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Space Shuttle Program

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Phase B Extension
TECHNICAL REPORT

Volume I

Contract NAS9-10960, Exhibit F
SD 71-342
12 November 1971

Space Division • North American Rockwell 12214 Lakewood Boulevard, Downey, California 90241

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SPACE SHUTTLE
PHASE B EXTENSION
TECHNICAL REPORT

Volume I

Approved by

B. Hello
Vice President and Program Manager
Space Shuttle Program

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Exhibit F
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FOREWORD

This report is submitted to NASA in accordance with contract NAS9-10960. The report documents the results of studies to define Space Shuttle programs that satisfy specific funding constraints and minimize technical risk. The studies were performed under the direction of the Space Division of North American Rockwell, Downey, California. Other members of the study team were Convair Aerospace Division of General Dynamics and Aerospace Division/Honeywell, Inc.

The report is provided in two volumes. Volume I presents the results of the effort accomplished during the months of July and August, 1971, when the following studies were made: external hydrogen tanks versus external hydrogen/oxygen tanks; variations on payload bay size; single engine orbiter impact; evaluation of various interim boosters and phased development programs; and low technology orbiter designs.

Volume II reports the results of the effort for September and October, 1971, during which the space shuttle systems were defined using a low technology orbiter combined with either an F-1 flyback booster or a pressure-fed booster.

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1.0 INTRODUCTION AND STUDY APPROACH

In order to define a system which would significantly reduce payload delivery costs compared to current launch systems, NASA contracted and directed a number of Phase B design studies of a fully reusable space system. These studies indicated that, while the fully reusable vehicle reduced operational costs, the annual expenditures were unacceptable. Significant technology advancement was also necessary with attendant program risk Contracted activities were therefore extended to investigate approaches that would reduce the risk and the peak annual expenditure for the space shuttle.

This report summarizes phase B study extension activities performed by the NR and GDC team during the period 1 July to 31 October, 1971, to investigate modifications to the reusable space shuttle concept. The investigation of various space shuttle system options were performed in two phases, each two months in duration. Phase 1 encompassed investigation of systems using orbiters with external propellant tanks and an interim expendable booster, which allowed phased development of the reusable orbiter and booster. Phase 2 studies were directed toward the definition of a program which would maximize the use of existing technology for the orbiter and which would also use existing hardware for the booster, as well as the simplification of designs for cost reduction.

During Phase 1, investigations addressed the following issues:

1. Merit of internal and external propellant tanks and impact of external LH_2 compared to LO_2 and LH_2

2. Impact of cargo bay size

3. Impact of abort

4. Merit of expendable booster options

5. Merit of a phased development program

The approach adopted in the study was to define the ultimate operational system with reusable boosters and orbiters with external LH_2 or LO_2/LH_2 tanks and to identify the characteristics of interim expendable boosters to achieve the desired interim performance capability. Studies showed that external LO_2/LH_2 on the orbiter minimized program cost and risk, and the use of a phased development with expendable solid rocket booster reduced peak annual expenditure.

Investigations also identified potential merit in a low technology advancement program. During Phase 2 of the study, activities were therefore directed toward the analysis of an orbiter with subsystems requiring minimal technology advancement and the use of a booster with existing hardware technology or simple pressure-fed system. The baseline program is illustrated in Figure 1-1. An orbiter was configured using a NASA/MSC concept as the baseline. The external LO_2/LH_2 orbiter tank size and flyable LO_2/RP and pressure-fed booster were then sized to satisfy Mark II system requirements. The capabilities of Mark I systems, using J-2 or J-2S engines on the orbiter, were also established.

Studies showed that the continued use of the J-2S engine on the orbiter reduced program cost. Preliminary analyses indicated that use of a pressure-fed booster also results in lower cost than a flyable LO_2/RP stage with F-1 engines, but confidence in this system needs to be increased and additional studies are therefore recommended.

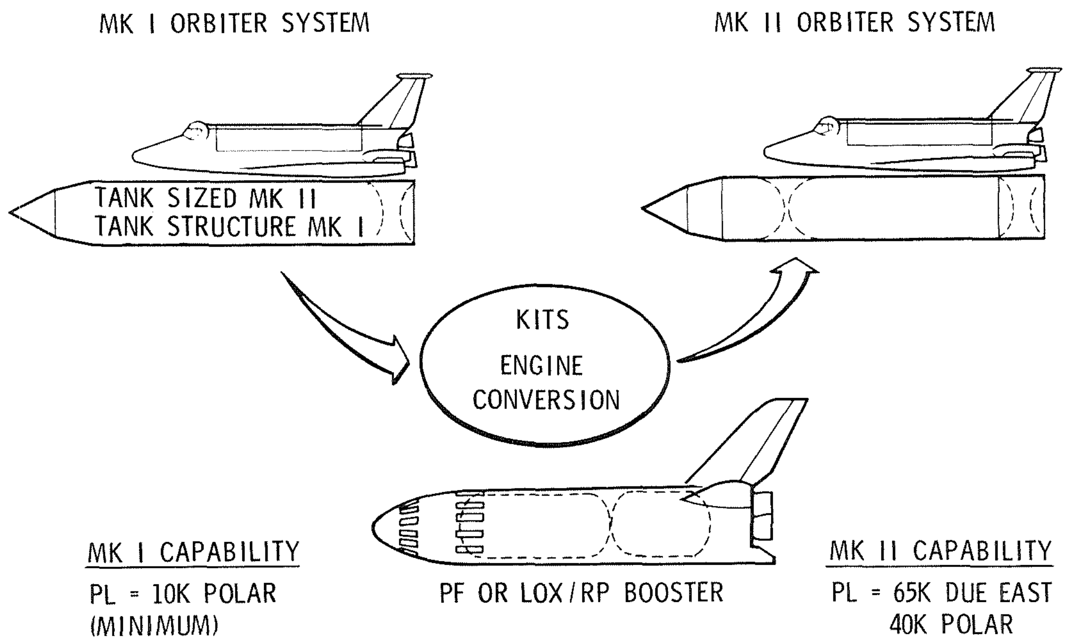


Figure 1-1. Baseline Program

2.0 SUMMARY

This document is a fourth month summary of the extension to the Shuttle Phase B Program Definition study accomplished by the North American Rockwell Corporation (NR) team under contract to NASA/MSC. The primary objective of the extension study was to identify a shuttle system and program which resulted in acceptable program costs and reduced expenditure rates. The goals of the extension study were: (1) select orbiter/main engine development approach; (2) select external tank (LH_2 vs LO_2/LH_2); (3) select interim and final booster; and (4) define the recommended program.

The approach to the activity is shown in Figure 2-1. The first two months of the activity considered phased development of the orbiter and booster with an expendable booster to be used for up to five years. External hydrogen and external hydrogen/oxygen tanks were considered for the orbiter, together with low technology risk, low-cost subsystems. The criteria to evaluate the alternatives were: (1) minimum peak annual funding; (2) mission capability; and (3) horizontal and vertical flight dates.

The NR recommendation on September 1, 1971, which is documented in SV71-40, Executive Summary Report, is illustrated in Figure 2-2. The orbiter used external LH_2/LO_2 in a single belly tank because this resulted in lower peak annual funding, lower risk, least orbiter weight sensitivity, and minimized fracture mechanics problems. Two separate orbiter designs were recommended which involved a 15- by 40-foot cargo bay for a Mark I,

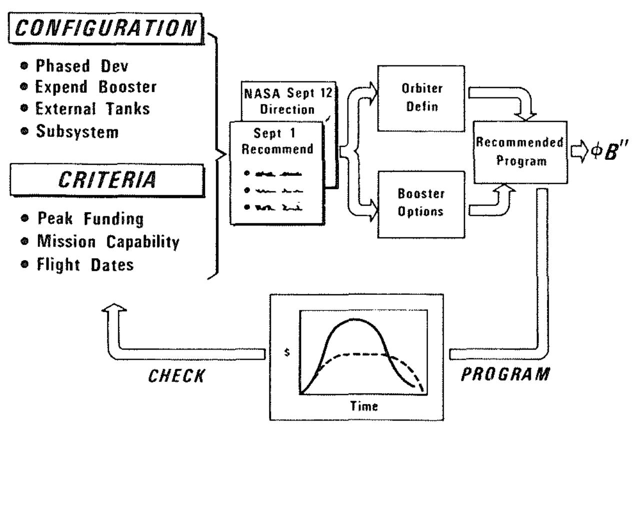


Figure 2-1. Phase B Activity

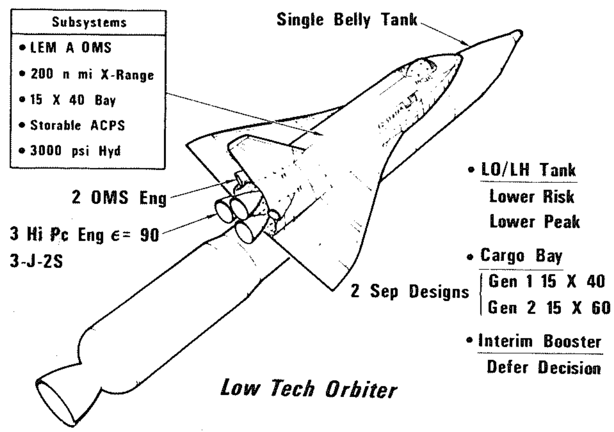


Figure 2-2. September 1 Configuration Recommendations

and 15- by 60-foot cargo bay for a Mark II orbiter. The orbiter used low technology risk subsystems such as an hypergolic orbit maneuvering system with two LEM ascent (LEMA) engines, hypergolic attitude control propulsion system, 3000 psi hydraulic systems, and three J-2S main engines. A 200-nautical-mile cross range thermal protection system was employed although the vehicle was designed aerodynamically to achieve 1100 nautical-mile cross range. The interim booster configuration decision could be deferred until six months into Phase C; the Mark II booster was the fully reusable heat sink (RHS) booster defined as part of the Phase B study.

The funding/phasing distribution is shown in Figure 2-3. The J-2S orbiter/expendable booster option uses this configuration for the entire program. The Hi Pc Booster is a program which uses high chamber pressure engines on the orbiter, and an expendable booster phased into an RHS booster.

NASA direction of September 12, 1971 (Figure 2-4) postulated a minimum technology risk orbiter with J-2 or J-2S engines for Mark I, and Hi Pc engine for Mark II with an external LH_2/LO_2 single belly tank. Two boosters were to be studied: (1) a flyable LO_2/RP recoverable booster using F-1 engines; and (2) a recoverable pressure-fed booster. The payload requirement for Mark I was to be 10,000 pounds minimum for a polar orbit; for Mark II it was to be 65,000 pounds for a due east orbit.

This volume reviews the technical definition of the boosters, examines the orbiter external tanks and the orbiter subsystems, and presents the configuration and program options.

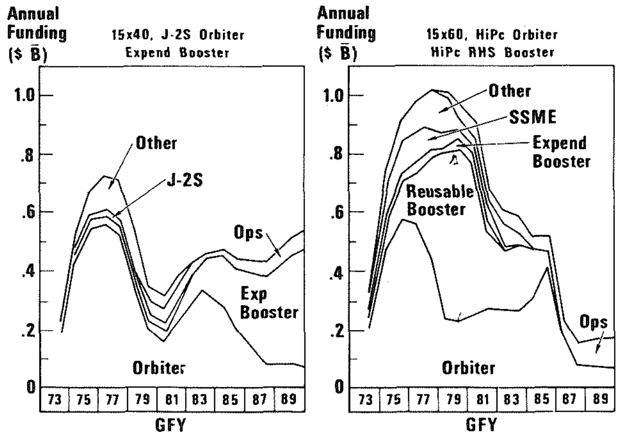


Figure 2-3. Funds Distribution/ Phasing (September 1)

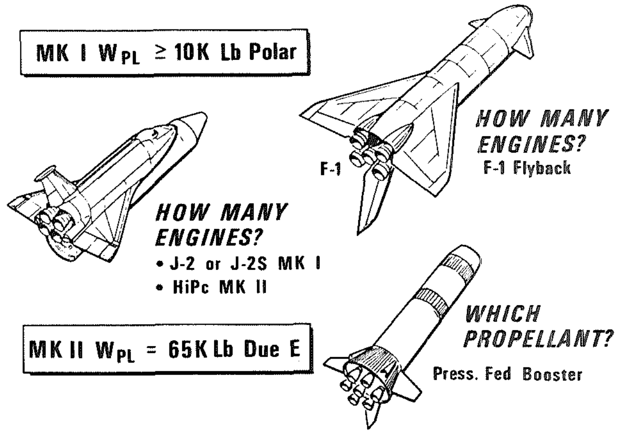


Figure 2-4. September 12 Direction

2.1 PRESSURE-FED BOOSTER

In the first two months of the Phase B extension, emphasis was placed on interim boosters including solids, Titan III L, S-IC, pressure-fed expendable, and the reusable heat sink booster defined in Phase B. As the result of the September 12, 1971 direction one booster to receive emphasis in the second two months of the Phase B extension was the pressure-fed booster (PFB) for which several propellant combinations were possible: (1) various storables; (2) LO_2/RP, and (3) LO_2/propane (C_3H_8). Studies showed that the storables result in the highest total program costs and are highly toxic. The LO_2/RP combination resulted in the lowest hardware weight; the LO_2/propane had engine and operational advantages; and both LO_2/RP and LO_2/propane had comparable total program costs for staging velocities of greater than 6000 fps. The LO_2/propane combination was selected as a baseline.

A tradeoff study was performed to determine the number of engines. Based on roll control, pakaging efficiency and delta V loss, the number was determined to be seven. The baseline PFB (Figure 2-5) has an overall length of 163.7 feet and uses seven 975,000-pound thrust engines. The LOX tank uses inconel 718 and operates at 315 psia pressurized by helium; the C_3H_8 (propane) tank uses inconel and operates at 300 psia and is pressurized by hydrazine (N_2H_4). The PFB uses three fins for control during launch.

The pressure-fed engine (Figure 2-6) is nongimbaled and uses liquid injection TVC to provide up to 5-degree effective angle of control. It operates at a mixture ratio of 2.8 and has a vacuum I_{sp} of 278 seconds. The engine has an overall length of 208 inches and an exit diameter (area ratio is 5) of 135 inches.

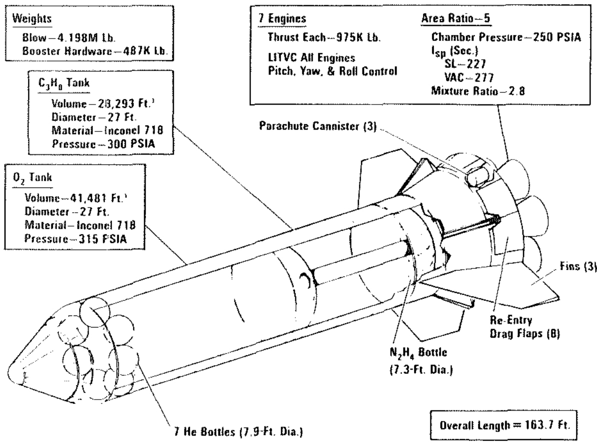


Figure 2-5. PFB Configuration

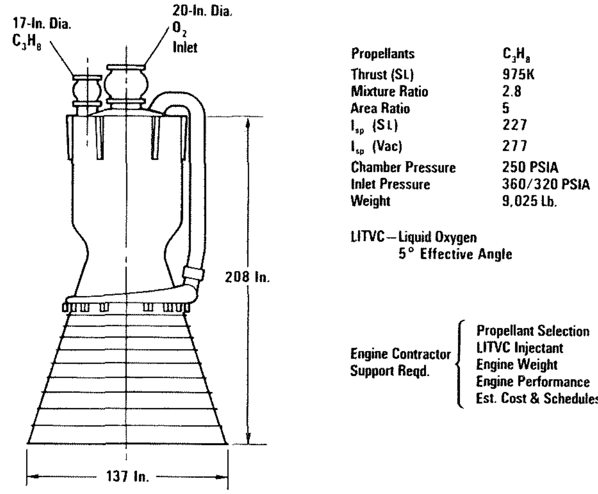


Figure 2-6. Pressure-Fed Engine

The engines are not throttled, but engine shutdown is used to restrict acceleration levels and dynamic pressure. At 30 seconds after launch, one engine is shut down; at 106 seconds, two engines are shut down; at 148 seconds two more engines are shutdown; and staging occurs at 150 seconds.

The recovery sequence for the PFB (Figure 2-7) uses drag flaps after separation, deploys drogue chutes at 32,710-foot altitude, and deploys three 104-foot main chutes (50 percent reefed) at 24,321 feet with the main chutes unreefed at 23,080 feet. The vertical impact speed is 150 fps.

The PFB booster is retrieved (Figure 2-8) by a modified landing ship dock (LSD) which floods the dock so the booster can be winched in. The LSD returns to Port Canaveral and the booster is then moved by barge to the VAB. The total turnaround time (Figure 2-9) is 28 calendar days or 46 shifts.

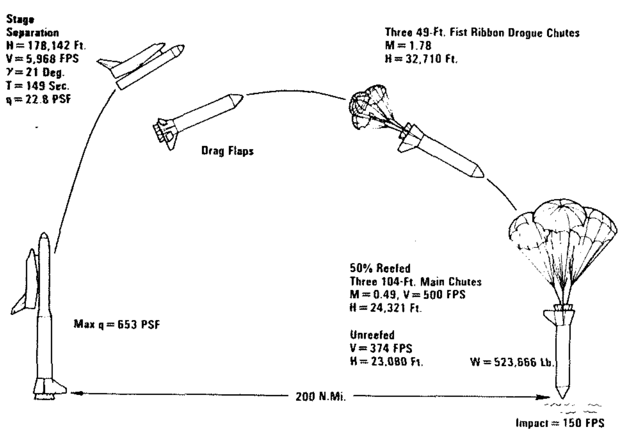


Figure 2-7. Recovery Concept

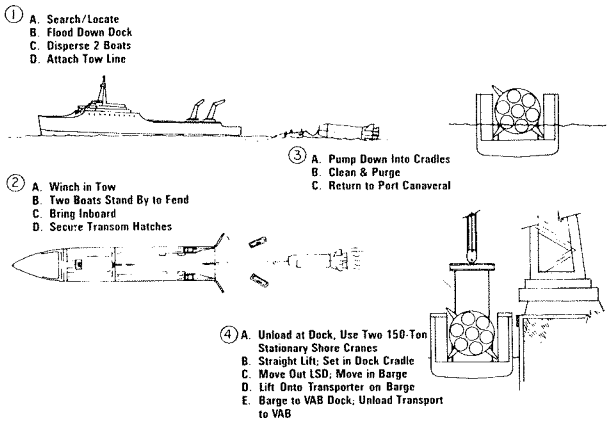


Figure 2-8. Retrieval Concept

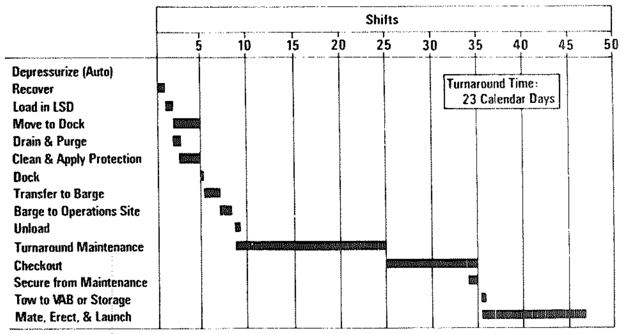


Figure 2-9. Pressure-Fed Booster Turnaround Time

2.2 F-1 FLYBACK BOOSTER

As a result of the 12 September 1971 direction, the F-1 flyback booster was added to the study list. This booster uses F-1 engines and, as far as possible, Saturn S-IC tank geometry. A number of configurations were examined (Figure 2-10) using S-IC tanks, S-IC tank geometry, new LO_2 tank, and new LO_2, RP tanks. The booster with S-IC tank geometry was initially selected and refined for balance and reduction in weight.

The preferred booster (Figure 2-11) configuration uses five F-1 engines and has an overall length of 188 feet and a wing span of 144.3 feet. It also employs canards. The comparison of the F-1 flyback booster and the S-IC is shown in Figure 12). The spacing between the LO_2 and RP tank has been increased from 2.5 feet to 10 feet to allow room for the installation of some of the airbreathing engines. Commonality with the S-IC is achieved in

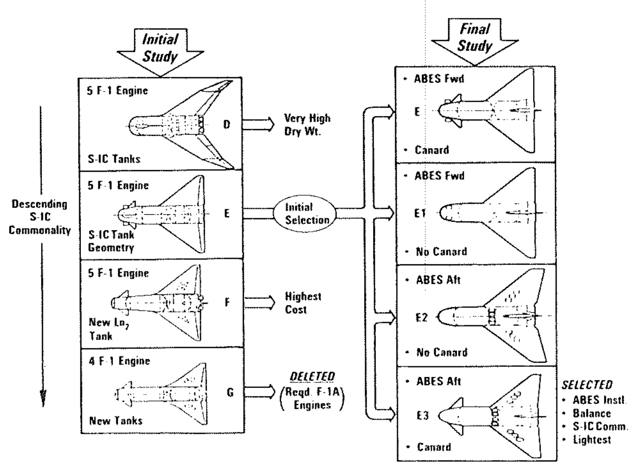


Figure 2-10. F-1 Flyback Booster Evolution

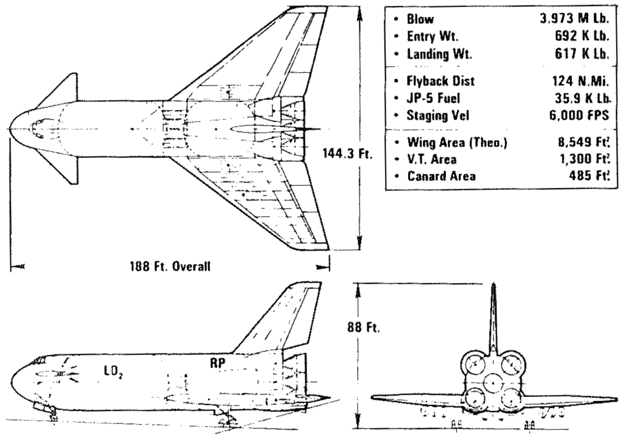


Figure 2-11. F-1 Flyback Booster Configuration (B-18-E3)

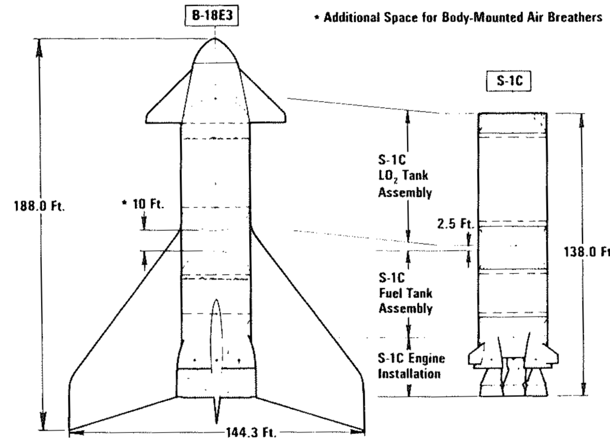


Figure 2-12. B-18-E3/Saturn Comparison

the main engines, most of the propellant system, and geometrically with the LO_2 and RP tank (Figure 2-13). The use of the airbreathers from the B-1 program and a number of common components with the orbiter achieves commonality with other aerospace vehicles.

The F-1 flyback booster uses 10 General Electric F101/F12B3 low bypass ratio turbofan jet engines for the flyback distance of 124 nautical miles. The attitude control propulsion system (ACPS) uses hydrazine monopropellant with 28 2200-pound thrust engines; the hydraulic power/electrical power generation system has four 778-horsepower APU's which also use hydrazine monopropellant. Both the APU's and ACPS's have a common hydrazine tank (one forward and one aft). The cockpit and the avionics systems achieve the same commonality with the orbiter as in Phase B.

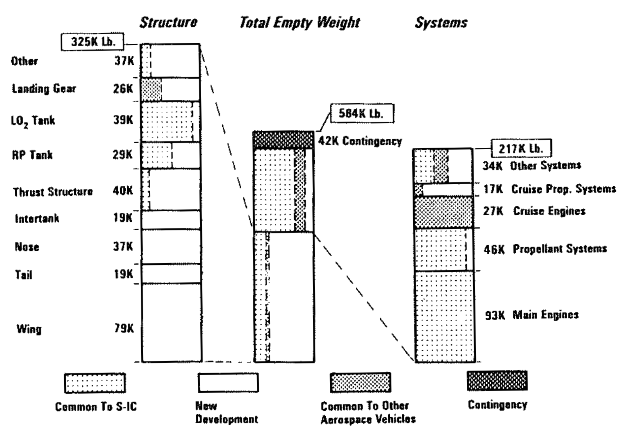


Figure 2-13. B-18-E3/S-1C Commonality

The structural concept uses aluminum for all the primary structure except as noted in Figure 2-14. Titanium is used in the leading edges of the wing and vertical structure and in part of the main engine installation compartment; Rene' 41 is used on the leading edge of the canards.

The ascent performance and control analysis (Figure 2-15) showed that a 90 percent thrust (10 percent throttle) is used for the first 40 seconds of the launch trajectory with one engine shutdown at 40 seconds; 100 percent thrust with four engines from 40 seconds to 116 seconds; 80 percent thrust until 138 seconds at which time two engines are shutdown; and the two remaining engines are at 100% thrust until staging at 140.6 seconds.

2.3 ORBITER EXTERNAL TANKS

Since the design and cost of the external tanks are key issues, detailed studies were performed including a number of design trades and a detailed parametric estimate of the tooling, facilities, and manufacturing process. Figure 2-16 compares the overall characteristics of the external tank designs with the fully reusable orbiter system that resulted from the original Phase B activity. The LH_2 orbiter is 13 feet longer, has a dry weight 43,000 pounds heavier, and a surface area 3738 square feet greater than the respective characteristics of the LO_2/LH_2 tank orbiter.

The selected concept has a single expendable LO_2/LH_2 tank mounted on the underside of the orbiter. This system, which uses expendable LO_2/LH_2 tanks, reduces sensitivity of the orbiter to weight growth, minimizes the technical risk associated with the fracture mechanics of the main propellant tanks, and results in a smaller orbiter which has a lower weight than previous configurations. This external tank design also provides a major advantage in reducing overall program costs with lower peak annual funding

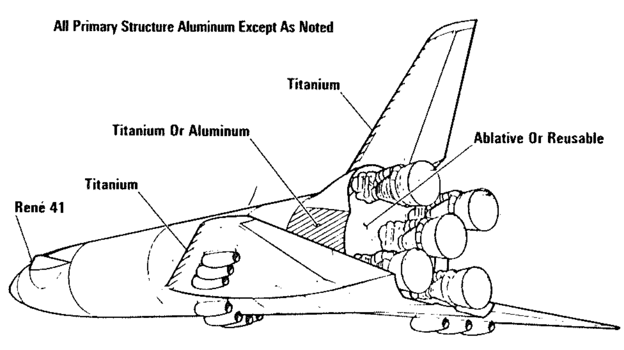


Figure 2-14. B-18-E3 Structural Concept

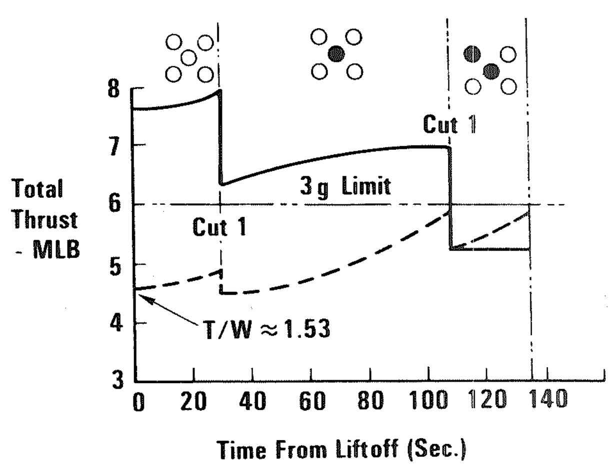
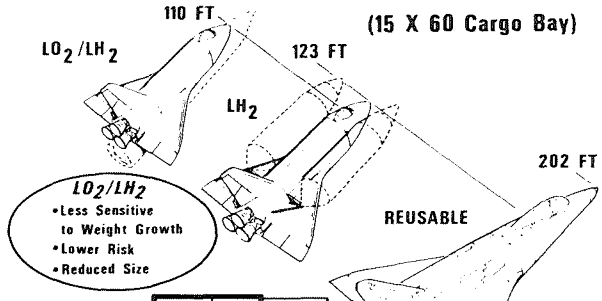


Figure 2-15. Engine Thrust



	LO_2/LH_2	LH_2	Reusable
Body Surface, ft^2	6,297	10,035	15,716
Dry Wt (k lb)*	148	191.2	293
3 Hi-Pc-Fvac Eng (k lb)	420	447.5	551

Figure 2-16. LO_2/LH_2 Orbiter Versus LH_2

than the other options considered. This design locates the oxygen tank forward and is connected with nonpressurized interstage structure to the liquid hydrogen tank aft. The external tank structure is used to interface with the boost vehicle. The external LO_2/LH_2 tank arrangement yields the lowest GLOW and its total program cost is slightly lower than the configuration with external hydrogen.

Several options for alternate main propellant tank arrangements and construction techniques were examined during the design studies. Figure 2-17 summarizes some LO_2/LH_2 tank arrangements which were studied in some depth. The selected concept employs simple monocoque construction. Despite its higher weight relative to skin-stringer-frame the monocoque approach was selected because of reduced production expenditure. Detailed studies relative to this concept, including design trades, manufacturing, tooling, and facilities/transportation, were then used to develop the production tank cost estimate.

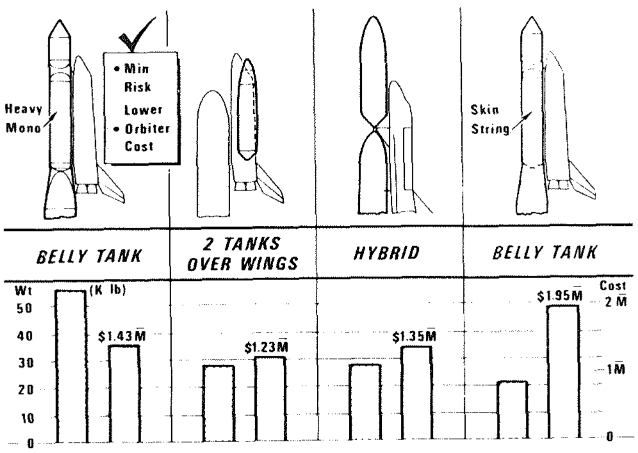


Figure 2-17. Some LO_2/LH_2 Tank Arrangements Studied

A key element in the production cost estimate of the external LO_2/LH_2 tank is the application of learning factors. In determining the cost, the appropriate learning curves for the assembly process involving metallics or nonmetallics, for the parts which are machined and chem-milled, for sheet metal, and for spray-on foam used are shown in Figure 2-18. In the cost verification a fabrication hour-per-pound structural weight comparison of the selected concept relative to other tank arrangements and hardware currently in use was made. The hour per pound for the first unit for the external LH_2 tank is 15 hours per pound, DC-10 is 10 hours per pound, S-II is 27 hours per pound, skin-stringer external LH_2/LO_2 tank is 11 hours per pound, the comparable number for the selected heavy monocoque external LH_2/LO_2 tank is 6 hours per pound.

Based on this detailed study, the September 1st estimate for the average unit cost (Figure 19) of the LO_2/LH_2 single belly tank was confirmed. The estimate of average unit cost at September 1 was $1,430,000; the current estimate is $1,400,000 per tank.

2.4 ORBITER SUBSYSTEMS DESIGN APPROACH

The baseline configuration (Figure 2-20) has been studied relative to body and wing shape, attitude control propulsion system (ACPS) location, manipulator location, canopy and cockpit requirements, and airlock design. The recommendations relative to changes to this configuration baseline are manipulator arms in cargo bay fairing, ACPS pods moved to wing tips to avoid elevon impingement, airlock external to cabin, and free-fall landing gear. The orbiter is designed with a 15- by 60-foot cargo bay (Mark I and HiPc for Mark II); two orbit maneuvering system (OMS) pods (LEMA engine for Mark I, and a new regenerative engine for Mark II); all aluminum structure, thermal protection (ablator for Mark I, and RSI for Mark II), and a canopy designed to provide forward and aft visibility.

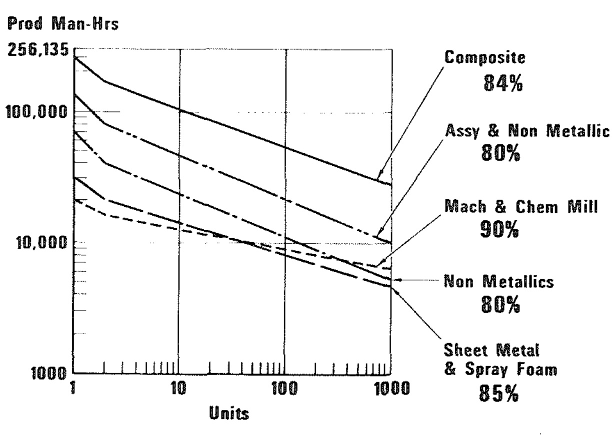


Figure 2-18. Learning Curves Selected for Each Element

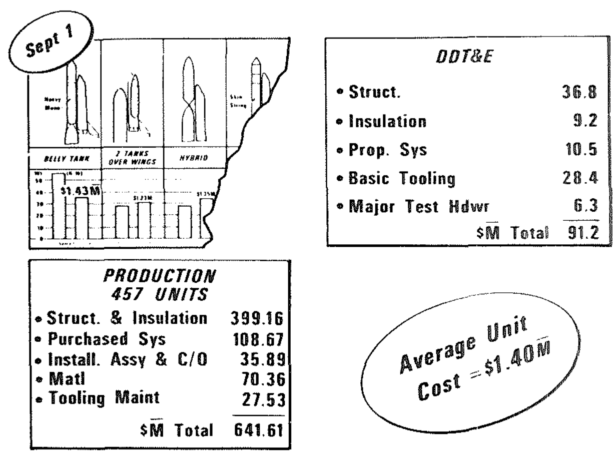


Figure 2-19. Detailed Studies Confirm September 1 Estimate

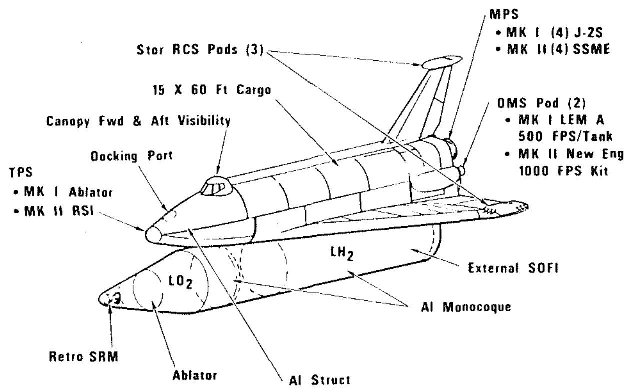


Figure 2-20. Phase B'' Baseline Orbiter

The structural design/development approach reflects basic similarities between Mark I and Mark II orbiter which requires design and qualification of only one basic airframe design. Geometric differences between Mark I and Mark II will be accomplished by the use of kits (i. e., Mark I versus Mark II base heat shield).

As indicated by Figure 2-21, to meet the requirement to fly an increased cross range (200 nautical miles for Mark I and 1100 nautical miles for Mark II) the Mark I ablative tiles are replaced with reusable surface insulation (RSI) tiles for Mark II missions. These tiles (ablator or RSI) are bonded directly to the common aluminum substructure for both Mark I and Mark II vehicles. The thermal protection system (TPS) weight comparison (Figure 2-22) for the Mark I ablator and Mark II RSI indicates a 7008 pound difference. The added weight penalty associated with the use of ablator for Mark II requirements is 4008 pounds.

Subsystem Requirements

MARK I	*MARK II*
Cross Range • 200 N MI Material • Ablative Bondline Temp • 350 F Substructure • Aluminum	Cross Range • 1100 N MI Material • RSI Bondline Temp • 350 F Substructure • Aluminum

DESIGN APPROACH

- Common Substructure for MARK I/II
- Ablative Tiles Replaced with RSI Tiles for MARK II
- Tiles Bonded on Al Skin

Figure 2-21. TPS Design Development Approach

ITEM	*MARK I* Ablator 200 N MI X Range	*MARK II* Ablator 1100 N MI X Range	*MARK II* RSI 1100 N MI X Range	*MARK II* RSI w/ Ti-Struct 1100 N MI X Range
Body (Lb)	8,793	15,179	12,950	8,241
Wing (Lb)	4,688	8,490	7,055	3,585
Tail (Lb)	648	1,476	1,132	589
Total (Lb)	14,129	25,145	21,137	12,415
Avg Unit Wt (Lb/Ft^2)	1.35	2.41	2.03	1.19

- Ref Wetted Area = 10,436 Sq Ft
- 300 F Bondline Temp
- Alum Structure

Figure 2-22. TPS Weight Comparison

Since the vehicle is sized for RSI Mark II, it is recommended that the single orbiter design be pursued using RSI and that the ablator be maintained as an option.

The orbiter crew station configuration (Figure 2-23) is based on commander and pilot side by side on the flight deck. The seats for the two additional crewmen are below. Restraint provisions also are provided at the manipulator operator station, for cargo handling and docking. This station is positioned on the centerline of the vehicle behind the crew seats on the flight deck. The avionics bays and food and waste management areas will be located against the sidewalls with unobstructed passageways for access during flight and ground maintenance. The design and development approach for the crew station for Mark I versus Mark II requires a flight test kit for Mark I. This includes ejection seats on the flight deck and a jettisonable hatch for ejection clearance. The flight test kit is designed to provide minimum impact for conversion to an operational vehicle.

The main propulsion engine configuration (Figure 2-24) represents the four-engine orbiter which uses J-2S engines for Mark I and HiPc engines for Mark II. The overall system characteristics for both engines are illustrated in Figure 2-25. The design approach uses the philosophy of one for one engine replacement with the design requiring common orbiter interfaces for both the J-2S and HiPc engines. Of primary importance is that conversion from Mark I J-2S to Mark II HiPc engine requires no airframe mold line or thrust structure changes.

The current avionics concept (Figure 2-26) includes dedicated subsystem; digital computer, GN&C only; hardwire controls; triple

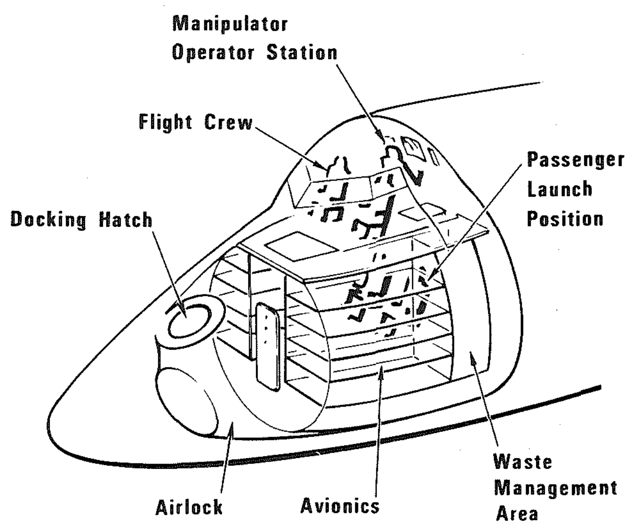


Figure 2-23. Crew Station Configuration

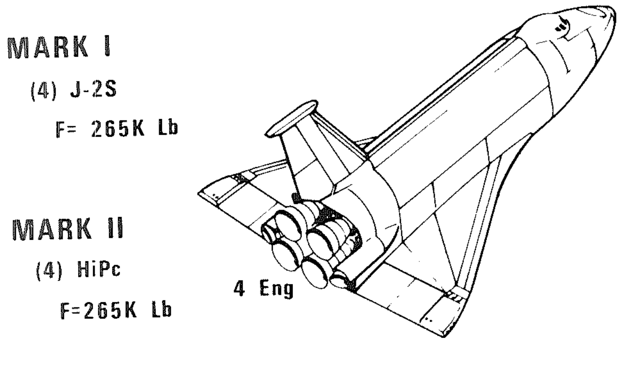


Figure 2-24. Main Propulsion Engine Configuration

System Characteristics

	MARK I	MARK II
Engine Type	J-2S	HiPc
No. of Engs	4	4
Thrust (K lbs)	265 (VAC)	265 (VAC)
Expansion Ratio	40:1	90:1
Isp (Sec)	436	453.4
Mixture Ratio	4.5:1-5.5:1	5.5:1-6.5:1
Throttle Capability	2:1	2:1
Pump Inlet Press.	0-250 PSIA	0-250 PSIA
Eng Wt (Lb)	3800	3125
Recirc Reqd	Yes	Yes

DESIGN APPROACH

- One for One Eng Replacement
- Design J-2S & HiPc Eng with Common Interfaces

Figure 2-25. Main Propulsion Design/Development Approach

redundancy; single string redundancy management; and manual aerodynamic control with control cables. Basic aircraft avionics will be provided for horizontal flight test with kits provided to progressively build up the avionics subsystem through vertical flight test and subsequently through operational flight (Figure 2-27).

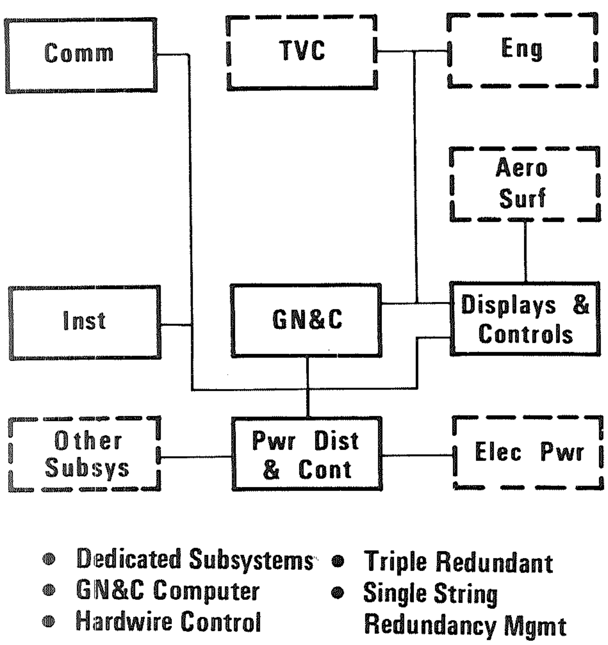


Figure 2-26. Avionics Configuration

REQUIREMENTS

MARK I	MARK II
HORIZ FLT TEST Basic Aircaft Avionics • Manual Flt Cntl (SAS) • Conv D&C • TACAN & UHF/AM • DFI T/M & Recorder VERTICAL FLT AddKit 1 • GN&C Computer, IMU, Star Tracker • C&D For: MPS, TVC, OMS, RCS, APU's & Fuel Cells • Manipulator Sta, Emergency Det Sys, MSFN Xponder	OPERATIONAL FLT Add Kit 2 • Autonomous Nav - Add Horizon Scanner • Docking Umbilical • Delete DFI

DESIGN APPROACH

- Progressive Build-up Using Kits

Figure 2-27. Avionics Design/Development Approach

2.5 CONFIGURATION AND PROGRAM OPTIONS

The first two months of the Phase B extension studies were devoted primarily to the phased development of candidate space shuttle systems employing various expendable boosters. In the phased approach an orbiter with reduced capability would be developed in the first phase, and an uprated orbiter with an RHS booster in the second phase. The program redirection of September 12, 1971 focused the main stream of attention on a basic orbiter, modifiable to an improved second-phase performance and employing either an F-1 flyback booster or a pressure-fed recoverable booster. The basic shuttle system requirements guiding this later two months of study are shown in Figure 2-28.

The design study approach was to size orbiter and booster propellant tanks to satisfy Mark II requirements. Using this booster and orbiter tank, the performance with the Mark I orbiter vehicle with J-2S engines was then analyzed to assure achievement of Mark I performance requirements.

The orbiter options (Figure 2-29) studied included use of the J-2, J-2S, and HiPc engines in both four- and five-engine configurations for Mark I, and upgrading (with the same number of orbiter engines) to J-2S or HiPc for Mark II. These orbiter combinations were assessed against an F-1 flyback booster with four and five engines, a recoverable pressure-fed booster, a phased booster program using first an S-IC stage and then F-1 flyback booster, and a phased program using first an expendable pressure-fed then a recoverable pressure-fed booster.

ITEM	*MARK I*	*MARK II*
• P/L Up (Lb)	10K Min Polar 25K Desired	40K Polar 25K Logs (with Abes) 65K Due East
• P/L Down (Lb)	25K	40K
• OMS ΔV (FPS)	650 Polar 900 Due East	650 Polar 1500 Logs 900 Due East
• OMS Tank Size (FPS)	1000 (Same Tank as MK II)	1000 + 1000 Kit in Cargo Bay
• Cargo Bay	15 X 60 Ft	15 X 60 Ft
• Cross Range (N MI)	Aero 1100 N MI/TPS 200 N MI	Aero/TPS 1100 N MI
• Abort	Intact Land Recovery	Intact Land Recovery
• Touchdown Vel (Knots)	150 for Design	Design for MK II Landing Wt

Figure 2-28. Shuttle System Requirements

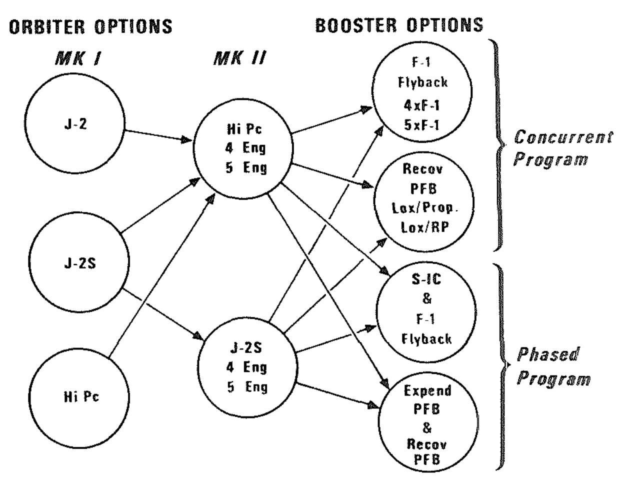


Figure 2-29. Shuttle Program Options

EOHT Orbiter/Pressure-Fed Booster Concurrent Program

This booster/orbiter which uses seven 975,000-pound-thrust (sea level) engines, LO_2/propane propellants, and a low-risk technology orbiter with a single external LO_2/LH_2 belly tank. The flight profile (Figure 2-30) for this configuration has a one-engine cutoff at 30 seconds, a two-engine cutoff at 106 seconds, a two-engine cutoff at 148 seconds, and staging at 150 seconds at 178,000 feet altitude at 6,000 fps. The booster is recovered by parachutes and retrieved by a modified landing ship dock (LSD). For this configuration option, the target dates of June 1976 for the orbiter first horizontal flight, and September 1978 for the first manned orbital flight were used.

A major parameter in the development of the basic orbiter was the selection of an engine and the number of engines. Initially, the HiPc engine was assumed for the Mark II orbiter and the Mark I orbiter was exercised for four or five J-2, J-2S, and HiPc engines. Performance/cost values for these cases (Figure 2-31) showed no clear-cut differences. Of significance, however, was the performance data for the J-2 indicating a five-engine configuration, whereas the J-2S and HiPc cases had a four-engine configuration. The increased cost and complexity of the J-2 orbiter configuration, coupled with the uncertainties associated with the Mark II modifications and supplemental development that would be required for its usage, eliminated the J-2 from contention.

Both the J-2S and HiPc engine cost/performance data favored the four-engine orbiter configuration. A comparison of these two engines showed the higher initial cost of using the HiPc from the outset was largely offset by retrofit and redesign costs of starting with the J-2S in Mark I and going to the HiPc in Mark II. Neither program showed a distinct advantage in total program cost, though peak funding and development funding requirements

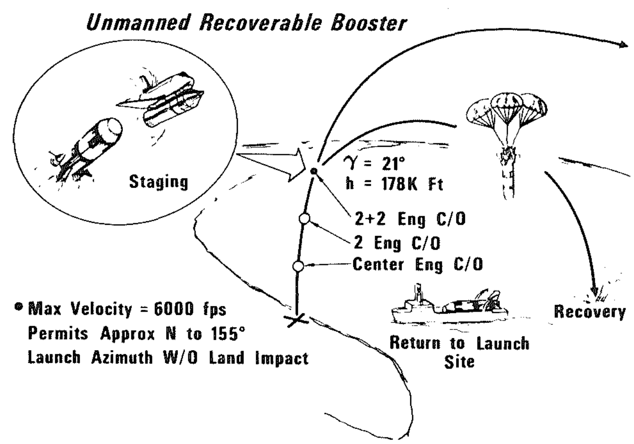
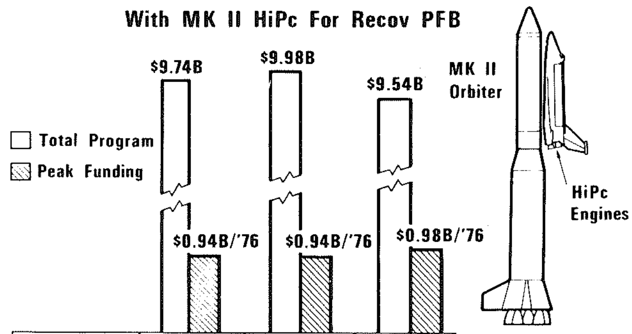


Figure 2-30. Flight Profile, Pressure-Fed Booster



MARK I	J-2S	J-2	HiPc	LOX/PROP. PFB
GLOW $\bar{M}$ Lb	5.173	5.010	5.253	5.253
PL (Polar) K Lb	25	12	49	40

Figure 2-31. Mark I Options

tended to favor the use of J-2S in the Mark I orbiter and coversion to HiPc in Mark II.

The next step in the study was to consider the case of using the J-2S for the entire program. Of concern here was the performance to be achieved in the Mark II mission with the lower specific impulse engine. For the 40,000-pound polar orbit payload capability, both the orbiter tanks and the booster need to be increased in size compared to the size required for the Mark II orbiters. Peak annual funding and development costs are less for the program using J-2S engines on the orbiter throughout the program. Total program costs however are higher with this system due to the costs associated with the larger external tanks and larger booster (Figure 2-32).

The remaining item of significance is the relative development risk involved in the two engines and the attendant confidence in program cost estimates. It is primarily this factor which led to the recommendation of the J-2S engine for both the Mark I and Mark II orbiter used with the pressure-fed booster.

EOHT Orbiter/F-1 Flyback Booster Concurrent Program

This booster/orbiter configuration consists of an F-1 flyback booster which uses S-IC propellant tank geometry and either four or five F-1 engines with a low-risk technology orbiter having a single external LO_2/LH_2 belly tank. The flight profile (Figure 2-33) and booster penalties associated with the interstage separation and/or retention were examined with the resulting recommendation to drop the interstage at staging. For this configuration option, the target dates of June 1976 for the first horizontal flight and September 1978 for the first manned orbital flight were used.

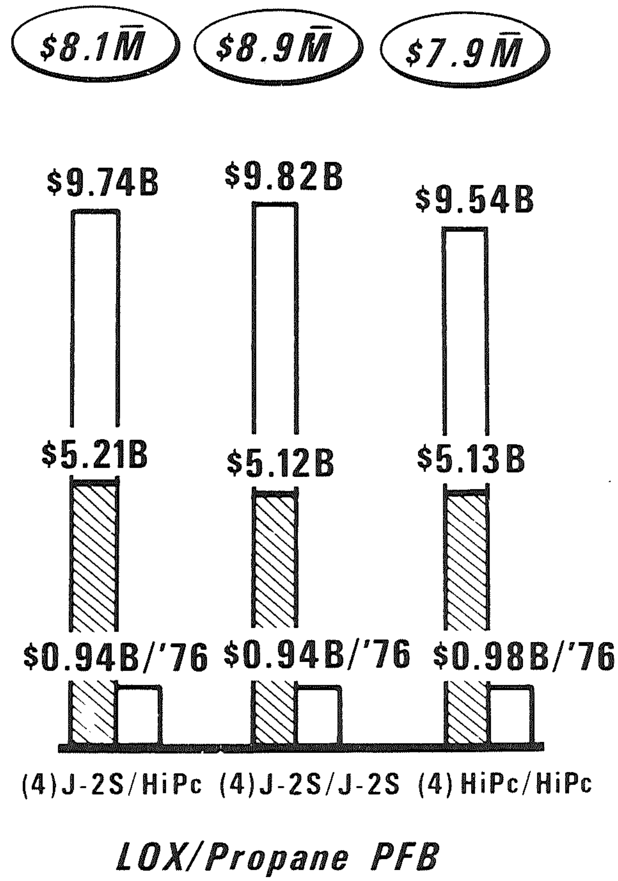


Figure 2-32. Program Cost Comparison - Concurrent Development

The booster options studied for the LO_2/RP system included the four versus five F-1 engine case. The results indicate a slightly lower program cost and acceptable performance for the four F-1 engine booster in each case. However, the risk involved in the one-engine-out case dictates a recommendation for the five-engine booster.

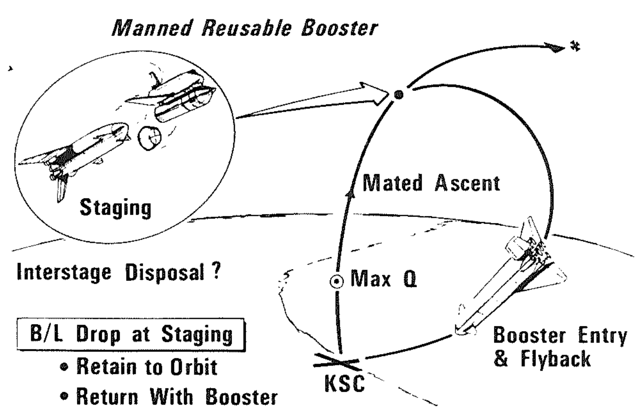


Figure 2-33. Flight Profile - Flyable LO_2/RP Booster

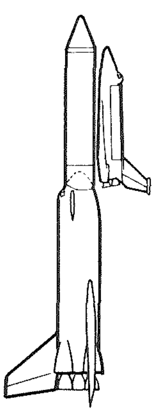
The same orbiter engine options were considered for the F-1 flyback booster as for the PFB. In this case, the J-2 and J-2S Mark I orbiters show a similar advantage in the four-engine configuration as opposed to five J-2's or J-2S's (lower mass fraction than PFB). Both were tolerant of a rather wide spread of staging velocities, though favoring velocities of 6000 fps or slightly higher (Figure 2-34). The use of HiPc engine on the orbiter throughout the program results in higher peak annual funding than use of J-2S engines on the Mark I orbiter (Figure 2-35). The options of retaining the Mark I J-2S engines in the Mark II orbiter or using HiPc engines in both orbiters were again assessed. These results (Figure 2-36) indicate approximately equivalent program costs for the all J-2S and the all HiPc configurations. Peak annual funding and development costs, however, are less for a program using J-2S engines in both Mark I and Mark II.

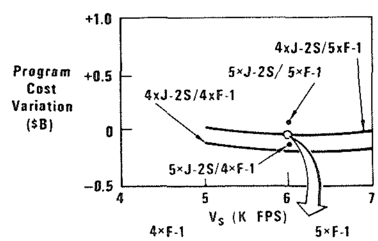
As with the PFB, no one orbiter system is clearly superior from a performance/cost standpoint, and the development risk and attendant confidence in cost estimate tends to favor a totally J-2S orbiter system for both Mark I and Mark II.

Phased Program Options

The remaining major option available in the matrix is a phased program employing S-IC expendable boosters or expendable PFB stages for the first 15 manned orbital flights. While the same FMOF date as the concurrent program is retained, achievement of an equivalent operational flight rate is delayed until 1983 and the advent of the Mark II orbiter (Figure 2-37).

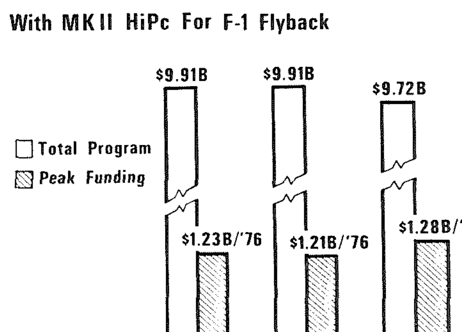
The F-1 flyback/S-IC phased program and the expendable/recoverable pressure-fed booster programs were next compared on a parallel basis for the J-2S/HiPc, J-2S/J-2S, and HiPc/HiPc Mark I/Mark II orbiters. A

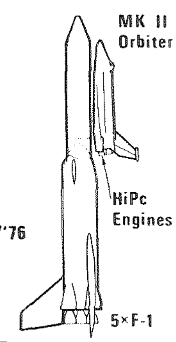




ITEM	MARK I	MARK II	MARK I	MARK II
GLOW M̄ Lb	4.58	4.69	4.98	5.09
BLOW M̄ Lb	3.58	3.58	3.97	3.97
OLOW M̄ Lb	1.00	1.11	1.01	1.12
PL K Lb	27 (Polar)	65 (East)	25 (Polar)	65 (East)
ABORT	X	X	✓	✓
V_S (FPS)	6,300	6,000	6,200	6,000

Figure 2-34. F-1 Flyback Booster/J-2S to HiPc Orbiter





MARK I	J-2S	J-2	HiPc	F-1 FLYBACK
GLOW M̄ Lb	4.98	5.48	5.09	5.09
PL (Polar) K Lb	25	17	49	40

Figure 2-35. Other Mark I Options

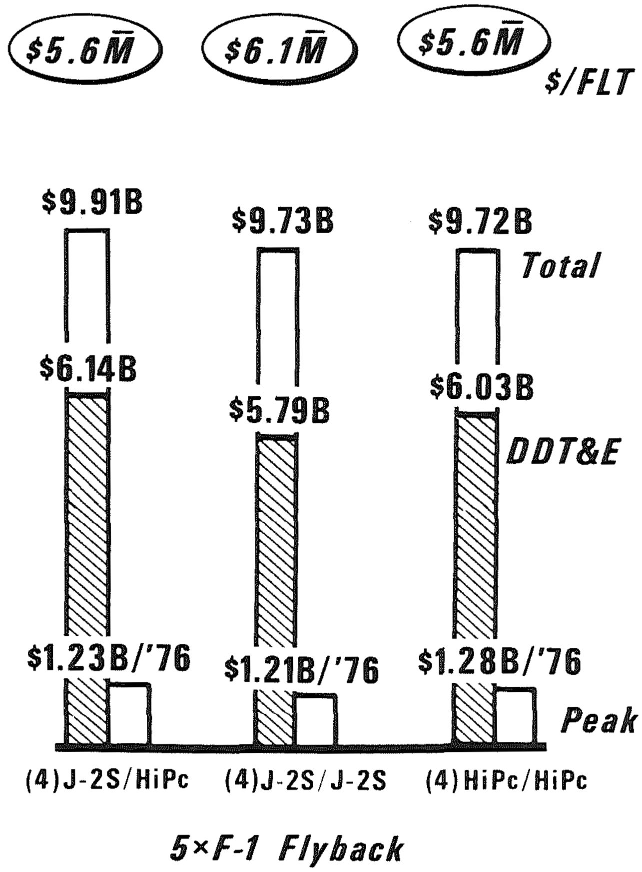


Figure 2-36. Cost Comparison - Concurrent Development

reduction in peak expenditure rates is achieved (Figure 2-38 as compared to Figures 2-32 and 2-36) at some penalty to schedule and with a commensurate increase in total program cost.

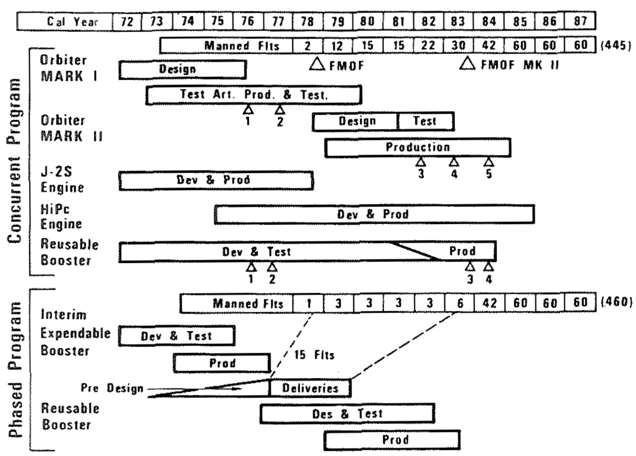


Figure 2-37. Concurrent Versus Phased Program Comparison

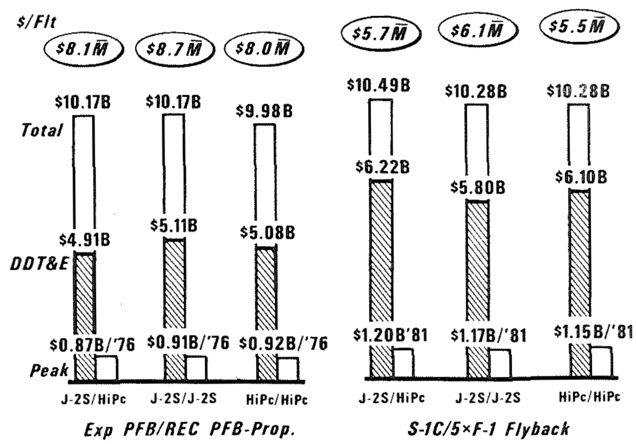


Figure 2-38. Program Cost Comparisons - Phased Development

Other Considerations

The relative merits and sensitivities of the programs studied are felt to be significant factors in their evaluation. Since engine cost is a major factor in the program cost comparisons, a sensitivity study was performed on F-1 engine life. The results (Figure 2-39) indicate a high degree of cost sensitivity to changes in the number of flights per engine. Also of significance was the relative insensitivity of cost to changes in the assumed baseline of five percent attrition rate for PFB components (Figure 2-40).

The space shuttle systems derived in the foregoing configuration studies were evaluated for mission capability by using the Flemming Mission Model, space station missions beginning in 1981 and a space tug in 1985. The Mark I and Mark II orbiters were used with and without the addition of up to 1000 fps or OMS delta velocity fuel in the cargo bay. The results of this evaluation (Figure 2-41) indicate an overall 93 percent mission capability for Mark I and more than 98 percent for Mark II.

Conclusions

The results of the analysis of configuration and program options are:

1. Low technology risk orbiters with J-2, J-2S, and HiPc main engines have comparable program costs and expenditure rates.

2. J-2 engine will require a larger vehicle with a five-engine orbiter.

3. The J-2S orbiter for both Mark I and Mark II achieves the lowest development risk.

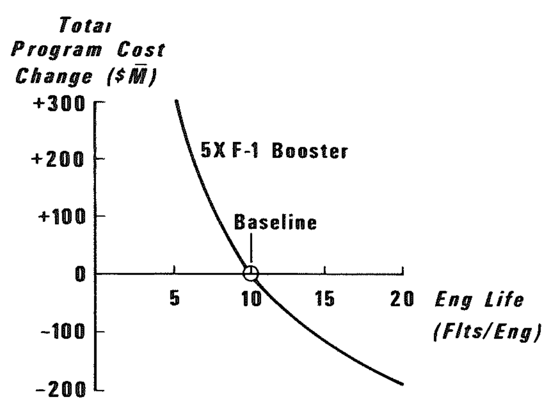


Figure 2-39. Program Cost Sensitivity - F-1 Engine Life

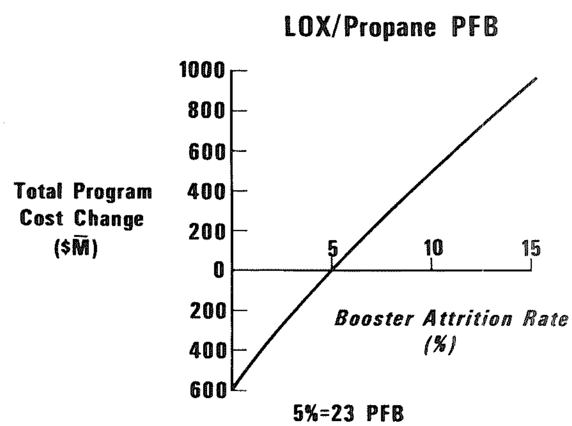


Figure 2-40. Program Cost Sensitivity - PFB Attrition

	28.5°/30° Incl	55° Incl	90° Incl	All Incl
No. of Missions	273	140	137	550
MARK I-NO ABES				
Within Capability	237	139	135	511
Percent	87%	99+%	98+%	93%
MARK II-NO ABES				
Within Capability	267	139	137	543
Percent	98%	99+%	100%	98+%

• OMS Kit Reqd on 28%

Figure 2-41. Shuttle Mission Capability Summary

4. The PRB booster yields lower program costs and expenditure rates than the F-1 flyback booster.

5. The phased and concurrent programs have comparable program costs and expenditure rates.

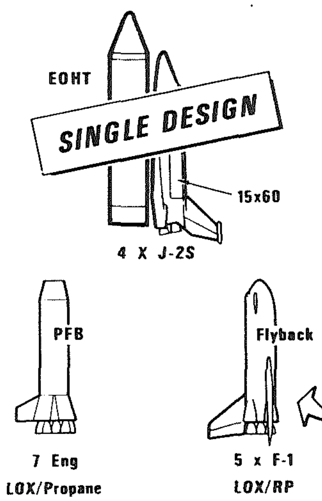
6. The J-2S orbiter for both Mark I and Mark II has the desired payload capability and captures 93 percent (Mark I) and 98 percent (Mark II) of the missions in the Flemming model.

2.6 SUMMARY AND FUTURE PLANS

The primary goals of the extension study were to (1) select orbiter/main engine development approach; (2) select external tank (LH_2 vs LO_2/LH_2); (3) select interim and final booster; and (4) define the recommended program.

The results of the last two months' effort have confirmed that the low-risk technology orbiter results in lower expenditure rates and total program costs. Comparison of the J-2, J-2S, and HiPc main engine show that these options have comparable program costs but that the J-2S development has the lowest risk since the approach starts with a known engine. The results of the special emphasis avionics study showed that dedicated aircraft and spacecraft avionics with maximum use of unmodified off-the-shelf equipment had lowest cost and risk. The TPS weights used in sizing the vehicle are those of RSI for high cross range and hence it is recommended that RSI be used in the vehicle design. The analysis of concurrent versus phased programs showed that the annual expenditures were comparable.

Therefore, for the selection of the orbiter/main engine development approach, the recommendation (Figure 2-42) is a single design orbiter with a 15-foot-diameter by 60-foot-long cargo bay, 40,000-pound polar/65,000-pound due-east payload, use of J-2S only, and use of high cross-range RSI.



Baseline for Follow-on:

- Concurrent Program
- 40K Polar/65K East
- J-2S Eng's Only
- HCR-RSI

Maintain Options for:

- HiPc Engine
- Ablator TPS

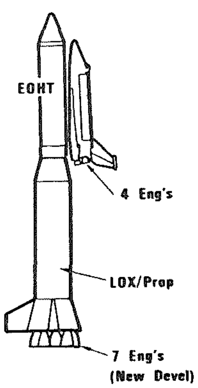
Continue Study for Dec 15th Selection

Figure 2-42. Orbiter/Main Engine/External Tank Approach

The development option for the HiPc engine and ablator TPS should be maintained.

More detailed studies of the external tanks, especially in manufacturing hours and learning curves, confirmed the costs presented at the September 1, 1971 review. The external LH_2/LO_2 single belly tank design has lower cost and lower risk and results in the least weight sensitivity for the orbiter vehicle.

The booster studies showed that the pressure-fed booster (Figure 2-43) and the F-1 flyback booster (Figure 2-43) and the F-1 flyback booster (Figure 2-44) yield comparable program costs, peak funding, and cost per



	MARK I	MARK II
GLOW	5.17$\overline{M}$ Lb	5.25$\overline{M}$ Lb
Payload	25K Polar	40K Polar
Orb Eng's	J-2S	HiPc
TPS	LCR/Abl	HCR/RSI
Abort	Down Range	Return to Site
Risk	Recov & Refurb	
Prog. Cost	$9.74B	
Peak	$0.94B	
Cost/Flt	8.10\overline{M}$	

Figure 2-43. Summary of Pressure-Fed System

	MARK I	MARK II
GLOW	4.98$\overline{M}$ Lb	5.09$\overline{M}$ Lb
Payload	25K Polar	65K East
Orb Eng's	J-2S	HiPc
TPS	LCR/Abl	HCR/RSI
Abort	Down Range B Prop Dump	Return to Site B Prop Dump
Risk	Flyback & Eng Life	
Prog Cost	$9.91B	
Peak	$1.23B	
Cost/Flt	5.6\overline{M}$	

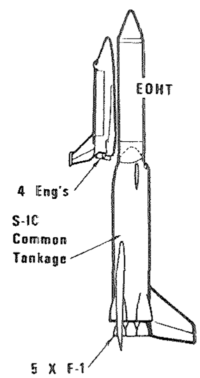


Figure 2-44. Summary of F-1 Flyback System

flight. The PFB recovery and refurbishment risk was comparable with the F-1 flyback booster engine life risk. Therefore, it is recommended that study of both boosters be continued.

Since the booster could not be selected at this time, no recommended program can be defined.

The future plans (Figure 2-45) include a requirements and preliminary system definition effort based on those systems illustrated in Figures 2-43 and 2-44, resulting in a requirements review on December 15. At this review, the booster option will be selected, the avionics design concept confirmed, and the J-2S design requirements defined. The following two and one-half months of effort consisting of subsystem definition and interface requirements and documentation will result in a final review on February 28, 1972. At this review, the Phase C/D system and program definition and cost estimates will be presented.

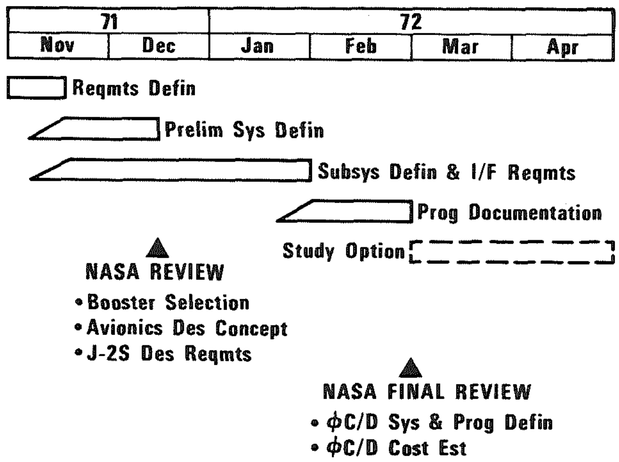


Figure 2-45. Phase B'' Study Approach

3.0 PHASE B EXTENSION - PHASE 1 STUDY ACTIVITY

Phase B studies resulted in definition of a fully reusable space shuttle system which required high annual expenditures for development, and necessitated significant advancement in technology. During the first phase of a study extension (July - September 1971), programs were analyzed which would minimize annual expenditures and technology risk. These programs used interim expendable boosters and orbiters with external propellant tanks. The objectives of the first phase of this study extension were as follows:

1. To evaluate the merits of external LH_2 tanks versus external LO_2/LH_2 tanks.
2. To determine the cost effectiveness of various cargo bay sizes.
3. To evaluate the cost savings associated with deleting the abort-to-orbit capability.
4. To evaluate the relative merits of various interim booster designs and related phased development programs.
5. To evaluate the merit of phased development programs compared to concurrent programs.

In addition, the program benefits from a low-technology approach to orbiter design were also established. The results of all of these studies are reported in this section.

3.1 PHASE 1 CONFIGURATION STUDY MATRIX

Expendable booster stages are used for the interim system (Generation 1) and an LO_2/LH_2 reusable booster is developed for the operational system (Generation 2). The orbiter configurations studied featured major variations in terms of external tank configuration (hydrogen only, and LO_2 and hydrogen), payload bay size (12 and 15 feet in diameter and 40 and 60 feet in length), and number of orbiter engines (once-around abort or no once-around abort). The Generation 2 booster is an LO_2/LH_2 reusable booster while the Generation 1 expendable boosters are either a 260 SRM, 120 SRM cluster, 156 SRM cluster, or an LO_2/LH_2 expendable booster. The overall study approach is to evaluate combinations of these major configurations. The vehicles studied in this contract extension phase are defined in Table 3-1.

Table 3-1. Configuration Study Matrix

Config No.	Development Approach	Tanks[2]	Cargo Bay	No. of Orb Engines	Expend Booster	Payload[3]
1	Gen 1 and 2	HO	15 x 60	3 or 2	260 SRM	65
2	Gen 1 and 2	HO	15 x 60	3 or 2	LO_2/LH_2 core	65
3	Gen 1 and 2	HO	15 x 60	3 or 2	Cluster 120 or 156 in.	65
4	Gen 1 and 2	H	15 x 60	3 or 2	260 SRM	65
5A	Gen 1	HO	15 x 40	3 or 2	260 SRM	45
5B	Gen 2	HO	15 x 60	3 or 2	—	65
6A	Gen 1	HO	12 x 40	1	260 SRM	45
6B	Gen 2	HO	12 x 60	1	—	65
6C	Gen 1	HO	12 x 40	3 or 2	260 SRM	45
6D	Gen 2	HO	12 x 60	3 or 2	—	65
7A	Gen 1	HO	12 x 40	3 or 2	LO_2/LH_2 core	45
7B	Gen 2	HO	12 x 60	3 or 2	—	65
8A	Gen 1	H	12 x 40	3 or 2	Cluster solids	45
8B	Gen 2	H	12 x 60	3 or 2	—	65
9	Gen 1 and 2	H	15 x 60	3 or 2	LO_2/LH_2 core	65

Note: [2]HO external hydrogen and oxygen tanks

[3]Up payload = 65 Klb; down payload = 40 Klb
Up payload = 45 Klb; down payload = 25 Klb

3.2 MISSION AND SYSTEM REQUIREMENTS

The mission requirements for Generation 1 and Generation 2 systems are listed in Table 3-2. The Generation 1 system is used for four years, after which time the Generation 2 system is introduced into the program. The payload capability for Generation 2 is exactly the same as that specified for the Phase B study. However, the payload capability requirement for the Generation 1 system is 45,000 pounds placed in a 100-nautical-mile orbit by 28-1/2-degree inclination orbit. The operational requirements for Generation 1 are identical to Generation 2 requirements with the following exceptions.

1. Aerodynamic cross range capability only is required for return to launch site from a single polar orbit mission.

2. Simplified subsystems may be used.

3. An expendable booster may be used.

Table 3-2. Mission Requirements

MISSION REQUIREMENTS

GENERATION 1	GENERATION 2
TRAFFIC MODES	
A) 3 FLIGHTS/YEAR FOR 4 YEARS	445 FLIGHTS/10 YEARS
B) FMOF 1978	FMOF - 1982
PAYLOAD	
PAYLOAD CAPABILITY BUILDUP WITHIN BUDGET CONSTRAINTS CARGO BAY EITHER 12 FT OR 15 FT DIA	CARGO BAY: DIAMETER 12 OR 15 FT; LENGTH 40 OR 60 FT

CARGO BAY	40 FT LONG	60 FT LONG
ORBIT	100 N MI x 28.5°	100 N MI x 28.5°
PAYLOAD	45K LB	45K LB
OMS ΔV	900 fps	900 fps
ABES	OUT	OUT
LANDING P/L	25K LB	25K LB

REFERENCE MISSIONS

	ORBIT INCLIN	P/L (LB)	ABES	OMS ΔV
NO. 1	100 N MI x 28.5°	65,000	OUT	900 fps
NO. 2	270 N MI x 55°	25,000	IN	1500 fps
NO. 3	100 N MI x 90°	40,000	OUT	650 fps

OPERATIONAL REQUIREMENTS

GENERATION 1	GENERATION 2
ABORT: INTACT ABORT TO ONCE-AROUND	INTACT ABORT ONCE-AROUND
CROSS RANGE: AERODYNAMIC CAPABILITY FOR RETURN TO LAUNCH SITE FROM SINGLE POLAR ORBIT MISSION	RETURN TO LAUNCH SITE FROM SINGLE POLAR ORBIT MISSION
	BOOSTER RETURN TO LAUNCH SITE ENCOUNTERING DIRECTIONAL WINDS
CREW COMPARTMENT: SIZE FOR 2 + 2 (400 FT^3)	SIZE FOR 2 + 2 (400 FT^3)

TECHNICAL REQUIREMENTS

GENERATION 1	GENERATION 2
SUBSYSTEMS: BASELINE ØB. CHANGES TO MINIMIZE COST	BASELINE ØB SUBSYSTEMS
REDUNDANCY: ORBITER - ØB - BOOSTER MAN RATED	SAME AS ØB
CONTINGENCY: 10% DRY WEIGHT EXCEPT MPS ENGINES	10% DRY WEIGHT EXCEPT MPS ENGINES
FPR: 1% IN ORBITER	1% IN ORBITER
WINDS: ØB WIND CRITERIA EXCEPT DIRECTIONAL WINDS FOR ASCENT	ØB WIND CRITERIA EXCEPT DIRECTIONAL WINDS FOR ASCENT
MAX AXIAL ACCELERATION - 3 g's	MAX AXIAL ACCELERATION - 3 g's
ORBITER NORMAL LOAD FACTOR, ENTRY 2.5 g's	ORBITER NORMAL LOAD FACTOR, ENTRY 2.5 g's
ABES IN CARGO BAY. WEIGHT CREDITED TO PAYLOAD	ABES STORED IN CARGO BAY WEIGHT CREDITED TO PAYLOAD
SHUTTLE Hi-P_C ENGINES	REUSABLE BOOSTER WILL BE HEAT SINK TYPE
OMS VOLUME ALLOCATION EQUIVALENT TO 2000 fps	SHUTTLE Hi-P_C ENGINES. THRUST TO BE DETERMINED. (RANGE 200K SL LB TO 600K SL LB)
OMS ΔV = 900 FT/SEC FOR 100 N MI x 28.50 INCLINATION ORBIT	OMS VOLUME ALLOCATION EQUIVALENT TO 2000 fps
OMS SYSTEM MAY BE BURNED DURING NORMAL OPERATIONS FOR ASCENT PROPULSION	OMS SYSTEM MAY BE BURNED DURING NORMAL OPERATIONS FOR ASCENT PROPULSION
EXPENDABLE BOOSTER WILL BE UNMANNED	REUSABLE BOOSTER WILL BE MANNED FOR OPERATIONAL FLIGHTS

3.3 COST GROUND RULES

In performing the following trade studies, costs were developed based upon the schedule milestones and ground rules shown in Table 3-3. These costs reflect significant variations in the configuration and program options studies. While learning for hardware fabrication was not applied to items of relatively low production rates, it was applied to items such as expendable tanks, expendable boosters, and engines where large costs and relatively high production rates would exist. Commonality between orbiter and booster was considered in the computations.

Table 3-3. Schedule and Cost Ground Rules

1. Phase C/D authority to proceed 4/72 (approx)
2. Orbiter first horizontal flight 5/77 (approx)
3. First interim manned orbiter flight 9/78 (approx)
4. 12 interim flights at 3 flights/year for 4 years
5. First reusable manned orbital flight 9/82 (approx)
6. Conduct 445 flights during 10 year (NASA mission model)
7. Flight dates fixed for all program options
8. No learning for reusable orbiter and booster vehicles
9. Costs include:

   Nonflyable propulsion test vehicle
   Main engine development and production
   Government facilities

10. Commonality considered for applicable program options

3.4 SCHEDULE GROUND RULES

The primary objective for evaluating a phased development concept was to reduce annual funding requirements without major increases in total program costs. A number of program schedules were developed to assess the merits of a phased development program. These schedules are described in the following paragraphs. For each alternative, Phase C/D authority to proceed (ATP) was assumed to occur on April 1, 1972.

3.4.1 Schedule A (Figure 3-1)

The parallel orbiter/booster development plan provides two orbiters and two boosters, in the initial phase, to support a first horizontal flight (FHF) date in April, 1977, the first manned orbital flight (FMOF) in September, 1978, and a 12-flight program for four years. The delivery of the remaining three orbiters and two boosters is delayed as late as possible consistent with meeting a flight date in September, 1982, and with fulfilling the requirements of the 445 flight traffic model. In addition, the fatigue testing and its associated costs were deferred as late as possible consistent with the higher flight rate that begins in September, 1982.

3.4.2 Schedule B (Figure 3-2)

The phased booster development plan is based on use reusable orbiters with cargo bays of either 15- by 40- or 15- by 60-feet to support a FHF in April, 1977, a FMOF in September, 1982, and a 4-year flight program with SRM expendable boosters.

The development program for the expendable boosters was keyed to support the FMOF in September, 1978, and fly four years in parallel with development of a reusable orbiter. This permitted deferring the development of a reusable heat sink booster by approximately four years with a considerable reduction in annual funding.

The deliveries of Orbiters 3, 4, and 5 were delayed as late as possible consistent with meeting a reusable booster flight date in September, 1982, and the requirements of the 445 flight traffic model.

3.4.3 Schedule C (Figure 3-3)

The interim orbiter/booster plan provides two orbiters with a 15- by 40-foot cargo bay for initial development with 260 SRM expendable boosters. The 260 SRM booster program ATP was developed to support a FMOF of September, 1978, and a 12-flight program for four years.

The remaining three orbiters are configured with a 15- by 60-foot cargo bay, and their development, along with the reusable booster program, is deferred consistent with supporting a launch date in September, 1982, and the 445-flight traffic model.

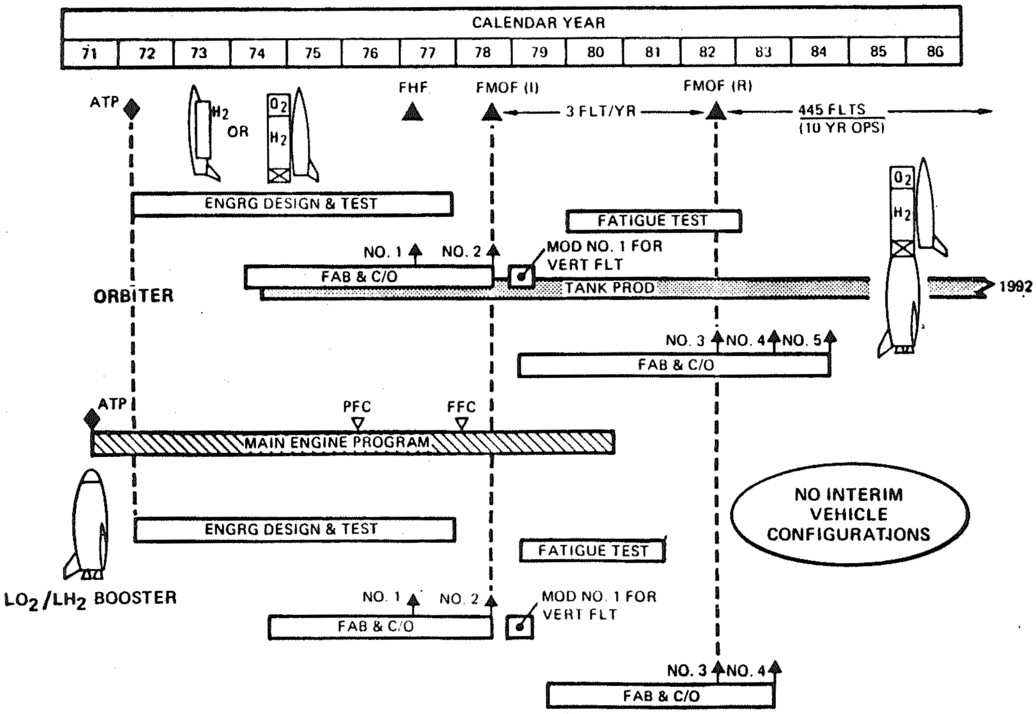


Figure 3-1. Program Schedule With Extended Initial Flights, Parallel Orbiter/Booster Development

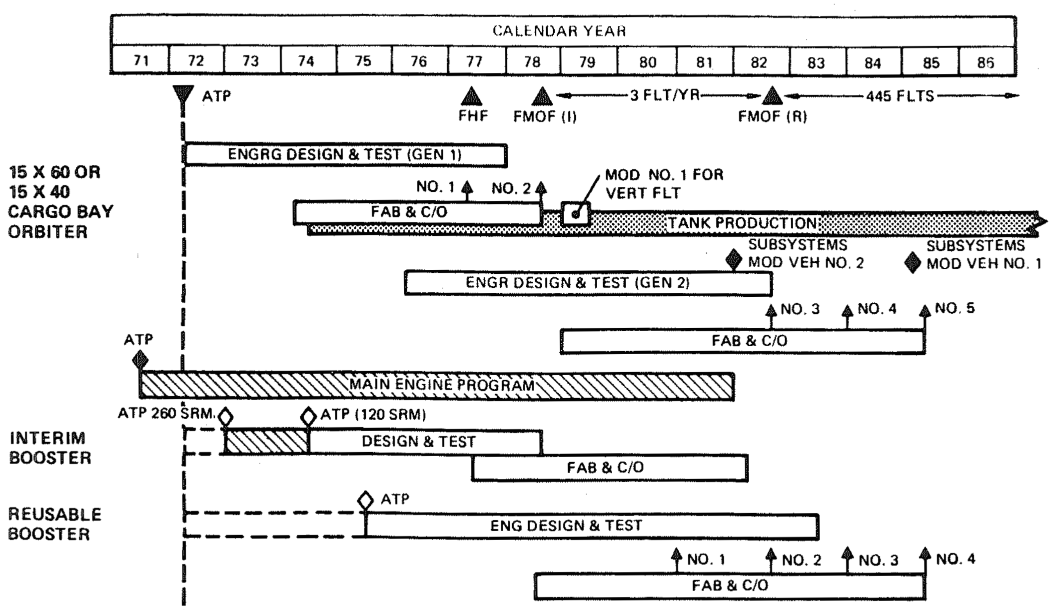


Figure 3-2. Program Schedule for Phased Booster Development, Interim SRM Booster/Reusable Booster

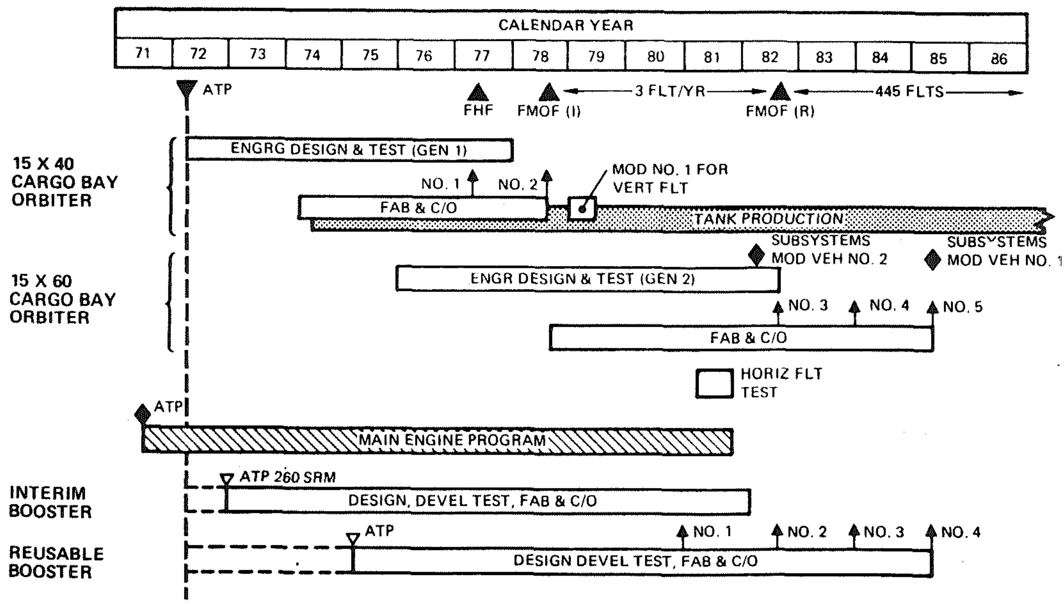


Figure 3-3. Program Schedule for Interim Orbiter/Booster, Orbiter (15 x 40) and 260 Inches, SRM/Orbiter 15 x 60 and Reusable

3.5 PHASE 1 STUDY LOGIC

The study approach for the first phase of the Phase B extension study is illustrated in Figure 3-4. Design studies were conducted to define an orbiter which minimizes weight and cost through aerodynamic and packaging improvements. In parallel with these studies, analyses of various external propellant tank designs were completed to determine (1) which propellants should be tanked externally, (2) where the tank should be mounted, and (3) the least costly method of tank construction. The results were utilized in sizing the orbiter propellant tankage and Generation 1 and 2 expendable and reusable boosters. Total system feasibility was established through studies of control requirements, abort and separation techniques, and evaluation of the program technical risk. The results were combined with studies of program phasing to establish least-cost total programs embodying minimum peak funding requirements and to evaluate the merits of the interim systems considered.

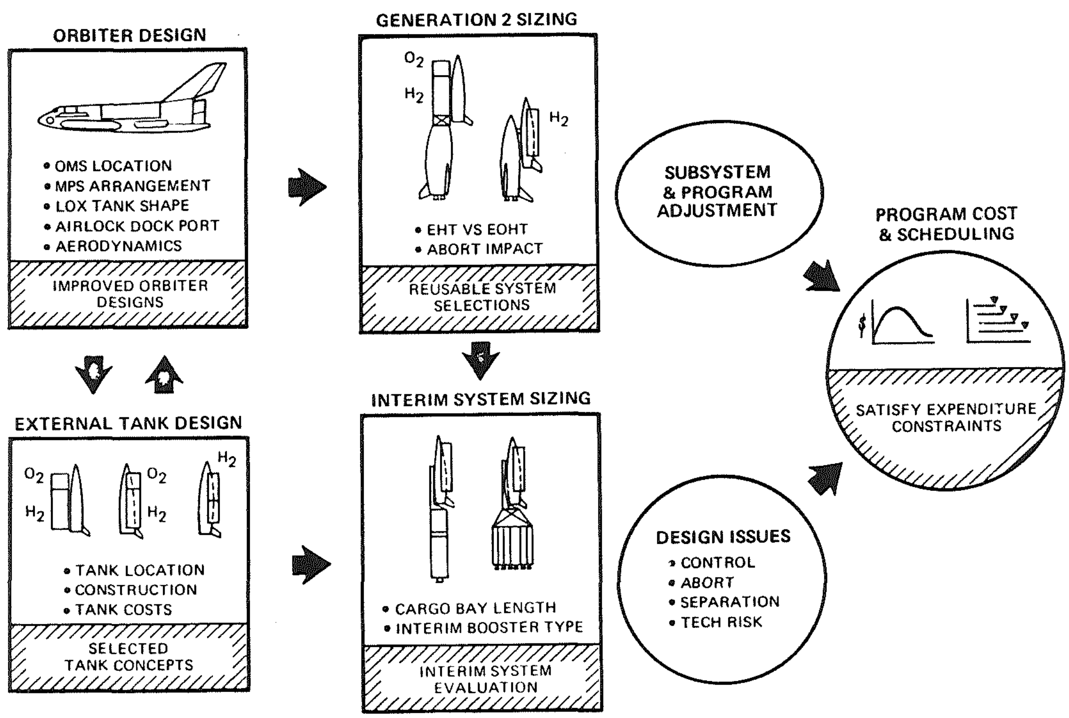


Figure 3-4. Study Approach

3.6 EHT VERSUS EOHT COMPARISON

At the conclusion of the first 12 months of the Phase B study, it was shown that the size of the integrated shuttle vehicle and orbiter could be reduced by carrying orbiter ascent LH_2 in external tanks that are expended following propellant depletion. The major advantages identified for the external tank concept were reduced peak annual expenditure and reduced technical risk associated with the fracture mechanics of reusable LH_2 tanks. Because of the study results stated above, a key orbiter configuration issue studied during the Phase B Extension was whether an external hydrogen tank (EHT) orbiter design or an external oxygen and hydrogen tank (EOHT) orbiter design best satisfies the overall NASA objectives.

To make the EHT-EOHT comparison, the following approach was adopted:

1. An orbiter configuration was developed for each concept
2. An integrated vehicle was developed for each concept
3. The significant configuration differences were evaluated (i. e., abort capability, fracture mechanics, test impact, facilities impact, technical risk, etc.)
4. Annual expenditure and total program cost were developed for each concept

The results of these studies and the recommended configuration are discussed in this section.

3.6.1 Orbiter Comparison (EHT and EOHT)

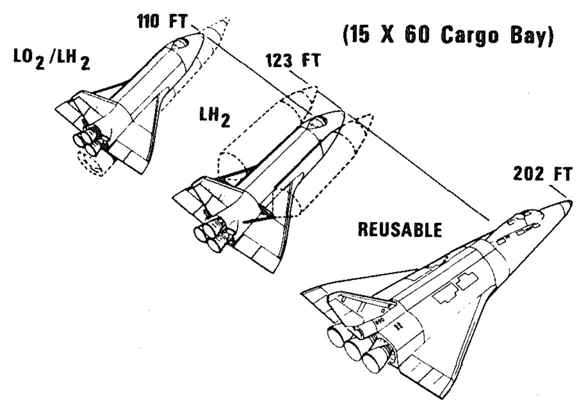
Many EHT and EOHT orbiter configurations were developed during the initial part of the study to evaluate orbiter general arrangements which provide minimum orbiter size and weight. The EHT and EOHT orbiter configurations selected for comparison are illustrated in Figure 3-5 with significant configuration characteristics listed. The EHT and EOHT orbiters are also compared with a three-engine reusable orbiter design with LH_2 and LO_2 stored within the entry vehicle.

The EHT orbiter design employs two LH_2 tanks mounted on the orbiter body above the wing. The EOHT orbiter design employs a single external tank mounted under the orbiter entry vehicle. Figure 3-5 shows that the EOHT orbiter is smaller in size and weight than the EHT. This results primarily from the requirement to package LO_2 within the EHT orbiter entry vehicle. The LO_2 tank (10,600 cu ft) is located forward of the wing carry-through structure and under the cargo bay and cabin. In addition to the tank volume requirement, the packaging efficiency of the EHT orbiter is inferior to that of the EOHT orbiter, requiring approximately 27,000 cu ft additional body volume. The combined dry weight of the EOHT orbiter (entry vehicle and external tank) is approximately 16,000 pounds less than the EHT orbiter and, therefore, provides a more efficient launch vehicle (less GLOW to deliver the same payload). This results in a smaller Hi Pc engine requirement for the EOHT configuration.

3.6.2 Aerodynamics

Aerodynamics activity during the first phase of the study consisted of an evaluation of external propellant orbiter configuration options to establish performance characteristics, size of aerodynamic surfaces, and to determine trim angle of attack and center of gravity (cg) limits.

Major orbiter configuration trade studies included propellant location and arrangement; payload bay size; and orbiter main engine number, size, and arrangement. The effects of these design variables on aerodynamic characteristics were compared and evaluated for the various configuration options. Aerodynamic design guidelines were selected on the basis of previous Phase B studies and NR preliminary design practice. Maximum



	EHT	EOHT	REUSABLE
BODY VOL, FT^3	58,029	31,450	116,097
PACKAGING EFFICIENCY	39.6%	46.9%	
BODY SURFACE, FT^2	10,035	6,297	15,716
WING AREA EXPOSED, FT^2	2,851	2,499	3,663
DRY WEIGHT, K LB (LESS TANKS & PAYLOAD)	191.2	148	293
ORBITER EXT TANK WT, K LB	19.7	48	0
3 Hi-Pc-F_{VAC}, K LB/ENG	447.5	420	551

Figure 3-5. EHT and EOHT Orbiter Configurations

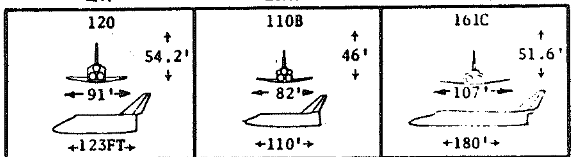
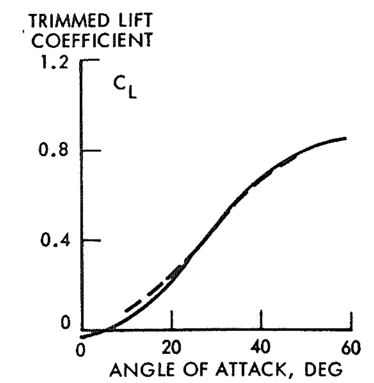
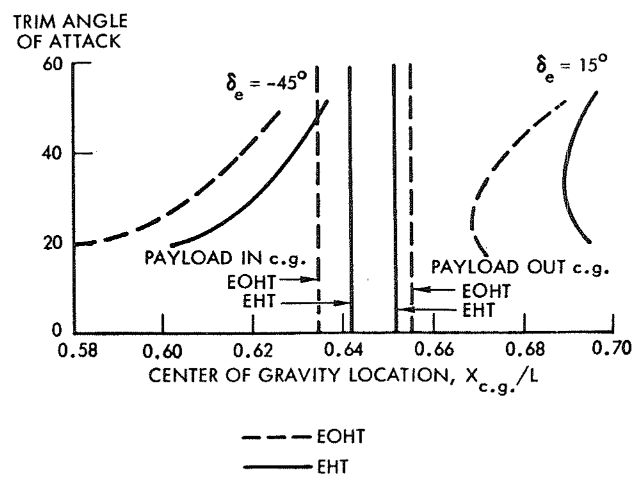
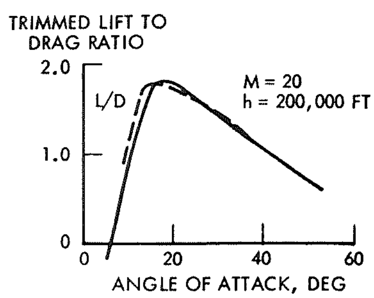
	EHT	EOHT	ØB REUSABLE
	120	110B	161C
			
DESIGN			
ENTRY WEIGHT K LB	234.1 (Landing)	192 (Landing)	268.7
DRY WEIGHT K LB	191.2	148	222.3
BODY VOLUME FT^3	58,029	31,450	85,655
PACKAGING EFFICIENCY %	39.6	46.9	
WING AREA, THEORETICAL FT^2	4,678	3,829	6,474
AERODYNAMICS			
HYPERSONIC L/D MAX	1.8	1.8	2.2
L/D AT α = 30°, M = 20	1.43	1.5	1.6
W/C_LS AT α = 30°, M = 20	111.0	109.0	83.0
SUBSONIC LIFT CURVE SLOPE, I/DEG	0.036	.0344	.035
SUBSONIC L/D MAX	6.0	5.7	7.2
TOUCHDOWN SPEED KNOTS	156	159	179
AFT C.G./LIMIT % L	65.2	65.6	68.7
FWD C.G./LIMIT % L	64.2	63.5	66.5

Figure 3-6. Comparison of EHT and EOHT Orbiters

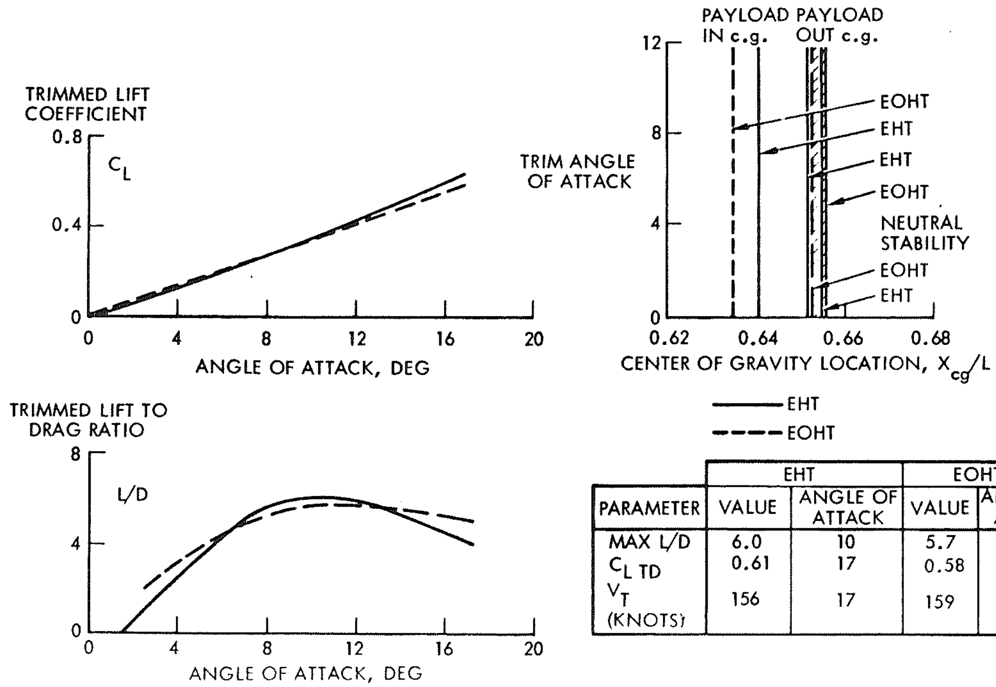






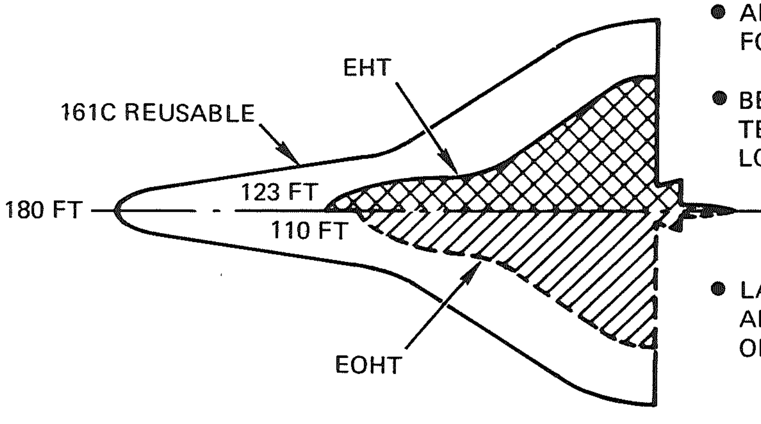
PARAMETER	EHT VALUE	EOHT VALUE
L/D	1.43	1.50
W/C_LS (PSF)	111.0	109.0
α (DEG)	30°	30°

Figure 3-7. Hypersonic Aerodynamics Characteristics Comparison (EHT-EOHT)



PARAMETER	EHT VALUE	EHT ANGLE OF ATTACK	EOHT VALUE	EOHT ANGLE OF ATTACK
MAX L/D	6.0	10	5.7	11
C_L TD	0.61	17	0.58	17
V_T (KNOTS)	156	17	159	17

Figure 3-8. Subsonic Aerodynamic Characteristics Comparison (EHT-EOHT)



FINDINGS

- AERODYNAMIC PERFORMANCE ADEQUATE FOR BOTH CONCEPTS
- BEST PACKAGING EFFICIENCY FOR EOHT TENDS TO INCREASE ENTRY LIFT LOADING

  30% GREATER W/C_LS COMPARED TO REUSABLE ORBITER
- LARGE C.G. SHIFT DUE TO PAYLOAD IN AND OUT - MAY REQUIRE REMATCHING OF WING & BODY GEOMETRY

  SWEEP

  WING POSITION

Figure 3-9. EHT Versus EOHT

3.6.3 Tank Selection

The various considerations that must be evaluated in the process of tank selection are shown in Figure 3-10.

The tank selection for the EHT or the EOHT tank concepts should yield the least-cost shuttle program whether it be the initial, yearly, or total costs. In reality, the program with the least initial cost is not always the program with the least total cost.

The configuration of the tank is closely involved with the overall configuration of the shuttle. The location of the tank(s), whether below, alongside, or ahead of the orbiter, affects the location of the orbiter with respect to the booster. During the EHT study, the orbiter-booster location was inviolately parallel, with the orbiter located on top of the booster, and the external hydrogen tanks could only be located alongside or ahead of the orbiter. A number of EHT tank options are shown in Figure 3-11. As illustrated, the use of two cylindrically shaped tanks without entry thermal protection resulted in the lowest weight.

The study ground rules for the EOHT concepts lifted the restriction of the orbiter being parallel to the booster, and the EOHT tank-location possibilities were increased. The location of the tank, therefore, determined the overall configuration and the internal location of the propellants.

For least cost, the basic structure of the tank should be monocoque; the method of tank support and load application to the internal propellants decreed semi-monocoque structure for some of the tank concepts. Monocoque structure (skins only) is desired instead of semi-monocoque (skins plus frames plus stringers) because the tank structure with the least number of parts is normally the cheapest to fabricate. The thermal protection system is based on boost requirements only; the original requirements for the thermal protection system were based on boost and entry for the tank. Analyses during the EHT tank study indicated that breakup of tanks at approximately 350,000 feet would result in tank fragment dispersion in the Indian Ocean (Figure 3-12) and this dispersion was considered acceptable. This design therefore allowed removal of thermal protection for entry. The systems on board the tank assembly were evaluated to perform the functions listed in Figure 3-10; the resulting systems were selected for least cost—which meant the use of the least number of systems and the installation of system components on board the orbiter, where they were reusable, rather than on board the tank where they were lost with each jettisoned tank. In addition, each of the tank concepts were designed for the Generation 2 shuttle with the recoverable booster and were to be adaptable to the Generation 1 shuttle with the expendable boosters.

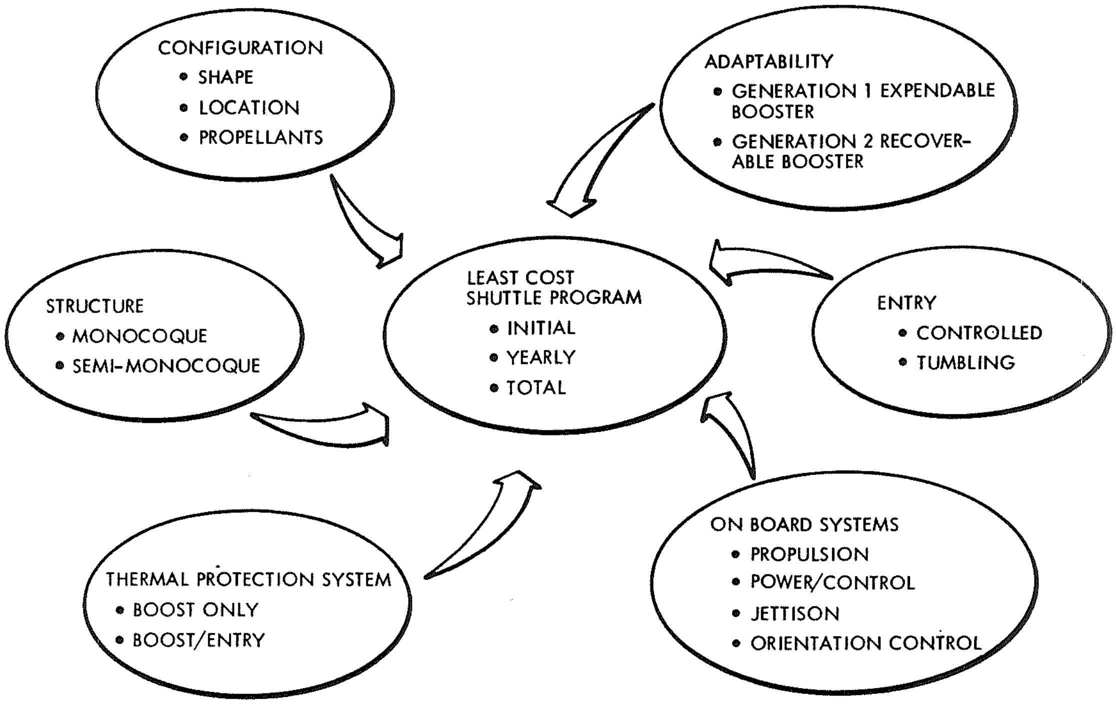
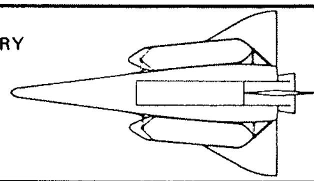
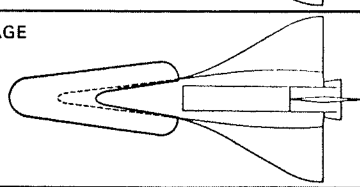
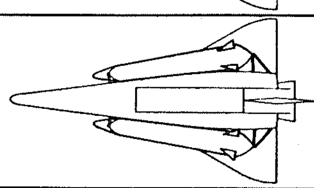
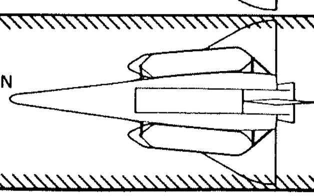


Figure 3-10. Tank Selection Process

	TANKAGE	CONFIGURATION	FEATURES	WEIGHT (LB)
INTACT ENTRY TO 200,000 FT	CYLINDRICAL	TANKAGE INTACT ENTRY 	• 2 TANKS • MONOCOQUE CONSTRUCTION • ABLATOR FOR ASCENT & ENTRY • SOFI INSIDE/OUTSIDE TANK • TUMBLING ENTRY • SPIN STABILIZED	20,870
	SPLAYED	TANKAGE 	• SINGLE ASSEMBLY • STRINGER/FRAME CONSTRUCTION • ABLATOR FOR ASCENT & ENTRY • SOFI INSIDE/OUTSIDE TANK • AERO STABILIZED ENTRY • INCREASED ORBITER LOADS	26,450
	CONICAL	TANKAGE 	• AERODYNAMIC FINS • MONOCOQUE/FRAME CONSTRUCTION COMPLEX • ABLATOR FOR ASCENT & ENTRY • SOFI INSIDE/OUTSIDE TANK • AERO STABILIZED ENTRY/FINS	21,470
BREAK-UP ALLOWED	CYLINDRICAL (NO ENTRY TPS)	TANKAGE NO ENTRY PROTECTION 	• 2 TANKS • MONOCOQUE CONSTRUCTION • SOFI OUTSIDE TANK • ABLATOR FOR ASCENT • TUMBLING ENTRY	14,705

• COMPATIBLE WITH PARALLEL REUSABLE BOOSTER
• PROPELLANT WEIGHT 113K LB

Figure 3-11. EHT Tank Configuration Options

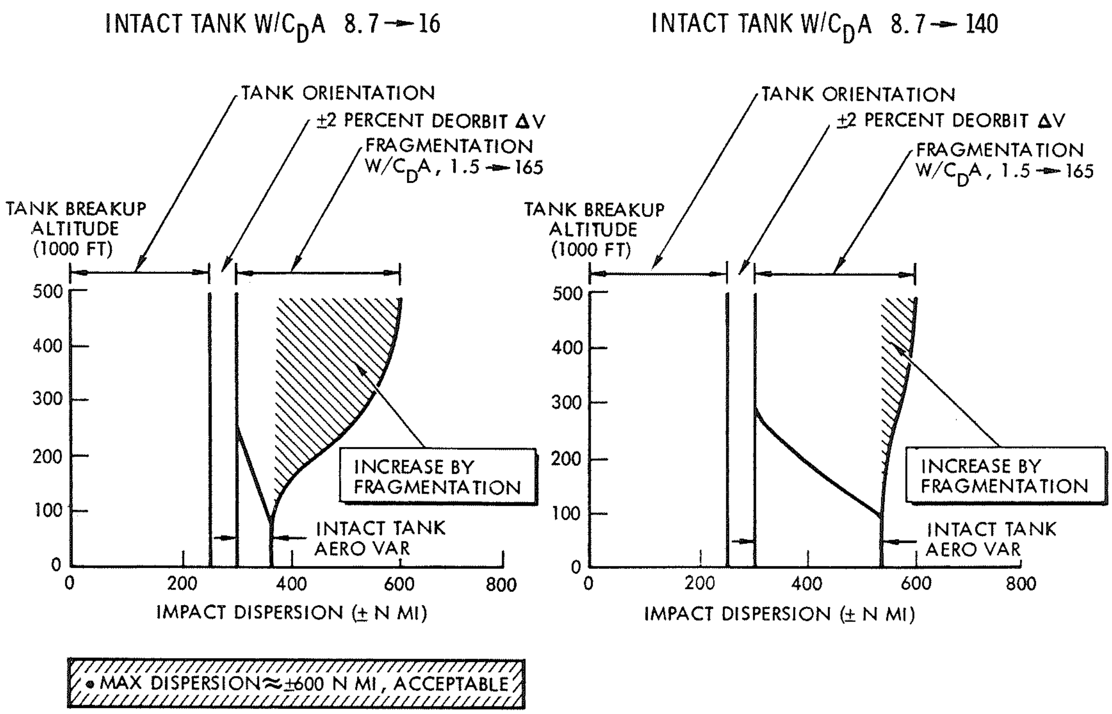


Figure 3-12. Tank Impact Dispersion

The tank selection process determined the EHT tank to be the cylindrical tank with no entry thermal protection system (for the reasons as shown in Figure 3-11). The EHT tank assembly is shown in its complete form on Figure 3-13.

Investigations of the EOHT tank concept encompassed an evaluation of the various tank configuration options (Figure 3-14). All of the tank configurations shown, except 7 and 9, fulfill the basic launch stack requirement that no balloon (pressure-stabilized) tank will be utilized if the tank affects the free-standing stack capability. In addition, prelaunch ground operations are not to be compromised by sequential propellant loading or sequential individual tank pressurization.

The 12 tank configuration options (Figure 3-14) were conceived by varying the number of tank assemblies, the location of the propellants within the tank(s), the location of the tank assemblies, and therefore the relationship of the orbiter to the booster. These variations were made in an attempt to locate the heavy masses (the LO_2 propellant which is approximately 66 percent of the total orbiter mass and the LO_2 and hydrogen propellant plus tank which is approximately 81 percent of the total orbiter mass) in the most direct thrust path with the booster, without penalizing either the orbiter or the booster.

The first tank option (No. 1, Figure 3-14) utilizes the LH_2 tank as the main structure. The tank is supported in tandem by the booster nose and parallel to the orbiter. The LH_2 is, therefore, part of the launch stack and because of the imposed loads, is of semi-monocoque construction. The LO_2 tank is monocoque. In option 2 the LH_2 is installed within a semi-monocoque outer shell (which is the main structure) in an attempt to utilize a free floating monocoque LH_2 tank. Option 3 utilizes two tanks alongside the orbiter with each tank containing half of the LO_2 and LH_2 propellants. The orbiter is mounted in parallel to the booster. The booster thrust load is transmitted through the orbiter to the aft end and directly into the LO_2 tank. Both tanks (LO_2 and LH_2) are monocoque structure. Option 11 is a variation of option 3 in that the LO_2 tanks are forward of the LH_2 tanks; the net result is that the LH_2 tanks are penalized for the boost reaction of the heavy LO_2 mass and must be either heavy-wall monocoque structure, semi-monocoque, or waffle construction.

Special options were considered in No. 4 in that the individual LO_2 and LH_2 tanks are end-supported below the orbiter to allow monocoque tanks for both propellants, but the booster load is transmitted to the tandem orbiter by means of long cantilevered arms to structure just forward of the engine compartment. The resultant load path penalizes both the booster and the orbiter. Option 5 applies the tandem booster thrust load directly into the

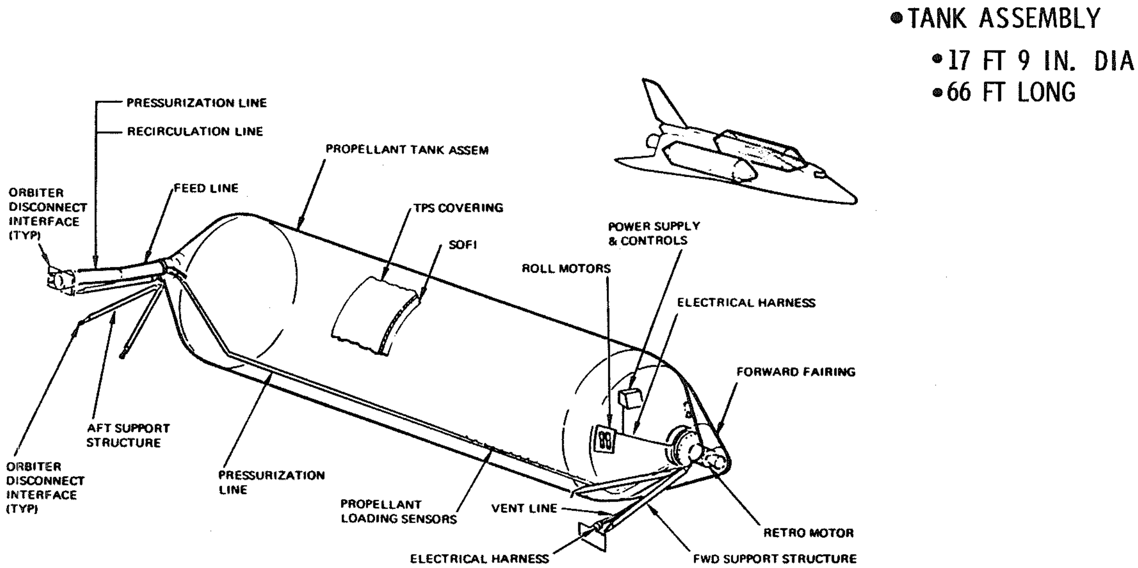


Figure 3-13. EHT Tank Assembly

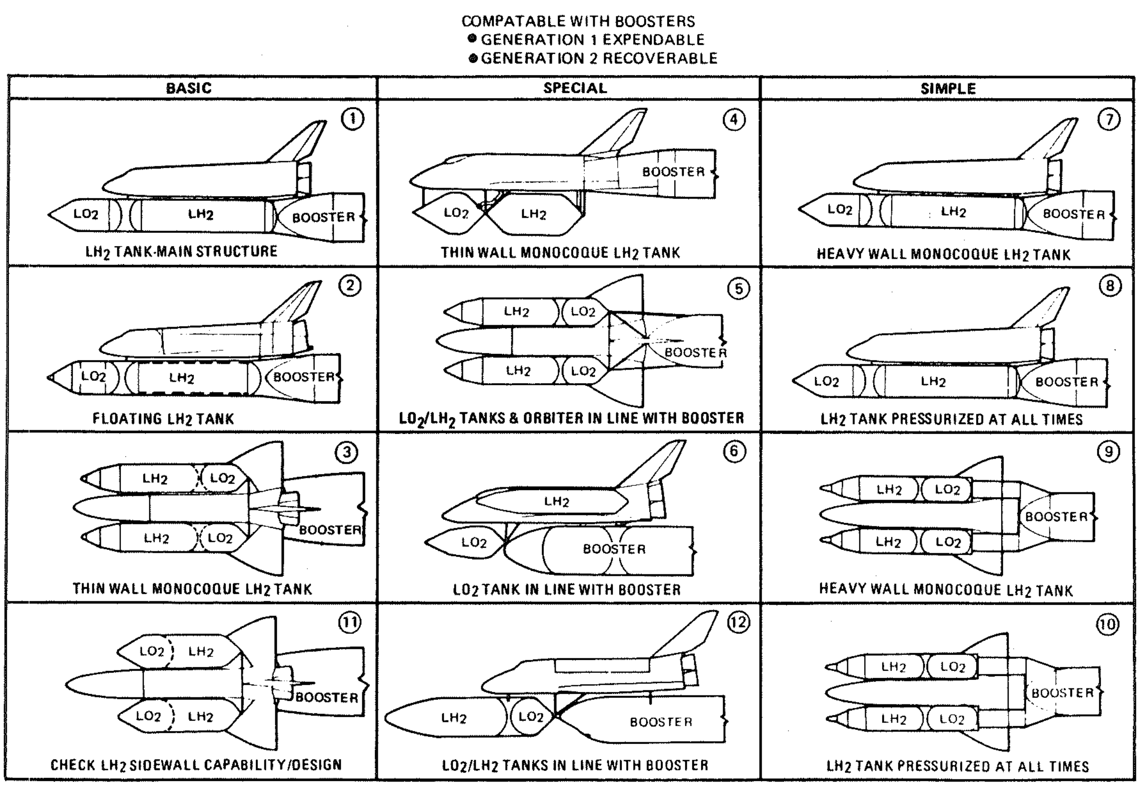


Figure 3-14. EOHT Tank Configuration Options

aft end of the orbiter and the side-mounted tanks to minimize the thrust load path penalties. The resulting thrust structure is complicated, indeterminate, heavy, and dependent on complex mechanisms. Option 6 utilizes two hydrogen tanks mounted alongside the orbiter with the single LO_2 tank below the orbiter and in line with the booster. The booster thrust is directly into the tandem LO_2 tank and through the parallel orbiter to the adjacent LH_2 tanks. A more reasonable option is No. 12, with the LO_2 and LH_2 tanks mounted above (tandem) to the booster, and the orbiter parallel to both the tank and the booster. This configuration utilizes monocoque propellant tanks; the booster load path is directly to the aft end of the LO_2 and the LH_2 propellants but the orbiter is the structural attachment across the tank assembly and the booster for all lateral loads.

Option 7 is similar to option 1, except that the LH_2 tank has been structurally over-simplified in that monocoque heavy wall structure is utilized for the prelaunch loads. The increased loads in the max $q\alpha$ regime are offset in all the options by the use of the pressure in the tanks required for propellant system operation. Option 8 utilizes pressure in the LH_2 tank during prelaunch operations and was studied to determine the weight penalties incurred for the ground rule prohibiting balloon tank or sequential loading or tank pressurization. Options 9 and 10 are for split propellant tanks supported in tandem by the booster (with the orbiter slung between the tanks) with the same design options as 7 and 8, respectively.

The weights of the various tank configuration options are compared in Figure 3-15. Although option 7 has the heaviest tank assembly, it was selected for continued study because the simplified construction potentially could result in the lowest initial program cost. Option 3 was attractive for the reduced tank weight (and simplified construction) but further effort could not appreciably reduce the orbiter (and booster) weight penalty and this option was eventually dropped. Option 12 was also selected for further study because the light monocoque tanks offered the promise of the lowest total program cost even though the increased structural requirement in the orbiter would increase the weight and cost of the orbiter.

The costs of the selected tank options for the external LH_2 and LO_2 propellant and for the external LH_2 propellant are compared in Figure 3-16. The production cost per set of tanks is compatible with the weight of the tanks; the total program cost of option 3 is excessive when compared to the total program costs of tank options 7 and 12, and therefore option 3 was eliminated.

TANK LOCATION	TANK CONFIGURATION	WEIGHTS (LB)			
		BOOSTER INTERSTAGE	TANK ASSEM	Δ ORBITER	Δ BOOSTER
SINGLE (BELOW ORBITER)	1	9750	39,125	0	0
	2	9750	47,325	0	0
	4	16,175	25,195	29,300	6500
	7 ✓	9750	56,415	0	0
	8	9750	47,075	0	0
DUAL (SIDE)	11	8100	31,035	22,178	8000
	3 ✓	8100	28,475	22,178	10,000
	9	29,000	36,590	2978	8000
	10	29,000	36,425	2978	8000
HYBRID	6	7500	28,035	13,146	-4500
	12 ✓	6910	28,340	8346	-2500

Figure 3-15. Weight Comparison, EOHT Tank Design

- WEIGHTS SHOWN DO NOT INCLUDE RESIZING
- COSTS SHOWN INCLUDE WEIGHT RESIZING

TANK			LEARNING CURVE (WT)	SHIP SET			COMMENT
PROPELLANT	CONFIGURATION	TOTAL QTY		WEIGHT (LB)	AVERAGE HOURS/FAB	PRODUCTION COST	
LH_2/LO_2	7 (LO2, LH2, BOOSTER)	457	84%	56,415	61,000	$1,430,000	THICK WALL MONOCOQUE
	3 (LH2, LO2, BOOSTER)	914	82%	28,475	54,000	$1,230,000	LARGER QTY REDUCES COST
	12 (LH2, LO2, BOOSTER)	457	84%	28,340	64,000	$1,350,000	THIN WALL MONOCOQUE
LH_2	(LH2, BOOSTER)	890	82%	20,915	33,000	$ 753,000	FOR COMPARISON PURPOSES

TANK CONFIGURATION	ORBITER		BOOSTER + INTERSTAGE		TANK		TOTAL PROGRAM
	WEIGHT (LB)	ΔCOST $M	WEIGHT (LB)	ΔCOST $M	WEIGHT (LB)	ΔCOST $M	
7	0	0	+9,750	0	56,415	0	0
3	+22,178	+122	+18,100	-3	28,475	-53	+66
12	+8,346	+46	+4,410	-124	28,340	-20	-98
PEAK FUNDING	1976		1980		COSTS SPREAD OVER OPERATIONS PERIOD		

Figure 3-16. Cost Comparison, Selected Tank Configuration Options

3.6.4 Integrated Vehicle Sizing

In order to compare the total system impact of EHT and EOHT orbiter designs, integrated vehicle sizes were established for each orbiter concept. The approach used in vehicle sizing was to establish the Generation 2 system (reusable heat sink booster and orbiter) size required to meet the mission requirements defined in Section 3.2. Once the Generation 2 system is sized, the orbiter (entry vehicle and external tanks) is used with an interim booster as a Generation 1 system. The interim booster system selected for comparison is a cluster of 120-inch-diameter solid rocket motors. As shown in the configuration study matrix (Section 3.1), this is the only interim booster in the study matrix which is common to both the EHT and EOHT orbiters.

The critical mission (Generation 2) for vehicle sizing is the 40,000 pound payload requirement to a polar orbit. Ground rules used in the sizing analysis were:

1. Initial T/W of 1.3
2. Common power head main engine assembly in booster and orbiter
3. Once-around abort capability with one orbiter engine inoperative at nominal staging
4. Nominal Generation 2 staging velocity limited to heat sink booster capability ($\approx$7800 fps)
5. Three-engine orbiter

The results of the vehicle sizing analysis for the integrated vehicle with EHT orbiter are shown in Figure 3-17 and the selected configuration is described further in Figure 3-18. Similar data for the integrated vehicle with EOHT orbiter are shown in Figures 3-19 and 3-20. The selected configuration for each concept is a compromise between minimizing system GLOW, total booster and orbiter dry weight (correlates to total cost), minimum main engine thrust, lower orbiter entry vehicle weight (correlates to reduced peak annual funding), and acceptable staging velocity.

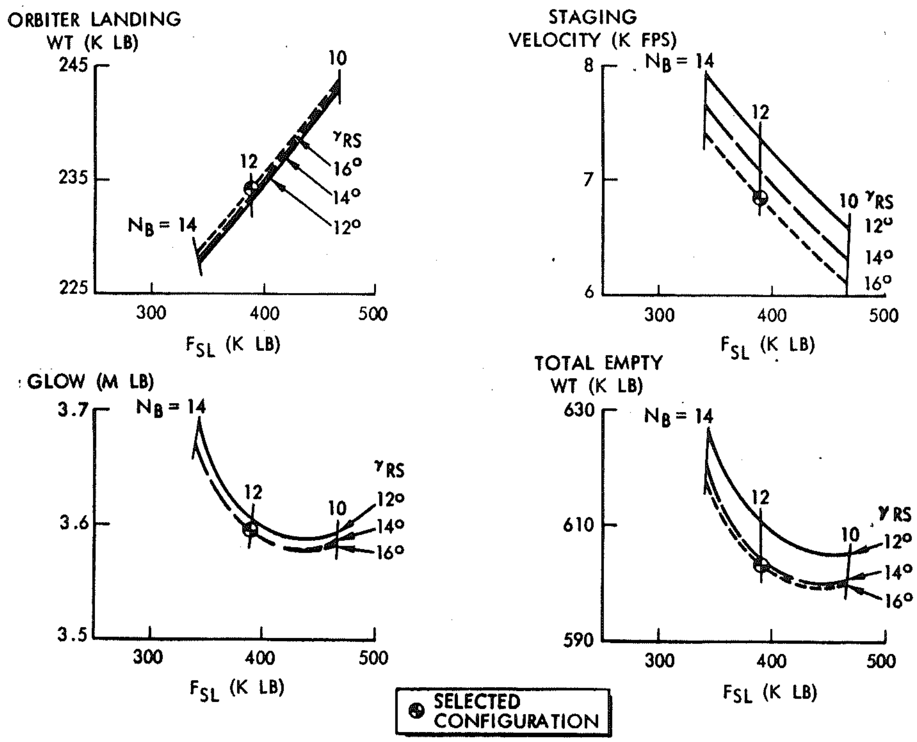
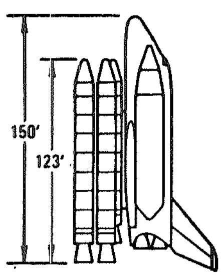


Figure 3-17. Reusable Booster/EHT Orbiter Sizing (15 x 60 Cargo Bay)



SYSTEM PARAMETERS		GEN 1		GEN 2	
CARGO BAY		15 FT X 60 FT		15 FT X 60 FT	
PAYLOAD	(KLB)	65K DUE E		40K POLAR	
OMS ΔV	(FPS)	900		650	
		BOOSTER	ORBITER	BOOSTER	ORBITER
SYSTEM GLOW	(MLB)	5.942		3.593	
T/W AT LO		1.35	1.16	1.30	1.20
STAGE GROSS WT	(MLB)	4.805	1.155	2.471	1.123
MAIN PROP. WT	(MLB)	3.000	0.832	2.016	0.832
OMS PROP. WT	(KLB)		22.9		16.5
FLYBACK FUEL	(KLB)			37.8	
FLYBACK RANGE	(N MI)			182.0	
*STAGE DRY WT	(KLB)	587.9	191.2	392.5	191.2
**STAGE LANDING WT	(KLB)		234.1	399.3	234.1
ORBITER EXT TANK WT	(KLB)		19.7		19.7
MAX $\bar{q}$	(PSF)	595		491	
STAGING γ_R	(FPS)	7,932		6,838	
	(DEG)	7		16	
MAIN ENG FSL/ENG	(KLB)	1,600		389.3	
FVAC/ENG	(KLB)	1,820	447.5		447.5
NO. OF MAIN ENGS		5/2	3	12	3

*PAYLOAD & EXPENDABLE TANKS NOT INCLUDED

**INCLUDES PAYLOAD

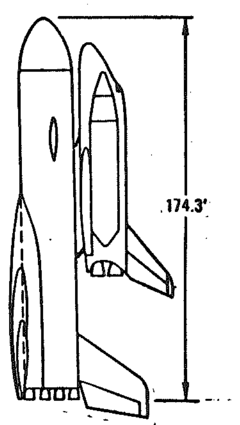


Figure 3-18. Integrated EHT System Description With 120-Inch SRM Interim Booster

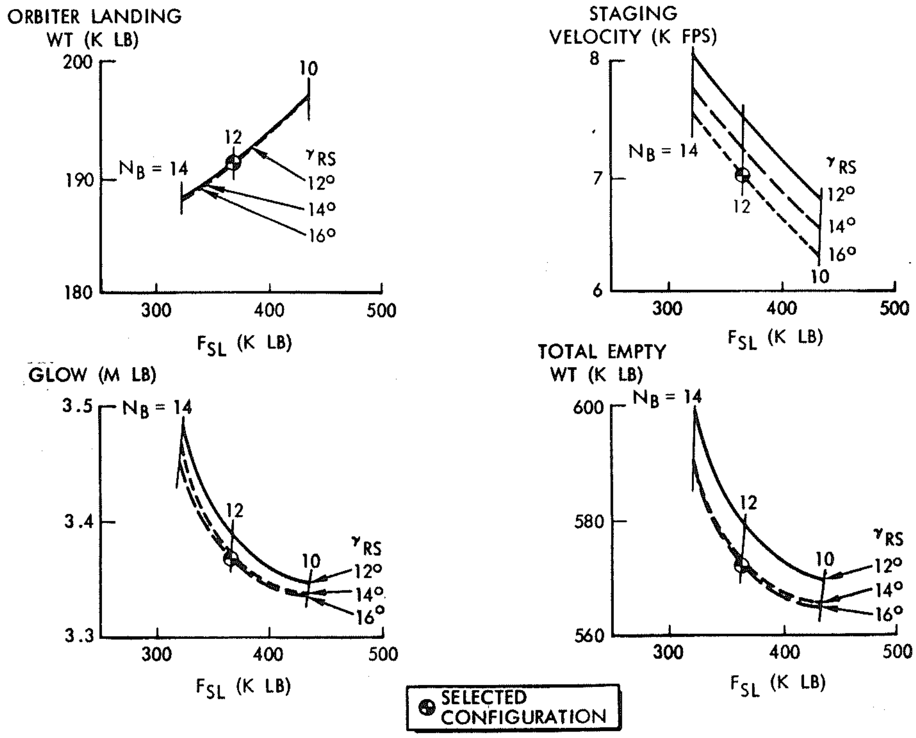
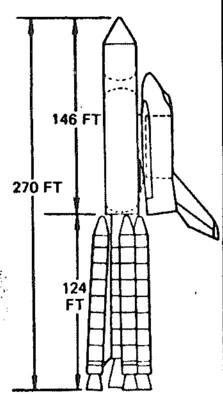
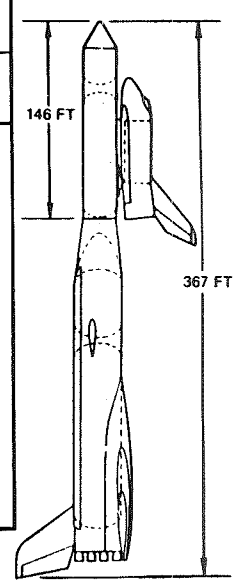


Figure 3-19. Reusable Booster/EOHT Orbiter Sizing (15 x 60 Cargo Bay)



SYSTEM PARAMETERS	GEN 1			GEN 2	
CARGO BAY PAYLOAD (KLB) OMS ΔV (FPS)	15 FT X 60 FT 74 DUE E 900			15 FT X 60 FT 40 POLAR 650	
	BOOSTER		ORBITER	BOOSTER	ORBITER
	1ST	2ND			
	5X1207	2X1207			
SYSTEM GLOW (MLB)	5.855			3.374	
T/W AT LO	1.37			1.3	
STAGE GROSS WT (MLB)	3.404	1.383	1.068	2.35	1.024
MAIN PROP. WT (MLB)	3.0	1.2	0.754	1.911	0.754
OMS PROP. WT (KLB)			22.6K		12.9
FLYBACK FUEL (KLB)				39	
FLYBACK RANGE (N MI)				195	
*STAGE DRY WT (KLB)	405	183	148	376	148
**STAGE LANDING WT (KLB)			177	421	192
ORBITER EXT TANK WT (KLB)			48		48
MAX $\bar{q}$ (PSF)	639			488	
STAGING V_R (FPS)	3663	8567		7029	
γ_R	33	6.3		16	
MAIN ENG F_{SL} (KLB)	8000			366	
F_{VAC} (KLB)		3800	420	401	420
NO. OF MAIN ENG	5	2	3	12	3



*PAYLOAD & EXPENDABLE TANKS NOT INCLUDED

**INCLUDES PAYLOAD

Figure 3-20. Integrated EOHT System Description With 120-Inch SRM Interim Booster

3.6.5 Abort

The ascent phase abort capabilities of EHT and EOHT orbiter configurations, which have three main engines and are sized to have "once around" abort capability in the event of an engine failure at staging, are essentially equal. They are also comparable to those of a fully reusable configuration sized to meet the same requirements.

The mission ascent phase is divided into three abort regimes. The first extends from liftoff to approximately 15 seconds into the flight. During this period, due to the low-altitude and velocity conditions and the time delay to orbiter engine full thrust, safe orbiter separation and flyaway are not possible. Adequate personnel safety and vehicle recovery are therefore dependent on booster reliability during this period.

After approximately 15 seconds and until normal staging conditions are reached, the orbiter is capable of flying back to the launch site after separation from the disabled booster. This abort flight mode requires use of the orbiter's main propulsion system to propel the orbiter to a position and direction from which it can glide to a normal approach and landing at the launch site. Aerodynamic forces are used during portions of the flight to assist in turning and maintaining altitude. The tank is separated after propellant depletion. The flight profile is planned to assure that the tanks will impact on water regardless of launch azimuth (KSC or WTR launches).

At staging, the orbiters (both EHT and EOHT) are designed to have "once around" abort capability in the event of an engine malfunction. Emergency power level (EPL) on the remaining two engines and burning of the OMS propellant are used to produce the ΔV required for insertion into an orbit that permits return to the launch site in one revolution.

The three abort regimes are illustrated in Figure 3-21. As indicated in the figure, after approximately 300 seconds into the flight, it is possible to abort to a 100-nm circular orbit or to the cross-around orbit.

3.6.6 Fracture Mechanics

The continued emphasis on program cost reduction has dictated a cryogenic tankage structural design of external expendable tanks of 2219 aluminum monocoque (Paragraph 4.4.4.1). This design minimizes the tankage fracture mechanics/control problem. In addition to eliminating the reuse requirement (no in-service inspection and greatly reduced cycles of loading), the roll-formed monocoque construction is less susceptible to fabrication-induced flaws and, overall, undergoes simpler stress fields than do the complex, stiffened constructions studied previously.

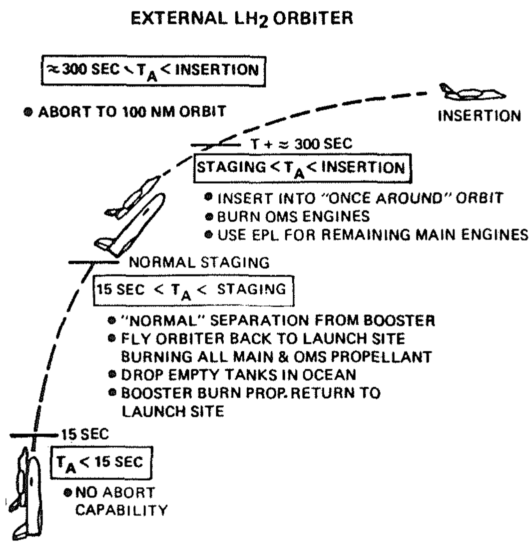


Figure 3-21. Abort Capabilities, EOHT and EHT

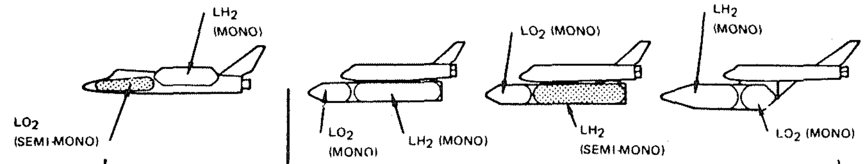
Figure 3-22 illustrates the nature of the EOHT fracture control review performed in support of the vehicle configuration studies. From a fracture control point of view, the selected baseline design is the most suitable configuration, primarily because of the low tensile stress levels in the LH_2 tank cylinder wall (compression stability is critical).

3.6.7 Test Impact

A comparison of the EHT and EOHT acceptance, structural, and separation qualification test efforts is presented in Table 3-4. The addition of the dual plane separation system in the EOHT interstage adapter will require additional in-process acceptance test effort, which will be offset by deletion of the rear rotating linkage orbiter/booster separation system associated with the EHT configuration.

The structural qualification test program for EOHT will require additional effort due to the increased size of the external tank—from 17-foot diameter and 72-foot length to 22-foot diameter and 167-foot length.

The EOHT separation qualification test effort will require increased test effort (compared with the EHT) as a result of the addition of the LO_2 fill, recirculation, and vent line disconnects as well as the addition of the



	EHT	EOHT		
LH_2 TANK				
• TENSION STRESS INTENSITY	HIGH	LOW	HIGH	HIGH
• STRESS FIELD COMPLEXITY	LOW	COMPRESSION	HIGH	LOW
• PEAK PRESSURE CYCLES	2	STABILITY REQMT	2	2
• THERMAL CYCLES	1	CRITICAL (CYL	1	1
• FAILURE MODE	LEAKAGE	WALL GAGE IS 4	RUPTURE	RUPTURE
• PROOF TEST COMPLEXITY	LOWEST	TIMES PRESSURE	HIGH	LOW
• POST PROOF INSPECTABILITY	HIGH	REQUIRED GAGE)	LOW	HIGH
• PROOF TEST CONFIDENCE	HIGHEST		LOW	HIGH
OVERALL RATING	GOOD	BEST	WORST	GOOD
LO_2 TANK				
• TENSION STRESS INTENSITY	HIGH	HIGH	HIGH	HIGH
• STRESS FIELD COMPLEXITY	HIGH	LOW	LOW	LOW
• PEAK PRESSURE CYCLES	200	2	2	2
• THERMAL CYCLES	100	1	1	1
• FAILURE MODE	RUPTURE	RUPTURE	RUPTURE	RUPTURE
• PROOF TEST COMPLEXITY	HIGH	LOW	LOW	LOW
• POST PROOF INSPECTABILITY	LOW	HIGH	HIGH	HIGH
• PROOF TEST CONFIDENCE	LOW	HIGH	HIGH	HIGH
• IN SERVICE INSPECTABILITY	POOR	NOT REQD	NOT REQD	NOT REQD
OVERALL RATING	WORST	BEST	BEST	BEST

Figure 3-22. Fracture Mechanics, EHT Versus EOHT

dual plane interstage adapter pyrotechnic separation system. Again, the rear rotating link orbiter/booster separation system is deleted in the EOHT system.

Structural testing of the orbiter airframe will be significantly reduced as a result of the external EOHT tank configuration due to the reduction in the overall size of the orbiter and the reduction in the test complexity from deletion of the internal LO_2 and LH_2 tanks. In addition, scheduling time will be gained in the Phase B' proposed program due to the opportunity for concurrent testing of the orbiter and tank as separate test articles.

3.6.8 Facility Impact

Facility requirements for tank acceptance tests involving hydrostatic tests of the LO_2 and LH_2 tank bulkheads and the LO_2 tank and pneumostatic test of the LH_2 tank will be essentially the same for the EOHT and EHT tanks.

The structural qualification test program for the EOHT tank will require a larger and stronger structural test tower due to its increased size and loading complexity.

Table 3-4. Comparison of Total Test Efforts

Test Requirement	EHT	Orbiter	EOHT	Orbiter
Acceptance Tests				
LO_2 tank	—	Yes	Equivalent	—
LH_2 tank	Yes	—	Equivalent	—
Separation mechanism	Yes	Yes	Equivalent	Equivalent
Power system	Yes	—	Equivalent	—
Deorbit SRM	Yes	—	Equivalent	—
Spin SRM	Yes	—	Deleted	—
Interstage adapter	—	—	Additional	—
Fluid management	Yes	Yes	Equivalent	Equivalent
Structural Qual Tests				
LO_2 tank	—	Yes	Equivalent	—
LH_2 tank	Yes	—	Increased	—
Aft skirt	—	—	Additional	—
Mid skirt	—	—	Additional	—
Orbiter structure	—	Yes	—	Reduced
Separation Qual Tests				
Tank/orbiter	Yes	Yes	Yes	Yes
Orbiter/booster	—	Yes	—	Deleted
Tank/booster	—	—	Additional	—
Disconnect fittings	Yes	Yes	Additional	Additional

Separation test and facility requirements are substantially different for the Phase B' EOHT tank due to the conceptual differences in the separation method. Adequate functional demonstration of the tank/orbiter disconnects and separation linkages and mechanisms may require a zero-g simulation facility with a minimum of three degrees of freedom. Since similar requirements exist for the docking and cargo-handling systems, it is proposed that a single air bearing facility be used to demonstrate all three subsystems.

The air bearing level flow would be approximately 100 feet long and 125 feet wide to accommodate anticipated test programs for the three subsystems.

In addition, a full-scale ordnance test facility will be required to adequately demonstrate the functional firing of the dual plane interstage adapter separation system.

Table 3-5 is a comparison of the facility requirements for EHT and EOHT testing.

3.6.9 Technical Risk

A comparison of the technical risk for the EHT and EOHT designs was made to identify any significant differences. Both external tank concepts were rated relative to the reusable orbiter design developed in the Phase B study. The results of this comparison are summarized in Figure 3-23.

A major technical risk concern is the impact of weight growth or I_{sp} degradation as the shuttle vehicle design evolves. The EOHT orbiter concept offers minimum risk, because the external tank design is essentially separated from the orbiter entry vehicle design. Potential loss in performance capability because of adverse weight and I_{sp} changes can be prevented by increasing the external tank and booster size, whereas the EHT orbiter design would require an increase in the internal LO_2 tank volume and modification of the orbiter entry vehicle mold lines.

The confidence in the current aerodynamic/aerothermal predictions are not as high as for the reusable system developed in the Phase B study. However, by the start of Phase C/D, equal data will be available, and the prediction can be made with equal confidence.

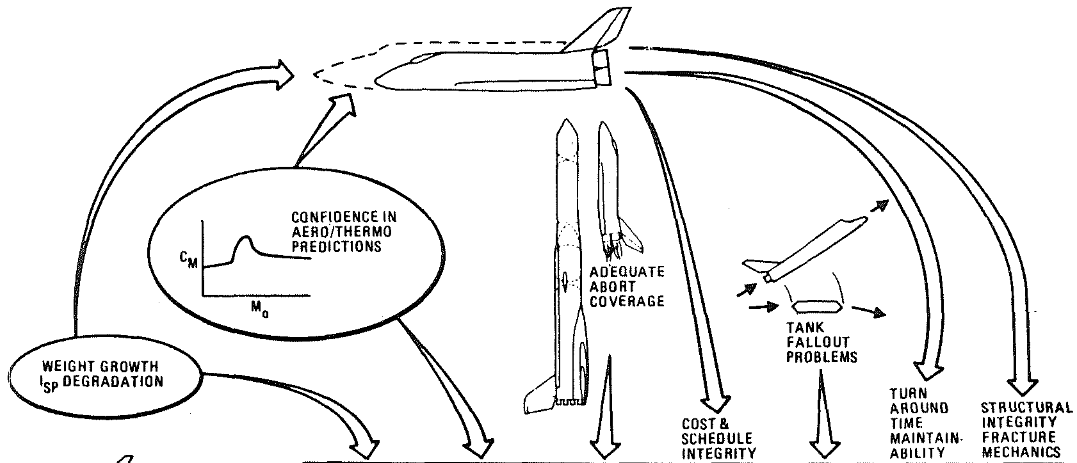
The EOHT orbiter design offers the least turnaround time because of the simpler design (no ascent propellant tanks). However, either concept is adequate.

The EOHT orbiter design caused the least concern over structural integrity and fracture mechanics. This is attributed primarily to the use of single-mission tankage instead of reusable tankage. Other important factors are the simplicity of the external tank structure and separation of body structure design from the tank structure design.

For the reasons stated, confidence in achieving the predicted shuttle system cost and development schedule is considered highest for the shuttle system with the EOHT orbiter design.

Table 3-5. Comparison of Facility Requirements

Test Requirement	EHT	EOHT
Acceptance Tests		
Bulkhead/tank hydrostat	Hydrostat facility	Hydrostat facility
Pneumostat	Open-field revetment	Open-field revetment
Structural Qual Tests	Structural test tower—90 feet high and 30 feet in diameter with 500,000-pound strongback column capability. LN_2 Dewar and pumping system with 75,000 gallon capacity	Structural test tower—200 feet high and 45 feet in diameter with 5-million pound strongback and bearing pad capability; LN_2 Dewar and pumping system with 150,000-gallon capacity
Separation Tests		
Tank/orbiter	Structural test tower with overhead cable suspension system	Air bearing floor, 100 feet long and 125 feet wide, for adequate zero-g simulation functional separation
Booster/orbiter	Static tests of components and computer analysis	Not applicable
Tank/booster	Not applicable	Full-scale ordnance test facility to accommodate 35-foot high by 33-foot-diameter specimen with applied airloads and inertia loads, rated at 0.50 pound TNT or equivalent.



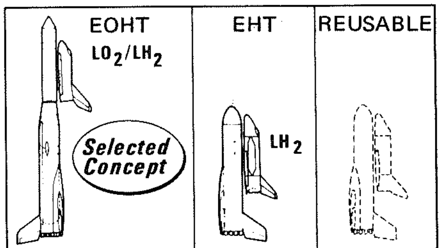
	WEIGHT GROWTH I_{SP} DEGRADATION	CONFIDENCE IN AERO/THERMO PREDICTIONS	ADEQUATE ABORT COVERAGE	COST & SCHEDULE INTEGRITY	TANK FALLOUT PROBLEMS	TURN AROUND TIME MAINTAIN-ABILITY	STRUCTURAL INTEGRITY FRACTURE MECHANICS
EOHT	LARGER EXTERNAL TANKS	BASED ON REUSABLE WT DATA	EJECT TANK FOR DOWN RANGE LANDING	SIMPLEST TANK DEVELOPMENT	FEASIBLE DEORBIT TIMING	BEST	LEAST PROBLEM
EHT	PROVIDE OVERSIZE LOX TANK OR MOD ORBITER	BASED ON REUSABLE WT DATA	BURN PROPELLANT LAND AT LAUNCH SITE	HYD TANK SIMPLE DEVELOPMENT	FEASIBLE DEORBIT TIMING	ADEQUATE	MODERATE PROBLEM
REUSABLE	LARGER ORBITER	MOST WT DATA NOW	BURN PROPELLANT LAND AT LAUNCH SITE	MOST COMPLEX TANKAGE DEVELOPMENT	NO PROBLEM	ADEQUATE	MOST PROBLEM
ALL THREE	LARGER BOOSTER	PHASE C EQUAL DATA	AFTER STAGING, ONCE AROUND	COMMON FLIGHT TEST PROGRAM	NOT APPLICABLE	PROVISIONS FOR ACCESS TO SUB-SYSTEMS	SAME EXCEPT FOR TANKS

Figure 3-23. Comparison of Orbiter Configuration on Basis of Technical Risk

3.6.10 Recommendation

The preferred system has a single LO_2/LH_2 tank mounted on the underside of the orbiter. This system results in a smaller, lower weight orbiter than the fully reusable orbiter with external LH_2 tanks. Use of the expendable LO_2/LH_2 tank reduces sensitivity of the vehicle to weight growth and minimizes program schedule risk. Many alternate propellant tank arrangements and construction techniques were examined. In the selected design, the oxygen tank is located forward and is connected with nonpressurized interstage structure to the liquid hydrogen tank aft. This external tank is used to interface with the boost vehicle. The simple monocoque tank was selected, despite its higher weight relative to the skin-stringer-frame semimonocoque approach, because of lower production costs.

The costs and other selection aspects associated with the external tank tradeoff are shown in Figure 3-24. This figure shows that the EOHT orbiter design yields the lowest GLOW for the Generation 2 system. Further, its total program cost is slightly lower than that for the configuration with an external hydrogen tank. It can be seen from the summary information in Figure 3-25 that the annual funding peak is under \$1.25 billion. This is for a program which includes an EOHT orbiter launched in 1978 by an interim booster. Ultimately, the reusable booster will be phased into the program to support a first manned orbiter flight date of 1982.



	EOHT LO_2/LH_2	EHT	REUSABLE
GLOW - 1000 lb	3374	3595	4479
Orb Dry W-1000 lb	148	191	293
Cost Total Prog - $B Peak Annual - $B	 10.71 1.22(77)	 10.74 1.23(77)	 11.31 1.98(76)
Development Risk	Least Wt/ & Sensitivity Min Fract Mech	Medium	Highest

Figure 3-24. EOHT Versus EHT

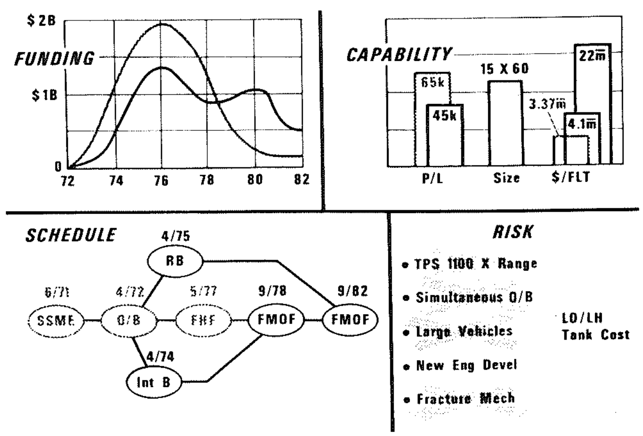


Figure 3-25. External Tank Summary

3.7 PAYLOAD BAY SIZE

Another orbiter configuration issue studied during the Phase B extension was whether the use of a 40-foot-long payload bay throughout the program or the use of a 40-foot-long payload bay in Generation 1 and a 60-foot-long payload bay in Generation 2 provided any significant cost benefit. In making this study, the following approach was used:

1. Orbiter configurations were developed with:

    a. 60-foot-long payload bay

    b. 40-foot-long payload bay

    c. 40-foot-long payload bay in Generation 1 and modified design with 60-foot-long payload bay in Generation 2

2. An integrated vehicle was developed for each concept

3. The significant configuration differences were evaluated

4. Annual expenditure and total program cost were developed for each concept

The results of this study and the recommended approach are discussed in the following paragraphs.

3.7.1 Orbiter Comparison (40- and 60-foot Payload Bay)

The EOHT orbiter concept was used in the study. Orbiter configurations were established for each payload bay length to support orbiter weight analysis, integrated vehicle sizing, system evaluation, and system cost analysis. A comparison of the orbiter configurations developed is shown in Figure 3-26. The reduction in orbiter length for the shorter payload bay is the same as the reduction in payload bay length. The reduced orbiter length results in an orbiter dry weight 8000 pounds less than that for the orbiter with a 60-foot-payload bay.

The information shown for the 40-foot-payload-bay orbiter represents one optimized for use in Generation 2. Although the 40-foot-payload bay is only a Generation 1 system, the Hi P_C engine thrust will be the same as for the Generation 2 system—420,000 pounds vacuum thrust, precluding dual engine development.

3.7.2 Aerodynamic Characteristics

Figure 3-27 illustrates the effects of changes in the major dimensions of the payload bays on the external configurations of the design study orbiters.

	CARGO BAY SIZE	
	15 FT X 40 FT	15 FT X 60 FT
BODY VOL, FT^3	25,050	31,450
PACKAGING EFFICIENCY	43.5%	46.9%
BODY SURFACE, FT^2	5,017	6,297
WING AREA, EXPOSED, FT^2	2,358	2,499
DRY WEIGHT, K LB (LESS TANKS & PAYLOAD)	140	148
ORBITER EXT TANK WT, K LB	46	48
3 Hi-Pc-F_{VAC}, K LB	404	420

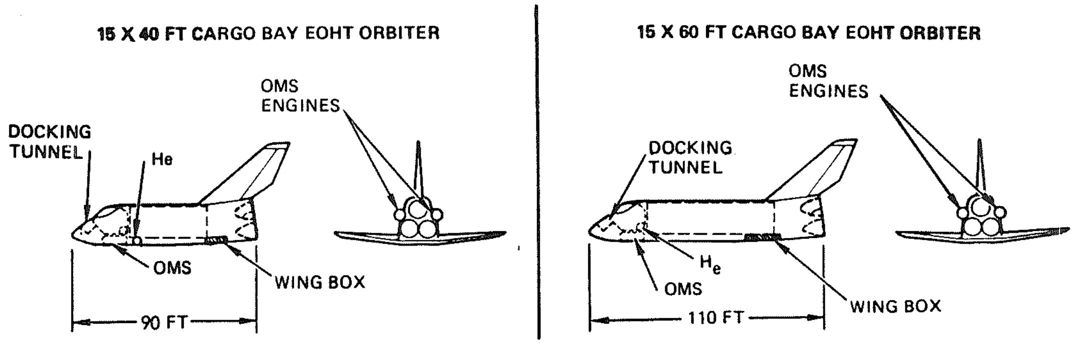
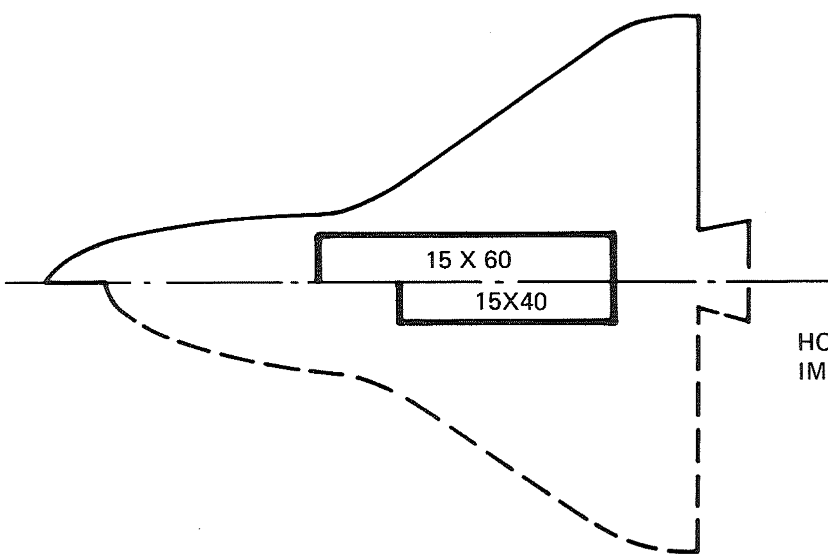


Figure 3-26. EOHT Description, 40- by 60-Foot Cargo Bay Vehicle



HOW DOES CARGO BAY SIZE IMPACT:

- VEHICLE WEIGHT AND SIZE?
- AERODYNAMIC PERFORMANCE AND BALANCE?

Figure 3-27. Cargo Bay Size

Vehicle size, weight, and fineness ratio are generally sensitive to the length dimension of the payload bays—all three parameters have higher values for orbiters with 60-foot payload bays. Payload bay dimensions have much more effect on the weight and size of the external-tankage orbiters than on the configurations of the all-reusable vehicles. Figure 3-28 further illustrates the impact of payload bay dimensional changes on orbiter design parameters—for a fixed diameter, payload bay length variations have a more significant effect on the single-engine EOHT orbiter design dry weight than does payload bay diameter.

The impact of payload bay sizes on hypersonic and on subsonic aerodynamic characteristics is presented in Figures 3-29 and 3-30. Aerodynamic performance parameters (C_L, C_{L_α}, and L/D) are not seriously affected by changing the length of a 15-foot-diameter payload bay from 60 to 40 feet. The slight decrement in aero performance associated with the shorter-bay configuration is caused by the increased hypersonic drag due to decreased body fineness ratio.

The impact of payload bay size changes on aerodynamic balance is more significant. Shortening the payload bay causes a loss in pitch trim capability at forward c.g. positions in the hypersonic flight regime (Figure 3-29). In the subsonic regime, the 60-foot-bay configuration can be neutrally stable or unstable in pitch depending upon the location of the aft c.g. (Figure 3-30). Subsonically, adequate control power is available for forward c.g. positions for either configuration.

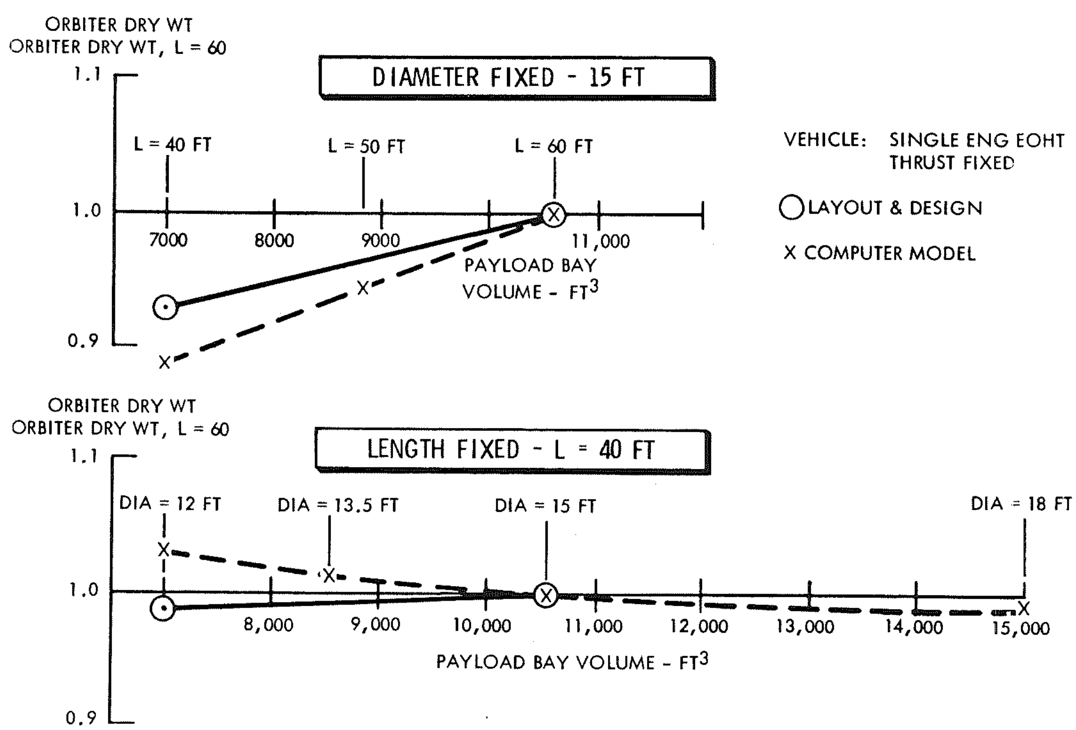
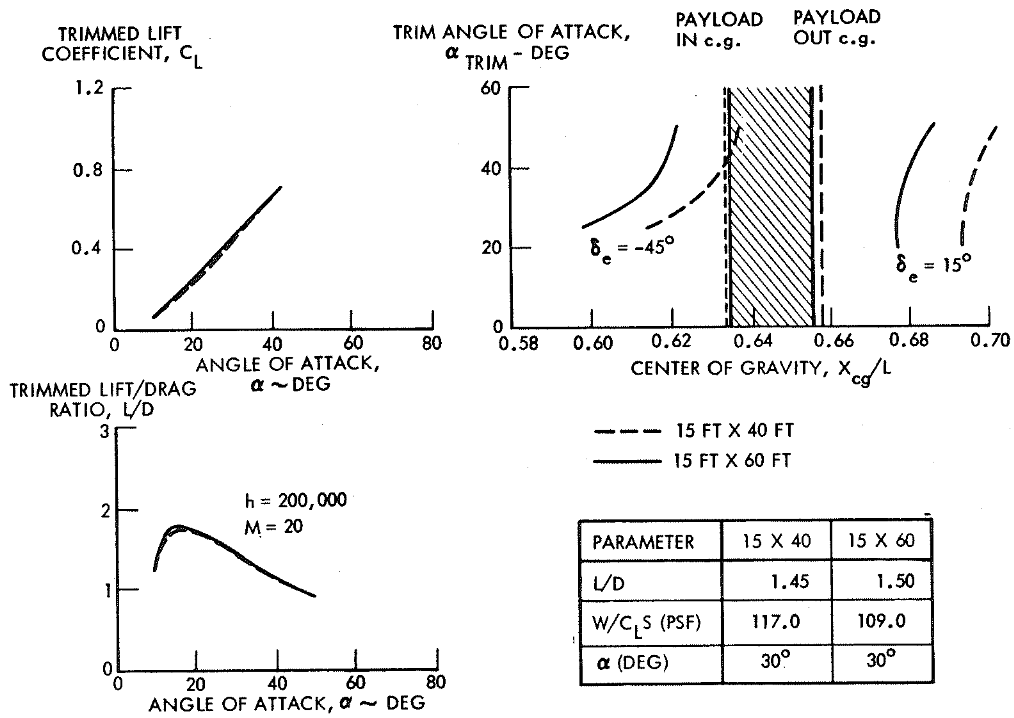
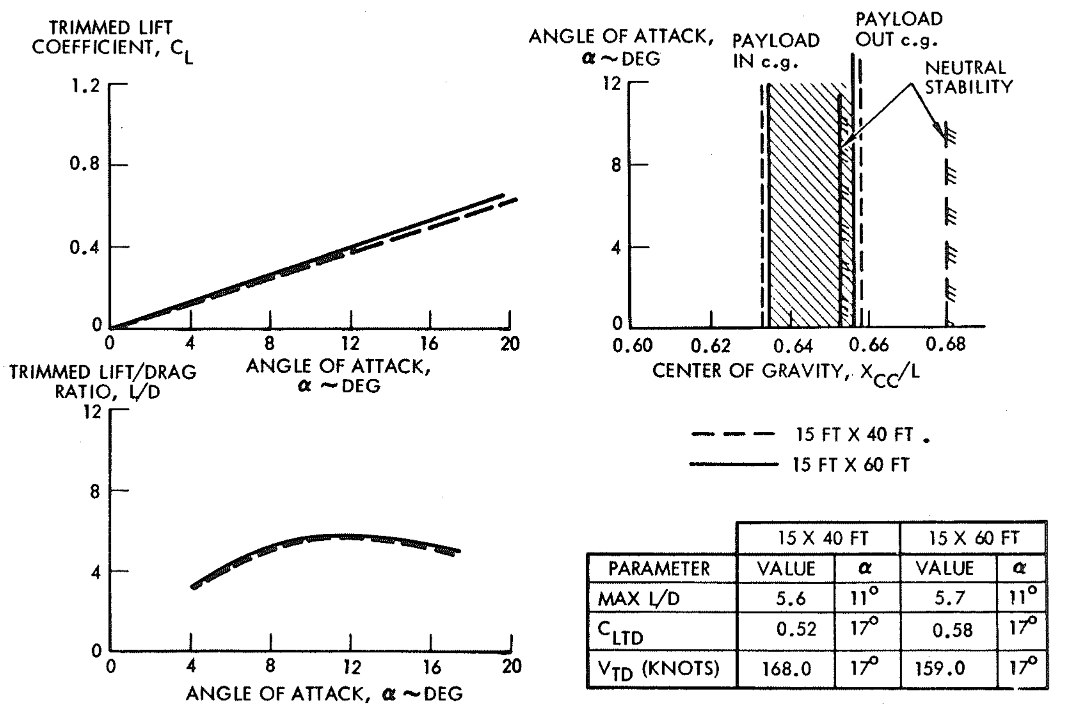


Figure 3-28. Effect of Payload Bay Size Change on Orbiter Dry Weight



PARAMETER	15 X 40	15 X 60
L/D	1.45	1.50
W/C_LS (PSF)	117.0	109.0
α (DEG)	30°	30°

Figure 3-29. Hypersonic Aerodynamic Characteristics Cargo Size Impact



	15 X 40 FT		15 X 60 FT	
PARAMETER	VALUE	α	VALUE	α
MAX L/D	5.6	11°	5.7	11°
C_{LTD}	0.52	17°	0.58	17°
V_{TD} (KNOTS)	168.0	17°	159.0	17°

Figure 3-30. Subsonic Aerodynamic Characteristics Cargo Size Impact

Figure 3-31 presents a comparison of 40-foot- and 60-foot-payload-bay orbiters, with emphasis on configuration design and aerodynamic parameters. Figure 3-32 summarizes the findings of the payload bay size study. Wing-body matching has been identified as a major design consideration for delta wing orbiter configurations.

3.7.3 Growth Considerations

3.7.3.1 Orbiter Provisions

Provisions in the orbiter to accommodate a payload by length increase from 40 to 60 feet must consider aerodynamic stability as well as physical size. This includes extension of the body length for the increase in the payload bay length and the location of the wing for the correct c.g. location relative to centroid of plan form area. The orbiter, designed for payload growth, is shown in Figure 3-33, and the payload bay options that were considered are shown in Figure 3-34. The decision for the selection of a payload bay growth option depends on the phased annual funding peak and the total program cost. The plugged option design has the highest peak annual funding, with a total program cost for either a single orbiter with only a 60-foot payload bay or two separate orbiters with different-length payload bays.

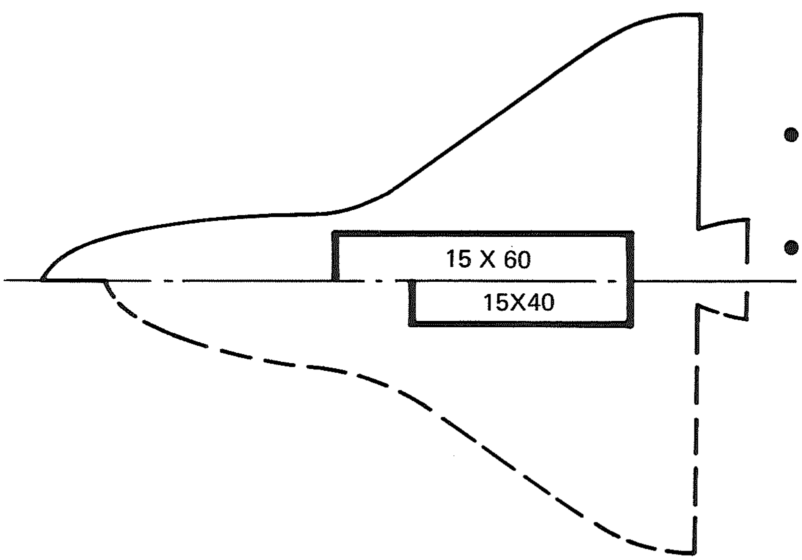
Some possible orbiter growth modifications are shown in Figure 3-35. These include the use of various fuselage body plugs and wing area additions. A single fuselage body plug and a wing root adapter provide the simplest, least-cost hardware approach to the orbiter growth problem—if two orbiter lengths are considered. The option selected for the orbiter design is to use a single orbiter for Generation 1 and 2 that will carry either the 40- or 60-foot payload. As shown on Figure 3-34, this option has an intermediate peak annual funding, but the lowest total program cost.

3.7.3.2 Tank Provisions

A payload length of either 40 or 60 feet also affects the external tank configuration. With respect to the tank configuration, the normal attachment of the 60-foot-payload-bay orbiter to the tank is at each of the bulkheads at both ends of the payload. A program incorporating a growth version (40 to 60 feet) has the following implications. The 40-foot-payload-bay orbiter is smaller and lighter—which in turn allows the use of reduced-size external tanks and a relocated forward tank attachment. However, the selected option is to size the tank for the Generation 2 orbiter (60-foot payload bay), retain the same tank for the Generation 1 orbiter (40-foot-payload bay) and fill up the tanks as required. A relocated frame (baseline and alternate #2 tank options) would be installed in the tank to correspond to the relocated forward

	HI P_c ENGINES		J2S ENGINE
	110B 15x60 46' 82' 110FT	110A 15x40 42.5' 80' 90'	98A 15x40 34.2' 66.7' 82.3'
DESIGN			
ENTRY WEIGHT K LB	192(Landing)	183(Landing)	124(Landing)
DRY WEIGHT K LB	148	140	
BODY VOLUME FT^3	31,450	25,050	
PACKAGING EFFICIENCY %	46.9	43.5	
WING AREA, THEORETICAL FT^2	3,829	3,653	2,727
AERODYNAMICS			
HYPERSONIC L/D MAX	1.8	1.75	1.8
L/D AT $\alpha = 30°$, M = 20	1.5	1.45	1.5
$W/C_L S$ AT $\alpha = 30°$, M = 20	109.0	117.0	91.0
SUBSONIC LIFT CURVE SLOPE, I/DEG	.0344	.0346	.031
SUBSONIC L/D MAX	5.7	5.6	6.6
TOUCHDOWN SPEED KNOTS	159	168	160
AFT C.G./LIMIT % L	65.6	65.8	61.8
FWD C.G./LIMIT % L	63.5	63.3	59.0

Figure 3-31. Comparison of 40- by 60-Foot Cargo Bay Orbiters



FINDINGS

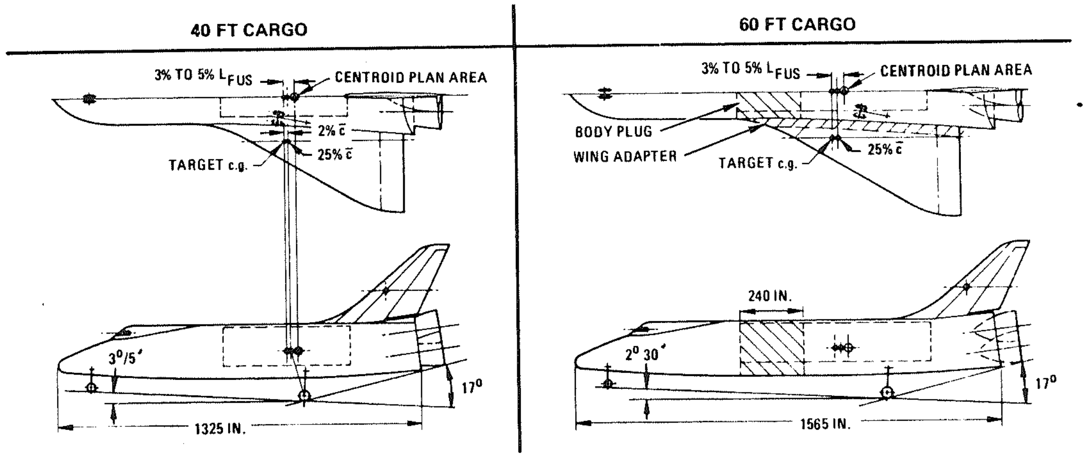
- CARGO BAY DIAMETER MINOR EFFECT ON VEHICLE WEIGHT
- SHORT CARGO BAY ORBITERS

  LARGE C.G. SHIFT

  DIFFICULT TO BALANCE

  REQUIRE FURTHER STUDY OF BODY-WING MATCHING

Figure 3-32. Cargo Bay Size Effects



RETAIN
- CONTROL STABILITY
  - HYPERSONIC
  - SUBSONIC
  - LANDING
  - TAKE OFF
  - CARGO IN
  - CARGO OUT
  - ELEVON ASSIST
- MAJOR COMPONENTS
  - FORWARD BODY
  - MID/AFT BODY
  - WING OUTER PANEL
  - LANDING GEAR
  - VERTICAL SURFACE

REQUIRES

PRE PLANNED
- DESIGN FOR LARGER SIZE/ HEAVIER VEHICLE

BODY PLUG & WING ADAPTER

Figure 3-33. Orbiter Configuration Development (Designed for Growth)

CONFIGURATION PLAN	DES REQUIREMENTS	RELATIVE SHUTTLE PEAK ANNUAL	TOTAL PROGRAM
• SINGLE ORB-GEN 1 & 2 60 FT	• 60 FT CARGO BAY • 1 WING SIZE/LOCATION • 1 SET LOADS • GEN 1 & 2 EQUIP. PKG	INTERMEDIATE	LOWEST
• PLUGGED DESIGN 40 FT 60 FT	• 40 FT & 60 FT CARGO BAY • 2 WING SIZES/LOCATIONS • BODY PLUG • GEN 1 & 2 EQUIP. PKG	HIGHEST	INTERMEDIATE
• GEN 1 40 FT • GEN 2 60 FT (DIFF CONFIG)	• SEPARATE DESIGNS • PHASED RELEASE • MINIMUM SCAR	LOWEST	HIGHEST

DECISION DEPENDS ON IMPORTANCE OF PHASED FUNDING/TOTAL COST

Figure 3-34. Payload Bay Growth Options

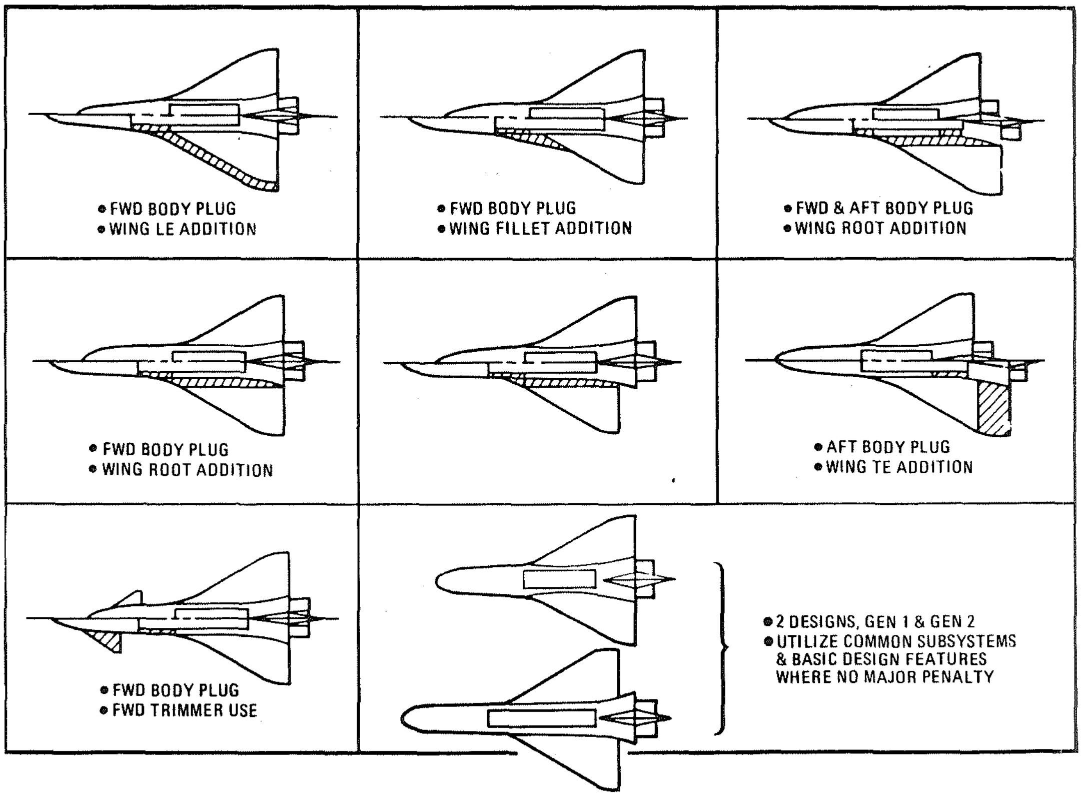


Figure 3-35. Orbiter Growth Modifications Possible Solutions

bulkhead in the orbiter. For the alternate #1 and #3 tank options, a relocated support in the nonrecoverable booster would be used to correspond with the shortened orbiter.

The alternate tank to orbiter compatibility options for the payload growth considerations are shown in Figure 3-36.

The final baseline selection of a single orbiter with the 60-foot payload bay eliminated the need for the relocated frame in the baseline and alternate #2 tank options and the relocated support in the nonrecoverable booster.

3.7.4 System Comparisons and Costs

A program in which the Generation 1 vehicle is composed of a 60-foot-payload-bay orbiter with an external LO_2/LH_2 tank mounted on a 260-inch SRM and a Generation 2 vehicle composed of the same orbiter and tanks mounted on a fully reusable heat-sink booster is illustrated in Figure 3-37. A single 260-inch SRM sized to provide a T/W of 1.3 is used in this system. Figure 3-38 illustrates a similar system, the difference being that the payload bay length is 40 feet in both Generation 1 and Generation 2. Figure 3-39 illustrates a system in which the orbiter has a 40-foot-long payload bay in

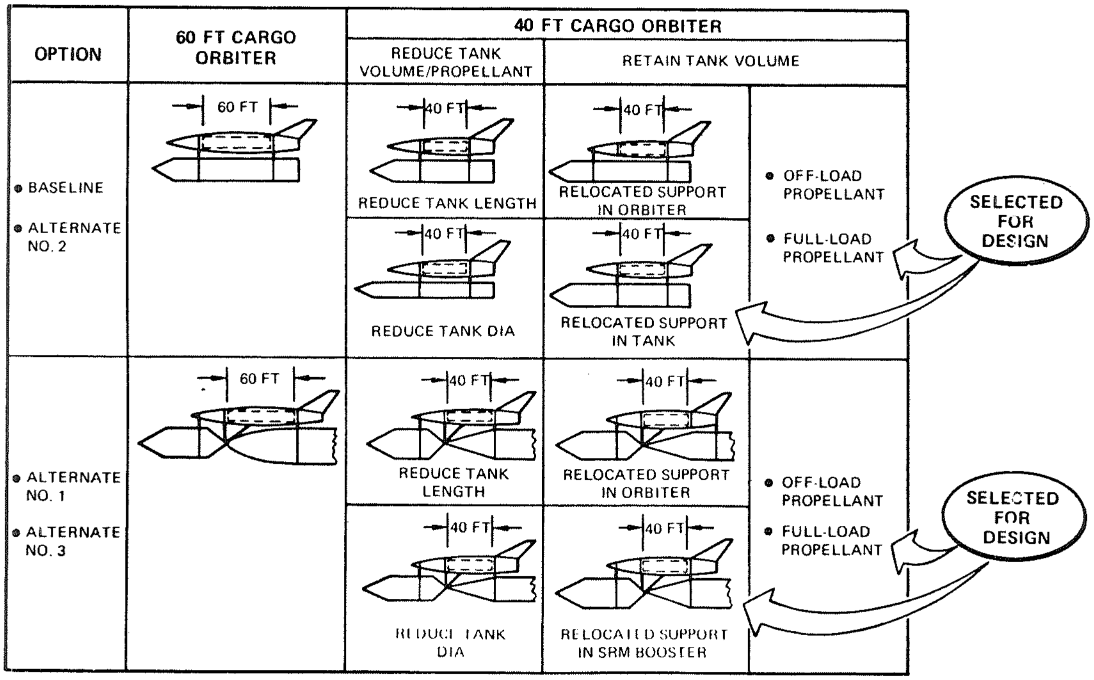
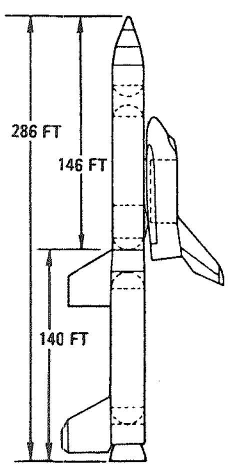
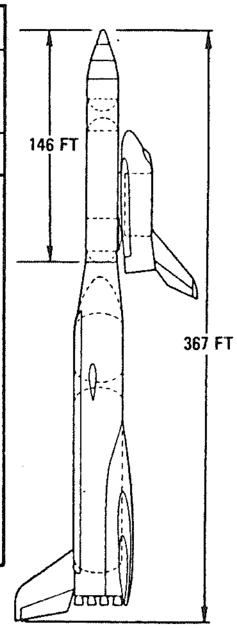


Figure 3-36. Tank to Orbiter Compatibility

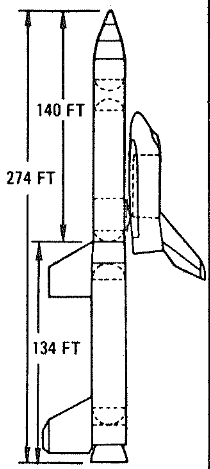


SYSTEM PARAMETERS		GEN 1		GEN 2	
CARGO BAY		15 FT X 60 FT		15 FT X 60 FT	
PAYLOAD	(K LB)	45 DUE E		40 POLAR	
OMS ΔV	(FPS)	900		650	
		BOOSTER	ORBITER	BOOSTER	ORBITER
SYSTEM GLOW	(M LB)	4.561		3.374	
T/W AT LO		1.3		1.3	
STAGE GROSS WT	(M LB)	3.527	1.034	2.35	1.024
MAIN PROP. WT	(M LB)	3.152	0.754	1.911	0.754
OMS PROP. WT	(K LB)	–	17.9	–	12.9
FLYBACK FUEL	(K LB)	–	–	39	–
FLYBACK RANGE	(N MI)	–	–	195	–
*STAGE DRY WT	(K LB)	375	148	376	148
ORBITER EXT TANKS	(K LB)	–	48	–	48
**STAGE LANDING WT	(K LB)	–	176	383	191
MAX $\bar{q}$	(PSF)	500	–	488	–
STAGING V_R	(FPS)	6177	–	7,029	–
γ_R	(DEG)	18.2	–	16	–
SRM L/D	(CYL)	3.8	–	–	–
MAIN ENG F_{SL}	(K LB)	5,925	–	366	–
F_{VAC}	(K LB)	–	420	401	420
NO. OF MAIN ENG		1	3	12	3

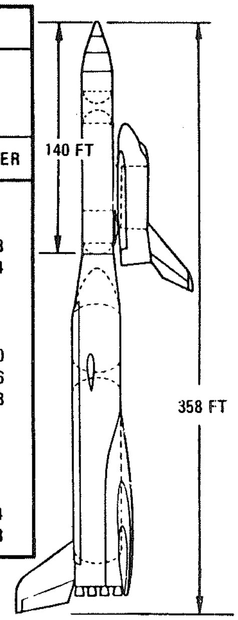


* PAYLOAD & TANKS NOT INCLUDED
** INCLUDES DOWN PAYLOAD WT: GEN 1, 15K 1B; GEN 2, 40K LB

Figure 3-37. Integrated System Description, EOHT 60-Foot Bay/ 260-Inch SRM Interim Booster

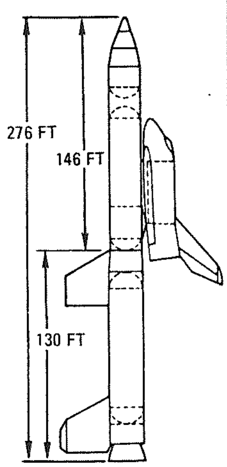


SYSTEM PARAMETERS		GEN 1		GEN 2	
CARGO BAY		15 FT X 40 FT		15 FT X 40 FT	
PAYLOAD	(K LB)	45 DUE E		40 POLAR	
OMS ΔV	(FPS)	900		650	
		BOOSTER	ORBITER	BOOSTER	ORBITER
SYSTEM GLOW	(M LB)	4.362	–	3.242	–
T/W AT LO		1.3	–	1.3	–
STAGE GROSS WT	(M LB)	3.369	0.993	2.260	0.983
MAIN PROP. WT	(M LB)	3.015	0.724	1.836	0.724
OMS PROP. WT	(K LB)		17.1	–	12.4
FLYBACK FUEL	(K LB)	–	–	37	–
FLYBACK RANGE	(N MI)	–	–	193	–
*STAGE DRY WT	(K LB)	354	140	364	140
ORBITER EXT TANKS	(K LB)	–	46	–	46
**STAGE LANDING WT	(K LB)	–	168	370	183
MAX $\bar{q}$	(PSF)	500	–	484	–
STAGING V_R	(FPS)	6184	–	7,004	–
SRM L/D	(CYL)	3.5	–	–	–
MAIN ENG F_{SL}	(K LB)	5,675		351	
F_{VAC}	(K LB)	–	404	386	404
NO. OF MAIN ENG		1	3	12	3

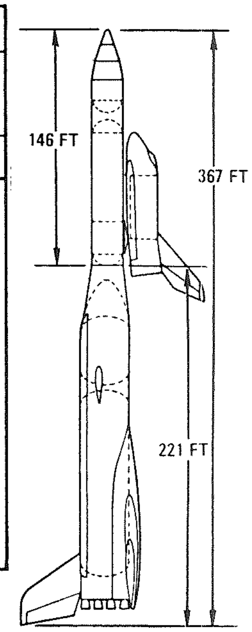


* PAYLOAD & TANKS NOT INCLUDED
** INCLUDES DOWN PAYLOAD WEIGHT: GEN 1, 25K LB; GEN 2, 40K LB

Figure 3-38. Integrated System Description, EOHT 40-Foot Bay/ 260-Inch SRM Interim Booster



SYSTEM PARAMETERS		GEN 1		GEN 2	
CARGO BAY		15 FT X 40 FT		15 FT X 60 FT	
PAYLOAD	(K LB)	45K DUE E		40K POLAR	
OMS ΔV	(FPS)	900		650 FPS	
		BOOSTER	ORBITER	BOOSTER	ORBITER
SYSTEM GLOW	(M LB)	4.175		3.374	
T/W AT LO		1.3		1.3	
STAGE GROSS WT	(M LB)	3.15	1.025	2.35	1.024
MAIN PROP WT	(M LB)	2.82	0.754	1.911	0.754
OMS PROP WT	(K LB)	–	17.1	–	12.9
FLYBACK FUEL	(K LB)	–	–	39	–
FLYBACK RANGE	(N MI)	–	–	195	–
*STAGE DRY WT	(K LB)	330	140	376	148
ORBITER EXT TANK WT	(LB)	–	48	–	48
**STAGE LANDING WT	(K LB)	–	168	383	191
MAX $\bar{q}$	(PSF)	500		488	
STAGING V_R	(FPS)	5,932		7,029	
γ_R	(DEG)	19		16	
SRM L/D	(CYL)	3.2		–	–
MAIN ENG F_{SL}	(K LB)	5440		366	
V_{AC}	(K LB)		420	401	420
NO. OF MAIN ENG		1	3	12	3



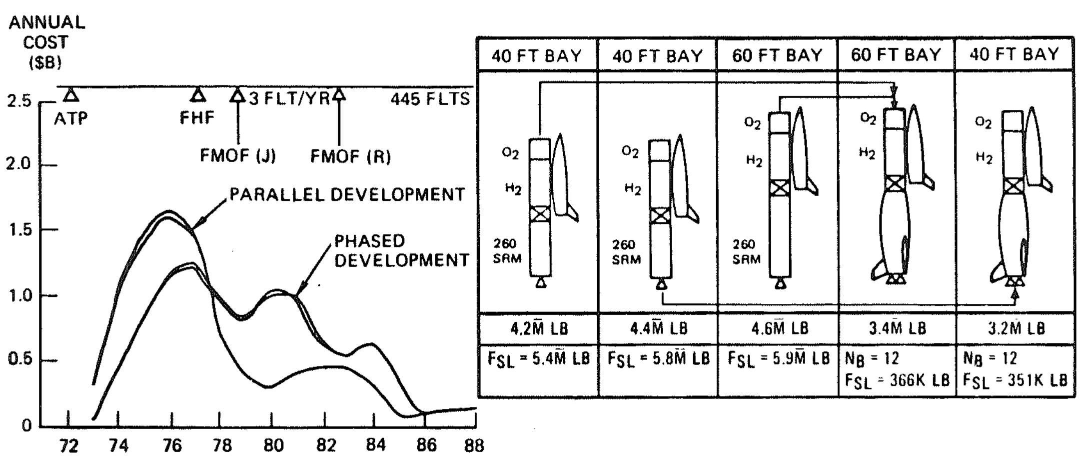
* PAYLOAD & EXPENDABLE TANKS - NOT INCLUDED
** INCLUDES DOWN PAYLOAD WEIGHT; GEN 1, 25K LB; GEN 2, 40K LB

Figure 3-39. Integrated System Description, EOHT 40- by 60-Foot Bay/ 260-Inch SRM Interim Booster

Generation 1 and a 60-foot-long cargo bay in Generation 2. Figure 3-40 illustrates program cost comparisons for three phased development programs in which 40-foot-payload-bay first generation and 60-foot second generation vehicles are compared with configurations in which the payload bay length is 40 feet without variation and 60 feet without variation. These programs include an interim booster for the first generation. A comparison is then made to two programs in which no interim booster is used, and the payload bay length is either 40 feet or 60 feet. In the phased development programs illustrated, it is assumed that the first-generation 40-foot orbiters are not refurbished to 60-foot payload bay lengths. It is seen that the phased development programs result in considerable reduction in peak annual funding, accompanied by a noticeable increase in total program costs. However, there is little difference between the three phased development programs examined. Also, the program including both a 40-foot and a 60-foot-payload-bay orbiter is the most expensive of the three phased development programs because of the additional orbiter design and test costs. However, for both the phased and parallel development programs, the continued use of an orbiter with 40-foot payload bay results in the least program cost.

3.7.5 Test Program Impact

Payload bay growth from 15/40 feet in the Generation 1 orbiter to 15/60 feet in Generation 2 requires additional structural and flight testing.



	PHASED DEVELOPMENT			PARALLEL DEVELOPMENT	
CARGO BAY LENGTH	40 FT/60 FT	40 FT/40 FT	60 FT/60 FT	40 FT	60 FT
PEAK ANNUAL FUNDING/YEAR	1.23 B$/77	1.22/77	1.23/77	1.62/76	1.67/76
ORBITER DDT&E	3.24	3.14	3.18	2.74	2.90
REUSABLE BOOSTER DDT&E	2.74	2.74	2.77	2.66	2.69
EXPENDABLE BOOSTER DDT&E	0.25	0.25	0.25	0	0
TOTAL PROGRAM	10.93	10.72	10.83	9.83	10.02

Figure 3-40. Program Cost Comparison, Cargo Bay Length Variation

Body and wing structural static tests are required to verify the new structure with added sections, the lengthened cargo bay doors require structural static tests, and structural tests incident to the changed location of external tank fittings must be performed. These tests must precede first manned orbital flight (MOF) for Generation 2.

Additional horizontal flight testing is required for stability and control, aerodynamic performance, and structural verification. These tests are estimated to require approximately 55 flight hours and six months to complete.

3.7.6 Mission Capabilities

The effects of payload bay size on mission capability are illustrated by a model built for the first 12 shuttle flights. For a phased program, the evolution of the shuttle's operational capability is spread over four years. The model was constructed from NASA-supplied data on the first 10 flights. Two flights were added at the beginning as solely flight tests, and the NASA data were modified to reflect new mission and payload-information. Manipulators are used for payload deployment and retrieval (Table 3-6A).

Table 3-6B shows that a 40-foot-long payload bay loses the space tug flights, with the tug's being replaced by expendable propulsive stage(s) to satisfy the same high-energy injection requirements. (The circles indicate deletions, and the crosses indicate the addition of the expendable stages.) This result is typical of mission model analysis: the tug is long—35 to 40 feet—and needs a long bay to accommodate the spectrum of tug payloads; assembly in space will allow the same tug flights with a 40-foot bay and additional shuttle launches. Apart from the tug, a 40-foot payload bay inhibits the launching of telescopes—such as Goddard's large space telescope—and DOD's large satellites, which run to 60 feet in length.

3.7.7 Recommendation

Figure 3-41 presents a summary of the significant results of this analysis. In general, small cost advantage—if any—is obtained through using a 40-foot payload bay in the first generation. However, it is felt that this system features a somewhat lower risk in acquiring flight experience and in developing better-defined requirements for the larger second-generation system.

Table 3-6. Initial Shuttle Missions Cargo Capability

A

Full Capability Orbiter
(15 FT X 60 FT Cargo Bay; Manipulators Installed)

ON-ORBIT TASKS	1978	1979		1980			1981			1982		
	FLIGHT NO.											
	1	2	3	4	5	6	7	8	9	10	11	12
C/O & OPERATE MANIPULATORS	SOLELY FLT TEST	SOLELY FLT TEST	X	X	X	X	X	X	X	X		X
EVA			X	X	X							
DEPLOY & RETRIEVE PASSIVE MODULES			X	X								
C/O & DEPLOY SATELLITES				X	X		X	X	X	X		X
RENDEZVOUS				X	X	X			X	X		
C/O & OPERATE PAYLOAD MODULES				X	X						X	
RETRIEVE SATELLITES						X						
C/O & DEPLOY PROPULSIVE STAGES								X				
C/O & DEPLOY TUG									X	X		
RETRIEVE TUG									X	X		
SUPPORT NON-ASTRONAUT PERSONNEL											X	
ONBOARD SCIENTIFIC EXPERIMENTS											X	

← SCIENTIFIC MISSIONS BASED ON MSC REPT 70 FM195 →

B

Full Capability Orbiter
(15 FT X 40 FT Cargo Bay; Manipulators Installed)

ON-ORBIT TASKS	1978	1979		1980			1981			1982		
	FLIGHT NO.											
	1	2	3	4	5	6	7	8	9	10	11	12
C/O & OPERATE MANIPULATORS	SOLELY FLT TEST	SOLELY FLT TEST	X	X	X	X	X	X	X	X		X
EVA			X	X	X							
DEPLOY & RETRIEVE PASSIVE MODULES			X	X								
C/O & DEPLOY SATELLITES				X	X		X	X	X	X		X
RENDEZVOUS				X	X	X			X	X		
C/O & OPERATE PAYLOAD MODULES				X	X						X	
RETRIEVE SATELLITES						X						
C/O & DEPLOY PROPULSIVE STAGES								X	X	X		
C/O & DEPLOY TUG									X	X		
RETRIEVE TUG									X	X		
SUPPORT NON-ASTRONAUT PERSONNEL											X	
ONBOARD SCIENTIFIC EXPERIMENTS											X	

← SCIENTIFIC MISSIONS BASED ON MSC REPT 70 FM195 →

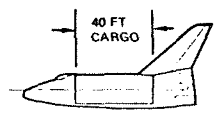
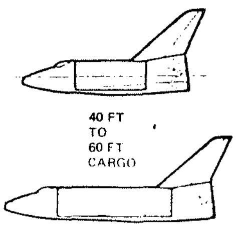
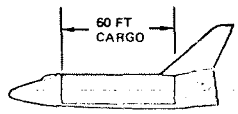
			
ORBITER DRY WEIGHT 1000 LB	140 ✓		148
COST • PROGRAM $B • PEAK ANNUAL $B	10.72 1.22	10.93 1.23	10.83 1.23
PROGRAM/ DESIGN	• CONSTANT ORBITER/TANK DESIGN POSSIBLE • DESIGN FOR MAX PAYLOAD • CONSTANT TOOLING • LOWER GEN 2 CARGO SIZE	• SIGNIFICANT REDESIGN • UTILIZE EXISTING TOOLING WHERE POSSIBLE • ADDITIONAL TESTING • 2 DESIGNS – COMMON FEATURES WHERE NO MAJOR PENALTY	• CONSTANT ORBITER/TANK DESIGN POSSIBLE • DESIGN FOR MAX PAYLOAD • CONSTANT TOOLING • LOWER GEN 2 CARGO SIZE • LARGE CARGO SIZE CAPABILITY
RISK	• MIN RISK • DESIRED MISSION CAPABILITY	• GEN 2 REQMTS BETTER KNOWN WHEN GEN 2 VEHICLE DESIGNED & BUILT • EXPERIENCE GAINED BEFORE DEVELOPMENT OF LARGE SYSTEM	• MISSION REQMTS MAY CHANGE

Figure 3-41. Summary and Recommendation Payload Size Effects (Phased Development Program)

3.8 SINGLE-ENGINE ORBITER

An investigation was conducted to determine the impact of restricting the number of orbiter engines to one. Design studies were carried out to determine orbiter weight, size, and aerodynamic characteristics, to define abort procedures, to compare costs and schedules of a single-engine orbiter and a three-engine orbiter system, and to assess the effect of a single-engine design on propulsion system thrust level and on external tank size.

3.8.1 Comparison of Vehicles With 12- and 15-Foot-Diameter Payload Bays

Figure 3-42 summarizes the comparison of single-engine orbiters with a three-engine orbiter. Each configuration had a 40-foot-long payload bay. However, for the single-engine orbiters, the payload bay diameter was varied (15 feet and 12 feet). The three-engine vehicle had a 15-foot-diameter payload bay. The data in Figure 3-42 indicate that little advantage is gained by using a 12-foot-diameter payload bay in a single-engine vehicle. A dry-weight saving of approximately 2000 pounds does not appear to warrant the loss of payload bay volume. No significant difference in engine thrust level is indicated. However, a significant reduction in dry weight can be achieved through the use of a single-engine vehicle instead of a three-engine vehicle— approximately 31,000 pounds.

3.8.2 Aerodynamic Characteristics

Two of the three single engine EOHT configurations evaluated aerodynamically during the Phase B extension are presented in Figure 3-42. For orbiters with 12-foot-diameter payload bays, payload bay lengths of both 40 and 60 feet were investigated. The orbiter with the 12- by 60-foot payload bay (Configuration -0075) was optimized for aerodynamic performance; the other two single-engine orbiter configurations were designed to improve packaging efficiency. The -0075 orbiter had a good hypersonic maximum L/D ratio of 2.37 and an L/D of 1.71 at an entry angle of attack of 30 degrees.

Aerodynamic performance and trim considerations for the orbiters with 15 by 40 and 12 by 40 payload bays are presented in Figures 3-43 and 3-44. Neither configuration has sufficient hypersonic L/D to achieve the desired maximum cross range with minimum TPS weight, as indicated by the relatively high values of W/C_LS. The aerodynamic performance deficiencies were caused primarily by reshaped and blunted noses, which were required to achieve better volumetric efficiencies. The blunted noses had negligible effect on hypersonic aerodynamic drag but caused significant hypersonic aerodynamic lift losses. Conversely, the subsonic L/D performance was improved, because the net subsonic drag was decreased due to reduced base drag. In addition, skin function drag becomes significant

	SINGLE ENGINE		3 ENGINE
	CARGO BAY SIZE		WITH ABORT EOHT
	15 FT X 40 FT	12 FT X 40 FT	15 FT X 40 FT
BODY VOL, FT^3	22,332	14,000	25,050
PACKAGING EFFICIENCY	42.4%	49.8%	43.5%
BODY SURFACE, FT^2	4,200	4,161	5,017
WING AREA EXPOSED, FT^2	1,705	1,702	2,358
DRY WEIGHT, LB (LESS TANK & PAYLOAD)	108,615	106,557	139,951
1 Hi-Pc-F_{VAC}, LB	605,203	597,832	3 x 404,000

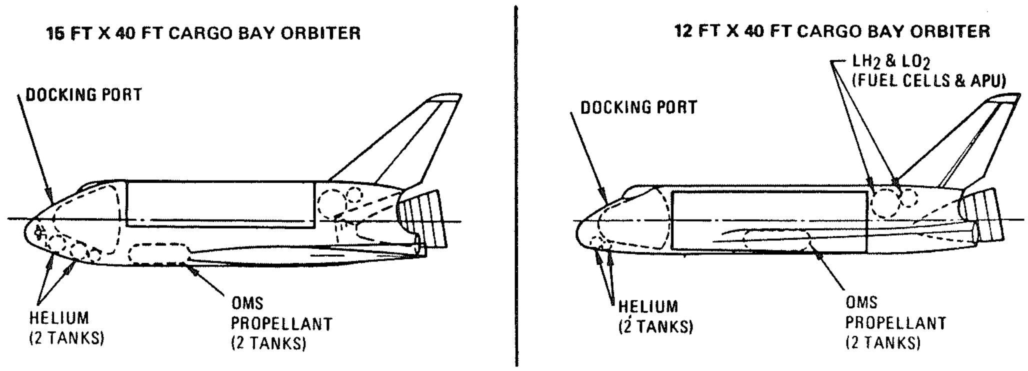
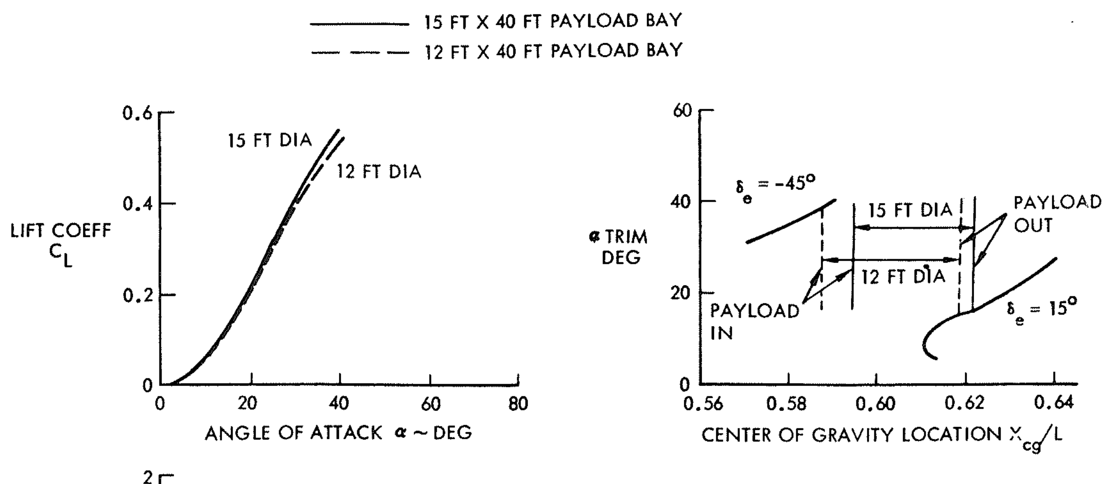
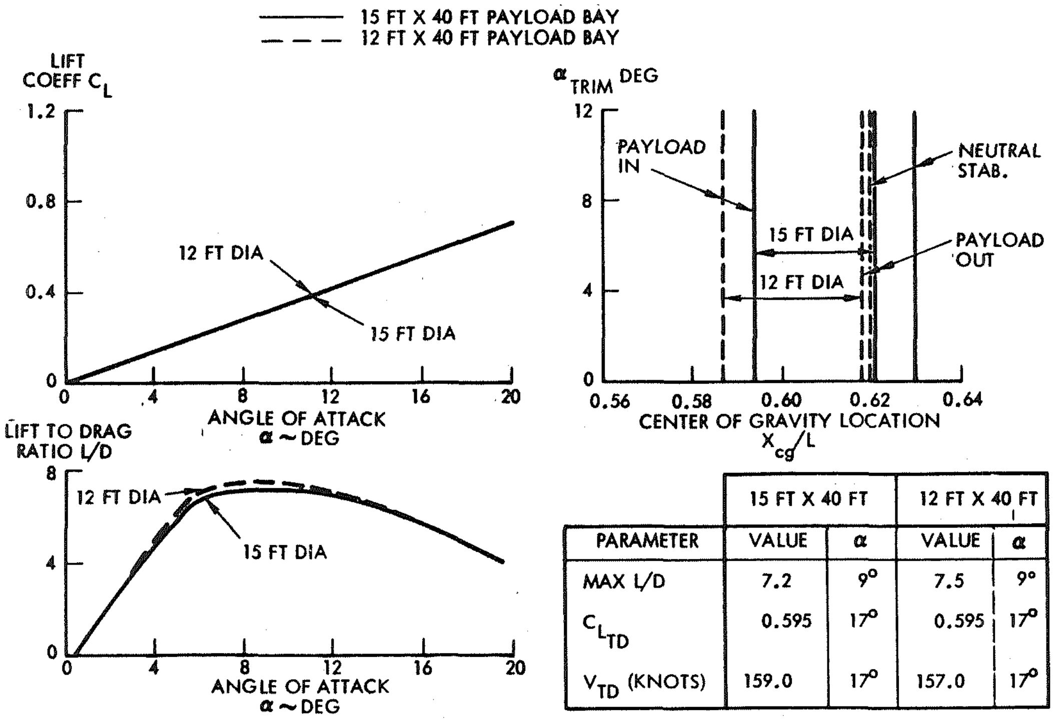


Figure 3-42. Single Engine EOHT Description, 12- by 15-Foot Cargo Bay Sizes



	15 X 40 FT	12 X 40 FT
PARAMETER	VALUE	VALUE
L/D	1.39	1.41
W/C_LS (PSF)	126	127
α (DEG)	30°	30°

Figure 3-43. Hypersonic Aerodynamic Characteristics Comparison of One Engine Orbiters (EOHT)



	15 FT X 40 FT		12 FT X 40 FT	
PARAMETER	VALUE	α	VALUE	α
MAX L/D	7.2	9°	7.5	9°
$C_{L_{TD}}$	0.595	17°	0.595	17°
V_{TD} (KNOTS)	159.0	17°	157.0	17°

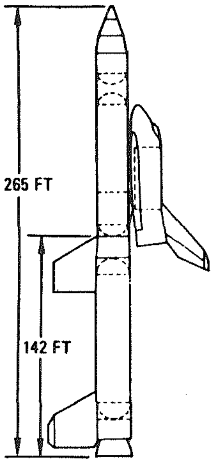
Figure 3-44. Subsonic Aerodynamic Characteristics Comparison of One Engine Orbiters (EOHT)

subsonically, and the higher-fineness-ratio -0075 configuration had more exposed surface area.

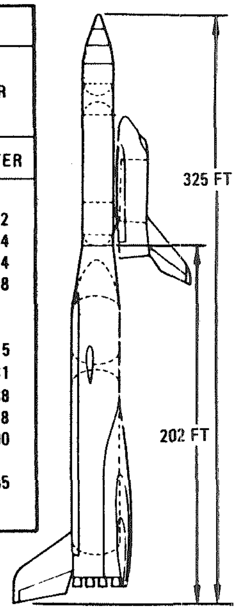
External tankage orbiters are more sensitive to the effects of payload weights and payload c.g. positions than are the all-reusable orbiters, because the payload constitutes a much higher percentage of the EOHT orbiters approach and landing weight. This results in lower maximum trim angles of attack at hypersonic velocities, with the payloads in, and in flights closer to neutral pitch stability limits, with payloads out and at subsonic velocities. The NR external-tankage configurations have been balanced for good flight performance in all flight velocity regimes by carefully positioning the wing with respect to the expected c.g. range and by shaping the fuselage nose to provide additional pitching moment, when needed, in the hypersonic flight regime.

3.8.3 System Comparison and Cost

Figures 3-45 and 3-46 show the system parameters for programs including a single-engine orbiter with a 15-foot-diameter payload bay and a single-engine orbiter with a 12-foot-diameter payload bay. Each program includes a 260-inch SRM as the interim booster. As anticipated, the system



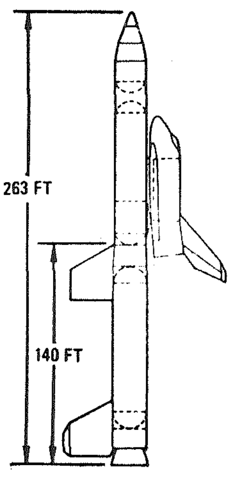
SYSTEM PARAMETERS		GEN 1		GEN 2	
CARGO BAY		15 FT X 40 FT		15 FT X 40 FT	
PAYLOAD	(K LB)	45K DUE E		40K POLAR	
OMS ΔV	(FPS)	900 FPS		650 FPS	
		BOOSTER	ORBITER	BOOSTER	ORBITER
SYSTEM GLOW	(M LB)	4,412,000		3,251,835	—
T/W AT LO		1.3		1.30	0.72
STAGE GROSS WT	(M LB)	3,561,000	851,000	2,409,870	841,964
MAIN PROP. WT	(M LB)	3,198,000	626,774	1,936,669	626,774
OMS PROP. WT	(K LB)	—	11,034	—	6,898
FLYBACK FUEL	(K LB)	—	—	61,827	—
FLYBACK RANGE	(N MI)	—	—	299	—
* STAGE DRY WT	(K LB)	263,000	108,615	386,651	108,615
ORBITER EXT TANK WT	(LB)	—	41,131	—	41,131
** STAGE LANDING WT	(K LB)	—	156,738	393,579	151,738
MAX $\bar{q}$	(PSF)	500	500	478	478
STAGING V_R	(FPS)			7,800	7,800
MAIN ENG F_{SL}	(K LB)	5,740,000		528,000	
VAC	(K LB)		607,185		607,185
NO. OF MAIN ENGINES		1	1	8	1



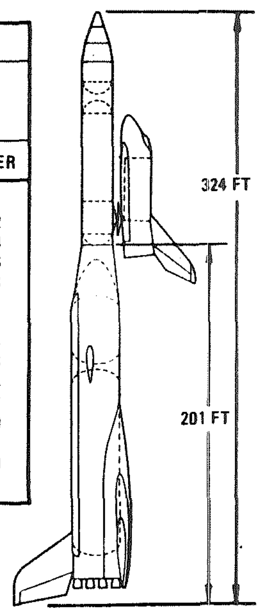
* PAYLOAD & EXPENDABLE TANKS & INTERSTAGE NOT INCLUDED

** INCLUDES PAYLOAD WT

Figure 3-45. Integrated EOHT System Description, 15- by 40-Foot Cargo Bay, Single Engine Orbiter With 260-Inch SRM Interim Booster



SYSTEM PARAMETERS		GEN 1		GEN 2	
CARGO BAY		12 FT X 40 FT		12 FT X 40 FT	
PAYLOAD	(K LB)	45K DUE E		40K POLAR	
OMS ΔV	(FPS)	900		650 FPS	
		BOOSTER	ORBITER	BOOSTER	ORBITER
SYSTEM GLOW	(M LB)	4,337,000		3,202,077	—
T/W AT LO		1.3		1.3	0.72
STAGE GROSS WT	(M LB)	3,501,000	836,700	2,374,144	827,933
MAIN PROP. WT	(M LB)	3,142,000	616,306	1,907,675	616,306
OMS PROP. WT	(K LB)	—	10,545	—	6,778
FLYBACK FUEL	(K LB)	—	—	61,173	—
FLYBACK RANGE	(N MI)	—	—	299	—
*STAGE DRY WT	(K LB)	259,000	106,494	381,381	106,494
ORBITER EXT TANK WT	(LB)	—	40,636	—	40,636
**STAGE LANDING WT	(K LB)	—	154,024	388,210	149,024
MAX $\bar{q}$	(PSF)	500	500	477	477
STAGING V_R	(FPS)			7,800	7,800
MAIN ENG F_{SL}	(K LB)	5,640,000		520.3	
VAC	(K LB)		598,000		598,000
NO. OF MAIN ENG		1	1	8	1



* PAYLOAD & EXPENDABLE TANKS & INTERSTAGE NOT INCLUDED

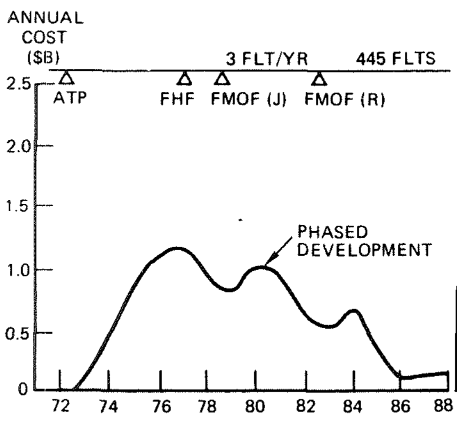
** INCLUDES PAYLOAD WEIGHT

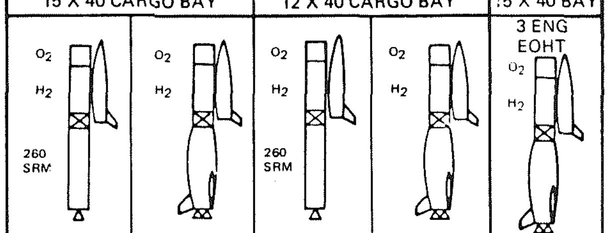
Figure 3-46. Integrated EOHT System Description, 12- by 40-Foot Cargo Bay, Single Engine Orbiter With 260-Inch SRM Interim Booster weights and main propulsion system requirements are similar for each. The SRM stages are small enough so that single stages are feasible within the length-to-diameter ratio specified by the SRM industry (less than 5.0).

Figure 3-38 provides the data for a program including a 40- by 15-foot-diameter-payload-bay orbiter, three-engine MPS, and a 260-inch SRM interin booster to be used in the comparison illustrated in Figure 3-47, where program costs are also shown. It is seen that the peak annual funding for each of the three programs is almost the same, with the three-engine system showing an increase of only $10 million during the peak annual funding year, 1977. Also, the total program costs are almost the same, with the three-engine system showing a slightly higher total program cost—approximately $100 million. It appears, therefore, that there is little cost advantage to a 12-foot-diameter payload bay or to a single-engine orbiter system. The loss in abort capability and payload volume is not warranted by the small cost saving.

3.8.4 Abort

The abort capability of a single-engine EOHT orbiter during the pre-staging ascent phase is similar to that of three-engine orbiters, except for the duration of the "no abort" regime (see Section 3.6.5). Because of the low T/W with only one engine, the time after lift-off when acceptable orbiter-alone flight can be initiated is longer. This is estimated to be approximately 20 seconds. The abort flight modes and procedures after 20 seconds are also





	15 X 40 CARGO BAY		12 X 40 CARGO BAY		15 X 40 BAY 3 ENG EOHT
GLOW	4,412,000	3,251,835	4,337,000	3,202,077	3,242,000
OLOW	851,000	841,964	836,700	827,933	983,000
ENG THRUST	5,740K	4,224K	5,640K	4,160K	4,212K
NO. ENG	1	8 AT 528K	1	8 AT 520K	12 AT 351K

	1 ENGINE		3 ENGINES
CONFIGURATION	15 X 40 CARGO BAY	12 X 40 CARGO BAY	15 X 40 CARGO BAY
PEAK ANNUAL FUNDING/YEAR	1.21 B$/77	1.21/77	1.22/77
ORBITER DDT&E	3.04	3.02	3.14
REUSABLE BOOSTER DDT&E	2.79	2.79	2.74
EXPENDABLE BOOSTER DDT&E	0.25	0.25	0.25
TOTAL PROGRAM	10.64 B$	10.61	10.72

Figure 3-47. Program Cost Comparison, Cargo Bay Diameter

slightly different because of the reduced T/W. However, the orbiter can be returned to the launch site by using the main propellant. Prestaging aborts are assumed to be caused by booster failures.

After staging, the loss of the orbiter engine negates the use of the main propellant for abort trajectory shaping. Consequently, the orbiter must be separated from the full (or partly full) tank and is committed to a suborbital reentry. Figure 3-48 shows the heating and loading problems engendered by this abort mode as functions of the abort initiation (engine-out) velocity. It is apparent that an orbiter designed for the normal missions could not withstand most of these aborts. The excessive axial load factors and dynamic pressures indicated in the figure are the result of modulating the angle of attack, as required, to maintain the normal load factor below 2.5 g's (design limit). A higher normal load factor design limit—4 g's, for example—would significantly reduce the peak axial load factors and heat rates.

3.8.5 Recommendation

The various considerations associated with a comparison of a single-engine orbiter and a multiple-engine orbiter system are summarized in Figure 3-49. The peak annual costs and program costs are quite similar. However, the multiple-engine vehicle offers more flexible abort modes. It is recommended, therefore, that the multiple-engine system be adopted.

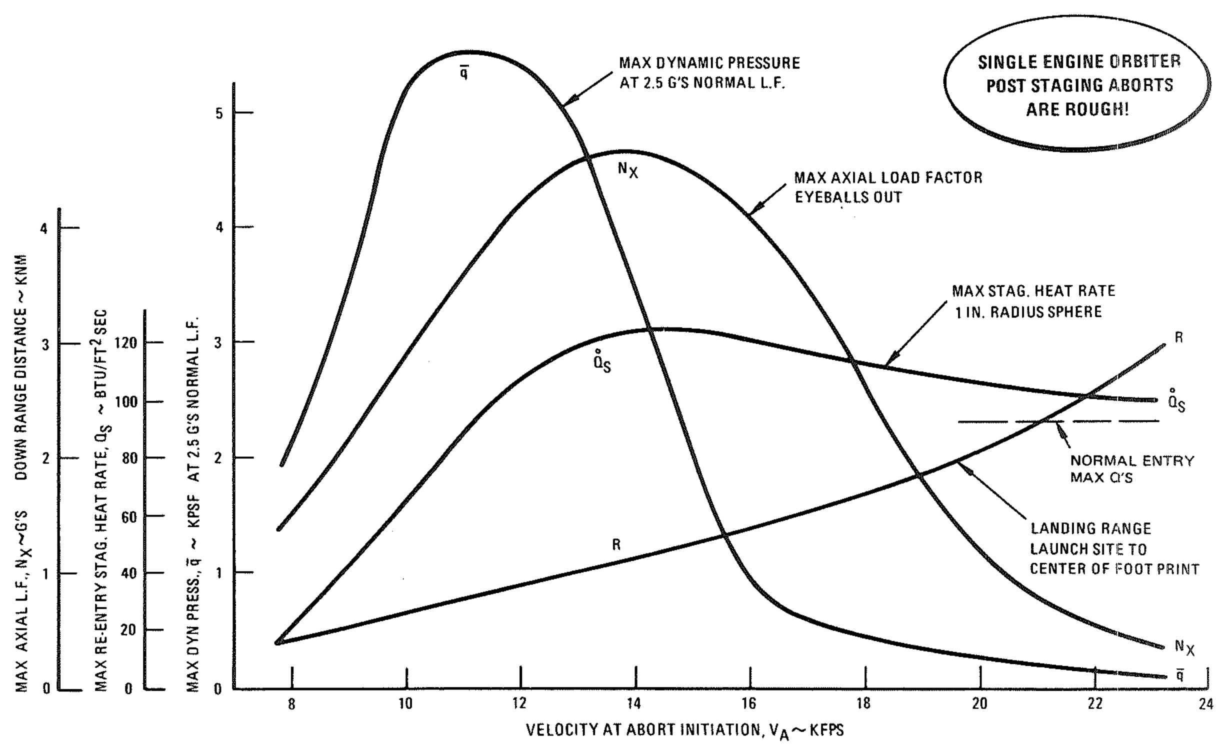


Figure 3-48. Post-Staging, Abort Reentry Characteristics, Single Engine Orbiter

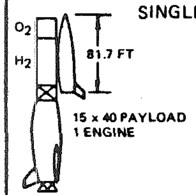
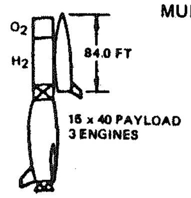
NO ONCE AROUND	ONCE AROUND
 SINGLE ENGINE ORBITER POSSIBLE 1 ENGINE F_{VAC} = 605K LB GLOW = 3,241K LB ORBITER DRY WT = 111K LB	 MULTIPLE ENGINES ENGINES F_{VAC} = 404K GLOW = 3,242K LB ORBITER DRY WT = 140K LB
COST PEAK ANNUAL $B 1.21 PROGRAM $B 10.64	COST PEAK ANNUAL $B 1.22 PROGRAM $B 10.72
ABORT MODE	
SINGLE ENGINE • PRE NORMAL STAGING • FLY ORBITER BACK TO LAUNCH SITE • NORMAL STAGING OR LATER • SEPARATE ORBITER & LAND AT ALTERNATE SITE OR DITCH	3 ENGINE • PRE NORMAL STAGING • FLY ORBITER BACK TO LAUNCH SITE • NORMAL STAGING OR LATER • ABORT ONCE AROUND
ALTERNATE MODE	
✓ MULTIPLE ENGINES •ABORT ONCE AROUND OR RETURN TO LAUNCH SITE	

Figure 3-49. Abort Recommendations

3.9 EXPANSION RATIO TRADE

A trade study was conducted to determine the effects on the orbiter, HO tank, and reusable LO_2/LH_2 heat sink booster associated with employment of a range of expansion ratios in the orbiter engines. The orbiter configuration had three high pressure engines. A two-position nozzle with ϵ = 150:1 was the baseline. A power head diameter of 90 inches was held constant for all cases. A two-position nozzle with ϵ = 120:1 and a fixed nozzle with ϵ = 90:1 were compared with the baseline. Both the engine weight variation and the effects on the base region of the orbiter were evaluated. To compare an orbiter with two-position nozzles with one having fixed nozzles, estimates were made for weight changes in the orbital maneuvering subsystem (OMS) compartment, the fuselage fairing, the base heat shield, and the main engines. Synthesis runs were made to size the vehicles of the system (orbiter, HO tank, and booster) for a payload of 40,000 pounds to low polar orbit.

Utilizing the 150:1 nozzle case as the baseline, the incremental dry weight figures shown in Table 3-7 were determined. The orbiter dry weight is reduced. HO tank weight increases due to increased propellant needed for the reduced specific impulse. The booster dry weight increases because of the larger orbiter liftoff weight for reduced expansion ratios, and because the optimum staging velocity is increased slightly for the reduced ϵ orbiter engines.

Table 3-7. Incremental Dry Weights Versus Engine Expansion Ratios

	150:1 (2-positions)	120:1 (2-positions)	90:1 (fixed)
Orbiter	0 (ref)	-550	-1,200
HO Tank	0	+270	+1,170
Reusable Booster	0	+4,555	+5,975
Total Delta Weight (pounds)		+4,275	+5,945

Incremental total program cost estimates were made, considering the above data. Results are given in Table 3-8.

Table 3-8. Total Delta Program Cost Estimates
(Millions of Dollars)

Orbiter	0 (Ref.)	-$ 3	-$ 1
HO Tank		+$ 1	+$ 2
Reusable Booster		+$13	+$22
SSME		-$ 5	-$42
Total Delta Cost		+$ 6	-$19

The 90:1 fixed nozzle was found to be the least expensive; it also would eliminate the requirement for a two-position nozzle on the orbiter engine, and the associated deployment mechanisms. This would lead to a less complex engine development program. From the standpoint of reliability, the possibility of failure of the nozzle to extend or retract when expected in flight would be eliminated by the fixed-nozzle selection.

3.10 EXPENDABLE BOOSTER SYSTEMS

The expendable boosters considered in Phase 1 of the study are listed along with the orbiter configuration on Table 3-9. In addition to the 260 SRM, 120 SRM cluster, and the LO_2/LH_2 core expendable boosters, three more boosters were studied: S-1C, Titan 3L, and an MCD (minimum cost design or "big dumb booster"). The intent was to find the lowest cost interim booster for the first four years of the flight program, so that the reusable booster development could be phased for its peak funding to occur after the orbiter funding peaked out. It was expected that the development of two reusable vehicles would not drive the total cost above on $1 billion in a single year. The orbiter flights would start in 1978 with interim booster and continue operationally until the reusable booster, with its development phased later, could be ready for flight.

Table 3-9. Candidate Configuration Matrix

CONFIG NO.	DEVELOPMENT APPROACH	TANKS[2]	CARGO BAY	NO. OF ORBITER ENGINES	EXPEND. BOOSTER	PAYLOAD[3]
1	GEN 1 & 2	HO	15 x 60	3	260 SRM	65
2	GEN 1 & 2	HO	15 x 60	3	LOX/LH_2 CORE	65
3	GEN 1 & 2	HO	15 x 60	3	CLUSTER 120 OR 156 IN.	65
4	GEN 1 & 2	H	15 x 60	3	260 SRM	65
5A	GEN 1	HO	15 x 40	3	260 SRM	45
5B	GEN 2	HO	15 x 60	3		65
6A	GEN 1	HO	12 x 40	1	260 SRM	45
6B	GEN 2	HO	12 x 60	1		65
6C	GEN 1	HO	12 x 40	3	260 SRM	45
6D	GEN 2	HO	12 x 60	3		65
7A	GEN 1	HO	12 x 40	3	LOX/LH_2 CORE	45
7B	GEN 2	HO	12 x 60	3		65
8A	GEN 1	H	12 x 40	3	CLUSTER SOLIDS	45
8B	GEN 2	H	12 x 60	3		65
9	GEN 1 & 2	H	15 x 60	3	LOX/LH_2 CORE	65

NOTE: 2 HO EXTERNAL HYDROGEN & OXYGEN TANKS
H EXTERNAL HYDROGEN TANKS
3 UP PAYLOAD = 65K LB, DOWN PAYLOAD = 40K LB
UP PAYLOAD = 45K LB, DOWN PAYLOAD = 25K LB

The complete expendable booster matrix is depicted on Figure 3-50.

3.10.1 260-Inch SRM Systems

Single-stage and two-stage 260-inch SRM expendable boosters were studied. Because of the higher complexity of a two-stage system, its slight weight advantage was not sufficient to overcome a large development cost. The most promising single stage 260-inch SRM boosters are shown in Figure 3-51 for a 40-foot cargo bay orbiter, and in Figure 3-52 for a 60-foot cargo bay orbiter.

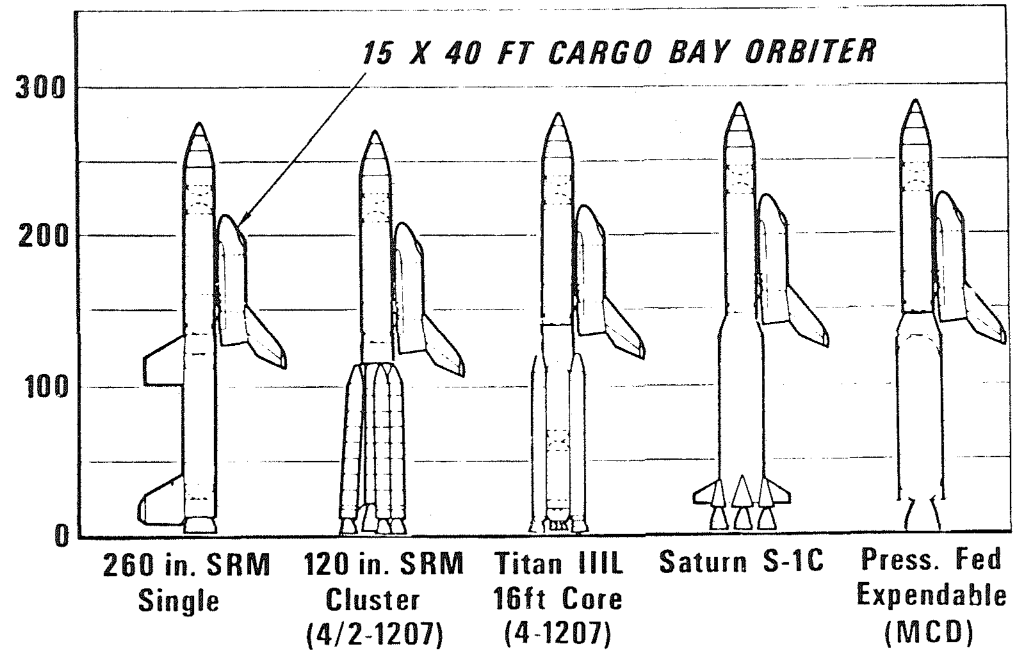
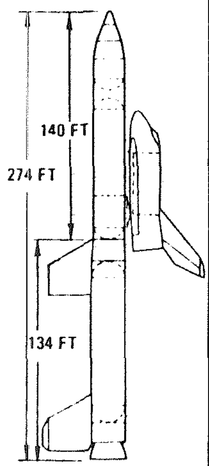


Figure 3-50. Interim Booster Combinations



SYSTEM PARAMETERS		GEN 1		GEN 2	
CARGO BAY		15 FT X 40 FT		15 FT X 40 FT	
PAYLOAD	(K LB)	45 DUE E		40 POLAR	
OMS ΔV	(FPS)	900		650	
		BOOSTER	ORBITER	BOOSTER	ORBITER
SYSTEM GLOW	(M LB)	4.362	–	3.242	–
T/W AT LO		1.3	–	1.3	–
STAGE GROSS WT	(M LB)	3.369	0.993	2.260	0.983
MAIN PROP. WT	(M LB)	3.015	0.724	1.836	0.724
OMS PROP WT	(K LB)		17.1	–	12.4
FLYBACK FUEL	(K LB)	–	–	37	–
FLYBACK RANGE	(N MI)	–	–	193	–
*STAGE DRY WT	(K LB)	354	140	364	140
ORBITER EXT TANKS	(K LB)	–	46	–	46
**STAGE LANDING WT	(K LB)	–	168	370	183
MAX q	(PSF)	500	–	484	–
STAGING V_R	(FPS)	6,184	–	7,004	–
SRM L/D	(CYL)	3.5	–	–	–
MAIN ENG F_{SL}	(K LB)	5,675		351	
F_{VAC}	(K LB)	–	404	386	404
NO. OF MAIN ENG		1	3	12	3

* PAYLOAD & TANKS NOT INCLUDED
** INCLUDES PAYLOAD

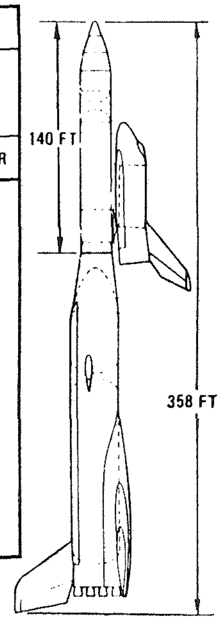
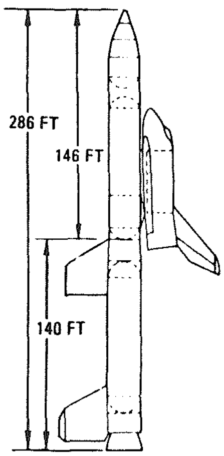


Figure 3-51. Integrated System Description, EOHT 40-Foot Bay/ 260-Inch SRM Interim Booster

In Figure 3-51, the orbiter had the short payload bay (40 feet versus 60 feet) and with its lower GLOW, the MPS was sized for a thrust of 404,000 pounds. The orbiter in Figure 3-52 with the larger payload bay (60 feet) and larger gross lift-off weight (GLOW) required more main propulsion subsystem (MPS) thrust (420,000 pounds). The final configuration shown in Figures 3-53, 3-54, and 3-55 used the MPS engines for Gen 2 in both Gen 1 and Gen 2.



SYSTEM PARAMETERS		GEN 1		GEN 2	
CARGO BAY		15 FT X 60 FT		15 FT X 60 FT	
PAYLOAD	(K LB)	45 DUE E		40 POLAR	
OMS ΔV	(FPS)	900		650	
		BOOSTER	ORBITER	BOOSTER	ORBITER
SYSTEM GLOW	(M LB)	4.561		3.374	
T/W AT LO		1.3		1 3	
STAGE GROSS WT	(M LB)	3.527	1.034	2.35	1.024
MAIN PROP. WT	(M LB)	3.152	0.754	1.911	0.754
OMS PROP. WT	(K LB)	–	17.9	–	12.9
FLYBACK FUEL	(K LB)	–	–	39	–
FLYBACK RANGE	(N MI)	–	–	195	–
*STAGE DRY WT	(K LB)	375	148	376	148
ORBITER EXT TANKS	(K LB)	–	48	–	48
**STAGE LANDING WT	(K LB)	–	176	383	191
MAX $\bar{q}$	(PSF)	500	–	488	–
STAGING V_R	(FPS)	6177	–	7,029	–
γ_R	(DEG)	18.2	–	16	–
SRM L/D	(CYL)	3.8	–	–	–
MAIN ENG F_{SL}	(K LB)	5,925	–	366	–
F_{VAC}	(K LB)	–	420	401	420
NO. OF MAIN ENG		1	3	12	3

* PAYLOAD & TANKS NOT INCLUDED
** INCLUDES PAYLOAD

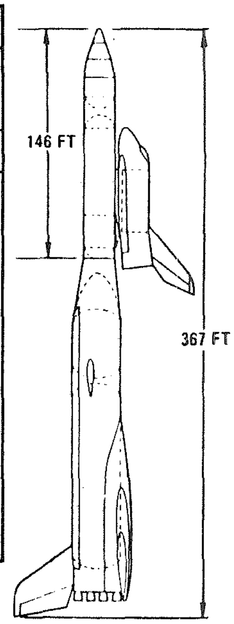


Figure 3-52. Integrated System Description, EOHT 60-Foot Bay/ 260-Inch SRM Interim Booster

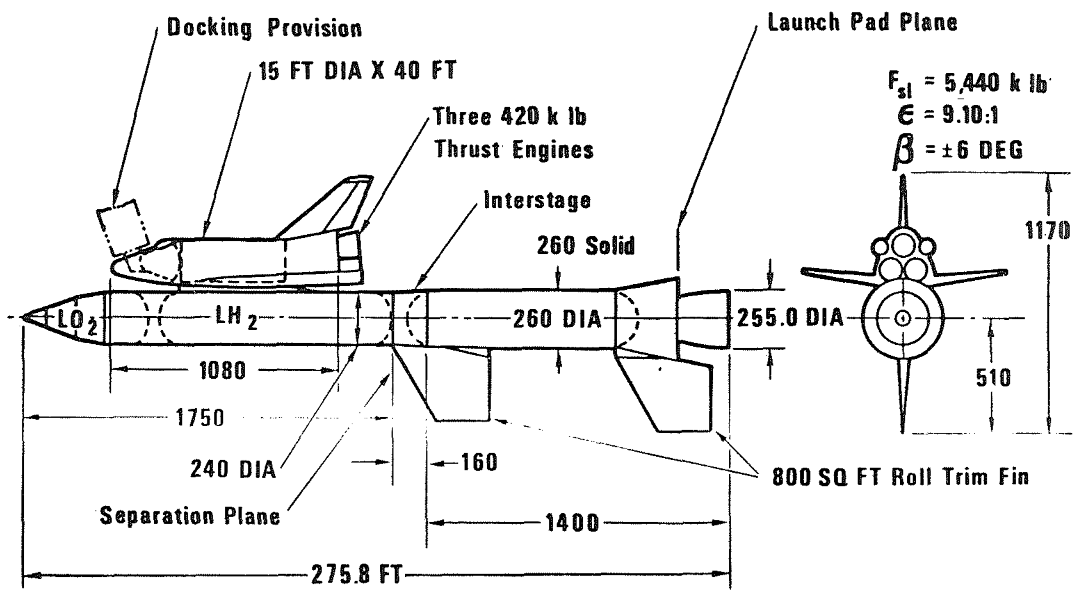
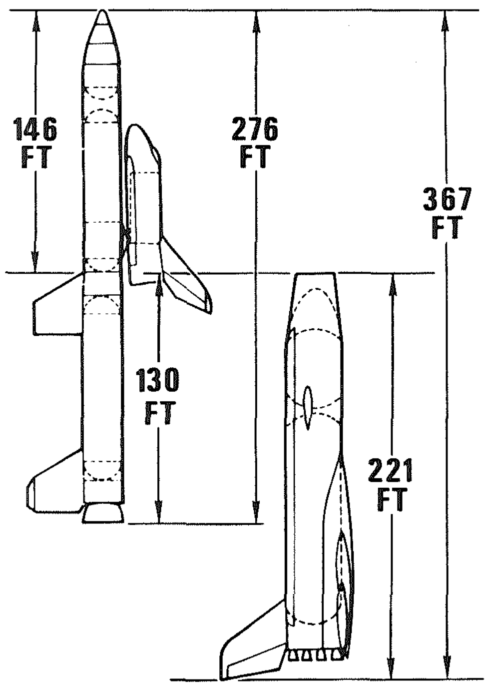


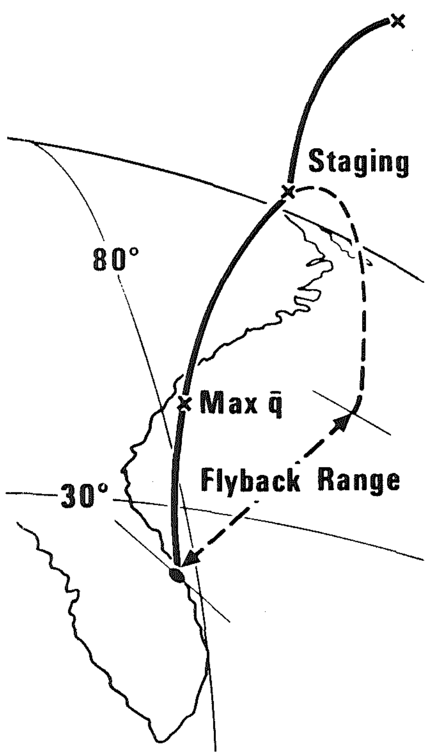
Figure 3-53. 260-Inch SRM Interim Booster



	Gen 1		Gen 2	
	B	Orb	B	Orb
Stage				
Gross Wt (m lb)	3.15	1.025	2.35	1.024
Dry Wt (k lb*)	330	140	376	148
Ext Tank				
Wt (k lb)		48		48
Main Engines				
Number	1	3	12	3
Thrust (k lb)	5440**	420	366**	420
Sys GLOW (m lb)	4.17		3.37	
P/L (Dn Wt)(k lb)	45 (25) Due E		40 (40) Polar	
OMS ΔV (fps)	900		650	

* Payload & Exp Tanks not Included
** Sea Level

Figure 3-54. 260-Inch SRM Interim Booster (15 by 40 and 15 by 60 Feet), 3-Engine Orbiter



	Gen 1		Gen 2	
P/L (Dn Wt) (k lb)	45 (25) Due E		40 (40) Polar	
OMS ΔV (fps)	900		650	
	B	Orb	B	Orb
Max $\bar{q}$ (psf)	500	500	488	488
Staging V_R (fps)	5932	5932	7029	7029
Staging γ_R (deg)	19	19	16	16
Flyback (n mi)			195	

* (3) ENGINE ORBITER

Figure 3-55. 260-Inch SRM Interim Booster (15 by 40 and 15 by 60 Feet)

3.10.2 Development Status

Current experience with 260-inch-diameter SRM's is limited to the demonstration of feasibility through the fabrication and successful static-test firing of three motors of this size. The test firings were conducted during the 1965-1967 time period by the Aerojet Solid Propulsion Company at their Dade County, Florida, facility under contract to the USAF and NASA. The motors were of monolithic construction with a single fixed nozzle and contained from 1.645 to 1.676 million pounds of hydrogen terminated polybutadiene (HTPB) propellant. The maximum thrust ranged from 3.141 to 5.884 million pounds. The primary accomplishments of these initial firings were the demonstration of predictable and reproducible motor performance, development of the low-cost HTPB propellant formulation, and the producibility of large, high-performance motor chambers and large nozzle ablative components. Through subsequent technology contracts with NASA, the feasibility of an acceptable thrust vector control (TVC) system was demonstrated by the fabrication and bench-testing of a typical 260-inch-diameter, flexible-seal, movable nozzle. An extensive study of transportation and handling requirements unique to this large-size SRM has also been completed under a NASA study contract.

Approximately four years are required for the design, development, and qualification of a 260-inch-diameter SRM to meet space shuttle requirements. The primary development issues would be the demonstration of an acceptable TVC system and the evaluation of motor and component acceptance criteria.

3.10.3 260-Inch SRM Booster Avionics Subsystem

The design concept of the 260-inch SRM booster avionics subsystem was driven by a number of key ground rules: (1) minimum cost; (2) minimum booster hardware; (3) all computations for booster flight control to be performed by the orbiter; (4) hardwire interface with orbiter; (5) orbiter provides electrical power prior to separation; and (6) orbiter provides RF uplink and downlink. Based on these ground rules, analysis of the booster mission requirements resulted in the minimum avionics subsystem illustrated in Figure 3-56. As shown in Figure 3-56, the following avionics subsystems are required: engine control; separation, including retrorocket motor ignition; flight control (roll control and rate gyros); power distribution; instrumentation; thrust termination; and malfunction detection. These subsystems are controlled by the booster subsystem controller, which provides the appropriate timing, permit logic, and inhibit logic required during booster operation.

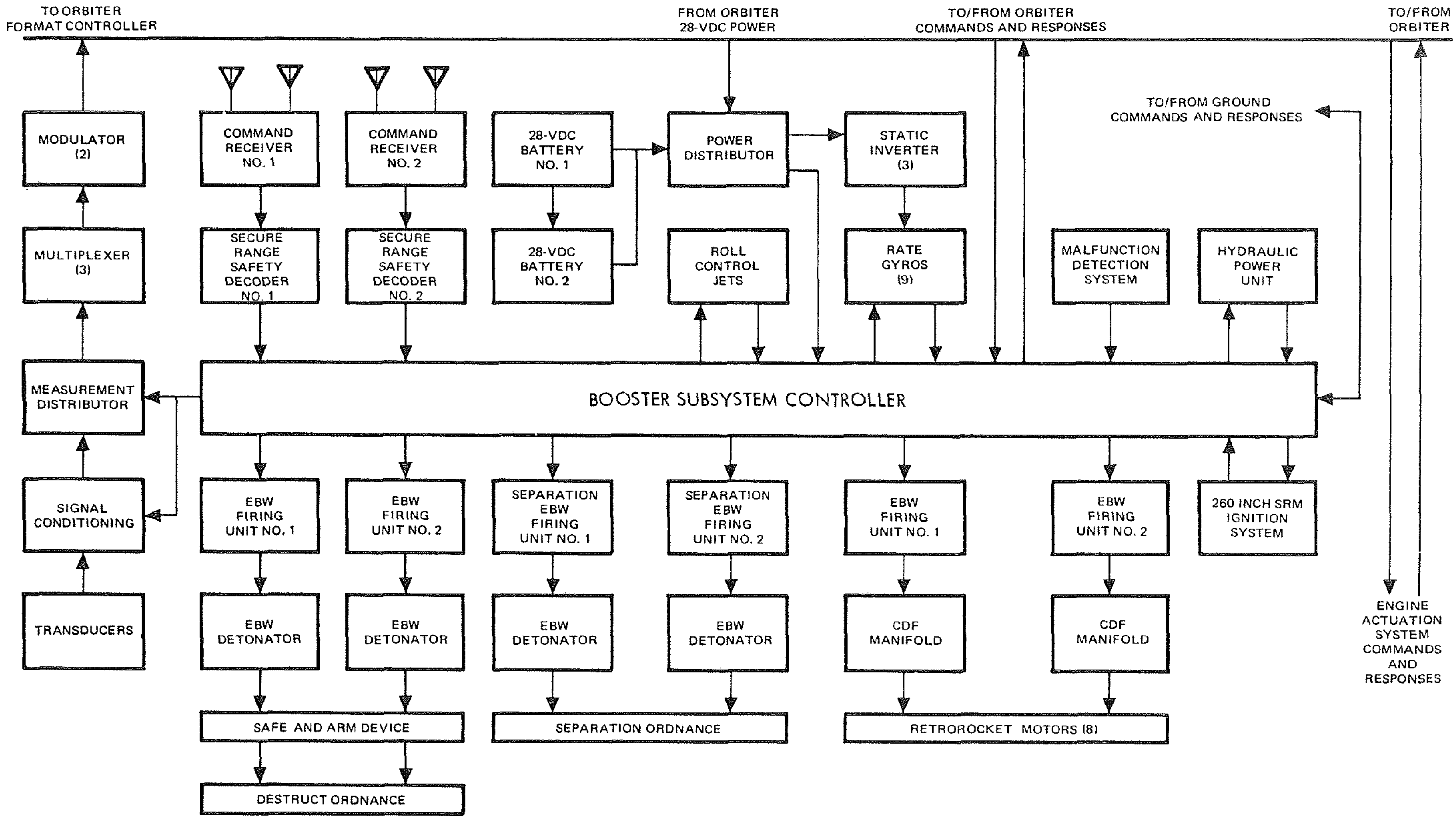


Figure 3-56. 260-Inch Solid Rocket Motor Booster Avionics Block Diagram

The separation, thrust termination, malfunction detection systems, and rate gyros are similar to those presently used on the Saturn V vehicle. The rate gyro subsystem consists of nine single-degree-of-freedom gyros in a a triple-redundant configuration and is used to sense the angular rate of movement of the booster about the roll, yaw, and pitch axes for attitude control and malfunction detection.

The instrumentation system, similar to Saturn S-II hardware, transmits serial digital data to the orbiter for recording and/or transmitting to ground stations. The system consists of a multiplexer for analog signals and a multiplexer for discrete measurements, with the outputs being transmitted to a pulse code modulator for transmission over twisted-shielded pairs to the orbiter. Vibration measurements are routed to a separate multiplexer and then to a frequency modulator for transmission to the orbiter.

As shown in Figure 3-56, engine actuation signals will be sent directly from the orbiter to the booster engine actuators for control during boost. Feedback signals will be sent directly to the orbiter via hardwire.

The above described subsystem meets the established requirements and ground rules for a low-cost avionics subsystem using proven off-the-shelf hardware. Only the minimum number of avionic components necessary for operational needs were included on the expendable booster, and all computations and functions possible were assigned to the orbiter.

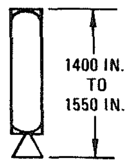
3.10.4 Booster Cost

The 260-inch SRM cost data are presented in Figures 3-57 and Table 3-10. Some analysis went into the question of monolithic or segmented construction; the final consensus was that the most feasible method of fabrication was monolithic. The costs given in Table 3-10 are for a monolithic structure.

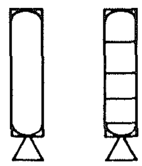
3.10.5 Mate and Erect 260-Inch SRM

This section addresses the problem associated with mating and erecting the 260-inch monolithic SRM at Kennedy Space Center. The basic impact on the mate-erect cycle, equipment, and facilities is determined by the weight, whether the SRM is segmented or monolithic. SRM's, or segments with handling gear weighing more than 1,000,000 pounds, will require erecting facilities exceeding the capability of the modified vertical assembly building (VAB) high-bay cranes. SRM's, or segments with handling gear weighing less than 1,000,000 pounds, can be erected in the VAB; the method recommended is discussed in Section 3.10.10.

- MOTOR SIZES - 3.0 TO 3.2 M LB PROPELLANT



- FLEXIBLE SEAL NOZZLE TVC SYSTEM
- MONOLITHIC OR SEGMENTED CONSTRUCTION



- 6 MOTOR DEV/PFRT PROGRAM
- COST BASIS - AEROJET GENERAL CORP NASA STUDY NAS7-513 DTD 4/67
- SRM CONTRACTOR DEV COST $83 M (1970 DOLLARS)

Figure 3-57. 260-Inch SRM Development Cost

Table 3-10. SRM Booster Cost (Millions of Dollars)*

Item	Development	Production	Total
Motor and static test	35.0		
Engineering and development	16.0		
Tooling and special test equipment	17.0		
Facilities and other	15.0		
Subtotal motor	83.0	103.0	186.0
Motor in-house	16.6		16.6
Structures	21.0	40.8	61.8
Roll control	6.0	4.3	10.3
Avionics	13.2	2.9	16.1
Stage tooling	36.0		36.0
Two ground test articles	28.2		28.2
Installation and assembly	2.0		2.0
Other design, development, test, and evaluation	56.1		56.1
Total	262.1	151.0	413.1

*Single 260-inch

The limitations of the present Saturn V LUT - C/T system influences the mate-erect technique. The C/T has a useful load capability of approximately 12.5 x 10^6 pounds. The total weight of the LUT plus Saturn V vehicle in the transport mode is approximately 12.3 x 10^6 pounds; therefore, the vehicle weight cannot appreciably exceed the current Saturn V weight—about 500,000 pounds. All the SRM configurations exceed this value. Two alternatives considered are:

1. Design a new LUT that can transport SRM-configured vehicles approaching 6 x 10^6 pounds gross weight; or

2. Modify the present Saturn V LUT by "splitting" it into two parts such that the loading limits of the C/T will not be exceeded.

Sufficient study has established the feasibility of splitting the LUT aft of Girder G14. The launch platform portion will weigh approximately 5 x 10^6 pounds, which will allow transport of vehicles weighing over 6 x 10^6 pounds. The umbilical portion (including most of the GSE and vehicle support systems) will weigh approximately 7 x 10^6 pounds and can be permanently installed on the launch pad with relative minor support modification.

Review of prior studies (Ref. 1) recommend erecting SRM's by overhead hoisting as opposed to other techniques. Movement of monolithic SRM should be reduced to the absolute minimum. Accordingly, the handling technique developed is shown in Figure 3-58. The SRM is lifted and rotated to the vertical directly from the delivery barge. The "split" LUT is driven under and the suspended SRM subsequently installed on the LUT. The SRM is transported to the VAB for mating with the orbiter.

Transfer to the launch pad is in the vertical attitude. The stiffness of the SRM should obviate the requirement for sway dampers. The technique differs significantly from current Saturn V procedures only in that the vehicle ground service connections must be accomplished at the launch pad rather than in the VAB. This is somewhat offset by elimination of launch pad to LUT interface connection requirements since the umbilical portion of the LUT is permanently installed at the launch pad.

The SRM erection facility could be located at the launch pad; however, this possibility was eliminated for the following reasons:

1. Launch pad modifications would be extensive and hence very costly;

2. Orbiter erection capability, presently available within the VAB, would have to be provided;

3. The orbiter mate cycle will be sensitive to weather conditions;

4. The utilization of the facilities as the program phases into the Gen 2 reusable booster becomes severely limited and awkward.

3.10.6 Facility Requirements and Ground Operations—260-Inch SRM

The sequence of operations for assembly of the 260-inch monolithic SRM was described in Section 3.10.5. Checkout of the booster will be accomplished on the LUT positioned in the VAB. After booster checkout is completed, the orbiter will be rolled into the mating bay and positioned for mating by use of the VAB crane system. From completion of mating until launch, all ground checkout activities are paced by the orbiter vehicle. Significant booster activities after mating will include installation of ordnance, participation in avionics overall test, and monitoring of electrical circuitry for undesirable EMI conditions. A timeline of the activities described is presented as Figure 3-59. Figure 3-58 is a sketch presenting the booster operations described above. Because the SRM is an expendable booster and does not require extensive servicing as a liquid booster does, an analysis was made of the deferrals for support equipment and operations

Reference 1: NASA report CR-72757, "Study of Storage and Handling of the 260" Solid Rocket Motor" - by Aerojet Solid Propulsion Company.

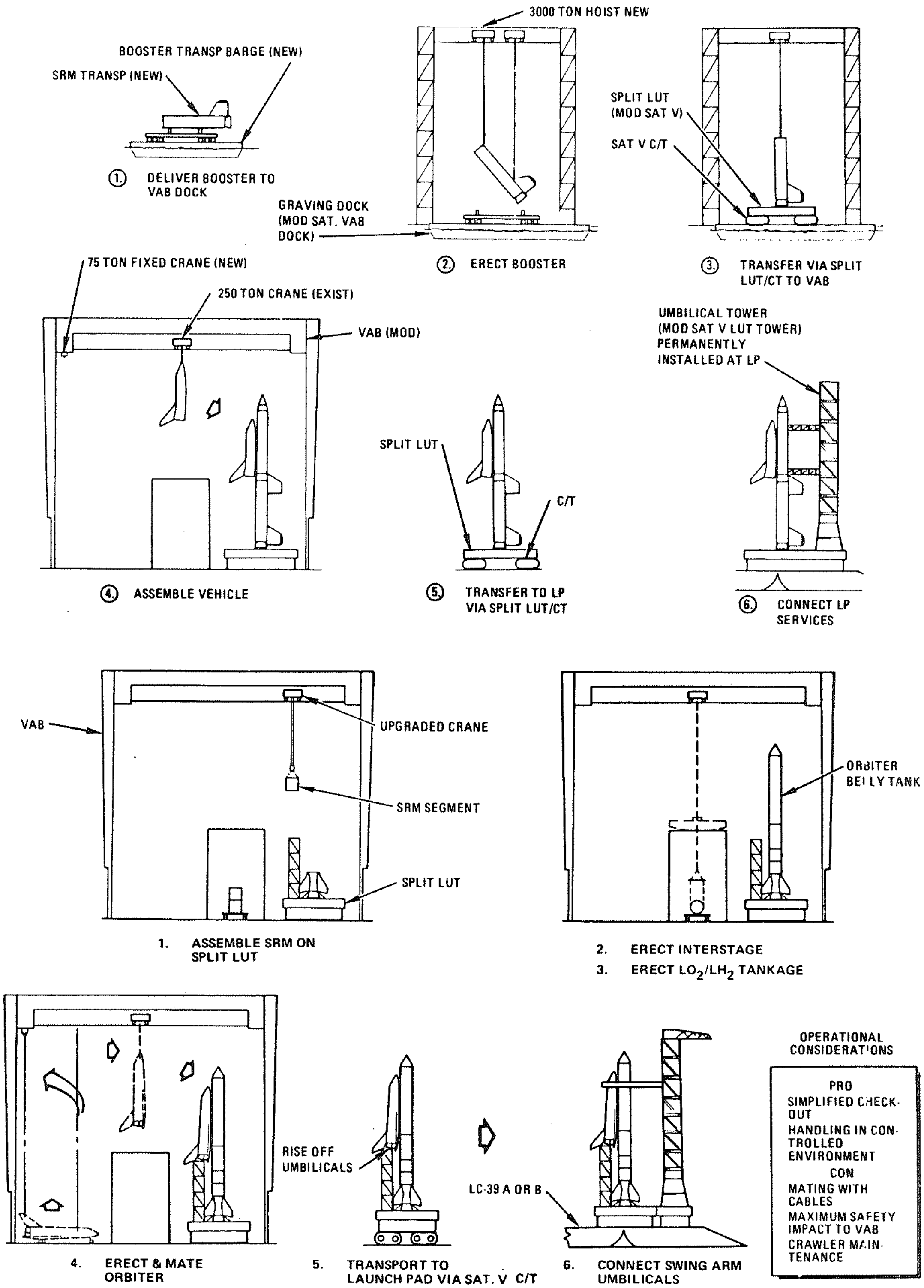


Figure 3-58. Mate/Erect Cycle, 260-Inch SRM

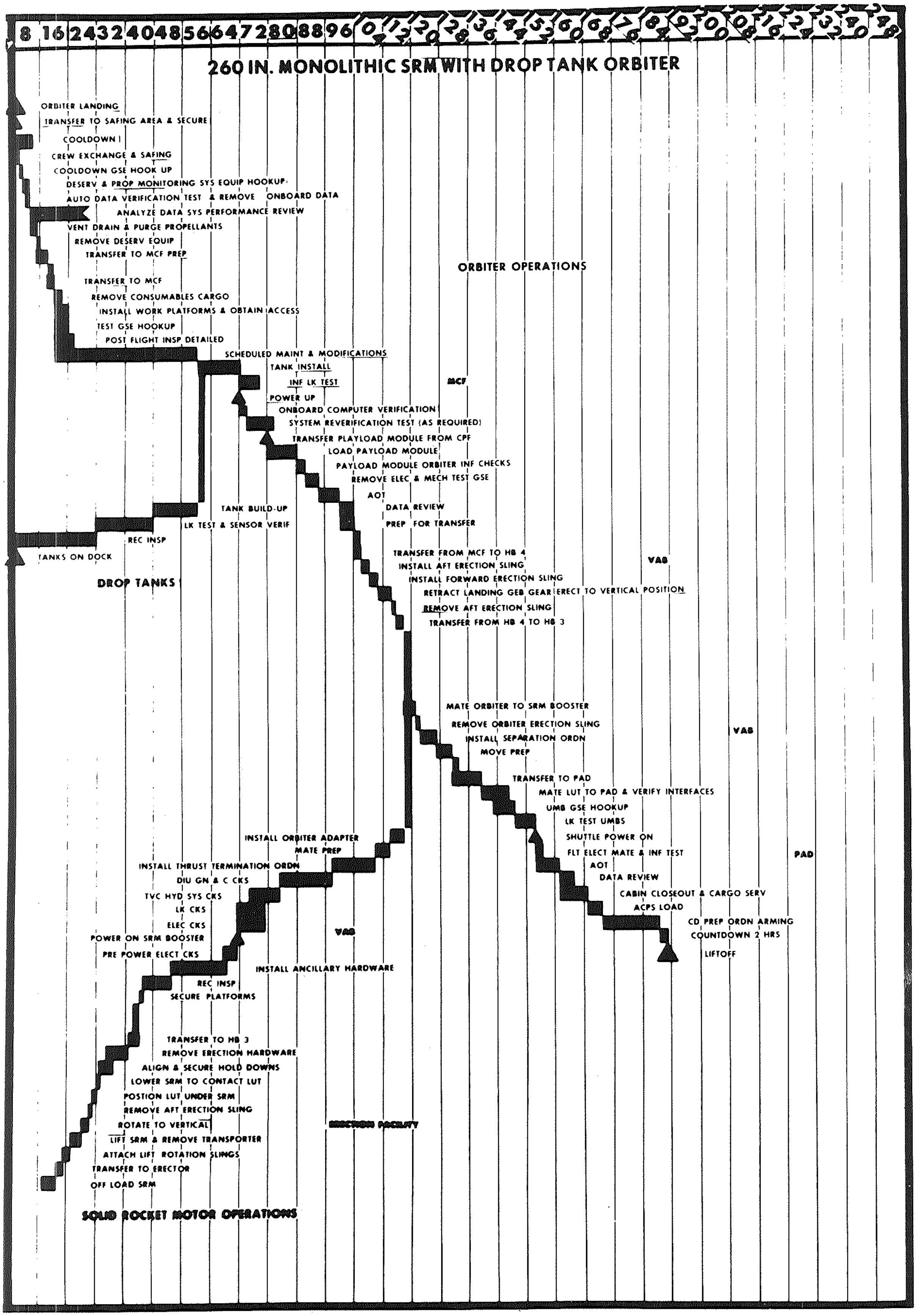


Figure 3-59. 260-Inch Monolithic SRM With Drop Tank Orbiter

costs that might be achieved using an interim booster. This analysis is summarized in Table 3-11.

Modification to the LUT was described in Section 3.10.5, modification to the barge turn basin and an extension to the crawlerway to provide for moving the split LUT to the dock will be required for this SRM configuration. A new bridge crane structure with a 3000-ton and a 1500-ton crane will also be required to erect and position the 260-inch monolithic SRM on the LUT. The split LUT concept will require additional support columns to be built at the launch pads in order to permanently install the tower portion of the existing LUT. Modifications to the VAB would be limited to the addition of a 75-ton crane in the VAB mating bay.

An analysis of the facilities and facility modifications that can be eliminated or reduced as compared to the required additions is presented in Table 3-12.

Table 3-11. Support Equipment and Operations Costs, Interim Booster System, 260-Inch SRM

Support Equipment	
• Items deferred until generation 2	
Reusable booster servicing, handling and checkout equipment	-$293 million
• Items added:	
SRM handling, servicing and checkout equipment	+$120 million
Operations	
• Items deferred until generation 2	>Costs of conducting total space shuttle operations to first manned orbital flight reusable booster
Reusable booster maintenance, servicing, checkout, launch, and flight operations	
• Items added:	
SRM handling, assembly, servicing, checkout, and launch operations	-$61.7 million

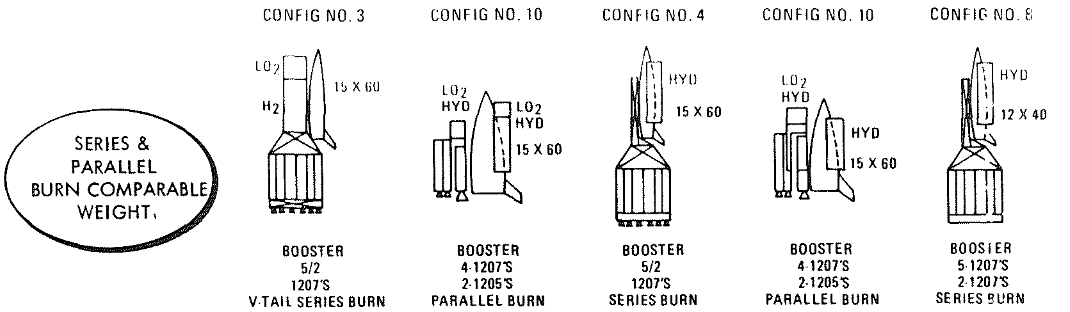
Table 3-12. Facility Reductions Versus Required Additions
Interim Booster System, 260-Inch SRM

Facility	(millions)
• Items deferred until generation 2	
Cryogenic service system modification	\$ 6.6
Landing facility reduced	4.0
Launch Pad B modification	12.0
LUT modifications for reusable booster	11.0
Maintenance, checkout, and mating facility modification reduced	12.0
Communications and data cabling reduced	3.0
Miscellaneous support facilities reduced	2.0
Flight crew training facility reduced	15.0
Central data processing equipment reduced	1.0
Design, supervisary, and administrative costs reduced	6.4
Activation costs reduced	20.0
Total	-\$93.0
• Items added	
LUT modification (structure splitting)	\$14.0
SRM erection tower at VAB dock	40.0
Total	+\$54.0

3.10.7 120-Inch and 156-Inch SRM Cluster Systems and Configuration

A large number of SRM cluster configurations were studied. The most feasible ones are described in Figure 3-60. The Configuration 10 series, parallel burn of orbiter and booster at lift-off, permitted the use of one less 120-inch SRM in the first stage boost but added the complexity of carrying first-stage propellant and cross-feeding it to the orbiter during first-stage boost. This approach was abandoned in favor of Configuration 4, series burn of a two-stage 120-inch SRM booster; the final configuration selected for the external hydrogen tank (internal LO_2) is shown in Figure 3-61.

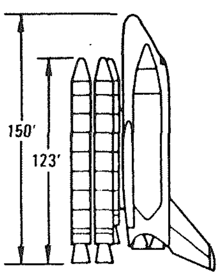
A 5/2 120-inch SRM cluster provided excess payload capability for an EOHT configuration with 15 x 60-foot cargo bay (Figure 3-62). By going to a 40-foot cargo bay orbiter, one first stage rocket could be deleted and a 4/2 120-inch SRM cluster was selected for the Gen 1 configuration. This is shown in Figures 3-63 and 3-64.



	CONFIG NO. 3	CONFIG NO. 10	CONFIG NO. 4	CONFIG NO. 10	CONFIG NO. 8
ORBITER GEN 2 SIZE	EXT LO_2 HYD	EXT LO_2 HYD	EXT HYD	EXT HYD	EXT HYD
DWG NO.	-0081	-0081	-0056	-0056	-0050A
PAYLOAD KLB	69 (65)	59.5 (65)	53.1 (65)	89.9 (65)	65 5 (45)
OLOW MLB	1.203	2.051 *	1.209	2.024 *	1.194
DRY WT KLB	226	226	240	240	223
ASCENT PROP. KLB	868	868	873	873	864
OMS PROP. KLB	15.7	20.1	17.8	15.1	17.4
BOOSTER - 1ST STAGE	5-UTC-1207	4-UTC-1207	5-UTC-1207	4-UTC-1207	5-UTC-1207
BLOW MLB	3.450	2.724	3.405	3.046	3.420
F_{SL}/MOTOR	1.656	1.656	1.656	1.656	1.656
ΔV_{IDEAL} FPS	5,541	5,575	5,633	5,500	5,604
BOOSTER - 2ND STAGE	2-UTC-1207	2-UTC-1205	2-UTC-1207	2-UTC-1205	2-UTC-1207
BLOW MLB	1.400	1.054	1.374	1.508	1.394
F_{VAC}/MOTOR	1.920	1.260	1.920	1.260	1.920
ΔV_{IDEAL}	5,579	7,370	5,637	7,180	5,625
STAGING VELOCITY	6,350	7,900	6,430	7,490	6,370
GLOW MLB	6.053	5.829	5.989	5.802	6.009
T/W AT LIFTOFF	1.368	1.348	1.383	1.338	1.378

* INCLUDES 1ST STAGE LOX-HYD

Figure 3-60. Interim 120-Inch SRM Integrated Vehicle Options and Sizes



SYSTEM PARAMETERS		GEN 1		GEN 2	
CARGO BAY		15 FT X 60 FT		15 FT X 60 FT	
PAYLOAD	(KLB)	65K DUE E		40K POLAR	
OMS ΔV	(FPS)	900		650	
		BOOSTER	ORBITER	BOOSTER	ORBITER
SYSTEM GLOW	(MLB)	5.942		3.593	
T/W AT LO		1.35	1.16	1.30	1.20
STAGE GROSS WT	(MLB)	4.805	1.155	2.471	1.123
MAIN PROP. WT	(MLB)	3.000	0.832	2.016	0.832
OMS PROP. WT	(KLB)		22.9		16.5
FLYBACK FUEL	(KLB)			37.8	
FLYBACK RANGE	(N MI)			182.0	
*STAGE DRY WT	(KLB)	587.9	191.2	392.5	191.2
**STAGE LANDING WT	(KLB)		234.1	399.3	234.1
ORBITER EXT TANK WT	(KLB)		19.7		19.7
MAX $\bar{q}$	(PSF)	595		491	
STAGING V_R	(FPS)	7,932		6,838	
	(DEG)	7		16	
MAIN ENG F_{SL}/ENG	(KLB)	1,600		389.3	
F_{VAC}/ENG	(KLB)	1,820	447.5		447.5
NO. OF MAIN ENGS		5/2	3	12	3

*PAYLOAD & EXPENDABLE TANKS NOT INCLUDED

**INCLUDES PAYLOAD

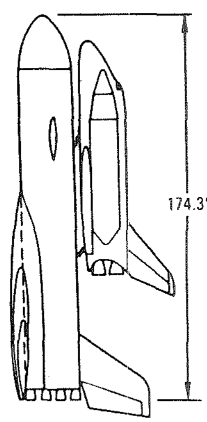
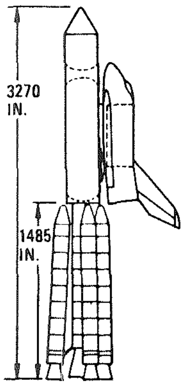
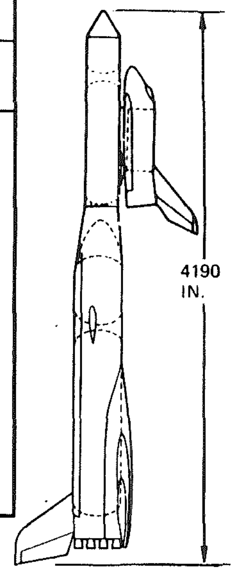


Figure 3-61. Integrated EHT System Description With 120-Inch SRM Booster

SYSTEM PARAMETERS		GEN 1			GEN 2	
CARGO BAY		15 FT X 60 FT			15 FT X 60 FT	
PAYLOAD	(KLB)	74 DUE E			40 POLAR	
OMS ΔV	(FPS)	900			650	
		BOOSTER		ORBITER	BOOSTER	ORBITER
		1ST	2ND			
		5X1207	2X1207			
SYSTEM GLOW	(MLB)	5.855			3.374	
T/W AT LO		1.37			1.3	
STAGE GROSS WT	(MLB)	3.404	1.383	1.068	2.35	1.024
MAIN PROP. WT	(MLB)	3.0	1.2	0.754	1.911	0.754
OMS PROP. WT	(KLB)			22.6K		12.9
FLYBACK FUEL	(KLB)				39	
FLYBACK RANGE	(N MI)				195	
*STAGE DRY WT	(KLB)	405	183	148	376	148
**STAGE LANDING WT	(KLB)			177	421	192
ORBITER EXT TANK WT	(KLB)			48		48
MAX $\bar{q}$	(PSF)	639			488	
STAGING V_R	(FPS)	3663	8567		7029	
γ_R		33	6.3		16	
MAIN ENG F_{SL}	(KLB)	8000			366	
F_{VAC}	(KLB)		3800	420	401	420
NO. OF MAIN ENG		5	2	3	12	3

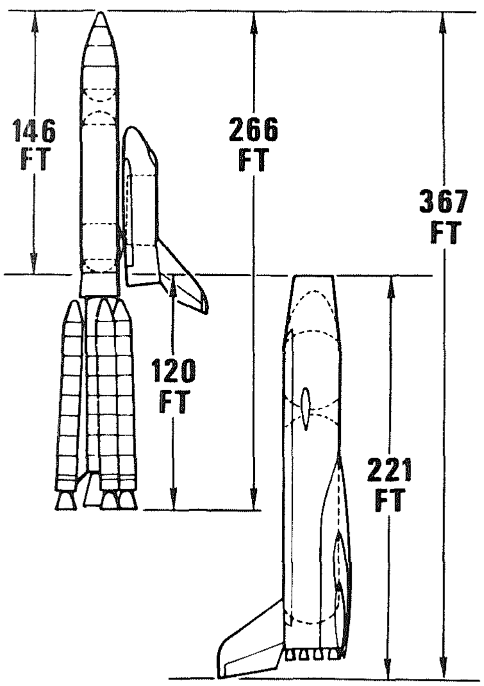




*PAYLOAD & EXPENDABLE TANKS NOT INCLUDED

**INCLUDES PAYLOAD

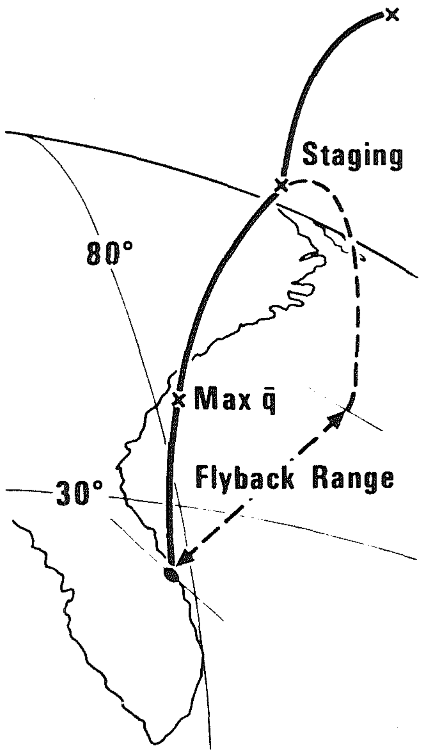
Figure 3-62. Integrated EOHT System Description With 120-Inch SRM Interim Booster



	Gen 1 B	Gen 1 Orb	Gen 2 B	Gen 2 Orb
Stage				
Gross Wt (m lb)	4.115	1.025	2.35	1.024
Dry Wt (k lb*)		140	376	148
Ext Tank				
Wt (k lb)		48		48
Main Engines				
Number	4/2	3	12	3
Thrust (k lb)	1600	420	366**	420
Sys GLOW (m lb)	5.14		3.37	
P/L (Dn Wt)(k lb)	45 (25) Due E		40 (40) Polar	
OMS ΔV (fps)	900		650	

* Payload & Exp Tanks not Included
** Sea Level

Figure 3-63. 120-Inch SRM Interim Booster (15 by 40 and 15 by 60 Feet)



	Gen 1		Gen 2	
P/L (Dn Wt) (k lb)	45 (25) Due E		40 (40) Polar	
OMS ΔV (fps)	900		650	
	B	Orb	B	Orb
Max $\bar{q}$ (psf)	455	455	488	488
Staging V_R (fps)	7401	7401	7029	7029
Staging γ_R (deg)	8	8	16	16
Flyback (n mi)			195	

Figure 3-64. 120-Inch SRM Interim Booster (15 by 40 and 15 by 60 Feet), 3-Engine Orbiter

Several 156-inch SRM clusters were studied (see Figure 3-65). The two-stage configurations did not match the payload requirement and a reasonable lift-off T/W value, so the two-stage approach was abandoned in favor of a single-stage cluster. The three-unit cluster could be considered as an alternate to the 260-unit SRM. Its thrust time curve would be tailored to a gradual thrust decay at maximum $q\alpha$, a build-up to a 3-g limit, and then thrust tail-off. The most satisfactory matching of maximum $q\alpha$ constraint, payload capability, and booster staging velocity constraint resulted in the performance shown in Table 3-13.

3.10.8 Development Status

Development of a five-segment, 120-inch-diameter SRM was initiated by the United Technology Center (UTC) Division of United Aircraft (UA) Corporation in late 1962 under contract to the USAF for use on the Titan IIIC program. A total of nine development and five preliminary flight rating test (PFRT) static test firings were successfully completed by 1965. Since that time, the motors have performed successfully on 17 Titan IIIC launches (a total of 34 SRM's). The five-segment motor (UA 1205) has an initial sea level thrust of 1.147 million pounds and contains 424,000 pounds of polybutadiene acrylonitrile (PBAN) propellant. A liquid injection thrust vector control (LITVC) system has been utilized.

CONFIGURATION		2 & 1 - 156 IN. CLUSTER			3 & 1 - 156 IN. CLUSTER		
CONCEPT		3-2			3-2		
GENERATION	(H, HO)	1			1		
CARGO ENVELOPE		15 FT X 60 FT			15 FT X 60 FT		
ORBIT		100 N MI - 28.6°			100 N MI - 28.6°		
PAYLOAD	(KLB)	65			65 PLUS		
OMS ΔV	(FPS)	900			900		
ABORT?		YES			YES		
		BOOSTER 1ST STAGE	BOOSTER 2ND STAGE	ORBITER	BOOSTER 1ST STAGE	BOOSTER 2ND STAGE	ORBITER
SYSTEM GLOW	(MLB)	6.05			7.65		
T/W AT LO		1.1	1.4	1.32	1.32	1.4	1.32
STAGE GROSS WT	(MLB)	3.22	1.62	1.21	4.83	1.61	1.21
MAIN PROPULSION							
PROPELLANT WT	(MLB)	2.81	1.4	0.868	4.21	1.4	0.868
DIAMETER	(INCHES)	156	156		156	156	
F_{SL}/MOTOR	(KLB)	3,410			3,410		
F_{VAC}/MOTOR	(KLB)		3,970	534.3		3,970	534.3
NUMBER		2	1	3	3	1	3
STAGE DRY WT	(KLB)	410	205	236	615	205	236

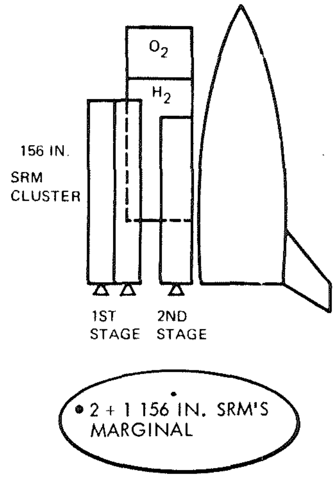


Figure 3-65. Interim 156-Inch SRM Integrated Vehicle Option and Sizes

Development of a seven-segment motor (UA 1207) was initiated in mid-1969, and four of eight planned development/PFRT static firing tests were successfully completed prior to termination of the MOL program in the third quarter of 1970. The UA 1207 had an initial sea level thrust of 1.397 million pounds and contained 593,000 pounds of PBAN propellant. The motor was of essentially the same construction as the UA 1205 and also utilized an LITVC system.

The 120-inch-diameter SRM proposed for space shuttle use will require a seven-segment configuration with modified motor ballistics (i.e., higher thrust and shorter burning time) in order to meet initial T/W requirements. The change in motor ballistics would be effected by a combination of higher propellant burning rate and higher motor operating pressure. The use of a flexible nozzle TVC system is also proposed on the basis of lower cost and higher stage mass fraction.

3.10.9 Booster Cost

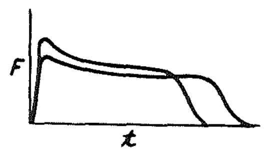
The 120-inch SRM costing was based on seven-segment motors already in development for the Grand Tour mission. A shorter burn time at greater lift-off thrust was considered since the propellant mix was only slightly modified for the change (see Figure 3-66). The costing (Table 3-14) was based on UTC 120-inch SRM cost data, but modified by addition due to system considerations for which NR would be responsible.

Table 3-13. Generation 1 J-2S Orbiter, 3-156-Inch SRM Cluster, Selected System Summary

System Parameters	Gen 1 Booster	Gen 1 Orbiter	137B Orbiter
Cargo bay		15 x 60	15 x 40
Payload (K lb)		(45) 35.4	(45) 57.6
OMS ΔV (fps)		900	900
Reference synthesis run - 9/7/71, Hour 15		Run 3	Run 1
System GLOW ($\overline{M}$ lb)	4.895		
T/W at LO	1.367		
Stage gross weight ($\overline{M}$ lb)	3.992	0.901348	
Main propellant weight, usable ($\overline{M}$ lb)	3.535	0.669363	
OMS propellant weight, usable (K lb)		15.044	13.207
Flyback fuel, usable (K lb)			
Flyback range (n mi)			
Stage dry weight (RV) (K lb)	438	117.709	93.320
Interstage weight (K lb)*	20		
Orbiter external tank weight (empty) (K lb)		50.861	50.824
Stage reentry weight (K lb)		140.526	120.899
Stage landing weight (K lb)		139.743	120.177
Max-Q (psf)	544.7		
Staging, V_i (fps)	8113		
h (K ft)	178.1		
γ_i (deg)	15.7		
q (psf)	~0		
Main engine, F_{SL} (K-lb ea)	2,230		
F_{VAC} (K-lb ea)	2,488	265	
$I_{SP_{VAC}}$		436	
Number of main engines	3	3	
Geometry			
Body wetted area (ft^2)		6440.5	4097.2
Body vol (ft^3)		27,668	17,465
Body L (ft)		112.92	82.3
Wing area (Theo) (ft^2)		3360.7	2404.7
Vert area (ft^2)		339.1	255.0
Orbiter burnout weight		181,583	161,217

*Dual plane separation - interstage weight included in booster gross weight but not in dry weight. Interstage separates with orbiter tank and is subsequently dropped.

- SEVEN SEGMENT SRM REQUIRED
- MODIFIED MOTOR BALLISTICS



- FLEXIBLE SEAL NOZZLE TVC SYSTEM
- CLUSTER HARDWARE INCLUDED
- COST BASIS-UNITED AIRCRAFT UTC 1207 SRM
- SRM CONTRACTOR DEV COST \$33$\bar{M}$ (1970 DOLLARS)

Figure 3-66. 120-Inch SRM Development Cost

Table 3-14. SRM Booster Cost*

Item	Development	Production	Total
Motor and static test	19.0		
Engineering and development	10.0		
Tooling and special test equipment	1.0		
Facilities and other	3.0		
Subtotal (motor)	33.0	129.4	162.4
Motor in-house	6.6		6.6
Structures	6.0	11.3	17.3
Avionics	13.2	2.9	16.1
Stage tooling	9.9		9.9
Two ground test articles	34.0		34.0
Installation and assembly	1.0		1.0
Other design, development, test and evaluation	26.0		26.0
Total	129.7	143.6	273.3

*Seven 120-inch

3.10.10 Mate and Erect 120-Inch SRM

The facility-LUT-C/T limitation discussed in Section 3.10.5 applies when considering methods for erecting segmented 120-inch SRM's. Several erection techniques were studied.

1. Erect at the launch pad.

2. Install the base structure and SRM and orbiter in the VAB on the LUT. Complete SRM assembly at the launch pad, utilizing the LUT hammerhead crane.

3. Erect in VAB on split LUT.

Alternates (1) and (2) are not recommended for the same reasons advanced against on-pad assembly in Section 3.10.5.

Alternate (3) is recommended and is depicted in Figure 3-58. The SRM clusters are built up segment by segment, followed by erection and mating of the orbiter to the booster assembly.

Transport to the launch pad is with the vehicle in the vertical attitude. It is assumed the cluster is sufficiently rigid that a sway damper system is not required. Orbiter ground system interfaces must be connected after the vehicle/launcher is installed at the launch pad.

3.10.11 Facility Requirements and Ground Operations for 120-Inch SRM

A study was made of the preflight operations required to assemble and prepare a 120-inch SRM booster and an orbiter with external propellant tank(s) for launch. Figure 3-67 presents a schedule of these orbiter and booster activities. The 120-inch SRM booster/orbiter configuration will require a split LUT concept, as presented in Figure 3-68, because of the weight of the SRM. The 120-inch solid motor will be shipped in segments, either by rail or by barge, and moved directly into the VAB on its transporter. Assembly of the SRM segment will be done on the LUT in the mating bay and booster inspection and checkout will be accomplished there. After the orbiter and booster vehicles are checked out individually, the orbiter will be brought into the mating bay, mated with its external full tank, and then mated to the booster. The assembled vehicle, on its LUT, will be transferred to the launch complex using the Apollo/Saturn crawler.

An analysis of support equipment and operations that differ from the reusable orbiter/booster baseline configuration was conducted, and a summary of the results is presented in Table 3-15.

Because of the reduction in, liquid propellant quantity, flight crews, data quantity, and checkout stations resulting from the use of a solid motor-type booster, some planned facility additions and modifications can be deferred until the Gen 2 booster time period. A summation of these items and their associated costs are presented in Table 3-16 along with required facility modifications caused by having the interim expendable booster.

3.10.12 Comparison of Development Status

With the exception of the qualified propellant system, all other elements of the 260-inch-diameter SRM have completed feasibility demonstration but are considered to be in the developmental stage (i.e., ignition system, motor case, flexible-seal TVC, thrust termination, etc.). In contrast, all elements of the 120-inch-diameter SRM, with the exception of the flexible seal TVC system and the thrust termination system, are considered to be qualified by virtue of identity or similarity to those components previously qualified in the UA 1205 (Titan IIIC) program. In both cases, however, a development/PFRT static test program of similar magnitude is required to demonstrate overall motor system performance. The technical risks are somewhat greater in the case of the 260-inch-diameter SRM, since the

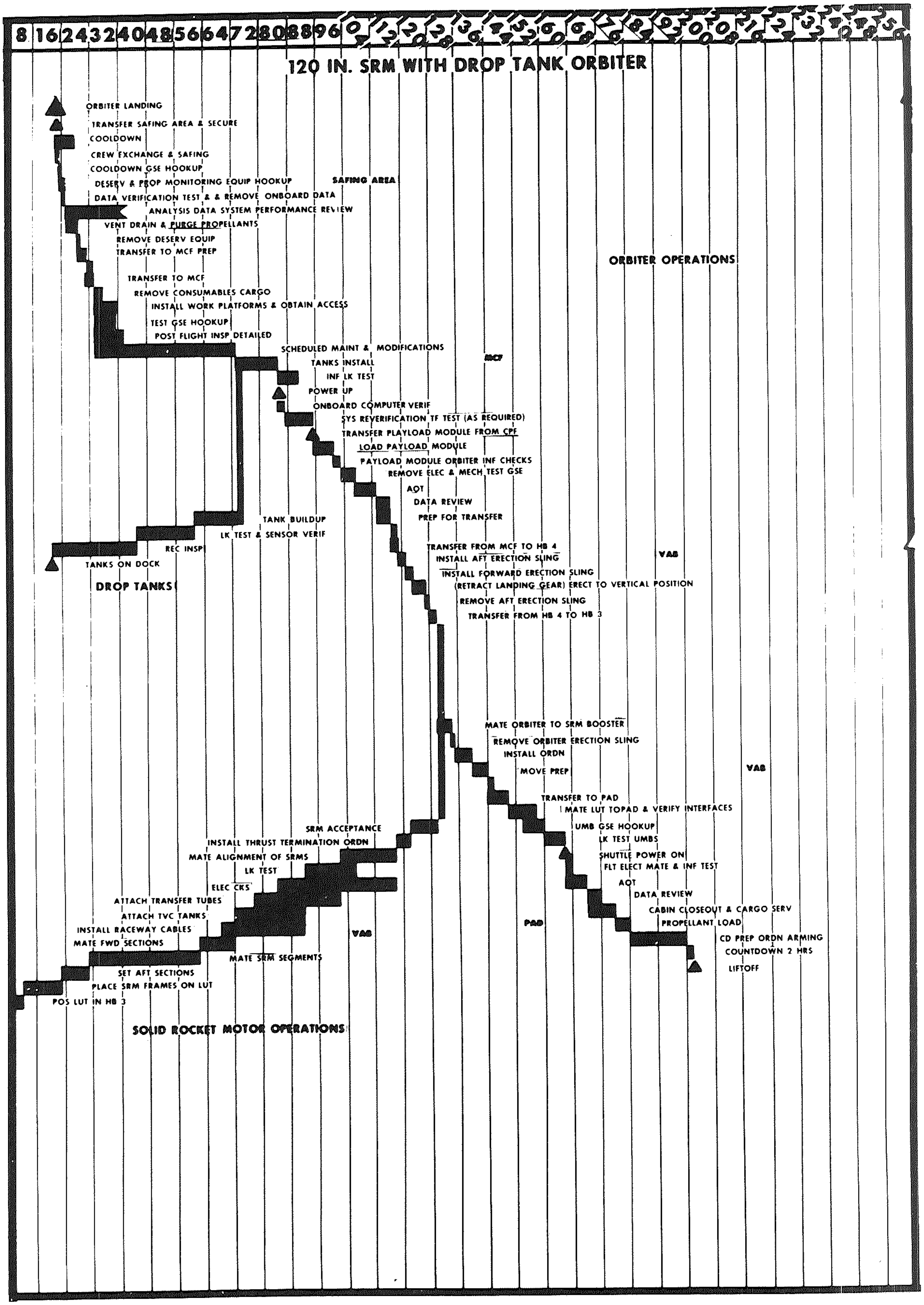


Figure 3-67. 120-Inch SRM With Drop Tank Orbiter

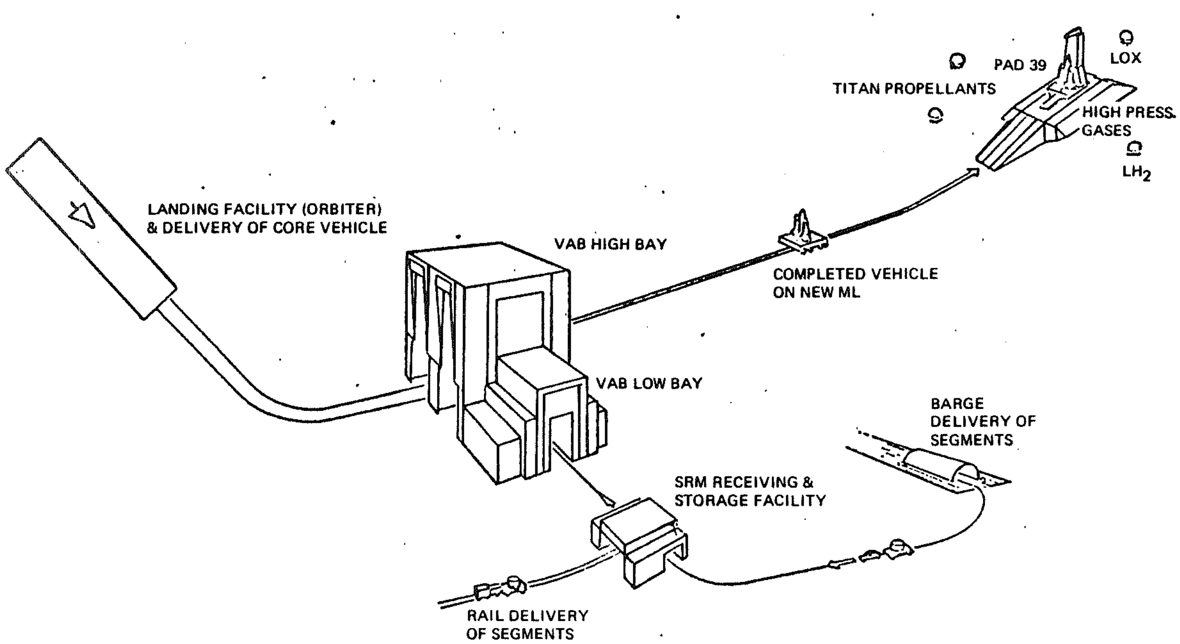


Figure 3-68. 120-Inch SRM Cluster/Orbiter Processing Concept, SRM Buildup in VAB

Table 3-15. Support Equipment and Operations Costs, Interim Booster System, 120-Inch SRM

Support equipment	
Items deferred until Gen 2:	
Reusable booster servicing, handling, and checkout equipment	-$293 million
Items added:	
SRM segment handling, servicing, and checkout equipment	+$120 million
Operations	
Items deferred until Gen 2:	
Reusable booster maintenance, servicing, checkout, launch, and flight operations	Costs of conducting total space shuttle operations to first manned orbital flight reusable booster
Items added:	
SRM handling, assembly, servicing, checkout, and launch operations	-$54 million

previous program was terminated in the R&D phase whereas the 120-inch-diameter SRM (UA 1205) is currently in limited production. A higher degree of confidence may therefore be placed in the cost and schedule integrity of the required 120-inch-diameter SRM development/PFRT program. The 260-inch-diameter SRM booster system does offer an advantage of less complexity and a higher degree of reliability of operation in that clustering and staging are not required for the single-element booster.

3.10.13 Environmental Effects

Preliminary SRM environmental data have been obtained in two specific areas: (1) ground level acoustic vibration, and (2) the characteristics and behavior of the products of combustion.

Table 3-16. Facility Reductions Versus Required Additions, Interim Booster System, 120-Inch SRM

Facility	(millions)
Items deferred until Gen 2:	
Cryogenic service system modification	\$ 6.6
Loading facility reduced	4.0
Launch Pad B modification	12.0
LUT modifications for reusable booster	11.0
Maintenance, checkout, and mating facility modification reduced	12.0
Communications and data cabling reduced	3.0
Miscellaneous support facilities reduced	2.0
Flight crew training facility reduced	15.0
Central data processing equipment reduced	1.0
Design, supervisory and administrative costs reduced	6.4
Activation costs reduced	20.0
Total	-\$93.0
Items added:	
LUT modification (structure splitting)	\$14.0
Modify VAB crane to 500-ton capacity	4.0
Total	+\$18.0

Comparative acoustic vibration data is provided in Table 3-17. The estimated levels for a space shuttle SRM booster were generated on the basis of thrust or energy level only. The values given for the 120-inch-diameter SRM cluster are considered to be conservative in that the attenuation effects due to clustering have not been considered.

The characteristics and behavior of the products of combustion for both the 120-inch-diameter and 260-inch-diameter booster are equivalent in that the propellant systems are essentially the same. The prime area of concern is the large quantities of hydrochloric acid (HCL) in the exhaust products. Information was obtained from the Aerospace Corporation relative to the measurements taken during Titan IIIC launches. A ground cloud containing approximately 20,000 pounds of HCL is formed from the first 10 to 12 seconds of motor burning. The gases being extremely hot and buoyant rise quite rapidly and clear the ground in less than one minute while expanding to a diameter of approximately 1600 feet (HCL concentration of 100 ppm) within two minutes. In the normally unstable atmospheric

Table 3-17. SRM-Generated Acoustic Environment Maximum Overall Sound Pressure Level (18-10,000 Hz)

Motor/Stage	DB Level	Distance (feet)
Measured Data		
260-inch-diameter SRM*	130	~800
120-inch-diameter SRM (T-IIIC)	138	~800
S-IC	155	800-1500
Estimated Space Shuttle Booster Levels		
260-inch-diameter SRM	135	800
120-inch-diameter SRM (5)	165	800
*Motor fired with nozzle up		

conditions at the ETR, cloud diffusion occurs in 20 to 30 minutes. Within five miles of the source, the concentration of HCL is reduced to 5 ppm. Plume dissipation is so rapid that attempts at measurement of HCL concentration have not been very successful. For example, measurements taken on the ground within 100 feet of the launch pad throughout the launch cycle show no indication of the presence of HCL. To date, no launch restrictions have been placed on the Titan IIIC from the ETR.

The expected ground cloud from a typical space shuttle SRM booster would contain approximately 50,000 pounds of HCL. Estimating techniques for determining rates of diffusion are available through NASA Study NAS8-2145. However, based upon the previous Titan IIIC experience, no launch restrictions from KSC are anticipated because of the prevailing offshore wind and normally unstable atmospheric conditions at KSC.

3.10.14 Orbiter/Booster Compatibility

Figure 3-69 shows that the interim and reusable boosters are attached to and separate from the orbiter in a similar manner. No essential difference in the interface between the orbiter and boosters exists.

3.10.15 Ascent Control

Ascent control trade studies were conducted to assess and compare the control requirements and capabilities of a 260-inch SRM booster system with clustered 120-inch SRM booster systems. Control authority for the 260-inch SRM booster in pitch and yaw is derived by TVC of the single engine. This authority is large enough that vehicle stability and adequate flight path control

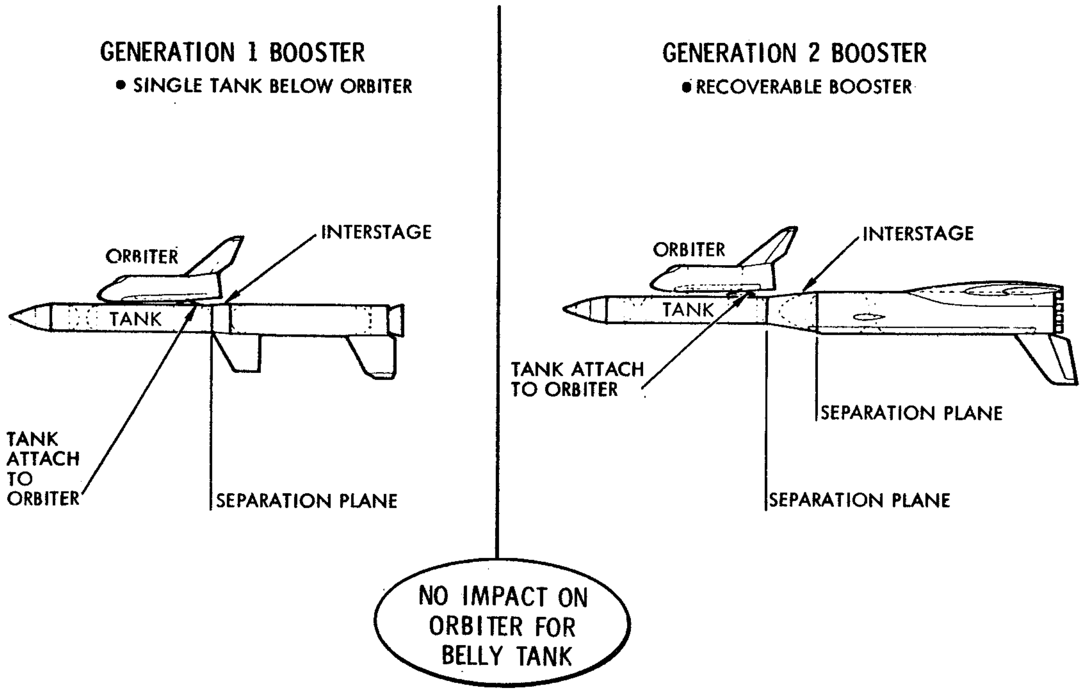


Figure 3-69. SRM Booster Impact on Orbiter

is maintained without a requirement for auxiliary aerodynamic surfaces on the booster. Even though the mated vehicle is statically unstable in both pitch and yaw, TVC deflection requirements do not exceed 5 degrees.

All 260-inch SRM configurations under investigation have a single engine nozzle. Roll control, therefore, cannot be achieved by TVC. During flight through the sensible atmosphere, roll stabilization and control is accomplished by using the orbiter elevons in a differential control mode. Due to the large Z-axis offset of the orbiter from the mated vehicle cg, however, aerodynamic rolling moments generated by sidewinds and gusts far exceed the hinge moment limits of the elevons. Redesign of the elevon actuators and elevon surface area to counter these aerodynamic moments is totally impractical. A ventral fin has, therefore, been added to the booster, sized to trim the orbiter rolling moment during flight through regions of high dynamic pressure. The elevons then provide the capability to handle mis-trims due to Mach number effects, and the authority for vehicle stabilization and maneuver control. The fin size requirement for each of the configurations evaluated is shown in Table 3-18.

In flight regions of low dynamic pressure such as pad liftoff and staging, reaction jets are installed on the tip of the ventral fin to provide roll control. They are sized by the crosswind magnitude and vertical gradient at liftoff. For the large orbiter (15 x 60 cargo bay), a 10,000-pound-thrust control authority is required.

All of the clustered 120-inch SRM boosters under investigation provide satisfactory control authority in all axes by TVC. No auxiliary aerodynamic surfaces on the booster are required. The maximum thrust vector deflections required are ±10 degrees in pitch and ±4 degrees in yaw or roll.

Table 3-18. Roll Trim Fin Size Requirements

Orbiter Cargo Configuration (feet)	Orbiter Engine Configuration	Roll Trim Fin Size (square feet)
15 x 60	3 high P_c	2000
15 x 40	3 high P_c	1600
12 x 40	1 high P_c	1200
15 x 40	1 high P_c	1240
260-inch SRM booster, EOHT orbiter		

3.10.16 Separation

The nominal separation sequence is shown in Figure 3-70. Prior to separation, the orbiter engines are prealigned to provide a pitch down moment on the orbiter. The sequence begins during booster engine thrust tailoff. When the sensed vehicle acceleration decays to 0.9 g, the orbiter engines are ignited. As the orbiter thrust builds up, the engine prealignment creates a pitch down moment on the mated vehicle. The booster engine is commanded to hold a zero pitch rate. Separation is initiated based on a time sequence which is set such that the orbiter T/W exceeds the booster T/W. The pitch down engine moment on the orbiter causes the overhang of the orbiter tail over the booster to pitch up and away from the booster. Adequate clearance is thus assured. The engine prealignment angle is sized so as to provide equal vertical accelerations of the orbiter and booster separation planes. The combination of (nearly) zero pitch rate of the mated vehicle and the (nearly) equal accelerations minimizes the tendency of interference transients at the separation plane. At separation the booster engines return to null. After a short time delay to allow a proper separation distance, the orbiter pitch control system is activated to recover from the separation transient.

In preliminary studies, the sequence described above was found to provide satisfactory separation both in and out of the atmosphere. Nominal staging, high $\bar{q}$ abort, and pad abort were investigated. It also is relatively insensitive to timing errors, thrust time history uncertainties, wind and gusts, cg location, and variations in the thrust vector control system. Some prestaging angle of attack control may be necessary at high $\bar{q}$.

3.10.17 Program Cost and Schedule Comparison - SRM's

A comparison of parallel versus phased development schedules can be made using Figures 3-71 and 3-72. The effect of using 120-inch SRM is that the interim booster decision can be delayed for 18 months longer than for 260-inch SRM (27 versus 9 months).

The cost comparisons are shown in Figure 3-73. The SRM interim booster recommendation is shown in Figure 3-74. The 120-inch SRM cluster is slightly superior from cost, schedule, handling, and transport viewpoints.

If the smaller orbiter (40-foot cargo bay) is considered, the 4/2 x 1207 SRM booster appears superior (see Figure 3-75).

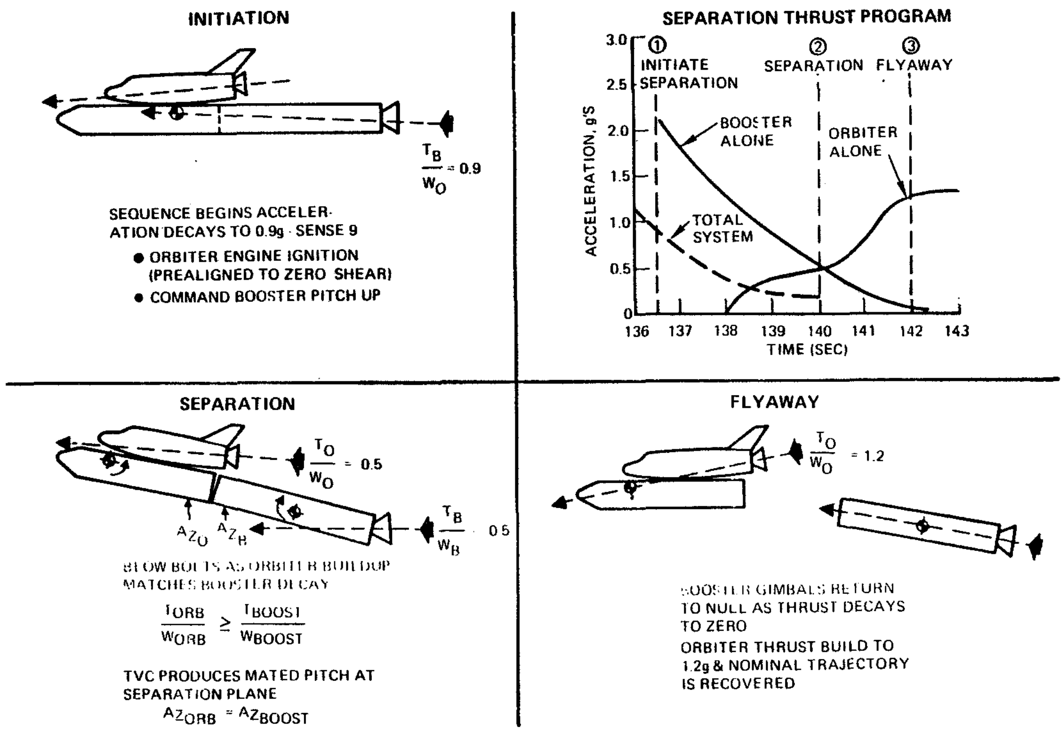


Figure 3-70. Separation

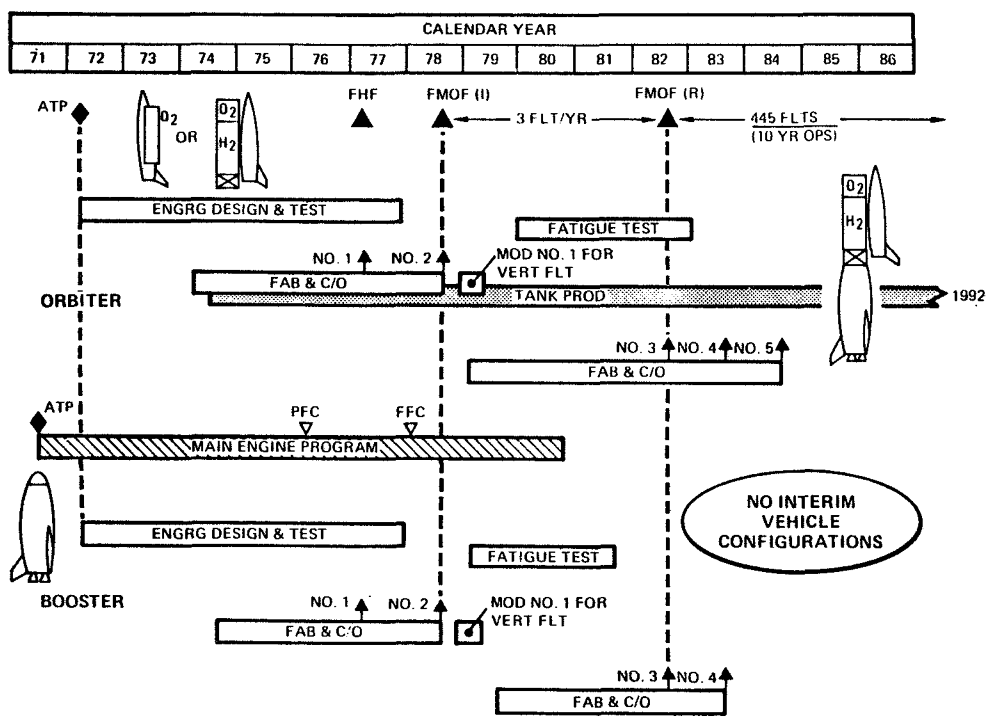


Figure 3-71. Program Schedule With Extended Initial Flights, Parallel Orbiter/Booster Development

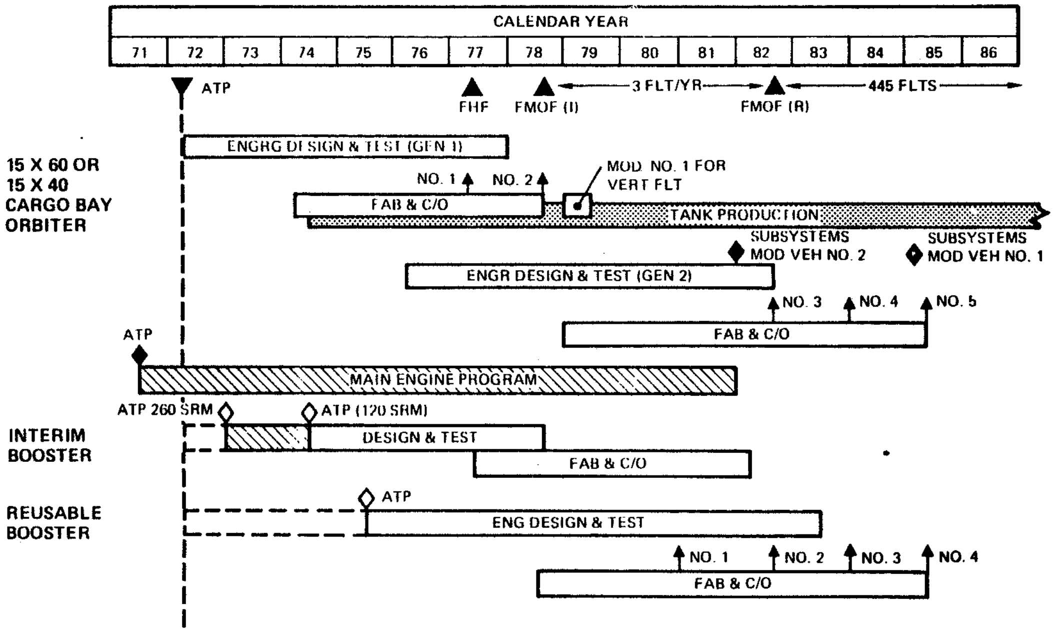
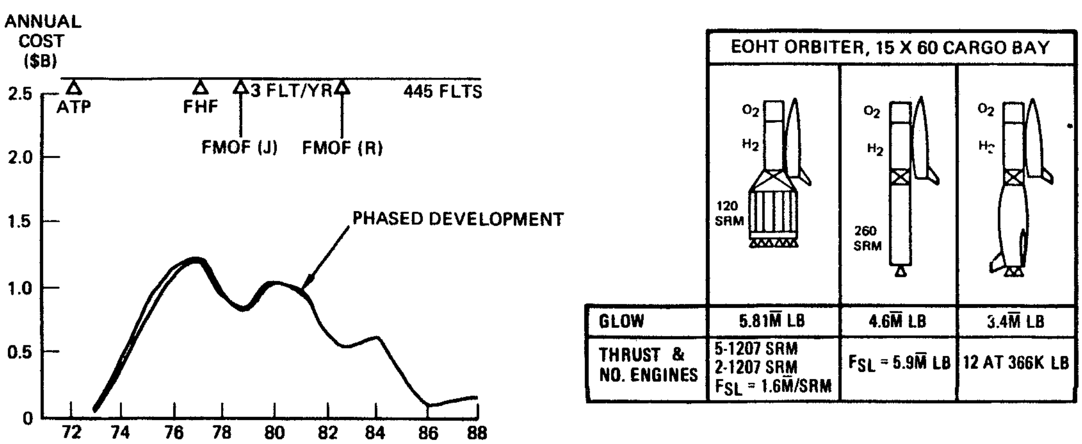
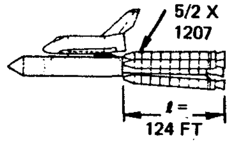
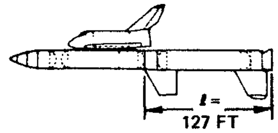


Figure 3-72. Program Schedule for Phased Booster Development, Interim SRM Booster/Reusable Booster



	PHASED DEVELOPMENT	
	120 CLUSTER	260 SRM
PEAK ANNUAL FUNDING/YEAR	1.22 B$/77	1.23/77
ORBITER DDT&E	3.18	3.18
REUSABLE BOOSTER DDT&E	2.77	2.77
EXPENDABLE BOOSTER DDT&E	0.14	0.25
TOTAL PROGRAM	10.71	10.83

Figure 3-73. Program Cost Comparison, Effect of SRM Selection (120-Inch Versus 260-Inch)

	 120 IN. SRM	 260 IN. SRM	OTHER BOOSTERS
COST (B$) SRM WT (MLB)	5.86 3.4/1.4	4.56 3.5	
COST (B$) PEAK ANNUAL PROGRAM	1.22 10.71	1.231 10.83	
FACILITIES IMPACT	• ASSY EQUIP./SPLIT LUT • LAUNCH FACIL COST = $23M	• ASSY EQUIP./SPLIT LUT • ACTIVATE MAN. FACILITY • LAUNCH FACIL COST = $84M • HANDLING DIFFICULTIES	
CONTROL	• 120 IN. SRMS ADEQUATE	• REQUIRES ORBITER ASSIST	
ABORT	• THRUST TERMINATION	• THRUST TERMINATION	
ORBITER DESIGN	• NO SIGNIFICANT IMPACT	• NO SIGNIFICANT IMPACT	
DISPOSAL	• LAUNCH CONSTRAINTS	• LAUNCH CONSTRAINTS	
DEVEL STATUS	• ON GOING	• R&D	
SCHEDULE IMPACT	• DEFER DECISION ≈ 27 MO	• DEFER DECISION ≈ 9 MO	
ENVIRON	• ACOUSTIC ACCEPTABLE • EXHAUST GASSES ACCEPTABLE	• ACOUSTIC ACCEPTABLE • EXHAUST GASSES ACCEPTABLE	

- ACCEPTABLE WITH ORBITER
- DECISION CAN BE DEFERRED
- CONTINUE STUDY
  - 120/156/260 SRMS

Figure 3-74. Interim Booster Recommendation

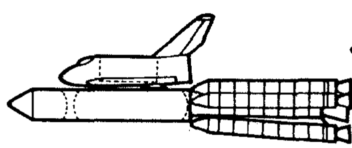
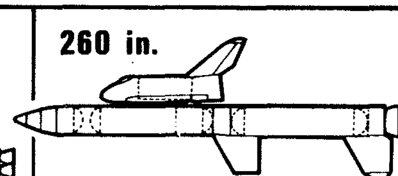
	4/2 X 1207 	260 in. 
SRM Wt (m lb)	**4.12**	**3.15**
Cost (B$) **Peak Annual** **Program**	**1.06** **11.15**	**1.08** **11.27**
Facil Impact	**Assy Equip/Split Lut** **Launch Facil Cost = $23m**	**Assy Equip/Split Lut** **Activate Mfg Facil** **Launch Facil Cost = $84m** **Handling Difficulties**
Control	**120 in. SRMS**	**Orbiter Assist**
Devel Status	**On Going**	**R&D**
Sched Impact	**Defer ≈ 27 Mo**	**Defer ≈ 9 Mo**

Figure 3-75. SRM Comparison Summary

3.10.18 LO_2/LH_2 Interim (Core) Booster

An early concept for an interim expendable booster system that would reduce early peak annual funding requirements and yet lead to a reusable system was the core booster. The core booster was conceived as the basic propulsion portion of the all-reusable booster, but omitting the wing and other aerodynamic surfaces needed for a flyback booster. The core components included the LH_2 and LO_2 tankage, the main propulsion feed system, high pressure space shuttle main engines, selected avionics, and a separation system. These elements of commonality are illustrated in Figure 3-76. After carrying out interim flights with the expendable core booster, the conversion would be accomplished to provide the fully reusable flyback booster for further operations.

Illustration of a tandem mounting arrangement for an orbiter with an HO tank (both hydrogen and oxygen in the external tank) is shown in Figure 3-77. For an orbiter equipped with either external hydrogen tanks mounted on the orbiter wings or HO tanks on the wings, a "piggyback" or parallel mounting is indicated. This is shown in Figure 3-78. These mounting arrangements and various vehicle combinations using the interim LO_2/LH_2 core booster are presented in Figure 3-79. In this figure the core concept and an "all-new" LO_2/LH_2 expendable booster (not readily convertible to a reusable system) are compared. For the same orbiter (with 15 x 60-foot cargo bay) the all-new booster is shown to be lighter, but the eventual requirement for a fully reusable booster precluded serious consideration of this case.

Another indication from Figure 3-79 is that the total program cost using the core booster is lower for orbiters with external HO tanks than for an orbiter with external LH_2 only.

Using the core booster, a study was made of the cost impact for having an early orbiter with a 40-foot cargo bay and a final orbiter with a 60-foot cargo bay. This is compared with a program in which the cargo bay is 60 feet from the beginning of the program. Figure 3-80 gives the cost comparison. The data show that comparable peak annual funding requirements would exist for the two approaches, but the total program cost for the growth orbiter (40-foot to 60-foot cargo bay) would be higher.

After completion of other interim expendable boosters, the total program cost was estimated for the several candidates. Figure 3-81 gives the comparison. Although the interim LO_2/LH_2 core booster offered the lowest gross weight system among all the candidates, the cost was higher than for several solid rocket motor interim boosters. The major cost difference was in the interim booster development cost. Because of this unfavorable trend, it was recommended that no further effort be devoted to the convertible core booster concept. NASA/MSC concurred with this recommendation.

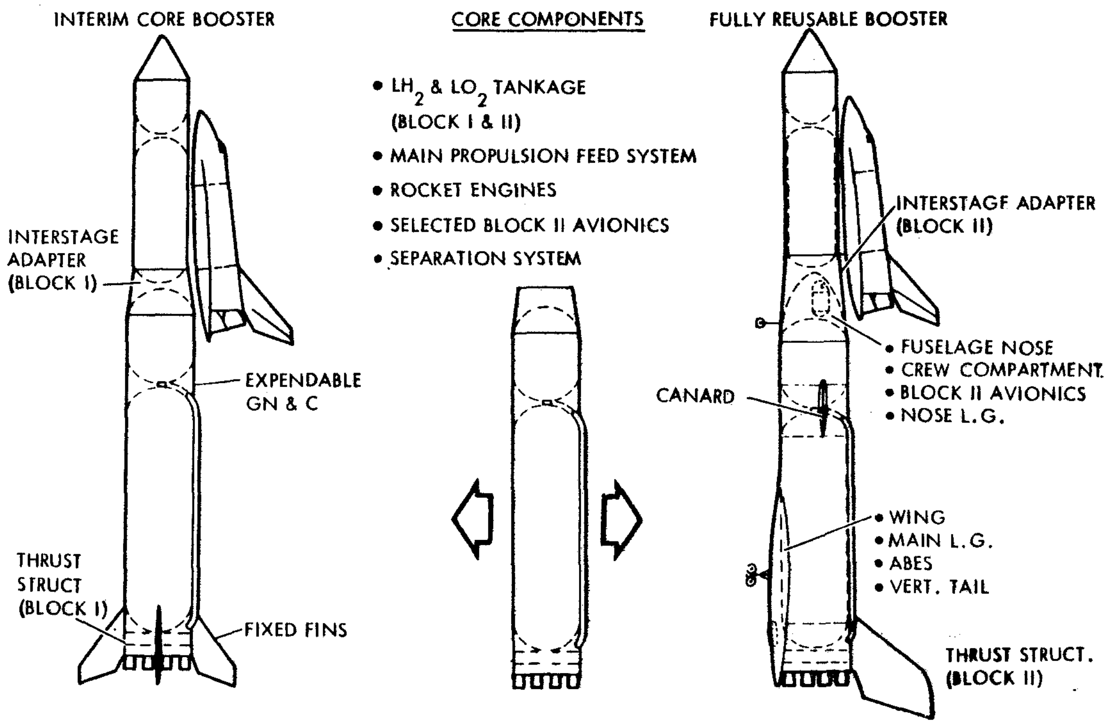


Figure 3-76. Basic Concept, Interim Core Booster

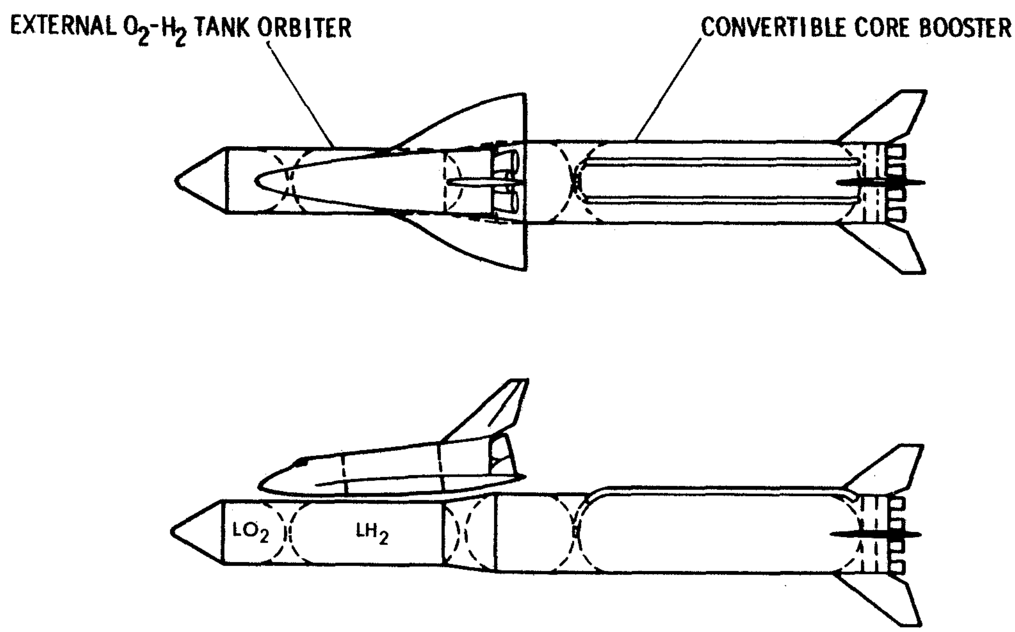


Figure 3-77. Interim Core Booster/Orbiter Tandem Configuration

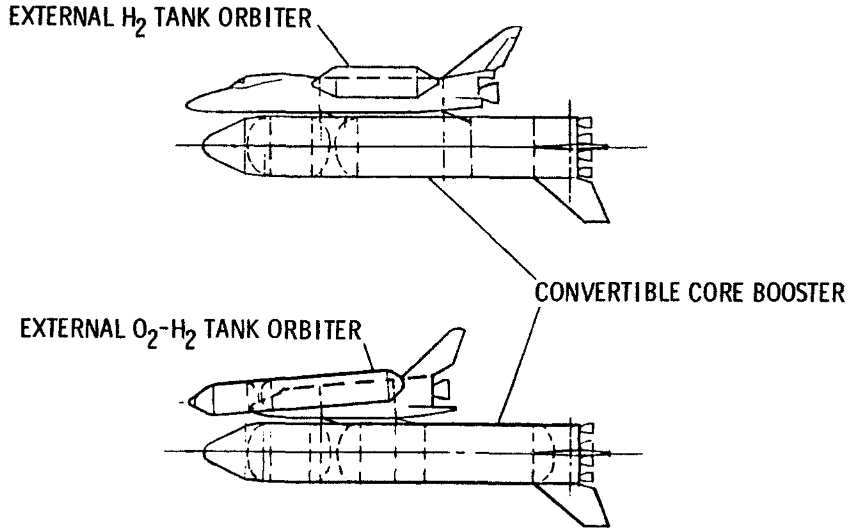
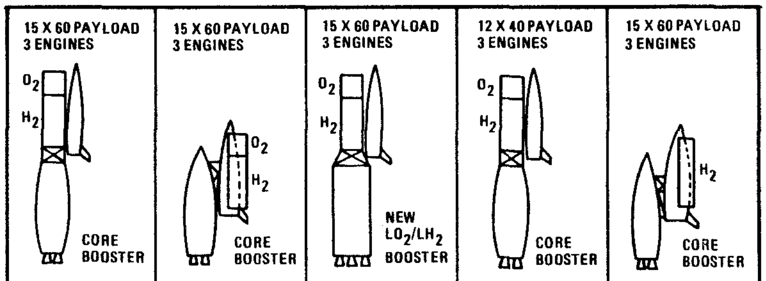


Figure 3-78. Interim Core Booster/Orbiter Parallel Configuration



	15 X 60 PAYLOAD 3 ENGINES	15 X 60 PAYLOAD 3 ENGINES	15 X 60 PAYLOAD 3 ENGINES	12 X 40 PAYLOAD 3 ENGINES	15 X 60 PAYLOAD 3 ENGINES
CONFIGURATION	2	2	2	7A	9
GLOW (MLB)	3.28	3.28	3.01	2.99	3.56
OLOW (MLB)	1.20	1.20	1.20	1.15	1.22
BLOW (MLB)	2.08	2.08	1.81	1.90	2.34
ORBITER W_{DRY} (KLB)	226	226	226	209	240
TOTAL W_{DRY} (KLB)	530	530	405	399	552
MPS F_{SL} (KLB)	507	507	489	503	467
NO. ENGS	8/3	8/3	8/3	8/3	10/3

COMMENTS

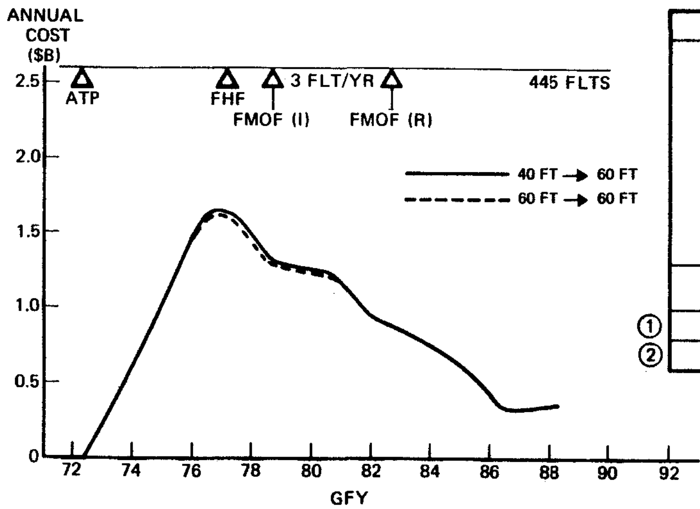
CONTROL

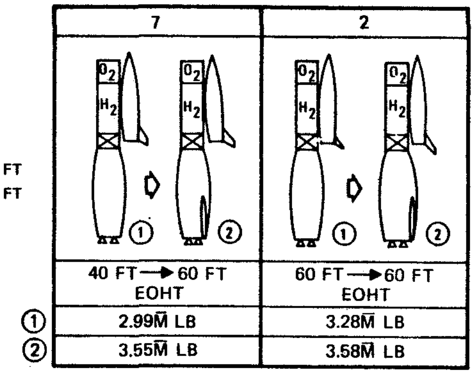
- ADD STABILIZING VERTICAL FIN (520 FT^2 - 1130 FT^2) TO CORE BOOSTER SYSTEM - CONTROL BOOSTER WITH ORBITER

FACILITIES

- NO SIGNIFICANT IMPACT TO GEN 2 FACILITIES FOR CORE BOOSTERS
- FACILITY MODIFICATIONS REQ FOR NEW LO_2/LH_2 BOOSTER

Figure 3-79. Interim LO_2/LH_2 Booster Integrated Vehicle Comparison

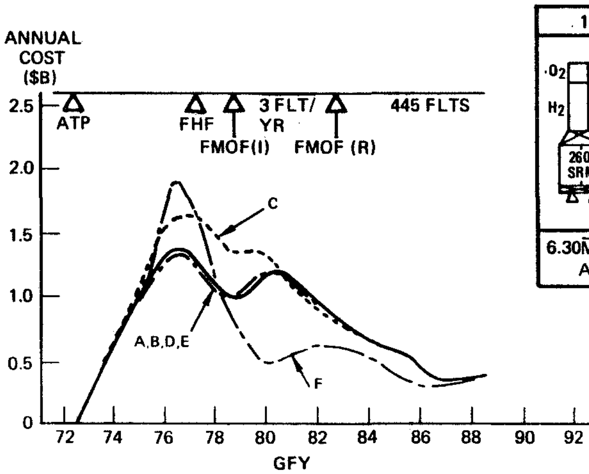


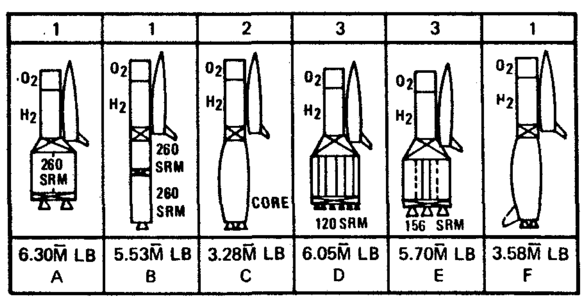


	40 FT → 60 FT	60 FT → 60 FT
PEAK ANNUAL FUNDING	1677/77	1652/77
TOTAL ORBITER DDT&E	3772	3437
INTERIM BOOST DDT&E	970	971
INTERIM OPS	300	300
FACILITY COST	486	483
FINAL OPS	1135	1132
TOTAL PROGRAM	15701	15455

- COMPARABLE PEAK ANNUAL FUNDING REQUIREMENTS
- HIGHER COST FOR GROWTH ORBITER

Figure 3-80. Cost Impact for 40-Foot Cargo Bay (For Interim Core Booster and EOHT Orbiter)





- HIGHER COST FOR CORE
- COMPARABLE COSTS FOR SRM'S

	A	B	C	D	E	F
PEAK ANNUAL FUNDING	1329/76	1374/76	1652/77	1330/76	1340/76	1866/76
TOTAL ORBITER DDT&E	3437	3437	3437	3437	3437	3455
INTERIM BOOST DDT&E	236	405	971	218	254	*
INTERIM OPS	250	250	300	265	265	400
FACILITY COST	548	548	483	506	506	485
FINAL OPS	1135	1135	1132	1134	1134	1068
TOTAL PROGRAM	14010	14171	15455	14103	14148	13626

* BOOSTER DDT&E–2700

Figure 3-81. Lowest Cost Interim Booster/EOHT Configuration (60-Foot Cargo Bay)

3.10.19 S-1C System

The S-1C stage of the Saturn V launch vehicle proves to be an attractive interim booster for the shuttle orbiter. The mated orbiter/S-1C is shown in Figure 3-82. The external HO belly tank of the orbiter system is mounted in tandem to the S-1C by means of a simple interstage structure.

The boost performance of the S-1C exceeds the baseline shuttle requirements by a significant margin. The orbiter constraints on loads, dynamic pressure, and axial acceleration are met by scheduling shutdown of booster engines and trajectory shaping. The full capability of the system exhibits the following payload capabilities.

Mission	Inclination (deg)	Payload (lb)
Due East	28.5	125K
Resupply	55	95K
South Polar	90	60K

The capability can be adjusted to the shuttle system design levels by means of off-loading propellant from either the orbiter or S-1C or by means of trajectory shaping. Orbiter propellant off-loading is preferred to S-1C propellant offloading because the resulting lower T/W makes the engine sequencing to control maximum dynamic pressure easier.

For ascent control considerations, the present S-1C fin size is adequate. The F-1 gimbal capability is increased to 6 degrees by repositioning the orbiter attachment to the stage structure, and the pitch and yaw plane is established in the plane of the fins in order to obtain the corner deflection capability in these planes. With these provisions, the vehicle is capable of control for a 95 percent omnidirectional design wind (75 M/sec) at the worst gust altitude (10 kM). Figure 3-83 shows the $q\beta$ and engine deflection for this case. The $q\beta$ in this case exceeds the allowable orbiter design, however, the incorporation of load limiting control policy and trajectory shaping will place the $q\beta$ within capability limits to be determined for the Generation 2 system.

Separation of the booster and interstage from the orbiter will be accomplished in a similar manner as the Generation 2 reusable booster. Studies of the reusable booster system have proved the feasibility of maneuvering the orbiter in a nose-down fashion in order to lift the tail overhang area away from the booster. For the S-1C, this scheme is enhanced by the use of the standard retro rockets to withdraw the booster in conjunction with the orbiter maneuver.

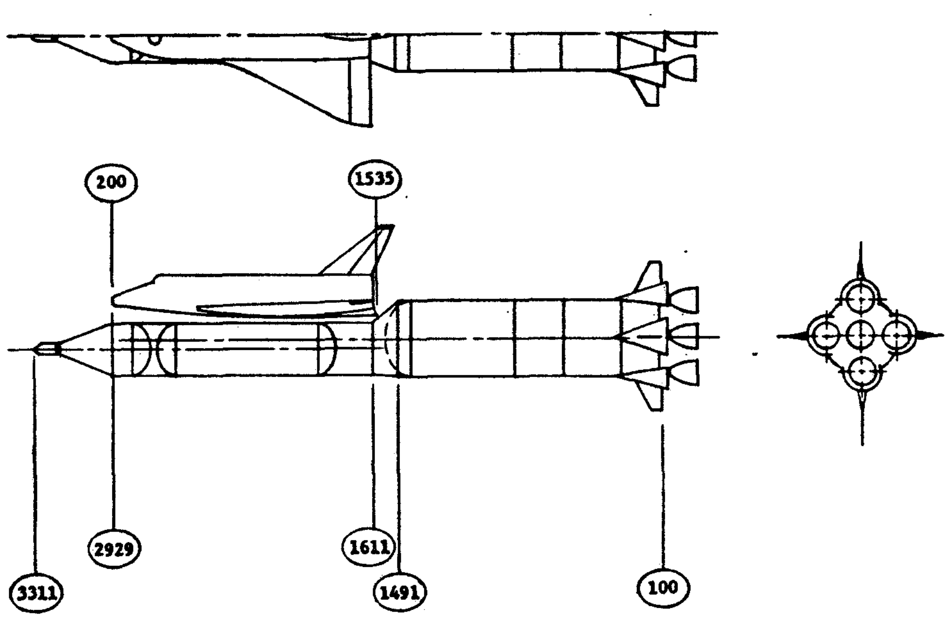


Figure 3-82. S-IC/NR Orbiter

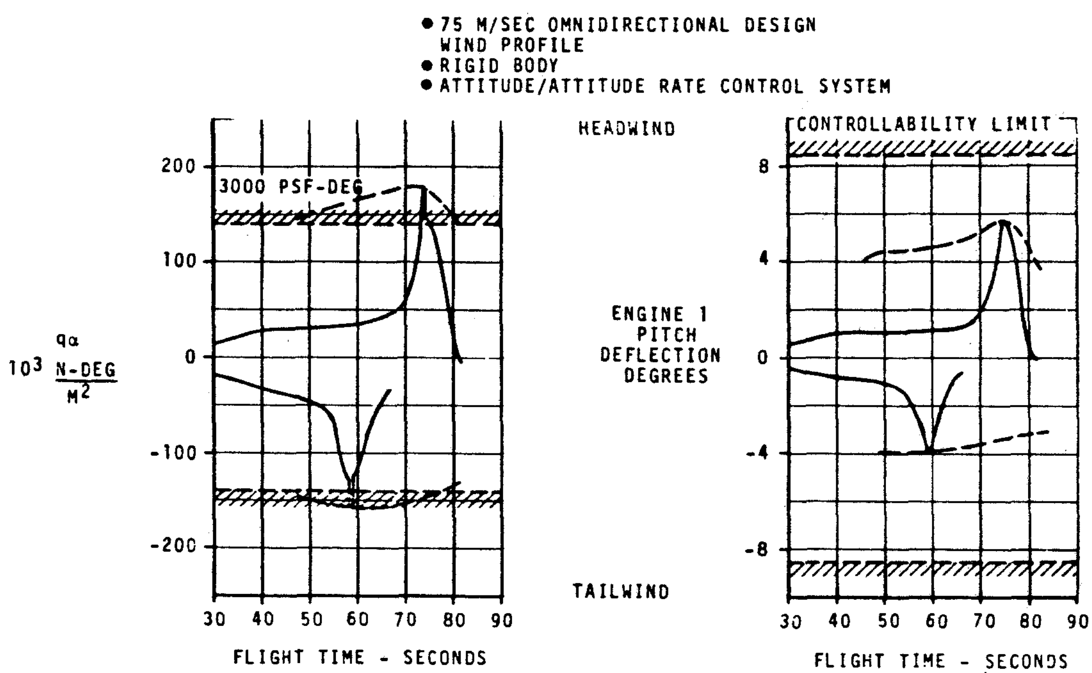


Figure 3-83. S-IC/HO Yaw Plane Design Wind Envelopes

The loads impact for the shuttle application of the S-1C affects primarily the forward skirt with an 1800-2300 pound structural beef-up required.

There is no increase in engine-induced acoustic and vibration environment over that considered for Generation 2 systems.

The S-1C expendable booster program cost is minimized by the following key features:

1. Minimum of required changes to basic S-1C

   Forward skirt structure
   Engine actuator attachment

2. Incorporation of simplifications

   Cost effective design changes
   Static firing of No. 1 only
   Vehicle operations

3. Lot fabrication and procurement

4. Use of refurbishment of existing components

   S-1C-14
   F-1 engines
   Spare components

The operations simplifications are illustrated in Figure 3-84. The post manufacturing and post-static firing checkout operations have been consolidated with similar operations at KSC. The engines are also installed at KSC.

The cost summary for the S-1C expendable booster is as follows.

DDT&E	$\$62\overline{M}$
Operations cost/launch	$\$3.49\overline{M}$
Production unit cost	$\$28.9\overline{M}$
Total cost - 12 launches	$\$451\overline{M}$
Peak annual funding (1978)	$\$82.6\overline{M}$

Figure 3-85 shows a typical S-1C expendable booster/LO_2 LH_2 reusable booster program.

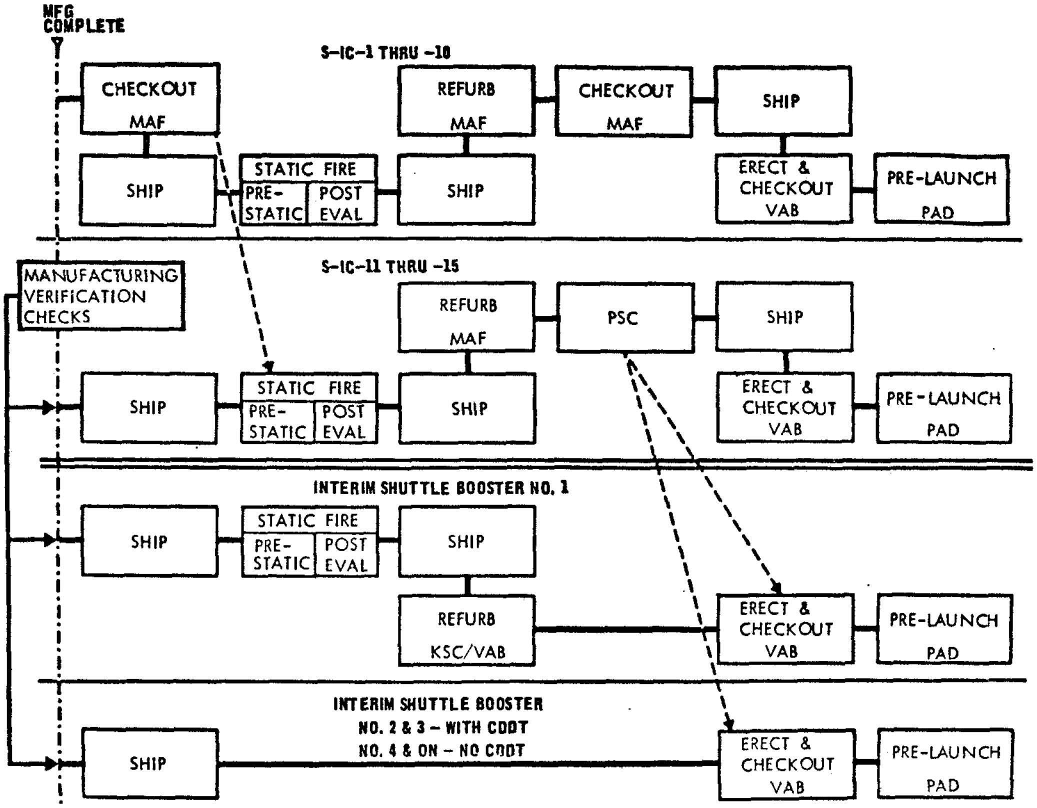
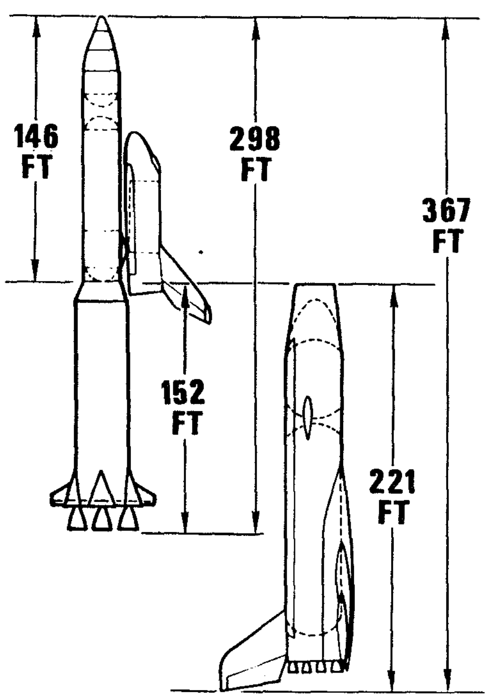


Figure 3-84. Vehicle Operations Concepts Comparison



	Gen 1		Gen 2	
	B	Orb	B	Orb
Stage				
Gross Wt (m lb)	5.0	0.771	2.35	1.024
Dry Wt (k lb*)	352.9	140	376	148
Ext Tank				
Wt (k lb)		48		48
Main Engines				
Number	5	3	12	3
Thrust (k lb)	** 1522	420	** 366	420
Sys GLOW (m lb)	5.77		3.37	
P/L (Dn Wt)(k lb)	45 (25) Due E		40 (40) Polar	
OMS ΔV (fps)	900		650	

*Payload & Exp Tanks not Included
**Sea Level

Figure 3-85. Saturn S-IC Interim Booster (15 by 40 and 15 by 60 Feet), 3-Engine Orbiter

3.10.20 Titan System

The Martin Company performed analyses using Titan III L boosters to launch the orbiter. The configuration is shown in Figure 3-86. The four 1207 SRM's were of the type planned (unmodified) for use in the Grand Tour program. All four SRM's plus five Titan LRM fire at liftoff and the SRM's drop away after burnout similar to the Titan IIIC operation. The performance of both the Generation 1 and Generation 2 shuttles is shown in Figures 3-87 and 3-88. To prevent the max-q value from getting too high, a nonoptimum performance type trajectory was used. A more optimum trajectory would have increased payload capability and max-q.

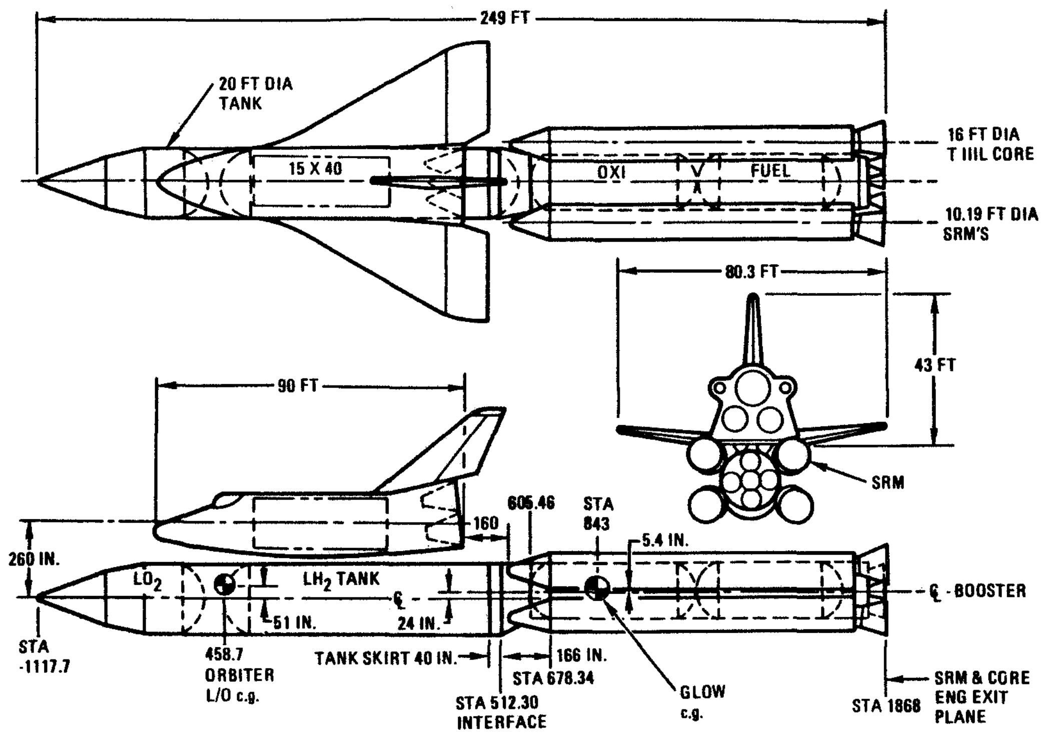
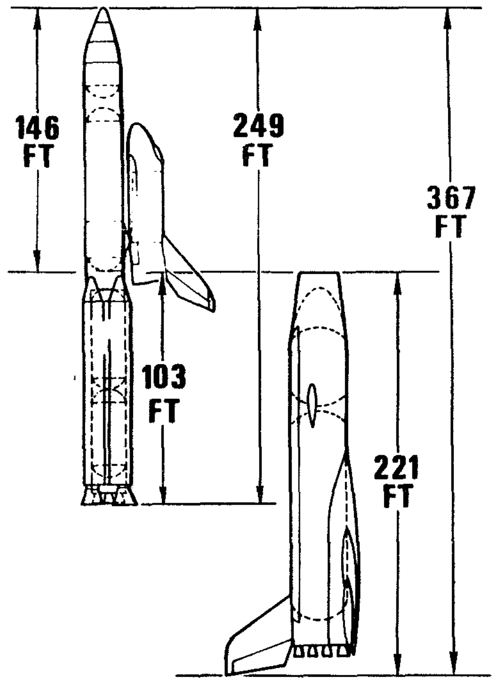


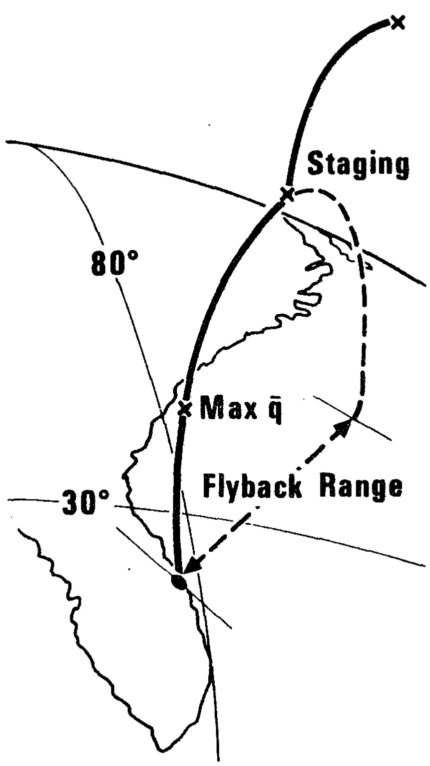
Figure 3-86. Titan III L Interim Booster



	Gen 1		Gen 2	
	B	Orb	B	Orb
Stage				
Gross Wt (m lb)	3.996	0.993	2.35	1.024
Dry Wt (k lb*)		140	376	148
Ext Tank				
Wt (k lb)		48		48
Main Engines				
Number	4/5	3	12	3
Thrust (k lb)	6773**	420	366**	420
Sys GLOW (m lb)	4.99		3.37	
P/L (Dn Wt)(k lb)	45 (25) Due E		40 (40) Polar	
OMS ΔV (fps)	900		650	

* Payload & Exp Tanks not Included
** Sea Level

Figure 3-87. Titan III L Interim Booster (15 by 40 and 15 by 60 Feet), 3-Engine Orbiter



	Gen 1		Gen 2	
P/L (Dn Wt) (k lb)	45 (25) Due E		40 (40) Polar	
OMS ΔV (fps)	900		650	
	B	Orb	B	Orb
Max $\bar{q}$ (psf)	580	580	488	488
Staging V_R (fps)	5771	5771	7029	7029
Staging γ_R (deg)	5.9	5.9	16	16
Flyback (n mi)			195	

Figure 3-88. Titan III L Interim Booster (15 by 40 and 15 by 60 Feet), 3-Engine Orbiter

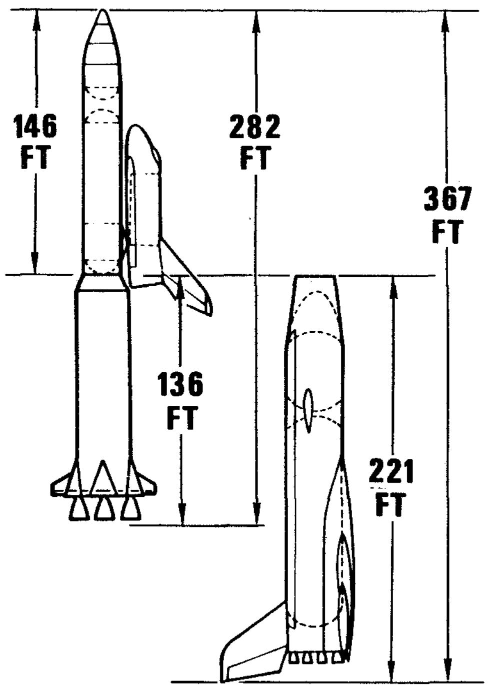
3.10.21 MCD System

The concept of an interim liquid-propellant booster based on the pressure-fed engine was investigated. This system also is referred to as the minimum cost design (MCD) booster. The booster propellant tank pressures are sufficiently high to produce a combustion chamber pressure of 300 psia without the use of turbopumps. Technical and cost parameters for this booster were developed by General Dynamics for applicability in the Generation 1 time period. The booster for Generation 2 was the LO_2/LH_2 reusable heat sink design, which employed 12 high pressure space shuttle main engines.

The results of this analysis are summarized in Figures 3-89 and 3-90.

3.10.22 Interim Booster Summary

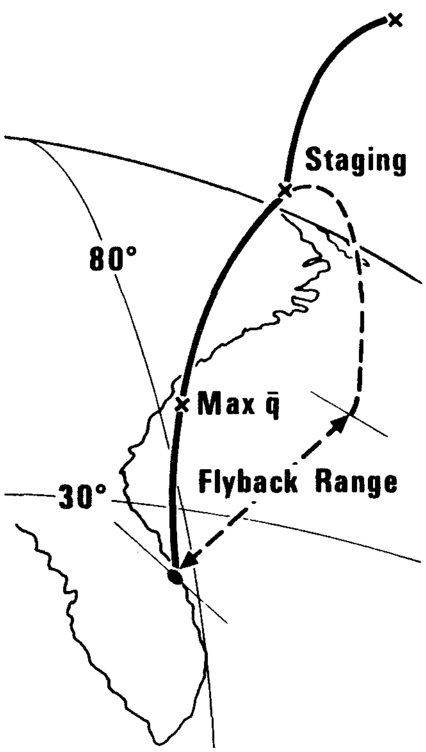
The major issues in the choice between parallel and phased development programs are summarized on Figure 3-91. The operational flight date for the reusable booster system is delayed from 1978 to 1982, but the annual peak funding is reduced from $2 billion to $1.2 billion. Payload capability in the reusable system is the same for both, but the interim system does have a smaller payload capability. The principal issues for the interim boosters are summarized on Figure 3-92 and the expendable booster costs are summarized on Figure 3-93. Referring to Figure 3-92, a comparison of booster lift-off weights (BLOW) shows a marked variation between the candidates. However, the program cost spread is only $210 million, which is shown also in the booster cost spread. The peak annual spending variation is only $20 million. From a facilities viewpoint, use of the S-1C would disturb the launch facilities least; use of the 260-inch SRM would require the greatest modifications. Ascent control is a problem with the single engine 260-inch SRM. The other candidates have satisfactory control characteristics with multiple nozzles with gimbal or LITVC capability for control. The S-1C is, of course, developed but would require some modifications; the 120-inch SRM units are in production but would require minor modification and assembly into a cluster. All the other candidates require development as the cost numbers on Figure 3-93 indicate. The 156-inch SRM cluster is considered as an alternate for the 260-inch SRM. As a single stage cluster it would avoid the ascent control problem of the single engine 260-inch SRM, the transport, and handling problems of the larger 260-inch grain, but would require motor development and clustering structure development. The 156-inch SRM cluster booster is a close second in cost to the 120-inch SRM cluster. The annual costing is shown on Figure 3-94. The 120-inch SRM cluster booster is superior from the cost standpoint (peak annual funding and total cost) and gives the most schedule relief (27-month start delay).



	Gen 1 B	Gen 1 Orb	Gen 2 B	Gen 2 Orb
Stage				
Gross Wt (m lb)	3.754	.993	2.35	1.024
Dry Wt (k lb*)	367	140	376	148
Ext Tank				
Wt (k lb)		48		48
Main Engines				
Number	7	3	12	3
Thrust (k lb)	881**	420	366**	420
Sys GLOW (m lb)	4.75		3.37	
P/L (Dn Wt)(k lb)	45 (25) Due E		40 (40) Polar	
OMS ΔV (fps)	900		650	

* Payload & Exp Tanks not Included
** Sea Level

Figure 3-89. MCD (Pressure Fed) Interim Booster (15 by 40 and 15 by 60 Feet), 3-Engine Orbiter



	Gen 1		Gen 2	
P/L (Dn Wt) (k lb)	45 (25) Due E		40 (40) Polar	
OMS ΔV (fps)	900		650	
	B	Orb	B	Orb
Max $\bar{q}$ (psf)	724	724	488	488
Staging V_R (fps)	6,276	6,276	7,029	7,029
Staging γ_R (deg)	12	12	16	16
Flyback (n mi)			195	

Figure 3-90. MCD (Pressure Fed) Interim Booster (15 by 40 and 15 by 60 Feet)

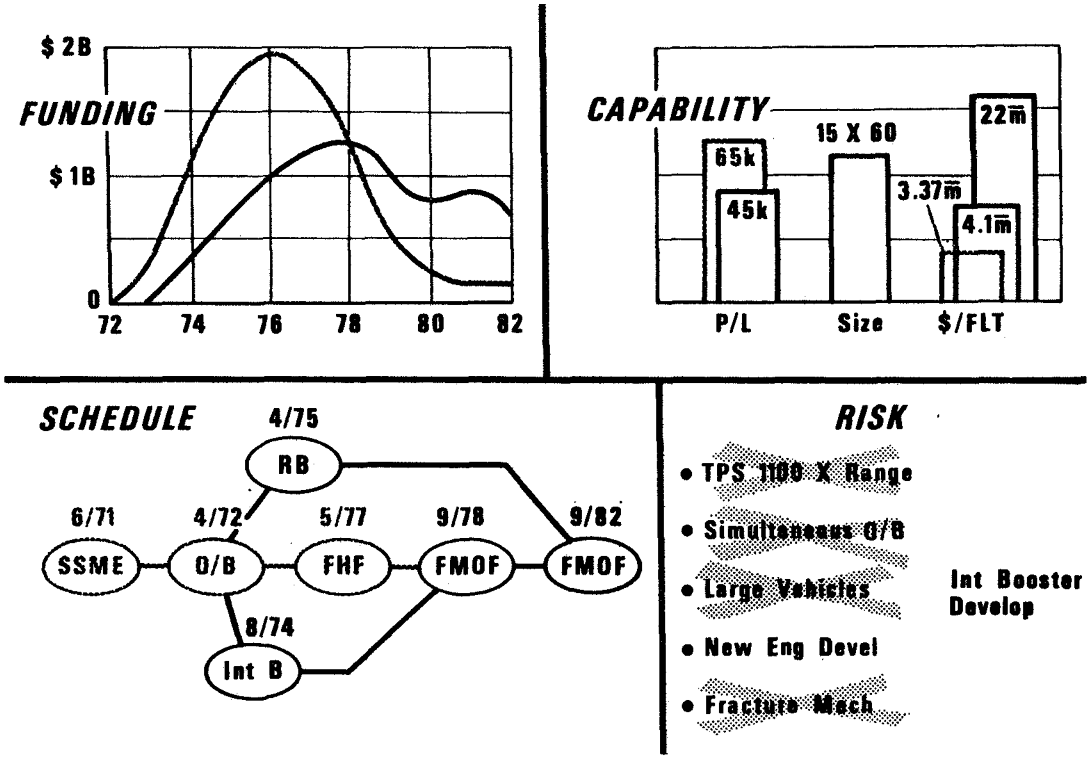
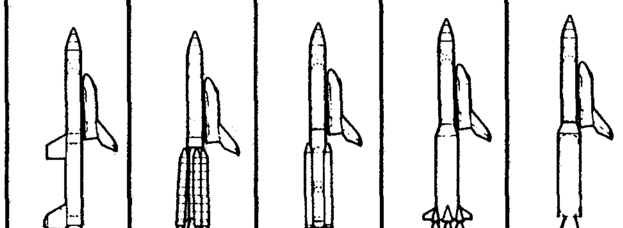


Figure 3-91. Interim Booster Summary

Orbiter Compatible With All

May Defer Decision



	260 In. SRM	120 In. SRM	Titan III L	S-IC	Press. Fed
BLOW M̄ LB / Cost $B	3.15 / 0.402	4.12 / 0.283	4.00 / 0.495	5.00 / 0.451	3.75 / 0.464
Cost $B Peak Annual (FY 78) / Prog	1.23 / 11.27	1.22 / 11.15	1.22 / 11.36	1.23 / 11.32	1.24 / 11.33
Facilities Impact (Booster)	Major $84M̄	Tools + Lut $23M̄	Tools + Lut $23M̄	Min	Solids/ Clusters
Control	Orb Assist (Gimbal)	SRM (Gimbal)	SRM (Gimbal)	Fins Adequate	LITVC
Devel Status	R&D	On-Going	Partial	Basically Complete	R&D
Schedule Impact — Decision Delayed (Mos)	9	18	6	18	6

Figure 3-92. Interim Booster Comparison

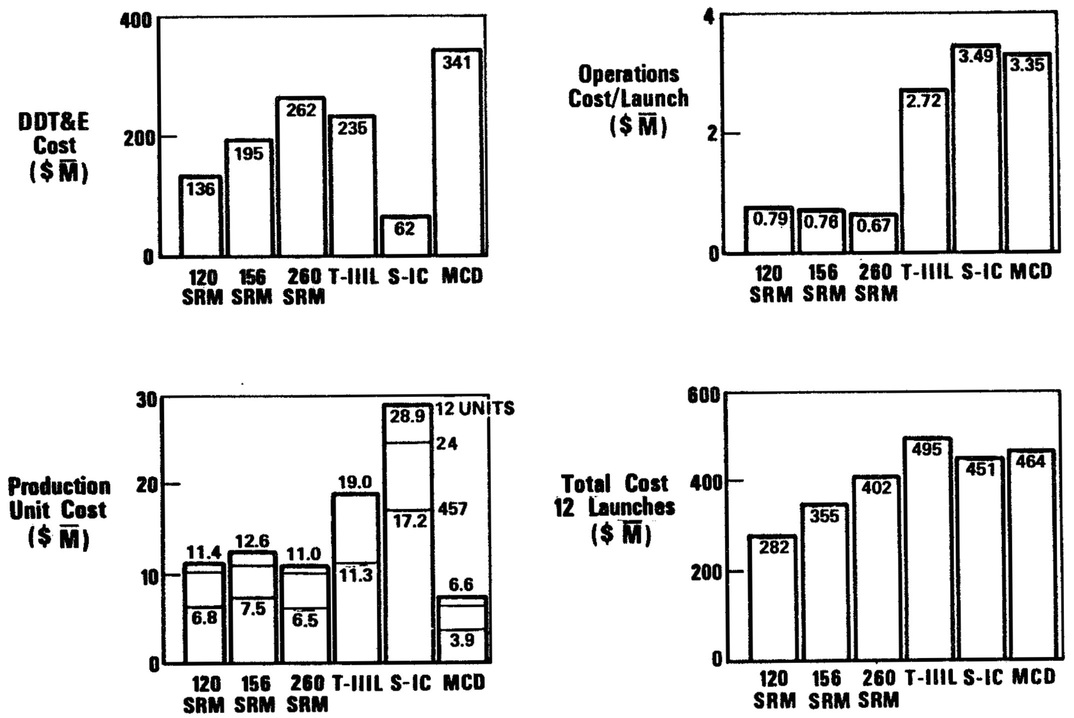


Figure 3-93. Expendable Booster Costs

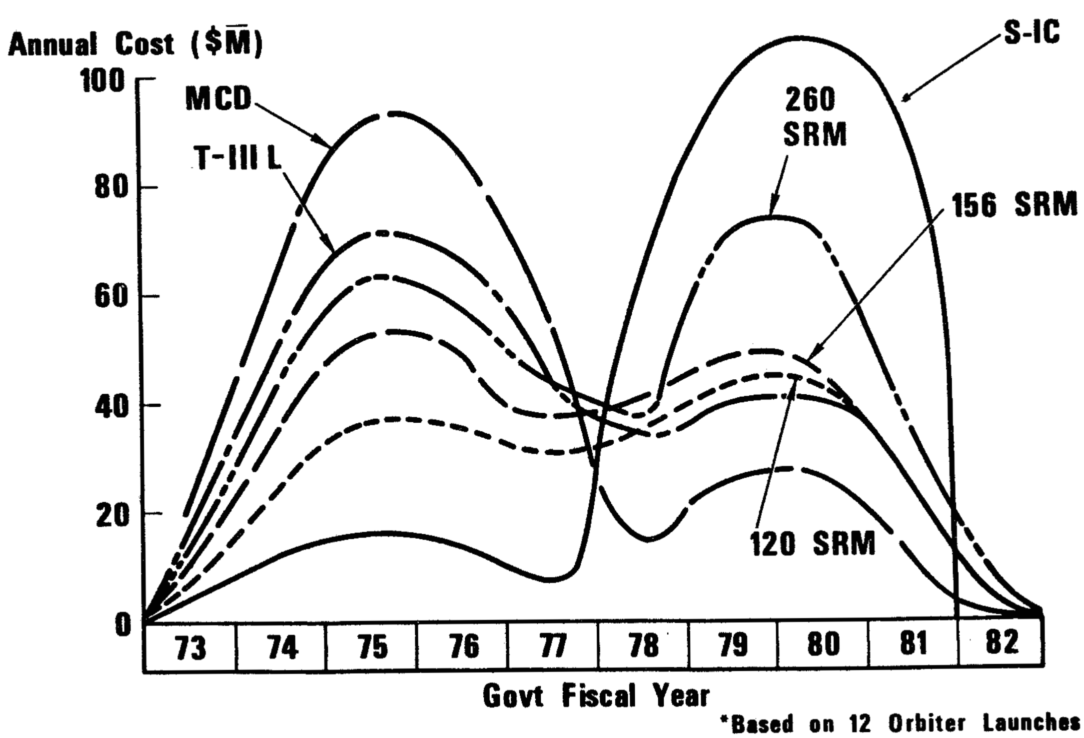


Figure 3-94. Expendable Booster Program Costs

From a development standpoint, the 120-inch SRM cluster booster shows superiority over the 260-inch SRM booster (Table 3-19). The technical risk comparison favors the 120-inch SRM cluster booster also (Table 3-20).

A low cost (40-foot cargo bay) expendable booster program was compared to the phased program (expendable Generation 1 and reusable Generation 2). The results (Figure 3-95) show a distinct cost advantage for the expendable booster study, but it does not have either the payload length (40-foot versus 60-foot) or weight capability of the Generation 2 reusable system.

Table 3-19. Comparison of Development Status

ITEM	120 IN. DIA SRM	260 IN. DIA SRM
PROPELLANT	QUALIFIED	QUALIFIED
IGNITION SYSTEM	QUALIFIED	DEVELOPMENT
MOTOR CASE FABRICATION	QUALIFIED	DEVELOPMENT
FLEXIBLE SEAL TVC	DEVELOPMENT	DEVELOPMENT
THRUST TERMINATION	DEVELOPMENT	DEVELOPMENT
CLUSTER ELEMENTS	* QUALIFIED	NOT REQUIRED
STAGING ELEMENTS	* QUALIFIED	NOT REQUIRED

*REQUIRES MODIFICATION

Table 3-20. Comparison of Technical Risk Issues, 260-Inch SRM Versus 120-Inch SRM

ISSUE	260 IN. DIA SRM	120 IN. DIA SRM
DEVELOPMENT STATUS	R&D	PARTIAL PRODUCTION
SYSTEM COMPLEXITY	SIMPLER SYSTEM	—
COST & SCHEDULE INTEGRITY		HIGHER CONFIDENCE
BOOSTER FALLOUT PROBLEMS	LESS PROBLEMS	
ABORT COVERAGE	NO FLEXIBILITY IF BOOSTER THRUST IS TERMINATED	SOME FLEXIBILITY IF ONE BOOSTER ENGINE IS OUT
RELIABILITY IN OPERATION	HIGHER	

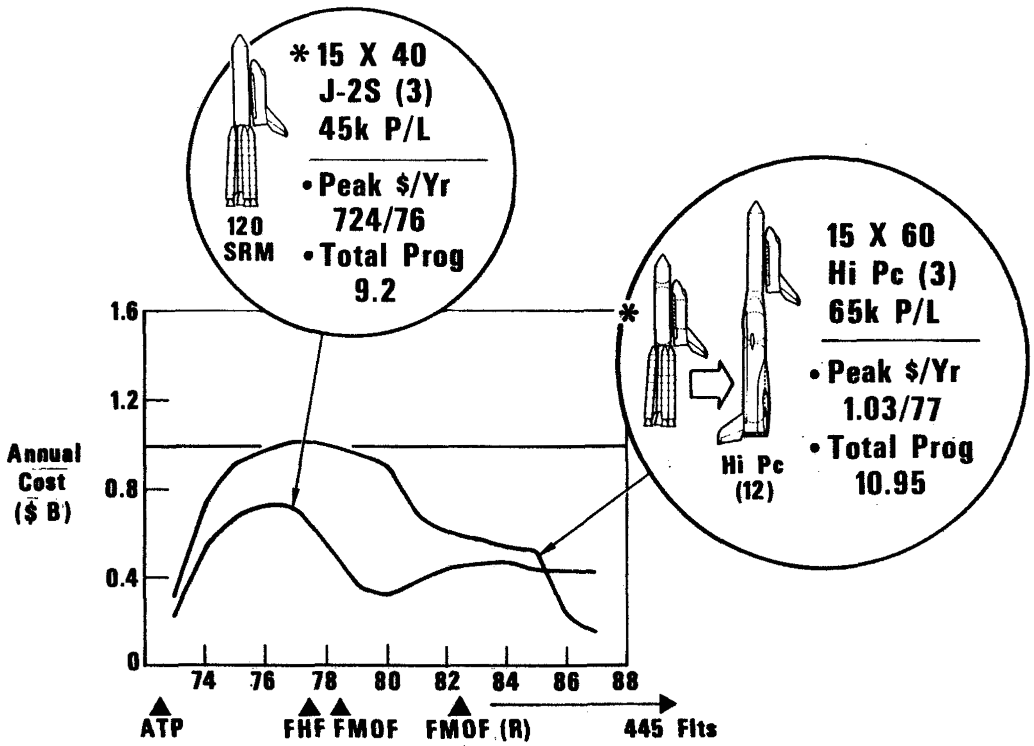


Figure 3-95. Expendable Versus Phased Reusable Booster

3.11 LOW TECHNOLOGY SYSTEM

3.11.1 Requirements

In addition to delaying the development of the booster and certain subsystems, lower annual funding and increased development confidence can be achieved by relaxing some of the program requirements for Generation 1 vehicles. The development of lower performance or less sophisticated subsystems, although resulting in somewhat heavier equipment, can also decrease the annual funding requirements in the initial phases of the program while achieving adequate performance capability. Figure 3-96 shows the deviations in program requirements that represent the cost minimization of Generation 1 from the ultimate Generation 2 (or Phase B).

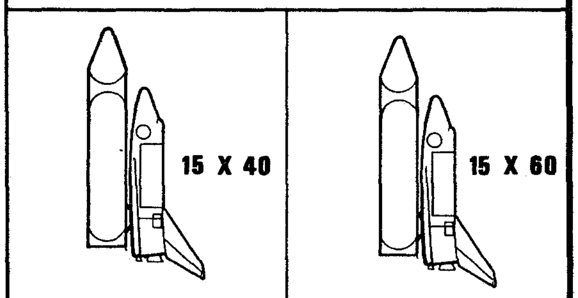
3.11.2 Subsystem Changes

3.11.2.1 OMS, Cryogenic to Storable

The orbit maneuvering system (OMS) design concept selected as the result of Phase B tradeoff studies employed liquid hydrogen and liquid oxygen propellants as specified in the Phase B Statement of Work. The primary objective of the tradeoff studies was to provide an OMS concept that not only met the operational and performance requirements but also yielded the lowest total Space Shuttle Program cost when employed in a fully reusable vehicle. The OMS concept evolved from these studies employed the designated propellant combination and featured integration of the OMS and the attitude control propulsion system (ACPS) propellant tankage and conditioning subsystems as a means of minimizing overall system weight and cost.

Subsequent to the completion of Phase B, changes in vehicle design concept and development program objectives necessitated a reexamination of both propellant selection and the attendant OMS design approaches. The key vehicle design concept variation that impacted OMS was the change from a fully reusable orbiter with internal main propellant tanks to one with expendable external tankage. This not only reduced on-orbit weight but also eliminated most of the internal volume previously available for OMS tank installation. In addition, the development program objectives were reoriented toward a phased, or incremental, vehicle operational schedule with the minimization of program expenditure rate (dollars/year) being the primary objective. The following discussion presents a summary of results pertinent to the reevaluation of OMS propellant selection in accordance with the revised design criteria and development program ground rules.

Generation 1



- 45k P/L Due East
- (3) J-2S Engines
- No Once-Around Abort
- 200 n mi X-Range
- OMS Tank Size 900 fps

Generation 2 Requirement

- 15 X 60 FT
- 65k lb
- Hi Pc
- Reusable Booster
- Once-Around Abort
- 1100 n mi X-Range
- 2000 fps

Figure 3-96. Requirements Deviations

Figures 3-97 and 3-98 present the cryogenic (LO_2/LH_2) and earth storable (N_2O_4/Aerozine 50) OMS concept schematics, respectively. Both system approaches employ two OMS engines and a fail-operational/fail-safe system design where that level of redundancy is required. The tankage is sized to provide 2000 ft/sec on-orbit ΔV for the space station logistics mission. The engine selection was directed toward maximum use of existing hardware for the first generation (Gen 1) vehicle with any inherent design life limits being offset by low development cost advantages. The development of fully reusable engines for the second generation (Gen 2) would be delayed until after peak vehicle funding level years. Tables 3-21 and 3-22 present the respective characteristics of the cryogenic and storable engines being considered.

Table 3-23 provides a comparison of the cryogenic and storable OMS system weights and tank volumes. In the key areas that affect orbiter vehicle design, namely OMS dry weight and tank volume, the storable system is lighter than the cryogenic system. Generation 1 and 2 storable systems are 3078 and 3150 pounds lighter, respectively, than the cryogenic system. In addition, the denser storable propellant requires 928 ft^3 less volume. The 6813-pound loaded weight advantage of the cryogenic system for the space station logistics mission affects only the booster vehicle and orbiter external propellant tank. Table 3-24 illustrates the effect of these weight and volume differences on the Gen 1 and Gen 2 vehicles. For the lower performance and lighter weight Gen 1 vehicle, the earth storable OMS weight and volume advantages more than offset the higher performance cryogenic OMS and results in a net shuttle system dry weight reduction of 5486 pounds. An analysis of the Gen 2 effects for the higher ΔV space station logistics mission (1500 ft/sec) revealed that cryogenic OMS performance superiority reduced this total vehicle dry weight advantage to 431 pounds.

The major advantage of employing an earth storable OMS arises from the low system development costs that are possible through the utilization of developed Apollo Program hardware by the first generation system. Table 3-25 illustrates the estimated Gen 1 and 2 OMS costs for the two candidate systems. A net total program savings of $56.62 million is projected if the earth storable system is employed. As shown in Figure 3-99, the major portion of this cost reduction occurs in the years of peak vehicle funding (1973-77) because of the phased development of the OMS. The advantages of the phased development approach over a concurrent development program are readily apparent.

In summary, it can be stated that an orbit maneuvering system employing earth storable propellants and utilizing existing Apollo Program hardware is superior to the cryogenic system concepts in the areas of volume, weight,

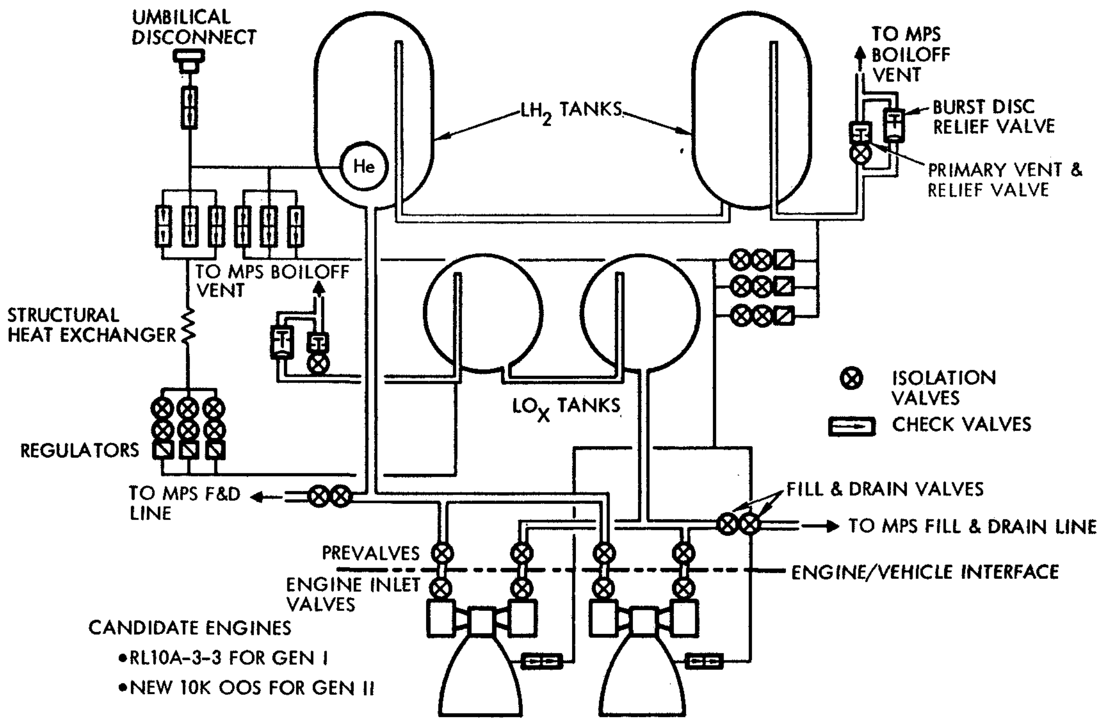


Figure 3-97. Cryogenic Orbit Maneuvering System (HO Configuration, Generations 1 and 2)

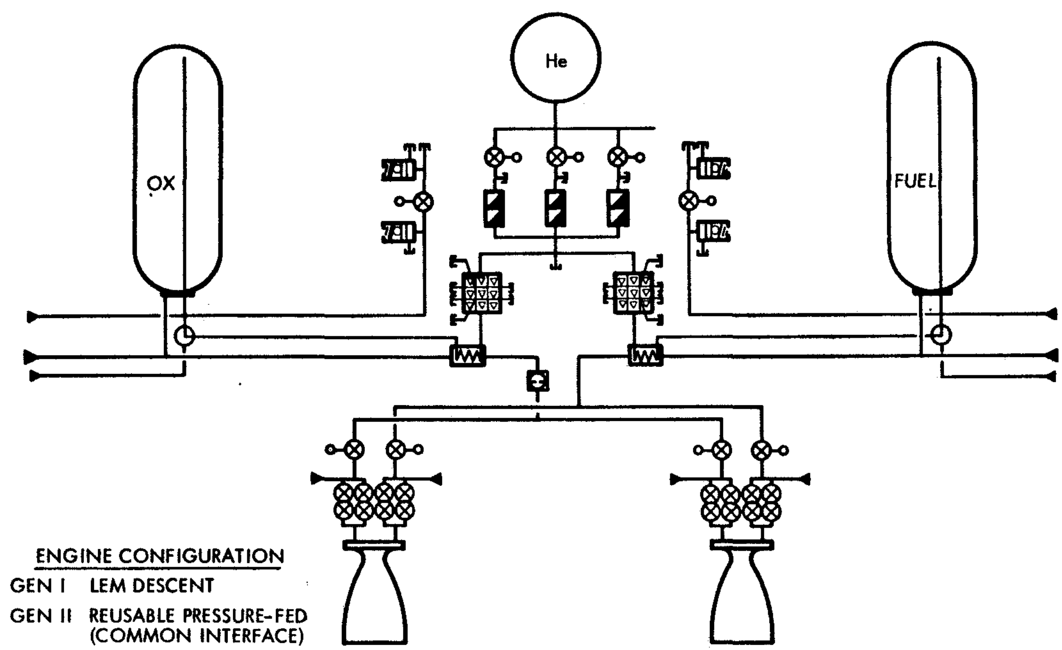


Figure 3-98. Storable Orbit Maneuvering System (HO Configuration, Generations 1 and 2)

and cost, as delineated in Table 3-26. In addition, because of the mature nature of systems employing earth storables and their inherent simplicity, they are considered to provide greater reliability and constitute a lesser development risk. It is recommended, therefore, that an earth storable orbit maneuvering propulsion system be employed by the space shuttle.

Table 3-21. Candidate Cryogenic Engines

Characteristic	Generation 1	Generation 2
Engine	RL10A-3-3	Orbit-to-orbit shuttle
Thrust (lb)	15,000	10,000
Specific impulse (sec (min))	439	464.6
Mixture ratio (deg/f)	5.0:1	6.0:1
Area ratio	57:1	400:1
Chamber pressure (psia)	400	1390
Cooling	Regenerative	Regenerative
Cycle type	Expander	Staged combustion
Chilldown	O.B. bleed	O.B. bleed
Weight (lb)	324	216
Gimbal angle (deg)	±4	±4
Length/diameter (in.)	70/39	78/44

Table 3-22. Storable OMS Engine Characteristics

	LEMDE	NEW 10K
PROPELLANTS	N_2O_4/A-50	N_2O_4/A-50
MIXTURE RATIO	1.6	1.6
THRUST LB	9,850	10,000
CHAMBER PRESSURE, PSIA	104	150
INLET PRESSURE, PSIA	222	225
NOZZLE AREA RATIO	47.5	68.5
NOMINAL I_{SP}, SEC	309	314
-3σ I_{SP}, SEC	305	311
ENGINE WEIGHT, LB	390	260
CHAMBER LIFE, SEC	1,000	INDEFINITE
CHAMBER DESIGN	ABLATIVE	REGEN
GIMBAL ANGLE, DEG	±6	±6
OVERALL LENGTH, IN.	90	90
EXIT DIAMETER, IN.	59	59

Table 3-23. OMS Weight/Volume Comparison

	GENERATION 1			GENERATION 2		
	HYPER-GOLIC	CRYO-GENIC	Δ CRYO-HYPER	HYPER-GOLIC	CRYO-GENIC	Δ CRYO-HYPER
✓ TANK VOLUME (FT^3)						
* FUEL	303	1,154	+851	303	1,154	+851
* OXIDIZER	303	380	+ 77	303	380	+ 77
* TOTAL	606	1,534	+928	606	1,534	+928
✓ SYSTEM WEIGHT (LB)						
DRY WEIGHT (INCL TANKS)	3,864	6,942	+3,078	3,604	6,754	+3,150
USABLE PROPELLANT						
ΔV = 650	12,317	8,520	-3,793	14,484	9,227	-5,257
ΔV = 900	19,534	13,739	-5,795	22,519	14,612	-7,907
ΔV = 1,500	-	-	-	31,873	21,295	-10,578
TOTAL LOADED WT						
ΔV = 650	16,992	17,088	+ 96	18,899	17,400	-1,499
ΔV = 900	24,210	22,307	-1,903	26,934	22,785	-4,149
ΔV = 1,500	-	-	-	36,288	29,475	-6,813

* GEN 2 TANK VOLUME (ΔV=2000 FPS) USED IN GEN 1 VEHICLE

Table 3-24. HO Vehicle Variations, Hypergolic Versus Cryogenic OMS

	GENERATION 1	GENERATION 2
* ON ORBIT V - FT/SEC	900	1500
WEIGHT CHANGE - (LB)		
OMS DRY WEIGHT Δ	+3078	+3150
FUSELAGE STRUCTURE Δ	+1091	+1091
FUSELAGE TPS Δ	+379	+379
WING STRUCTURE Δ	+500	+500
WING TPS Δ	+188	+188
TOTAL ORBITER DRY WEIGHT Δ (LB)	+5236	+5308
EXTERNAL TANKS Δ	+20	-397
TOTAL ORBITER + TANKS DRY WEIGHT Δ	+5256	+4911
OMS PROPELLANT Δ	-4995	-9923
BOOST PROPELLANT Δ	+678	-12922
ORBITER WEIGHT AT SEPARATION Δ	+939	-17934
BOOSTER DRY WEIGHT Δ	+230	-4480
GROSS LIFT-OFF WEIGHT Δ	+2579	-49311
BOOSTER + ORBITER DRY WEIGHT	+5486	+431

* Δ = CRYO WEIGHT - HYPER WEIGHT

Table 3-25. OMS Cost Comparison, HO Vehicle

	STORABLE $\$ \times 10^6$	CRYO $\$ \times 10^6$	CRYO - STOR Δ $\$ \times 10^6$
GENERATION I			
TFU	3.93	7.18	+3.25
SSD&E	22.33	38.43	+16.10
DDT&E	36.09	63.55	+27.46
PRODUCTION			
OPER REPLACEMENT	1.40	0.72	-0.68
TOTAL	37.40	64.27	+26.78
GENERATION II			
TFU	4.46	9.91	+ 5.45
SSD&E	120.61	125.07	+ 4.46
DDT&E	124.00	129.53	+ 5.53
PRODUCTION (3 NEW + 2 REFURB)	16.23	34.09	+17.86
OPER REPLACEMENT	4.50	10.95	+ 6.45
TOTAL	144.73	174.57	+29.84
TOTAL OMS COST	182.22	238.84	+56.62

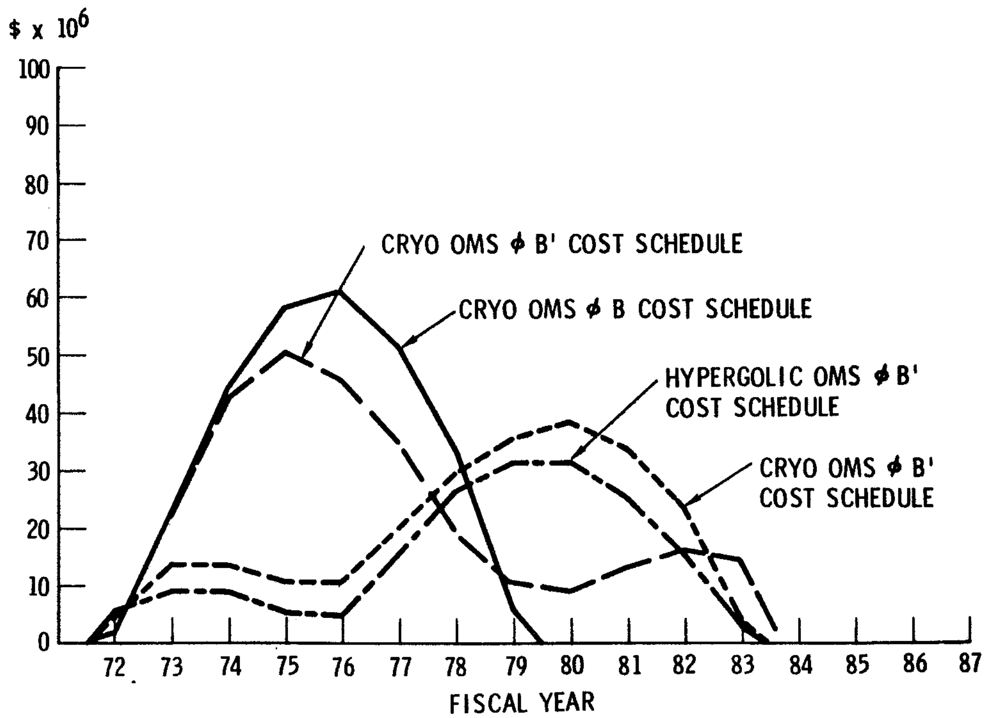


Figure 3-99. Cryogenic Versus Hypergolic OMS (HO Orbiter)

Table 3-26. Summary Comparison, Hypergolic Versus Cryogenic OMS

HYPERGOLIC SYSTEM ADVANTAGES

	GEN 1	GEN 2
REDUCED TANK VOLUME	928 FT^3	928 FT^3
REDUCED SYSTEM DRY WEIGHT	3078 LB	3150 LB
REDUCED ORBITER DRY WEIGHT	5256 LB	4911 LB
REDUCED STACK DRY WEIGHT	5486 LB	431 LB
REDUCED PROGRAM COSTS	26.78\overline{M}$	29.84\overline{M}$

ADDITIONAL CONSIDERATIONS

- REDUCED TECHNICAL RISK
- GREATER INHERENT RELIABILITY

3.11.2.2 ACPS, Cryogenic to Storable

Upon completion of Phase B study effort, a cryogenic attitude control propulsion system (ACPS) was selected as the orbiter baseline system. This concept was integrated using common propellant storage and machinery in conjunction with the orbit maneuvering system (OMS). The required integration of the selected cryogenic propellant storage and pumping systems introduces numerous areas of major technical risk representing an extensive technology/design/development effort and resulting in a large system cost.

Subsequent analysis was performed during Phase B' to define two reaction control systems: (1) a nonintegrated cryogenic system and, (2) a comparable earth storable hypergolic system. Trade study ground rules and assumptions are defined in Table 3-27. Attitude control system comparisons for three vehicles performing the due east mission are presented in Table 3-28.

Generally, the storable systems are lighter in terms of loaded vehicle weight and require a significant reduction in vehicle volume. The secondary impact of the reduced vehicle volume is a reduction in vehicle weight and surface area, which will reduce basic vehicle cost. A vehicle employing external oxygen and hydrogen (HO) tanks for main propulsion was selected as the baseline for a detailed system tradeoff study.

The storable ACPS was configured in three modules located in the nose, on the bottom fuselage surface, and in the aft fuselage. The forward module provides ± roll, - pitch, ± yaw rotation, and -X, +Z, and ± translation assist. The bottom fuselage module affords -Z translation; it controls pitch translation during docking maneuvers. The aft module supplies ± roll, + pitch, ± yaw translation, and +X, +Z, and +Y translation assist. Figure 3-100 shows the forward module schematic and component arrangement for the fail-operational/fail-safe system design.

The cryogenic ACPS uses common propellant storage with the OMS propellant tanks. Separate turbopumps, gas generators, and heat exchangers are used to condition the subcritical liquid to a gaseous state, after which it is stored in high-pressure accumulators. Figure 3-101 is a schematic of the cryogenic ACPS for the fail-operational/fail-safe system design.

Thruster orientation (for both systems) and the system weight comparison are shown on Figure 3-102. The storable ACPS is 3378 pounds lighter than the cryogenic system in terms of subsystem dry weight and 2079 pounds lighter in total loaded weight. The reduction in total hardware requirements plus the reduction in technical risk related to cryogenic

Table 3-27. ACPS Ground Rules

- Due east mission (sizes propellant quantity and tankage)
- All systems sized for generation 2 vehicles
- Thrust level sized by entry yaw acceleration = 1.3°/sec^2
- Fail operational/fail safe

Storable System	Cryogenic System
Three independent modules Forward, midship, and aft Propellant weights based on: I_{sp} = 290 seconds 5% contingency and 98% expulsion efficiency System tankage Tank pressure 425 psia Tank SF = 2.0 and He tank = 1.5 Spherical tanks	Integrated system with common Propellant storage - OMS tank Separate TP's and H-X's (3 each) Propellant weight based on: I_{sp} = 370 seconds (system delivered) Startup ≈ 80 cycles System tankage ΔOMS tank volume, weight and cost Charged to ACPS

Table 3-28. ACPS Comparison

			STORABLE SYSTEM				CRYOGENIC SYSTEM			
VEHICLE	ENTRY (1) WT (K LB)	RCS THRUST LEVEL	USABLE PROPEL-LANT	SYS WT	LOADED WT	VOL (FT^3)	USABLE (2) PROPEL-LANT	SYS WT	LOADED (3) WT (4)	VOL (FT^3)
H	281	1,650	6,950	2,800	9,750	95.5	5,790	6,026	11,816	392
HO	247	1,550	6,950	2,757	9,707	95.0	5,790	5,996	11,786	390
NATO	217	2,000	6,550	2,880	9,530	93.0	5,550	6,273	11,823	408

NOTES -
1. CAPABLE OF 65K PAYLOAD DUE EAST MISSION
2. INCLUDES START-UP LOSSES, RESIDUALS & IMPULSE PROPELLANT
3. INCLUDES OMS TANKS ΔWT FOR RCS PROPELLANT STORAGE
4. OMS TANK SIZED FOR 2000 FT/SEC ΔV

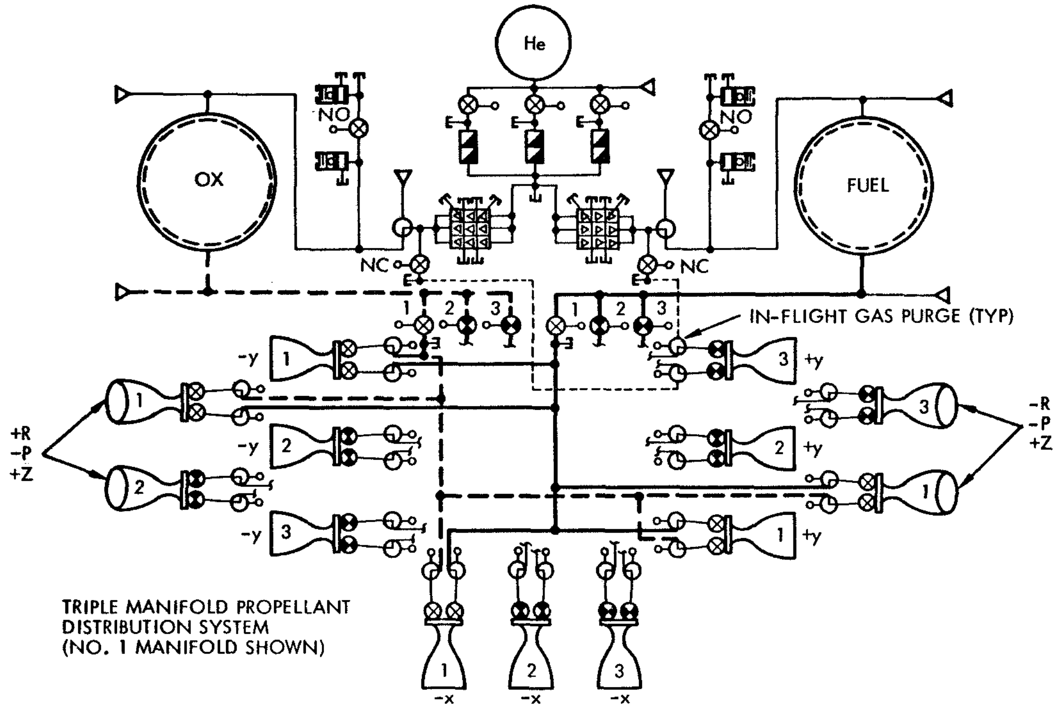


Figure 3-100. Storable Orbiter ACPS (Forward Module)

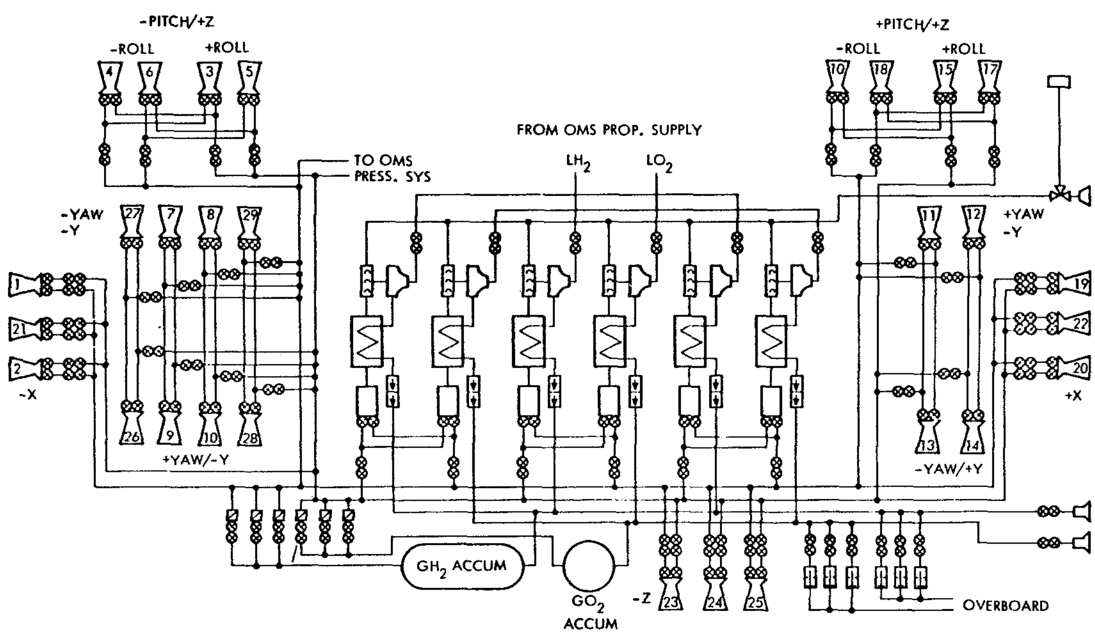
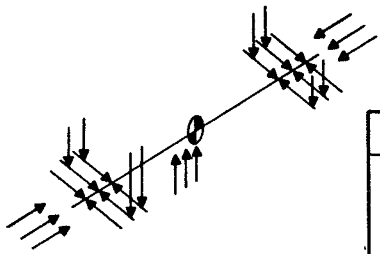


Figure 3-101. Cryogenic Orbiter ACPS (HO Vehicle Configuration)



SYSTEM CHARACTERISTICS

THRUST/THRUSTER - 1550 LB

THRUSTERS/MODULE

FORWARD	13
-Z	3
AFT	13

SYSTEM WEIGHT	STORABLE	CRYO
PROPELLANT		
IMPULSE	6,620	5,190
CONTINGENCY	330	–
RESIDUAL	139	600
START-UP LOSSES	–	
SYSTEM		
THRUSTERS (29)	798	915
TANKS & DISTRIBUTION	1,074	5,081
PRESSURIZATION	746	-
TOTAL	9,707	11,786

Figure 3-102. ACPS Configuration (HO Orbiter)

3.11.2.3 Hydraulic System Pressure Tradeoff, 3000 Versus 4000 Psi

A study was conducted to determine the optimum pressure for the Phase B' EOHT space shuttle configuration hydraulic system because optimum pressure is sensitive to vehicle configuration. Past studies have shown that a 4000-psi hydraulic system is lighter in weight than a 3000-psi system (910 pounds for the final Phase B configuration). The configuration studied on the EOHT was a four-system arrangement, resulting in a differential hydraulic weight of 255 pounds in favor of the 4000-psi system. (See Figure 3-103 for results of pressure tradeoff studies.) The weight differential advantage of the 4000-psi system was judged to be offset by the availability of 3000-psi hardware, such as pumps, check valves, relief valves, filters, and solenoid control valves, suitable for the EOHT configuration. Although additional testing and possible modification of these components may be necessary to meet requirements unique to the space shuttle vehicle, this cost is less than the cost of designing, developing, and testing new hydraulic components of similar configuration for a 4000-psi system. The hydraulic system operating pressure of 3000 psi was recommended on the basis of minimum cost and minimum technical risk.

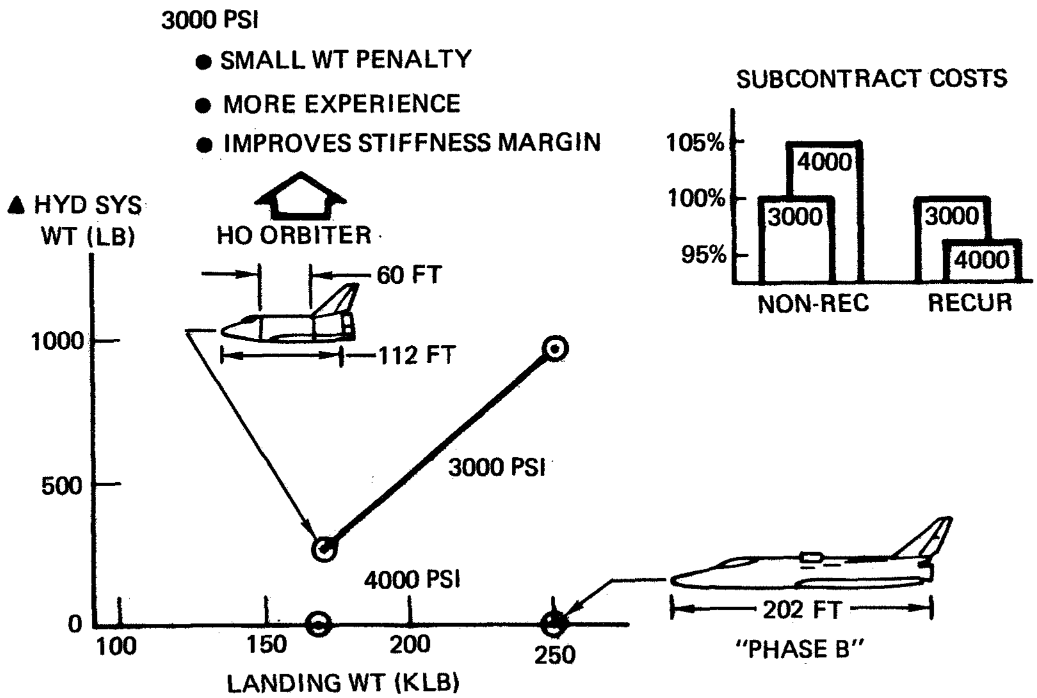


Figure 3-103. 3000-psi Hydraulic System Lower Risk

turbopump and heat exchangers result in significant cost differentials, as summarized in Table 3-29, which indicates that a cryogenic ACPS will incur a program cost on the order of 357 million dollars compared to 173 million for a storable system. This cost differential of 184 million together with the extensive technical risk in cryogenic system development results in recommending the storable ACPS for space shuttle orbiter application. The design characteristics of the storable ACPS are summarized in Table 3-30.

Table 3-29. Cost Comparison for HO Orbiter

	CRYOGENIC SYSTEM	STORABLE SYSTEM
THEORETICAL FIRST UNIT	16.3	10.3
SYSTEM & SUBSYSTEM DEVELOPMENT ENGINEERING	224.9	89.5
DESIGN DEVELOPMENT TEST & ENGINEERING	282.2	125.55
PRODUCTION	59.5	37.49
OPERATION	16.5	10.40
TOTAL PROGRAM	357.2	173.44

Table 3-30. Storable ACPS Characteristics

Thrust (lb)	1,550
P_c (psi)	300
Expansion ratio	20:1
Weight (lb)	27.5
Number of thrusters	29
Thruster specific impulse (sec)	290
System del specific impulse (sec)	290
Propellants	N_2O_4/A-50
System hardware weight (lb)	2,757
Impulse propellant (lb) (due east mission)	6,950
Total system weight (lb)	9,707
WBS program cost	173.44M

3.11.2.4 Avionics

The objective of the avionics low-cost study was to develop a shuttle avionics system configuration that could be designed, developed and manufactured at annual and total costs lower than the costs estimated for the Phase B baseline configuration. The scope of the study included the definition of a phased development approach. The study encompassed the following areas:

1. Orbiter avionic systems, including software
2. Expendable and reusable booster avionic systems, including software
3. Impact on:
   a. Other vehicle subsystems - ECLSS, electrical power, structures
   b. Ground systems, including checkout and launch equipment, ground software, MCC, MSFN.

The following study ground rules were developed to assure the maximum benefit from the avionic system cost reduction studies.

1. Define a two-generation phased development program (Mark I, Mark II).

2. Avoid block changes.

3. Use Apollo concepts in such areas as EDS, C&W, and failure detection.

4. Evaluate redundancy management concepts and fault tolerance requirements for reduction.

The orbiter avionics system requirements and expendable booster concepts were evaluated, with maximum emphasis placed on the definition of a phased development program and use of off-the-shelf equipment. Cost advantages, either in the form of deferred development or deleted costs, were then analyzed in order to define the phased development program and select a candidate system.

Interface requirements were delineated for the new configurations, particularly for the avionics interfaces between the orbiter and the external HO tank and between the tank and the expendable solid rocket motor boosters being studied during the first phase of the Phase B extension studies. Figure 3-104 shows these interfaces.

The avionic studies were conducted assuming the same requirements used in the Phase B studies, except for: considerations related to phased development; the checkout and fault isolation subsystem optimized to support onboard redundancy management, with fault isolation to the functional path level; fail-safe configuration assumed and traded up; and initial considerations of an expendable booster. With these study requirements, it was concluded that the main advantage that could be achieved was in the deferment of development costs. The results of the time-phased cost analysis conducted in this study phase are shown in Figure 3-105, indicating the effect on total avionics costs of the phased development program.

The study results indicated the necessity for performing a reiteration of the avionics system study, with greater latitude permitted regarding requirements to be satisfied. Results of this subsequent study are discussed in paragraph 4.4.3.9.

3.11.2.5 Main Propulsion System

Lower payload requirements for Generation 1 vehicles lead to the possibility of using a lower performance main engine if significant reduction in early funding can be realized. A number of engines are available as

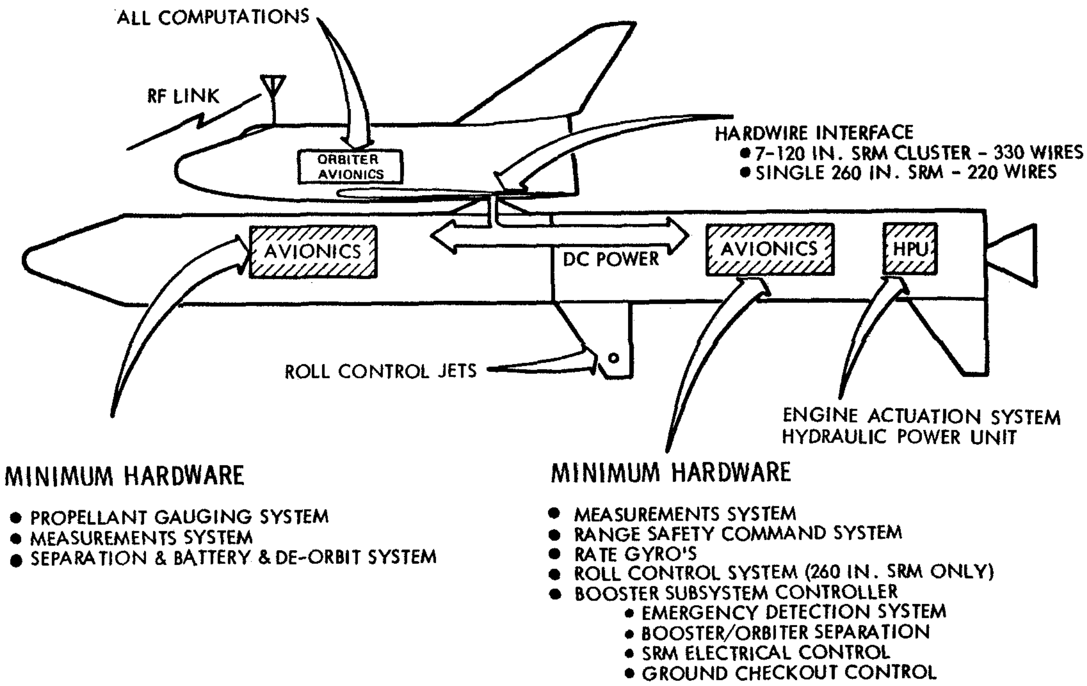


Figure 3-104. Expendable Booster/Drop Tank Avionics

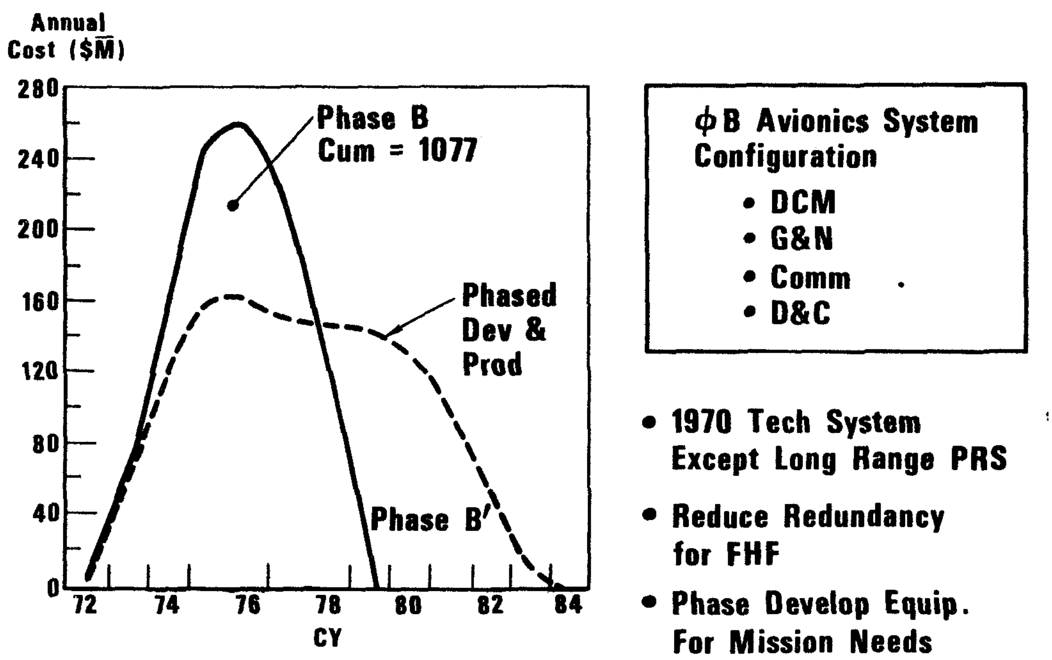


Figure 3-105. Avionic Changes for Lower Peak Funding

off-the-shelf, or nearly so, in terms of development and testing of components. Both the J-2 and J-2S engines are immediate candidates because of thrust level and specific impulse only slightly lower than a HiPc engine but providing significant cost reduction. For example, Figure 3-106 presents a comparison of the costs of the J-2S engine program with a HiPc engine program with and without deferred development.

The Generation 1 requirements result in the following effects.

1. Reduced payload permits a shorter orbiter by 20 feet and subsequent smaller wing.

2. The use of J-2S engines delays the development of the HiPc engine.

3. Reduced orbiter tanks and booster result from elimination of the once-around abort constraint.

4. Low cross range allows the use of ablator TPS, thereby deferring development of RSI.

5. 900-fps OMS tank size reduces orbiter volume. A kit in the cargo bay can provide the logistics mission ΔV = 2000 fps requirements when necessary.

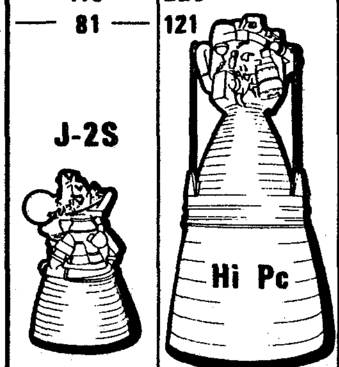
In addition, the two-week turnaround time for final operational vehicles has been relaxed to one month for interim operations.

A more detailed comparison of the performance of the three candidate engines is presented in Table 3-31. The impact of the performance differences are discussed in sections of this report dealing with system analyses and trade studies.

In addition to performance differences in the candidate engines, the baseline orbiter design requires certain modifications to the J-2 and J-2S engine fluid systems. These design factors and required engine modifications are summarized in Table 3-32 and significant differences are discussed in the following paragraphs.

The start sequence on J-2 requires an initial LO_2 tank ullage pressure of 62 to 64 psi. The J-2S requires an LO_2 tank ullage pressure of 39 to 41 psi. This difference is dictated by the requirement to maintain J-2 NPSP start requirements despite the inlet pressure slump resulting from the fast engine start sequence and column flow acceleration requirements during start. This pressure results in an LO_2 tank weight increase of approximately 2700 pounds for the J-2 versus J-2S engine. As a result of the high tank

	J-2S	Hi Pc
Length IN.	119	226
Diameter IN.	81	121



	J-2S	Hi Pc
Thrust (vac k lb)	265	420
I_{sp} (vac sec)	436	459
M.R.	5.5	6.0
ϵ	40	150
Weight (lb)	3,800	6,200

Engine Costs

	J-2S	Hi Pc
Non-Recur	58$\bar{m}$	394$\bar{m}$
Recurring	2$\bar{m}$	3$\bar{m}$

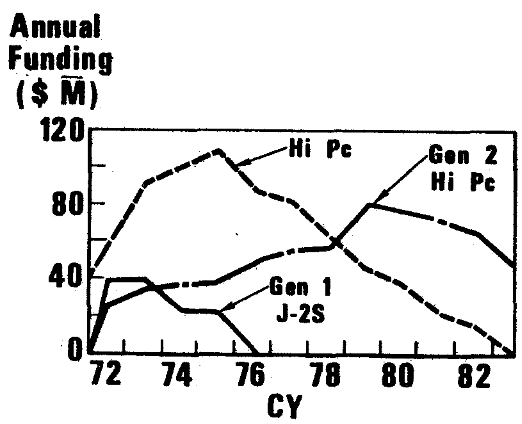


Figure 3-106. J-2S Engine Reduces Peak Funding

Table 3-31. Nominal Engine Performance Parameters

Engine Characteristic	Engine Model		
	J-2	J-2S	SSME
Thrust (VAC)	230K	265K	265K
I_{sp} (sec) (VAC)	425	436	453.2
Mix ratio	5.5	5.5	6.0
Range	4.5-5.5	4.5-6.0	5.5-6.5
LO_2 flow rate (lb/sec)	458	514.3	501.1
LH_2 flow rate (lb/sec)	83.3	93.5	83.6
Chamber pressure (psia)	780	1247	3000
Area expansion ratio	27.5	40	90
LO_2 pump inlet pressure (psia)	39	39 (33)	25
LH_2 pump inlet pressure (psia)	30	30 (27)	20
Recirc flow rates (lb/sec)			
LO_2	0.75	0.75	4.0
LH_2	1.0	1.0	1.7
Throttle range	0	50-100%	50-109%

pressure required to meet minimum NPSP requirements during the fast start sequence, a relatively high surge (approximately 300 psia) is encountered on the J-2 engine. To modify the J-2 engine for a slower start sequence would require extensive engine changes and testing; i.e., main LO_2 valve, gas generator, etc. This does not appear to be a problem on a J-2S, since it employs a slower start sequence.

The surges that will be experienced at engine cutoff far exceed the allowable surge pressure of 132 psia for the J-2 engine as a result of its fast shutdown sequence (approximately 450 psi) and exceed the allowable J-2S engine surge pressure although not as severe because of a slower shutdown sequence. An evaluation of this condition indicates the most desirable solution to the problem is to modify the J-2 engine shutdown sequence or incorporate the 2:1 throttling capability in the J-2S, with a resultant reduction in propellant flow at cutoff, which would prevent excessive surges.

The J-2 will require extended fuel bleed times as part of the start sequence for thrust chamber chill. During the fuel bleed cycle, the augmented spark igniter (ASI) operates with both oxidizer and fuel flow. The bleed cycle is initiated before separation to accommodate the separation sequence. Under these conditions, an unacceptable off mixture ratio condition results in the ASI from the high LO_2 pump inlet pressure. To accept this condition, extensive modifications would be required to the J-2 engine. The J-2S is also subjected to this condition; however, since the condition

Table 3-32. Fluid System Comparison

Condition	Ops Value	SSME Des	J-2S		J-2	
			Des	Remarks	Des	Remarks
LO_2 pump inlet pressure for engine start (psi)		25	39	39-41 psi ullage pressure required	39	62-64 psi ullage pressure required mod main LO_2 valve, gas generator, etc.
LO_2 pump static inlet press - 1st stage boost (psi)	190 + ullage	275	132	Mod to inlet bellows and LO_2 pump	132	Mod to inlet bellows and LO_2 pump
LO_2 pump inlet pressure 2nd stage boost (psi)	>200	275	52	Head suppression sys or mod as above	55	Head suppression sys or mod as above
LO_2 pump inlet pressure surges at ECO (psi)	~450 for J-2	2:1 throttling capability	132	Design for 2:1 throttling	132	2:1 throttling required (extensive mod required)
LO_2 pump inlet pressure during dual bleed cycle	190 +			Mod to ASI feed sys		Extensive mod to ASI feed sys
Thrust chamber chilldown		Idle mode		Idle mode		>8 sec LH_2 lead time required
Turbine start		Normal tank pressure		Solid propellant		Start bottle with increased insulation
Propellant reconditioning		Recirc sys		Mod required for recirc sys		Recirc sys
Sea level start		OK		Minor mod for side loads		Mod to turbine bypass valve and thrust chamber and nozzle
Max-g level during boost	3	2:1 throttling capability		Design for 2:1 throttling		Extensive mod for 2:1 throttling or engine shutdown required

exists on the J-2S for only one second and since its ASI propellant feed system is less sensitive to pump inlet variations, less significant modifications will be required.

The thrust chamber temperature and start bottle pressure and temperature requirements for a J-2 engine at start will require modification to the engine to provide extended fuel lead time (8 + seconds) and increased thermal protection for the start tank, or modification to the orbiter vehicle to incorporate an expendable closeout cover for the rocket engine nozzles, which would incur extensive design, test, weight, and cost penalties and does not appear technically feasible. The J-2S does not require a special thrust chamber chill (normal fuel bleed idle mode will attain the ambient thrust chamber temperature required for start) and does not use a start bottle for start energy (a solid propellant turbine starter is used in lieu of a start tank).

The orbiter configuration with the LO_2 tank forward requires an approximate 100-foot long LO_2 feed line. The LH_2 feed line incorporates an approximate 40-foot long run to the engine interface. The present configuration of the J-2S engine does not incorporate provisions for propellant recirculation. Without recirculation, the volume of LO_2 and LH_2 that would either be two-phase or saturated fluid within lines of this configuration cannot be accommodated with an acceptable start sequence. It is recommended to incorporate recirculation systems to recirculate propellants from the propellant tanks, through the engine pumps and high-pressure ducting, and return to the tanks.

Neither the J-2 nor the J-2S engine have the capability of starting below 40,000 feet altitude without restraints on the thrust chamber for start transient separation sideloads. It is not considered feasible to modify the J-2 engine to accommodate sea level and altitude starts, since an orifice change in the turbine bypass valve is required between sea level and altitude and a completely redesigned thrust chamber would be required to achieve the desired nozzle contour and structural strength for sea level starts. A modification to strengthen the J-2S thrust chamber fuel injection manifold attach points and actuator mounts would accommodate sea level and altitude starts, because the sea level start side loads are reduced on the J-2S and the thrust chamber is a beefed-up design.

The SSME is designed to accommodate the space shuttle requirements of reusability and low postflight maintenance to allow a quick turnaround, unless the J-2/J-2S were originally designed for single missions only. The relative degrees of maintenance requirements for the three engines are indicated in Table 3-33, which shows the relative complexity factors for

Table 3-33. Postflight Engine Servicing Requirements Relative Complexity Factors

Requirement	Engine Model		
	J-2	J-2S	SSME
Vent and purge propellant feed and engine servicing systems	6	4	4
Vent and purge H_2 start tank	3	N/A	N/A
Remove expended SPTS	N/A	1	N/A
Purge LO_2 dome and thrust chamber	3	3	3
Purge turbopumps	2	2	2
Purge gas generator	3	N/A	N/A
Dry thrust chamber, LO_2 dome, turbopump seal cavities, gas generator and hot gas system	24	N/A	N/A
Perform subsystem gross LX check	48	N/A	N/A
Perform electromech sequence test	8	N/A	N/A
Flight instrument test	12	8	4
Mixture ratio control val test	2	N/A	N/A
Install engine covers	4	4	4

postflight maintenance of engine components and systems. The J-2 has the most maintenance items and the greater number of complexity factors. The J-2S is a considerable improvement over the J-2 by offering a more simplified system, which, by design, requires less maintenance attention.

3.11.3 Low-Cost Program Options

To assure that the proposed orbiter test program is cost effective, a reevaluation of the Phase B baseline was performed. This review was to reestablish the validity of the requirements and the optimum approach to support a phased program. The items reviewed are identified on Table 3-34. The significant conclusions of this review are as follows:

1. Utilize a modified MPS test article.

2. Utilize a single cabin for ECLSS functional and static testing.

3. Utilize Orbiter Number 1 for HFT and Orbiter Number 2 for VFT supplemented by Orbiter Number 1.

4. Eliminate the fatigue test article.

5. Utilize "bare bone" orbiters for flight test.

Table 3-34. Summary of Phase B Baseline Reevaluation

Item	Conclusion	Rationale
FLIGHT TEST PROGRAM		
Eliminate the horizontal flight testing on orbiters subsequent to Orbiter 1.	The HFT program can be accomplished by Orbiter 1. Horizontal flights on subsequent orbiters will be limited to airworthiness demonstration only.	The Master Program Schedule allows sufficient time for the HFT program table conducted on Orbiter 1. The revised launch rate during the early years allows Orbiter 2 and Orbiter 1 to satisfy the requirements.
Limit the flight test program to 2 orbiters, use Number 1 for HFT, then modify and schedule for MOF; use Number 2 orbiter for FMOF. Also, delay delivery of Orbiters 3, 4, 5.	The flight test program will be structured for Orbiters 1 and 2 only.	Eliminate the horizontal flight testing on orbiters subsequent to Orbiter 1.
Minimize the number of subsystems on Orbiters 1 and 2 to only those essential to support the test program.	The identification and implementation of "bare bones" orbiters for HFT and VFT will be implemented.	The HFT can start 3 months earlier due to the reduction on time for fabrication and assembly. The uninstalled system allows deferment of \$50.7M.
Provide incremental vertical flight testing to eliminate any requirements for unmanned vertical flights and manned scaled prototypes.	The first and subsequent HFT and VFT will be implemented. Depressed trajectories and not recommended.	The ground test program and HFT are sufficient to allow the first MOF to be manned. Depressed trajectory tests are not cost effective.
MAJOR TEST ARTICLES		
Elimination of separation testing.	Retain ground separation testing.	Ground separation testing is a key element in the cost-effective approach to manned launchs.
Utilize the ECLSS crew cabin test article for static structural testing.	The test requirements and scheduling are compatible for a single cabin with dual usage, and the program will be planned in this manner.	The elimination of one cabin test article will save \$4.5M.
Eliminate the facility checkout vehicle.	A facility checkout vehicle is not required.	The requirements for fit check can be satisfied by the MPS article, first production articles, static article, and some small check plates.
Eliminate the fatigue test article.	A dedicated fatigue test article is not required.	Fatigue testing will be performed on these elements for which the design was dictated by fatigue criteria. The elimination of the fatigue test article will save \$33.2M.
Build one propulsion test article, phased to support development of MPS, APS.	Use separate articles.	Interdependence of operations creates an excessive schedule risk. Operations flexibility minimized.
Use a modified MPS test article.	The MPS test article will utilize flight tanks, ducting, MPS and aft boat tail assembly and heavy duty structure to replace the orbiter fuselague.	The modified MPS will save \$31.8M. All test objectives can be met.
Use MPS test article after the test program to: 1. Support T/V chamber tests. 2. Fulfill fatigue test program. 3. Deliver as Orbiter 3.	Concept not to be implemented. The engines will be refurbished after the MPS test program and used as spares.	Large section T/V chamber testing is not required. Fatigue test article is not required. The MPS will be a modified configuration.
Review thermal control article.	Scope of TCS is reduced by external tanks. Vent and purge tests are reduced in scope. Delete vehicle purge tests from MPS cluster; MPS purge only.	Integrated thermal control testing reduction parallels vehicle design changes. Save \$.48M.
Review hydraulic/power article.	Maintain as in Phase B. Locate in same area as landing gear and turbo for articles to share personnel.	Maintain operational flexibility.
Eliminate the flight readiness static firing on Orbiters 4, 5.	Static fire (flight readiness) Orbiters 4, 5.	Required for functional verification of MPS. No significant cost savings.

Table 3-34. Summary of Phase B Baseline Reevaluation (Cont)

Item	Conclusion	Rationale
FACILITIES		
Consider one site for all propulsion testing.	Use approach per Phase B.	Maximize use of existing facilities. Provide operational flexibility.
Consider one test site for all subsonic and vertical flight testing.	Use EAFB and KSC as per Phase B.	Use of existing EAFB facilities allows deferment of dollars for KSC facility modifications. EAFB provides close proximity to design and assembly, and extensive safety features.
Consider single site for final assembly, HFT, VFT, M&R, etc.	Use approach per Phase B.	Maximizes use of existing facilities. Allows deferment of dollars for KSC facility modifications.
SUPPORT EQUIPMENT		
Adopt Saturn static firing pad for orbiter.	Feasible if MPS and flight readiness tests not at KSC.	MTF SIC stand can support firing schedules. Aerodynamic surfaces of orbiter must be removed for clearances.
Adopt Saturn propellant loading system for orbiter.	The use of the Saturn PLS is planned in the baseline. Maintains existing maximum flow rate.	Cost deferral can be obtained if the Mark I booster is not LO_2/LH_2.
Determine the facility/support equipment impact of orbiter assembly at the operations site.	Use Phase B baseline.	Duplication of major support equipment items to un-process manufacturing. Reallocation of large number of manufacturing personnel to operations site. Limited manufacturing fabrication rework capabilities at operations site.
Use Apollo ACE for Gen 1 and 2 orbiter checkout support.	Use Phase B baseline.	Extensive refurbishment required. Lower reliability of equipment. Two modifications required: CSM to Gen 1 Gen 1 to Gen 2. Requires more operational manpower.
Reduce ACE to SE control and monitor, and computer memory information retrieval for Gen 2.	Use the same system Gen 1 and Gen 2. Use Phase B baseline.	Higher total program cost. Schedule interference for Gen 2 equipment installation and certification.
Maintenance support equipment is modified Apollo BME for Gen 1.	Use Phase B baseline.	Refurbishment and modifications required to Apollo BME. Test and Control data gathered not compatible with UTE.

3.12 SCHEDULE AND COST OPTIONS

The most desirable system options were selected and program schedules developed to determine the near-term, peak annual and total funding associated with each. Funding and development schedule characteristics are presented for the low technology orbiter system with a typical interim booster (120-inch SRM) which would have a Generation 1 first manned orbital flight in September, 1978. Principal Generation 1 system variables considered were the expendable booster, the launch rate of the interim operational period, orbiter engine, cargo bay length, and payload weight capability. These Generation 1 systems were then phased into the second generation by one or more of the following: (1) upgrading the cross range and payload weight performance (all cases), (2) increasing the payload bay size, (3) changing the main engine, and (4) including the use of a reusable heat sink booster (RHSB). The principal Generation 2 system variables considered in deriving program cost profiles and development schedules were the date of first manned orbital flight (FMOF) of the second generation system and the number of operational orbiters and boosters (Figure 3-107).

Generation 1 program requirements are associated with developing a low-cost and low-risk program with adequate capability to demonstrate the key shuttle features associated with payload operations and low-cost turnaround (Figure 3-108). Achieving low cost includes minimizing the 1973 and 1974, as well as the peak, funding requirements. In addition, it is desirable that a Generation 1 system (orbiter and booster) combined with the Generation 2 system result in a low total program cost. The requirement for least risk implies the maximum use of existing subsystems and technology on the smallest vehicles. Further, the Generation 1 system should be able to grow easily to the full capability needed to ensure obtaining the shuttle benefits. This growth includes the compatibility of the orbiter with the RHSB to ensure

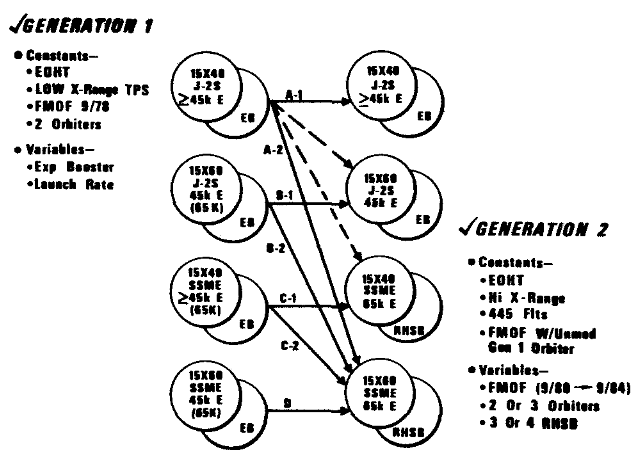


Figure 3-107. Options Exercised

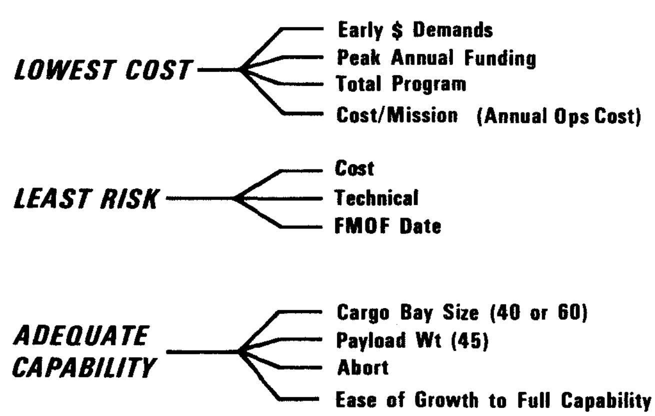


Figure 3-108. What Are We Really Looking For?

low refly cost during Generation 2, as well as the growth potential to payload weight and size of 65,000 pounds and 60 feet long. This cargo bay size and weight capability are needed to obtain the benefits associated with reusable synchronous orbit payloads and reusable orbit-to-orbit shuttles.

The assumed peak annual funding (PAF) for the shuttle is $1 billion. The estimated government program support which is assumed to be included within this constraint is approximately 20 percent of the total funding. Thus the PAF requirements associated with the total contracted funding of the shuttle are approximately $800 million (including government facilities) with the profile shown in Figure 3-109.

Preliminary cost estimates of expendable boosters compatible with the Generation 1 performance requirements with an EOHT. 15- by 40-foot cargo bay orbiter are shown in Figure 3-110. From a program standpoint the difference between the various expendable boosters for an interim program of 12 launches is not significant. The 120-inch SRM is used in this section as a typical expendable booster in comparatively evaluating the options considered.

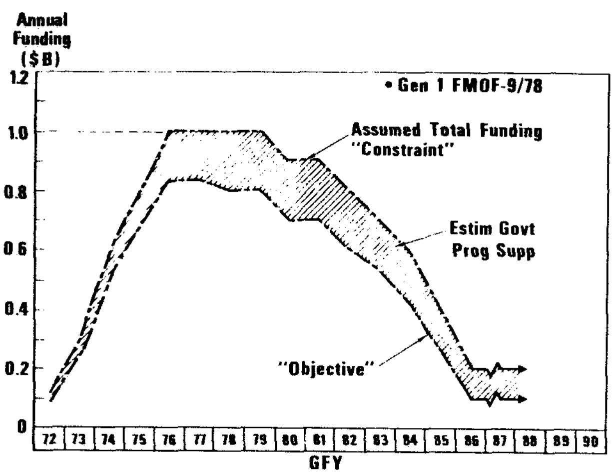


Figure 3-109. Funding Requirements Versus "Targets"

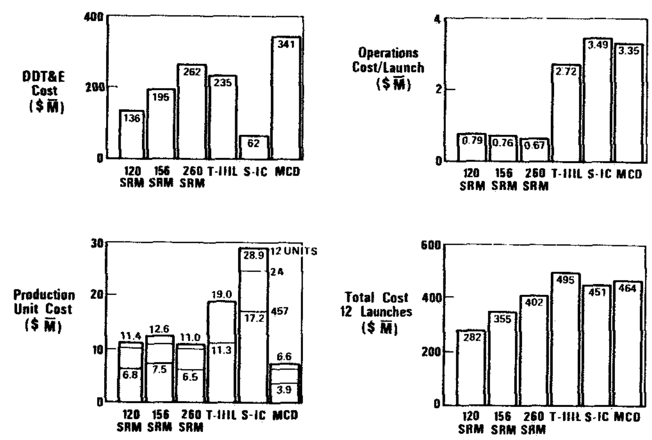


Figure 3-110. Expendable Booster Costs

The key results of the cost and schedule study are shown in Table 3-35. Note that the only two options within the peak funding "objective" (i.e., allowing for government program management and support) exclude the development of a reusable heat sink booster (RHSB) and the high P_c engines. These two options, in addition to having very low early and peak funding requirements, have the lowest total program cost for the 445 flights. Note, however, that continued operation of either of these two generation systems beyond 1985 results in a high annual cost (over $500 million) because of the high cost associated with the expendable boosters. Also note that the options which require orbiter block changes to increase payload bay size (A-2, C-2) or orbiter engine (B-2) result in higher total program cost than the options which do not require an orbiter block change (C-1 and D).

The time-phased funding requirements are compared with the assumed funding target in Figure 3-111. System and subsystem development and production start dates and span times were selected to minimize peak annual funding (PAF) and to avoid large changes in total funding requirements. (Note that the start dates shown in previous sections of this summary indicate the latest point in the program that a particular element can be initiated rather than that required to minimize PAF.) As illustrated, none of the options satisfy the funding objective for the total program duration from 1970 through 1989. The expendable booster Generation 2 option exceeds the funding objective from 1984 and beyond, while those options which include a reusable heat sink booster violate the funding objective from 1974 through 1981.

Table 3-35. Key Results-Cost/Schedule Study

Program Option	Early Funds Reqmts		Peak Funding		Total Program Cost	Approx Annual Cost >1985
	FY 73	FY 74	Amount	GFY		
A-1	$226M	$519M	$724M	1976	$9.62B	$500M
A-2	312	688	1080	1977	11.33	165
B-1	234	535	736	1976	9.72	500
B-2	302	689	1049	1977	11.16	165
C-1	325	704	1017	1977	10.85	165
C-2	343	739	1060	1977	11.15	165
D	333	719	1025	1977	10.95	165

Fee & Govt Prog Support Excluded From Above Figures

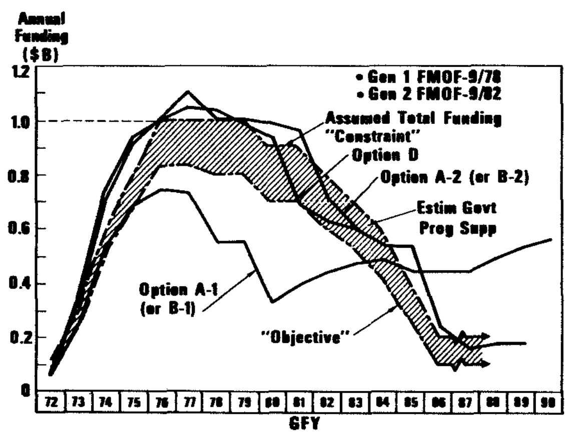


Figure 3-111. Funding Requirements Versus Target and "Objectives"

An attempt was made to meet the funding constraint with those programs which include a PHSB by deferring the FMOF of the Generation 2 system. The second generation FMOF was varied from September 1982 to September 1984. Although the objective of $0.8 billion (excluding government program support) was not met, deferring the Generation 2 FMOF can be used to reduce the total peak funding by $100 to $150 million to approximately $0.9 billion.