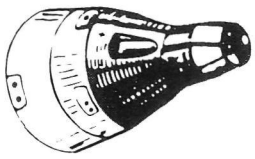
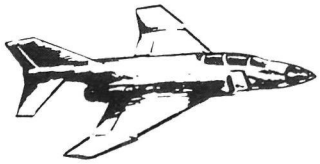


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MDC E0558

Phase B System Study Extension

FINAL REPORT

Part II - Volume III

TECHNICAL REPORT

MMC ACTIVITY

HIGH VALUE

SPACE SHUTTLE

LOW COST



(NASA-CR-143908) SPACE SHUTTLE PHASE B SYSTEM STUDY EXTENSION. PART 2, VOLUME 3: MMC ACTIVITY Final Report, 1 Nov. 1971 - 15 Mar. 1972 (McDonnell-Douglas Corp.) 57 p

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SPACE SHUTTLE PHASE B SYSTEM STUDY EXTENSION FINAL REPORT

15 MARCH 1972

MDC E0558

Part II - Volume III

TECHNICAL REPORT MMC ACTIVITY

Submitted Under NASA
Modification 24
Supplemental Agreement
to Contract NAS 8-26016
CDRL Item MA-04



FOREWORD

The Space Shuttle Phase B Extension Study was conducted under Modification 24, Supplemental Agreement to Contract NAS 8-26016 by a team headed by the McDonnell Douglas Corporation. The period of performance for the study was 1 November 1971 through 15 March 1972. Members of the study team included the McDonnell Douglas Corporation, the Martin Marietta Corporation, TRW Incorporated, Pan American World Airways Incorporated, and Sperry and Norden as subcontractors.

This document constitutes Part II Volume III of the final documentation for the study which consists of the following:

CDRL Item MA-04 Final Report
- Part I Executive Summary
- Part II Technical Report
  - Volume I System and Orbiter
  - Volume II System and Booster
  - Volume III MMC Activity
- Part III Development Requirements
- Part IV Cost and Scheduling Data

CDRL Item SE-03
- Part V Design Data Book
  - Volume I Program and System Baseline
  - Volume II Drawings
  - Volume III Orbiter Aero Design Data Book
  - Volume IV Booster Aero Design Data Book (CDRL Item SE-01)

CDRL Item SE-04
- Part VI Final Mass Properties Report

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INTRODUCTION

This volume of the technical report summarizes the booster design activities conducted by the Martin Marietta Corporation during the second extension to the Phase B Shuttle subcontract to McDonnell Douglas Astronautics Company (MDAC) under PO YO3083, NASA Prime Contract NAS8-26016.

The work conducted since December 15 on tank end-loaded pressure-fed and solid rocket motor boosters has received special emphasis in Sections 3.0 and 4.0, but very brief summaries of the retroglide and flyback booster studies are presented in Sections 1.0 and 2.0. The winged booster work on the retroglider and flyback canard was closed out in December and was comprehensively presented in briefing chart booklets dated December 2, 1971, and December 15, 1971, respectively. The retroglide concept was also issued as Design Note B-MMC-CD-5002, dated December 2, 1971.

1.0 Retroglide Booster Studies

1.1 System Description

The retroglide booster is a concept* that utilizes excess fuel capacity of the S-IC booster to perform a boost-glide return of the S-IC to its launch site following release of a Space Shuttle second stage. The booster mission duration is less than 20 minutes from liftoff to landing and is performed entirely within view of the launch site. Aerodynamic heating during ascent, retroboost, and reentry is within allowables for aluminum structure over more than 90% of the vehicle surface. The Saturn structure, tankage, and propulsion system require about 20% modification, with more than 90% use of existing tooling.

Additions to the S-IC structure consist principally of nose, wing, and directional fin structures; the nose and wing structures incorporate the landing gear and aerodynamic control systems. The main engines are modified to incorporate step throttling to 1.0 million pounds equivalent sea-level thrust to limit the acceleration during boost and stage separation.

Figure 1.1-1 illustrates the booster flight profile. Following normal ascent during which the main engines are successively throttled back, stage separation is accomplished by shutting down the four outer booster engines and igniting the orbiter engines. The staging ordnance is fired and the orbiter is accelerated ahead of the booster by a solid-rocket assist system attached to the drop tank. Following separation the booster is rotated 145 degrees to its retrofire orientation at which time the throttled center engine is advanced to maximum thrust. The retro impulse lofts the booster to an apogee of 413,000 feet about 66 nautical miles downrange (maximum downrange excursion is 72 n.m.); the booster then reenters at maximum lift to a nominal uncontrolled impact point located away from the launch site as a safety measure. Aerodynamic forces result in an entry pullout above 50,000 feet at transonic speed, followed by aerodynamic glide and deceleration to subsonic speed. The nominal

*_Retroglide Booster Concept, A Briefing Booklet._ Design Note B-MMC-CD-5002. Martin Marietta Corporation, Denver, Colorado, December 2, 1971.

trajectory ends at 25,000 feet over the recovery site. A circling high-energy landing approach is made under automatic control, utilizing the technique developed and demonstrated by NASA-FRC and USAF at Edwards Air Force Base.

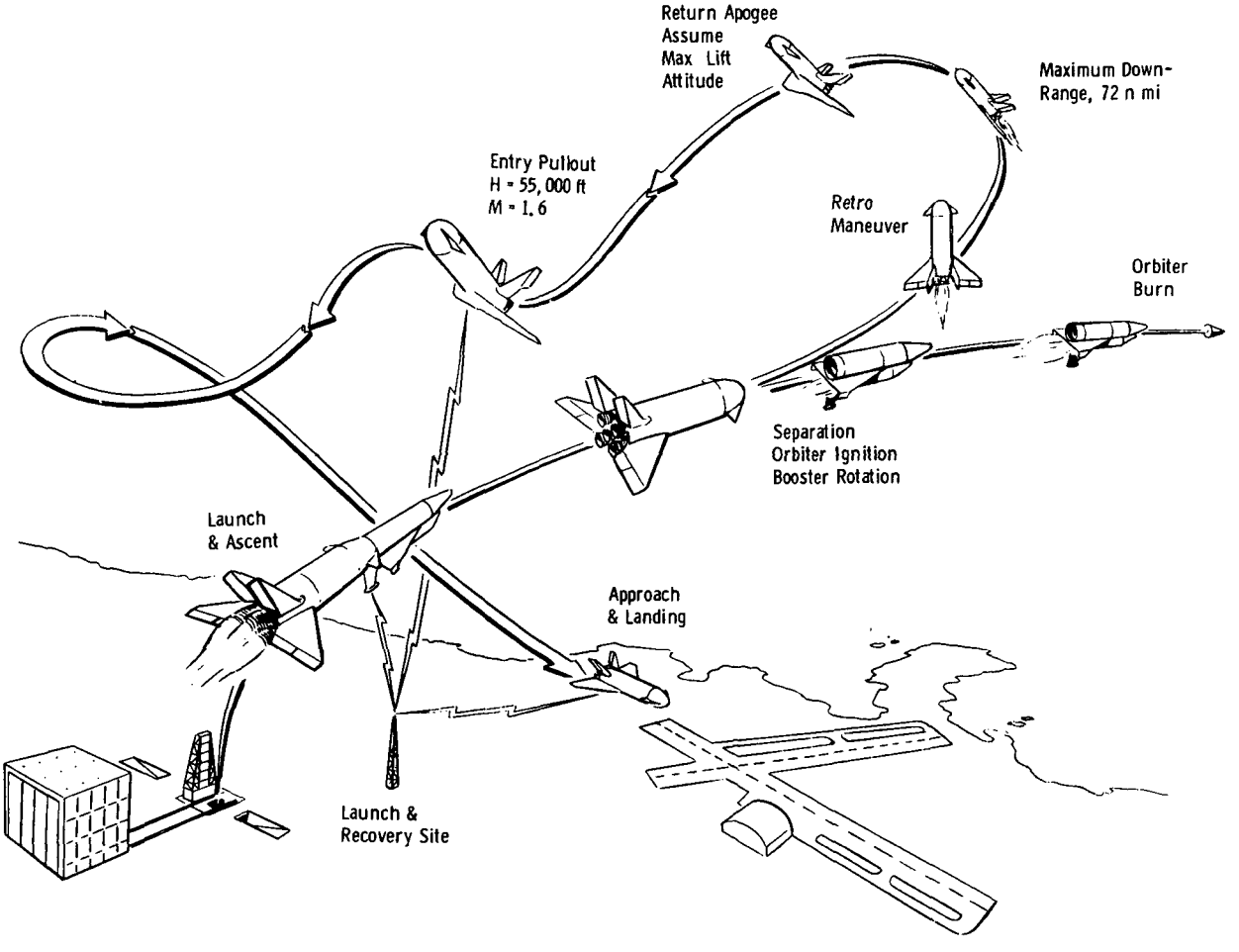
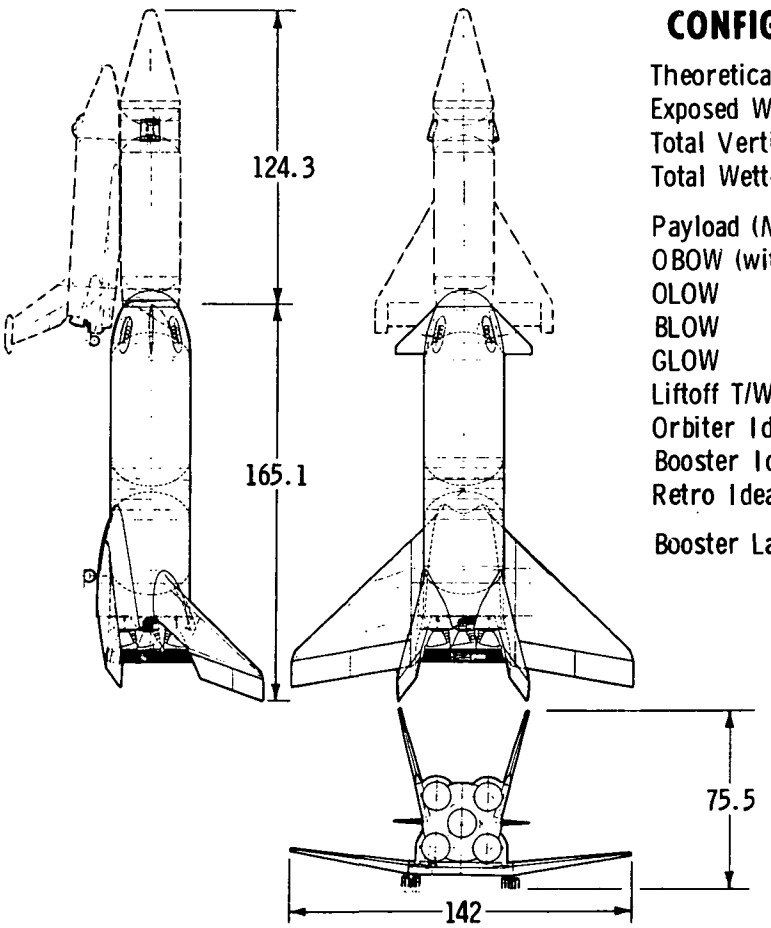


Figure 1.1-1 Retroglide Booster Flight Profile

The booster configuration is shown in Figure 1.1-2, with the orbiter and drop tank shown in phantom outline in the launch configuration. The wing and canard are sized to considerations of landing (10,000 ft FAA runway rating) and longitudinal stability and control throughout the re-entry flight regime.



CONFIGURATION DEFINITION

Theoretical Wing Area	6,000 sq ft
Exposed Wing Area	3,890 sq ft
Total Vertical Tail Area	1,291 sq ft
Total Wetted Area	25,800 sq ft
Payload (Mark II Polar)	40,000 lb
OBOW (with Drop Tank)	252,329 lb
OLOW	1,085,640 lb
BLOW	5,126,620 lb
GLOW	6,212,260 lb
Liftoff T/W	1.225
Orbiter Ideal ΔV	21,080 fps
Booster Ideal ΔV	10,396 fps
Retro Ideal ΔV	6,260 fps
Booster Landing Wt/cg	488Klb/70.5%

Figure 1.1-2 Retroglide Booster Configuration

1.2 System Performance and Mission Profile

The retroglider can perform the mission over a fairly wide range of staging ideal velocities. Figure 1.2-1 shows the variation in gross liftoff weight (GLOW) required to perform the polar mission with a 40,000-pound payload using the high chamber pressure orbiter engines. The upper limit on staging velocity is determined by the GLOW resulting in the minimum desired thrust-to-weight (T/W) ratio at liftoff or the S-IC tank capacity. For this configuration, the tank capacity and the GLOW limitation both occur at an approximately 10,800-fps ideal staging

velocity. The booster propellant for both ascent and retro increases with staging velocity while the orbiter propellant decreases.

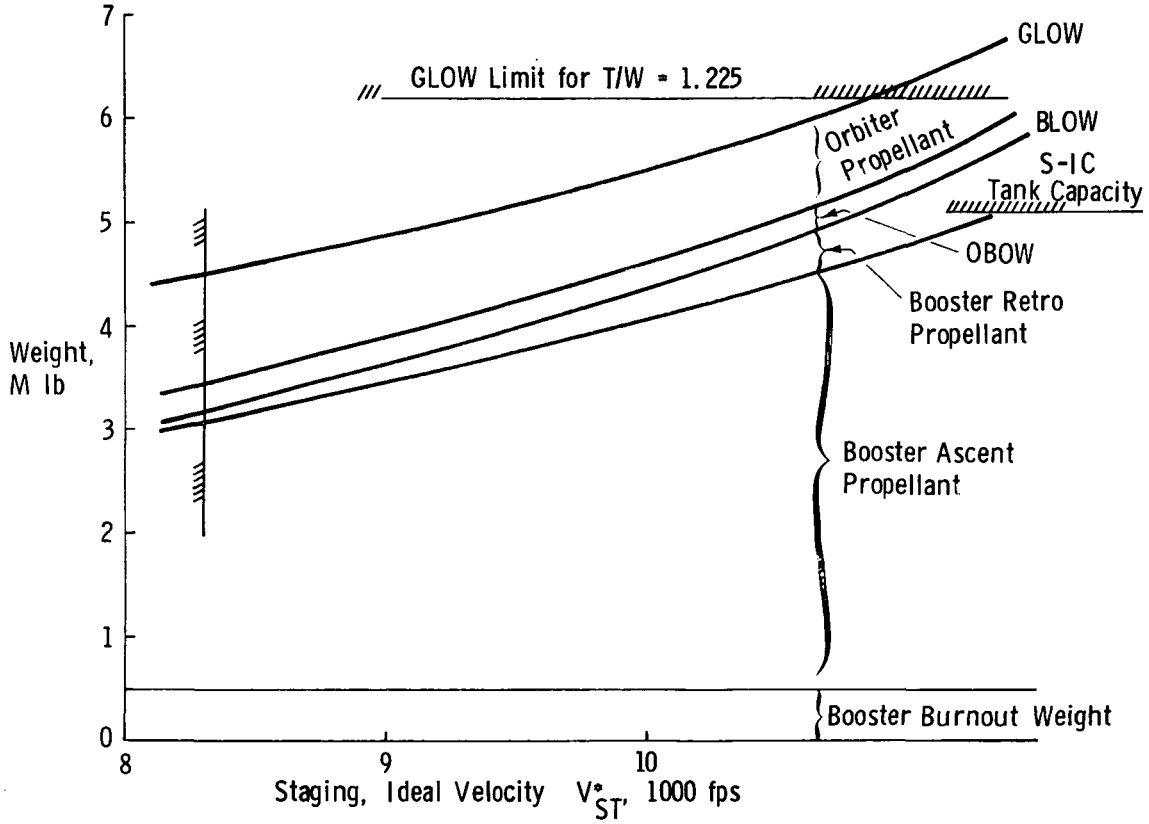


Figure 1.2-1 GLOW Variation with Staging Velocity

The lower limit on staging velocity results from the T/W of the orbiter. This limit is picked at 0.9 and occurs at a staging velocity of about 8300 fps. These data are based on a fixed booster weight with propellant off-loaded for the lower staging velocities. More performance margin is available if the configuration is designed for a low staging velocity.

1.2.1 Mission Profile

The range/altitude variations shown in Figure 1.2.1-1 illustrate the retroglide mission profile. The total vehicle lifts off at T/W = 1.225 and flies to a staging point at 187,000 feet and 35 nautical miles downrange. The relative velocity is 5860 fps and the flightpath angle is 28.1 degrees. The high staging flightpath angle is somewhat higher than normal to reduce the impulse required for the retro maneuver.

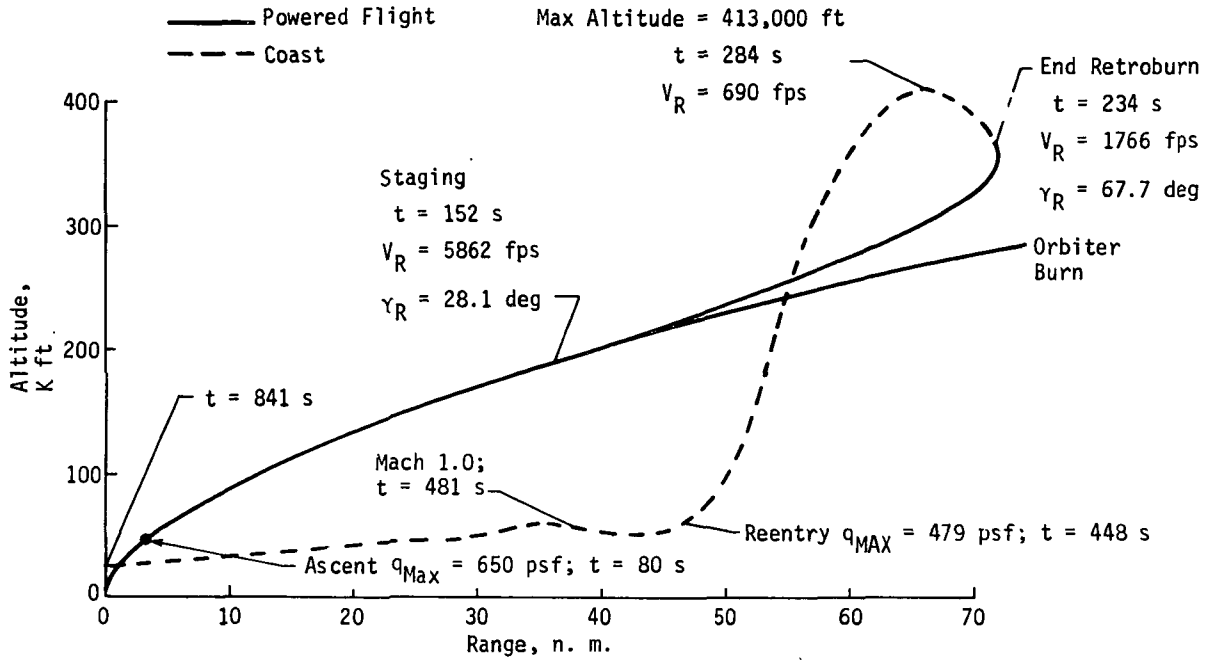


Figure 1.2.1-1 Booster Altitude/Range Profile

After staging, the orbiter continues into orbit and the booster is re-oriented to a pitch attitude of approximately 160 degrees (20 deg above the horizontal with the vehicle pointed back toward the launch site). During the retro maneuver, the booster reaches a maximum downrange of about 72 nautical miles. The state vector at booster burnout allows the vehicle to coast through apogee and glide to the landing site.

1.3 Cost Summary

Cost studies indicate that the retroglide booster can be developed for approximately $480 million less than the minimum-cost flyback configuration. These reduced RDT&E costs are attributable to the elimination of the cruise propulsion system, crew compartment, and life support system, the reduced thermal protection system requirements, and the overall lower system weights. Inherent in these differences are the elimination of significant risk factors associated with the development of the cruise engines and the development of a thermal protection system for the more stringent reentry environments of the flyback system.

Retroglide booster operational costs are also predicted to be lower than the flyback booster by about $66 million due to elimination of the cruise propulsion system, crew compartment, and life support systems, all resulting in lower ground operation, maintenance, and spares costs.

1.4 Conclusions

The concept is feasible. The principal feasibility issue is staging the orbiter from the booster. Alternative solutions to most problems (including the separation problem) are available. Existing technology can be used to a greater extent than for any other fully recoverable system, thus improving confidence in the feasibility of the concept. Costs are approximately 18% less than for a comparable air-breathing flyback system. The concept is applicable not only to the reusable S-IC, but to any other propulsion system contemplated for a winged recoverable booster.

2.0 Flyback Booster Study

2.1 System Description

The flyback booster concept* was to make maximum use of S-IC components, designs, and tooling to minimize development cost. This is accomplished by using the existing S-IC engines (five F-1 engines), propellant tanks ,and thrust structure.

The major modifications of the S-IC structure consist of adding a cradle-type wing carrythrough structure, "rabbit-ear" fins, aero canard, ACPS, and crew compartment. The cradle or underslung assembly consists of a composite structure that includes the wing, main landing gear, nine air-breathing flyback engines (ABES), and the ABES inlet door.

The booster was sized to deliver 40,000 pounds of payload to a polar orbit. A staging velocity of 6500 fps was chosen to be consistent with a heat sink booster concept. The low wing was selected to minimize entry heating and to keep from structurally penetrating the S-IC propellant tanks. The "rabbit ear" fins are effective up to a 45-degree angle of attack and provide directional stability throughout the entire entry profile. The fins' effectiveness also tends to reduce the duty cycle and ACPS propellant requirements. Since the aero canards provide trim control without degrading lift, a significantly smaller wing can be used.

The launch configuration (Fig. 2.1-1) is a tank end-loaded arrangement with the orbiter side-mounted on the drop tank with its vertical fin diagonally opposite to the booster vertical fin. This arrangement carries the thrust loads through the drop tank, minimizes rolling moment due to sideslip, and allows cancellation of pitching moments through selection of the orbiter incidence angle. The weight sequence is also tabulated for the design mission.

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*_Flyback Booster Study_. Design Note B-MMC-TI5006 (Rev A). Martin Marietta Corporation, Denver, Colorado, December 15, 1971.

†Except for shortened barrel sections.

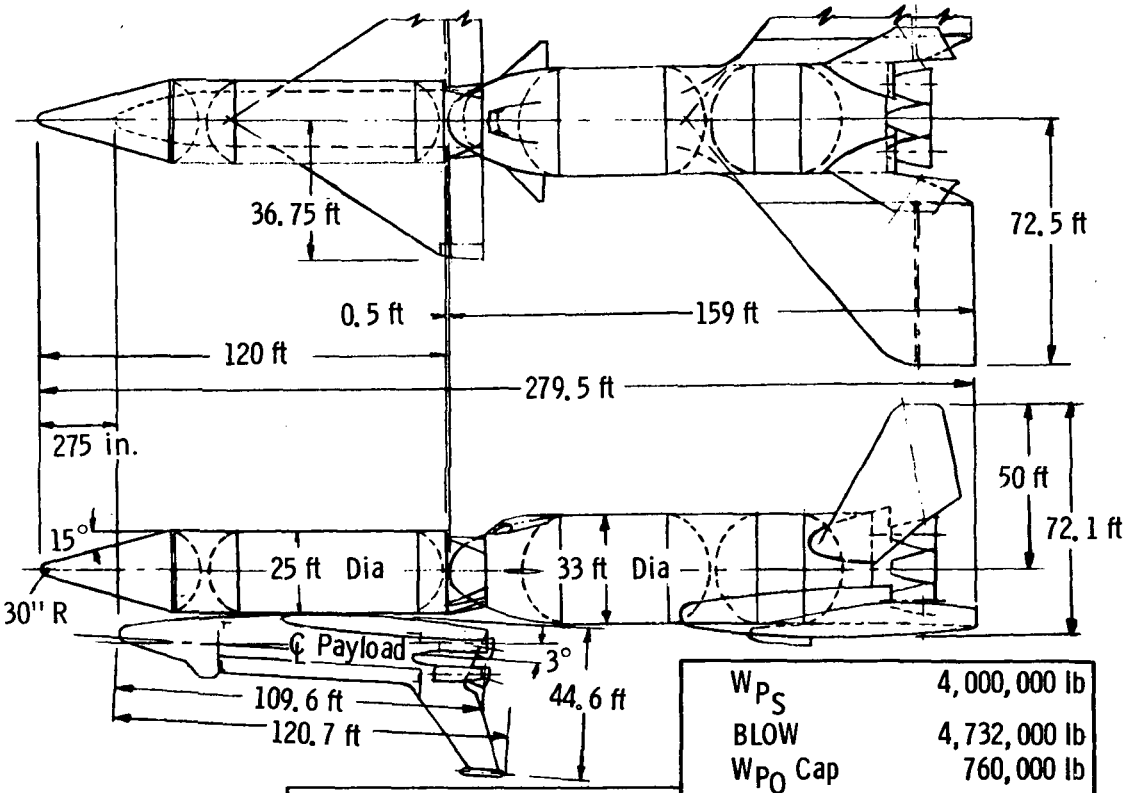


Figure 2.1-1 Launch Configuration - Aero Canard RF-1

2.2 System Performance

The use of throttling to satisfy the maximum dynamic pressure constraint ($q_{max} \approx 650$ psf) is much more effective than pure trajectory shaping. Throttling yields more desirable staging conditions for booster reentry and increases the performance capability by 550 fps (inertial velocity). The propulsion modifications required to throttle the F-1 engine in flight were not baselined and this approach should be weighed against the performance advantages and increased cost prior to making final decision.

2.2.1 Ascent Performance Parametrics

The effect of maximum dynamic pressure on the ascent performance is shown in Figure 2.2.1-1. [redacted]



Figure 2.2.1-1 F-1 Flyback Ascent Performance Parametrics (MK II Orbiter)

[redacted]

The entry is initiated at a 60-degree angle of attack until peak heating is passed, after which banking is commenced to reduce flyback range and α is modulated to reduce peak g (4.9 max). The turn is completed and subsonic flight commences at an altitude over 50,000 feet; cruise commences at an altitude of 20,000 feet, 209 miles from the launch site.

The highly lofted trajectory gives staging conditions that result in an apogee altitude of about 400,000 feet. High apogee altitudes give higher g, dynamic pressure, and heating rates during booster reentry. The trajectory can be reshaped to lessen the reentry problems and lower the apogee altitude to 300,000 feet for an additional performance penalty of 90 fps.

2.3 Cost Summary

Table 2.3-1 for the flyback booster demonstrates the effect of the configuration on total program costs. The aero canard configuration was established to offer the potential of maximum use of S-IC parts, primarily in the structures/TPS systems and in the propulsion area.

Table 2.3-1 Flyback Booster Cost Summary
(Costs in $Millions)

	Delta	Canard (Nominal)	Canard (Maximum S-IC Use)
Subsystem Development	1709	1460	1362
System Support	1261	1209	1143
Flight Test	152	152	152
Total RDT&E	3122	2821	2657
Investment	663	546	546
GFE Main Engine	18	18	18
Total	681	564	564
TOTAL PROGRAM	5019	4307	4143

The potential total program savings established by this study may be determined from the data in the summary chart. The maximum RDT&E savings potential over the MDAC baseline Delta wing configuration is $465 million, of which $164 M is attributable to design for maximum use of S-IC parts. The aero canard can reduce investment costs by $117 M, and $294 reductions have been attributed to operations, the majority associated with pure heat sink or reusable overcoat in lieu of replaceable ablators for the MK 1 booster.

2.4 Conclusions

Use of S-IC components, design, and facilities is feasible for a flyback booster system. By constraining the design to a heat sink concept and using existing hardware, designs, and facilities, it has been demonstrated that $1.855 billion in RDT&E can be saved in comparision to the performance-optimized flyback system that uses high-chamber-pressure LO_2/LH_2 main propulsion and a thermal protection system compatible with high staging velocities. The aero canard configuration also represents a lower cost option than the previously established Delta wing configuration, and could potentially reduce RDT&E costs by 14.9% and operations costs by 24.2% for a total program cost reduction of 17.5%.

3.0 Series-Burn Pressure-Fed Liquid Booster

3.1 General Configuration Description

The selected series-burn pressure-fed liquid-propellant booster is a water recoverable booster stage mounted in a tandem arrangement with the orbiter attached to the side of the hydrogen-oxygen (HO) tank as shown in Figure 3.1-1.* The booster uses RP-1 fuel from a forward-mounted fuel tank and liquid oxygen (LO_2) from an aft-mounted oxidizer tank to feed five boost engines at a mixture ratio of 2.4 (by weight). The engines are arranged in a cruciform pattern, positioned 45 degrees off the pitch and yaw axes. The center engine is fixed, and the four outer engines are hinged to permit motion in a tangential direction only.

The outboard configuration of the booster is heavily influenced by the booster reentry and recovery mode. The recovery approach is based on ballistic (zero angle of attack) entry using a passive (open-loop) entry mode in the belief that such a system offers, through simplicity, the best potential for low cost and high reliability. Subsection 3.11 summarizes the tradeoffs considered for ballistic versus lifting entry modes. The ability to passively self-capture nose-first and remain in a stable attitude during atmospheric reentry has caused the booster to be configured to a 25-foot main tank diameter with a large fixed-flare engine compartment. The addition of six extendable speed brakes enhances the capture capability and also provides stability and deceleration capability during reentry. During the terminal descent phase, altitude sensing initiates speed brake retraction and firing of retrorockets to further retard velocity in advance of water impact. Because of the water impact, the booster nose (fuel tank dome) is designed to be cone shaped to minimize initial impact forces. A common bulkhead between main propellant tanks provides a uniformly high strength body from nose to engine compartment, further strengthened by retaining booster burnout tank pressures through the reentry phase. The high strength level of the

*More detailed data regarding this configuration may be found in the following briefing booklets: *MSFC Systems Analysis/Flight Mechanics/Aero Working Group Meeting, SRAO (PFB) and (SRM)*, February 1, 1972; *MSFC Configuration and Structures Working Group Meeting, SRAO (PFB) and (SRM)*, February 1, 1972; *MSFC Propulsion Working Group Meeting, SRAO (PFB) and (SRM)*, February 1, 1972. *MSFC Design Review, Phase B System Study Extension*, February 11, 1972 by McDonnell Douglas/Martin Marietta Corp Team.

Design Characteristics
(East Mission)
GLOW = 6,057,800
BLOW = 4,806,900
OLOW = 1,250,800

Booster

Impulse Propellant	4,001,800 lb
λ'	.8325
Jettison Weight	805,100 lb
Impact Weight	759,100 lb
Isp (SL/Vac) (Min)	221/279 sec
T/W_0	1.25
Staging Velocity (VR)	4976 fps

Orbiter

Impulse Propellant (Incl. FPR)	957,200 lb
λ' (Drop Tank)	.9474
Jettison Weight	53,111 lb
Core Weight (65 K lb P/L)	240,570 lb
Isp (Min)	456 sec
T/W_i	1.12

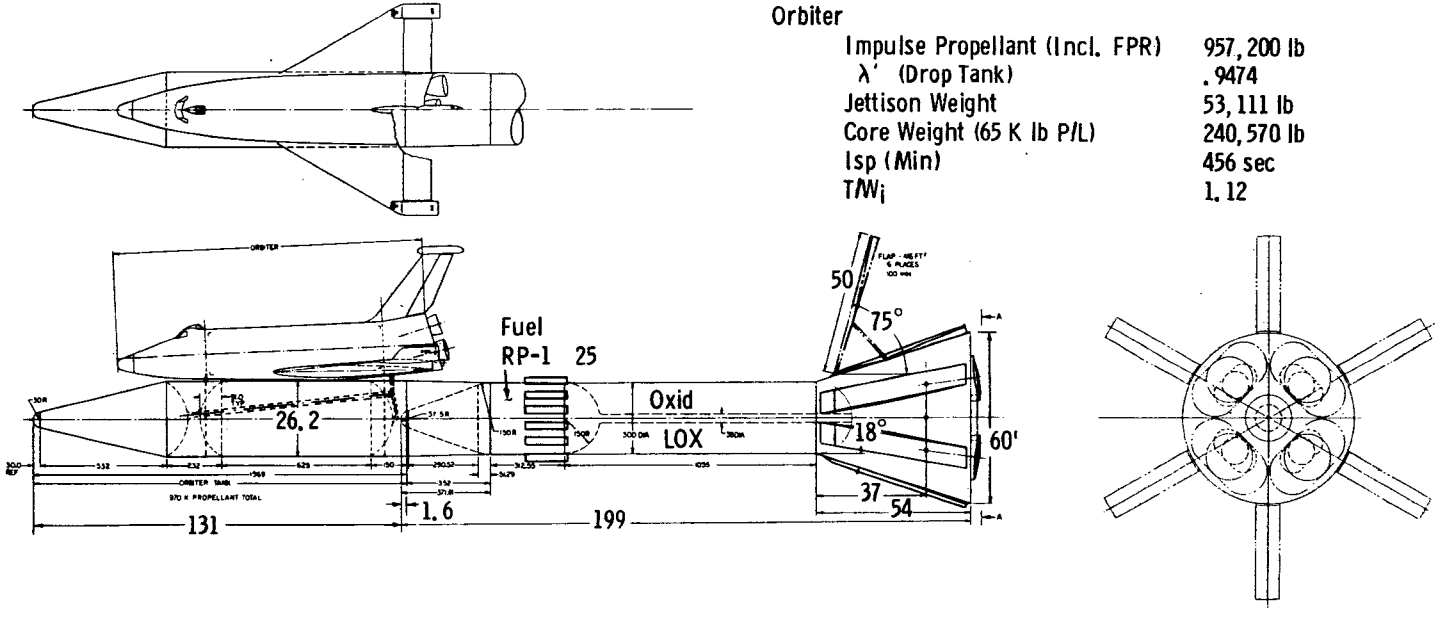


Figure 3.1-1 Pressure-Fed Booster General Description

tanks required for propellant pressurization and feed provides the means of surviving the large water impact loads. The booster engines are protected from direct water impingement loads during water entry and water rebound conditions by the engine compartment fixed flare skirt, which extends to near the engine exit plane. The moveable engines are hinged near their center of mass to have minimum design impact on engine and actuator systems under lateral inertia loads during water entry. The engine compartment is closed off with an aft bulkhead located near the engine swivel station, providing a closed, protected environment for subsystem installations.

The following sections describe in more detail the operational characteristics and the subsystem design definitions for the recoverable pressure-fed booster.

3.2 System Performance

3.2.1 Baseline Sizes and Size Sensitivities

Table 3.2.1-1 presents the configuration characteristics of a "point design" and "current" baseline configuration. The point design reflects initial weight estimates and served as a common basis for booster system and subsystem point design studies. The impact of weight changes resulting from the design studies is shown by the "current" column in Table 3.2.1-1. Staging velocity was held constant in updating the point design to the current vehicle.

The growth sensitivities for the current design included in Table 3.2.1-2 reflect two modes of growing the vehicle. For a fixed booster, the sensitivity of orbiter liftoff weight (OLOW) to orbiter and booster mass fractions and specific impulses are shown. The same sensitivities for booster liftoff weight (BLOW) are shown for a fixed drop tank. Since these sensitivities are nonlinear (especially OLOW) and baseline dependent, they should be applied with caution.

3.2.2 Mission Profile and Trajectory Histories

The mission profile for the point design baseline vehicle is presented in Figure 3.2.2-1. The engines were sized to provide a liftoff T/W of 1.25. [redacted]

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Table 3.2.1-1 Sizing Impact of Weight Changes (East Mission)

DESIGN CHARACTERISTICS		POINT DESIGN	CURRENT
GLOW	M lb	6.058	6.473
BLOW	M lb	4.807	5.222
OLOW	M lb	1.251	1.251
BOOSTER			
Impulse Propellant	M lb	4.002	4.276
λ'		0.8325	0.8189
Jettison Weight	M lb	0.805	0.946
Impact Weight	M lb	0.759	0.900
I_{sp} (sl/vac) (minimum)	s	221/279	221/279
T/W_o		1.25	1.25
Real Staging Velocity	fps	4980	4980
ORBITER			
Impulse Propellant (including FPR)	K lb	957.2	957.2
λ' (Drop Tank)		0.9474	0.9474
Jettison Weight	K lb	53.1	53.1
Core Weight (65 K lb Payload)	K lb	240.6	240.6
I_{sp} (minimum)	s	456	456
T/W_i		1.12	1.12

Table 3.2.1-2 Growth Sensitivities for Current Baseline

Constant Booster Propellant	Constant Drop Tank
$\frac{\partial\ OLOW}{\partial \lambda_{DT}} = 11 \times 10^6$ lb	$\frac{\partial\ BLOW}{\partial \lambda_{DT}} = 98 \times 10^6$ lb
$\frac{\partial\ OLOW}{\partial \lambda_{B}} = 3.8 \times 10^6$ lb	$\frac{\partial\ BLOW}{\partial \lambda_{B}} = 43 \times 10^6$ lb
$\frac{\partial\ OLOW}{\partial\ I_{sp_0}} = 10{,}600$ lb/s	$\frac{\partial\ BLOW}{\partial\ I_{sp_0}} = 87{,}800$ lb/s
$\frac{\partial\ OLOW}{\partial\ I_{sp_B}} = 7{,}600$ lb/s	$\frac{\partial\ BLOW}{\partial\ I_{sp_B}} = 65{,}300$ lb/s

[redacted]. The staging relative velocity was under 5000 fps. Trajectory histories of the ascent are shown in Figure 3.2.2-2.

The high aspect ratio drag brakes were deployed at staging and a ballistic entry followed after an apogee altitude of 261,000 feet was reached. The maximum dynamic pressure reached during entry was about 1200 psf. [redacted]

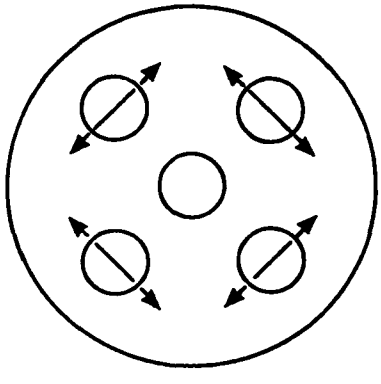


3.3 Aerodynamics

The ascent aerodynamic characteristics of the pressure-fed booster (PFB) single rocket assist orbiter (SRAO) configuration are presented in Table 3.3-1. Note that the normal force and side force derivatives are similar in value and result in unstable moments about the cg. They also are far aft for the SRAO arrangement. This instability and the rolling moment derivative impose large thrust vector control angle requirements on the booster.

[redacted] The use of chutes, retrorockets, and water rebound suppressors will be required for terminal descent, deceleration, and water impact. These, and associated flotation gear have been sized and are included in weights, but their final sizing is subject to further analysis and tradeoff studies.

[redacted]



3.4 Flight Control Comparisons

The baseline pressure-fed booster uses hinged engines and requires a ±16-degree deflection. The engine arrangement is shown. The baseline configuration does not use orbiter aerodynamic surfaces for ascent control.

Control system analyses were performed for several different pressure-fed booster configurations. Time-varying control system gains were established to provide a drift-minimum control system in pitch and yaw with a rigid-body frequency of 1.5 rad/s and 0.6 damping. Six-degree-of-freedom simulations were run to determine vehicle response to winds. Wind direction and the altitude of the peak wind were varied to determine the worst case. For some of the configuration variations, peak engine deflections were estimated on the basis of previous 6-D simulations and the ratios of aerodynamic moment coefficients and control moment coefficients.

The pitch and yaw deflection requirements for this configuration are small (less than 2 deg); the roll deflection requirements are large because of large roll moments due to sideslip. Postseparation simulations of booster flight were conducted to determine booster "capture" characteristics for reentry. The results showed that for nominal staging conditions and nominal aerodynamics the booster would capture at staging tipoff rates up to 12 deg/sec. However, an uncertainty of 5% in cp-cg distance resulted in "backward" capture at rates of 8 deg/s. Actual tipoff rates may be in the 4 to 10 deg/s range.

If the aerodynamic characteristics of the booster (C_m as a function of α over a 360-degree α range) are such that only one stable capture point exists (including cp and cg tolerance), then satisfactory capture is assured. If more than one stable capture point exists, then much more detailed analysis would be required to determine capture probability, considering uncertainties in cp, cg, staging tipoff rates, staging dynamic pressure, etc.

It is recommended that more aerodynamic configuration activity be devoted to tests that will assure a single trim point to avoid any question concerning the need for an ACPS to capture.

3.5 Structural Design

The structural concept of the pressure-fed booster is based on pure monocoque propellant tanks, sized for the internal pressures required for feeding propellant to the booster engines. To this structure is added an aft engine mount/launch pad interface structure and a forward adapter structure to interface with the orbiter HO tank. Installation of subsystem components and modification for water recovery requirements completes the design. Our preliminary design studies have identified

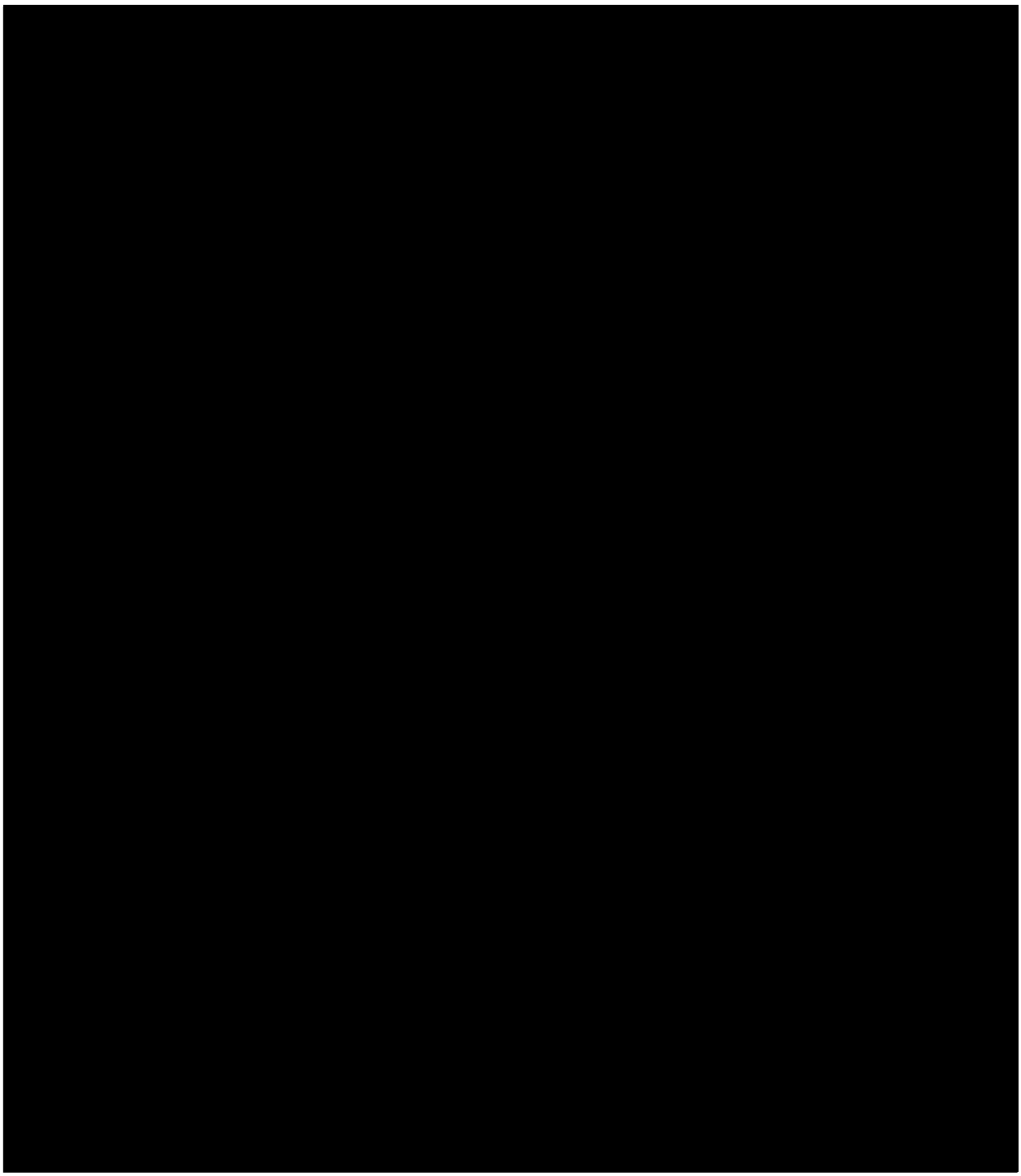
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two design areas that are critical to the feasibility of this structural concept. The first critical issue concerns the material selection for fabrication of the main tanks, especially for the cryogenic LO_2 tank. The second issue concerns the structural requirements and design impact of the reentry/recovery phase environment.

The basic tank wall thicknesses are derived from internal pressure requirements. Figure 3.5-1 presents the development of the design pressure curves, based on an engine operating chamber pressure of 250 psia. Combining chamber pressure with the pressure losses contributed by the feed system, engine cooling, and injection losses, the fluid head pressures, and the design factors of safety, the ultimate design pressure levels of the tanks are in the 500- to 600-psi range. These pressure levels in such large tanks make the selection of tankage material extremely critical to stage efficiency and fabricability. Initial material surveys have converged on three candidates that appear to best fit the design requirements. However, each has its disadvantages. Figure 3.5-2 summarizes the tank design impact of each of the candidates--maraging steel Grade 200, aluminum alloy 2219-787, and Inconel 718. The baseline design configuration has a common dome. Additional tradeoff studies will determine the final selection of common versus separate domes; if separate domes are selected, the use of different materials for each propellant tank may be preferred. More detailed data must be generated in the critical problem areas for each of the material candidates before an intelligent selection can be made. In the interim maraging 200 has been used as the point design baseline, even though aluminum tanks appear to be lowest risk.

The pressurized tank assembly with a common dome provides an inherently strong body to withstand applied loads. Preliminary body loads have been estimated for conditions believed to be potentially critical to booster design. These conditions include prelaunch, launch, maximum ascent phase airloads, maximum ascent phase acceleration, and recovery phase water impact loads. These loads are summarized in Figure 3.5-3 as ultimate line loads per inch of shell circumference versus booster station. The load curves show the combined effect of axial load and bending moment. The water impact loads dominate all other load conditions through the booster structure, except for the engine compartment.

The ultimate load capability of the pressurized tank, assuming booster burnout pressures retained, exceeds the impact applied load but by only a small margin.



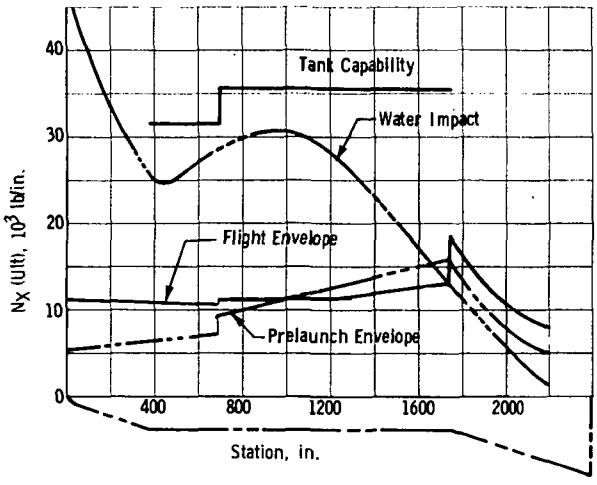


Figure 3.5-3 Design Loads Summary

A second critical load imposed by water recovery (not shown in Fig. 3.5-3) is the locally distributed pressure loads that occur along the body in various magnitudes and distributions both during initial impact and penetration and during rebound and the "slapdown" phase of water recovery. The preliminary design levels are shown in Figure 3.5-4. The major concern of this type of load is the asymmetrical aspect of the pressure distribution around the booster circumference. The high-pressure loads on the nose section caused by entry attitude dispersions will require additional fuel tank nose section stiffening in the form of several internal ring frames. The slapdown phase pressure of 20 psi applied asymmetrically to the main body and engine compartment has been selected as a reasonable level for body design. Implicit with the 20-psi value is the requirement to provide a rebound retarding device on the booster to slow the buoyant rebound velocity that could cause pressures far above 20 psi if not impeded. Our baseline ballistic entry booster configuration has weight allowances for ring reinforcement for initial impact loads as well as for a rebound retarding mechanism.

In addition to the tank assembly, there are three major structure assemblies in the booster stage--the orbiter adapter, the engine compartment, and the speed brake. These are all made of aluminum 7075-T73 alloy skin, stringers, and frames, using mechanical joining methods.

The adapter is an expendable structure that is jettisoned shortly after booster/orbiter separation. The separation sequence is shown in Figure 3.5-5. The adapter is first released from the booster and pulled away from the booster by the orbiter. After clearing the booster nose, it is released from the orbiter. Rotational restrainers prevent the adapter contacting the orbiter during the staging event.

The engine compartment is a conical aluminum semimonocoque structure that bolts to a short skirt on the tank. An internal steel truss supports the center engine and the inboard hinge pin of the four hinged engines. Eight major longerons distribute thrust loads and launch mount loads into the shell. Five major ring frames support speed brake hinge, actuator, and linkage loads, engine thrust loads, and heat shield loads. Twelve secondary longerons distribute speed brake longitudinal loads. A sealing bulkhead with base heating insulation closes the compartment near the hinge plane of the engines, providing a controlled environment for subsystem installation.

Six speed brake panels, each 100x600 inches, provide a total brake area of 2500 ft^2 for entry deceleration (Fig. 3.5-6). During ascent they are folded against the fixed engine compartment. After staging, pneumatic actuators extend an overcenter linkage to deploy the panels to their entry configuration. Terminal deceleration is provided by retrorockets located externally near the common dome. The speed brakes are then retracted, initiated by the retrorocket ignition signal. During retraction, aerodynamic loads assist, and inertia loads resist, panel closure. Closure time is short (less than 4 s); the closure condition will design the actuation system.

[redacted]

3.6 Propulsion Features and Tradeoff Studies

3.6.1 Pressure-Fed Booster Engine

The baseline engine shown in Figure 3.6.1-1 features a full regeneratively cooled chamber and nozzle, integral hinge attachments, and utilizes propellant line bellows for hinge movement accommodation.

Other than the absence of hinge provisions on the center (fixed) engine, all five engines in the cluster are identical and have self-contained throttle controls and a hinge angle capability of ±22°. The nominal performance is indicated.

Sea Level Parameter	Sea Level	Vacuum
Thrust, M lb	1.52	1.93
I_{sp}, s	225*	283*
MR, --	2.4	
P_c, psia	250	
A_e/A_t, --	6	
Weight, lb	24,600	

*Baseline flight mechanics analyzed with $I_{sp_{sl/vac}}$ = 224/282 s (MR = 2.6).



F

3.6.2 Thrust Vector Control (TVC)

Several TVC methods were evaluated for this study--gaseous injection, liquid injection, and gimbal and hinge. The cg hinge system of Figure 3.6.1-1 was baselined as a viable candidate that met the high control requirements as well as the water impact requirements of ruggedness and water closure. Figure 3.6.2-1 compares hinged engines, gaseous injection TVC, and gimbaled engines. The GI TVC and LI TVC systems were eliminated due to the inability to efficiently satisfy the high control requirements. The head end and cg gimbal systems were found to be viable for configurations requiring six or seven engines.

ENGINE-RELATED CONFIGURATION IMPACT (5 VS 7)

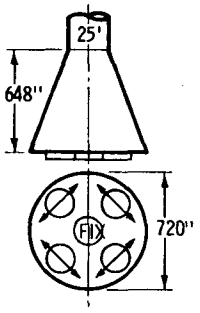
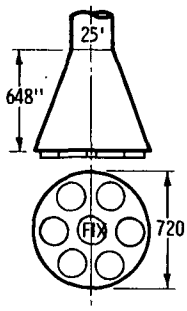
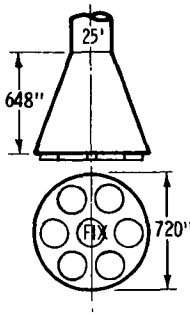
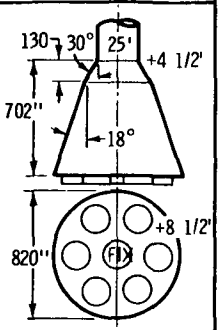
	Point Design (Hinged - One Plane)	Fixed (GITVC)	CG Gimbal	Head Gimbal
Body Diameter Flare Length Flare Diameter				
Geometry Basis	Clears ±22° Hinge	Fixed	Slaved - Clears Fail Null	Slaved - Clears Fail Null
BLOW	Point Design	+465K 8 sec Isp Loss 50K lb Increase Wt	+130K (45K lb Increase Wt)	+115K (40K lb Increase Wt)
δ Limit / δ Reqd (with Orb Surfaces)	±22°/±16° (±10°)	±7° Practical/±8° (±5°)	±12°/±8° (±5°)	±12°/±8° (±5°)
Inert Wt	Point Design	+50,000 lb • Valve Manifolds • Extra Tank Volume	+45,000 lb • Gimbal Rings • Actuators	+40,000 lb • Actuators • Flare Size

Figure 3.6.2-1 TVC for Pressure-Fed Booster

3.6.3 Throttling

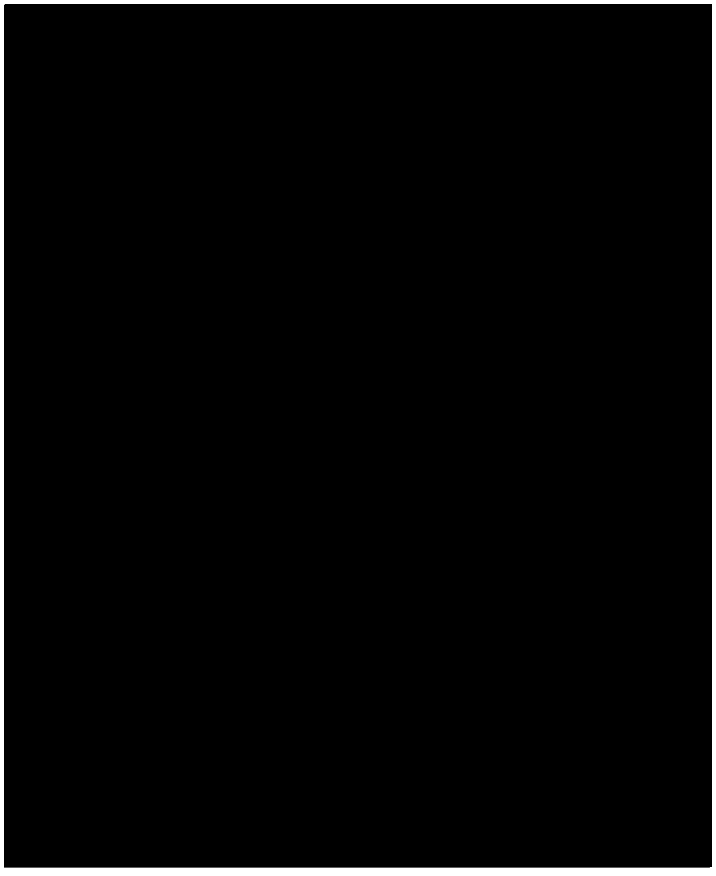
[redacted]

This technique required that the center engine be capable of withstanding 82 seconds of base heating and that throttlability be an engine requirement. The fully regenerative nozzle and the 3-g constraint need further investigation.

3.6.4 Pressurization System

Several pressurization system concepts were studied before the baseline shown in Figure 3.6.4-1 was selected. Cold gas systems were excessively heavy and volume limited. Hot gas systems using engine heat exchangers or independent heat exchangers were compared. Although engine heat exchangers offer a readily available source of energy and were compatible with pressurization system energy requirements, the increased development risk and complexity associated with the engine interface made them undesirable. Combinations of independent heat exchangers and gas generators were examined from several design standpoints, including development risk, simplicity, cost, reusability, and weight.

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The recommended pressurization system uses decomposed N_2H_4 for the RP-1 tank pressurant. Heat from the N_2H_4 decomposition process is used to vaporize LO_2 for the LO_2 tank pressurant. This system minimizes additional fluids for pressurization and does not present a water condensation problem in the LO_2 tank.

3.6.5 APU for Pressure-Fed Booster

The auxiliary power unit consists of two independent modules that derive power from the catalytic decomposition of hydrazine. Each of the two modules consists of a 252-SHP turbopower unit, a gearbox, a 200-HP hydraulic pump, and a 14-HP electric alternator.

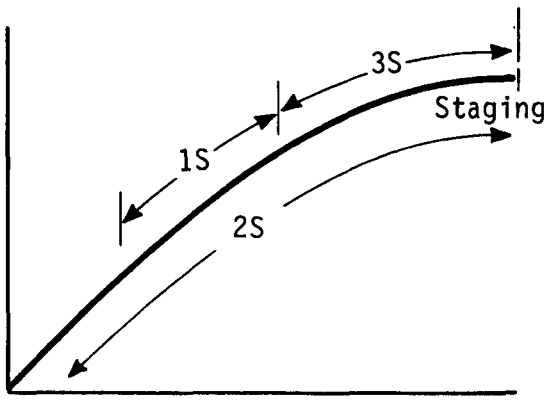
Each APU module can supply the total booster power requirement and is completely autonomous, having its own pressurization and propellant supply. Positive propellant expulsion will be achieved through the use of a bellows or bladder storage tank.

3.7 Mass Properties

Tables 3.7-1 and 3.7-2 and Figure 3.7-1 summarize the point design weights and mass property data that were used by the design groups in the performance, flight controls, loads, and structural analyses. As a result of performing these analyses, it was concluded that the weight and mass property data should be revised or updated to obtain a consistent baseline configuration. The recommended modifications and revised weights for the updated configurations are also shown.

3.8 Reliability, Safety, and Abort

The alternative intact abort approaches for ascent are shown,



where

1S = shut down booster engines, separate orbiter from drop tank, glide to landing strip,

2S = shut down booster engines and use impulse system on drop tank to separate tank and orbiter from booster and to provide NPSH for orbiter main engines. Ignite orbiter engines, burn to acceptable state vector, separate from tank, and glide to landing area.

3S = shut down booster engines, separate orbiter from drop tank, glide to water.

During the early portion of ascent flight, the orbiter has insufficient energy for aerodynamic glide. This abort gap could be eliminated by installing solid motors on the drop tank. These motors would be sized to burn a minimum of 4.5 seconds to allow the orbiter main engines to come up to full thrust, and to impart sufficient energy that the orbiter can achieve the altitude where the thrust/weight is greater than 1.

During the late portion of flight, the vehicle momentum directed downrange is too great to permit a glide return to the landing strip. This gap is also eliminated if solid motors are installed on the drop tank to provide the necessary ullage for main engine start.

An alternative scheme for eliminating the intact abort gaps is to install solid motors on the orbiter. This scheme permits a more complete abort capability since it permits escape from failures of the drop tank.

The hazardous conditions identified that require emergency action are given in Table 3.8-1. The main propulsion system is considered to be the *major* source of abort-forcing failure. The total probability of failure of this system is estimated to be 5438 ppm.

3.9 Manufacturing

Various design concepts were evaluated and preliminary tooling and fabrication plans were developed for major hardware elements. Plate size and weld criteria were established for 2219-T87 aluminum, HY 140 steel, maraging 200 steel, and Inconel 718 steel; weld land thicknesses and weld pass requirements were established.

Forming and handling problems are extreme due to the thicknesses and weights of the aluminum or steel, depending on which material is selected. Much of the detail fabrication and subassembly would require subcontracting to companies that have the necessary facilities and equipment to handle and form heavy materials.

3.10 Operations and Test Considerations

The water recovery and reusability feature of the pressure-fed booster requires operations and test requirements that are unique to this configuration.

3.10.1 Recovery Operation

The booster will impact the water approximately 166 nautical miles from the launch complex. The vehicle could then be towed back to a port or picked up and carried by a ship. Each of these techniques has advantages and disadvantages that must be carefully weighed before a final selection is made.

The towing operation is simple in theory but the large physical size and weight of the booster causes several problems leading to high-risk areas. The tow cable hookup, flotation system, sea state status, and time duration that the vehicle is subjected to salt water are typical examples.

Although pickup and retrieval by ship is feasible for configurations of this size, a new ship or vessel would be required. In addition to the cost and development problems of designing a larger vessel, the pickup technique and vehicle safing are high-risk items when they are combined with high sea states.

Table 3.8-1 Failure-Induced Hazards, Pressure-Fed Booster

Primary Failure Mode	Primary Effect	Action Required	Possible Secondary Effect	Consequence/ Possible Solutions
1. No Start	Low Vehicle Thrust	Launch Hold. Shutdown Remaining Engines	None	Use Majority Vote TCPSs For Failure Detection
2. Catastrophic Leakage (O_2, RP-1 Hydrazine, Hydraulic Fluid, Chamber Exhaust Products)	Fire: Possible Explosion	Isolate RP-1 Source. Shut Down Engines and Separate	1. Burn Through Wire Bundles: MDS, TVC, Control Valves 2. Burn Through Hydrazine Tanks, Heat Exchanger, Fluid Lines, Decomposition Chamber	1. Loss of MDS, TVC, or Control Valve Commands. Insulation Wrap will Protect Wire Bundles. 2. Can be Catastrophic, Leak or Fire Detection System Required. Thermal Protection of Components Probably Required.
3. Heat Exchanger Burnthrough	Fire: Possible Explosion	Shut Down Engines and Separate		Can be Catastrophic. Large Quantities of N_2H_4/LOX Available. Require Explosionproof Shroud Around Heat Exchanger
4. Tank Underpressure	Thrust Degradation and Control Degradation	1. If Sufficient T/W and Control, Continue Flight 2. Insufficient T/W of Control, Shut Down Engines and Separate	Combustion Stability Limits May be Violated	Thrust Chamber Rupture. Majority Vote Detection System
5. Tank Overpressure	Structural Failure	Shut Down Engines and Separate Prior to Violating Structural Limits	Fire or Explosion	Redundant Relief Valves, Redundant MDS Sensors
6. TVC Failure	Degradation of Control	1. If Sufficient Control, Continue Flight 2. If Insufficient Control, Shut Down Engines and Separate		Fail Operational TVC
7. Inadvertent Destruct	Fire/Explosion	Engine Shutdown and Separate		Probably Catastrophic. Redundancy Protection in Range Safety System
8. Loss of Nozzle	Thrust Degradation and Control Degradation	1. If Sufficient T/W and Control, Continue Flight 2. Insufficient T/W or Control, Shut Down Engines and Separate		Majority Vote Detection System
9. Engine Out	Thrust Degradation and Control Degradation	1. If Sufficient T/W and Control, Continue Flight 2. Insufficient T/W or Control, Shut Down Engines and Separate		Majority Vote Detection System

3.10.2 Refurbish/Retest Operation

The most significant item identified is recertification of the airframe for subsequent flight. The technique of accomplishing recertification is to install the vehicle vertically in a new cleaning and pressure test facility. The booster would be leak-checked at flight pressures. Although X-ray, ultrasonic, and other NDT techniques were investigated, no clear approach emerged. It is probable that final techniques will be established on early development vehicles.

3.11 Lifting vs Ballistic Entry

Lifting (high α) versus ballistic (low α) entry concepts considered are depicted in Figure 3.11-1. The lifting entry configuration obtains velocity attenuation during entry by flying at high angle of attack, using basic body drag. The ballistic entry configuration enters at near-zero angle of attack, using large flaps for the required drag braking.

The advantage of lifting entry is that the body fins required to obtain high angles of attack are also helpful in obtaining a near-neutral stability configuration for the ascent phase. The disadvantages relate to the need of an ACPS for capture, and active fin control systems for entry. The fins are also vulnerable to damage at water impact.

The advantage of the ballistic entry system lies with its ability to capture and enter passively, requiring no control system. Retraction of the flaps just before water impact minimizes the concern for flap damage.

Table 3.11-1 summarizes the findings of studies comparing the two systems. The ballistic configuration is recommended on the basis of least complexity, less technical risk and uncertainty, and less of refurbishment requirements for the operational program.

3.12 Conclusions

The conclusions reached relative to design approach and major concerns are discussed in this subsection.

3.12.1 Entry Mode

The ballistic (low α) configuration is significantly simpler and offers less technical risk than the lifting entry (high α) design. Booster entry weight (for a booster sized for 4 M lb of LO_2/RP-1) can grow to 1 million pounds and still be viable for ballistic entry using the basic point design approach shown.

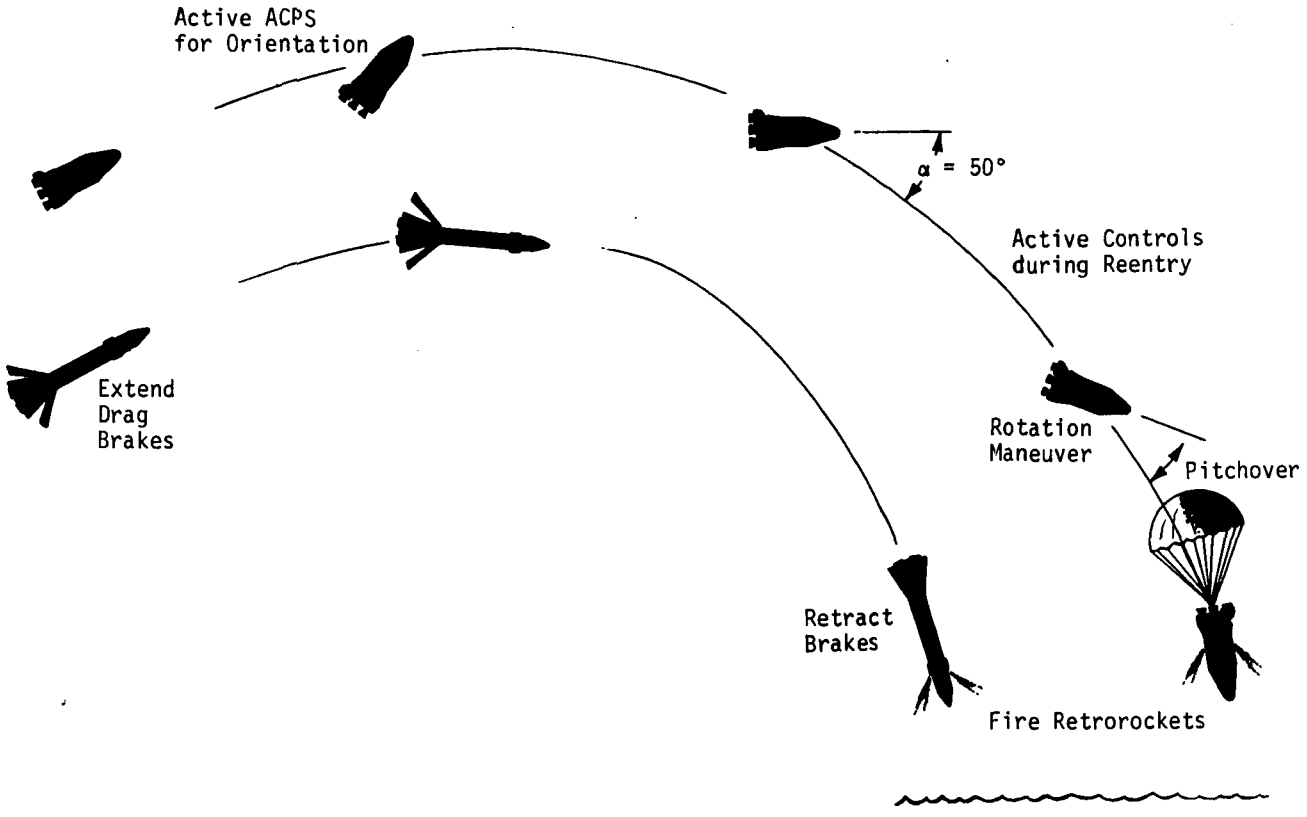


Figure 3.11-1 High α versus Low α Entry

Table 3.11-1 Lifting vs Ballistic Entry, Summary

	LIFTING	+	BALLISTIC	+
Decelerator Technique	High α (Asymmetric Fins)	-	Low α (Deployable Panels)	-
Ascent Control Requirements TVC	Stable/Fins (Minimize δ_e) GI TVC, Hinge, or Gimbal	✓	Unstable (Larger δ_e) Hinge or Gimbal Engines	
Reentry Capture Requirements	ACPS, 3-Axis Reference System		Deployable Panels	✓
Reentry Attitude Control	Movable Fins, Control System, Sensors, Actuators and APU		Passive, Deployable Panels	✓
Preimpact Requirements	Rotation Maneuver (60-ft Chute)		Terminal/Ballistic	✓
Terminal Decelerators	Retrorockets	-	Retrorockets	-
Expendable Elements	Fins, Retrorockets		Retrorockets	✓
Recommended Configuration: Least Complexity, Risk and Refurbishment			✓	

3.12.2 Thrust Vector Control

The hinged engine approach is viable and preferred for a five-engine configuration with the center engine fixed. The preference is based on (1) plumbing simplicity, (2) lateral impact load resistance for a hinge located at the engine cg, and (3) little or no throttling required for a five-engine configuration.

While GI TVC is adequate when orbiter surfaces are used to minimize δ_e requirements, the I_{sp} degradation results in unnecessarily large penalties and development risk and complexity are greater. Therefore GI TVC is not recommended.

Center-of-gravity or head-end engine gimbaling are viable options and should be used when six or seven engine clusters are proposed.

3.12.3 Concerns Remaining

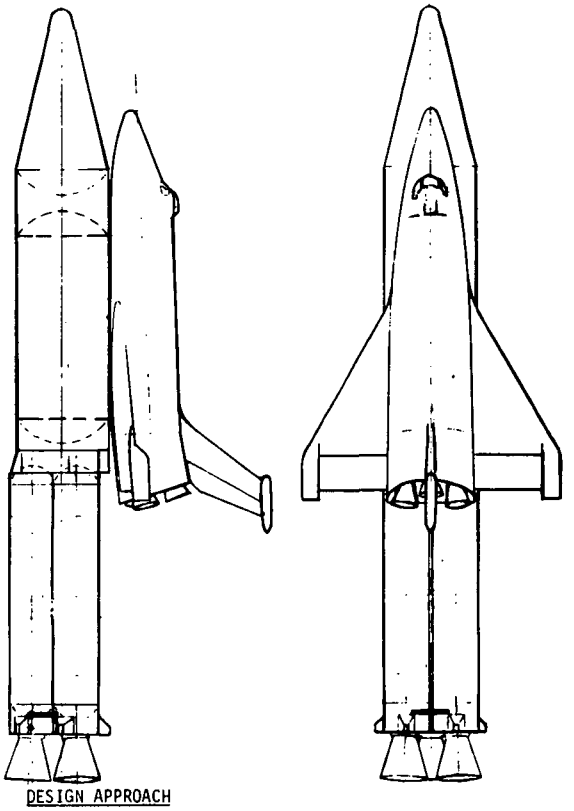
Major concerns that offer substantial risk to the proposed development phase design and/or costs include:

1) Materials selection and joining (aluminum with 4-in.-thick welds or mechanical joints, maraging 200, and Inconel are the only known candidate materials/joining approaches for the LO_2 tank);
2) Engine design/combustion stability;
3) Water impact/loads uncertainty.

4.0 SRAO (SRM) Booster

4.1 Configuration Arrangement and Rationale

This configuration consists of a cluster of three 156-inch SRMs to make up a single stage booster that burns in series with the orbiter.* The forward SRM adapter fixes the SRMs and distributes loads to the aft end of the orbiter tank. Tie links at the aft end of the SRMs permit independent motor growth.



DESIGN APPROACH

- Aft Attachment, Tangential Links, & Shear Fitting
- Forward Adapter Fixes SRMs & Distributes Loads
- SRM Cluster on Tank Centerline
- Launch Support on Pad through SRMs
- Nozzle ε = 10, Clears Gimbal of ±8°, With Fail Null, One Nozzle

Figure 4-1 Configuration Definition, (SRM) 156-inch Three-SRM SRAO

The configuration arrangement is shown in Figure 4-1. The centerline of the SRM cluster is on the orbiter tank centerline, providing an even thrust load distribution to the orbiter tank through a rigid adapter to maintain an adequately high first bending mode frequency for satisfactory control gains. Motors are mounted closely together to minimize attachment structure weight, necessitating a 3-degree precant of SRM (ε = 10) nozzles to clear a ±8-degree gimbal pattern with one nozzle

*More detailed data regarding this configuration may be found in the following briefing booklets: (1) *MSFC Systems Analysis/Flight Mechanics/Aero Working Group Meeting, SRAO (PFB) and (SRM)*, February 1, 1972; (2) *MSFC Configuration and Structures Working Group Meeting; SRAO (PFB) and (SRM)*, February 1, 1972; (3) *MSFC Propulsion Working Group Meeting, SRAO (PFB) and (SRM)*, February 1, 1972.

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failed null. Booster subsystem requirements include (1) APU-driven actuators for the gimbaled nozzles, (2) APU-mounted generators, (3) the APU system, (4) provisions for separation and inadvertent separation destruct, and (5) abort/thrust termination (TT) systems.

The configuration performance data are based on the MDAC 15 December 1971 baseline orbiter, a drop tank with W_p = 958,500 pounds, and a mass fraction (λ) of 0.9474. The drop tank diameter is 26.4 feet and the length is 126 feet.

4.2 System Performance and Mission Profile

The mission profile is summarized in Figure 4.2-1. After liftoff at a T/W = 1.39, the vehicle reaches a max q of 628 psf at a 41,400-foot altitude and mach 1.51. Staging occurs at 135 seconds at an altitude of 124,400 feet, a relative velocity of 4338 fps, and a flightpath angle of 19 degrees. The orbiter is then ignited and is injected into a 50x100-nautical mile orbit 4565 seconds after liftoff. The booster impacts approximately 147 nautical miles from the launch point.

4.2.1 SRM Thrust Shape

The thrust profile is shaped to provide the mission capability within the q and acceleration constraints. Curve A in Figure 4.2-2 was chosen as the baseline and is selected to provide a liftoff acceleration of approximately 1.4. The regression out to 70 seconds is selected to limit q to 650 psf and the thrust level at the start of tailoff produces a 3-g acceleration. Performance can be improved for the same SRM impulse by reducing the SRM burn time by increasing the thrust after q_{max} is passed. In the case of Curve B, the thrust was increased until the acceleration limit was reached. The thrust was then reduced to maintain a nearly constant 3-g limit.

The increased orbiter burnout weight obtained by using this thrust modification is shown as a function of max q in Figure 4.2-3. These data show that the B shape thrust curve increases the payload by about 4000 pounds.

4.3 Aerodynamics

The ascent aerodynamic characteristics of the 156-inch SRM SRAO configuration are given in Table 4.3-1. The cg of this ascent configuration is considerably aft. This configuration also tends to require more gimbal deflection capability than nearly neutral stable configurations. The greatest demand for gimbal angle, however, comes from the roll due to the sideslip associated with side or quartering tailwinds. The aileron controls of the orbiter could be used, however, to reduce the control requirement.

Table 4.3-1 Ascent Characteristics

M	0.5	1.1	1.5	3.0
C_{N_α}	0.20	0.27	0.20	0.12
$C_{N_{\alpha=0}}$	-0.58	-1.00	-0.56	-0.25
C_{m_α}	-0.77	-1.10	-0.78	-0.45
$C_{m_{\alpha=0}}$	1.0	2.6	1.2	0.8
$C_{M_{C_{N=0}}}$	-2.5	-3.2	-2.2	-0.90
$C_{A_{\alpha=0}}$	0.71	1.60	1.32	0.91
C_{Y_β}	-0.19	-0.20	-0.18	-0.13
C_{n_β}	0.60	0.70	0.63	0.47
C_{ℓ_β}	-0.17	-0.22	-0.19	-0.12
$C_{\ell_{\delta_a}}$				

Note: 1. Derivatives,
S_{Ref} = 547 ft^2,
L_{Ref} = 26.4 ft,
MRP at HO tank nose on ℄,
Orbiter incidence = -1.5°,
S_{Fin} = 485 ft^2.
2. All derivatives are per degree.

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4.4 Flight Control

The baseline SRAO configuration uses gimbaled SRM nozzles and orbiter aerodynamic surfaces for ascent control. The nozzle gimbal capability is ±8 degrees. The required deflections are ±6.9 degrees for the gimbaled nozzles and ±25.6 degrees for the orbiter ailerons.

Thrust deflection requirements were determined by 6-D simulations using a control system design based on rigid-body analysis. The deflection requirements are tabulated.

Wind Direction		Peak Autopilot Commands, deg			
		Engines			Ailerons
		Pitch	Yaw	Roll	Roll
No Ailerons	Head	+3	--	--	--
	Tail	+2.5	--	--	--
	Quartering Tail	+1.8	-1.0	-9.7	
	Cross	+1.2	-1.7	-13.3	
With Ailerons	Quartering Tail	+1.8	-1.0	-2.5	-12.7
	Cross	+1.2	-1.7	-3.3	-16.3

Past analysis has shown that the deflection requirements from the simulation must be multiplied by a factor of 1.6 to account for static-elastic effects and off-nominal conditions.

The tabulation shows that pitch and yaw deflection requirements are fairly small but that the roll deflection requirements are large in the "no aileron" case. Ailerons (with a 5:1 aileron/engine roll mix) were therefore used as the baseline for this configuration.

4.5 Structural Design

The SRAO booster structural configuration consists of a cluster of three 156-inch SRMs, with interconnecting structure at forward and aft ends. The forward end of the booster interfaces with the HO tank aft skirt, forming the main structural element of the boost configuration. This orbiter/booster interface structure must carry overall body bending moments, shears, axial loads, and torsions. The booster forward structure (adapter) provides for the transition of these loads from the single body HO tank to the three SRM cases. To provide for differential strains between SRM cases without inducing large stresses, as well as to allow for tolerances and alignment, the SRM cluster aft

structural tie consists of only lateral ties and linkages. These include three sets of shear tie fittings located at each of the inner points where the SRMs approach mutual contact, and three linkage bars connecting the SRMs near the points of outer tangency between adjacent cases.

Load studies performed for this configuration are summarized in Figure 4.5-1 for both prelaunch and flight conditions. The loads on the SRM case are not critical for either the unfired (no internal pressure) condition at prelaunch or the flight condition. The HO tank loads have been compared to the allowable strength of an H_2 tank if it were designed as a monocoque tank with an internal pressure of 36 psia. Flight loads far exceed this capability, especially at the aft end. It is concluded that this tank should be integrally stiffened or pressure stabilized to withstand flight loads.

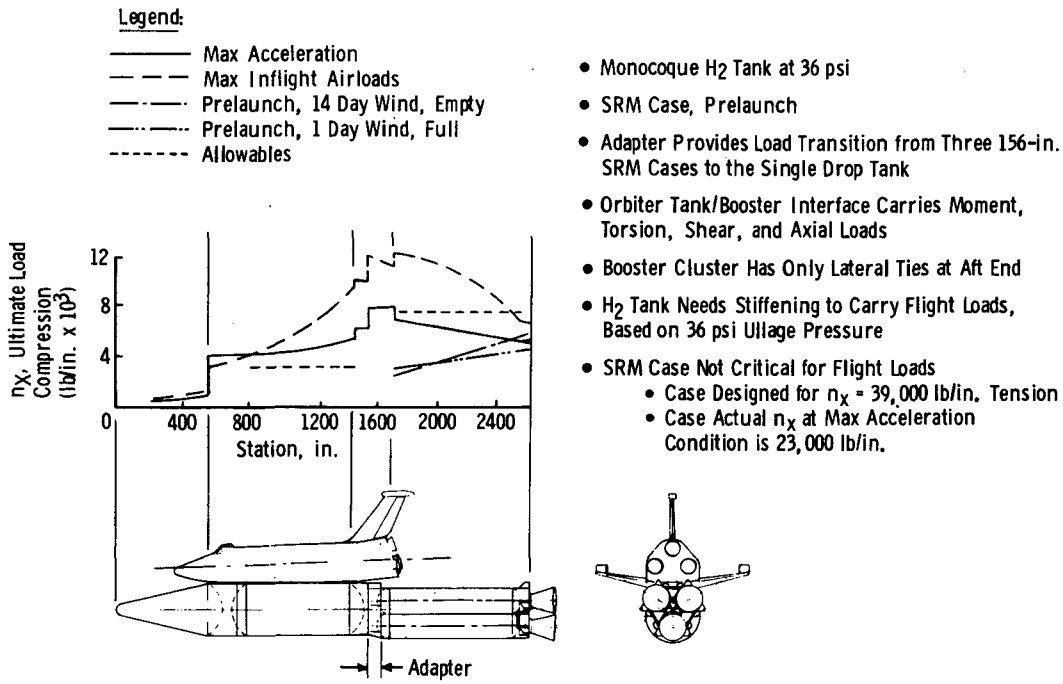


Figure 4.5-1 SRAO Preliminary Design Loads

There are no structural configuration issues that bear on basic configuration feasibility. The design of the adapter to minimize load peaking in local areas of the SRM cases as well as the modeling of the clustered body redundant load paths to permit realistic load predictions are critical design issues.

4.6 Propulsion Features and Tradeoff Summary

The SRAO configuration uses three expendable 156-inch-diameter solid rocket motors (SRMs) of the configuration shown in Figure 4.6-1. The forward adapter fixes the SRMs and distributes the thrust and bending loads. Each SRM has thrust termination, destruct, malfunction detection, and thrust vector control capabilities.

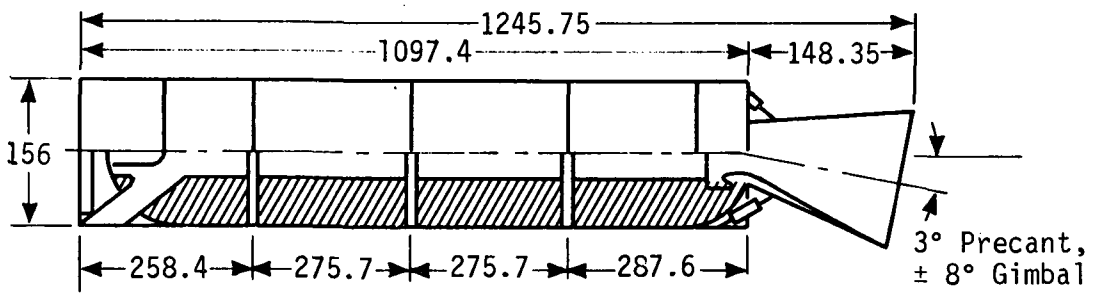


Figure 4.6-1 SRAO SRM Configuration

The weight for the selected SRM configuration is summarized in Table 4.6-1.

Table 4.6-1 SRM Weight Summary

Component	Weights
Forward Case Segment	18,816
Center Case Segments (2)	39,000
Aft Case Segment	23,100
Nozzle	11,900
Insulation & Liner	6,600
Igniter	360
Thrust Termination	710
TVC	750
Destruct	40
Total/Inerts	101,276
Propellant Weight	992,650
Total Weight	1,093,926
Motor Mass Fraction	0.907

The nominal vacuum performance characteristics for this motor are presented in Table 4.6-2.

Table 4.6-2 Nominal SRM Performance at 70°F

Delivered Effective I_{sp} (Vacuum) Minimum	264 lb_f-s/lb_m
Delivered Vacuum Total/Impulse	262,059,600 lb_f-s
Burn Time	125 s
Average Thrust (Vacuum)	2,096,477 lb_f
Average Chamber Pressure	745 psia
Average Nozzle Throat Diameter	46.17 in.
Average Nozzle Expansion Ratio	10
Nozzle Exit Diameter	146.0 in.

4.7 Mass Properties

Table 4.7-1 reflects the final configuration used in the flight performance data and includes three SRM motors with their associated structural skirts, linkage, and subsystems.

4.8 Reliability, Safety, and Abort

The orbiter intact abort mode alternatives have been considered.

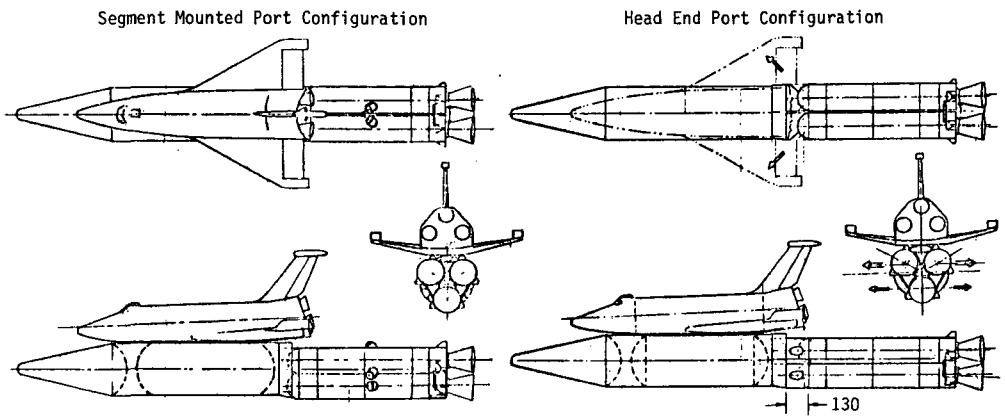
During the first 30 seconds, the orbiter has insufficient velocity to sustain an aerodynamic glide to the landing strip. After 100 seconds, the downrange momentum is too large to permit a glide return. These abort gaps can be eliminated by incorporating two abort staging rockets on the drop tank. Thrust neutralization (thrust termination) is achieved by cutting additional vent areas in the SRM chamber. This action lowers the SRM chamber pressure and provides components of force to oppose the thrust force through the nozzle. The requirements for safe implementation of the thrust neutralization system are (1) avoid collision of the port cover and orbiter vehicle, (2) avoid plume impingement on the orbiter or drop tank, and (3) reduce the net forward thrust to an acceptably low level.

Table 4.7-1 SRAO Characteristics

Mass Properties							
Conditions	Weight	X	Y	Z	I_X	I_Y	I_Z
BBOW	425,290	1814	0	48	1.4	12.0	12.0
Expendables	16,000						
Ascent Propulsion	2,961,300						
BLOW	3,402,590	1741	0	48	9.7	50.0	50.3
OLOW	1,250,000						
GLOW	4,652,590	1364	0	48	14.0	497.5	495.3

Weight Summary (lb)	
Body	
Forward Skirt	5,700
Aft Skirt and Attachment Structure	46,640
Motor Case	
Forward Shell	26,820
Segments	200,640
Aft Shell	22,020
Nozzle Assembly	51,000
Insulation	17,000
Igniter	2,100
Separation	10,000
APU/Hydraulics	4,800
Avionics	3,300
Booster/Drop Tank Adapter	11,400
Growth	23,870
BBOW	425,290
Expendables	2,977,300
BLOW	3,402,590

The candidate thrust neutralization schemes are depicted in Figure 4.8-1.



FEATURES
- No Port Cover or Plume Impingement
- Unorthodox Port Design
- SRM Weight Increase
  - Port Openings Cause Nonuniform Load Paths
  - Large Bending Moments in TT Cutout Region
  - May Require Separate Segment

Features
- No Port Cover or Plume Impingement
- Conventional Port Cover Design
- Identical Motors for PFRT
  - Requires Left-Hand, Right-Hand, Lower
  - Motor Assembly Buildup
  - May Require Two TT Static Firings
- Adapter Weight Increase
  - 88 in. Additional Length
  - Port Openings Cause Nonuniform Load Paths
  - Adapter Must React TT Forces of Upper Motors

Figure 4.8-1 SRAO TT Implementation Options

The hazardous conditions identified that required emergency action are given in Table 4.8-1. The probability of mission-ending failures induced by the solid motors is estimated at 1734 ppm. This estimate was derived from UTC failure rates on the existing 120-inch five-segment motor used in the Titan program.

4.9 Manufacturing

Various adapter hardware configurations for interfacing the solid rocket motors to the HO drop tank were investigated. Tooling and fabrication concepts were developed and tooling and assembly sketches were prepared for tradeoff evaluations of bulkhead, frame, skin panel, and main fixture assemblies. The tooling for handling and transportation and the master tools for the forward adapter skirt and the aft thrust structure were established and listed. Weld machining and heat-treat requirements for the shear tie and stabilizing linkage were investigated. No significant manufacturing difficulties were discovered.

Table 4.8-1 Failure-Induced Hazards, SRM Stage

Primary Failure Mode	Primary Effect	Action Required	Possible Secondary Effect	Consequence/ Possible Solutions
1. No Start	Tipover Moment	1. Hold Down Vehicle 2. TT SRM and Separate Orbiter Core 3. TT SRM and Separate Orbiter and Cryo Tank	1. High Thermal Loads on Pad 2. Cryo Tank Rupture 3. SRM Disintegration	1. Water Deluge System 2. Extensive Pad Damage 3. Minor Pad Damage
2. Case Burnthrough or Segment O-Ring Leak	Flame Penetrates SRM Case. Burn-through Hole Causes Chamber Pressure Decay and Moment About CG	TT SRMs to Relieve Chamber Pressure and Neutralize Thrust. Separate Orbiter	1. Burn Through Wire Bundles: MDS, Thrust Termination, TVC	1. Loss of MDS, TT Command Capability, TVC Commands. Insulation Wrap Will Protect Wire Bundles
			2. Ignite TT System Ordnance, Initiate Destruct Action	2. Could be Catastrophic. Detected by SRM Pressure Decay
			3. Impingement on Flammable Fluid Lines	3. Small Fire; Limited Quantity of Flammable Fluids. Can Insulate Lines if Required
			4. Overheat Attach Structure. Ignite Separation Ordnance	4. Inadvertent SRM Separation Local Thermal Protection if Required
			5. Impingement on Cryo Tank (TRAO Only)	5. Probably Catastrophic
3. Loss of Nozzle	Thrust Degradation and Control Degradation	Continue Flight	Loss of TVC for One Motor	TT and Separate
4. TVC Failure	Loss of Control	TT and Separate		Redundant TVC System per Motor
5. Flammable Fluid Leak	Loss of Control	TT and Separate	Small Fire	Probably Not Catastrophic Due to Limited Quantity of Flammable Fluids. TT and Separate
6. Inadvertent Destruct	Fire/Explosion	TT and Separate		Probably Catastrophic. Redundancy Protection in Range Safety System

4.10 Test Considerations

The test program considered for the 156-inch, three-SRM SRAO booster configuration consists of five test phases--design/development, qualification, manufacturing, acceptance, and launch operations.

Development testing of the SRM propellants, case, liner, TVC system, and thrust termination will require full-scale static firings. Thrust versus time histories and dispersion data will also be obtained from these firings. Data concerning the resistance of SRM propellant to cracking and liner separation will be required during the development phase. Vibration and aging tests will aid in determining these data. Providing a low cost and effective SRM development program requires careful analysis of total data requirements so each test will satisfy the maximum number of parameters.

Qualification testing (including life cycle, vibration, and margin tests) demonstrates that a production flight article is functionally reliable in the predicted flight environment. Therefore, qualification test articles must be essentially the same as production items. To the maximum extent possible, development test results should be used to satisfy qualification test requirements.

The manufacturing tests verify that there are no built-in manufacturing defects. Included are X-ray, wiring harness continuity, and megger tests, and fit checks.

Acceptance tests verify that the vehicle meets all specifications. These tests are functional subsystem and combined system tests to verify that the vehicle systems operate within the specified parameters.

Launch operations testing is performed to reverify systems that may have been disturbed by packing and shipping operations and to verify that the booster vehicle is ready to be mated with the orbiter vehicle. Successful completion of a complete booster systems test, with the orbiter vehicle electrically simulated, is a prerequisite to mating with the orbiter.

With the orbiter electrically and mechanically mated to the booster, interface verifications culminating in an integrated systems test in the VAB is a prerequisite to actual mating with the orbiter and transporting the assembled vehicle to the pad.

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4.11 Conclusions

The conclusions and/or technical concerns reached for the SRAO 156-inch configuration are discussed in this subsection.

4.11.1 Stack Arrangement

The three SRMs are required to provide a stable prelaunch stack configuration and to avoid case-buckling-critical conditions during a preignition ground wind condition. While the prelaunch condition for the three-SRM arrangement does not exceed the unpressurized case allowables, the margin is not large and this condition should continue to be monitored as design progresses. Longer SRMs or a larger orbiter planform area could create a critical design condition.

4.11.2 Control

The ±8-degree control gimbal provision is adequate when orbiter ailerons are also used to help with roll control.

4.11.3 Separation

Conventional separation techniques are satisfactory for normal flight separation. However, large separation/abort rockets are required on either the drop tank or orbiter to provide a significant abort-and-safe-return capability.

4.11.4 Abort

Abort and safe return requires thrust termination and large abort/separation rockets on either the orbiter or orbiter drop tanks. Provisions for thrust termination requires either (1) a significant weight penalty for the booster adapter to keep TT ports in the head end of the SRMs and some compromise to SRM head port production commonality, or (2) unconventional TT ports in the barrel section of the SRM located 50% or more of the SRM length aft of the forward closure.

4.11.5 Concerns

The major concerns center around the cost competitiveness of the SRAO versus the twin rocket assist orbiter (TRAO), even with TRAO TVC. The heavy adapter of the SRAO plus three SRMs cannot be competitive with the TRAO approach on a recurring cost basis. The potential difficulty with SRAO thrust termination is also of concern, posing a difficult structural design/development problem for either the adapter or the SRM case.

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