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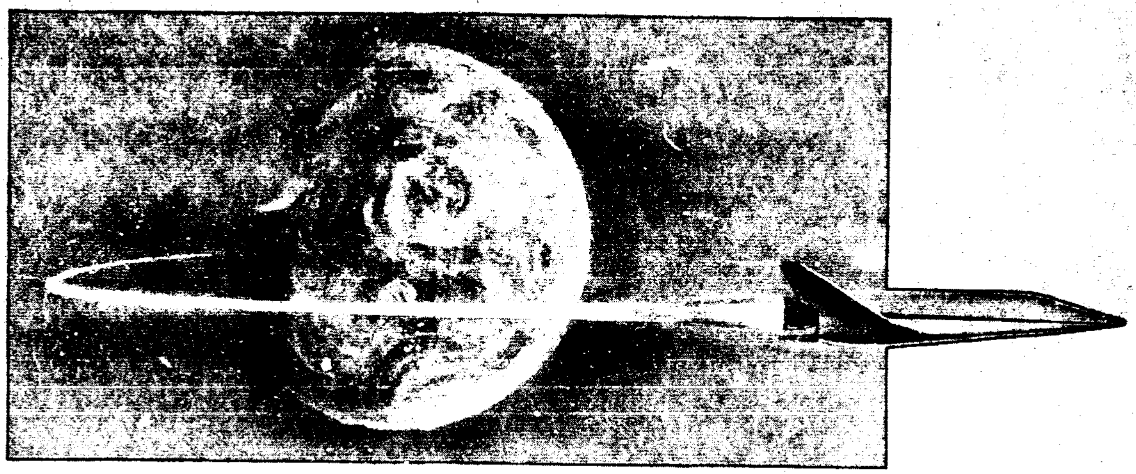
SPACE SHUTTLE



LOCKHEED
LMSC
SPACE SYSTEMS
DIVISION

FINAL REPORT
INTEGRAL LAUNCH AND REENTRY VEHICLE

EXECUTIVE SUMMARY

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FINAL REPORT

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EXECUTIVE SUMMARY

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FOREWORD

The Integral Launch and Reentry Vehicle (ILRV) Study was conducted by Lockheed Missiles & Space Company for the NASA Marshall Space Flight Center under contract NAS9-9206. The Final Report, dated December 22, 1969, and bearing the number LMSC-A959837, is contained in three volumes, as follows:

I Configuration Definition and Planning
II Technology Identification
III Special Studies

Highlights of the study are presented in this Executive Summary.

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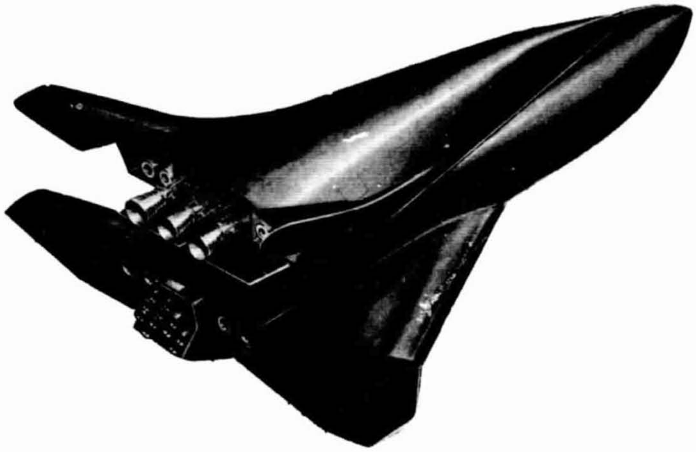
I. INTRODUCTION

The Lockheed Integral Launch and Reentry Vehicle Study reflects a 10-year span of company interest in reusable launch and space operations concepts, backed by extensive company-funded study and development effort in lifting entry spacecraft aerodynamics and heat transfer.



To meet the projected national space transport requirements, a Two-Stage fully reusable launch system, based on advanced hydrogen-oxygen propulsion systems, has been derived in the study. The booster is comparable in size and weight to the C-5A aircraft, and the lifting body orbiter incorporates a propulsion system comparable to the Saturn S-II stage. As alternative designs were investigated, several candidate systems were eliminated. Two principal concepts were investigated initially in the study. In one, the Stage-and-One-Half, a single orbiting stage and low-cost drop tanks are featured. In the other, the pure Triamese, commonality of orbiter and booster primary structure and propulsion systems was adopted. The objective in both concepts was to eliminate a separate parallel booster development program.

Midcourse in the study, the drop-tank concept and the commonality concept were abandoned as the cost trends resulting from high traffic rates and increased payload weight and volume requirements became apparent. Accordingly, the remainder of the effort was concentrated on the preferred fully reusable Two-Stage Shuttle and the information presented in this report pertains primarily to that configuration.

II. STUDY OBJECTIVES

The overall objective of the study was to generate conceptual designs for a Space Shuttle system that would bring about an order-of-magnitude cost reduction in the logistics support of large orbital space stations and other space operations planned by NASA for the late 1970s and the 1980s. In addition, the study was to provide analytical data for use in identifying development, manufacturing, procurement, and testing requirements for RDT&E and operational phases; identifying requirements for research and technology development; and contributing significantly to advances in safety methods. A third objective was to develop information beyond the scope of Phase A in the following special emphasis study areas:

- Reentry heating and thermal protection
- Approach and terminal landing
- Propulsion system parameters
- Vehicle/propulsion system interfaces
- Integrated electronic system
- Ferry operations

III. RELATIONSHIP TO OTHER NASA EFFORTS

As a result of transportation studies conducted since 1963, NASA has identified the Space Shuttle as a focal point for all other space operations with respect to earth-to-orbit-and-return logistics. Among the studied systems for which Space Shuttle may provide logistic support are Space Station/Base, Space Tug, Orbit-to-Orbit Shuttle, LM-B, Lunar-Base Module, Nuclear Shuttle, and unmanned probes.

IV. METHOD OF APPROACH AND PRINCIPAL ASSUMPTIONS

Computer programs closely coupled with analytical design were employed to reach a multiplicity of design solutions to the central problem of orbiter and booster sizing and to accommodate large variations in system requirements and design parameters. These programs aided in integration of interdependent ascent propulsion, lifting body aerodynamics and reentry trajectory, and subsystem characteristics of the vehicle. The central program of this series is called MAGIC. Early in the study, cost estimation subroutines were incorporated into the system synthesis programs to permit cost trades to be included with technical evaluation of alternate approaches. Throughout the study, these programs were continuously updated and refined from the results of conceptual configuration design, propulsion, aerodynamic, aerothermodynamic, trajectory, structural, and weight studies.

Following trade study evaluations, which led to final baseline requirements and design parameters, four baseline vehicles were sized on the basis of Two-Stage and dissimilar Triamese concepts and payload values of 25,000 and 50,000 pounds. Each of these baselines was then designed in sufficient depth to accomplish a meaningful comparative evaluation. The results of mission analyses, program planning, cost and schedule analyses, sensitivity analyses, technology evaluation, and the six special emphasis studies were also used in refining the conceptual designs to provide definition of an optimal Space Shuttle system. The final technical assumptions are reflected in the baseline requirements. The principal programming assumption was an initial operational capability in 1976 or 1977.

V. BASIC DATA GENERATED AND SIGNIFICANT RESULTS

Selected technical characteristics and trade studies displayed in this report substantiate selection of the baseline Two-Stage configuration with 50,000-pound payload capacity and 10,000-cubic foot volume. Principal conclusions of the study involve interrelated technical and management aspects, particularly the large system size and acute sensitivity to system requirements. Since the RDT&E cost impact of large design payload appears to be small, the best insurance against development risk would seem to be an approach based on a large system size with growth potential to accommodate advanced technology and with crossrange potential to provide flexibility for alternate missions. In assessing the total development program environment, it is essential that fundamental system requirements be defined with adequate margins and rigorously stabilized throughout development.

Baseline System Requirements

The ILRV baseline vehicles have been configured to meet all of the NASA desired characteristics delineated in this statement of system requirements. The Space Shuttle is to be operationally flexible and capable of accepting a large variety of payloads, either cargo only or including passengers and mission-peculiar crew members, to accomplish these mission types: Space Station/Base logistics; placement and retrieval of unmanned satellites; delivery of propulsive stages and payloads; delivery of propellants; satellite service and maintenance, and short-duration orbital missions.

Primary Sizing Parameters

System Concepts:
- Two Cases – Two-Stage and Dissimilar Triamese

Payload Capability (Up & Down):
- Two Cases – 25 K and 50 K

Primary Propulsion
- Bell type, high P_c, H_2/O_2
- 400 K SL thrust
- Sequential burn
- Three orbiter engines

Ascent Reference Orbit
- 45 x 100 nm, 55° incl.

On-Orbit ΔV
- Main tanks 858 ft/sec
- Orbit tanks 1142
- 2000 ft/sec*

Flight Performance Reserve
- 0.75% of ascent ideal velocity

Design Contingency
- 10% of all dry weight

Mission Duration
- 7 days self-sustaining
- 30-day capability

Payload Size
- 15 ft dia, 60 ft long
- 22 ft dia, 30 ft long

Crew Accommodation
- Two-man crew
- Cabin for four men*
- 10-psi atmosphere*

Reentry Crossrange
- Aerodynamic configuration: 1500 nm or more
- Thermal protection: 400 nm

Maximum Acceleration
- Ascent: 3 g with passengers
  - 4 g with cargo
- Reentry: 2 g

Go-Around Capability*
- Four airbreathing engines – all required
- JP-4 fuel*

*Areas of possible change in future designs

These requirements are the primary sizing determinants. With the exception of the 400-nm crossrange, they are the most severe requirements proposed for Space Shuttle system sizing.

While most of these requirements are those defined by NASA, some were established by LMSC on the basis of understanding of Space Shuttle requirements. These are as follows:

- Three orbiter engines, specified to meet the NASA fail-operational requirement (this has no effect on the 50,000-pound vehicles but imposes a small penalty on the 25,000-pound case.)
- The use of a payload bay for propellant storage, housing either a 15 by 60-ft or a 22 by 30-ft payload
- The four-man cabin, with the additional room available for mission-peculiar personnel when needed (These men could be housed elsewhere, and although used in the baseline the four-man cabin is subject to further study.)
- The requirement that the aerodynamic shape offers a potential for growth to 1500-nm crossrange (The use of a 400-nm heat shield saves 8,000 to 10,000 pounds of orbiter inert weight.)

Items considered for change in future design include a reduction to 1500 ft/sec on-orbit delta velocity, change to a two-man cabin, change to 14.7-psi atmosphere if this value is used for the Space Station, possible removal of go-around capability for operational vehicles, and the use of hydrogen rather than JP-4 for jet engine fuel. This last change might apply to booster fly-back as well as for go-around engines.

Additional Design Parameters

Docking Capability
- Piloted hard docking
- Automatic hard docking
- Shuttle-to-shuttle docking

Guidance and Control
- Autonomous capability
- Limited duration attitude restriction
- H_2/O_2 RCS
- One deg/sec² rotation
- One ft/sec² translation

Passenger/Cargo Accommodation
- Payload support out of payload capability
- Up to 50 people
- Shirtsleeve transfer
- Small cargo through hatches
- Large cargo – no EVA

Landing Capability
- 10,000-ft runway
- 160 knots or less touchdown

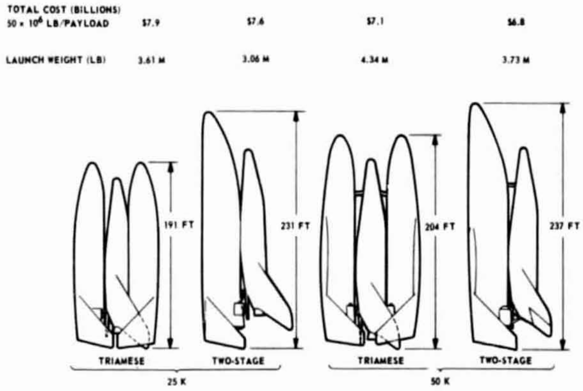
Reliability and Safety
- All systems: fail-operational, fail-safe
- Electronics: fail-operational, fail-operational, fail-safe
- No single engine-out dead band
- Intact abort
  - Primary mode through orbit
  - Alternate mode for early ascent-return to base
- Automatic landing capability
- Landing visibility comparable to that of high-performance aircraft
- Vehicle and GSE for rapid egress for launch pad abort
- Vehicle systems for safe egress after landing (no GSE required)

These requirements, along with those that determine vehicle size, complete the definition of the principal capabilities for which the ILRV baselines have been designed. Most parameters were defined by NASA. Those added by LMSC include the last three under guidance and control, the limit of 160 knots touchdown velocity, and the definition of intact abort modes.

Baseline Evaluation

ILRV Baselines

Sketches of the final four ILRV baselines are arranged here in the order of decreasing total cost to deliver 50 x 10^6 pounds of payload over a 10-year operational life. This is the equivalent of 100 flights per year for the 50,000-pound payload systems with full payload on each flight. There are significant advantages of the Two-Stage over the Triamese in reliability, safety, and operations. Parametric cost studies show that any possible RDT&E cost saving to develop a somewhat smaller booster for Triamese can be expected to be offset by the requirement for more boosters for development testing. Triamese recurring costs would exceed those of Two-Stage by almost 20 percent.



Estimated Weight Growth (Acquisition Phase)

To determine the possible program effects of inert weight growth, these worst-case estimates of possible growth were based on the history of growth of space vehicle systems. First-generation systems have typically grown 35 percent during the acquisition phase; so first-generation portions, such as the heat shield, have been given this factor. Other portions of the system have been given lower factors, appropriate to their degree of maturity.

The 16.2 percent weighted average growth for the booster and 17.7 percent for the orbiter are considered worst case, because the history applied was based on space vehicle systems only. The Space Shuttle incorporates many airplane features as well as space system features, and airplane growth has not been as severe. For instance, the C-5A growth has been less than 2 percent. It should be noted, however, that it has been very expensive to maintain C-5A growth within this value.

BOOSTER

	Percentage of Dry Weight	Estimated Growth Factor
Wing	18.0	1.15
Body Structure	26.0	1.20
Environ. Protection	8.4	1.35
Interstage Structure	4.7	1.15
Landing Gear	4.6	1.06
Rocket Engines	18.2	1.15
Jet Engine and Nacelle	10.0	1.06
Rocket Engine System	3.7	1.06
Jet Engine System	1.0	1.06
Orientation and Control	1.4	1.35
Electronic System	0.6	1.15
Environmental Control	0.6	1.06
Power System	2.8	1.06
Total Dry Weight	100.0	1.162

ORBITER

	Percentage of Dry Weight	Estimated Growth Factor
Aero Surfaces	7.4	1.15
Body Structure	25.8	1.20
Environ. Protection	19.2	1.35
Interstage/Dock Struct.	3.0	1.15
Landing Gear	5.6	1.06
Rocket Engines	6.8	1.15
Jet Engine and Nacelle	11.3	1.06
Rocket Engine System	14.4	1.10
Jet Engine System	0.5	1.06
Orientation and Control	1.1	1.35
Electronic System	1.1	1.15
Environmental Control	1.0	1.06
Power Systems	2.8	1.06
Total Dry Weight	100.0	1.177

Payload Effect of Weight Growth (50K Two-Stage)

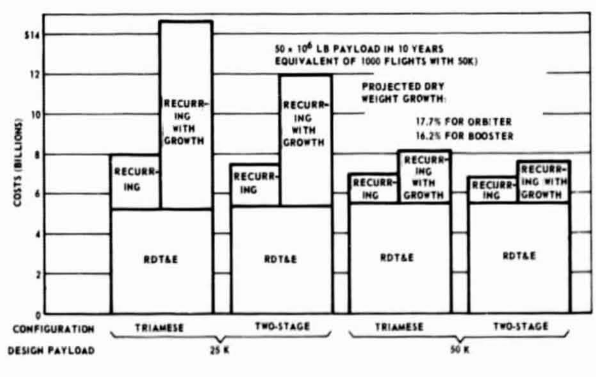
	BOOSTER	ORBITER	TOTAL
Dry Weight	322,550	183,830	
Estimated Growth	52,200	32,500	
Payload Effect	-7,900	-32,500	-40,400
10% Contingency	32,300	18,400	
Net Growth	19,900	14,100	
Net Payload Effect	3,000	14,100	-17,100
Net Payload			32,900

Program Cost Comparison

Parametric cost estimates were used to evaluate the development risk associated with possible weight growth. The 50,000-pound Two-Stage RDT&E cost is $5.51 billion; adding $1.26 billion recurring cost for 1000 flights gives $6.77 billion to deliver 50 x 10^6 pounds of payload with no growth effect. If the worst-case estimated growth occurred, payload capability would be reduced to 32,900 pounds and the recurring cost to deliver the same total payload would be $1.91 billion, giving a total of $7.42 billion. This penalty could be considered marginally acceptable. The corresponding penalty for the 50,000-pound payload Triamese would be significantly greater, and that for either of the 25,000-pound payload systems would be clearly unacceptable.

The relatively small additional RDT&E cost for a 50,000-pound payload system is a very good investment to ensure against heavy potential penalties that could result from inert weight growth.

The evaluation of the four ILRV baselines results in the conclusion that the 50,000-pound payload Two-Stage system is significantly superior to the other three.

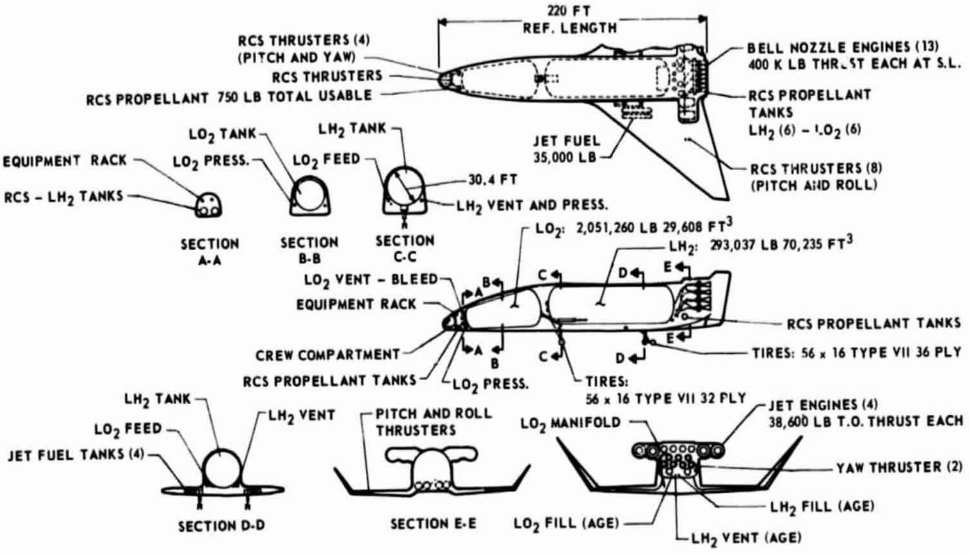


Configuration Features

The 50, 000-pound payload Two-Stage booster baseline design incorporates a number of desirable features. For example, propellants are in efficient integral tanks; the hydrogen tank is cylindrical and the oxygen tank is conical. The fixed delta wings, which are of dry structure, carry the landing gear and non-integral tanks for the flyback jet fuel. The thirteen 400, 000-pound thrust engines are mounted high to minimize center-of-gravity offset effects with the orbiter attached. The four air-breathing engines are on fixed mounts at the rear so will not affect flow over the body and wings.

Some weight penalty may be imposed by the use of fixed delta wings rather than straight wings or variable-geometry wings, but the lower risk with this approach is considered worth the penalty. Low-risk reentry heating results from minimizing interference effects as compared to the straight-wing case and from the relatively low reentry planform loading as compared to the variable-geometry case. The proven high risk of variable-geometry system weights and the existence of an additional failure mode (failure to deploy) is also avoided.

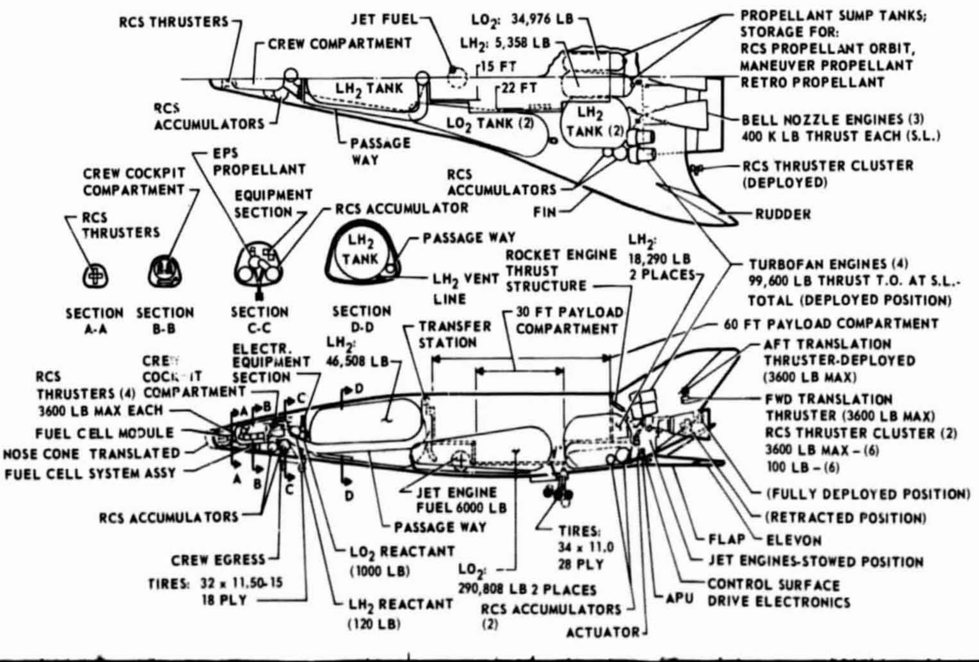
BOOSTER



The delta lifting body orbiter design affords efficiency in packaging propellants as well as desirable aerodynamic characteristics in all three flight regimes. Acceptable mass fraction is obtainable with nonintegral tankage, and this results in less manufacturing risk and in much better inspection and maintenance characteristics than would be the case with an integral tank design. The relatively large fins improve the subsonic lift-to-drag ratio as well as hypersonic stability. They also make it possible to mount the jet engines at the rear, where they will not affect the flow over the lifting body.

A promising alternate is a cylindrical body with delta wings and with a very similar planform. The propellant tankage would be simplified, but the vehicle would have to be made longer to contain the propellants. The overall weight effect would involve a penalty, but the subsonic lift-to-drag ratio would be improved from 5 to about 7. Approach and landing studies indicate that 5 is an adequate lift-to-drag value; but if improved subsonic characteristics are desired, the delta wing approach appears to be the most attractive alternative to the baseline.

ORBITER



Probability of Crew Survival

With the total mission probability of success of 0.995, a probability of success (P_i) is allocated for each of seven mission phases. The Q_{2i} values represent the probability of catastrophic failure in each mission phase, obtained by considering 12 different types of hazards. Of the total 128.6×10^{-6} occurrences per mission, the dominating source is rocket engine catastrophic failure, which accounts for 46.0×10^{-6} occurrences per mission. Q_{1i} is abort decision probability, that is, probability that decision is made in i^{th} phase to change objective of flight from "mission completion" to "safe return." This is the balance allocation remaining ($Q_{1i} = 1 - P_i - Q_{2i}$). The calculation of crew survival probability, shown at the bottom of the chart, is based on

$$P_{cs} = M_5 + \sum_{i=1}^{5} M_{i-1} \times Q_{1i} \times R_{Ai} ,$$

where

$$M_k = \prod_{i=1}^{k} P_i = \text{probability of completion of } k^{th} \text{ phase}$$

$$R_{Ai} = \text{reliability of abort equipment for } i^{th} \text{ phase}$$

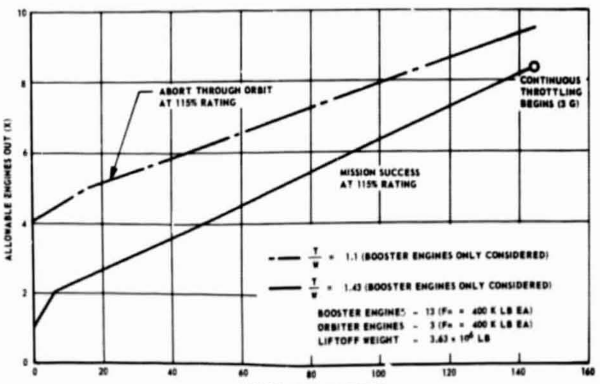
The message here is that the 190 catastrophic failures per million flights, which is the difference between P_{cs} and 1, may be considered as a failure rate goal for the Space Shuttle system.

TERM \ PHASE	1 PRE-LAUNCH	2A ASCENT 0 TO 20 SEC	2B ASCENT 20 TO 120 SEC	2C ASCENT TO INJECT	3 ON-ORBIT	4 REENTRY	5 APPROACH AND LANDING
M_i	0.999950	0.999825	0.999700	0.999500	0.995659	0.95300	0.99500
$Q_{1i}\left(\times 10^{-6}\right)$	40	108.7	70.8	188.7	3832	341	290.2
$Q_{2i}\left(\times 10^{-6}\right)$	10	16.3	54.2	11.3	18	9	9.8
P_i	0.999950	0.999875	0.999375	0.999800	0.996150	0.999650	0.999700
R_{A_i}	0.99	0.90	0.96	0.995	0.995	0.98	0.96

$$P_{cs} = 0.995 + 0.004810 = 0.999810$$

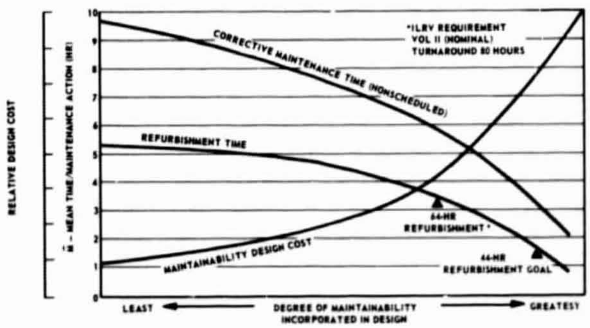
Allowable Engine Out

Since the engine operation is of major concern, a plot was made to determine the number of booster engines that can fail non-catastrophically and still achieve either mission success or an abort through orbit. As an example, at 10 seconds from lift-off two engines can be lost and the system can still achieve mission success or full mission capability if the remaining engines are increased to 115 percent rating. Alternatively, an abort through orbit can be achieved by increasing the engines to 115 percent rating if as many as five engines are lost at 10 seconds.



Maintainability/Refurbishment

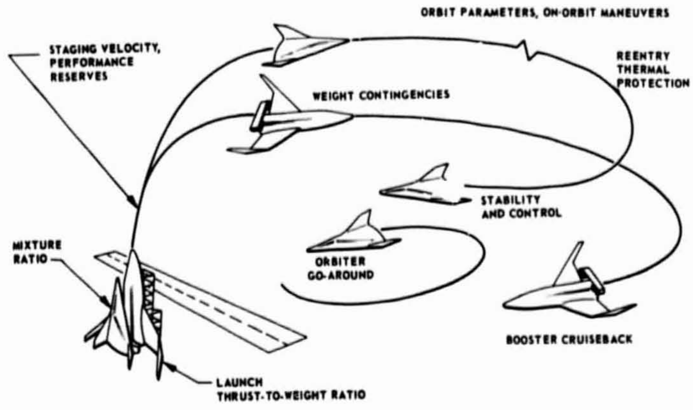
Low inherent maintenance time means higher design cost because of incorporating design features that may not have any influence on the function. However, maintainability cost effective analysis indicates that it is possible to achieve the 44-hour refurbishment goal, which is below the 64-hour Space Shuttle requirements.



System Synthesis

MAGIC System Synthesis Approach

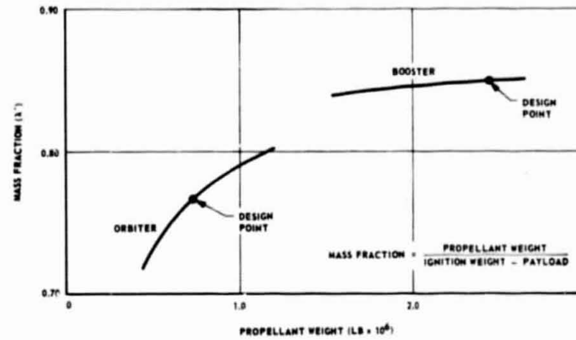
MAGIC is a system synthesis computer program developed by LMSC. It was used throughout the study to model the interrelated effects of various parameters in determining launch system size. Key sizing parameters can be identified with each mission phase of both orbiter and booster operation, and these influences lead to interdependent w ght and size determination as a function of system requirements. As an example, orbiter thermal protection system weight is strongly influenced by wing loading, which in turn is derived from lifting body planform geometry in relation to weights of primary structure and subsystems involved in performance of both ascent and on-orbit mission functions. These include many nonpropulsive spacecraft housekeeping, mission support, and life support requirements. The booster size reflects interaction with all of the orbiter sizing requirements and constraints; and both system elements are sized by means of incremental weight buildup, subsystem by subsystem. This program has been in use for approximately 2 years, and excellent agreement between computer and detailed analytical results have been achieved through continuous refinement of the computer model with configuration layouts and aerodynamic, propulsion, structural, and subsystem analytical designs.



Variation of Mass Fraction With Propellant

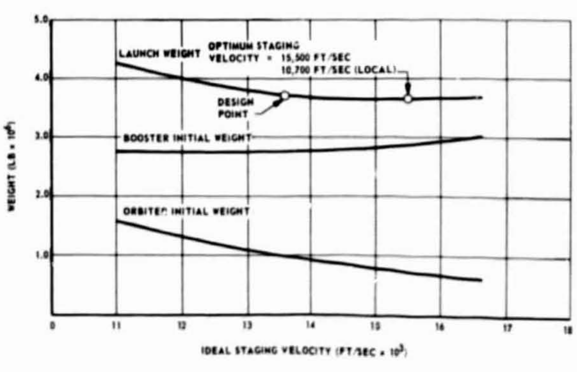
The variation of mass fraction was accomplished through an actual incremental vehicle weight buildup by subsystem by use of MAGIC. In the case of the booster, the mass fraction is basically insensitive to size over the propellant range of interest. The orbiter, however, shows a marked variation in mass fraction with propellant load, resulting from the large payload bay and the fixed weight of subsystems required for its multiplicity of functions.

In the hypothetical approach to determining the effect of staging velocity on launch system weight, the mass fraction for the orbiter and booster developed for the design point is held fixed and independent of stage sizes or staging velocity. Results show that the launch vehicle size is extremely sensitive to staging velocity and the optimum staging point occurs at an ideal velocity of 18,750 ft/sec (13,750 ft/sec actual). The discrepancy between the actual baseline design point and the idealized staging point is due mainly to the hypothetical assumption of fixed mass fraction. The basis for selection of a design point at an ideal velocity between 13,000 and 14,000 ft/sec is shown in the variable mass fraction staging effects in the figure below.



Variable Mass Fraction Staging Effects

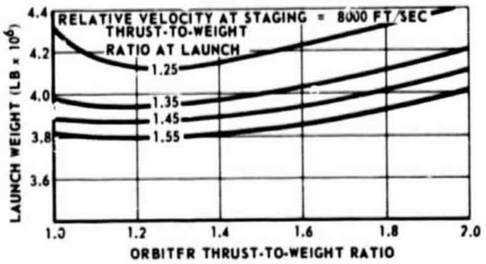
Applying a weight buildup technique for mass fraction results in significantly different conclusions as to the effects of staging velocity than would be derived from the less realistic assumption of fixed mass fraction. As indicated, the launch system size is not sensitive to staging velocity. The ideal staging velocity for minimum launch weight is 15,500 ft/sec, as compared to 18,750 ft/sec for the fixed mass fraction study. The final issue to be considered is the implication of staging velocity on stage size. As staging velocity increases, the orbiter size is reduced. This difference represents a reduction of 6 percent in spacecraft length between the design point and the optimum. The difference, however, is significant from the standpoint of payload packaging; the smaller vehicle would require a more forward location of the payload bay. This in turn would create a less desirable structural arrangement for accommodation of the large payload bay. The orbiter for the baseline system was therefore selected on the basis of design considerations and involves little penalty from the optimum staging case.

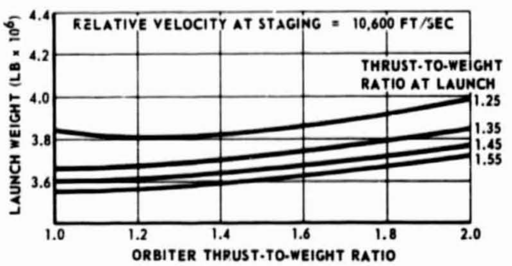


Effect of Thrust-to-Weight Ratio

The orbiter, with three 400, 000-pound thrust rocket engines, would encompass a range of thrust-to-weight ratios of 1.4 to 1.8. Since these values imply minimum launch system weights for the expected range of staging velocities, a compatible orbiter design has been accomplished.

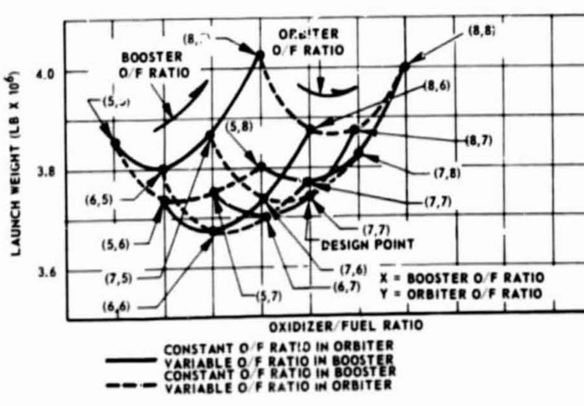
In the case of booster thrust-to-weight ratio, however, two significant effects have not been incorporated. These are the effects of thrust-to-weight ratio on structural weight and on cruiseback lift-to-drag ratio. These two effects can easily override the implications shown in the figure as to the benefits of increasing booster thrust-to-weight ratios. The issue of booster thrust-to-weight ratio therefore remains open until more definitive structural and aerodynamic analyses are completed.



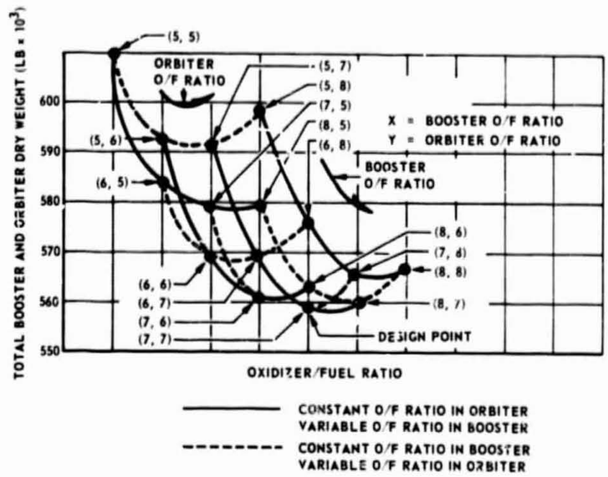


Effect of Oxidizer-Fuel Ratio

The study of the effect of propellant mixture ratio is one in which structural efficiency due to changing propellant bulk density is compared to propulsive efficiency (specific impulse). As shown, there is a minimum launch weight for mixture ratios of 6:1 in both the orbiter and the booster. The difference between 6:1 and 7:1 is small, reflecting a change in launch weight on the order of 1-1/2 percent.

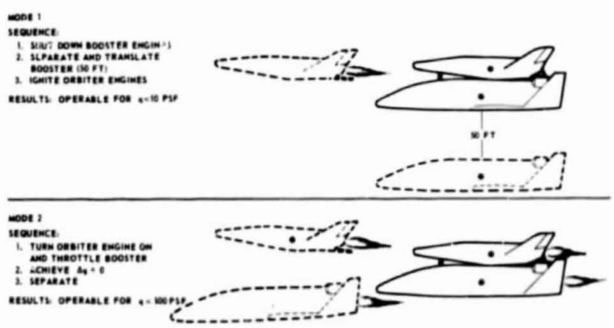


A second measure of launch system size is the dry weight of the launch system. Here minimum dry weight occurs at mixture ratios of 7:1. Again, the weight difference between 6:1 and 7:1 is on the order of 1-1/2 percent. Since for mixture ratios between 6:1 and 7:1 there is little effect of the selected ratio on both launch weight and dry weight, it must be considered from the standpoint of vehicle performance that mixture ratio is not a key issue. However, since weight growth is expected and weight is proportional to volume, the high mixture ratio (7:1) would be the preferred choice.



Staging Analysis

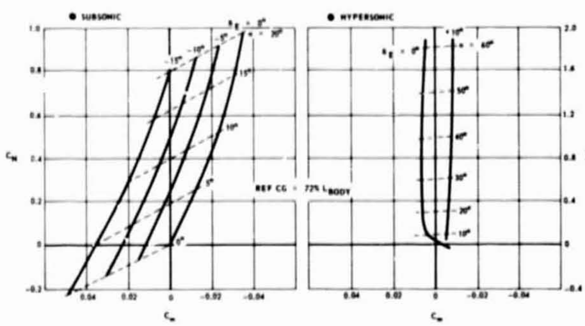
Two staging modes have been examined analytically. In simulation, estimates of aerodynamic interferences were used for the induced interstage loads. The first mode, which is considered nominal, is operable for normal staging at low dynamic pressures. The second mode is operable at higher dynamic pressures, but in it the vehicles may rotate to larger angles of attack (controls fixed). The large fuel penalties involved may relegate this mode to abort situations.



Aerodynamics and Performance

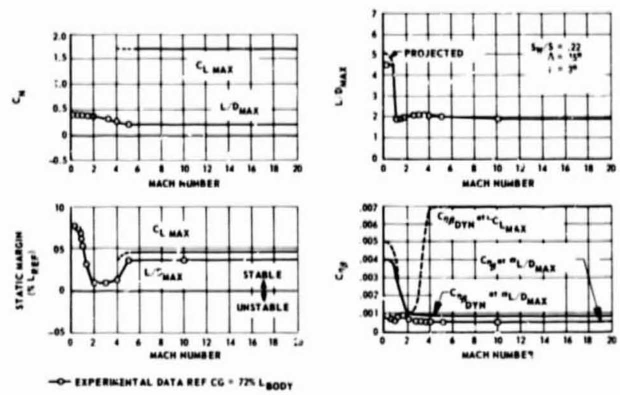
Booster Longitudinal Stability (Estimated Data)

These data are based on estimates of the booster subsonic and hypersonic longitudinal stability. The vehicle is stable and trimmable at subsonic speeds and neutrally stable at high speeds. Initial test data now being obtained at the MSFC 14-inch Trisonic Wind Tunnel tend to verify these estimates.



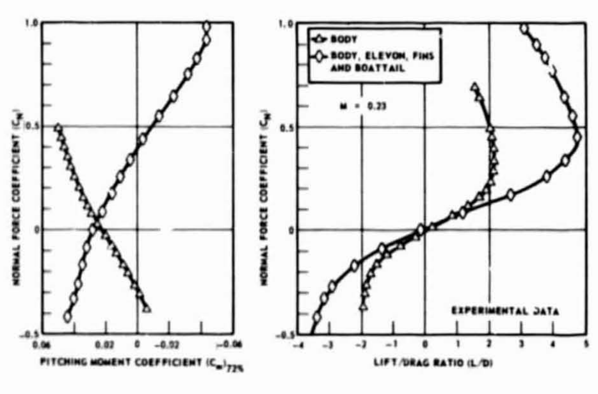
Orbiter Trim Characteristics

Summary trends of the major aerodynamic characteristics indicate that the orbiter is capable of stable trim at all Mach numbers and at high entry angles of attack. All of the data for trimmed L/D maximum conditions are based on wind tunnel test results; therefore there is high confidence in the validity of this configuration.



Configuration Buildup

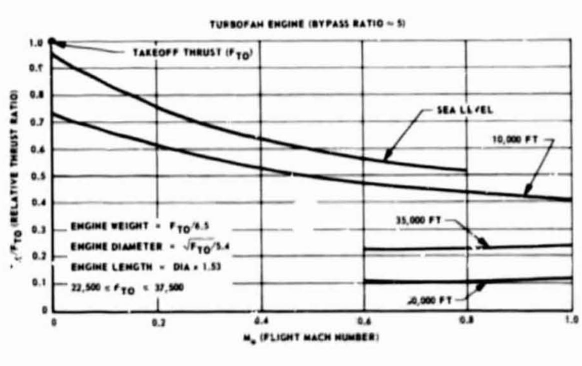
The lifting-body LMSC orbiter receives substantial increments in lift and longitudinal stability from the addition of elevons and fins. The fins, highly swept to reduce leading edge heating, provide the aft lift necessary to develop a 4-percent trimmed static margin at a reference center of gravity of 72 percent body length. The fin-reduced lift and boattail-reduced drag also permit experimentally measured trim L/Ds of 4.7 to be achieved. It is confidently expected that a best-on-best buildup of already tested configuration elements plus further turning will enable realization of trimmed subsonic L/Ds of 5 or greater.



Estimated Thrust Performance

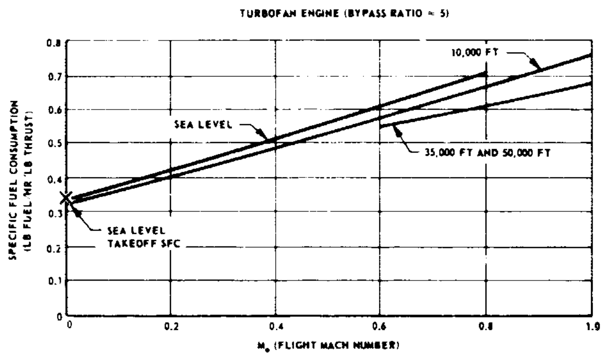
Establishment of jet engine weight is developed from the operating thrust required and operating condition. The figure shown is used to convert these conditions to a takeoff static thrust (takeoff thrust = thrust required/$\left(\frac{Fx}{Fto}\right)$, whereupon the engine is then weighed and installation penalties are applied.

These same data have been used for sizing the cruiseback engines on the booster.



Estimated SFC Performance

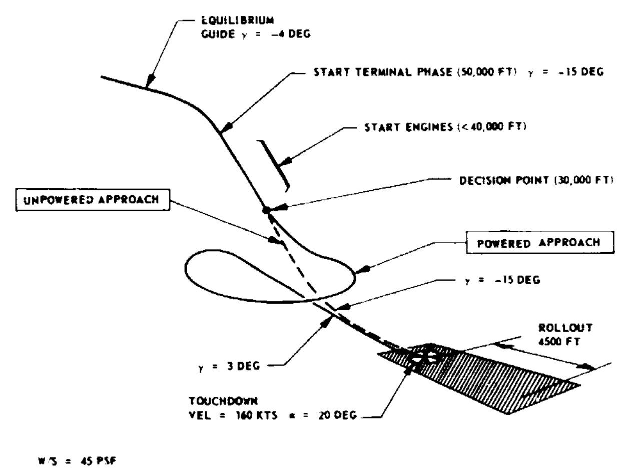
The data shown reflect the model used in determining the fuel requirement for the various operating conditions that may be applied to the performance envelopes of either the booster or the orbiter.



Typical Orbiter Terminal Flight Phase

An important area for further study is the approach and landing technique. The concept shown indicates a decision point at which either a poweroff direct approach, at a flight path angle of about 15 degrees, or a powered airline-type approach could be chosen. The original intent of this concept was to allow for the power-off approach in case the jet engines failed to start. However, experience in lifting body and low L/D aircraft landing approaches indicates that use of the steeper approach may be better. The rationale is that, by using the steep approach, the runway is made all the way in from the decision point. which may be preferable to committing to a shallow turning approach and depending on engines that have been running only about 2 minutes at the decision point. It is a fail-operational, fail-safe matter. Engine failure on the low flight path angle approach is not considered fail-safe.

A major significance of this consideration is that, if a power-off approach is preferred, later removal of the go-around requirement could mean complete removal of the jet engine system. This removal, which could be considered after sufficient confidence in the landing systems and techniques has been developed, would result in a payload improvement of about 27,000 pounds. Some confidence is already being developed through the Air Force testing of the F-111. This type of approach on instruments (making use of the F-111 inertial system) at low lift-to-drag values simulates the approach angle of the delta lifting body orbiter. Experience indicates spot landing capability every time within a few hundred feet.



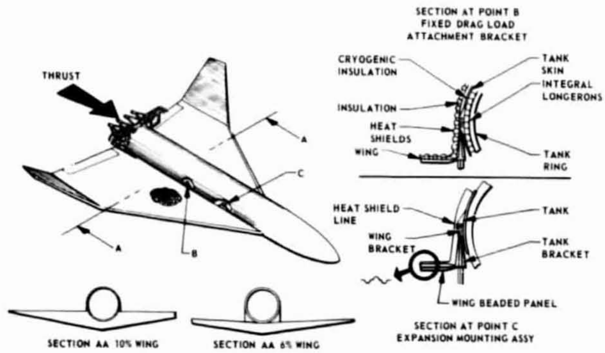
Effect of Jet System Criteria (Standard Day, Four-Engine Operation)

In the case of glide slope improvement, a lift engine exhibiting the lightest weight for a given thrust requirement would be selected. The short operating time (less than 6 minutes) allows specific fuel consumption to be of secondary importance. For go-around, marked improvement in specific fuel consumption, combined with the operating time, makes the turbofan the best candidate. This situation holds for cruise and ferry modes.

A go-around profile defined for these calculations was based on pullup to maximum rate of climb at maximum L/D, 180-degree turn at an altitude of 1000 feet and climb to 2000 feet, rollout to 5 nm downwind leg, and 180-degree turn to final approach.

The final selection of jet system criteria will severely impact the launch system size and operational requirements, such as possible provision of an engine kit for ferry operations.

	IMPROVED GLIDE SLOPE	GO-AROUND	CRUISE 400 NM	FERRY 400 NM
Power Setting	Takeoff	Takeoff	Max Cont.	Takeoff
Operating Altitude (Feet)	Sea Level	3,000	10,000	Sea Level
Selected Engine	Lift Fan	Turbofan	Turbofan	Turbofan
Number – Takeoff Static Rating (Pounds)	4–10,000	4–25,000	4–25,000	4–40,000
Installed Engine Weight (Pounds)	3,200	20,600	20,600	33,000
Fuel (Pounds)	4,700	6,000	46,200	49,000
Total System Weight (Pounds)	7,900	26,000	66,800	82,000

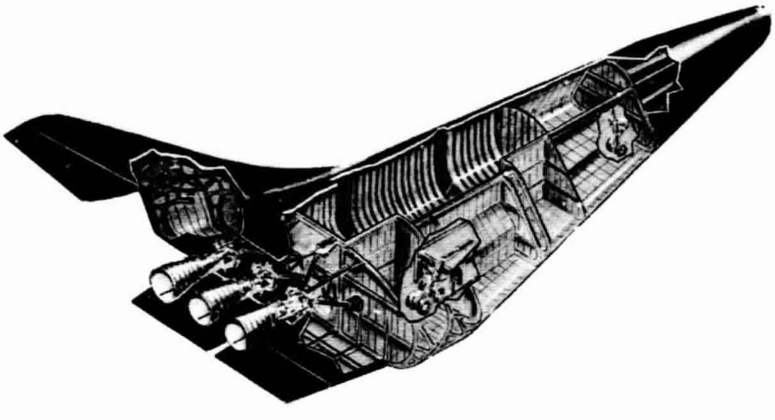


Booster Structural Arrangement

The engine thrust loads and fuselage bending moments can be accepted by an aluminum load-carrying fuel tank. The wing attachment provides forward and aft flexible mounting assemblies to allow for the movement of the fuselage fuel tank in relation to the dry wing. The wing thickness could be increased to accommodate wing tankage.

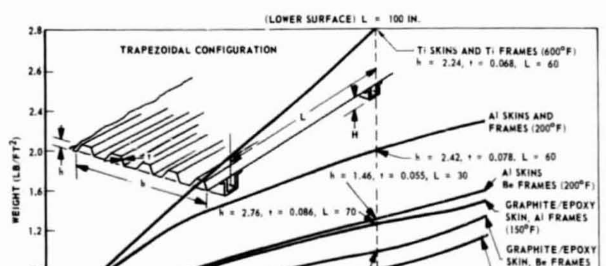
Orbiter Structural Arrangement

The leading orbiter concept is a delta lifting body. Characteristic design features are the large unobstructed payload bay in midbody; the deep aft section frames, which carry the integral fin spars and the concentrated booster and primary rocket engine thrust loads into the main longerons and thence to the integrally stiffened skin, stringer, and frame members of the forebody; and the subdivision of the forebody into tankage and equipment bays by the webs of the major subframes of the vehicle, which also bear the landing loads. Typically, the subframe webs are penetrated with cutouts for the nonintegral main propellant tanks and propellant feed systems. An advantage of the canted fins is substantial static margin of stability, as a result of the lift produced at the aft station. This permits fixed location of the turbojet engines within the after body section. This, in turn, permits using the aft deep section frames to carry the turbojet engines as well as the primary rocket engines, resulting in less weight penalty.



Variation of Orbiter Primary Structure Weights

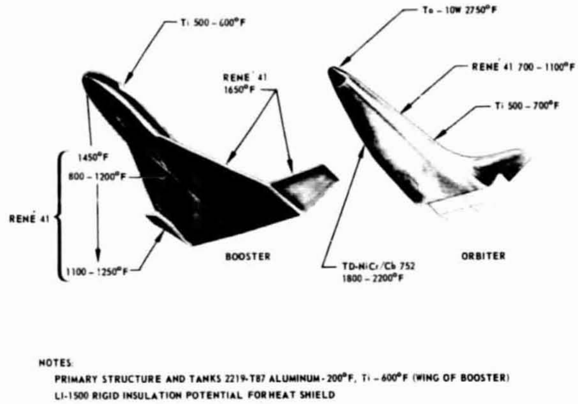
At a maximum αq design condition for the lower surface, the load intensity is 3000 lb/in.; and a titanium structure (600°F) is about 0.8 lb/ft^2 heavier than the baseline aluminum structure (200°F). Panels of the graphite/epoxy composite (150°F) with beryllium frames are about 1.0 lb/ft^2 lower in weight than aluminum panels. Beryllium is about 1.2 lb/ft^2 lower in weight than aluminum. While the cost and development risk are higher, beryllium and the composites warrant serious consideration because of the significant potential weight savings they afford.



Summary of Materials and Predicted Temperatures

Either LI-1500 or titanium (upper surface) and René 41 (lower surface) can be used for the booster heat shields. Either LI-1500 or a combination of TD-NiCr/Cb-752 (lower surface), René 41 (forward upper surface), and titanium (aft upper surface) can be used for the orbiter heat shields.

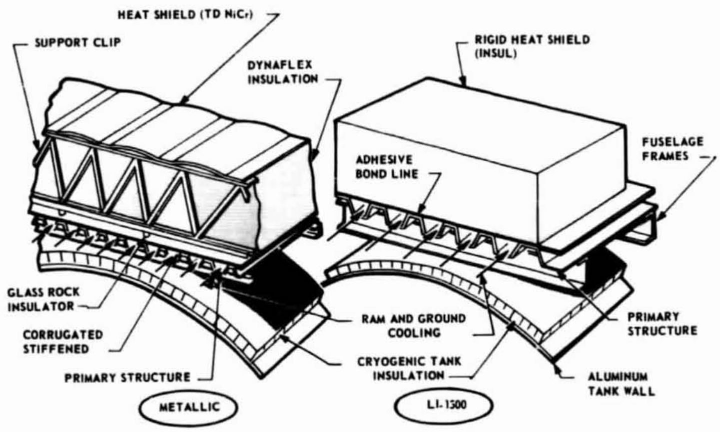
LI-1500, which was used for the ILRV sizing and costing, is an advanced, resulable, lightweight insulation material being developed for potential application in reentry vehicles. Because of its low cost, weight advantages, and its 2500°F capability, it appears to be suitable for use in all Space Shuttle surfaces except the orbiter nose.



Typical Heat Shield Approaches

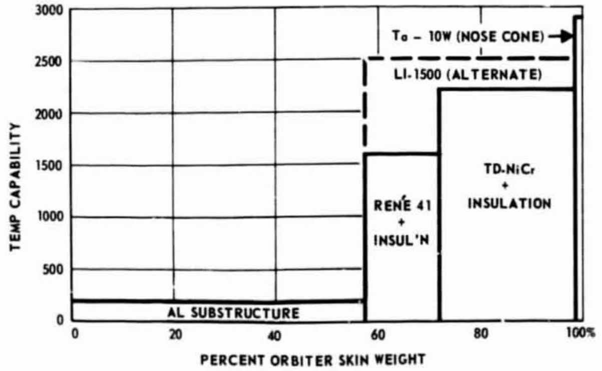
Recent studies indicate that large corrugated heat shields with multiple clip supports are lighter in weight than post-supported integrally stiffened heat shields. A flat provided between corrugation arcs enables attachment of the continuous support clip. Mechanical fasteners and resistance spot welding are used to attach the TD-NiCr and René 41 corrugated heat shields. The clips are attached by glass rock insulators to the primary aluminum structure. Blanket-type insulation of dynaflex or microquartz is packaged between the shield and the structural panel.

The basic approach permits keeping options open. If LI-1500 were used, it would be bonded to the primary structural panels or, in alternate designs, which appear more attractive, to secondary removable panels designed for about 200°F higher temperature than the primary structure.



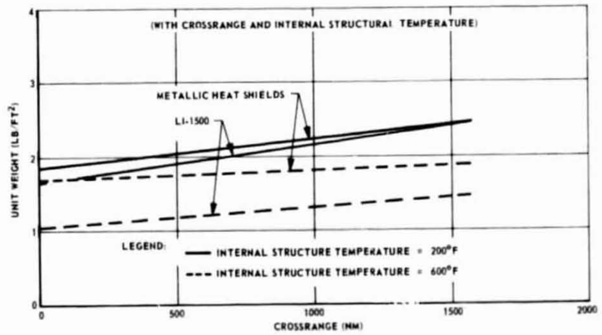
Orbiter Skin Structural Materials

The aluminum substructure of the orbiter skin, which carries the primary loads, is maintained at a temperature below 200°F by the heat shield, representing over 40 percent of the vehicle skin weight. If a metallic heat shield is used, the upper surface and sides (about 50 percent of the total surface) are covered with René 41, while the lower surface and leading edge surfaces are of TD-NiCr. If LI-1500 is used, as in the baseline heat shield, it serves both as insulation and surface material. The nose cone, with a predicted maximum temperature of 2750°F, consists of a tantalum-tungsten alloy, Ta-10W.



Variation of Orbiter Heat Shield Weight

For a maximum structural temperature of 200°F the metallic and LI-1500 heat shield weights are competitive. However, for the 600°F structural temperature (titanium) the LI-1500 weight is lower because the fibrous insulation of the metallic heat shield has approximately the same conductivity as LI-1500 (but a lower density and a lower weight decrease with backface temperature). The increased titanium structural weight may cause the total thermal-structural cross-section for heat shield concepts with aluminum and titanium primary structures to be of approximately the same weight.



Engine Characteristics

Both the aerospike and the bell-type engines were examined in various phases of the study; however, the information presented here is for the bell-type engine only.

The approach was to provide the desirable commonality in the orbiter and booster engines with regard to turbo-machinery and thrust chamber, with the variations in the nozzle. The thrust of the booster engine with an optimized 35:1 area ratio nozzle was established at a sea level thrust of 400,000 pounds. While the baseline uses a mixture ratio of 7:1, a nominal mixture ratio at any value between 6:1 and 7:1 is acceptable.

Using an optimized nozzle for the booster rather than the 35:1 base of an extendible (orbiter) nozzle increases the sea level specific impulse by about 13 seconds. The vacuum specific impulse when the booster engine power head is used is increased by using a 35/150 or 100/200 area ratio nozzle on the orbiter engine. While a 35/100 nozzle was used in the baseline configuration, the 100/200 case is recommended. A specific impulse increase of 3.5 seconds is achieved by using the 100/200 nozzle over the 35/150 nozzle. The stowed and extended lengths of the 100/200 nozzle are 128 in./256 in. as compared to 217 in./270 in. for the 35/150 nozzle. This significant length reduction is achieved with only a 2 percent increase in weight and an 11.5 percent increase in dimeter of the 100/200 nozzle over the 35/150 nozzle. The increase in diameter does not increase the orbiter base area, since the propellant tank configuration is the determining factor in the base area.

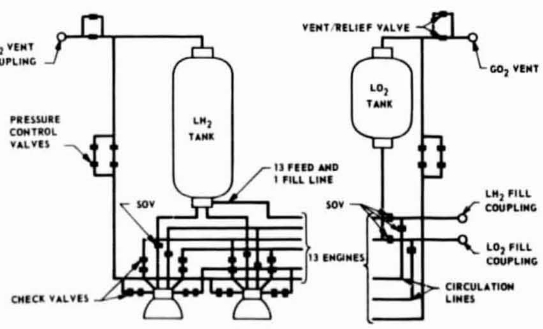
Booster Propellant System

The booster propellant system has only one liquid oxygen tank and one liquid hydrogen tank. The propellants are used rapidly after launch; therefore the propellant system requirements are very similar to those for the Saturn V. The propellant tanks, which may be insulated with foam-type insulations, are pressurized prior to launch. Pressurization gas is supplied during ascent by means of engine bleed from only three to five engines.

The 22-1/2-inch diameter feedline from the liquid oxygen tank becomes two 15-inch diameter manifolds for lateral distribution in the vehicle base area. The liquid oxygen lines from this manifold to the engines are 7 to 8 inches in diameter.

The liquid hydrogen lines, which are approximately 8 inches in diameter, are connected directly from the tank to the engines.

The propellant conditions in the feedlines are maintained through recirculation from a ground supply system.



Orbiter LO_2 System

Attaining the relatively high propellant fraction characteristics of the baseline design necessitates a multiple-tank system.

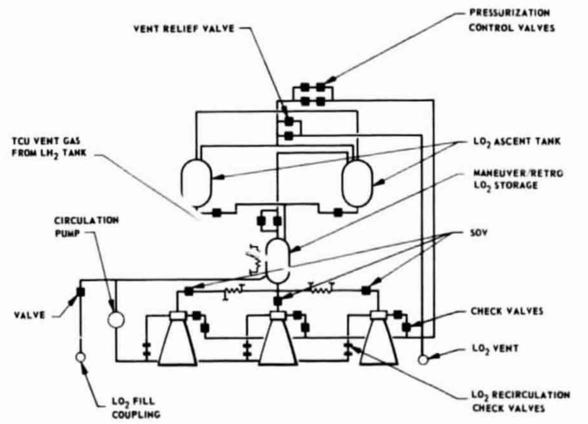
Since propellants for ascent are expended rapidly, there are no requirements for extended storage. Therefore, as in the case of the booster propellant system, these ascent propellants may be contained in tanks with foam-type insulation. The tanks may be pressurized prior to launch. Pressurization gas is supplied by the orbiter engines during ascent.

Propellants for orbital transfer, maneuver, and retro must be stored for extended periods of time in tanks with multilayer insulation.

Weight savings in plumbing and residuals are possible through directing the propellants from the ascent tanks through the orbital storage tanks and then to the engines. The orbital storage tanks can be made slightly oversize to assure that the ascent tanks are emptied.

Flow from two liquid oxygen tanks is directed through the orbit storage stanks. A venting system assures that during fill and ground hold, when the tanks are pressurized and boiling is suppressed, possible gas regions or bubbles are eliminated in the orbital storage tank.

Propellants are recirculated by use of a pump during ground and ascent to assure that proper conditions are maintained in the feedlines during ascent.



Orbiter LH_2 System

The liquid hydrogen propellant system for the orbiter is very similar to the liquid oxygen system. Propellant from the forward ascent tank is directed to the other ascent tanks and then to the orbital storage tank. This arrangement reduces the complexity of plumbing and manifolds considerably.

The storage tank has a thermal conditioning unit, which accepts either liquid or gas, passes this through an expansion valve, and cools the liquid hydrogen and oxygen.

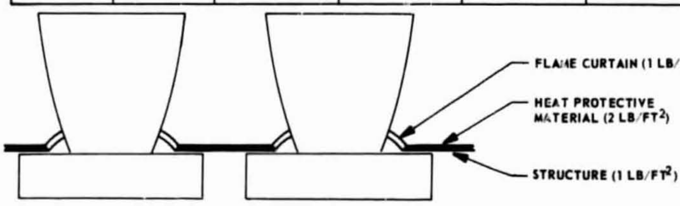
Base Heating and Thermal Protection

Plume impingement and base heating were among the items analyzed. It was found that the temperatures produced on the orbiter control surfaces at full thrust are less than 2500°R. Since the control surface can be moved out of the way in vacuum and the engines will be throttled in the flight profile, it was concluded that this problem could be minimized. (The booster is not subject to plume impingement effects.)

The most serious heating problem results from the booster base heating. The base heating estimates are consistent with Saturn technology.

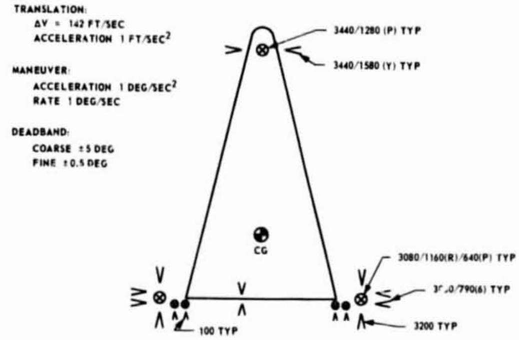
A possible nonresuable thermal protection system consists of reinforced silicone elastomers and a honeycomb structure. A possible reusable thermal protection system could be produced from coated tantalum.

	NUMBER OF ENGINES	LOCATION MEASURED FROM NOZZLE EXIT PLANE, IN.	CONVECTIVE HEAT RATE BTU/FT² SEC, COLD WALL (T_W = 70°F)	COMBUSTION EFFECT ON BASE ENVIRONMENT, BTU/FT² SEC	TOTAL HEAT BTU/FT² SEC
BOOSTER	13	68	10	10 – 30	20 – 40
ORBITER	3	208	<2.5	SMALL	<5



Orbiter RCS Thruster Arrangement

Shown is a consolidation of NASA and DOD requirements. The throttled thrusts result from the angular acceleration rate of 1 deg/sec. Throttling is considered to be more desirable than pulsing the engines at this thrust level. The minimum impulse bits of the engines are large, and the specific impulse would be degraded by pulsing. The minimum impulse bit of the large engines is too great for efficient limit cycling, so smaller thrusters (with thrust levels of 100 lb or less) were selected.



The selected arrangement provides for 18 large thrusters for maneuvering and translation and 12 smaller thrusters for limit cycling. The thruster area ratios are 30:1.

Orbital Operational Mode

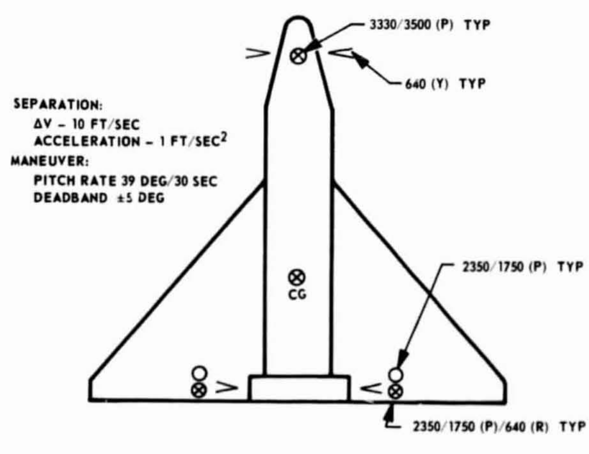
The orbital transfers, maneuvers, and retros may be performed with one engine at 10 percent thrust or lower or by the reaction control thrusters. Five engine restarts are required for the reference logistics mission. Tradeoffs associated with the three modes of obtaining a ΔV of 2000 ft/sec in five restarts are very sensitive to the specific impulse and cooldown propellant assumptions. (These findings are based on the most recent results obtained from engine contractors.)

Mode	Specific Impulse (sec)	Impulse Propellant (lb)	Other Propellant (lb)	Total (lb)
Pumped Idle	+66.5	36,240	2500	38,740
Unpumped Idle	–	40,760	80	40,840
RCS Thrusters	+6	40,480	3520	44,000

The pumped idle (or throttled) mode of operation affords an advantage in that engine bleed is available for tank pressurization; this is not available in the unpumped idle mode. The pumped idle mode requires that liquid propellants be provided to the engine immediately after cooldown. This suggests some type of propellant orientation system. Though not essential to the pumped idle mode, an unpumped idle (or pressure fed) mode would be an aid in that the cooldown propellants could be burned to gain this impulse and to assist in the orientation of propellants during engine start. The use of the reaction control system at maximum thrust for these maneuvers would require continuous supply of propellants at a rate of approximately 25 lb/sec. The operating time required on the individual nozzles would be approximately 1600 seconds for these extra operations.

Booster RCS Thruster Arrangements

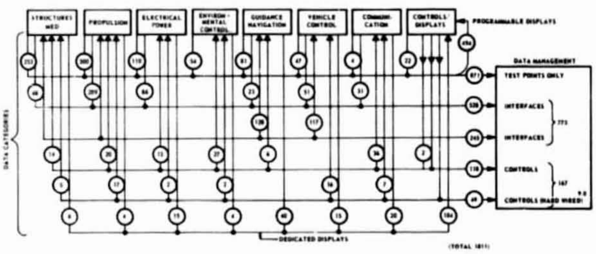
The basic reaction control system for the orbiter appears to be suitable for the booster also. The differences are that only 12 thrusters are used in the booster system and the area ratio is reduced to 20:1.



Integrated Electronics

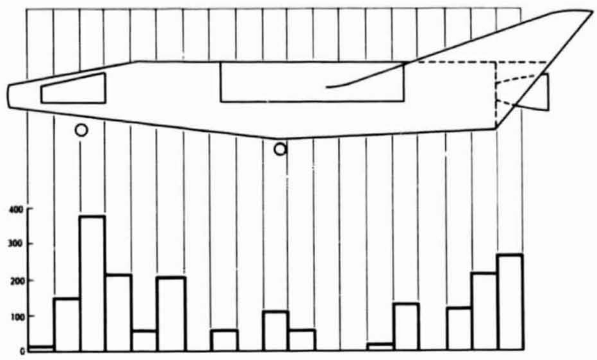
Subsystem Data Traffic

In this depiction of the data traffic to and from all vehicle subsystems, the circled numbers denote the number of data channels. The six horizontal lines correspond to data categories.



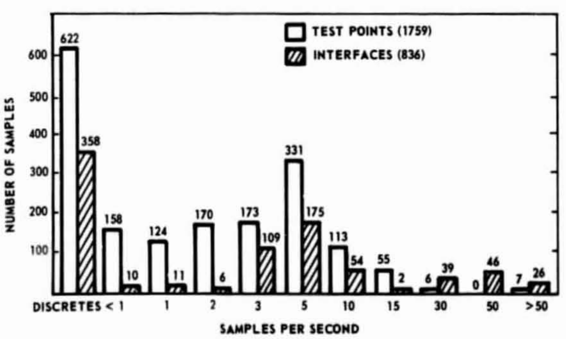
Data Distribution

As indicated, there are two concentrations of data channels: one up front in the crew compartment region and one aft in the vicinity of engines and actuators. Knowledge of this data distribution is important in determining the best grouping of signal acquisition units and the routing of data busses.



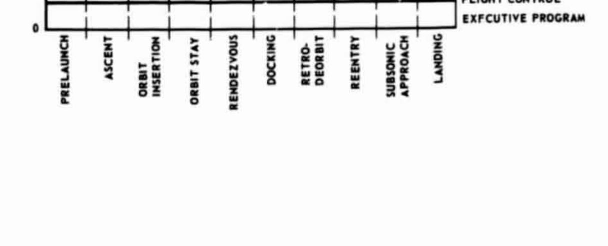
Sampling Spectrum

Indicated here for the complete system are the number of data channels sampled at particular sampling frequencies. Similar plots were made for each of the nine subsystems.



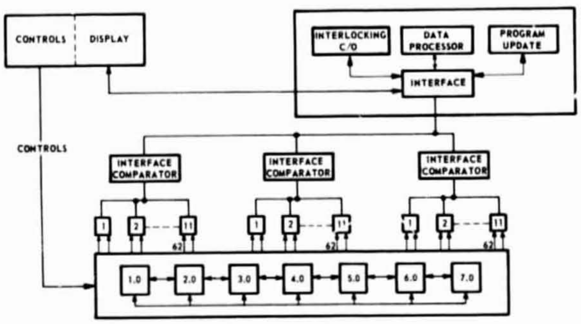
Computer Instruction Rate (10^3 instructions per second)

The computer speed vs mission phase is illustrated. The data storage requirements were estimated to total 274,000 32-bit words, with 64,000 32-bit words required for on-line memory.



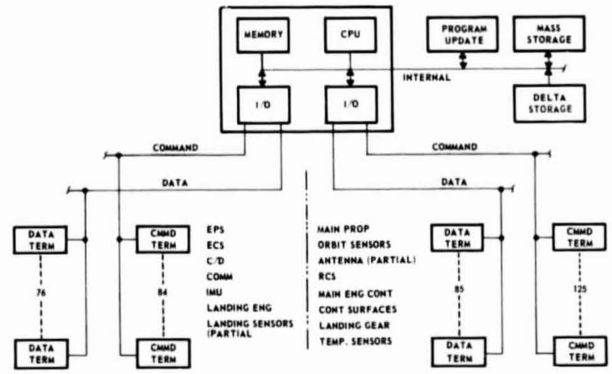
Autonomous Hardware Option

The first alternative IES, which consisted of conventionally interconnected subsystems, served as a baseline IES configuration. Manual command and control inputs to subsystems were routed directly from control display rather than through a data management subsystem. Configuration control and sequencing functions were performed by respective subsystems upon command, either manual or programmed as appropriate. Also, each subsystem was responsible for its own performance and provided diagnostic information to the data management subsystem. The functions of onboard checkout and fault isolational abort warning operations support were accomplished in the first alternate.



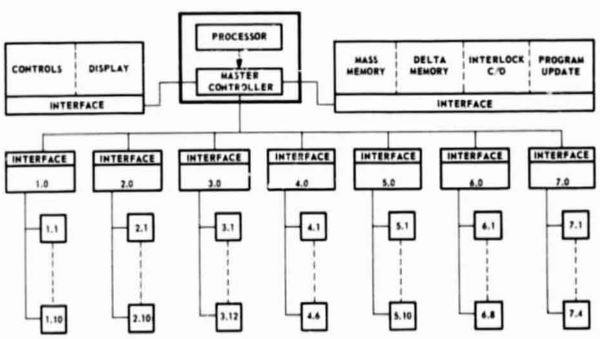
Centralized Multiplexed Option

In the third alternative IES, integration by means of a central computer complex was investigated. All functions performed in Alternative 2, plus computation for subsystem functions, were performed by the centralized system. The only constraint imposed on this alternative IES was the technology of electronic components as projected to the end of 1972. This technology projection was made as part of this study.



Decentralized Multiplexed Option

In the second alternative IES, the functions of Alternative 1 were performed and, in addition, subsystems and major components were interconnected through standardized interfaces and multiplexed data busses, with information and data flow controlled. The control/display data processing was performed by the data management subsystem.



Integrated Electronic System Comparisons

Alternatives 2 and 3, with less cabling, are more reliable than Alternative 1, even though multiplexing circuits are added. Alternative 3 may also involve use of software to achieve graceful degradation in performance through priority control of information retained in storage after a failure of memory.

Electronics component technology does not present a technical risk. Only the software development required for the third alternative is considered to be a technical risk item.

Alternatives 2 and 3 both provide flexibility through incorporation of standard interfaces, multiplexed busses, and the use of software in place of hardware.

	Autonomous Hardware	Decentralized Multiplexed	Centralized Multiplexed
Weight	Baseline	Cable Decrease (-552 lb)	Equipment and Cable Decrease (-630 lb)
Power	Baseline	Slight Increase	Decrease
Reliability	Baseline	Less Cabling	Less Cabling Graceful Degradation
Technical Risk	Current Technology	1969–1970 State of the Art	New System Interface Software
Flexibility	Least	Standard Interface Software	Standard Interface Software

Test and Operations

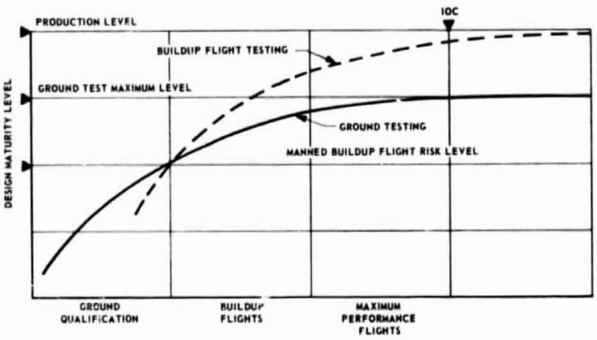
Hybrid Concept of Development Tests

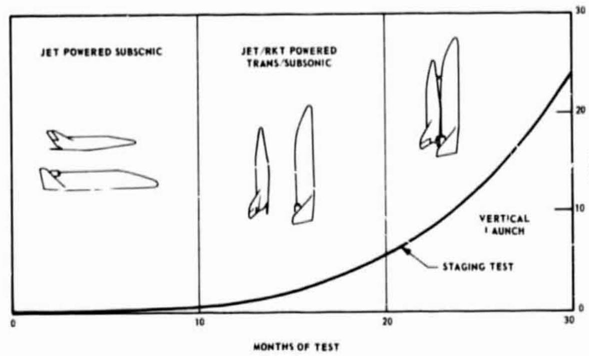
Shown by the solid line is a typical growth curve, which indicates that design maturity increases as development testing is performed. The intersection with the dotted line shows initiation of flight testing. Later flight testing would entail less risk but at the penalty of delaying IOC beyond the 1976-77 period.

The hybrid concept of tests has been proposed to achieve IOC maturity rapidly with low cost and short schedule position. This can be accomplished by combining ground and flight testing programs when sufficient maturity is reached through the ground test program to assume flight risk. One of the major areas requiring further investigation is how to establish the crossover between the flight test and the ground test curves.

Flight Test Concept

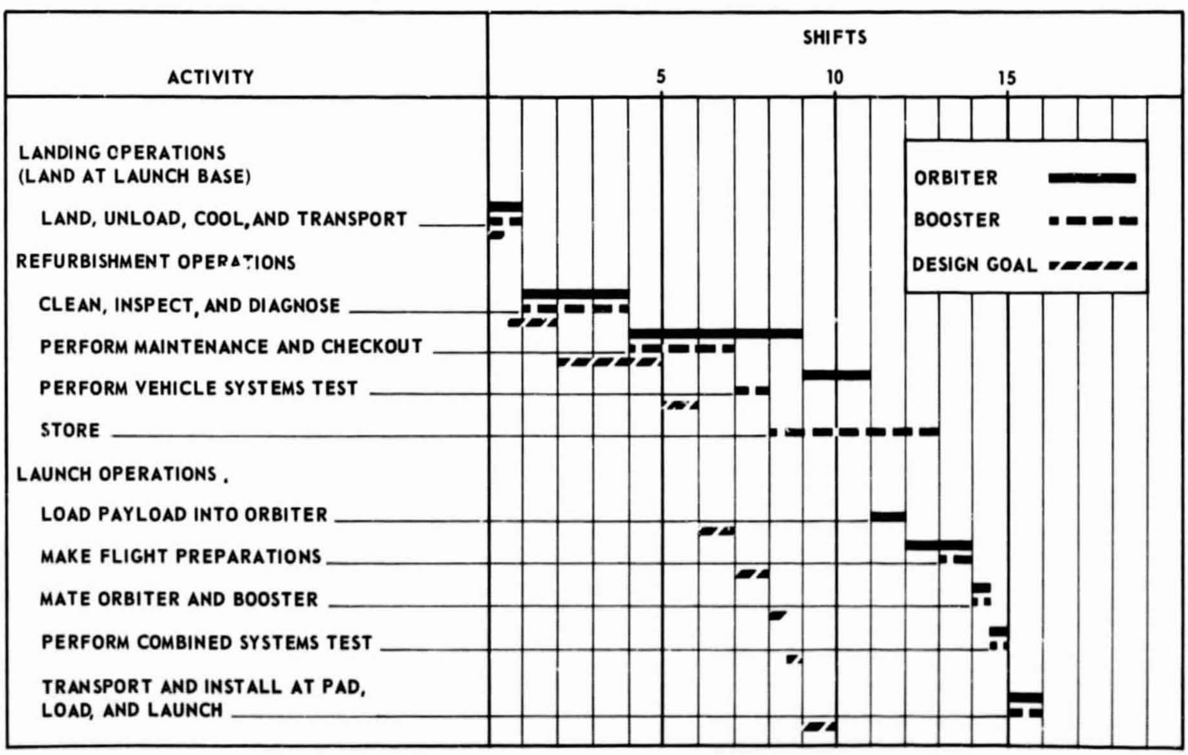
The first period of subsonic testing with jet engines may also be pushed to supersonic speeds. Mach levels as high as 10 to 12 may be reached in the second period with separate launching of each stage for both the orbiter and the booster. In addition to flights involving the basic booster and orbiter, support aircraft will be required. Preliminary studies indicate that 75 to 80 percent of the avionics can be checked out on a variable stability aircraft, such as the Boeing 707-80. It is anticipated that such aircraft will be used to check out critical parts and components for the orbiter and the booster prior to installation.





Turnaround Estimates for Two-Stage Vehicle

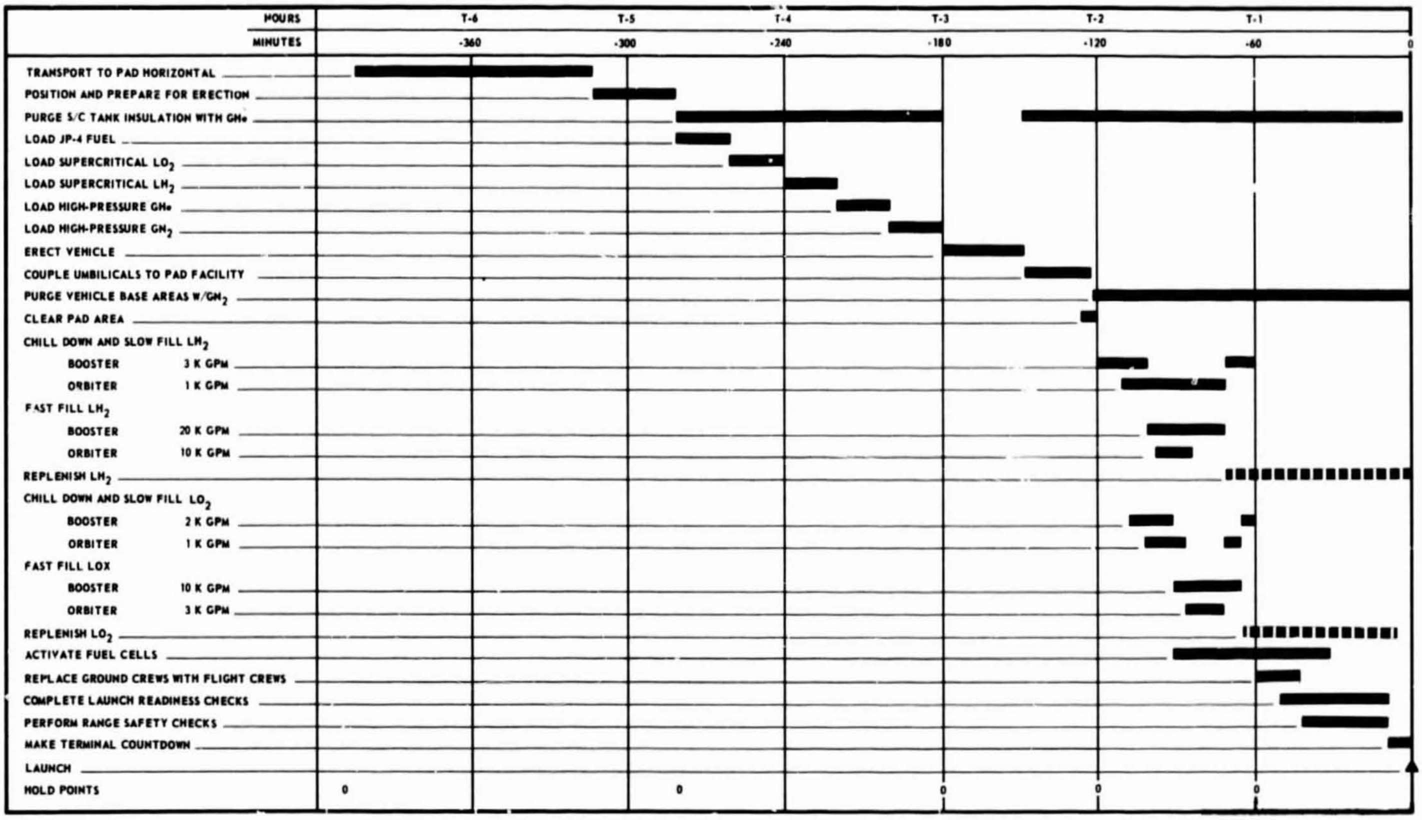
Analysis has shown the possibility of turnaround within 16 shifts for normal turnaround time or 10 shifts on an expedited basis. In this estimate, the orbiter appears as the pacing vehicle, because the booster returns immediately after launch, permitting ground maintenance while the orbiter is in flight.



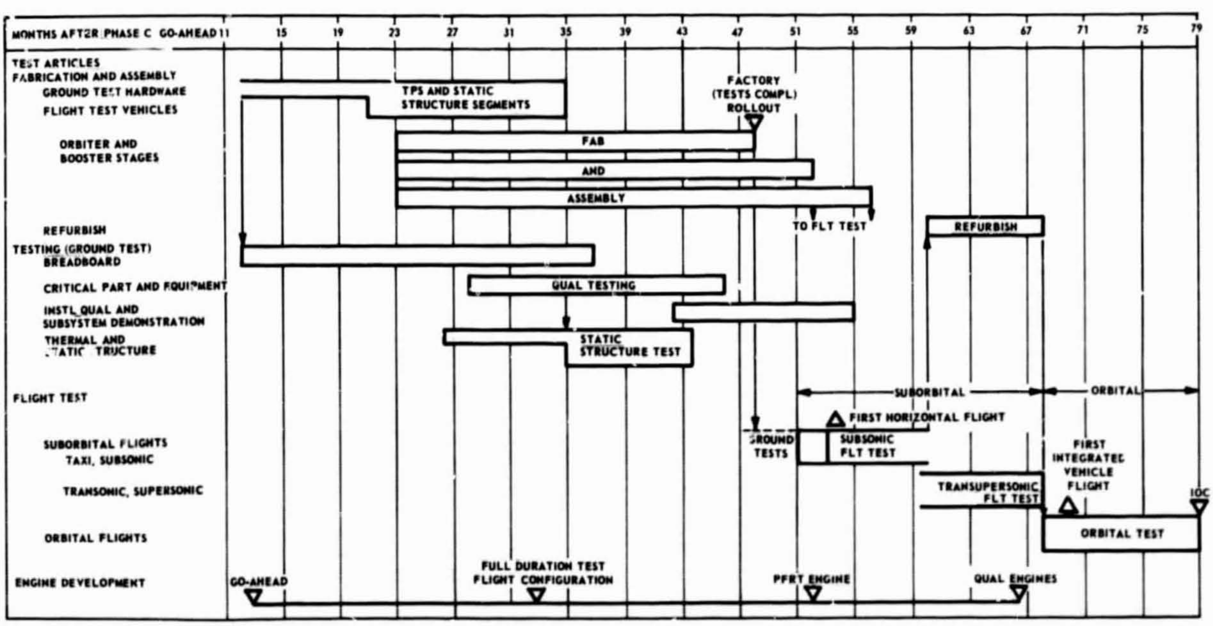
Launch Time Line Two-Stage Space Shuttle

As the vehicle is delivered from the Maintenance and Assembly Building ready for pad erection, vehicle hold is determined by primary battery limitations and weather protection capability. The next hold occurs after the supercryogens have been loaded. Hold limit will be established on the basis of permissible cryogen boiloff, as more visibility into cryogen tank capability is obtained. The next hold point occurs after main propellant loading. Hold time would be established by permissible propellant boiloff and possible cooling effect on the vehicle.

Within this entire sequence, launch scrub could be performed at any time prior to 90 percent thrust buildup and release of holddown.



Schedule



50K Two-Stage Costs

Key Costing Assumptions

- RDT&E

  Equivalent of five orbiters and five boosters required for test
  175 horizontal test flights for each stage
  25 vertical test flights
  Booster/orbiter commonality (%)

Heat shield	20
RCS	80
EPS	50
Avionics	80
ECS	80
Rocket engine	97

  Modify existing airbreather engines

- Recurring

  Orbiter has nominal on-orbit stay time of 7 days
  14-day ground turnaround time for orbiter
  7-day ground turnaround time for booster (double shifts)
  Two orbiters and two boosters from development phase used in operational phase

RDT&E Cost Distribution

ILRV Cost Model (Dollars in millions)

ORBITER			2498.5
Structures		719.0	
Airframe	373.8		
Heat Shield	345.2		
Propulsion		482.3	
Rocket	452.3		
Airbreather	30.0		
Other Subsystems		311.9	
Facilities		175.0	
Test Hardware		310.6	
Systems Support		499.7	
BOOSTER			1990.3
Structures		769.9	
Airframe	509.7		
Heat Shield	260.2		
Propulsion		43.5	
Rocket	13.5		
Airbreather	30.0		
Other Subsystems		94.2	
Facilities		191.4	
Test Hardware		493.2	
Systems Support		398.1	
GSE			363.0
LAUNCH OPERATIONS AND FACILITIES			243.8
FLIGHT OPERATIONS			86.0
PROGRAM MANAGEMENT			330.8
TOTAL RDT&E			5512.4

Recurring Cost for 1000 Flights

ILRV Cost Model (Dollars in millions)

HARDWARE COST		501.7
Orbiter (five vehicles)	305.6	
Booster (two vehicles)	196.1	
OPERATIONS COST		753.5
Launch Operations	438.2	
Flight Operations	315.3	
TOTAL RECURRING		1,255.2

Cost Effectiveness Summary

TOTAL RECURRING COST/FLIGHT	$ 1.255 M
TOTAL RECURRING COST/LB OF PAYLOAD	$25.10
OPERATING COST/FLIGHT	$ .754 M
OPERATING COST/LB OF PAYLOAD	$15.1

Operations Cost Per Flight Breakdown

Operation	Cost ($ x 10^3)
Launch Operations	84.0
Propellants and Gases	312.6
Ground Control and Tracking	16.0
Refurbishment	232.7
Subsystem Support	66.5
Program Management	41.7
Total Operations Cost Per Flight	753.3

Cost Comparison

Comparison of two parametric cost models indicates some significant areas for future study. The ILRV and STS cost models are in substantial agreement on RDT&E cost when allowance is made for STS CERs expressing horizontal flight test cost. For each test flight, the STS model yields cost of $2.5 million, whereas the ILRV model, based on other experience, indicates about $.25 million. This difference alone accounts for $400 million difference in the total RDT&E. The largest difference in the models is operations cost. In the STS model 3/4 percent of first unit cost per flight is assumed for maintenance, which amounts to about $1.8 million per flight, equivalent to 120,000 manhours, whereas the maintenance and refurbishment projection for Space Shuttle indicates a 10-day to 2-week turnaround. If this turnaround time line is realistic, it is hard to envisage such an expenditure, which is roughly equivalent to a 1500-man work force. The comparison indicates that the STS model is closer to the ILRV bottom-up estimate for first unit cost, in which an 83 percent learning curve rather than a 90 percent curve is applied for production hardware. The differences wash out somewhat in comparing total production hardware and total recurring cost. Overall, the ILRV bottom-up total system cost compares very favorably with the ILRV cost model and the STS cost model is considered pessimistic in operations cost.

(Dollars in Millions Except Where Noted)

	PARAMETRIC ILRV Model	PARAMETRIC AF Model	BOTTOM-UP
RDT&E	5,512	6,220	5,468
Production Hardware	502	1,013	688
Operations Cost	754	2,830	937
Total System Cost	6,768	10,063	7,093
Vehicle First Unit	186.4	232.7	258.6
Total Recurring Cost/Flight	1.255	3.84	1.625
Total Recurring Cost/Pound Payload ($/lb)	25.10	76.80	32.50
Operating Cost/Flight	0.754	2.83	0.93
Operating Cost/Pound Payload ($/lb)	15.10	56.60	18.70

Note: Recurring costs based on 100 flights/year for 10 years

VI. STUDY LIMITATIONS

Even though subject to the normal Phase A limitations of time and changing conditions, the study has succeeded in strongly re-affirming that a Space Shuttle system is feasible and can exceed the goal of an order-of-magnitude reduction in space transportation costs. Following are brief discussions of limitations in areas that warrant further study.

- The requirement that main propellants for both stages of the Two-Stage vehicle be LO_2/LH_2 – Limited investigation of alternate propellants for the Stage-and-One-Half and the Triamese pointed strongly to LO_2/LH_2 as the optimal choice. The same result would clearly apply to the orbiter of a Two-Stage vehicle. However, since study of alternate booster propellants was not carried out, it has not been affirmed that LO_2/LH_2 propulsion provides the most cost effective system for the booster.
- Emphasis in the study on delta-body orbiters and delta-wing boosters – These configurations certainly possess desirable characteristics for the Space Shuttle, but delta-wing and straight-wing orbiters and straight-wing boosters are also alternatives worthy of further consideration.
- The assumption that all missions considered for Space Shuttle would in fact be flown on the Space Shuttle – While the great reduction in space transportation costs afforded by the Space Shuttle system would seem to justify this assumption, an examination of all launch system alternatives for each mission may be needed to verify the assumed traffic levels for the Space Shuttle system, a parameter that is very important in demonstration of low recurring costs.
- The assumption that an optimal approach is based on minimum cost or minimum launch weight for a best case system design to meet specified operational requirements – Significant factors ignored in this assumption are development risk and mission flexibility. For instance, the high sensitivity of booster sizing to orbiter dry weight suggests a desensitized booster design (e.g., stretchable booster), even though it would involve some penalty to launch weight, in order to alleviate development risk. Weight penalties could be allowed in the orbiter to increase mission flexibility and therefore help maximize system utilization; this was not permitted in the Phase A study but should be given serious consideration in the subsequent phases of development.

VII. IMPLICATIONS FOR RESEARCH

Propulsion System

Development of thrusters, propellant feed system, and propellant orientation system for the oxygen/hydrogen reaction control system require technological advancements, as does the development of the auxiliary power unit (whose primary function is to supply hydraulic power for control surface actuation during reentry).

Leakage detection, which represents a major problem, also entails advancement.

Aerodynamics

Although the Space Shuttle aerodynamics effort will be considerable, the basic aerodynamic technology exists. The areas expected to represent major development effort are subsonic flight, hypersonic viscous effects on vehicle performance and control, plume-induced phenomena, composite launch vehicle interference effects, and staging and abort.

Aerothermodynamics

Listed below are key technology issues related to prediction of aerodynamic heating distributions and resulting surface temperatures:

- Prediction of flow field and laminar and turbulent heat transfer distributions (Wind tunnel tests should be conducted to resolve the discrepancies among various flow field and heating prediction techniques.)
- Prediction of boundary layer transition (Experimental boundary layer transition studies should be conducted for each candidate configuration.)
- Evaluation of heat transfer increase due to surface irregularities
- Evaluation of location and heat transfer increases associated with shock wave/boundary layer interactions
- Prediction of base heating rates (Scale model tests of the booster/orbiter configuration employing nozzle hot flow could provide necessary convective and radiative heat transfer design data.)

Structures

Primary technology areas requiring additional investigation or basic research are reusable thermal protection systems (which can maintain their orbital thermal properties) and their interfaces and efficient joining techniques, along with adequate analytical methods. Specific areas include materials; thermal protection systems; joining methods; movable elements in a high-temperature environment, such as control surfaces; structural mechanics analysis; cryogenic technology (particularly reusable insulation development like "breathing" multilayer systems); and orbital vehicle surface thermal properties.

Avionics

Avionics requirements for the Space Shuttle do not indicate the need for a breakthrough in technology; however, improvements in technology are required to achieve the objective of a low-cost transportation vehicle. Integration of the avionics functions is the primary technique to be used in significantly reducing system weight, power, and cost. The two most challenging areas are in the development of survivable antennas and a link for communications via a relay satellite. Both may require advancement in the state-of-the-art. The technology plan will require an extensive simulation test program, both ground and airborne, to validate design integrity of the integrated system. These tests should occupy the span from mid-1970 through 1971. It would be desirable to intorduce critical Space Shuttle communications equipment on the AAP/ATS-F relay link experiment to provide experience and confidence in the design early in Phase D of the shuttle development program.

Bioastronautics

Environmental control/life support system technology areas identified as being desirable for expansion or refinement to meet the needs of the Space Shuttle are radiator design, noise control, chemical oxygen systems, waste management and personal hygiene, reusable ECS, multimode ECS, suit-loop elimination, and equipment temperature control system design.

Crew system areas requiring further work include determination of vehicle flying qualities by simulation techniques, assessment of crew visibility requirements for various mission phases along with empirical evaluations of alternative visibility provisions, development and evaluation of candidate crew safety and escape features for crew and passengers, assessment of crew workload distribution and validation of the ability of the specified two-man crew to accomplish all assigned mission activities, and human factors effectiveness evaluations of novel integrated display concepts to allow the crew to exercise a manual override and takeover function.

Technology Funding

The advanced development funding, required to support technology programs prior to initiation of Phase D system development, is estimated to total less than $200 million.

VIII. SUGGESTED ADDITIONAL EFFORT

Now that Phase A results have demonstrated the economic advantages of Space Shuttle, no time should be lost in proceeding to Phase B. Significant delay in the IOC would unnecessarily prolong the cost penalties incurred by continued reliance on throw-away boosters and the absence of a capability to recover and refurbish orbiting systems. Areas requiring particular emphasis in the Phase B study include mission compatibility, configuration definition, and development planning.

Mission Compatibility

For maximum cost effectiveness, the Space Shuttle must be designed to accommodate a large percentage of the anticipated missions. Initial assessments of the interface requirements with projected mission elements indicate that the Space Shuttle payload capability should be at least 50,000 pounds and that the cargo bay should be 60 feet long with a 22-foot diameter extending over 45 to 50 feet of its length. Refining these requirements necessitates definition of the Space Tug, LM-B, Nuclear Shuttle, Space Station, and high-velocity stages used for deep-space probes. Space Shuttle contractors should be intimately involved in this work so that they can arrive at the most suitable payload/shuttle interfaces. This area of additional effort bears directly on verification of the expected traffic rates and high utilization demanded of an effective Space Shuttle.

Configuration Definition

Alternate configurations to the delta lifting body emphasized in the Phase A study are winged cylindrical bodies with either straight wings or delta wings. In-depth analysis of each alternative must be accomplished in Phase B in accordance with a common set of system requirements and design criteria. The primary advantages of the winged cylindrical body approach are simplification of propellant tankage and plumbing and improvement of the subsonic lift-to-drag ratio. However, the greater length required for storing sufficient propellants would result in overall weight penalty. In addition to further wind-tunnel testing, analyses to determine whether the benefits justify the weight penalties should be supported by both computer and physical simulation of the propellant handling systems and of approach and landing techniques with various lift-to-drag characteristics. Similar analyses and tests are required to arrive at a choice among the booster configuration alternatives, which include straight-wing and possibly variable-geometry wing alternatives as well as the delta wing concept emphasized in the study.

Other significant configuration considerations warranting additional study include:

- Resolution of orbiter landing modes and operational requirements for powered go-around engines
- Allowance of overdesign or stretchable design penalties to desensitize the system against potential performance loss
- Allowance of appropriate weight penalties in the orbiter to provide high mission flexibility, e.g., possible use of a four-man cabin, providing convenient space for two mission-peculiar crew members, and convenient provisions for storage of extra expendables required on some missions
- Provision of a cargo bay larger than the current 15 x 60 ft and 22 ft x 30 ft size
- Redundancy provisions in all subsystems needed to provide adequate safety and mission reliability
- Configuration design effects of providing optimal abort capabilities
- Possible use of hydrogen as the fuel for booster fly-back engines and orbiter go-around engines rather than the JP-4 assumed in the ILRV baseline designs (Appreciable orbiter cruise range after an abort as well as system weight advantages could result from this approach.)

Development Planning

Primary considerations in development planning involve studies to establish an optimal IOC date and programming to alleviate development risk. The IOC date for Space Shuttle should be determined on the basis of the proper balance between the potential cost impact of assuming a degree of development risk and the predictable cost penalty of delaying the IOC (i.e., the cost of continuing for a longer period to launch expendable space systems on expendable boosters). Alleviation of development risk should be emphasized in planning as well as design. Early concentrated effort on the development of heat shield materials could produce results that significantly reduce the current estimates of development risk in this area. One possibility, for instance, is that further development and testing will confirm the present indications that LI-1500 can withstand 2500°F with a margin of up to 500°F. If, in addition, feasible methods of manufacture and attachment of this material were demonstrated, such results would significantly reduce the current concern that the predicted reentry temperature of 2200°F has an uncertainty of 200° or 300°F.