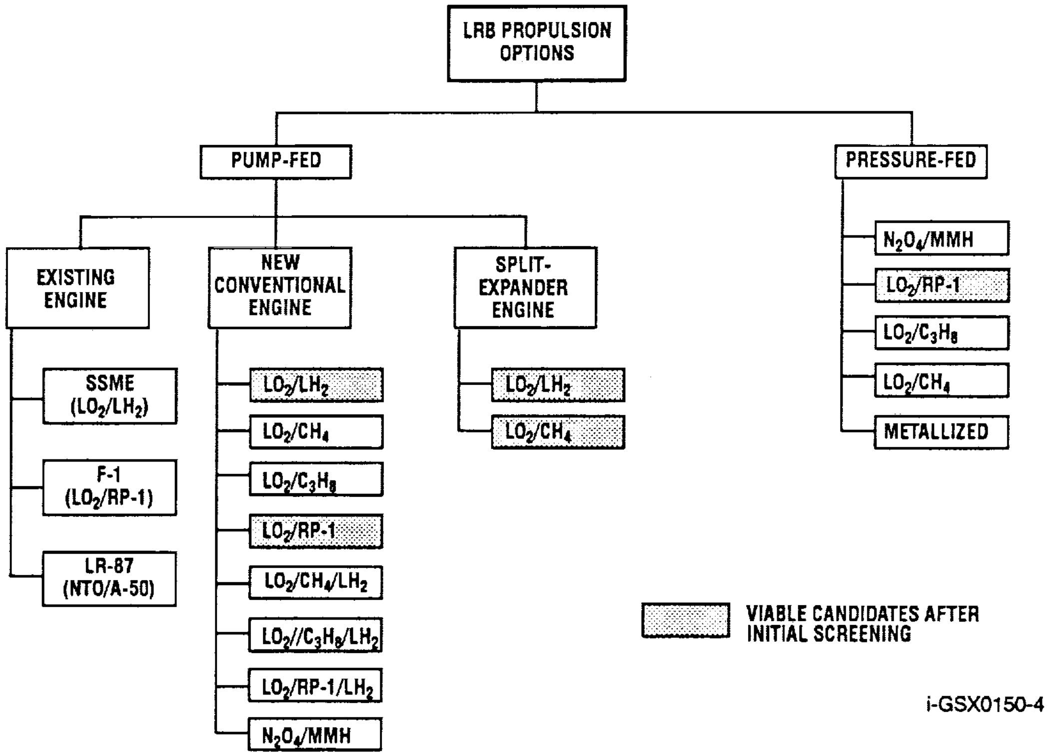



*Figure 2-1. Range of LRB concepts evaluated.*

Booster Engine (STBE) and Space Transportation Main Engine (STME) studies. Pressure-fed engine data was provided by the engine subcontractors, Rocketdyne and TRW. We also looked at metallized propellant systems, which offer high density-impulse characteristics, but require technological advancements.

Figure 2-2 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 concerns caused by large LRBs. These concerns were later dispelled, as discussed in Section 4.

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 predesign 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 overstressing the ET $LO_2$ tank aft bulkhead.

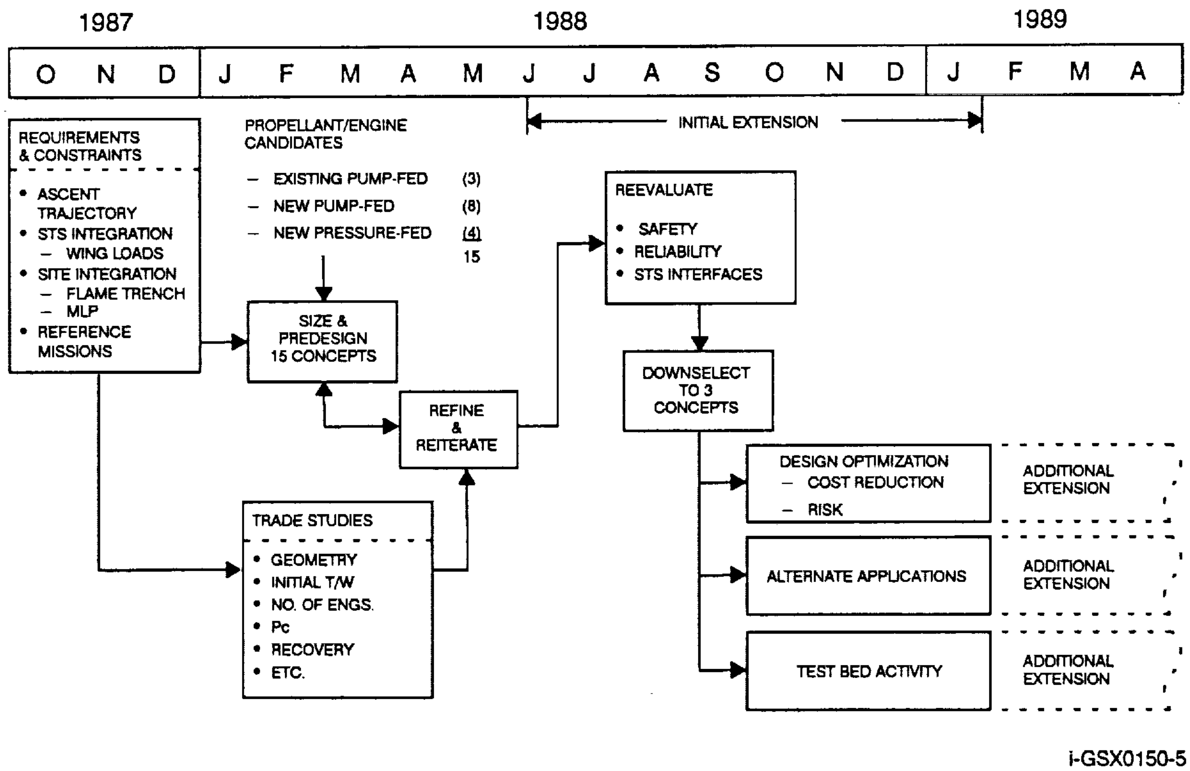



*Figure 2-2. Approach to LRB concept selection.*

To prevent prematurely eliminating a booster class from further consideration, we structured our approach to ensure that three concepts were carried through the evaluation process for further definition: one pressure-fed concept, one pump-fed concept with existing engines, and one pump-fed concept using new engines. An additional concept was added later that uses a new pump-fed engine of the split-expander type.

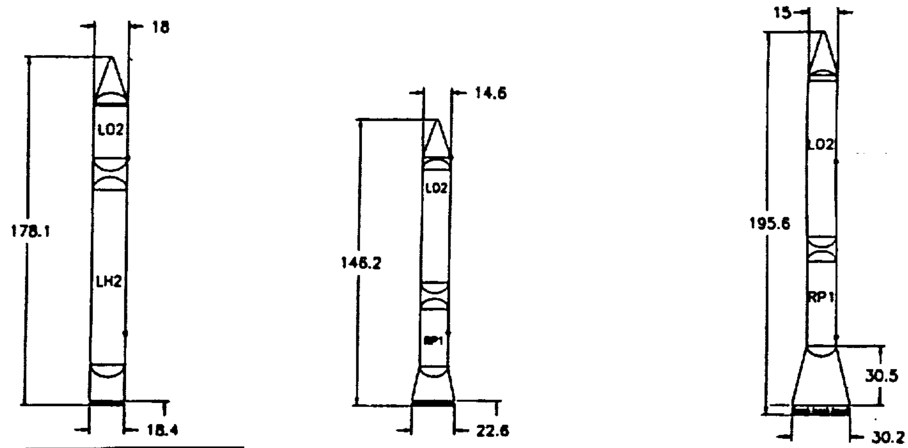
Cost was initially considered a secondary selection criterion. As concepts became better defined, the accuracy of cost estimates improved and their importance increased. Cost was the key to the recoverability trade.

Growth potential was also treated as a secondary selection criterion, but was significant in some later choices.

During the study extension from June 1988 to January 1989, we concentrated on three concepts that all met the basic performance, safety, and compatibility requirements. We reduced cost estimates through a number of trades. For instance, lower-thrust engines can reduce vehicle costs even though the vehicle becomes larger. We evaluated the cost and schedule risk in each approach. We predesigned numerous growth options, finding potential commonality with ALS.

Figure 2-3 shows our final concept evaluation based on risk, growth, and cost. In the end, the final selection criteria boiled down to versatility. A liquid oxygen/liquid hydrogen LRB was the most versatile primarily because of its commonality with ALS applications. An $LO_2$/RP-1 pump-fed LRB has features particularly attractive to STS application and should remain under consideration pending the acceptance of a common LRB/ALS program.

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 |

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*Figure 2-3. Concept evaluation summary.*

# 3 ✦ VEHICLE DEFINITION

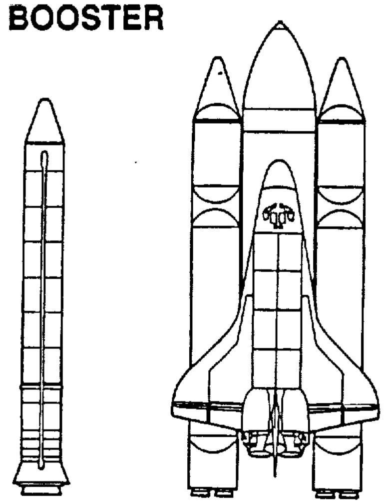
The $LO_2/LH_2$ pump-fed configuration was recommended as the best final concept for the LRB (see Figure 3-1). This concept offers low technical risks, minimal environmental impacts, propellant commonality with the current STS, and most importantly, commonality with ALS concepts. The $LO_2/LH_2$ LRB life cycle cost can be significantly reduced by the ALS sharing of DDT&E costs in engine or booster development and production rate effects.

The engine selected as the baseline for this vehicle concept is the $LO_2/LH_2$ gas-generator cycle engine. The split-expander cycle engine is an alternative that seems promising in reducing costs and improving reliability. 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.

## 3.1 INCREASED PERFORMANCE CAPABILITIES

From the start of the study our goal has been to build robust performance capabilities into LRB. Even though our design requirement of 70.5 Klb to 160-nmi altitude is above the landing weight capability of 58 Klb, LRBs permit delivery of 65 Klb to the Space Station at 220 nmi.

**THE STS LIQUID ROCKET BOOSTER**



**Features**

- LH2/LO2 propellants
- 2219 aluminum tankage
- New low-cost, pump-fed engines
- 4 engines per booster
- Expendable (engines may be recovered)
- Existing technologies

| | SRB | LRB |
|---|---|---|
| Length (ft) | 149 | 178 |
| Diameter (ft) | 12.2 | 18 |
| Booster dry weight (lb) | 146,000 | 122,000 |
| Booster gross weight (lb) | 1,250,000 | 821,000 |
| Engine thrust at sea level (lb) | 2,912,000 | 4 × 515,000 |

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*Figure 3-1. Summary of selected $LO_2/H_2$ vehicle with new pump-fed engine.*

weight capability of 58 Klb, LRBs permit delivery of 65 Klb to the Space Station at 220 nmi. Studies also indicated that in order to provide high reliability, the LRBs require engine-out capability. Therefore, the LRBs are sized to achieve a safe orbit (abort-to-orbit) with one LRB engine out from liftoff. The performance capability relative to the current revised solid rocket motors (RSRMs) is shown in Figure 3-2. Performance analysis also indicated that the large performance capabilities:

- Enhance abort capabilities with large payloads
- Enable SSMEs to operate at nominal 100% instead of 104% throttle (reduces engine overhaul and extends life)
- Eliminate boost stage SSME throttling requirement and reduce engine critical failure modes
- Increase mission planning flexibility (standardized trajectories)

### 3.2 IMPROVED ABORT CAPABILITIES

One of the first objectives of the LRB program is to improve Shuttle safety by expanding and improving current abort coverage. There is currently no abort option for solid booster failure during ascent. The LRB, however, is designed to have hold-down capability on the pad and engine-out capability during ascent to protect against booster engine failures. LRBs will also

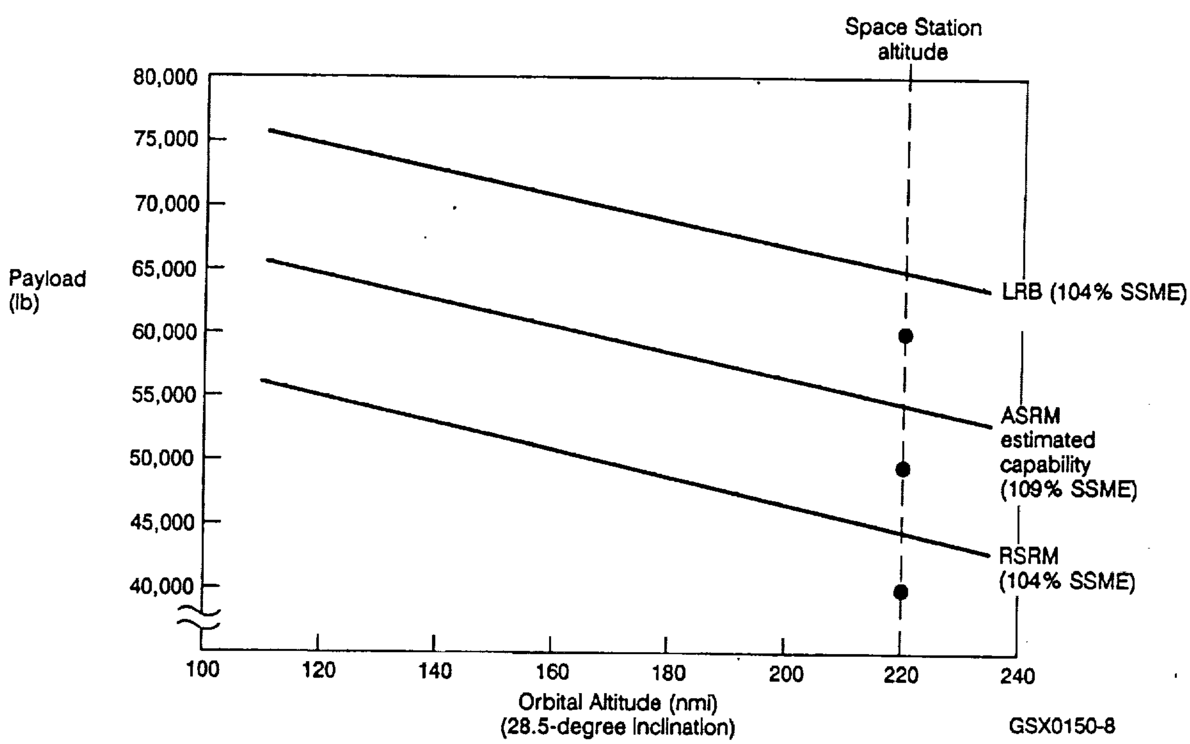



*Figure 3-2. LRB performance comparison.*

improve existing abort modes by making use of a liquid rocket's inherent capability to throttle or shut down on command and the increased performance margin included in the design.

As shown in Figure 3-3, the LRB will improve and expand engine failure abort modes. Perhaps most importantly, the health of the LRB engines can be verified on the pad before the vehicle is released. Even if a booster engine should fail on one or both LRBs following release, the Orbiter is still able to abort to orbit (ATO), that is, still able to achieve a safe earth orbit only slightly below the target. This engine-out capability is a significant improvement over solid boosters, which have no recourse should a booster fail.

Current intact abort modes for a single SSME failure during ascent include return to launch site (RTLS), a transatlantic abort landing (TAL), and an ATO. RTLS and ATO aborts are preferred because of their quicker accessibility and lower turnaround time. The added performance of LRBs will allow ATO and RTLS aborts to be initiated much sooner in the flight than currently available. RTLS and ATO windows can be made to nearly overlap, thereby reducing the current dependence on TAL to fill the gap. LRBs can also provide a more favorable environment for multiple Orbiter SSME failures.

LRBs can greatly improve the chance of recovering the crew and the Orbiter. In the case of multiple SSME shutdowns, the LRBs may be immediately throttled down to moderate the attach loads and can use engine gimbaling to reorient the Orbiter on a glide path to return to

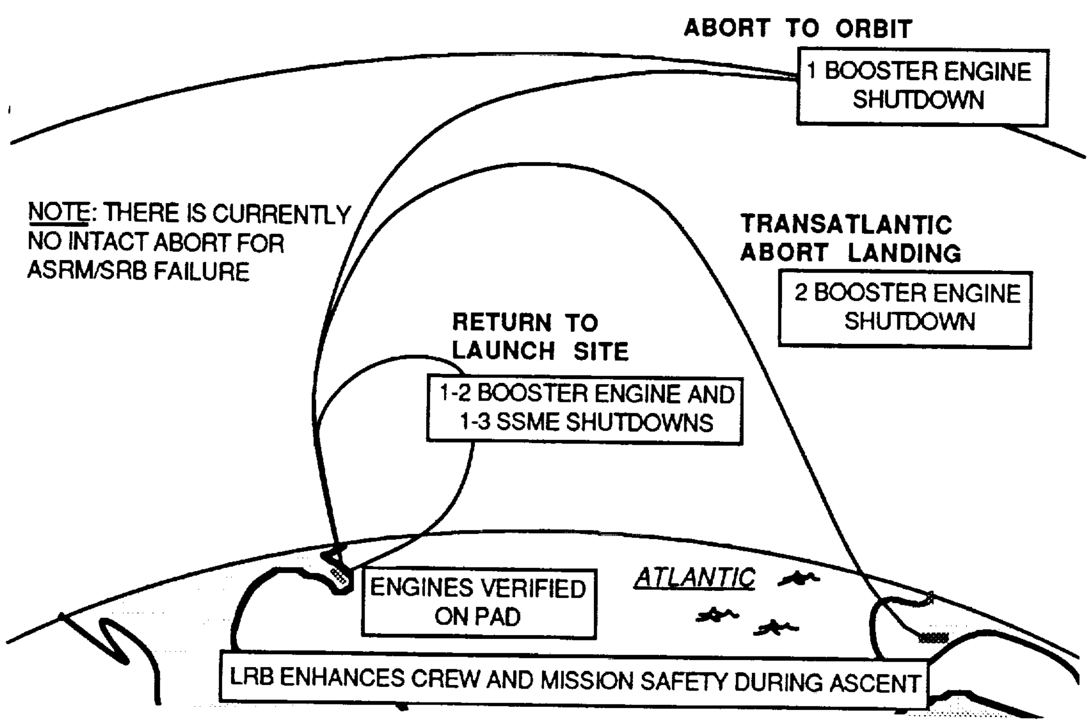



*Figure 3-3. First-stage contingency abort analysis.*

land at KSC. Should a major booster failure occur, the capability of LRBs for commanded premature shutdown followed by a fast Orbiter separation from the LRBs and ET will allow the Orbiter to perform a wide banking turn that will return it to the landing site. With the current SRBs, such a fast separation could not even be initiated until after booster burnout and separation, at which time the Orbiter would be too far downrange.

At present, the only option for multiple SSME failures after 20 seconds into the flight is to perform a downrange ditch after booster burnout (assuming that attach points do not fail). As mentioned before, the ability to shut down the LRB engines may allow earlier fast separation of the LRBs from the stack during an emergency. This ability will also improve the downrange ditch abort mode by allowing separation of the LRBs while continuing the thrusting of the SSMEs until an Orbiter reorientation can be accomplished. At present, the only option for multiple SSME failures after 20 seconds into the flight is to perform a downrange ditch after booster burnout. Currently, a risky maneuver called the Split-S could be attempted, should multiple SSMEs fail prior to 20 seconds into the flight. The Split-S involves lofting the trajectory nearly straight up until SRB burnout, which occurs at around 150,000 ft. The Orbiter would return to land at KSC after performing a very high g pullout turn. LRBs will allow a Split-S to be performed even up to 40 seconds into the flight.

The safer new contingency options introduced by the LRB (discussed above) will made the downrange ditch and the Split-S last-resort choices.

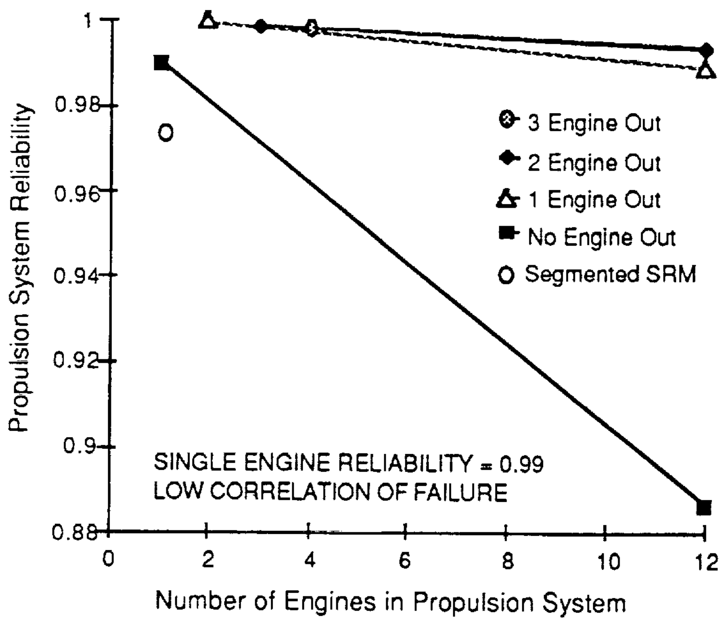
To summarize, current intact abort modes have been improved and the option to ATO is available sooner because of the performance increase designed into the LRB. Two very important booster features that did not previously exist with SRBs — hold-down and engine-out capabilities — have been included in our designs. Finally, due primarily to the ability to throttle and shut down the LRB engines in real time, current contingency aborts will be made safer and two new methods for returning the crew and Orbiter to the launch site in the event of emergency will be possible.

### 3.3 BOOSTER ENGINE SELECTION

We believe that LRBs must have mission reliability superior to SRBs if the program is to be viable. The Shuttle crew must be able to safely perform an intact abort if one LRB engine fails. Figure 3-4 illustrates that high propulsion system reliability can be achieved with engine-out capability. By comparison with these theoretical numbers, large segmented solid rocket motors have an actual flight reliability of 0.9765 (based on one Titan failure in 174 flights and one STS failure in 50 SRB flights). Qualitatively, more booster engines reduces the decrease in thrust drop if one engine fails, but increases the chances of failure. On the other hand, fewer

• RELIABILITY SHOULD BE BASED ON AFTER IGNITION & TRANSIENT EFFECTS (BECAUSE OF HOLD-DOWN CAPABILITY OF LRB)

• SAFETY & RELIABILITY ARE IMPROVED WITH ENGINE-OUT CAPABILITY




| SEGMENTED SRM DATA | | |
|---|---|---|
| VEHICLE | NO OF FLIGHT /FAILURE | RELIABILITY WITH 90% CONFIDENCE LEVEL |
| TITAN | 174/1 | 0.9775 |
| STS | 50/1 | 0.921 |
| TOTAL | 224/2 | 0.9765 |

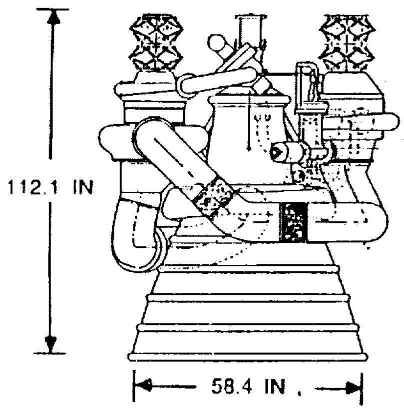
GSX0150-11

*Figure 3-4. Reliability vs. number of engines.*

thrust drop if one engine fails, but increases the chances of failure. On the other hand, fewer engines requires that each engine have a larger thrust, and throttling capability, and wider gimballing range. We found that two engines provides a minimum of redundancy, but with a large thrust loss per failure. In contrast, if six engines are used the thrust loss per engine is smaller, but the probability of failure is higher, and the plumbing complexity and ground checkout is increased. Therefore, we selected four engines as the best compromise between reliability, controllability, and throttling range.

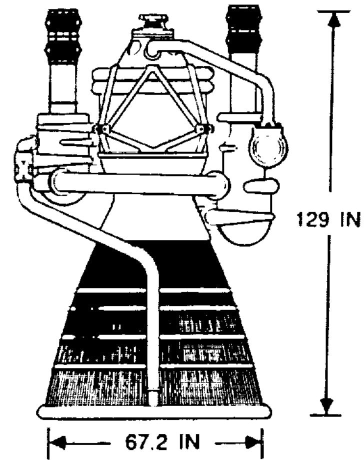
## 3.4 MAIN PROPULSION SYSTEM

The characteristics of the baseline LRB $LO_2/LH_2$ gas generator pump-fed engine and the alternate split expander engine configuration are shown in Figure 3-5. The engines were sized to meet the requirement of abort to orbit with one LRB engine out, a minimum thrust-to-weight at launch to clear the tower with one engine out, and nominal thrust to weight ratio at launch for minimum cost. The design characteristics of the selected system are virtually identical to the present Space Transportation Main Engine (STME) configuration now being studied by Rocketdyne and Pratt & Whitney under a separate STME/STBE contract for NASA and the ALS program.




**BASELINE**

- **Pc = 2250 psia**
- **Tvac = 558 K Lbf**
- **AR = 20:1**
- **$Isp_{vac}$ = 411.4**




**ALTERNATE**

- **Pc = 968 psia**
- **Tvac = 563.9 K Lbf**
- **AR = 10.6:1**
- **$Isp_{vac}$ = 409.5**

| ISSUE | REQUIREMENT | RATIONALE |
|---|---|---|
| NUMBER OF ENGINES | 4 | BEST RELIABILITY, CONTROL & THROLLTLE COMPROMISE |
| TYPE | EXPENDABLE | LOWER DDTE COST & RISK |
| CYCLE | BASELINE: GAS GENERATOR<br>ALTERNATE: SPLIT EXPANDER | MATURE TECHNOLOGY<br>POTENTIAL COST BENEFITS & BENIGN FAILURE MODES |
| MR | 6 | LOW TECHNICAL RISK |
| CHAMBER PRESSURE | 2250 PSIA (GG) | LOWER COST & COMPLEXITY |
| BOOSTER PUMPS | NONE | LOWER COST & COMPLEXITY |
| BLEEDS | NONE | OPERATIONAL SIMPLICITY |
| THROTTLING | STEP | LOWER COST & HIGHER RELIABILITY |
| CONTROL | OPEN LOOP | LOWER COST & HIGHER RELIABILITY |
| EXPANSION RATIO | 20 (GG) | LOWEST VEHICLE LIFE CYCLE COST |
| INLET PRESSURE | LO2 = 65 PSIA; LH2 = 45 PSIA | LOWER COST & SMALLER TURBOPUMPS |
| GIMBAL SYSTEM | EMA | OPERATIONAL SIMPLICITY |
| GIMBAL ANGLES | ± 6 DEG; SQUARE | CONTOL CONSIDERATIONS |

*Figure 3-5. Selected $LO_2/LH_2$ engine requirements and features.* GSX0150-12

Engine performance and cycle type were studied and balanced for the configurations, and the resultant parameters were used to establish the pertinent combustion chamber, injector, nozzle, and turbopump characteristics that led to the recommended configuration and physical design. The integration of the engine and other propulsion features is shown in Figure 3-6. The smaller size of the nozzles for the $LO_2/LH_2$ concept permits a straight skirt design and therefore fewer modifications of the launch platform.

### 3.5 STRUCTURAL DESIGN CONSIDERATIONS

The inboard profile and aft skirt and engine features, shown in Figures 3-7 and 3-8, identify the significant features of the booster structure and systems. The basic structure consists of two propellant tanks, nose cone, intertank adapter, and engine compartment skirt. The forward tank contains $LO_2$, the aft tank $LH_2$. The engine compartment comprises skirt shell, thrust, and launch support structures. All aluminum materials were selected based on established manufacturing practices and technologies, resulting in low risk and cost.

Both propellant tanks are of integral skin stringer construction, with internal ring frames. The tank material is 2219 aluminum alloy and all external seams are welded. This type of construction is fail-safe, since it provides crack tear stopper and alternate load paths. Aluminum/lithium alloys were considered for tankage but not recommended because of questions about $LO_2$ compatibility and higher cost.

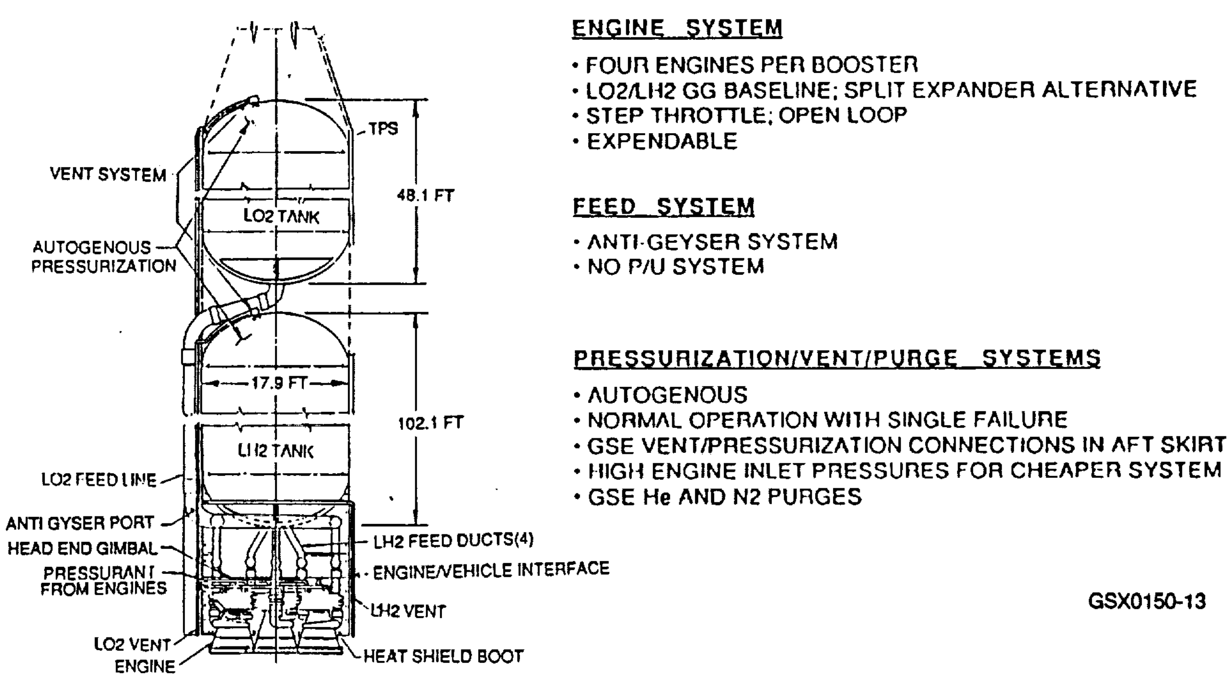



*Figure 3-6. $LO_2/LH_2$ propulsion design integration features.*

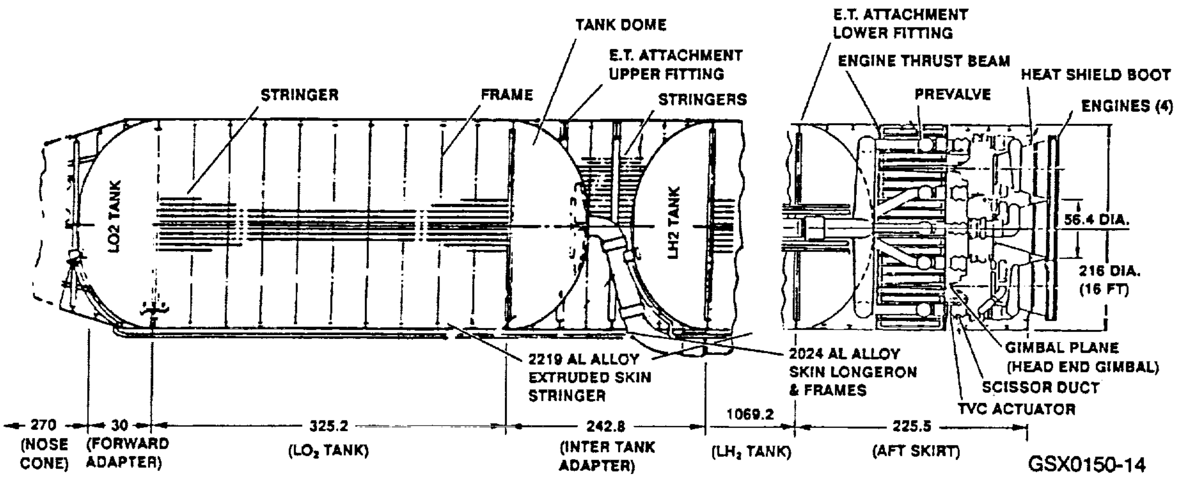



*Figure 3-7. Inboard profile.*

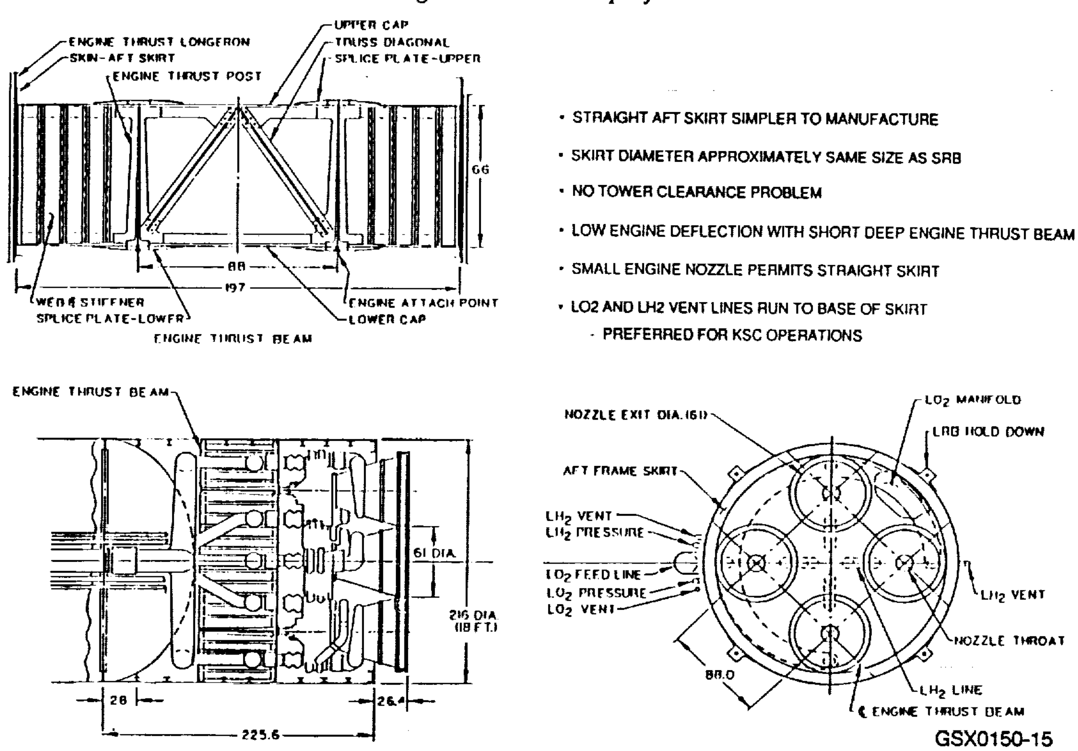



*Figure 3-8. Aft skirt and engine features.*

The intertank adapter is a skin stringer with ring frame construction. The upper attachment (thrust fitting) to the external tank is located in this section. The external tank thermal contraction loads are minimized by designing the LRB attachment frame so it can deflect and relieve the load.

The aft skirt shell structure consists of skin, frames, hold-down members, and engine thrust beam members. The thrust beams are short and deep to maintain low deflections.

The structural design of the LRB was also influenced by the desire to:

- Minimize impact on ET structure during the prelaunch and ascent flight trajectory
- Keep impacts to ground support equipment at the launch pad to a minimum

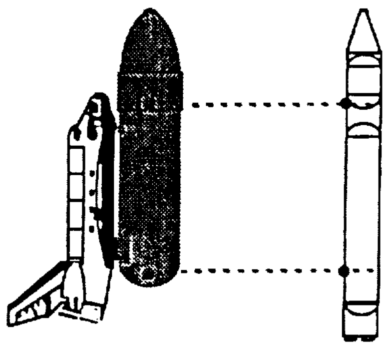
The $LO_2/LH_2$ vehicle has the lowest weight and thrust of all vehicles studied, and therefore requires the lowest overall attach loads. The structural analysis performed indicated that the attach loads were considered to be within current limits and therefore, as shown in Figure 3-9, no changes to the existing ET are required. Load analysis was also used to design the LRB structure. For example, the loads that occur during launch (SSME thrust buildup, causing swaying of stack) primarily governed the strength design of the aft skirt and the aft $LH_2$ tank, while the $LO_2$ tank design was primarily governed by the loads during conditions at Max-g.

Another design consideration was the $LH_2$ cryogenic tank shrinkage during fueling. The shrinkage results from loading the attach struts between the LRB and the ET, while the LRBs are still bolted to the pad and support the entire stack. Figure 3-9 illustrates the amount of shrinkage expected in the ET and LRB $LH_2$ tanks. The preliminary analysis results indicate the 18-ft-diameter tank of the LRB can accommodate the strut loads by local shell deflection, which actually results in lower strut loads than those currently experienced with the less flexible steel case of the SRB.

In order to minimize impacts to the existing ground equipment such as the ET umbilicals, structures were designed to maintain minimum booster bending deflection while the stack is still held down and the SSMEs are fired. A stiffness criterion of .24 Hz first bending frequency was derived for this purpose. Figure 3-10 shows what is needed in wall thickness versus diameter to maintain this minimum stiffness. Boosters below 16.5 feet in diameter incur a penalty in weight as the thickness is increased to improve stiffness. The $LO_2/LH_2$ vehicle with a diameter of 18 feet incurs no weight penalty to maintain this required stiffness.

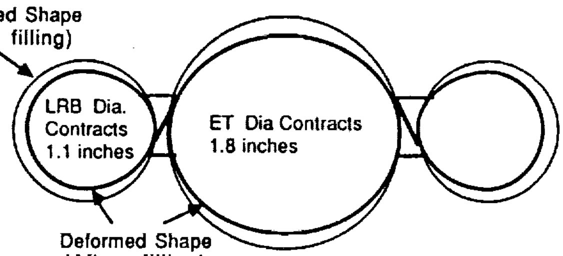
the system, data going to the Orbiter avionics and commands coming from the Orbiter avionics can be manipulated to ensure consistency with the present Orbiter data protocol.

**USE EXISTING ET INTERFACE ATTACH POINTS**
**MINIMIZE ET INTERFACE MODIFICATIONS**




| Loads | LOW THRUST, LOW WEIGHT LOWEST ATTACH LOADS |
|---|---|
| Fwd Attach Fitting | IN UNPRESSURIZED INTERTANK AREA |
| Aft Attach Struts/Frame | ATTACHED TO LH2 TANK TANK SHRINKS BUT ATTACH LOADS STAY WITHIN ET LIMITS |

- LRBs are fixed to MLP
- Both ET and LRB LH2 tanks shrink during fueling
- Tensile loads develop in the aft attach struts
- STS-26 Strut Loads measured prior to lift-off small compared to limit loads :
  40 KLBS upper and lower struts, 62 KLBS diagonal struts
- Limit loads :
  265 KLBS lower strut, 393 KLBs diagonal strut
  332 KLBS upper strut
- LRB shell stiffness is lower than SRB - larger diameter ,lower wall thickness , Aluminum structure
  Cross-section tends to deform into egg shape
  Relieves Strut Loads
  Lower loads on ET




**LRB Aft Attach Loads are lower than SRBs** GSX0150-17

*Figure 3-9. Structural interfaces.*

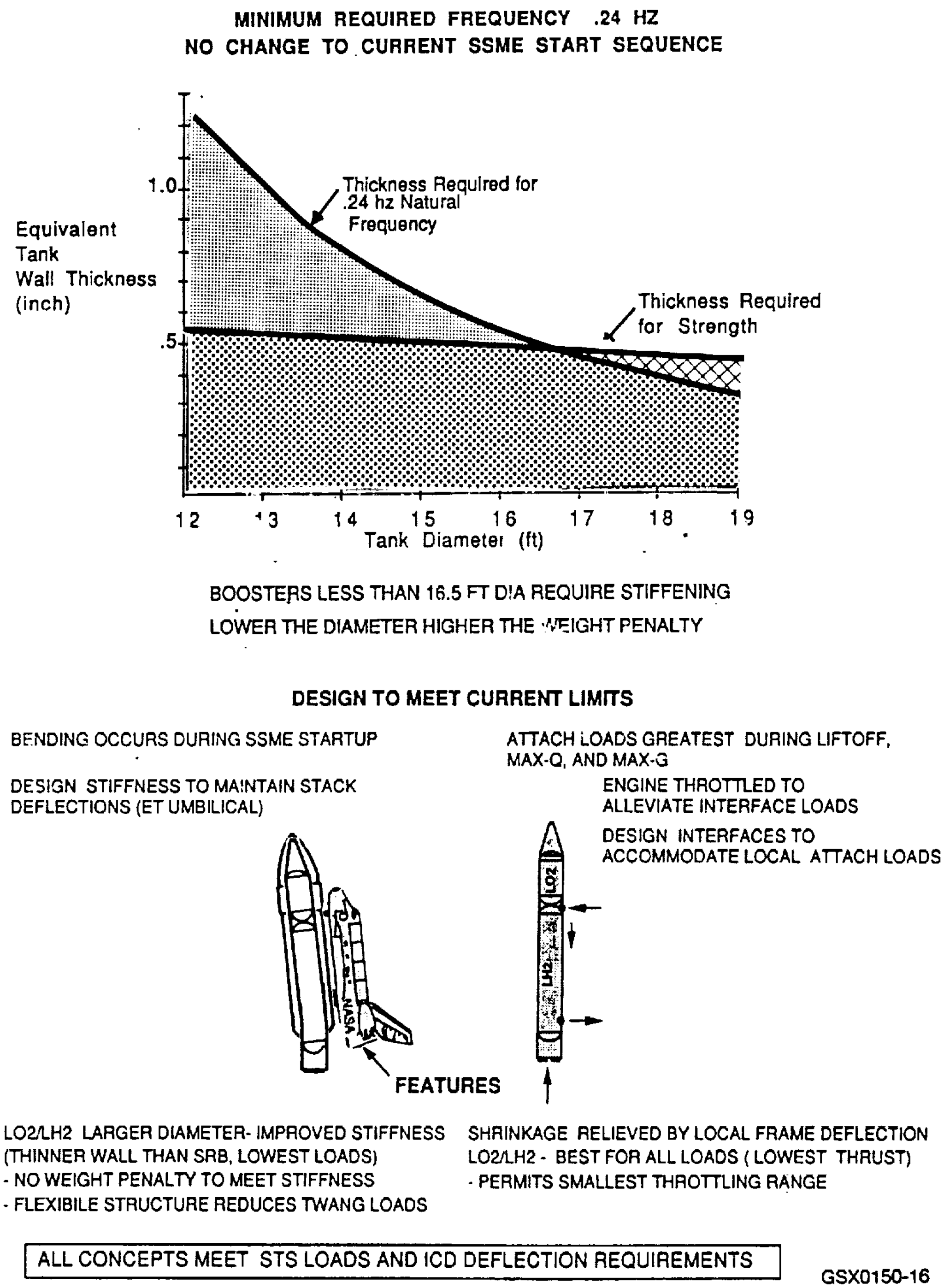



*Figure 3-10. Structural design considerations.*

## 3.6 AVIONICS SYSTEMS ARCHITECTURE SELECTION

The top objectives for the LRB avionics are to improve STS system reliability, while minimizing Orbiter software and hardware impacts. The system had to provide command and monitoring capabilities for the more complex liquid engines and subsystems, and reduce ground operations and support requirements, while minimizing development cost and risk.

The architecture adapted as the baseline is illustrated in Figure 3-11. This system provides a redundant three-string system to achieve the large reliability needed for the STS mission. In addition, each LRB uses its own RF telemetry system, thus supporting the increased telemetry requirements for the more complex LRB. By incorporating data processing capabilities into the system, data going to the Orbiter avionics and commands coming from the Orbiter avionics can be manipulated to ensure consistency with the present Orbiter data protocol.

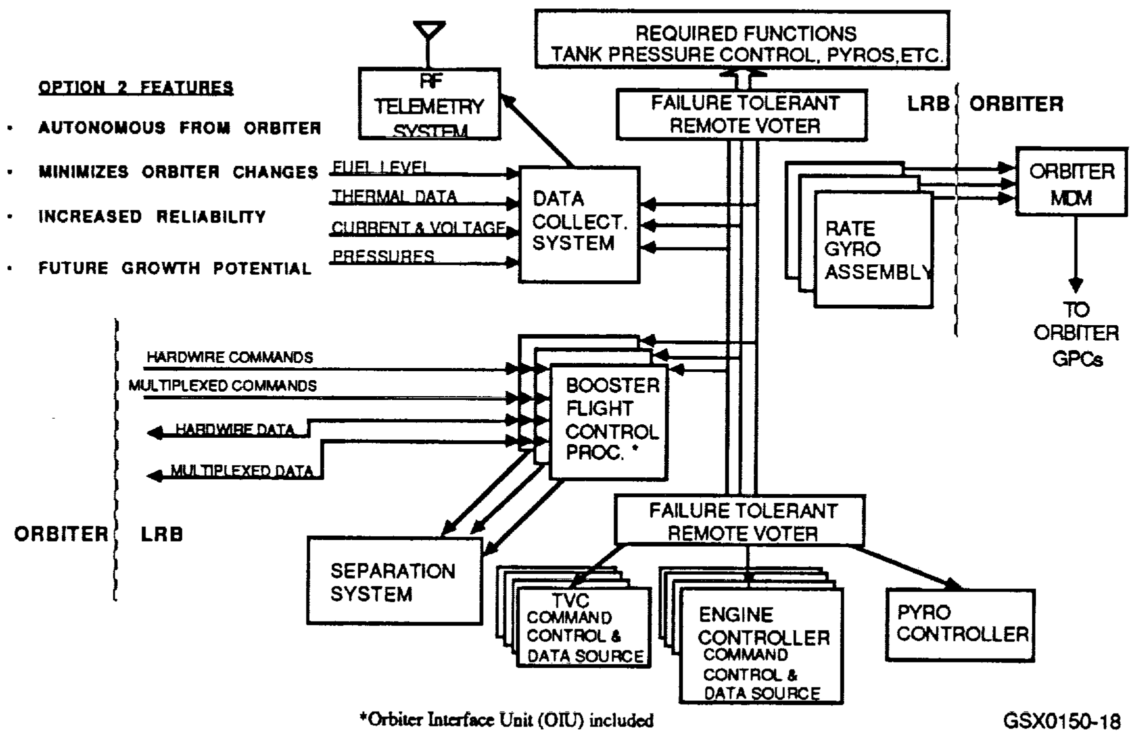



*Figure 3-11. Baselined LRB avionics architecture.*

# 4 ✦ TRADES AND ANALYSES

During the first six months of the study, our engineering effort concentrated on trades and analyses. Twelve of these were formally reviewed by an internal engineering review board (ERB) at inception, midpoint, and conclusion. In addition to providing a consistent philosophy and ground rules, these meetings were an opportunity to discuss interrelated findings among the LRB study team. There have also been a number of detailed trades late in the study.

Three central questions on LRB concepts (summarized in subsections 4.1 through 4.4 and Section 7) are:

- Which propellants best suit LRB?
- Should LRB be reusable?
- Which concepts best support alternate applications such as ALS?

Table 4-1 summarizes the results of these major trades. These issues are key to the LRB concept selection process. After the early trades and analyses had eliminated some concepts, we focused on three that could meet the requirements. In addition to the recommended $LO_2/LH_2$ concept, $LO_2$/RP-1 pump-fed and pressure-fed concepts are viable. The pressure-fed concept is and will continue to be too high a risk until the technology is demonstrated as planned in the Civil Space Technology Initiatives (CSTI) program under Booster Technology. Once updated technology (combustion efficiency, chugging, pressurization, etc.) allows, the pressure-fed concept might be an acceptable candidate. At this time we recommend an $LO_2/LH_2$ pump-fed LRB for STS and ALS, and development of one major rocket engine for both programs. The following trades and analyses lead up to and support this concept selection.

*Table 4-1. Major trade study results.*

| Trade | Recommendation | Rationale |
|---|---|---|
| Propellants | $LO_2/LH_2$ has least environmental impact. LOX/RP-1 and $LO_2/CH_4$ acceptable | Storable propellants in such large quantities are too risky |
| Geometry | Cylindrical shape with diameters up to 18 ft | Clocking, hammerheads, or tandem tanks are too complex. New aero data 18 ft acceptable |
| Reusability | Expend LRBs for flight rate < 15/year. Reconsider limited engine reuse | Increased development cost and reduced ascent reliability not justified at expected rates |
| Engine type | Pump-fed gas generator; split-expander cycle alternative | Pressure-fed concepts offer great potential, but need technology development |

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Table 4-2 summarizes the numerous trades used to establish the configurations and refine the concepts.

*Table 4-2. Concept refinement trade studies.*

| Trade | Recommendation | Rationale |
|---|---|---|
| **Pressure fed** | | |
| Chamber pressure | 334 psi | Minimum weight with acceptable combustion stability |
| Pressurization | Helihyox* with catalytic heating | Simplicity and minimum weight |
| **Pump fed $LO_2/LH_2$** | | |
| Number of engines | 4 on each LRB, with engine out capability | 3 require large throttling range, 5 or 6 have lower reliability |
| Chamber pressure | 2250 psi | Minimum cost vehicle |
| Expansion ratio | 20 | Minimum life cycle cost |
| Initial T/W | 1.2 with 1 engine out | Minimum thrust = minimum cost and still clear tower |
| Throttling | Two settings: 100% & 75% | Less expensive than continuous throttling |
| **Structure** | | |
| Stiffness | Equivalent to SRB including thrust structure, and larger drain | SSME ignition stagger too large an STS impact |
| Material, construct | 2219 ring-stringer | Stiffer and lighter at same cost, 2090 Al-Li not $LO_2$ compatible |
| **Avionics** | | |
| Architecture | Modified triple string | High reliability |
| Thrust vector control | 20 H.P. electro-mechanical actuators | Simplify KSC operations |

*Helium (He) mixed with small amounts of hydrogen ($H_2$) and oxygen ($O_2$) in non-flammable quantities

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## 4.1 LRB PROPELLANT SELECTION

Data from this trade study was a major element in concept selection. Initially, propellant density was clearly a driver; low-density hydrogen resulted in very large LRBs, whereas storable propellants allowed LRBs nearly as small as the current SRBs. Table 4-3 summarizes our initial propellant screening.

*Table 4-3. Initial propellant screening.*

| | |
|---|---|
| Storables | – Eliminated due to environmental and safety concerns |
| Metalized fuels | – Eliminated due to advanced technology status |
| Tripropellants | – Eliminated due to vehicle and facility complexity |
| Propane | – Eliminated. Preferred methane, which has better reusability and less spill hazard due to lighter vapor |

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Table 4-4 shows the characteristics of the major pump-fed propellant groups with respect to key selection criteria. Thus far, no substantial discriminators exist between propellant systems concerning launch facility compatibility. Consequently, that criterion is not shown.

*Table 4-4. Pump-fed propellant system comparison.*

| Criteria | $LO_2/LH_2$ | $LO_2$/HC | LO2/HC/$LH_2$ | NTO/hydrazines |
|---|---|---|---|---|
| **Safety** | • Detonation hazard | • Good | • Detonation hazard | • Detonation hazard, hypergolic<br>• Toxic, corrosive, carcinogen |
| **STS Compatibility** | • Large LRB<br>• Existing tankage system<br>• Additional storage required | • New HC system, not a major impact | • 3 storage transfer, monitoring systems | • Smallest LRB<br>• New facility system major impact |
| **Risk** | • Low: Extensive experience & established data base | • Low ($LO_2$/RP-1)<br>• Medium ($LO_2$/CH4 & $LO_2$/C3H8 | • Highest engine tech. & schedule risk | • High, due to safety/environmental protection reqts<br>• Extensive experience & established database<br>• Limited propellant production capability |
| **Operational Complexity** | • More complex than $LO_2$/HC | • Simple | • Complex engine servicing/cycle<br>• Complex LPS control | • Slow & complex launch process due to safety<br>• Slow & complex recov. and return |
| **Environmental Impact** | • Benign exhaust products | • Medium, $CO_2$ exhaust affect ozone | • Medium<br>• $CO_2$ exhausts affect ozone | • High, in case of accident |

GSX0150-21

The $LO_2/LH_2$ system is physically the largest, and therefore may have more impacts on the STS and launch facilities (although thus far, no major problems have been uncovered). On the positive side, $LO_2$ and $LH_2$ have the most environmentally favorable exhaust products, are existent within the STS, and result in the lightest weight vehicle.

At the other extreme are the storables, which pose serious safety and environmental hazards but are small and simple to operate. The $LO_2$ hydrocarbon systems are reasonably compact, are operationally simple, and exhibit good safety and environmental characteristics. The $LO_2$/RP-1 system, like $LO_2/LH_2$ and storables, operates routinely on launch vehicles today.

The tripropellant combinations are reasonably compact, but the introduction of a third propellant and the need to develop a new type of engine results in high program risks.

As a group, the pressure-fed propellant systems, with the exception of those with metal additions, are larger than the pump-fed (see Table 4-5). $LO_2/LH_2$ was found to be excessively large and heavy, and therefore is not being considered in this comparison.

Of course, comments from Table 4-4 pertaining to the propellants themselves, such as safety and environmental issues, apply equally here.

*Table 4-5. Pressure-fed propellant system comparison*

| | $LO_2$/HC | NTO/Hydrazine | Metallized/gel |
|---|---|---|---|
| **Safety** | • Good | • Detonation Hazard Hypergolic<br>• Toxic,carcinogen corrosive | • Better than storables |
| **Reliability/STS compatibility** | • Lower than storables<br>• Large booster size<br>• ($LO_2$/RP1 < $LO_2$/C3H8 < $LO_2$/CH4<br>• New HC tanking system | • Highest simple propulsion sys<br>• Smaller booster size than $LO_2$/HC<br>• New facility system | • Lower than storables (under study)<br>• Booster similar to SRB<br>• New facility system |
| **Risk** | • Low schedule risk<br>• $LH_2$/RP1 and $LO_2$/C3H8 engines tested, higher tech. risk than hpergols | • Higher schedule risk than $LO_2$/HC<br>• Low technical risk<br>• Engine tested | • Technological issues still unresolved<br>• Highest technological and schedule risk |
| **Operational complexity** | • Simple (Best RP-1) launch processing | • Complex launch processing due to safety | • TBD |
| **Environmental impacts** | • Medium, $CO_2$ exhaust affect ozone layer | • High (In case of accidents) | • High for hypergol gels<br>• Medium for $LO_2$/RP-1 gels |

GSX0150-22

Metallized propellants are the only systems that offer the potential of LRBs that are equal to or smaller than the present SRBs. Many unresolved operational and technological issues remain. Further development is required prior to their introduction.

Liquid hydrogen was a controversial choice from the beginning. At first look, it had major advantages: no environmental impact from a spill or exhaust products and high $I_{sp}$. The hydrogen-burning SSME is the only engine available in the half-million-pound thrust class if LRB were needed very soon. But there were serious concerns that hydrogen LRBs were too large and that the wider diameter would cause excessive aerodynamic loads on the Orbiter wing and/or that long lengths (with $LO_2$ tank forward) would cause excessive loads into the ET interfaces. Numerical data on these loads was not available until recently. They are manageable. Secondary criteria such as cost consequently became more important. Liquid hydrogen vehicles tend to be more expensive than $LO_2$/RP vehicles based on General Dynamics' long experience with Centaur and Atlas. There is still a strong interest on the ALS program in $LO_2/LH_2$ systems, and commonality between ALS and LRB should yield cost savings. Therefore, we continue to recommend $LO_2/LH_2$ as an LRB candidate.

## 4.2 REUSABILITY

This was a very broad trade study that compared the technical merits and cost effectiveness of about 15 recovery concepts. These options included full recovery (appropriate for pressure-fed LRBs whose tanks are strong enough to survive water impact such as an SRM), partial recovery (for expensive engines like SSME), and no recovery. Flyback boosters were felt to be candidates only for growth options because of their major impact on current STS compatibility. Work concentrated on water recovery using parachutes or parawings to slow the descent. Our subcontractor, Pioneer, provided data on these deployable aerodynamic devices.

The downselection process identified two recovery concepts that we considered the most desirable on a technological basis: the downrange parachute recovery of a pressure-fed LRB and the RTLS propulsion/avionics (P/A) module parawing towback of a pump-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 and therefore should provide the most recoveries. The parawing towback concept of the P/A module for RTLS recovery presents the lightest parawing recovery system. The water landing system is considerably lighter than airstrip or platform landing, which in turn reduces complexity and risk, and enhances reliability. The LRB concepts were upsized to accommodate the added weight of recovery devices, sea water sealing systems, disconnects, etc.

Rocketdyne and our ALS program then developed the idea that expendable engines should demonstrate inherent life of at least five flights during qualification testing. To apply this concept to LRB, a low-cost engine recovery module was examined, as shown in Figure 4-1. The $LO_2/LH_2$ LRB was upsized about 3% to accommodate about 9,000 lb of separation and recovery systems. We recommend further study and support of the ALS Booster Recovery Module ADP. Until this data is available, we continue to recommend that the LRB be expendable.

The cost effectiveness of recovery is dependent upon three factors: attrition (number of units lost due to high seas, bad parachutes, etc.), cost to refurbish, and DDT&E.

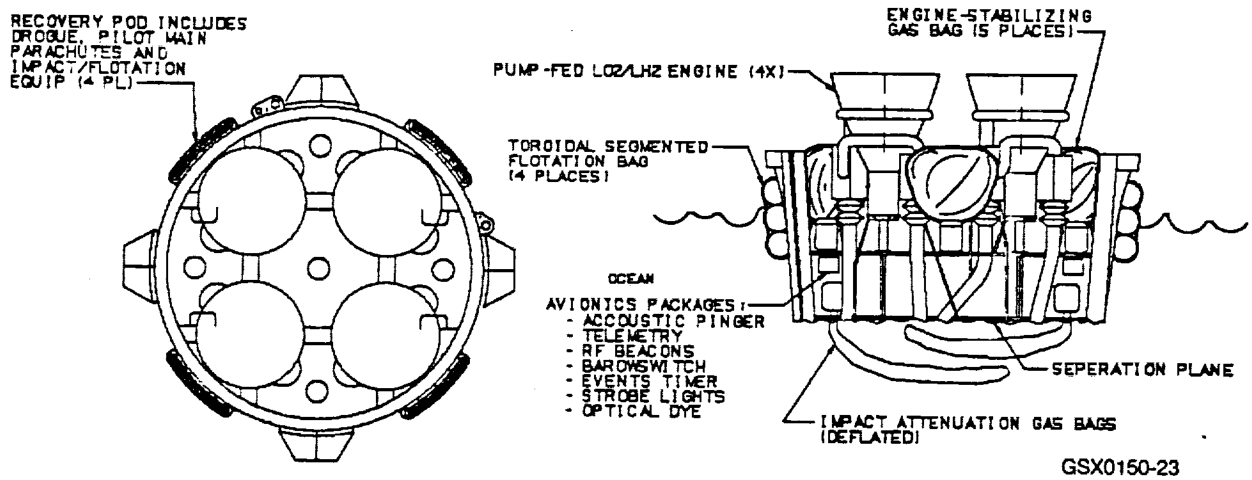


*Figure 4-1. Low-cost engine recovery module concept for LRB.*

Cost sensitivities determined that the cost effectiveness of recovery is very sensitive to the following three issues:

- The additional complexity of designing and manufacturing a component for extended man-rated life vs. one flight
- The assumed recovery reliability
- The cost required to refurbish an item that has been successfully flown and recovered

Trade studies indicated that parachute recovery was more cost-effective than towback, tossback, or flyback on a LCC basis. The parachute recovery concepts were also less costly on a LCC basis than their respective expendable concepts, although the magnitude of these estimated cost savings is considered marginal and nearly disappears when a 3% discount factor is applied.

For the $LO_2/LH_2$ concept, we traded expendable vs. two recoverable concepts based on recovering only the engine. The recoverable options were a limited-life engine that has the inherent ability to perform five missions (four reuses) and an engine designed for a life of 20 missions. The engine was chosen as the reusable element because a shipset of engines represents approximately 40% of the average unit cost of an expendable, pump-fed $LO_2/LH_2$ vehicle.

In completing the expendable vs. recoverable trade study a number of engine assumptions were made, as shown in Table 6-3. This table indicates that the engine DDT&E cost for the recoverable limited-life engine and recoverable engine with life of 20 missions is estimated to cost 10% and 52% more than the expendable engine, respectively. In addition, the cost of refurbishing a recoverable engine was assumed to be 25% of the engine TFU cost.

*Table 4-6. Expendable vs. recoverable engine assumptions.*

| | EXPENDABLE ENGINE | RECOVERABLE (LIMITED LIFE ENGINE) | RECOVERABLE (ENGINE LIFE = 20 MISSIONS) |
|---|---|---|---|
| NUMBER OF ENGINE USES | 1 | 5 | 20 |
| ENGINE COST COMPARISON | | | |
| DDT&E | 1.00(Ref.) | 1.10 | 1.52 |
| PRODUCTION(TFU) | 1.00(Ref.) | 1.10 | 1.62 |
| % OF ENGINE TFU FOR REFURBISHMENT | N/A | .25 | .25 |

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Based on these engine assumptions, a comparison of the LCC of the expendable vs. the two recoverable options was completed as shown in Figure 6-2. Note that the crossing point for the LCC of the recoverable vehicle with limited-life engine and the expendable vehicle occurs before 50 LRB flights. At this point the recurring savings of the recoverable vehicle with limited-life engine are large enough to offset the additional DDT&E required for the recoverable vehicle over the expendable vehicle. At 14 STS missions per year (244 LRBs) the LCC of the expendable vehicle is estimated to be about 7% less than the recoverable with engine life of 20 missions and about 8% more than the recoverable vehicle with a limited-life engine. In addition, the LCC comparison does not appear to be very sensitive to the number of STS missions per year (between 6 and 25).

The cost required to refurbish an engine is not well understood. As indicated in Figure 6-3, we have assumed a refurbishment factor of 25% for the pump-fed $LO_2/LH_2$ engines. This sensitivity analysis indicates that an engine refurbishment factor of 10% would be needed to make the recoverable vehicle with engine life of 20 missions cost-competitive with the expendable

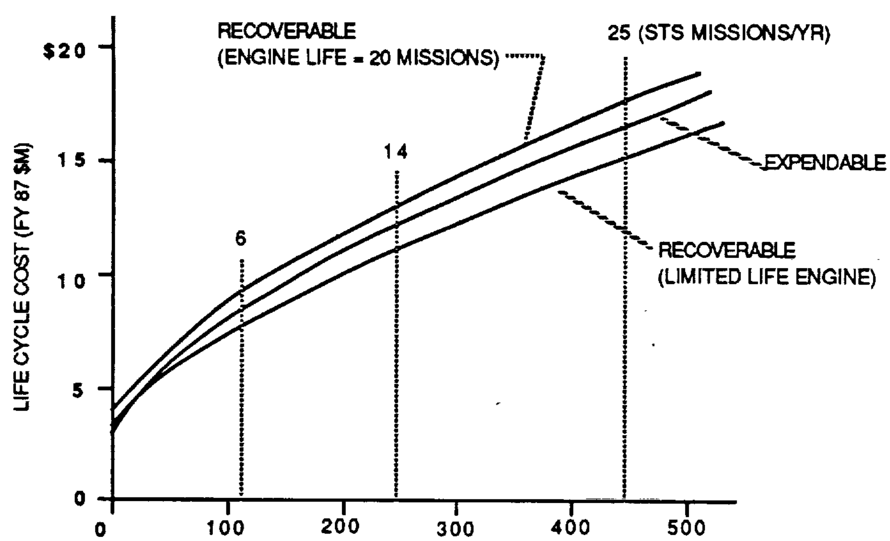


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*Figure 4-2. Expendable vs. recoverable LCC comparison.*

LRB. The likelihood of attaining a 10% refurbishment factor is quite low. In addition, the engine refurbishment factor could increase to 50% before the LCC of the recoverable vehicle with limited-life engine approaches that of the expendable vehicle. Although the engine refurbishment factor could be greater than 25%, we feel it is unlikely that it will approach 50%.

It is equally important to understand the sensitivity of cost to recovery reliability. A 90% recovery reliability was assumed in our trade study analysis, as shown in Figure 6-4. This figure addresses the difference in the sensitivity of cost to recovery reliability for the two recoverable

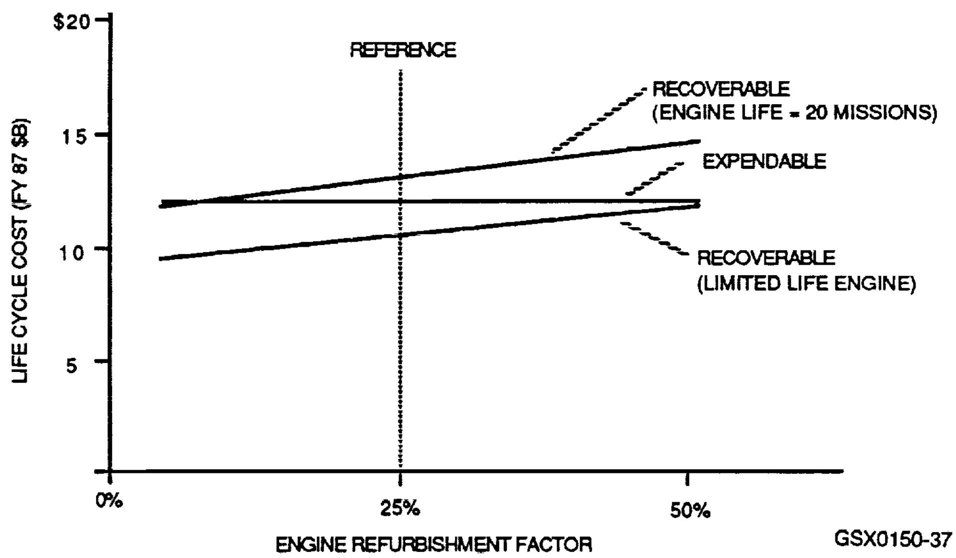


*Figure 4-3. Engine refurbishment cost sensitivity.*

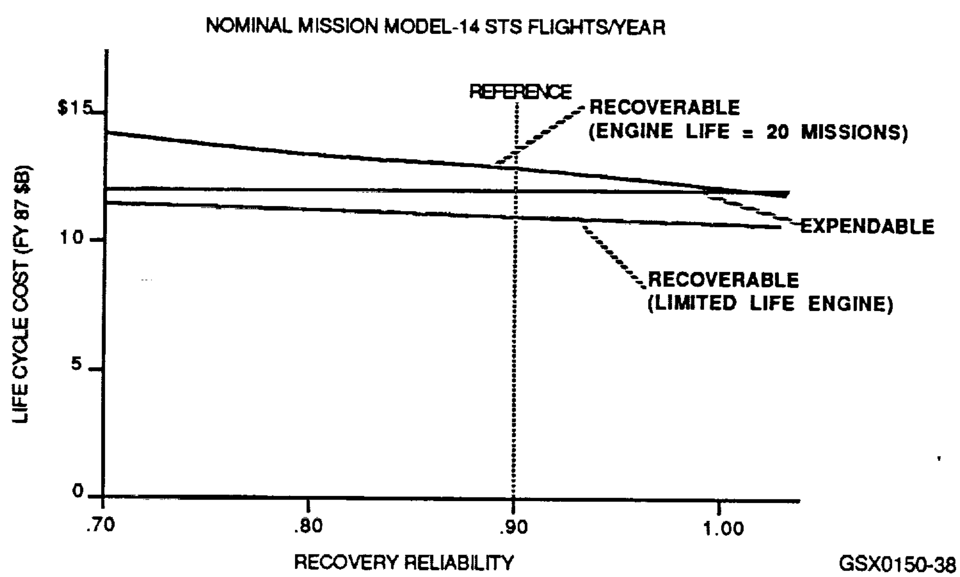


*Figure 4-4. Recovery reliability cost sensitivity.*

concepts. The recoverable vehicle with a limited-life engine is less sensitive to recovery reliability than the recoverable vehicle with an engine life of 20 missions. When an engine designed for 20 missions is unrecoverable, the engine lost is an expensive one which may have had up to 19 reuses left in it. On the other hand, when a limited-life engine is unrecoverable, the engine lost is a less expensive one which had four or less reuses left in it. So the cost of replacement engines is significantly different for the two recoverable concepts.

From this analysis we conclude that recoverable systems with engines designed for long life cannot compete effectively with inexpensive expendable engines. In addition, the limited-life, recoverable engine concept appears to offer some potential cost benefits over expendables. There are a number of issues such as recovery reliability, engine refurbishment, and verification of reuse of an expendable engine that still must be addressed.

We recommend that LRB be expendable. The additional development expenditure of approximately $1 billion will be nearly paid back after 100 flights. This is based on developing engines and the whole propulsion module for approximately 20 reuses. For vehicles with higher flight rates, recovery and reuse may be cost-effective.

### 4.3 LRB GEOMETRY, LENGTH AND DIAMETER

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. Multidiameter (such as hammerhead) and nonsymmetrical (clocked) arrangements were also tested.

The choice of LRB geometry is a complex problem involving aerodynamics, control, and structural loads on the whole STS stack as well as the LRB itself.

Although optimal length-to-diameter ratios (L/D) have not been established, a typical fineness ratio (L/D = 12.3) value has been examined for interface loads, and the results indicate that this ratio is acceptable. KSC facility-derived limits on diameter (19 feet maximum) and length (200 feet maximum) have also been identified. An LRB less than 170 feet long avoids interference with the ET GOX vent arm. Aerodynamic and aerothermal effects have been examined, and LRB lengths between 175 and 185 feet have the highest drag. These constraints are illustrated in Figure 4-2. Variations in length and diameter (for a given volume) have a small (less than 1%) impact on vehicle gross weight.

Based on the results of wind tunnel tests and our analysis, we recommend flying at q-alpha profiles that produce acceptable wing loading (LEMSCO Memo APO 208 4/27/88, "Minimizing Orbiter Wing Root Loads Impacts Due to the Incorporation of LRBs.") rather than using

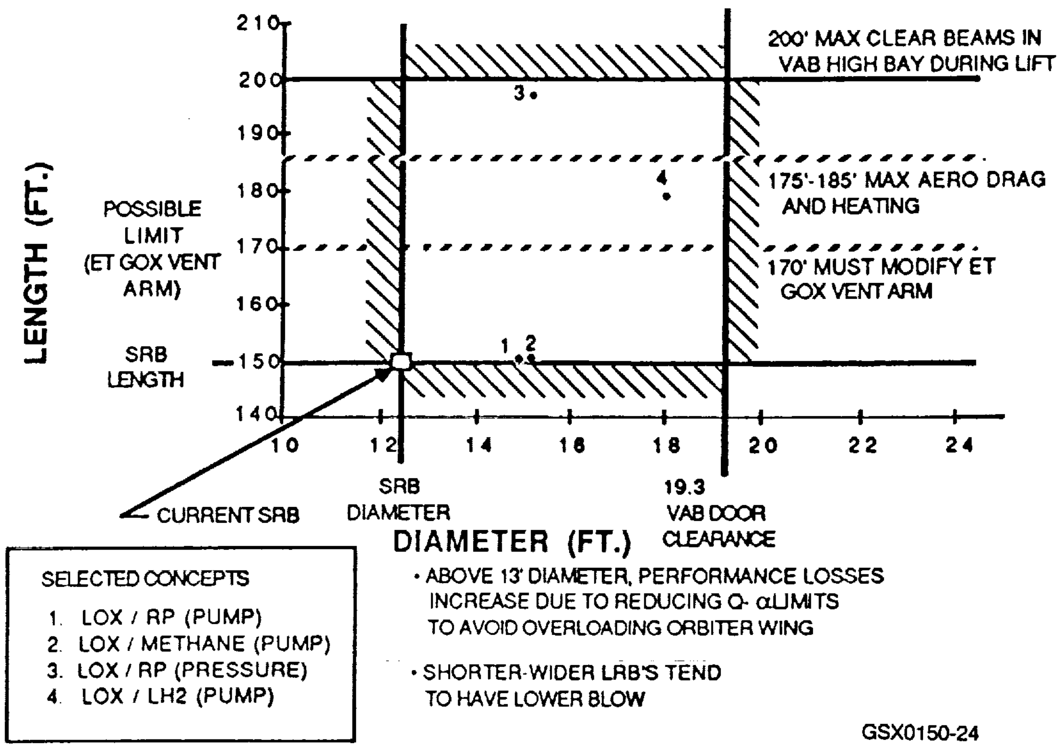


*Figure 4-5. LRB length and diameter constraints.*

unconventional geometries such as hammerhead. We believe that LRBs up to 19 feet in diameter and 200 feet long are feasible within a L/D range of 9 to 14.

SRMs have protruding rings and an electronic box that increase their effective diameter to nearly 16 feet. Therefore, diameters of 18 feet force only a modest reduction in max q-alpha and max q (see Section 4.4).

**4.3.1 KSC FACILITY CONSIDERATIONS** — The VAB doors are 871.5 inches wide. When the ET diameter (331 inches) is subtracted from this distance and provisions are made for dynamic clearances, the maximum LRB diameter possible is 19 feet.

The only limit on LRB length inside the VAB is imposed by the clearance above the support beams that separate highbays 2 and 4 from the VAB transfer aisles. A fully assembled LRB will have to be lifted through one of these openings before stacking on an MLP. The height (considering crane apparatus requirements) is about 257 feet — we recommend limiting the LRB to 200 feet. Taller LRBs would require modification of these beams. Because these beams are major structural members, their removal or modification is unlikely. In addition, inside the VAB high bays, numerous changes will have to be made to service platforms for all our LRB designs. The number and extent of modifications required increase as the length and diameter increase, but we do not feel that work platform impacts should be used to constrain LRB size.

At the launch pads (39A and B) the ET GOX vent arm is located at elevation 265, and the maximum length a LRB could attain and fit under the arm is about 170 feet. This limit is not firm. A modified GOX vent arm could be constructed, tested offline, and subsequently be changed out on the pad in a relatively brief operation. Wider LRBs encounter difficulties with LRB engines extending outside the flame trench and interfering with the $GH_2$ vent arm. The flame trench problem is significant and requires further analysis. As shown in Figure 4-3, preliminary indications are that these problems can be solved by designing specialized flame holes in the new MLPs. The $LO_2/LH_2$ LRB aft skirt appears to be only slightly larger than the current SRB skirt and, coupled with the smaller 74-inch engines, it potentially eliminates the problem for this configuration. The $GH_2$ vent arm impact problem applies to all LRB configurations. More analysis of actual LRB trajectories is required before the problem is completely resolved. All parties involved agree that the problem can be solved through trajectory shaping and redesign of the T-0 umbilical.

**4.3.2 AERODYNAMIC CONSIDERATIONS** — The aerodynamic considerations on LRB length and diameter were approached in two groups: effects on the Orbiter only (primarily wings), and effects on the mated vehicle (aerothermal heating, drag, and stability).

When nominal STS trajectory design is used with larger LRBs, loading on the Orbiter wings becomes a problem resulting from Orbiter wing root shear at max q. The ascent trajectory can

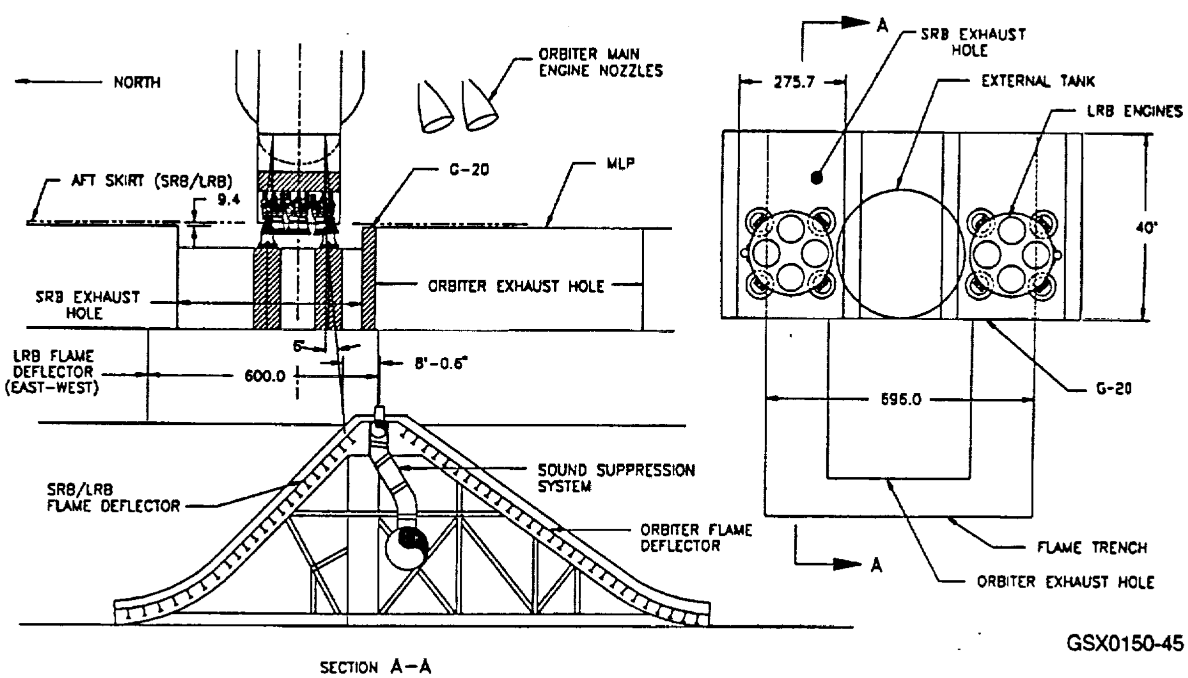


*Figure 4-6. $LO_2/LH_2$ flame trench design.*

be modified such that the dynamic pressure or alpha is reduced and the overall wing loading is kept within limits. As the dynamic pressure is lowered, however, a performance loss results that drives booster sizing to a slightly larger volume. Figure 4-4 shows that the performance penalties are not significant with diameters up through 18 feet, as long as there are no external protuberances on the LRB under the wing.

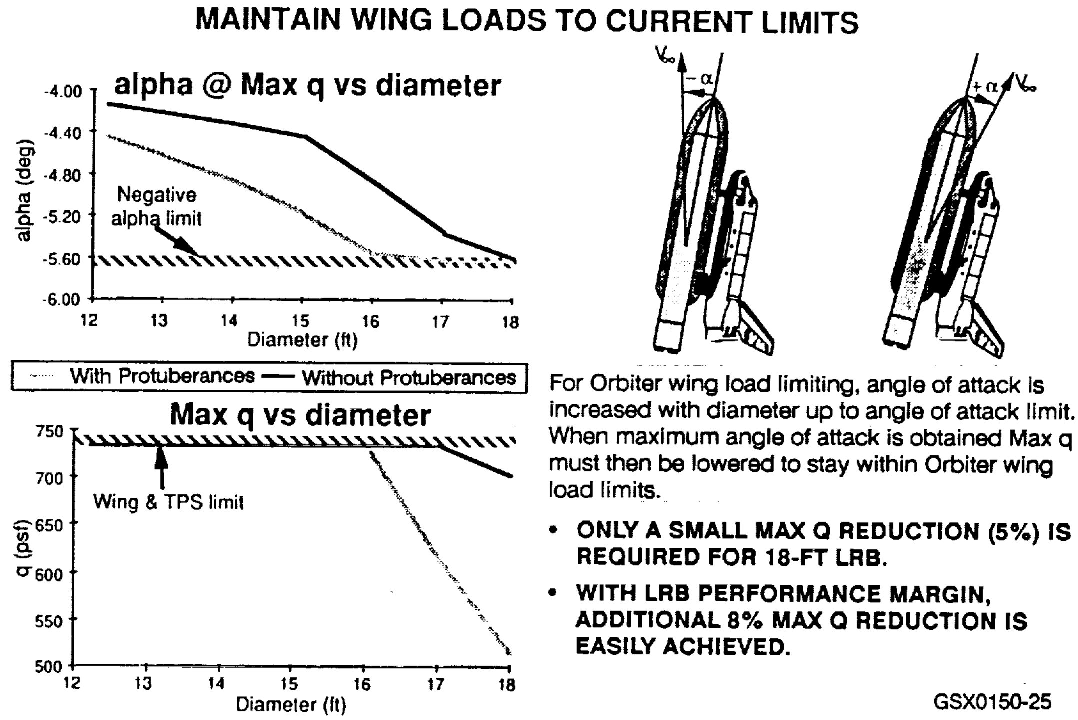


*Figure 4-7. LRB diameter effects on angle of attack and dynamic pressure.*

## 4.4 AVIONICS SYSTEMS ARCHITECTURE SELECTION

Our objective for the LRB avionics is to improve STS system reliability while minimizing Orbiter software and hardware impacts. The system had to provide command and monitoring capabilities for the more complex liquid engines and subsystems, while reducing ground operations and support requirements. The system also had to minimize development cost and risk.

Several different avionics system architectures were evaluated, as shown in Table 4-7.

An improved technology Centaur avionics-based system was selected. We also recommend implementing failure-tolerant techniques to provide the high reliability needed. Flight control commands such as engine start/shutdown, TVC, and separation will come from the Orbiter avionics as is presently done for SRBs. This will allow the use of the existing command lines and therefore provide the least impact to the present Orbiter hardware and software configuration.

The architecture adapted as the baseline is illustrated in Section 3, Figure 3-11. This system provides a redundant three-string system to achieve the large reliability needed for the STS mission. In addition, each LRB uses its own RF telemetry system, supporting the increased telemetry requirements for the more complex LRB. By incorporating data processing capabilities into the system, data going to the Orbiter avionics and commands coming from the Orbiter avionics can be manipulated to ensure consistency with the present Orbiter data protocol.

*Table 4-7. LRB avionics architecture trade comparisons.*

| Criteria / Options | SRB-based | Modern avionics | MPRAS-based |
|---|---|---|---|
| Vehicle interface impacts | H/W - high<br>• More control lines<br>• More ATVC channels<br>• More data lines<br>• Higher power requirements<br>S/W - high<br>• Complex booster engine control/monitor<br>• Pressure control<br>• Caution & warning | H/W - low<br>• Autonomy incorporated as needed<br>S/W - medium<br>• Caution & warning<br>• Booster to booster communication | H/W - low<br>• Autonomy incorporated as needed<br>S/W - medium<br>• Caution & warning<br>• Booster to booster communication |
| Ground interface impacts | H/W - low<br>• Through Orbiter<br>S/W - medium<br>• Booster checkout<br>• Booster monitor/command | H/W - medium<br>• Booster umbilicals<br>• Booster communication<br>S/W - medium<br>• Booster checkout<br>• Booster monitor/command | H/W - medium<br>• Booster umbilicals<br>• Booster communications<br>S/W - medium<br>• Booster checkout<br>• Booster monitor/command |
| Operational complexity | High<br>• Orbiter-dependent | Low<br>• Autonomous | Low<br>• Autonomous |
| Program risk<br>– Schedule<br>– Availability | <br>Low<br>Medium | <br>Low<br>Low | <br>High<br>Medium |
| Reliability | Meets requirements | Meets requirements | Meets requirements |
| Cost<br>– Development<br>– Recurring | <br>Medium<br>Low | <br>Low/medium<br>Low/medium | <br>High<br>Low |
| Growth potential | Limited | Good | Excellent |

Discriminators used to eliminate avionics options.

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# 5 ✦ PROJECT PLANNING

During this study, preliminary plans were established for the full-scale development (FSD) phase of the LRB program. This section summarizes these plans, which are addressed in greater detail in the Preliminary Project Implementation Plan, Appendix 2.

## 5.1 PROJECT MANAGEMENT

The LRB project will be organized according to the program Work Breakdown Structure (WBS). A preliminary WBS for the LRB program is shown in Figure 5-1. Each functional organization supporting the LRB program will be assigned responsibility for certain WBS elements and will receive a separate budget for each WBS element it must support.

Since the objective of the LRB program is to enhance the performance and safety of a launch system that is already operational, timely implementation of the LRB program is particularly vital to its success. A preliminary LRB project master schedule is shown in Figure 5-2. Since a final selection between pump-fed and pressure-fed LRBs has not yet been made, this schedule shows engine development milestones for both concepts.

## 5.2 SYSTEMS ENGINEERING AND INTEGRATION

Integration responsibilities of the LRB prime contractor include definition and refinement of system requirements, development of compatible interface designs, support for the development of an integrated STS system verification program, and support for the development of

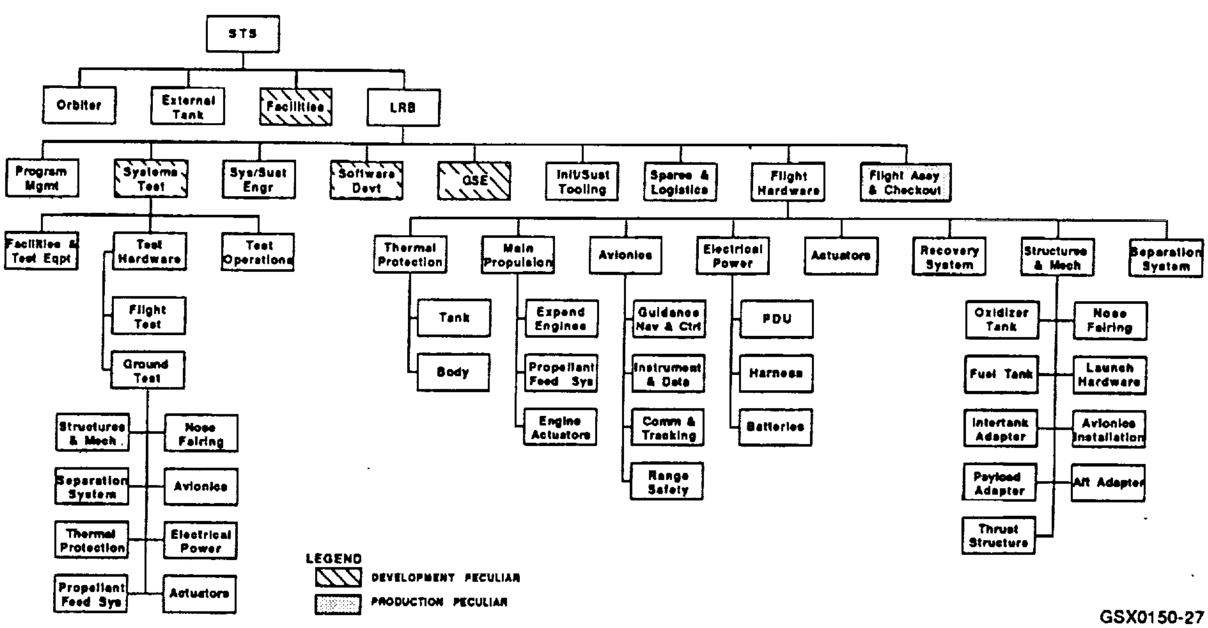


*Figure 5-1. LRB work breakdown structure (development and production phases).*

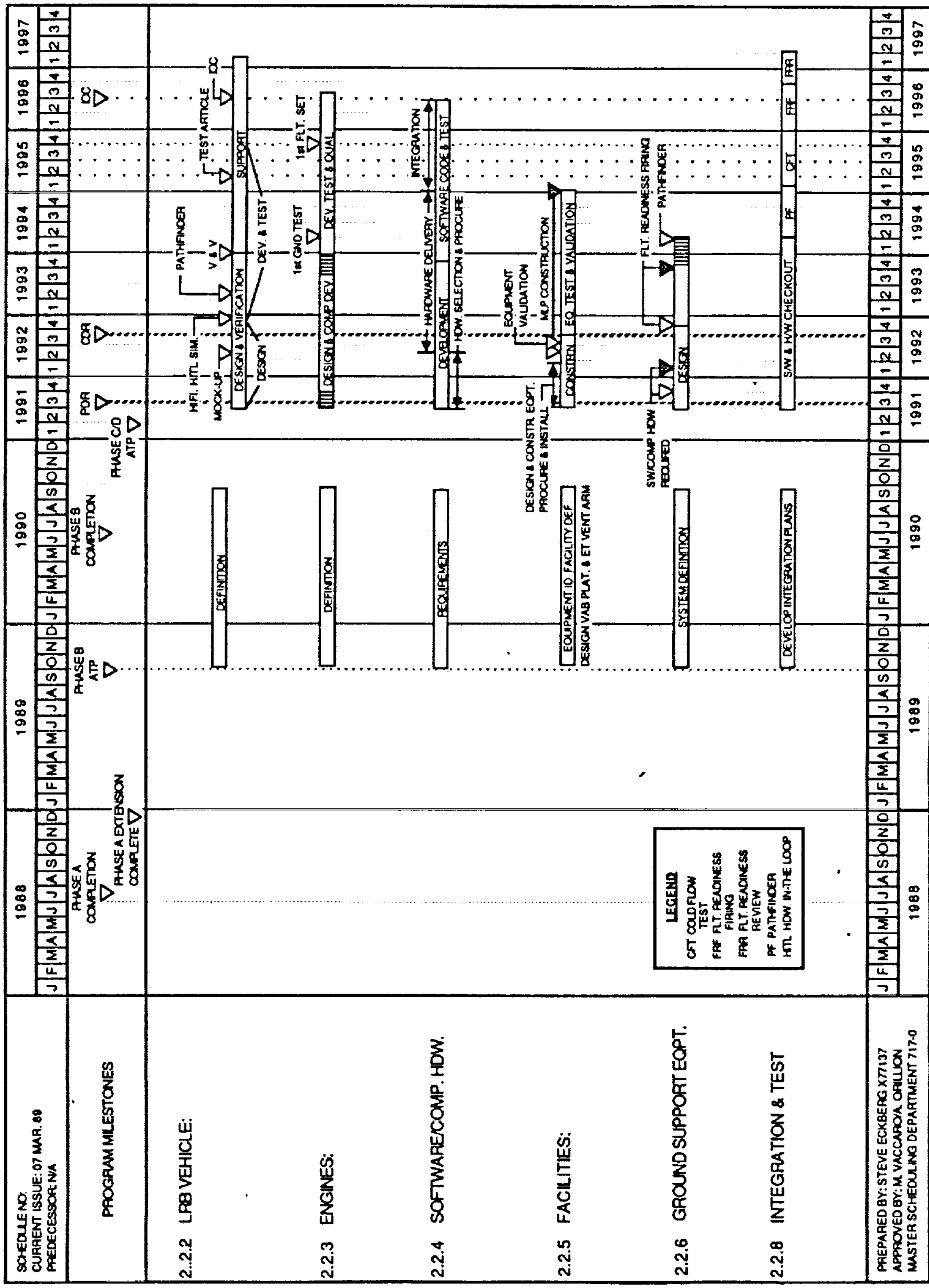


*Figure 5-2. LRB master schedule.*

launch site operations. The role of the systems integrator on this program is especially critical, due to the LRB's integration into an existing manned launch system. Major LRB system requirements that will drive the vehicle design and operational procedures include:

- Ascent performance
- Intact abort
- Safety
- STS compatibility
- Minimization of development and life cycle costs
- Evolution and growth

## 5.3 DESIGN AND DEVELOPMENT

The primary objective of the LRB design and development effort is to generate a safe, reliable, low-cost design that can be readily integrated with the STS. Our recommended design approach is to maximize the use of proven design concepts that offer sufficient performance margins. Early trade-off and sensitivity studies have addressed performance vs. complexity, weight, cost, and risk; these analyses need to be continued through the early part of the FSD phase. A commonality plan should be completed prior to the Preliminary Requirements Review (PRR).

## 5.4 MANUFACTURING APPROACH AND FACILITIES

The LRB manufacturing approach is based upon strong cooperative efforts between design and producibility engineers. Producibility engineers will actively participate in all trade studies, engineering reviews, and concept reviews to assure that manufacturing considerations are incorporated into the design. The manufacturing approach for the LRB is to provide in-house assembly, test, and checkout of the vehicle and to procure from subcontractors the detail components that go into those assemblies.

The optimal location for the LRB final assembly facility has not yet been determined. Locating the factory at or near the launch site would eliminate costly and potentially hazardous transportation problems. If the final assembly facility is not in the immediate KSC vicinity, it will need to have waterway access to the launch site, since the LRB will be too large for land or air transportation. If transportation is required, certain checkout procedures may need to be performed twice — at the factory and again at KSC. These factors favor location of the LRB final assembly plant close to the launch site, although economic or political factors may ultimately determine that the factory be placed somewhere else.

## 5.5 TEST AND VERIFICATION

The LRB test verification approach must take into consideration the many facets that drive the verification activities in the development of requirements and the application of methods to meet requirements. Figure 5-3 shows the major elements of the LRB test program. Particular development tests are listed in Table 5-1. The following is a list of key program features that will affect verification requirements and procedures:

- The LRB will be used for STS manned missions
- The LRB will provide very high thrust levels
- The main engine area will be subjected to a very high vibration/acoustic environment
- The LRB will perform during a very critical period of flight
- Two LRBs will perform in parallel during their normal use
- The LRBs are an integral part of the Space Shuttle system

## 5.6 LAUNCH OPERATIONS AND FACILITIES

There will be three major tasks associated with LRB ground operations (see Figure 5-4): LRB checkout and NASA acceptance, vehicle integration, and integrated Shuttle vehicle checkout

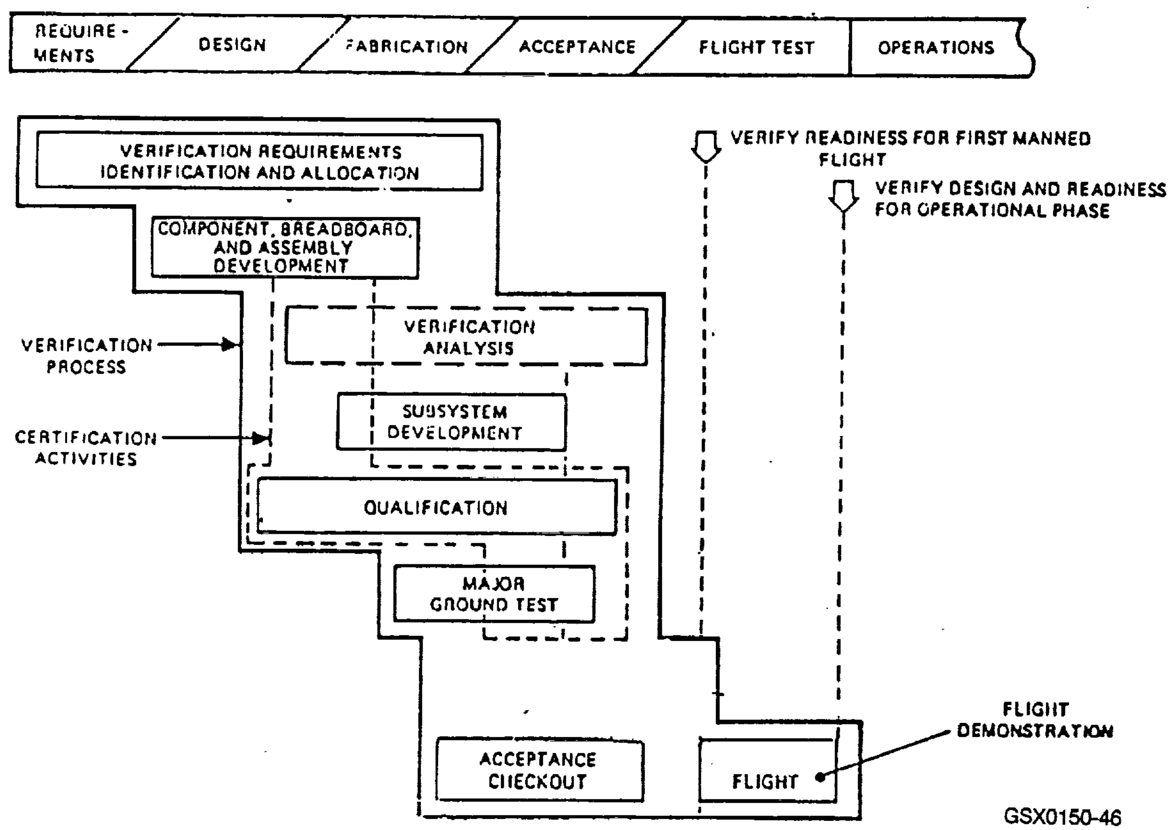


*Figure 5-3. Major elements of the LRB test program.*

*Figure 5-3. Major elements of the LRB test program.*

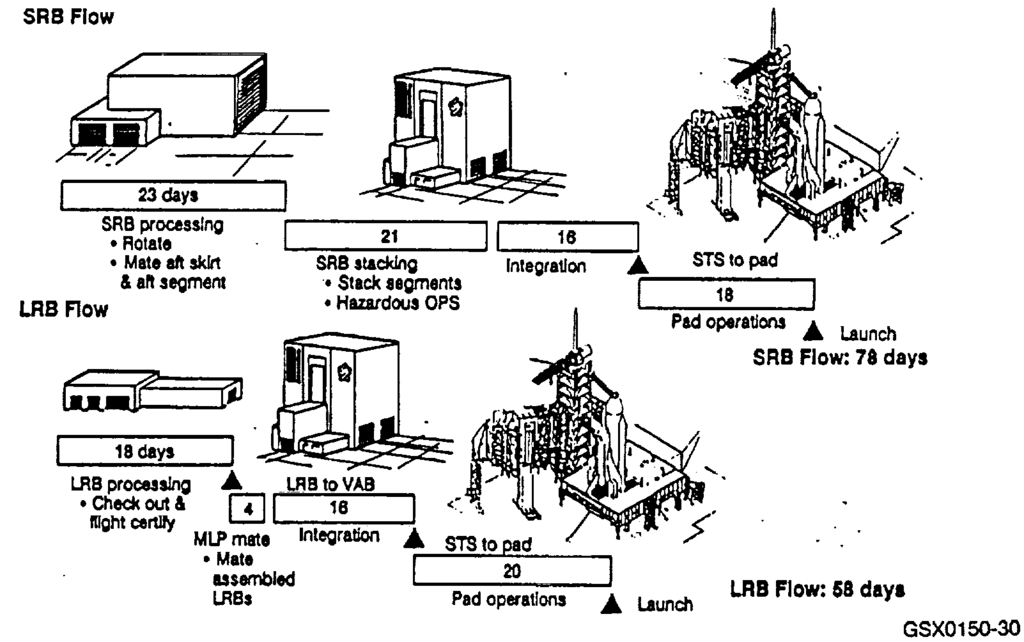


*Figure 5-4. Launch operations time table.*

and launch operations. Once final checkout and NASA acceptance have been completed, the LRB will be delivered to the Vehicle Assembly Building (VAB) transfer aisle. The LRB will then be mated to the MLP and all the appropriate systems such as data, fuel, environmental control, and purge will be connected. Once the hookups are verified, the LRBs will be aligned and the ET mated. The Orbiter will then be mated. Once the system is fully integrated, there will be a Shuttle Integrated Test (SIT) to verify compatibility and to test the fully integrated vehicle. The vehicle will then be transported to the launch pad, where payload integration and additional testing will be performed.

The LRB will introduce several significant improvements to STS launch operations. The use of LRBs will eliminate many hazardous operations in the VAB because they will not be processed with live fuel as with the SRBs. In addition, the LRB will arrive fully assembled, eliminating the time-consuming task of booster assembly from the critical path of Shuttle processing. In fact, the LRB schedule should reduce current processing time lines by over 20 days. This reduction will greatly increase the likelihood of attaining the desired 14 launches per year. The LRB also improves the STS launch windows by increasing the temperature range at which the Shuttle can launch.

The existing launch processing facilities must be modified to accommodate the larger size of the LRBs and to provide a propellant servicing capability at the launch pads. The principal facilities that must be modified are the launch pad and the VAB. The principal change to the launch pad will be the installation of new propellant storage and transfer systems. The $GH_2$ vent arm T-0 umbilicals may also need to be modified. For certain LRB configurations, the ET GOX vent arm would have to be modified to enable them to "wrap around" the LRB. The principal changes required to the VAB will be modification of interior platforms to accommodate the larger-diameter LRB. In addition, GDSS recommends that LRB checkout and acceptance take place at the assembly site to minimize new facilities needed at KSC.

New equipment required will include two MLPs (one new, one modified existing MLP). The new MLPs will feature systems and designs to accommodate an LRB such as: a new propellant system, a new holddown system to provide a soft release for the STS, and enlarged flame holes for the larger LRB engine plumes.

In addition to these facilities, KSC recommends addition facilities and modifications as shown in Figure 5-5 to allow further improvements in Shuttle processing and less potential for schedule impacts during the transition from SRB operations to LRB operations. These modifications include removal of ET processing to a joint ET/LRB processing facility and the activation of VAB highbay 4.

*Table 5-1. LRB development tests (preliminary list).*

| | |
|---|---|
| **Wind Tunnel**<br>Aerodynamic wind tunnel pressure & loads test<br>Aerodynamic wind tunnel stability & control test<br>Wind tunnel captive trajectory test<br>Base heating/recirculating wind tunnel test<br>Aerodynamic heating<br>**Structures and Mechanisms**<br>Engine gimbal frequency response test<br>Aft skirt structural test<br>Model firewall test<br>LRB jettison test<br>Forward attach fitting load test<br>Aft attach fitting load test<br>Separation explosive device functional test<br>Engine boot material heating test<br>Nose cone material heating test<br>Tank insulation characteristics<br>Welding process development tests<br>• Weld coupons test<br>• Weld joint cyclic load tests<br>Component development testing (rings, domes, baffles, etc.)<br>**GSE**<br>Model flame bucket/flame deflector test<br>Launcher operating load/deflection test<br>$LO_2/LH_2$ line retract tests<br>Rise-off panel test | **Engines**<br>Injector<br>• Mixture ratio<br>• Ignition stability<br>• Throttling characteristics<br>Thrust chamber assembly<br>• Thermal characteristics<br>• Materials selection<br>• Errosion<br>Controller<br>• Thermal environment<br>• Control Algorithms<br>• Response Characteristics<br>• BIT/BITE requirements<br>Nozzle<br>• Cooling<br>• Inertia<br>• Gimbal Limits<br>• Flex Mechanism<br>Thrust vector control system<br>• Hydraulic system (pumps & lines)<br>• Force requirements<br>• Response characteristics<br>Throttling device<br>• Mixture ratio<br>• Response Characteristics<br>Component development testing |

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## 5.7 MISSION OPERATIONS SUPPORT

The LRB prime contractor will have an ongoing role supporting NASA mission planning, operations, and analysis. This task will include development of LRB flight requirements and constraints, analysis of LRB mission performance, and support of NASA/MSFC and the STS Program in accomplishing these functions for the integrated launch vehicle. During the LRB flight test program, additional tasks will be required, including definition of flight test requirements and comprehensive analysis of mission data. These responsibilities will be carried out under the coordination of the SE&I organization and will include support of all LRB project technical groups.

*Table 5-2. Recommended facility changes.*

| Facility | Estimated Cost* |
|---|---|
| VAB High Bays 1 & 3 | 16.8 |
| Activate VAB HB 4 | – |
| Launch Control Center | 13.2 |
| New MLP | 122.3 |
| Mod MLP 3** | 80.0 |
| Pad mods | 96.2 |
| LETF | 26.7 |
| MLP Parksite | 2.4 |
| Power dist | 16.9 |
| ET/LRB processing facility/GSE | – |
| Total | $374.5 |

* In FY87 $M

** No new MLP required – modify existing MLP (18-ft $LO_2/LH_2$ vehicle)

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# 6 ✦ COST ANALYSIS

Cost analysis results have been an important input to LRB trade study downselections and for developing program planning cost data. Life cycle cost (LCC) estimates were one of the key discriminators in downselection decisions on the SSME vehicle, expendable vs. reusable vehicles, numerous engine propellant combinations, and propellant tank material/construction selection.

## 6.1 COST ESTIMATING METHODOLOGY

In completing cost estimates and trade studies it is essential to recognize all of the design and operational differences between the various alternatives. For example, to complete a vehicle propellant tank trade study between two alternative material types (2219 aluminum and graphite/epoxy) Engineering would resize (performance and mass properties) the vehicles based on each tank material. The resizing effort provides the cost analyst new vehicle mass properties and engine thrust levels based on the two different tank materials. The respective vehicle mass properties, engine thrust levels, and appropriate development and manufacturing complexities that distinguish the two tanks are inputs to the model for each trade study alternative. The resulting LCC estimates combined with numerous technical considerations become the discriminators used to downselect to one tank material. This is the normal trade study method that we have employed to complete LRB vehicle downselections.

## 6.2 COST ESTIMATES

The selected vehicles for the final report are expendables with the following propellant combinations: pump-fed $LO_2/LH_2$, pump-fed $LO_2$/RP-1, and pressure-fed $LO_2$/RP-1 (see Table 6-1). The cost estimates are based on the 14 STS flights per year mission model (244 LRBs) and are in FY 87 dollars in millions excluding contractor fee, government support, and contingency. Table 6-1 presents the vehicle DDT&E and production cost by subsystem for the selected concepts. As has been the case during this study, the major subsystem cost contributor for DDT&E and average unit cost (AUC) for each of the vehicle concepts is the main engine. The only exception is the pressure-fed $LO_2$/RP-1 vehicle, in which the AUC of the structure/TPS subsystem is the major cost contributor.

The three selected vehicle LCC estimates are presented in Table 6-2 and include nonrecurring cost, recurring production cost, and recurring operations cost. The nonrecurring cost includes the vehicle DDT&E, Orbiter and external tank modifications, new and modified facilities, and STS systems engineering and integration (SE&I). The LCC of the three selected LRB vehicles is approximately $10 to 12 billion.

*Table 6-1. DDT&E/production cost summary.*

| Cost Element \ Concept | Pump-fed New $LH_2/LO_2$* | Pump-fed New RP-1/$LO_2$* | Press-fed New RP-1/$LO_2$* |
|---|---|---|---|
| **DDT&E** | | | |
| Structures/TPS | 231 | 206 | 248 |
| Separation system | 23 | 23 | 30 |
| Propulsion system | 146 | 169 | 388 |
| Main engines | 1007 | 878 | 435 |
| Avionics/electrical power | 70 | 70 | 70 |
| Tooling/test ops/GSE/ S/W | 462 | 424 | 433 |
| Systems engr/program mgmt | 218 | 204 | 188 |
| TOTAL | 2157 | 1974 | 1792 |
| **Average Unit Cost** | | | |
| Structures/TPS | 8 | 7 | 9 |
| Separation system | 1 | 1 | 1 |
| Propulsion system | 3 | 4 | 10 |
| Main engines | 13 | 9 | 5 |
| Avionics/electrical power | 3 | 3 | 3 |
| Sustaining tooling/final assy | 3 | 2 | 4 |
| Systems engr/program mgmt | 2 | 2 | 2 |
| TOTAL | 33 | 28 | 34 |

*Costs are in FY 87 $M

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*Table 6-2. Life cycle cost summary.*

| Cost Element \ Concept | Pump-fed New LH2/$LO_2$* | Pump-fed New RP-1/$LO_2$* | Press-fed New RP-1/$LO_2$* |
|---|---|---|---|
| **Nonrecurring** | | | |
| Vehicle DDT&E | 2157 | 1974 | 1792 |
| Orbiter modifications | 229 | 229 | 229 |
| ET modifications | 20 | 20 | 20 |
| Facilities | 413 | 357 | 372 |
| STS SE&I | 105 | 105 | 105 |
| **Total nonrecurring** | **2924** | **2685** | **2518** |
| **Recurring** | | | |
| Vehicle production | 8001 | 6873 | 8362 |
| Launch operations | 830 | 818 | 830 |
| **Total recurring** | **8831** | **7691** | **9192** |
| **Total LCC** | **11755** | **10376** | **11710** |

*Costs are in FY 87 $M

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The concept downselection from three selected vehicles to a pump-fed $LO_2/LH_2$ vehicle considered all data developed during the study, especially propellant combination data, safety, technical risk, LCC, and evolution and growth potential. The attributes of the pump-fed $LO_2/LH_2$ vehicle were judged to be best when considering all of these issues.

A schedule of annual funding of all nonrecurring cost for the pump-fed $LO_2/LH_2$ vehicle is based on the overall program schedule and is shown graphically in Figure 6-1. The nonrecurring cost of the pump-fed $LO_2/LH_2$ vehicle is spread by fiscal year into engine nonrecurring, other nonrecurring and total nonrecurring less engine nonrecurring. This figure indicates that the vehicle reaches peak annual funding of about $750 million during Phase C/D. In addition, if the USAF continues to fund the ALS engine development, which is common to LRB needs, then the peak funding drops to about $550 million.

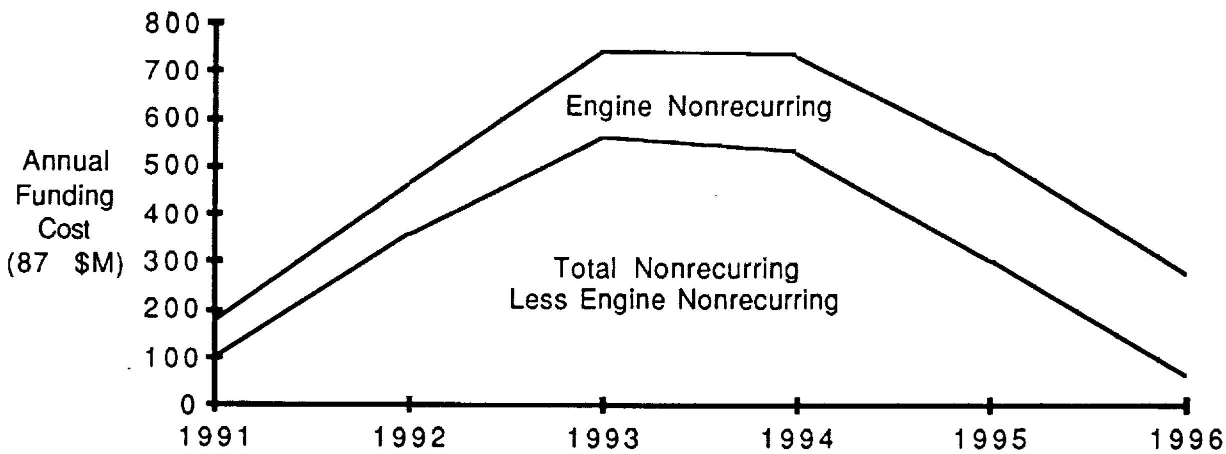



*Figure 6-1. Pump-fed $LO_2/LH_2$ annual funding schedule.*

### 6.3 ALS/LRB SYNERGISM

A cooperative USAF/NASA LRB program could provide significant cost benefits that result from the use of a common LRB on the ALS and the STS. In addition, the synergism that results from the use of the LRB's pump-fed $LO_2/LH_2$ engine on the ALS core could provide more cost benefits. A common development program could provide DDT&E cost savings of up to $1.6 billion compared with funding two independent programs. A common production program could provide additional learning and rate effects, which would result in a decrease in the average unit cost of LRBs for the STS.

### 6.4 COST ANALYSIS CONCLUSIONS

- LRB vehicle DDT&E cost is approximately $2 billion
- LCC is $10 to 12 billion
- Engines remain the major cost contributor
- ALS/LRB synergism could provide significant cost savings

# 7 ✦ EVOLUTION AND GROWTH

A close look at several LRB concepts used for growth STS (Shuttle-C), standalone boosters, and ALS was incorporated into this Phase A study. The $LO_2/LH_2$ option emerged as the clear favorite.

All three ALS contractors recommend a $LO_2/LH_2$ core with an engine thrust in the half-million-pound class. Closer comparison with the GDSS concept for ALS showed that the same engine could be used for STS LRB and the ALS core, except for differences in the expansion ratio and throttling needed. At MSFC, technology work is proceeding on the $LO_2/LH_2$ engine for ALS called the Space Technology Engine Program (STEP). There are a number of potential benefits to both NASA and the USAF if one large new rocket engine were developed for both ALS and STS. This commonality with ALS was a major factor in our recommending a liquid oxygen/liquid hydrogen concept for the STS LRB.

## 7.1 SUMMARY OF RESULTS

This study has established that the LRB concept can be used successfully in many alternate applications. This flexibility provides additional benefits to the basic STS-LRB program such as potential LRB development cost savings due to DDT&E cost sharing with other programs and reductions in production unit cost because of increased rates of production to support multiple applications.

Major conclusions of the alternate applications study are summarized in Figures 7-1 and 7-2, and are listed on the following pages. For further information on LRB alternate applications and evolutionary growth, refer to Volume II, Appendix 9, Book 5.

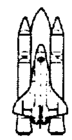
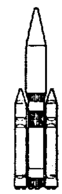
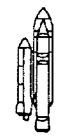
| Application / Requirement |  STS LRB |  ALS |  Shuttle-C |  Standalone |
|---|---|---|---|---|
| Payload (Klb) (nominal mission model) | 7.5 (160 nmi 28.5°) | 80 (80x150 nmi 28.5°) | 100-150 (220 nmi 28.5°) | TBD (150 nmi 28.5°) |
| Performance (total booster impulse) | 540 m lb sec | 640 m lb sec | ~500 m lb sec | 250+ m lb sec |
| Man-rated | Yes | No | No | No |
| Flight rate/year | 14 | 10 (capability to 20) | 2-3 | ~10 |
| Engine-out capability | Yes | Yes | Yes | TBD |
| Booster reusability | No | Engine recovery | TBD | No |
| IOC | 1995 | ~2000 | 1993 | 1995-1996 |

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*Figure 7-1. LRB applications to other vehicle programs.*

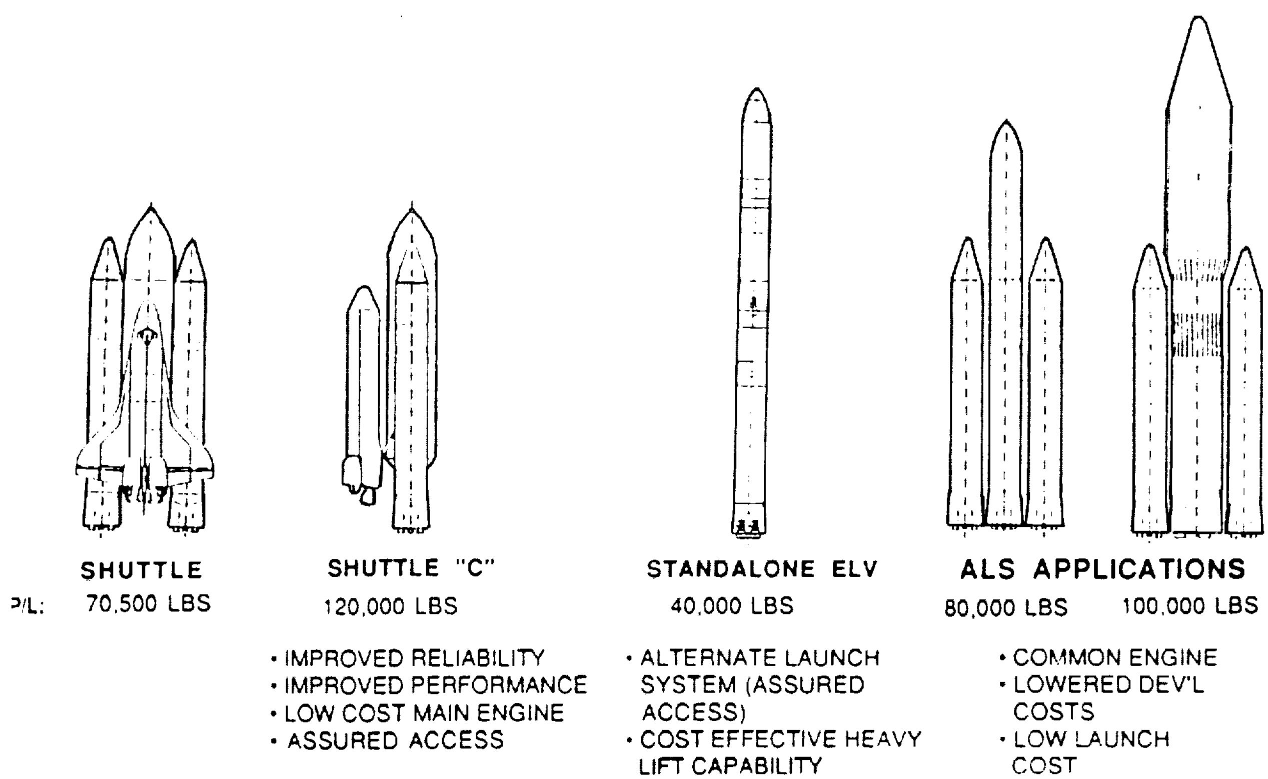
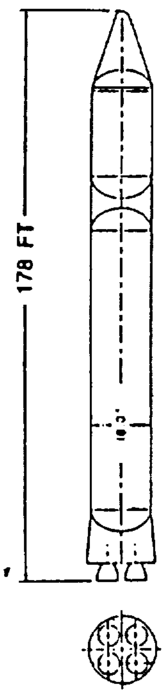



*Figure 7-2. LRB evolution and growth results.*

**LRB applications to ALS:**

- The $LO_2/LH_2$ LRB is best suited for ALS because of common propellants.
- The $LO_2/LH_2$ LRB has virtually the same engines as the ALS, and therefore a common engine development program is feasible (see Figure 7-3).
- A family of vehicles with payload capabilities ranging from 50 to 200 Klb can be formed by varying the number of LRBs used and the number of engines per LRB. One such possibility is shown in Figure 7-4.
- Use of LRBs for ALS can reduce NASA's LRB DDT&E and recurring production costs by sharing program costs with USAF.




| | LRB | ALS Booster |
|---|---|---|
| Propellants | LO2/LH2 | LO2/LH2 |
| Dry weight (Klb) | 129 | 186 |
| No. of engines | 4 | 7* |
| Engine-out capability** | 1 | 1 |
| **Engine** | | |
| Thrust, vac NPL (sec) | 558 | 612 |
| Isp, vac NPL (sec) | 411.4 | 429.4 |
| Pc NPL (psia) | 2250 | 2000 |
| Mixture ratio | 6.0 | 6.0 |
| Area ratio | 20:1 | 38.8 |
| Throttleability | 100% or 75% | None |
| Reliability (90% confidence) | 0.99 | 0.99 |

* Common engine for core & booster
** With mission-complete capability

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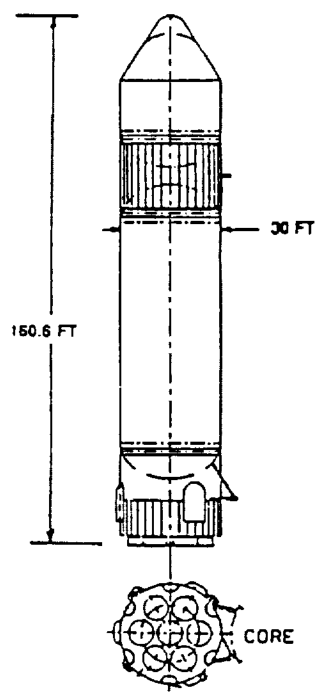
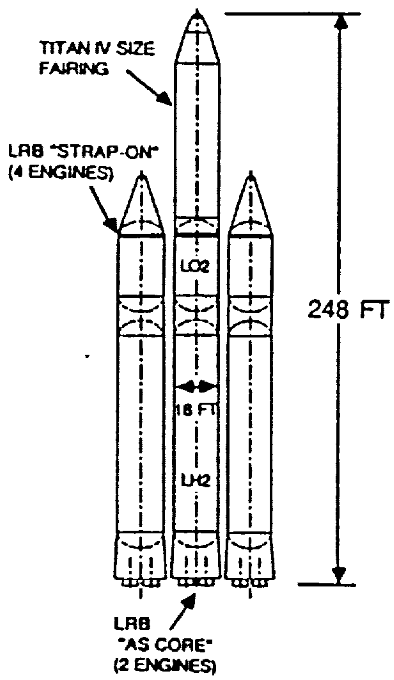



*Figure 7-3. LRB/ALS booster comparison.*




| | |
|---|---|
| **AVAILABLE:** | 1997 TO 1998 |
| **PAYLOAD:** | |
| • LEO (80 X 150 NMI, 28.5 DEG) | 80,000 LB |
| **VEHICLE:** | |
| • GLOW | 2.54 M LB |
| • INERT WT | |
| LRB (AS CORE) | 115,890 LB |
| LRB (AS STRAP-ON BOOSTER) | 130,600 LB |
| **TRAJECTORY:** | |
| • MAX Q | 850 PSF |
| • MAX G | 5.5 g's |
| • STAGGING VELOCITY | 11,633 FT/SEC |
| • TOTAL DELTA VELOCITY | 28,857 FT/SEC |

**MAJOR MODIFICATIONS TO LRB (FOR USE AS CORE):**

- REMOVE 2 ENGINES AND ATTENDANT TVC
- MODIFY THRUST STRUCTURE
- ADD GN&C, AND TELEMETRY AVIONICS
- ADD NEW FLIGHT SOFTWARE
- REMOVE NOSE CONE AND ADD PAYLOAD FAIRING ADAPTER
- ADD BOOSTER STRUCTURAL ATTACHMENTS
- ADD ACS SYSTEM AND DEBOOST MOTOR

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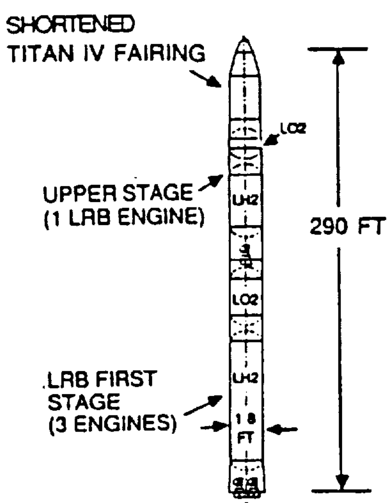
*Figure 7-4. LRB core with two strap-on LRBs.*

**LRB applications to Shuttle-C:**

- LRBs provide approximately 20 Klb greater payload capability than SRBs for Shuttle-C.
- Use of LRB engines as SSME replacements may lower Shuttle-C costs per flight.
- Applicability of LRBs and LRB engines to Shuttle-C provides NASA with an additional measure of assured access to space.
- The LRB provides many of the same benefits to the Shuttle-C that it provides the Shuttle such as improved reliability (i.e., engine-out capability) and safer operations (i.e., hazardous propellants are removed from the VAB).

**LRB application to standalone expendable launch vehicles:**

- LRB standalone expendable launch vehicles can be used as an initial building block for ALS in the lower payload range.
- New LRB standalone launch vehicles provide an additional measure of assured access to space — an alternative to Titan IV, as shown in Figure 7-5.
- The $LO_2/LH_2$ LRB has the best performance of candidate LRB designs for standalone launch vehicle applications.
- The recommended LRB standalone launch vehicle is a core-to-orbit concept that uses one or two LRB boosters in a modular approach to deliver 25 to 80 Klb of payload to low Earth orbit.




**AVAILABLE:** 1997 TO 2000

**PAYLOAD:**
- LEO (80 X 150 NMI, 28.5 DEG) 40,000 LB

**VEHICLE:**
- GLOW 1,260,370 LB
- INERT WT
  UPPER STAGE 53,051 LB

**TRAJECTORY:**
- MAX Q 413 PSF
- MAX G 5.5 g's
- STAGGING VELOCITY 4,903 FT/SEC
- TOTAL DELTA VELOCITY 32,408 FT/SEC

**MAJOR LRB MODIFICATIONS**
- REMOVE 1 ENGINE AND ATTENDANT TVC
- MODIFY THRUST STRUCTURE
- CHANGE FLIGHT SOFTWARE
- REMOVE NOSE CONE AND ADD INTERSTAGE ADAPTER

**UPPER STAGE FEATURES:**
- SINGLE LRB BOOSTER ENGINE
  (CONSIDERING USING NOZZLE EXTENTION FOR HIGHER AR)
- PROVIDES ENTIRE VEHICLE GN&C

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*Figure 7-5. LRB with new upper stage.*

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