

DRA



MDC E0558

Phase B System Study Extension

FINAL REPORT

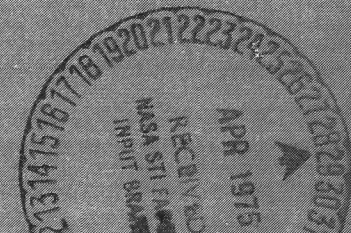
Part II - Volume I

TECHNICAL REPORT
SYSTEM AND ORBITER

HIGH VALUE

SPACE
SHUTTLE

LOW COST



N75-76032

Unclas

00/98 17875

(NASA-CR-143906) SPACE SHUTTLE PHASE B
SYSTEM STUDY EXTENSION. PART 2, VOLUME 1:
TECHNICAL REPORT, SYSTEM AND ORBITER Final
Report, 1 Nov. 1971 - 15 Mar. 1972
(McDonnell-Douglas Corp.) 244 p

MCDONNELL DOUGLAS

TRW



CORPORATION

COPY NO. 225

SPACE SHUTTLE PHASE B SYSTEM STUDY EXTENSION FINAL REPORT

15 MARCH 1972

MDC E0558

Part II - Volume I

TECHNICAL REPORT SYSTEM AND ORBITER

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

MCDONNELL DOUGLAS

TRW



CORPORATION

15 MARCH 1972

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 I 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

Table of Contents

<u>Section</u>	<u>Title</u>	<u>Page</u>
A	(Shuttle System)	
1.0	Summary	A.1-1
2.0	General System Requirements	A.2-1
3.0	Shuttle Configuration Evolution	A.3-1
4.0	Key Issues and Trade Studies	A.4-1
4.1	Advantage of the Mark I → Mark II Approach	A.4-1
4.2	Parallel vs Series Burn and Booster Selection	A.4-3
4.3	Mating Arrangement	A.4-3
4.4	Orbiter Engine Selection	A.4-7
4.5	Payload Bay Size	A.4-9
4.6	Flight Performance Reserve Summary	A.4-16
4.7	Engine Mixture Ratio Range	A.4-17
5.0	System Sizing and Performance	A.5-1
5.1	Analysis Technique	A.5-1
5.2	Effect of Staging Velocity	A.5-1
5.3	Growth Accommodation	A.5-3
5.4	Cost Summary	A.5-3
6.0	Space Shuttle System Description	A.6-1
6.1	Configuration Characteristics	A.6-1
6.2	Ascent Trajectories	A.6-5
6.3	Ascent Control	A.6-12
6.4	Separation	A.6-26
7.0	Abort Analysis	A.7-1
7.1	Abort Modes	A.7-1
7.2	Abort Separation	A.7-8
7.3	System Impact	A.7-10
8.0	Operation	A.8-1
8.1	Ground Operations	A.8-1
8.2	Flight Operations Support	A.8-3

Table of Contents - Continued

<u>Section</u>	<u>Title</u>	<u>Page</u>
A	(Shuttle System) - Continued	
9.0	Recommended System	A.9-1
9.1	Recommended System Description	A.9-1
9.2	Selection - Rationale	A.9-3
<u>Section</u>		
B	(Orbiter Vehicle)	
1.0	Summary	B.1-1
2.0	General Characteristics	B.2-1
2.1	Physical Characteristics	B.2-1
2.2	Orbiter, Weight Summary	B.2-4
2.3	Aerodynamics and Performance	B.2-6
2.4	Flight Control Characteristics	B.2-12
3.0	Subsystems	B.3-1
3.1	Airframe	B.3-1
3.2	Propulsion	B.3-43
3.3	Avionics	B.3-63
3.4	Crew Station	B.3-79
3.5	Power Supply	B.3-83
4.0	Orbiter HO Tank (Series Burn - Pressure Fed)	B.4-1
4.1	Phase B Extension Final Design	B.4-1
4.2	Analysis Baseline	B.4-2
5.0	Orbiter HO Tank Parallel Burn Two 156" SRM	B.5-1
5.1	Phase B Extension Final Design	B.5-1
5.2	Analysis Baseline	B.5-2
6.0	Test and Manufacturing	B.6-1
6.1	Development Testing	B.6-1
6.2	Manufacturing	B.6-3

LIST OF PAGES

Title Page	
ii	through v
A.1-1	
A.2-1	
A.3-1	through A.3-5
A.4-1	through A.4-18
A.5-1	through A.5-5
A.6-1	through A.6-29
A.7-1	through A.7-12
A.8-1	through A.8-8
A.9-1	through A.9-4
B.1-1	
B.2-1	through B.2-14
B.3-1	through B.3-89
B.4-1	through B.4-29
B.5-1	through B.5-18
B.6-1	through B.6-4

15 MARCH 1972

1.0 Summary

The most promising candidate Shuttle configurations to evolve from the Phase B extension study are:

- o Series Burn Pressure Fed Booster with a large Orbiter (15 ft dia by 60 ft payload bay)
- o Series Burn Pump Fed (F-1) Booster with a large Orbiter
- o Parallel Burn Twin SRM Booster with a large Orbiter
- c Parallel Burn Twin SRM Booster with a smaller Orbiter (14 ft dia by 45 ft payload bay)

A wide range of Shuttle systems was investigated to find a viable system that would result in minimum costs and acceptable peak annual funding. Both single and dual expendable external hydrogen and hydrogen/oxygen tanks were studied for the Orbiter to reduce weight and cost.

Expendable boosters were first studied as interim systems only. They were to be replaced eventually by fully reusable (flyback) boosters. However, the flyback Booster was eventually dropped to reduce RDT&E costs and a water recoverable (non-flyback) Booster substituted. The interim Boosters investigated were expendable solid rocket motors and variations of the Titan III launch vehicle. In these investigations, parallel burn ascent was found to be competitive with series burn systems, and was particularly attractive for solid rocket motor boosters.

Intact abort capability was defined for each Shuttle configuration and the cost of eliminating all gaps was determined. The cost of providing on-pad abort capability, the most critical condition, is approximately \$250 million.

A parallel burn twin 156" SRM Booster with a large cargo bay Orbiter is the recommended system. This configuration, with a gross lift off weight of 4.6 million lb, has been sized to deliver a 65,000 lb payload into an Easterly orbit and return with 40,000 lb. The initial thrust-to-weight ratio for the system is 1.4. Recovery of the solid rocket motors is economically attractive and hence recommended. The Orbiter provides thrust vector control as well as guidance and navigation for the mated configuration. The stages are separated at a relative velocity of 4000 fps. The Orbiter main propulsion is provided by three HiPc 470,000 lb thrust engines. An optional go-around and ferry capability is provided by removable airbreathing engines and tankage.

2.0 General System Requirements

- o The design missions shall be:

	PAYLOAD (K LB)	ORBIT		ON-ORBIT X-TRANS ΔV (FPS)	ABES
		ALT (N.MI.)	INCL (DEG.)		
EASTERLY	65	100	28.5	900	OUT
RESUPPLY	25	270	55	1500	IN
POLAR	40	100	90	650	OUT

- o The Orbiter shall have sufficient cross range capability to return to the launch site after one revolution.
- o The payload bay shall have a clear volume of 15 ft dia. x 60 ft (nominal requirement).
- o Turnaround time from landing at the launch site to launch readiness shall be less than 14 calendar days.
- o Rapid egress before lift-off shall be provided and intact abort after lift-off is a design goal.
- o Resultant load factors for shuttle launch and Orbiter entry shall not exceed 3 g's.
- o The Shuttle shall be capable of operating in all flight phases and throughout the payload range (zero to maximum) without ballast.
- o Mating interchangeability shall exist between Orbiters and Boosters.
- o The Shuttle system shall have the capability to be launched from a standby status within 3 hours and to stay in a standby status for 2 hours.
- o The Orbiter and Booster vehicles ascent performance shall be based on nominal engine performance. Minimum engine performance shall be used for sizing the propellant capacity of both vehicles.
- o Orbiter main propellant tankage shall be sized to include 1% flight performance reserve.
- o Uncertainty/growth weight allowance shall be as follows:
 - Orbiter: 10% of dry weight less ascent engines (and ABES, when installed)
 - HO Tank: 2% of dry weight

15 MARCH 1972

Booster

Pump Fed System: 10% of Dry Weight (Less Ascent Engines)
Pressure Fed System: 10% of Dry Weight
SRM System: 2% of motor case, nozzle and insulation
10% of Booster structure (skirts, etc.) and
subsystem dry weight

- o All flight vehicle subsystems, except primary structure and pressure vessels, shall be designed to fail safe as a minimum. For systems essential to crew/vehicle survival, the subsystems redundancy level shall be determined by analysis.
- o The Shuttle systems design shall retain the option for unmanned operations.
- o The mated ascent, guidance, and navigation function will be provided by the Orbiter Vehicle.

3.0 Shuttle Configuration Evolution

The two four-month Study Extensions were conducted to find ways of reducing the high RDT&E costs and peak annual funding requirements derived during the original Phase B study.

Booster concepts examined during this period varied from the Phase B, fully reusable, flyback canard using a hydrogen/oxygen HiPc engine to throwaway Solid Rocket Motors (SRM's). The payload weight and cargo bay size of the Orbiter were also investigated. The major variations in the system requirements are outlined in Figure 3-1. Only Orbiter crossrange capability and the airbreathing engine concept remained essentially unchanged throughout the study extensions.

During the first four-month study extension, the MDAC team investigated interim expendable Boosters which evolve to a fully reusable Booster system, and also Mark I (limited technology) Orbiters that could grow to Mark II (advanced technology) vehicles. Some of the primary concepts studied during this phase are illustrated in Figure 3-2. Orbiter variations included various delta and swept-wing designs, external H₂O and H tanks, payload bay sizes from 12 x 40 to 15 x 60 ft, various mating arrangements, and various main and auxiliary engine systems. Interim Booster variations included: TIIL, clustered solids, single 260" solid, and series and parallel burn ascent modes. Reusable Booster variations included: Flyback, Retro Glider, and Water recoverable designs; Saturn IC F-1, Pressure Fed, and HiPc engines; Delta and Canard wings; and parallel and series burn ascent modes. The

SHUTTLE REQUIREMENTS EVOLUTION

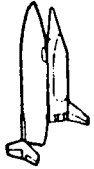
	JUNE	JULY	AUG	SEPT	OCT	NOV	DEC	JAN	FEB
STUDY	PHASE B ▽ 1ST 4 MONTH EXTENSION ▽ 2ND 4 MONTH EXTENSION ▽								
CROSSRANGE	1100 NM								
ORBITER ABES	IN/OUT GO-AROUND				NO GO-AROUND				
ABES FUEL	JP								
PAYLOAD - ASCENT (K LB)	65 EAST	INTERIM 45 EAST			MARK I - 25 POLAR			65 EAST	
- LANDED (K LB)	40 POLAR	FINAL 65 EAST-40 POLAR			MARK II - 65 EAST-40 POLAR			45 EAST	
	40							40 FOR 15x60 BAY 25 FOR 14x45 BAY	
BOOSTER - SERIES	H1Pc	H1Pc INTERM.		TIIIL SRM		H1Pc FI		PF	
- PARALLEL	NONE	INTERM. SRM				PF SRM			
BOOSTER RECOVERY - LIQUID - SRM	FLYBACK		FLYBACK/WATER RECOVERY				WATER RECOVERY		
			SRM'S NOT RECOVERED				RECOVERED		
PAYLOAD BAY SIZE (FT)	15x60	15x60 15x40	12x60 12x40	15x60			15x60 14x45		
ORBITER TANKAGE	INT/EXT H	INT/EXT H & HO			EXTERNAL HO				
REDUNDANCY	FO/FO/FS-AVION (WITH DEVIAT.)				CONTRACTOR RECOMMEND				
ABORT	TO ORBIT	MINIMIZE GAPS - ASSESS HIGH Q						ASSESS OFF-THE-PAD	
ORBITER MAIN ENGINE (K LB)	550 H1Pc	250-600 H1Pc			J2S, J2, 250-600 H1Pc			470 H1Pc	
REUSABLE BOOSTER ENGINES	SAME AS ORBITER				F-1, PF, H1Pc				

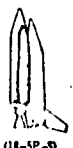
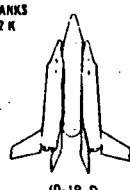
SRM Boosters included various segments of 156", 180", 260", and 120" solids; parallel and series burn; and various numbers of solid units per stage.

The matrix in Figure 3-3 indicates the breadth of system combinations investigated during the second four-month study extension. Principal system candidates are illustrated in Figure 3-4. During this study the Orbiter candidates were reduced to two: large bay with heavy payload and small bay with light payload. The Orbiter propulsion system was held at three engines of 470,000 lb thrust each. Booster candidates consisted of various solid rockets, the pressure fed Booster and two Boosters with pump fed engines.

The use of an interim Booster was found to reduce peak annual costs, but total program costs remained high. The switch to F-1 engines in the reusable Booster gave similar results. Changing the Booster recovery mode from flyback to a water recovery concept reduced the RDT&E and total program costs, maintained a low peak annual funding, and had only a slight penalty in cost per flight. The use of SRM Boosters allowed further reduction in peak annual funding at low total program costs although cost per flight was increased. This concept, using a parallel burn 156" SRM with a large cargo bay Orbiter, is the preferred system.

FIGURE 3-2
CANDIDATE CONFIGURATIONS STUDIED
DURING FIRST FOUR MONTH EXTENSION

SYSTEM	NO SERIES BURN					
FINAL SYSTEM	① 2-NO TANKS 12-3 415 K ε = 35-90 40 POLAR 3,835 B-15-OS 1,263 928 27 196 45 Y-15-OS 2,566 2,074 413 	40 POLAR 3,484 B-15-T 1,109 792 25 182 39 Y-15-T 2,375 1,884 405 	② 1-NO TANK 12-3 415 K ε = 35-90	③ 2-NO TANKS 12-3 415 K ε = 35-90 40 POLAR 3,589 (B-15-S) 1,163 840 25 185 41 (Y-15-S) 2,426 1,940 407 		
INTERIM SYSTEM	T-III L 1207-4 45 EAST 6,331 B-15-OS 1,253 928 0 196 45 (13-15-OS) 5,278 1,134 3,556 55 513 41 	T-III L 1207-4 45 EAST 5,868 B-15-T 1,089 792 0 182 39 (13-15-T) 3,999 1,213 2,371 55 742 71 	1-280" 45 EAST 6,232 B-15-T 1,089 792 0 182 39 (26-15-T) 4,843 6,343 (SRM) 	3-156" SRM 45 EAST 6,232 B-15-T 1,089 792 0 182 39 (13-15-T) 4,067 1,213 2,371 55 742 71 	T-III L 1207-4 45 EAST 6,210 (B-15-S) 1,163 840 0 185 41 (13-15-S) 4,995 1,213 2,371 55 742 71 	3-156" 45 EAST 6,338 (B-15-S) 1,163 840 0 185 41 (13-15-S) 4,995 1,213 2,371 55 742 71 

MINI NO SERIES BURN	NO PARALLEL BURN		H SERIES BURN
⑦ 1-NO TANK 18-2 415 K ε = 35-90 45 EAST 3,082 (C-75-T) 928 644 0 138 36 X-75-T 2,153 1,668 398 	⑤A 2-NO TANKS 1-180" 4/550 K ε = 90 (RATO) 40 POLAR 4,734 (E-5P-S) 2,558 2,131/29.7 228.92 (18-SP-S) 2,176 1,934 (SRM) 	⑥A 3-NO TANKS 5-5/572 K ε = 55 (IVC) 40 POLAR 4,508 (D-1P-S) 1,640 1,217 33 248.53 (D-1P-S) 2,868 2,434 264/125 	⑧ 2-NO TANKS 12-3 415 K ε = 35-45/150 40 POLAR 3,402 (A-45-S) 1,091 716 24 205 20 (2-45-S) 2,311 1,839 404 
T-III L 1207-4 45 EAST 6,328 5,067 (C-75-T) 928 644 0 138 36 (13-75-T) 3,999 1,213 2,371 55 742 71 	1-280" 45 EAST (E-5P-S) SAME VEHICLE 	1-280" 45 EAST (D-1P-S) SAME VEHICLE 	T-III L 1207-4 45 EAST 6,364 (A-45-S) 1,076 776 0 205 20 (15-45-S) 5,544 1,134 2,371 55 742 71 

CANDIDATE CONFIGURATIONS INVESTIGATED DURING SECOND FOUR-MONTH EXTENSION

STUDY EMPHASIS

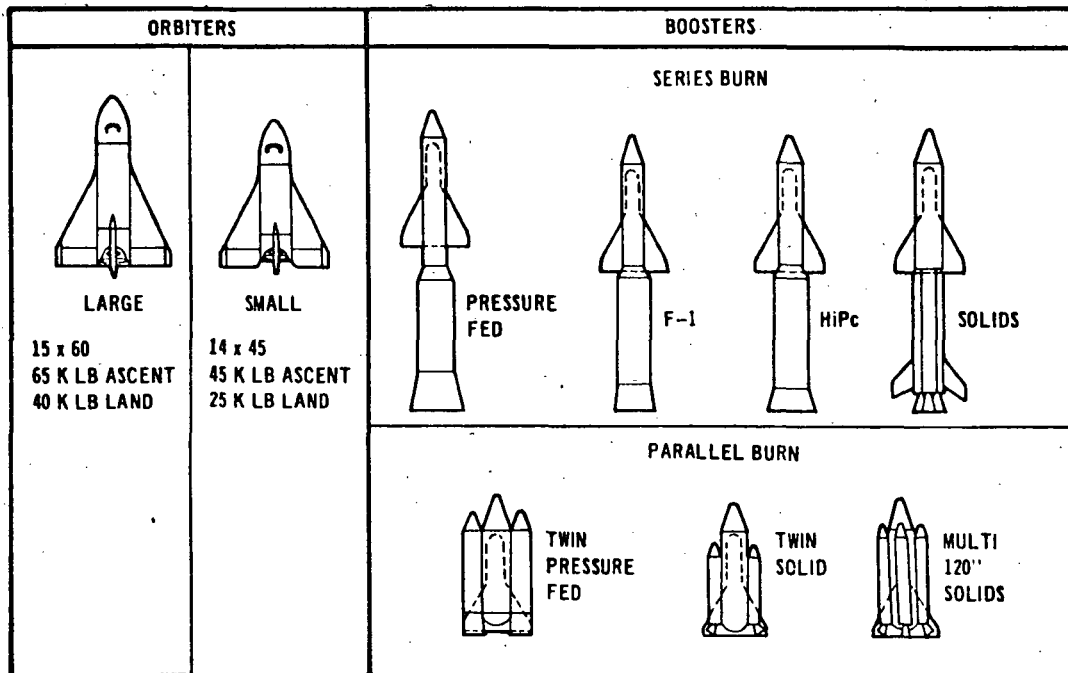
✓ POINT DESIGN WORK DONE
OTHERS PARAMETRICALLY DERIVED

SYSTEM		BURN MODE	TYPE	BOOSTER		HO TANKS	
REQUIREMENTS				ENGINES	MATING		
BASELINE ORBITER	BAY (FT) 15 x 60 PAYLOADS (K LB): EAST 65 RESUPPLY 25 + ABES + OMS POLAR 40 LAND 40 ✓	SERIES	SINGLE LIQUID	LOX/RP-1 → PR. FD ✓ 6 F-1 ✓ 4 HiPc ✓ 10	TANK END LOAD (TEL)	SINGLE BELLY MOUNTED (SBM)	
			SOLID CLUSTER 3 - 156"	3 SRM ✓			
		PARALLEL	TWIN LIQUID LOX/RP-1	PR. FD. 2 x 6 ✓ 1207 4 OR 5 SRM ✓ TWIN 156" 2 - SRM ✓ SINGLE 180" 1 SRM SINGLE 260"	TANK SIDE LOAD (TSL)		TBM
			SOLID →	TEL	SBM		
	ALTERNATE ORBITER	BAY (FT) 14 x 45 PAYLOADS (K LB) EAST 45 RESUPPLY 25 (NO ABES) POLAR - AS AVAILABLE LAND 25 ✓	SERIES	SINGLE LIQUID	LOX/RP-1 → PR. FD 6 F-1 4 HiPc 9	TANK END LOAD (TEL)	SINGLE BELLY MOUNTED (SBM)
				SOLID CLUSTER 3-156"	3 SRM		
ALTERNATE ORBITER		PARALLEL	TWIN LIQUID LOX/RP-1	PR. FD. 2x6 1207 3 OR 4 SRM TWIN 156" 2 SRM SINGLE 180" 1 SRM SINGLE 260"	TANK SIDE LOAD (TSL)	TBM	
			SOLID →	TEL	SBM		

* TBM = TWIN BELLY MOUNTED

FIGURE 3-4

PRIMARY CONFIGURATIONS STUDIED DURING SECOND FOUR-MONTH EXTENSION



15 MARCH 1972

Sizes and costs for the main contenders at the end of this study are summarized in Figure 3-5. The four candidates that are emphasized in this table are discussed in more detail later in this report.

FIGURE 3-5

SIZE AND COST SUMMARY OF PRIMARY SHUTTLE CANDIDATES

BOOSTER MATING ENGINE SYSTEM		SERIES BURN			PARALLEL BURN		
		LIQUID		SOLID CLUSTER	TWIN LIQUID	TWIN SOLID	
					TEL	TSL	
		6/PF	4/F-1	10/HiPc	3-156"	2x6. PF	2-156"
LARGE ORBITER 15 x 60	GLOW (K LB)	5,777	4,458	3,144	4,539	4,800	4,315
	RELATIVE STAGE V (FPS)	4,000	5,000	5,000	4,000	4,000	4,000
	COSTS (\$M)						
	TOTAL PROGRAM	10,798	10,002	10,599	11,335	9,980	10,207
	RDT&E	5,676	5,018	5,217	4,215	4,948	4,126
	PER FLIGHT	7.50	7.56	7.91	12.58	7.47	10.49
SMALL ORBITER 14 x 45				(9/HiPc)			
	GLOW (K LB)	5,000	3,485	2,760	3,886	3,917	3,617
	RELATIVE STAGE V (FPS)	4,000	5,000	5,000	4,000	4,000	4,000
	COSTS (\$M)						
	TOTAL PROGRAM	10,351	9,800	10,272	10,788	9,404	9,681
	RDT&E	5,432	4,904	5,066	4,081	4,676	4,006
PER FLIGHT	7.23	7.44	7.73	11.76	7.07	9.83	

TEL = HO TANK END LOADED; TSL = HO TANK SIDE LOADED BY BOOSTERS
ALL CONFIGURATIONS HAVE 3-470 K HiPc ORBITER ENGINES

4.0 Key Issues and Trade Studies

The key system issues studied during the Phase B extension study were:

- o Advantage of the Mark I → Mark II Approach
- o Parallel vs Series Burn and Booster selection
- o Mating Arrangement
- o Orbiter Engine Selection
- o Payload Bay Size
- o Flight Performance Reserve Analysis
- o Engine Mixture Ratio Range

The study approaches and results are discussed in the ensuing paragraphs. The primary evaluation criteria were cost and performance. A major consideration for payload bay size selection was also mission capture capability.

4.1 Advantage of the Mark I → Mark II Approach

The possibility of phasing system requirements was investigated as a means of reducing costs. This philosophy resulted in a Mark I → Mark II concept that utilized a system with a reduced performance capability for 125 flights of a 445 flight program. The payload requirement was lowered to 45,000 lb for the easterly mission and 10,000 to 25,000 lb for the south polar mission, in order to permit use of a modified existing Orbiter main engine (J2/J2-S). The Mark I system also utilized an ablative TPS, a simpler avionics system, and an expendable Booster. TPS replacement and additional ground control checkout requirements resulted in doubling the required turnaround time. However, it was planned that during the Mark I operational time span, development of reusable surface insulation TPS, HiPc engines, additional avionics sophistication and Booster recovery would be accomplished, to allow transition to full Mark II performance capability for the remaining 320 flights.

The baseline system used for comparing a Mark I → Mark II approach with an early Mark II transition concept was a Parallel Burn Twin Pressure Fed Shuttle System. Figures 4.1-1 and 4.1-2 compare the evolution time frame and risk of the two development approaches. It was found that although there is a PAF increase of \$28M in 1978, total program cost can be reduced almost \$1 Billion if the Mark I phase is eliminated. Costs are reduced because only one Orbiter engine development program is required, the operational costs associated with reusable TPS and additional onboard checkout capability are less, and retrofit of Mark I vehicles to Mark II is eliminated.

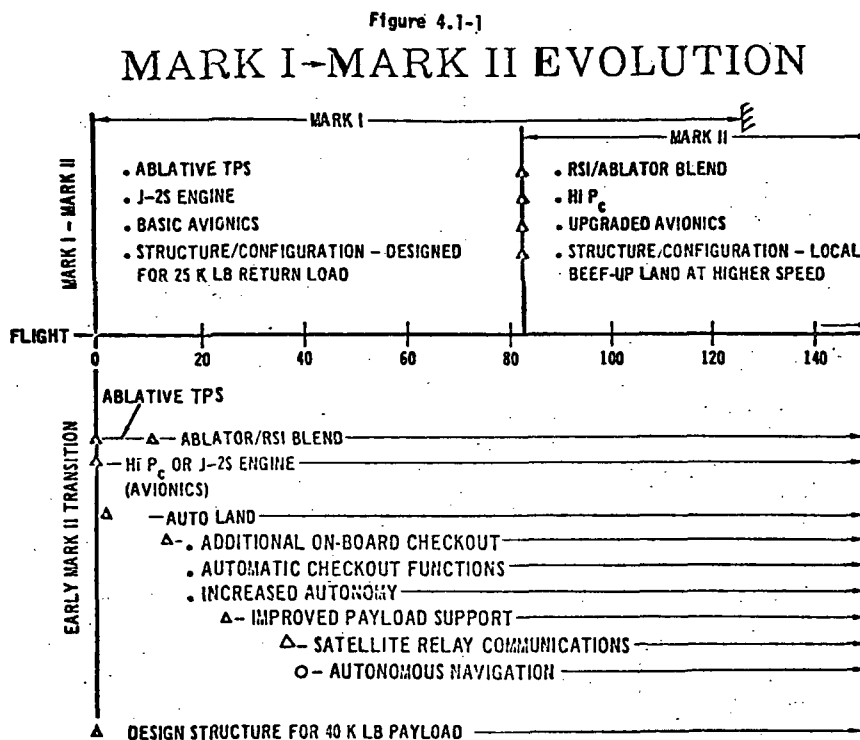


Figure 4.1-2

PROGRAM RISK COMPARISON

	MARK I — MARK II	EARLY MARK II TRANSITION
PAYLOAD REQUIREMENT (SOUTH POLAR)	DESIRED: 25 K LB — 40 K LB MINIMUM - 10 K LB • INABILITY TO CAPTURE TRAFFIC FOR (4) YRS	40 K LB
ORBITER PROPULSION	J-2S — Hi Pc • USE OF MODIFIED EXISTING ENGINE INITIALLY REDUCES RISK OF SCHEDULE DELAY	J-2S • LEAST RISK Hi Pc • SOME RISK IF DEVELOPMENT PROBLEMS OCCUR • GREATEST COST SAVINGS
TPS	ABLATOR — ABLATOR & RSI BLEND • EASES RSI DEVELOPMENT TIME TABLE • USE OF ABLATOR FOR 125 FLIGHTS INCREASES COSTS	ABLATOR — ABLATOR & RSI BLEND • FIRST FLIGHTS UTILIZE ALL - ABLATOR SYSTEM DESIGN • EARLY FLIGHTS WILL VERIFY ENVIRONMENT & FLIGHT TEST RSI (IN NON-FLIGHT SAFETY CRITICAL AREAS) • USE OF EXISTING ABLATOR SYSTEM FOR BACKUP ELIMINATES RISK • REDUCED COST
AVIONICS	MINIMUM SYSTEM — UPGRADED SYSTEM • MINIMAL RISK • LOWERS PEAK ANNUAL FUNDING INITIALLY • REDUCES ORBITER CAPABILITY	INTRODUCE MARK II REQUIREMENTS IN STEP SEQUENCE COMMENSURATE WITH PEAK FUNDING CONSTRAINTS • MINIMAL RISK • PROVIDES INCREASED CAPABILITY
STRUCTURE	DESIGN FOR 25 K LB — STRETCH DESIGN PHILOSOPHY • LIGHTER VEHICLE BUT SOME RISK • LIMITED GROWTH POTENTIAL	DESIGN FOR 40 K LB PAYLOAD • MINIMAL RISK • PROVIDES GROWTH POTENTIAL
CONFIGURATION	LAND AT ≈ 150 KT — LAND FASTER WITH 25 K LB PAYLOAD — LAND FASTER WITH 40 K LB PAYLOAD • MINIMUM WEIGHT • LIMITED GROWTH POTENTIAL	LAND FASTER WITH 40 K LB PAYLOAD AND MARK I WING • MINIMUM WEIGHT • LIMITED GROWTH POTENTIAL

15 MARCH 1972

4.2 Parallel vs Series Burn and Booster Selection

During the study, selection of the Booster and mated vehicle configuration was emphasized, with particular attention to costs. Figure 4.2-1 summarizes the weights and costs of various parallel-burn solid-rocket systems.

Use of the off-the-shelf seven-segment 120" SRM's is relatively expensive because of the number of SRM's per stage. Thus, even though the RDT&E cost of 120" SRM's is relatively low, the program cost becomes unacceptable. These and other considerations are summarized in Figure 4.2-2.

In Figure 4.2-3, the concepts for series and parallel burn are compared under those items which are considered to be the main discriminators. The parallel burn mode has several very important advantages. To cite two: verified ignition of all engines eliminates all concern for engine start failures, and RDT&E and program costs are significantly lower.

In Figure 4.2-4, the technical risks of three series-burn liquid Booster systems are compared with those of the twin 156" parallel-burn system. The results are as follows:

- o Least Technical Risk: Parallel Solids
- o Most Risk: Series HiPc
- o Intermediate and very nearly equal risk: Series pressure-fed and series F-1 pump-fed

This technical risk assessment, along with the cost comparisons summarized in Figure 4.2-5, has led us to select the twin-156" parallel-burn design as our prime choice and the pump-fed F-1 series-burn as the second choice.

4.3 Mating Arrangement

Studies performed during August 1971 concluded that: (a) external Orbiter H₂O tanks were preferred to external H tanks, (b) a tank-end-loaded (TEL) Booster-to-Orbiter mating configuration was preferred for the external H₂O tank concept, and (c) side-by-side (SS) mating was preferred for the external twin H tank concept. Design effort during September and October concentrated on external H₂O tank concepts, primarily for a configuration using a single H₂O tank, TEL mating, and series burn. This configuration and a side by side arrangement are shown in Figure 4.3-1.

FIGURE 4.2-1

SOLID-ROCKET PARALLEL-BURN SYSTEMS COMPARISON

BAY SIZE	15 x 60			14 x 45	
BOOSTER	2-156 IN.	4-120 IN.	5-120 IN.	2-156 IN.	4-120 IN.
WEIGHTS (K LB)					
PAYLOAD (EAST)	65.0	65.0	65.0	45.0	45.0
GLOW	4315.2	4662.4	5246.1	3617.4	4382.2
OLOW	1898.6	1853.1	1733.8	1691.2	1572.4
USABLE PROPELLANT	1577.9	1533.8	1417.7	1407.7	1292.3
ORBITER DRY	148.8	148.8	148.8	139.1	139.1
TANK DRY	64.8	63.5	60.6	60.3	57.2
BLOW	2416.5	2809.8	3512.3	1926.2	2809.8
USABLE PROPELLANT	2 x 1064.5	4 x 607.4	5 x 607.4	2 x 848.5	4 x 607.4
BOOSTER INERT	(2 x 143.8)	(4 x 95.0)	(5 x 95.0)	(2 x 114.6)	(4 x 95.0)
COSTS (\$M)					
TOTAL PROGRAM	10207	11826	12817	9681	11628
RDT&E	4126	4006	4021	4006	3906
PEAK FUNDING (YR)	946(76)	922(76)	975(81)	896(76)	887(76)
PER FLIGHT (TOTAL)	10.49	14.11	16.11	9.83	13.94
OPERATIONS	3.80	4.06	4.19	3.78	4.01
TANKS	1.49	1.47	1.42	1.41	1.35
EXPENDABLE BOOSTER HARDWARE	5.20	8.58	10.50	4.64	8.58

FIGURE 4.2-2

PARALLEL SOLID BOOSTER ISSUES

ITEM	RESULTS																				
ALTERNATIVES 2-156 IN. SRM'S VS 4-120 IN. 7 SEGMENT VS 5-120 IN. 7 SEGMENT	• Δ COSTS (\$M):<table><tr><td></td><td></td><th colspan="2">120 IN. 7-SEGMENT</th></tr><tr><td></td><td>2-156 IN. SYSTEM</td><th>4 SRM SYSTEM</th><th>5 SRM SYSTEM</th></tr><tr><td>PROGRAM</td><td>REF</td><td>+ 1619</td><td>+ 2610</td></tr><tr><td>RDT&E</td><td>REF</td><td>-120</td><td>-105</td></tr><tr><td>PER FLIGHT</td><td>REF</td><td>+ 3.62</td><td>+ 5.62</td></tr></table> • CONCLUSION: NUMBERS SRM'S PER STAGE HAS STRONGER IMPACT ON COST THAN SIZE OF INDIVIDUAL SRM			120 IN. 7-SEGMENT			2-156 IN. SYSTEM	4 SRM SYSTEM	5 SRM SYSTEM	PROGRAM	REF	+ 1619	+ 2610	RDT&E	REF	-120	-105	PER FLIGHT	REF	+ 3.62	+ 5.62
		120 IN. 7-SEGMENT																			
	2-156 IN. SYSTEM	4 SRM SYSTEM	5 SRM SYSTEM																		
PROGRAM	REF	+ 1619	+ 2610																		
RDT&E	REF	-120	-105																		
PER FLIGHT	REF	+ 3.62	+ 5.62																		
ENVIRONMENT EFFECTS	• NOT A TECHNICAL PROBLEM BUT MAY BECOME A POLITICAL ISSUE - DISPERSION MODELS INDICATE LOW PROBABILITY OF DANGEROUS CONCENTRATIONS - LAUNCH IN RAIN STORMS MAY BE CONSTRAINT																				
RECOVERY OF SRM'S	• SIGNIFICANT COST SAVINGS<table><tr><td>Δ COSTS (\$M)</td><td></td></tr><tr><td>PROGRAM</td><td>-646</td></tr><tr><td>RDT&E</td><td>+ 46</td></tr><tr><td>PER FLIGHT</td><td>-1.33</td></tr></table>	Δ COSTS (\$M)		PROGRAM	-646	RDT&E	+ 46	PER FLIGHT	-1.33												
Δ COSTS (\$M)																					
PROGRAM	-646																				
RDT&E	+ 46																				
PER FLIGHT	-1.33																				
RELIABILITY OF SRMS	• ADEQUATE DESIGN MARGINS & GROUND DEVELOPMENT TESTING SHOULD PROVIDE RELIABILITY OF ~ .9981 • ABORT CAPABILITIES SHOULD COVER REMAINDER OF FAILURES																				

FIGURE 4.2-3

DISCRIMINATORS: SERIES BURN LIQUIDS vs PARALLEL BURN SOLID BOOSTERS

ITEM	SERIES LIQUIDS	PARALLEL SOLIDS	REMARKS															
• ENGINE GIMBAL TRAVEL - ORBITER - BOOSTER - THRUST MISALIGNMENT VARIATIONS	✓ ✓	✓	NO ASCENT CONTROL BY ORBITER IN SERIES BURN. NO TVC ON PARALLEL BURN SOLIDS. THRUST IMBALANCE SOLID TO SOLID, BUT ORBITER ENGINES CAN HANDLE.															
• CONFIDENCE ORBITER ENGINE START		✓	IGNITED FROM LIFT-OFF															
• NO TANK MASS FRACTION		✓	LOADING CONDITIONS MORE FAVORABLE FOR TWIN SOLIDS															
• WEIGHT		✓	PARALLEL LIGHTER IN GLOW 1.46 M LB PR.FD. 0.143 M LB F-1															
• RECOVERY, REFURBISHMENT		✓	RECOVERY NOT REQUIRED FOR SOLIDS															
• STRUCTURAL DYNAMICS	✓	✓	LONG, AND FLEXIBLE FOR SERIES, VS ASYMMETRICAL FOR PARALLEL															
• BASE HEATING	✓		BASE OF NO TANK AND ORBITER EXPOSED. 500 LB TPS PENALTY															
• ORBITER DRY WEIGHT	✓	✓	PARALLEL SAVES 55 LB (INSIGNIFICANT)															
• ACOUSTICS - PAYLOAD - BASE OF ORBITER	✓ ✓		PAYLOAD FARTHER FROM BOOSTER ENGINES, BUT TWIN SOLID ENVIRONMENT O.K. OVERALL SOUND PRESSURE LEVEL ON ORBITER HIGHER FOR PARALLEL, BUT RSI ATTENUATES TO NO PENALTY.															
• ABORTS - PROBABILITY OF LOSS OF THRUST OF A BOOSTER ENGINE - DIFFICULTY OF THRUST TERMINATION	✓ ✓ ✓	✓ ✓	NO ABORT CAPS. BOTH BOOSTERS REQUIRE THRUST TERMINATION SOLIDS HIGHLY RELIABLE															
• SEPARATION	✓		SIV-B TYPE SEPARATION vs HINGED ROTATING METHOD															
• GROWTH CAPABILITY	✓	✓	NEARLY EQUAL AT $V_0 = 4000$ FPS															
• COSTS - TOTAL PROGRAM - RDT&E - COST/FLIGHT	✓ ✓ ✓	✓ ✓ ✓	<table><tr><th>PARALLEL SOLIDS</th><th colspan="2">SERIES LIQUIDS</th></tr><tr><td></td><th>PF</th><th>F-1</th></tr><tr><td>10,207</td><td>10,798</td><td>10,002</td></tr><tr><td>4,126</td><td>5,676</td><td>5,018</td></tr><tr><td>10.49</td><td>7.50</td><td>7.56</td></tr></table>	PARALLEL SOLIDS	SERIES LIQUIDS			PF	F-1	10,207	10,798	10,002	4,126	5,676	5,018	10.49	7.50	7.56
PARALLEL SOLIDS	SERIES LIQUIDS																	
	PF	F-1																
10,207	10,798	10,002																
4,126	5,676	5,018																
10.49	7.50	7.56																

SYSTEM TECHNICAL RISK COMPARISON

15 MARCH 1972

MDC E0558

PART II - VOLUME I

• LARGE ORBITER, 15' x 60' BAY

• EASTERLY MISSION, 65 K LB PAYLOAD

FIGURE 4.2-4

ITEM	WEIGHT FACTOR	SERIES								PARALLEL SOLIDS			
		PRESSURE FED			PUMP FED								
		6 PRESSURE FED	BASE RISK FACTOR	WTD. RISK FACTOR	4 F-1	BASE RISK FACTOR	WTD. RISK FACTOR	10 HiPc	BASE RISK FACTOR	WTD. RISK FACTOR	TWIN 156"	BASE RISK FACTOR	WTD. RISK FACTOR
BOOSTER SIZE	1	HEAVIEST	4	4		3	3	LIGHTEST	2	2	LIGHT & EXPEND-ABLE	1	1
ASCENT FLIGHT MECHANICS & CONTROL	2	FINS?	2.5	5	FINS? MARG'L ENG. OUT	3	6	FINS?	2.5	5	TVC?	2	4
ASCENT AERODYNAMICS	1	AERO DRAG	3	3	SAME AS PRESSURE FED	3	3	SAME AS PRESSURE FED	3	3	LOW DRAG	1	1
LOADS	1	BODY BENDING	2	2	SAME AS F-1	2	2	SAME AS F-1	2	2	ACOUSTIC HIGHER αq	4	4
SEPARATION	1	CONVENTIONAL	2	2	CONVENTIONAL	2	2	CONVENTIONAL	2	2	WING CLEARANCE	4	4
ABORT	2	LOW PROB-ABILITY	1	2	HIGHER PROBABILITY	3	6	HIGHER PROBABILITY	3	6	LOW PROB-ABILITY THRUST TERMINATION REQUIRED	3	6
BOOSTER ENGINE	2	NEW	4	8	EXISTING	2	4	MOD. FROM ORBITER	3	6	SOME DEVEL- OPMENT SIMPLE SYSTEM	1	2
BOOSTER PROPULSION ATTACHMENT & BOOSTER STRUCTURE	1	HEAVY RESIDUALS	3	3	FEWER ENGINES	2	2	LARGEST NO. OF ENGINES	4	4	NO REUSE REQ'D	1	1
BOOSTER RECOVERY SYSTEM OPERATIONS	2	RUGGED TANK MATERIAL	4	8	WATER IMPACT CRITICAL	2.5	5	SAME AS F-1	2.5	5	BREAKUP ACCEPTA- BLE	1	2
	2	ROCKETS	2	4	WITH SUSTAINER ROCKETS	3	6	WITH SUSTAINER ROCKETS	4	8	EXPEND-ABLE	1	2
	2	BOOSTER CERTIFICA- TION	2	4	ALSO ENGINE WATER DE-VELOPMENT	3	6	SAME AS F-1	4	8	EXPEND-ABLE	1	2
TESTING AND DEVELOPMENT DEPTH OF ANALYSIS	2		3	6		3	6		3	6	LESS TESTING	1	2
	1	SUB- STANTIAL	1.5	1.5	LOWER	3	3	LOWEST	4	4	SUBSTAN- TIAL	1.5	1.5
TOTAL			34	52.5		34.5	54		39	61		22.5	32.5

FIGURE 4.2-5

SYSTEM COST COMPARISON

1-216A

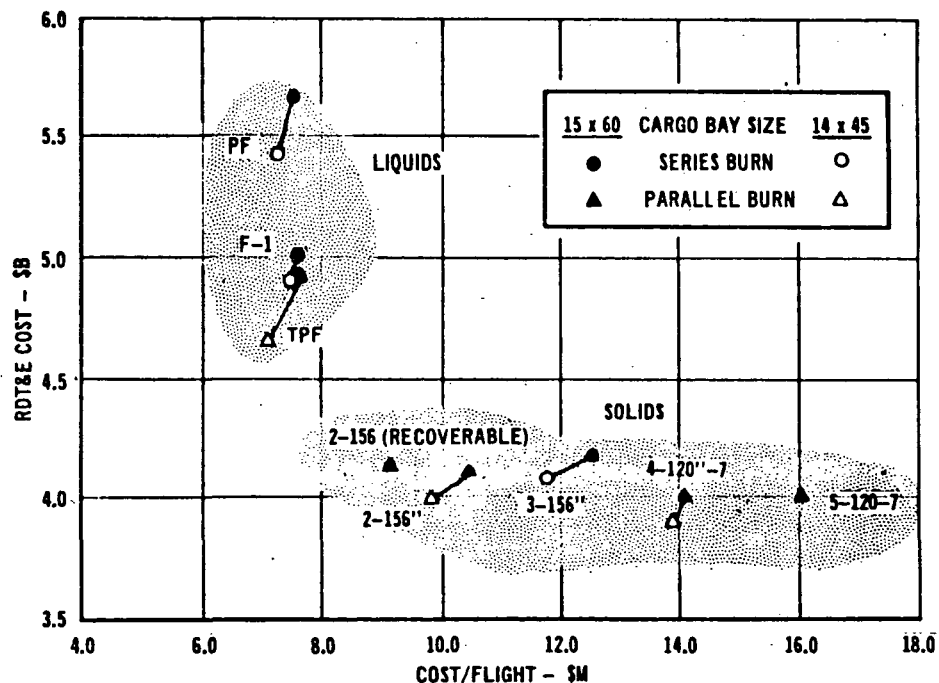
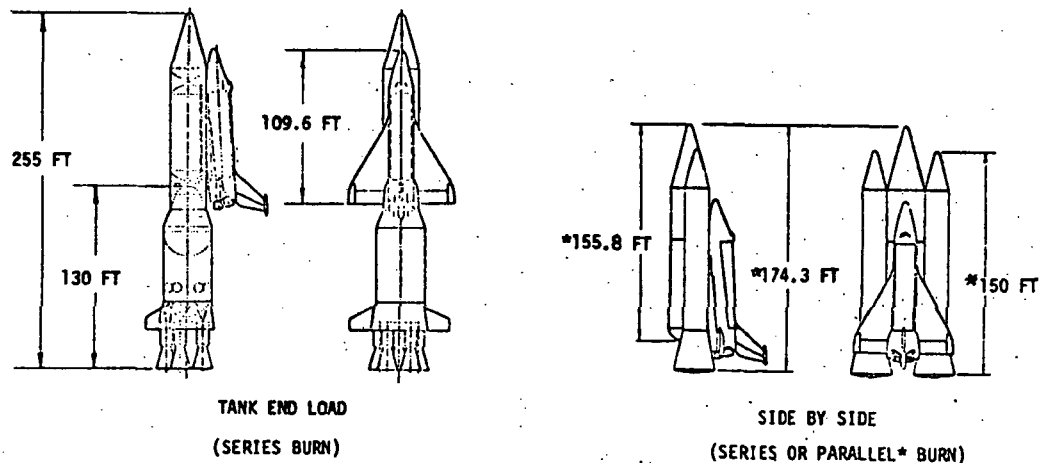


FIGURE 4.3-1
UNMANNED RECOVERABLE PRESSURE-FED
BOOSTER LAUNCH ARRANGEMENTS



The comparison criteria which were considered in the TEL vs SS configuration trade are summarized in Figure 4.3-2. The check marks indicate a preference and the dashes indicate that no clear preference could be identified. The items in favor of TEL definitely predominate for the series burn system. Details of analyses supporting these assessments were published in Design Note No. I-East-TI-34, dated Nov 71.

The one important item that tended to favor SS mating was Orbiter-with-tank abort separation prior to depletion of Booster propellant. This factor did not impact the unmanned pressure-fed system since it could be expendable in an emergency. The manned RF-1B Booster would require modifications (such as engine restart capability or addition of a dump system) if the abort gap for the Booster is to be minimized.

4.4 Orbiter Engine Selection

The recommended main engine configuration for the baseline (15 x 60 ft payload) Orbiter is three high chamber pressure, oxygen/hydrogen engines, each of which delivers 470,000 lb of vacuum thrust. This selection is the result of a series of trade studies in which various J-2 derivative engine designs, as well

FIGURE 4.3-2

TANK END LOADED VS SIDE BY SIDE MATING COMPARISON SUMMARY

- TEL = TANK END LOADED: SINGLE HO TANK
- SS = PARALLEL SIDE MATING: TWIN SHOULDER HO TANKS

CRITERIA	RF-1B		PFRB		COMMENTS
	TEL	SS	TEL	SS	
WEIGHTS:					TEL Δ WEIGHT (K LB):
- GLOW	✓		✓		RF-1B PFRB
					-148 -401
- DRY ORBITER	✓		✓		-5 -5
- DRY TANKS	✓		✓		-2.5 -3.3
- DRY BOOSTER		✓	✓		+28 -33
PROGRAM COST	-	-	✓		TEL RF-1B = + \$3 M Δ COST
					TEL PFRB = -\$140 M Δ COST
TANK COST	✓		✓		TEL SINGLE TANK CHEAPER
LAUNCH OPERATIONS & FACILITIES	-	-	-	-	PROS & CONS ABOUT BALANCED
CARGO ACCESS ON THE PAD	✓		✓		SS TANKS OBSTRUCT BAY DOORS
MATED ASCENT DRAG	✓		✓		TEL MATED ASCENT $C_D = 2/3$ OF SS C_D
ASCENT GUIDANCE & CONTROL:					TEL LESS C.G. GIMBAL TRACKING &
- MATED BOOST	✓		✓		LARGER CONTROL MOMENTS
- ORBITER ONLY ASCENT		✓		✓	TEL HAS LARGER YAW/ROLL COUPLING
					GIMBAL ANGLE REQUIREMENT
ASCENT HEATING	✓		✓		TEL LESS INTERFERENCE HEATING
STRUCTURAL DYNAMICS:					
- INTERFERENCE FLUTTER	✓		✓		SS HAS POTENTIAL WING INTERFERENCE
					PROBLEM
- FLEXIBLE BODY	✓		✓		FIRST BENDING MODE > 1.1 Hz FOR BOTH.
					SS ROLL MODE MAY HAVE COUPLING
					PROBLEM
ABORT:					
- SEPARATE AT BOOSTER BURNOUT	-	-	-	-	TEL & SS ABOUT EQUAL
- SEPARATE BEFORE BOOSTER					
DEPLETION -					
ORBITER & TANK FLY BACK		✓	-	-	TEL REQUIRES THRUST REDUCTION TO
					60-10% FOR ORBITER TO OUTFRAN
					BOOSTER
ORBITER ONLY GLIDE BACK	✓		✓		TEL REQUIRES ONLY ONE SEPARATION
					OPERATION; SS REQUIRES THREE

15 MARCH 1972

as the HiPc SSME, were evaluated and variations in engine number and thrust level were assessed.

The engine selection studies began in response to Technical Directive 71 which specified either a J-2 or a J-2S engine for a Mark I Orbiter and development of an interchangeable, high Pc engine for Mark II. As reported in Design Note I-EAST-TI-36, the J-2S engine was recommended for Mark I and four 306,000 lb HiPc engines (constrained envelope design) for Mark II. However, the study results also indicated that use of a single engine design for both Mark I and Mark II would provide a significant cost advantage.

The evaluation of candidate Orbiter engine designs is summarized in Figure 4.4-1. Both J-2S and HiPc engines were considered, along with variations in the number of engines and thrust per HiPc engine. The recommended configuration was three 450,000 lb vacuum thrust HiPc engines. The vacuum thrust was subsequently revised to 470,000 lb to be compatible with the 415,000 lb reference sea level thrust of the powerhead in the original ICD engine used as an early point design. The selection of 470,000 lb was later validated for revised Orbiter and Booster design characteristics. The results are shown in Figure 4.4-2 for parallel and series burn vehicles and for both a 15 x 60 and a 14 x 45 ft payload bay. Details of the sizing/performance analysis are presented in Section 5.0. For the baseline 15 x 60 ft payload bay, 470,000 lb per engine provides near minimum cost for either parallel or series burn. This was based on the NASA specified minimum allowable staging velocity of 4000 ft/sec. Lower staging velocities could reduce costs but would require more thrust from the Orbiter engines. If a 14 x 45 ft payload bay is used the optimum-cost thrust level for 4000 fps staging would be reduced to about 420,000 lb per engine.

4.5 Payload Bay Size

Reducing the size of the payload bay has been repeatedly studied as a device for reducing Shuttle Program costs. Early in Phase B such a study indicated that program costs could be reduced by about \$200M but accommodation of payloads would be extensively restricted. As each major change in Shuttle concept was introduced, reduction of payload bay size was again studied: notably in the consideration of alternate tankage concepts and again in a study of alternate Booster concepts. These two studies are described below.

15 MARCH 1972

FIGURE 4.4-1

ORBITER MAIN ENGINE SYSTEM SELECTION

CONSIDERATIONS

• ORBITER THRUST OPTIONS

4-6 265K J-2S
4 320K J-2S UPGRATED
3-4 300K - 450K Hi Pc

• STUDY PARAMETERS

- ORBITER GEOMETRY/WEIGHT EFFECTS
- V_{STG} VS ORBITER THRUST LEVEL
- V_{STG}, GROWTH CAPABILITY, THRUST LEVEL INTERRELATIONSHIPS
- PROGRAM COSTS

COST COMPARISON

BASELINE CONCEPTS	BASELINE J-2S → HiPc	DELTA'S FROM BASELINE			
		J-2S (ALONE)	HiPc (ALONE)		
			4-306 K LB (VAC)	3-450 K LB (VAC)	4-450 K LB (VAC)
FLYBACK/SERIES BURN					
RDT&E	7,398	-430	-209	-131	-85
PER FLIGHT	6.16	+0.06	0	-0.01	-0.04
TOTAL PROGRAM	12,863	-493	-287	-214	-172
PAF	1,319 (76)	-12	+37	+49	+51
SINGLE RAO/SERIES BURN					
RDT&E	5,726	-262	-209	-199	-180
PER FLIGHT	6.46	+0.03	0	0	+0.02
TOTAL PROGRAM	10,894	-330	-287	-320	-300
PAF	1,046 (78)	+22	+37	+36	+32
TWIN RAO/PARALLEL BURN					
RDT&E	5,199	-213	-209	-208	-258
PER FLIGHT	6.87	+0.10	0	-0.01	-0.01
TOTAL PROGRAM	10,469	-120	-287	-217	-807
PAF	939 (78)	+32	+37	+25	+16

THREE ENGINES PREFERRED TO FOUR

• ASSUMING CONSTANT TOTAL ORBITER THRUST

CRITERION	NUMBER OF ENGINES		
	3	4	
SYSTEM RELIABILITY	✓		• SYSTEM RELIABILITY HIGHER WITH FEWER ENGINES PROVIDED
ABORT GAPS	-	-	• NO SIGNIFICANT DISCRIMINATOR BETWEEN 3 AND 4 ENGINE SYSTEM
DESIGN	✓		• FEWER ENGINES ALLOW VEHICLE INTERFACES TO BE SIMPLIFIED
MAINTENANCE	✓		• MANHOURS - 3 HiPc 96 MMH/FLT 4 HiPc 128 MMH/FLT 4 J-2S 160 MMH/FLT
	✓		• FIELD OPERATIONS (SUPPORT, SPARES, TURNAROUND TIME) FAVOR FEWER ENGINES
FLIGHT OPERATIONS	✓		• DATA MONITORING, INSTRUMENTATION, ELECTRICAL POWER CONSIDERATIONS FAVOR FEWER ENGINES
ORBITER WEIGHT		✓	• THREE ENGINE SYSTEM 100 LB HEAVIER

CONCLUSIONS

- SINGLE ENGINE DEVELOPMENT PREFERRED
- 4 J-2S ENGINE SYSTEM FEASIBLE FOR BOTH SINGLE AND TWIN RAO (PRESSURE-FED) V_{STG} > 7000 FPS
- HIGHER ORBITER THRUST ALLOWS REDUCED V_{STG} AND LOWER COSTS
- "UPGRATED" J-2S BETTER THAN J-2S, BUT HiPc IS SUPERIOR
- 3 - 450 K ENGINES REPRESENTS GOOD COMPROMISE FOR BOTH SINGLE AND TWIN RAO (PRESSURE-FED) SYSTEMS
- 4-450K ENGINES WOULD SAVE MORE THAN \$100 M IN SYSTEM COST FOR PARALLEL BURN. EXPENSE OF ORBITER AERO/CG DESIGN PROBLEMS NOT ACCOUNTED FOR.
- 3 - 450 K ENGINES IS RECOMMENDED SELECTION

15 MARCH 1972

4.5.1 Payload Size with Alternate Tank Concepts - Three concepts of Orbiter propellant tankage were investigated to determine their sensitivity to payload bay size. These three Orbiters had the following characteristics when defined in their 15 x 60 ft payload bay configuration:

15 x 60 Bay Configurations	Single H0 Tank	Twin-H Tanks	No External Tank
Volume (Cu Ft)			
Internal Main Propellant	0	11,050	25,398
Occupied Fuselage Volume	17,255	28,135	43,340
OML Volume	39,172	57,400	76,095
Packaging Efficiency	0.44	0.49	0.56

Each Orbiter design was modified to accommodate several payload bay sizes. Allowances were made for resizing aero surfaces, surface controls, landing systems, and the auxiliary propulsion system. These modified Orbiters, their expendable tanks and fully reusable Boosters were then resized to provide 40,000 lb payload in a polar orbit.

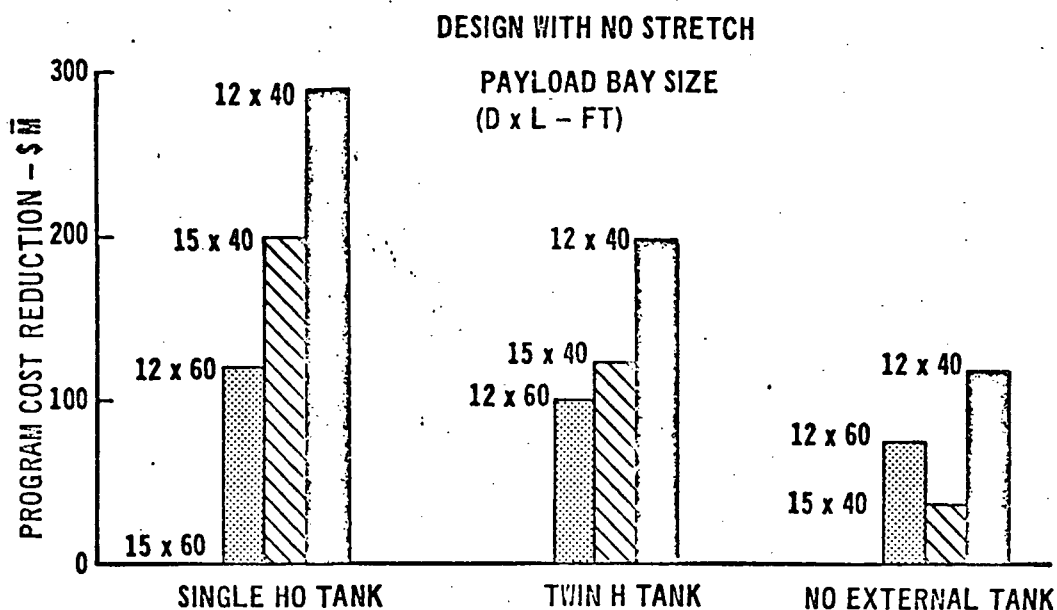
The resulting program cost trends are shown in Figure 4.5.1-1. Vehicles with larger volumes of internal main propellant are less sensitive to changes in payload size. For example, when the payload bay of the H0-tank Orbiter (zero internal propellant) is shortened by 20 ft, the fuselage is likewise shortened by 20 ft and considerable unused volume beneath the bay can be eliminated. On the other hand, when the bay is shortened on the all-internal propellant configuration the tankage eliminated under the bay must be contained in a larger fuselage.

At the time of this study, the Shuttle Program was defined as a two-phase program with an interim expendable Booster and Mark I Orbiter design. The initial premise was to design the Orbiter with a short payload bay and evolve to a longer one (with no diameter change) later in the program. The longer bays would then be employed with the fully reusable Booster. It was determined that stretching the delta-wing Orbiter was impractical. Because of the blended wing-body, aerodynamic balance and wing fairing requirements precluded transition to a longer bay without costly design compromises.

15 MARCH 1972

FIGURE 4.5.1-1

EFFECT OF PAYLOAD BAY SIZE ON COST



The swept-wing Orbiter, however, was found to be amenable to stretching, but the cost benefits of the small bay were severely limited even for the HO tank system. A 12 x 40 bay program which grows to a 12 x 60 bay would cost the same as an entire program at 12 x 60, plus an additional \$30M cost for the dual vehicle design which must be carried along. With the smaller payload bay diameter, the program peak funding is reduced by only \$13M, but about 55% of the anticipated NASA and DoD payload traffic opportunities are lost.

It was therefore recommended that shuttle designs continue with the baseline 15 x 60 ft payload bay.

4.5.2 Payload Size for Current Orbiter Configuration - Two payload weight and size options were specified for detailed comparison during the final weeks of the study.

Baseline: 15 x 60 ft bay
65,000 lb easterly ascent payload
40,000 lb payload landed

15 MARCH 1972

Alternate: 14 x 45 ft bay
45,000 lb easterly ascent payload
25,000 lb payload landed

These dimensions are the allowable payload envelope for which the actual Orbiter bay design provides a minimum radial clearance of three inches and about one foot clearance at each end.

The baseline 15 x 60 ft bay Orbiter is the result of in-depth weight analysis of design drawings (basic drawing - 068B) based on several iterations of aerodynamic and structures analyses. The alternate 14 x 45 bay Orbiter design (drawing - 085A) was derived primarily by parametric analyses. The validity of this approach was checked by comparing dry weight trends with those of commercial aircraft (Figure 4.5.2-1), thus enhancing the confidence in the results.

Figure 4.5.2-2 summarizes the basic dimensional and weight changes realized when both the payload bay and weight are reduced. Figure 4.5.2-3 expands the definition of the smaller Orbiter (right two columns) and also shows an intermediate case, in which the large bay is retained and only the payload weight is reduced. This intermediate change reduces the dry weight of the Orbiter 4720 lb whereas the smaller bay results in a reduction of 9655 lb. The attendant Orbiter cost reductions are \$104M, half of which is attributable to reduced payload size and the other half to reduced landed weight.

Boosters and HO tanks were resized and costed with the minimum Orbiter. Program cost savings ranged from \$202M for the F-1 Pump-Fed System to \$576M for the Parallel Burn Pressure Fed System. Cost savings for the Series Burn Pressure Fed and Parallel Burn Twin 156" SRM are \$447M and \$526M, respectively. About 3/4 of this savings results from payload weight reductions, and only 1/4 from bay size reductions.

Bay size reduction is extremely restrictive to mission accomplishment. Use of the 14 x 45 ft bay would preclude about one sixth of the NASA missions. About two thirds of the DoD missions could be accomplished with the smaller bay, but most would require dual launch.

15 MARCH 1972

FIGURE 4.5.2-1
WEIGHT TREND

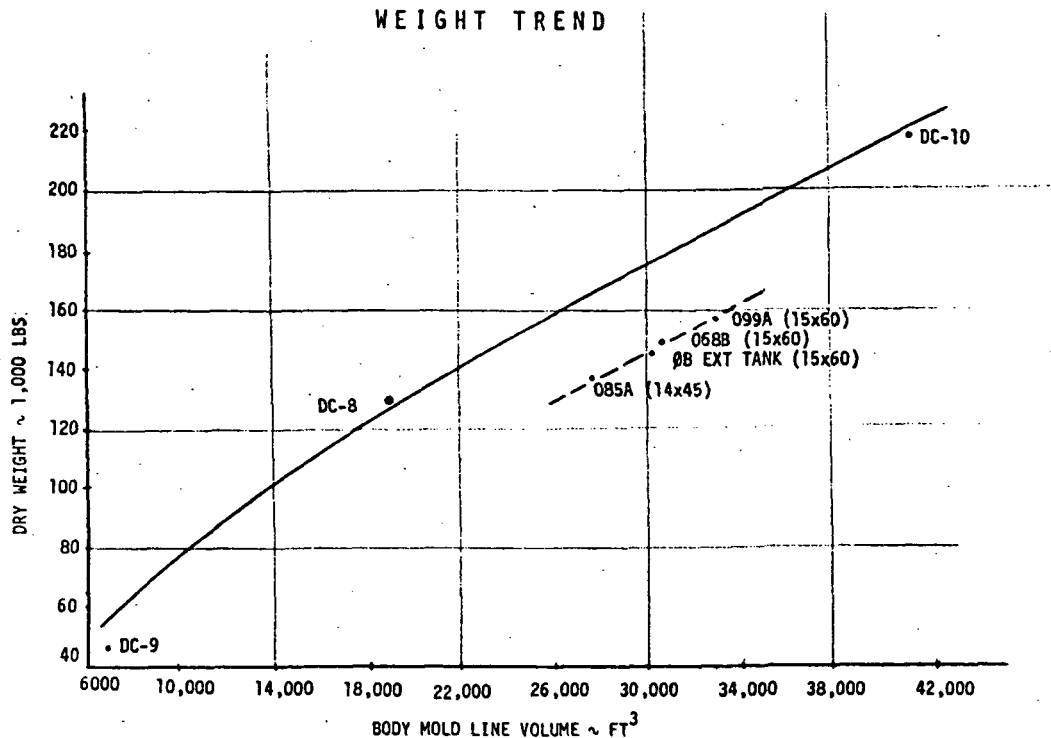


FIGURE 4.5.2-2
ORBITER BAY SIZE
DESIGN COMPARISON

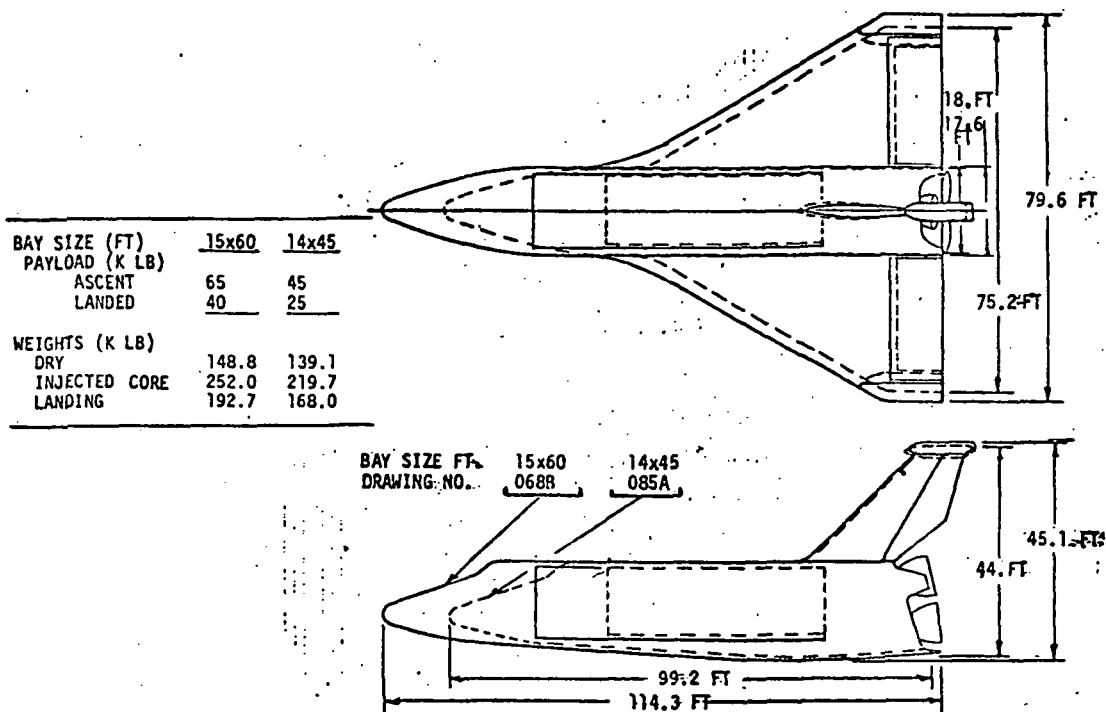


FIGURE 4.5.2-3
ORBITER GROUP WEIGHT COMPARISONS

• EASTERLY MISSION WEIGHTS - LB

BAY SIZE	15 x 60		15 x 60		14 x 45
PAYLOAD WEIGHT REQUIREMENT	65K	45K	ΔWT	45K	ΔWT
WING GROUP	16,214	14,474	-1,740	14,305	-1,909
TAIL GROUP	2,766	2,706	-60	2,643	-123
BODY GROUP	28,620	28,620	-	26,120	-2,500
INDUCED ENVIRONMENT PROTECTION	25,602	24,042	-1,560	22,771	-2,831
LANDING, RECOVERY, DOCKING	7,977	7,447	-530	7,249	-728
PROPULSION-ASCENT	23,000	23,000	-	23,000	-
PROPULSION-CRUISE	0	0	-	0	-
PROPULSION-AUXILIARY	6,483	6,283	-200	6,261	-222
PRIME POWER	4,020	4,020	-	4,020	-
ELECT. CONVER. AND DISTRIBUTION	2,530	2,530	-	2,530	-
HYDRAULIC CONVER. AND DISTRIBUTION	1,510	1,510	-	1,510	-
SURFACE CONTROLS	4,997	4,797	-200	4,525	-472
AVIONICS	5,730	5,730	-	5,730	-
ENVIRONMENTAL CONTROL	6,550	6,550	-	6,550	-
PERSONNEL PROVISIONS	800	800	-	800	-
RANGE SAFETY AND ABORT	0	0	-	0	-
BALLAST	0	0	-	0	-
GROWTH UNCERTAINTY	12,008	11,578	-430	11,128	-880
SUBTOTAL (DRY WEIGHT)	(148,807)	(144,087)	(-4,720)	(139,142)	(-9,665)
PERSONNEL	1,200	1,200	-	1,200	-
CARGO	65,000	45,000	-20,000	45,000	-20,000
ORDNANCE	0	0	-	0	-
RESIDUAL FLUIDS	2,156	2,156	-	2,156	-
SUBTOTAL (INERT WEIGHT)	(217,163)	(192,443)	(-24,720)	(187,499)	(-29,665)
RESERVE FLUIDS	13,915	12,605	-1,310	12,472	-1,443
INFLIGHT LOSSES	5,443	5,443	-	5,443	-
PROPELLANT-ASCENT	0	0	-	0	-
PROPELLANT-CRUISE	0	0	-	0	-
PROPELLANT-MANEUVERS ACS	15,501	14,325	-1,176	14,290	-1,211
TOTAL (GROSS-WEIGHT) LB	(252,022)	(224,816)	(-27,206)	(219,703)	(-32,319)
ENTRY WEIGHT - LB	197,710	177,990	(-19,720)	173,045	(-24,665)
LANDING WEIGHT - LB	192,659	172,939	(-19,720)	167,994	(-24,665)
LANDED CARGO	40,000	25,000	(-15,000)	25,000	(-15,000)

4.6 Flight Performance Reserve Summary

The total impulsive velocity reserves required to overcome statistically predictable deviations of parameters that affect vehicle performance have been evaluated for several vehicle configurations. The vehicle configurations analyzed were:

Parallel Burn	2 x 156" SRM ($V_s = 4000$ fps)
Parallel Burn	Pressure Fed Booster ($V_s = 4000$ fps)
Series Burn	Pressure Fed Booster ($V_s = 4000$ fps)
Series Burn	Pressure Fed Booster ($V_s = 5000$ fps)
Series Burn	F-1 Pump Fed Booster ($V_s = 5000$ fps)

The parameters that contribute to the error sources include vacuum specific impulse and thrust, mixture ratio, total propellant loading, inert weight, G & N error, winds, and axial force coefficient. The effect of each on Orbiter velocity reserve was evaluated individually, and the reserve was then calculated as the "root sum square" of the individual values. The analysis treated both closed-loop

15 MARCH 1972

and open-loop PU systems for the liquid propellant systems. The results are compared in Figure 4.6-1 to the 1% of total velocity which has been used for sizing.

The flight performance reserve requirements for the parallel SRM configuration and for both the series burn pressure fed booster and pump fed (F-1) booster configurations compare favorably with the available reserves. Requirements for systems with closed loop PU in the Orbiter (baseline) fall below that available for these configurations. Only in the case of the twin pressure fed boosters with parallel burn does closed loop PU in the Booster show a substantial advantage. Detailed FPR computations can be found in Part C, Sections 1, 2 and 3.

4.7 Engine Mixture Ratio Range

A study was made to define the engine mixture ratio (EMR) requirements for the HiPc engine. To permit closed loop propellant utilization, the engine must be capable of commanded operation over a range of mixture ratios for error correction. The range required depends upon the tank mixture ratio error, which is a statistical combination of the loading and EMR errors, as well as on the nominal EMR and on the percent of total burn time allocated to correction. The design values of the loading and EMR errors for the Orbiter are: LOX loading, ± 1 percent; LH_2 loading, ± 1 percent; engine EMR, ± 2 percent. Those S-IV and S-IVB flights with closed loop propellant utilization were reviewed, and this led to selection of a value of 50 percent of total burn time for correction.

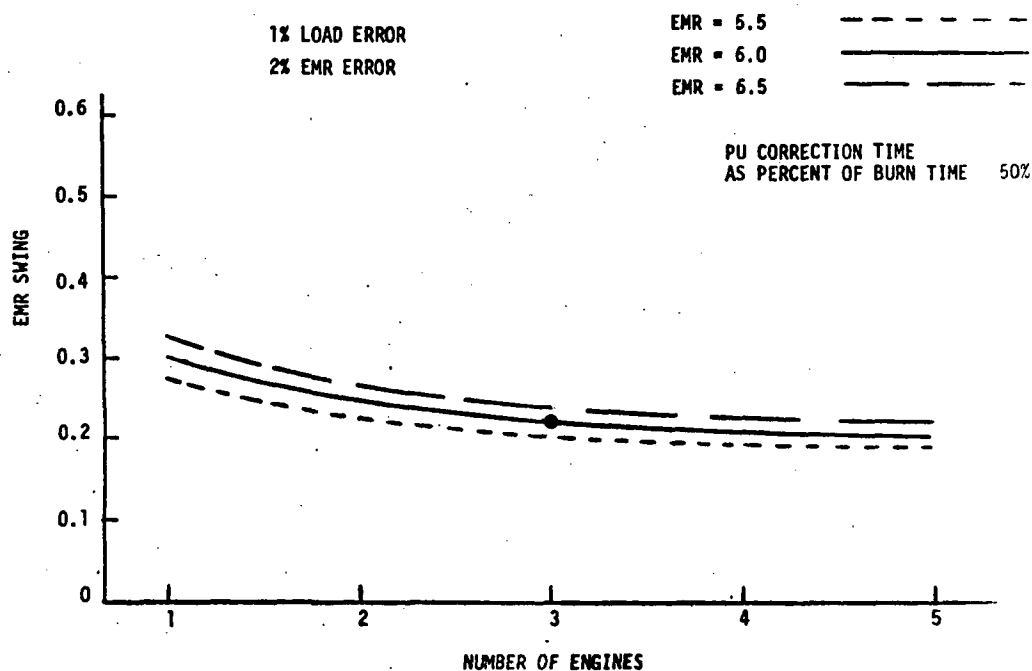
The required EMR range for these conditions is presented in Figure 4.7-1. As shown, an EMR range, or swing, of about $\pm .25$ mixture ratio units is adequate for a three engine Orbiter.

FIGURE 4.6-1
FLIGHT PERFORMANCE RESERVE SUMMARY

CONFIGURATION	Δ ORBITER VELOCITY RESERVE	
	AVAILABLE FROM SIZING (FPR 1%)	REQUIRED
PARALLEL BURN - TSRM ($V_s = 4000$ FPS)		
WITH ORBITER PU:		NOM -220 FPS TO NOM +278 FPS
WITHOUT ORBITER PU:	NOM +307 FPS	NOM -220 FPS TO NOM +397 FPS
PARALLEL BURN - TPFB ($V_s = 4000$ FPS)		
WITH BOOSTER PU:		
WITH ORBITER PU:		NOM -182 FPS TO NOM +252 FPS
WITHOUT ORBITER PU:	NOM +308 FPS	NOM -182 FPS TO NOM +381 FPS
WITHOUT BOOSTER PU:		
WITH ORBITER PU:		NOM -182 FPS TO NOM +1102 FPS
WITHOUT ORBITER PU:		NOM -182 FPS TO NOM +1138 FPS
SERIES BURN - PFB ($V_s = 4000$ FPS)		
WITH BOOSTER PU:		
WITH ORBITER PU:		NOM -143 FPS TO NOM +215 FPS
WITHOUT ORBITER PU:		NOM -143 FPS TO NOM +299 FPS
WITHOUT BOOSTER PU:		
WITH ORBITER PU:	NOM +308 FPS	NOM -143 FPS TO NOM +247 FPS
WITHOUT ORBITER PU:		NOM -143 FPS TO NOM +317 FPS
SERIES BURN - F-1 ($V_s = 5000$ FPS)		
WITH BOOSTER PU:		
WITH ORBITER PU:		NOM -139 FPS TO NOM +197 FPS
WITHOUT ORBITER PU:		NOM -139 FPS TO NOM +263 FPS
WITHOUT BOOSTER PU:		
WITH ORBITER PU:	NOM +302 FPS	NOM -139 FPS TO NOM +243 FPS
WITHOUT ORBITER PU:		NOM -139 FPS TO NOM +299 FPS
SERIES BURN - F-1 ($V_s = 5000$ FPS)		
WITH BOOSTER PU:		
WITH ORBITER PU:		NOM -127 FPS TO NOM +199 FPS
WITHOUT ORBITER PU:	NOM +300 FPS	NOM -127 FPS TO NOM +274 FPS
WITHOUT BOOSTER PU:		
WITH ORBITER PU:		NOM -127 FPS TO NOM +218 FPS
WITHOUT ORBITER PU:		NOM -127 FPS TO NOM +288 FPS

FIGURE 4.7-1

ENGINE MIXTURE RATIO RANGE REQUIREMENT



5.0 System Sizing and Performance

Following detailed analytical studies, a final iteration of the parametric analyses of alternative system concepts was undertaken for the Series Burn Pressure-Fed and Pump-Fed Boosters with 15 x 60 ft payload bay Orbiters, and Twin Solid Parallel Burn Boosters with both 15 x 60 ft and 14 x 45 ft payload bay Orbiters. Comparison data for 4-120" SRM parallel systems are also provided in sizing and cost summary charts.

5.1 Analysis Technique

Figures 5-1 and 5-2 show the booster mass fraction parametrics with the design points noted. The mass fraction of the Parallel Burn 156" motor (SRM) used in the final sizing was 0.881, independent of propellant load. Figure 5-3 shows the variation of the mass fraction of the Orbiter H0 tank with propellant load, with the four design points noted. These curves, plus the Orbiter gross weights from Figure 4.5.2-3, allow the system to be sized for any desired staging velocity, orbital inclination, thrust-to-weight ratio, etc.

The design points sizing summaries based on easterly missions are presented in Figure 5-4 for the following conditions:

<u>System</u>	<u>V_s</u>	<u>$(T/W)_1$</u>
Pressure Fed Series	4000	1.35
Pump Fed (F-1) Series	5000	1.35
SRM Parallel (156")	4000	1.405

5.2 Effect of Staging Velocity

Figure 5-5 shows the variation of lift off weight and total program cost as a function of staging velocity. Minimum cost for the pressure fed series and the 156" SRM parallel cases occurs at staging velocities below the 4000 fps specified as minimum. The pump-fed F-1 series burn case exhibits an opposite trend, because engine hardware costs do not increase with increased staging velocity. Minimum GLOW occurs at about 5000 fps for the F-1 system. This, together with the fact that growth accommodation of the F-1 system at $V_s = 5000$ fps is about equal to that of the pressure-fed system at $V_s = 4000$ fps (see Para. 5-4), led to the selection of these different staging velocities for the purpose of comparing the competing series burn systems.

15 MARCH 1972

FIGURE 5-1

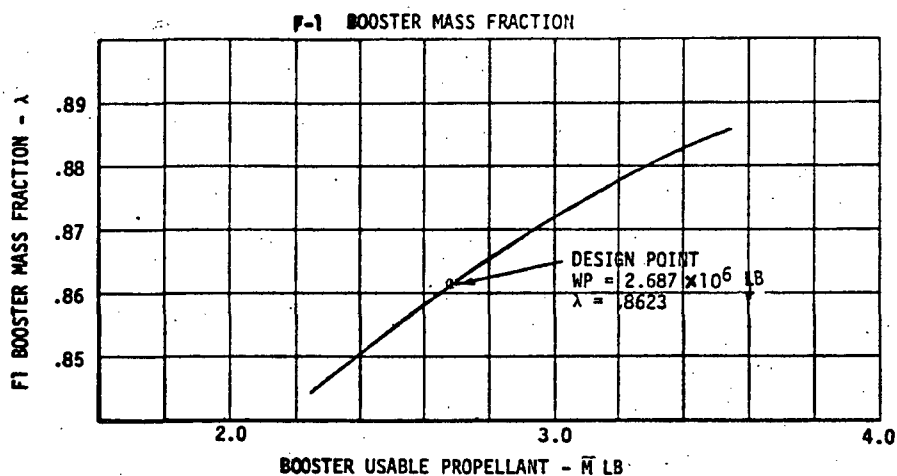


FIGURE 5-2

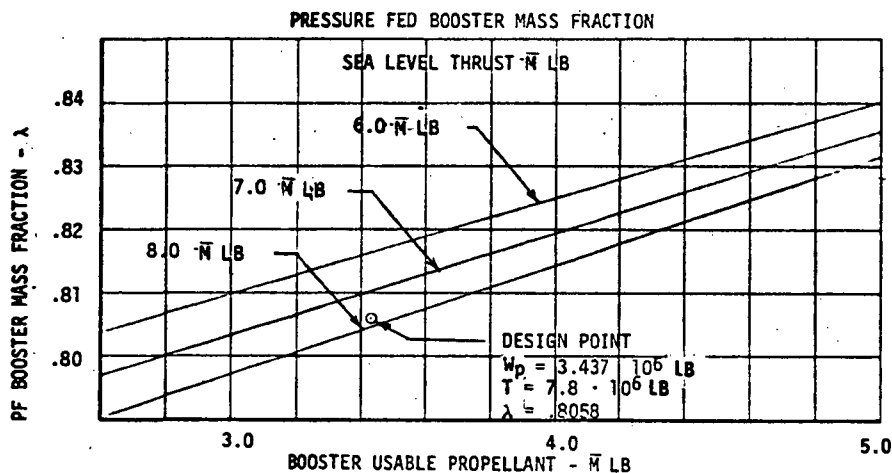
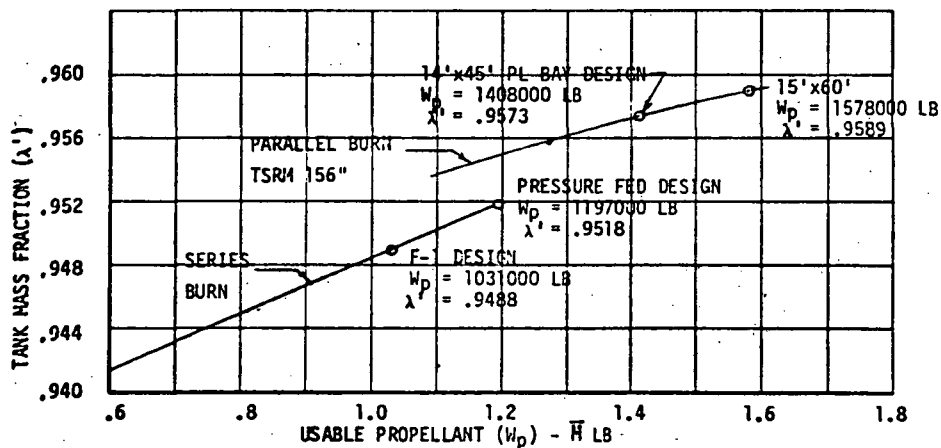


FIGURE 5-3
EXTERNAL TANK MASS FRACTION



15 MARCH 1972

FIGURE 5-4
SYSTEM SIZING SUMMARY

	15' x 60' PL BAY				14' x 45' PL BAY	
	SERIES BURN		PARALLEL BURN			
	PRESSURE FEED	F-1	2-156" SRM	4-1207	2-156" SRM	4-1207
PAYLOAD (K LB): EASTERLY RESUPPLY POLAR	65.0 31.2 40.6	65.0 31.3 40.8	65.0 31.2 40.4	65.0 31.3 40.6	45.0 22.9 23.3	45.0 23.0 24.0
MISSION DESCRIBED	EASTERLY					
ORBITER STAGE:						
CORE WEIGHT (K LB)	252.0	252.0	252.0	252.0	219.7	219.7
ORBITER DRY WT (K LB)	148.8	148.8	148.8	148.8	139.1	139.1
USABLE PROPELLANT (K LB)	1196.6	1031.2	1577.9	1533.8	1407.7	1292.3
START PROPELLANT (K LB)	2.2	2.2	--	--	--	--
TANK MASS FRACTION (LB/LB)	.9518	.9488	.9589	.9586	.9573	.9560
TANK DRY WT (K LB)	58.1	53.5	64.8	63.5	60.3	57.2
ENGINES NO./VAC. THR. PER ENG. (K LB)	3/470	3/470	3/470	3/470	3/470	3/470
VACUUM MIN Isp (SEC)	455	455	455	455	455	455
GLOW (K LB)	1512.2	1341.7	1898.6	1853.1	1691.2	1572.4
BOOSTER STAGE:						
BOOSTER MASS FRACTION (LB/LB)	.8058	.8633	.881	.8647	.881	.8647
INERT WEIGHT (K LB)	828.5	429.2	2x143.8	4x95.0	2x114.6	4x95.0
DRY WEIGHT (K LB)	734.9	372.3	--	--	--	--
PROPELLANT WEIGHT (K LB)	3436.9	2687.3	2x1064.5	4x607.4	2x848.5	4x607.4
ENGINES NO./S.L. THR. PER ENG. (K LB)	6/1300	4/1505	2/2575	4/1397	2x2067.8	4/1397
VACUUM MIN Isp (SEC)	279	304.1	264	262.8	264	262.8
BLOW (K LB)	4265.4	3116.5	2416.5	2809.8	1926.2	2809.8
GLOW (K LB)						
VELOCITIES (FPS)	5777.6	4458.3	4315.2	4662.4	3617.4	4382.2
RELATIVE STAGING						
1ST STAGE IDEAL	4000	5000	4000	3692	4000	4837
2ND STAGE IDEAL	8110	9033	7969	8394	7958	9316
T/W: LIFT-OFF	22984	21468	22760	22553	22448	21424
STAGING (VAC)	1.35	1.35	1.405	1.425	1.405	1.516
	.934	1.053	0.927	.944	1.071	1.162

5.3 Growth Accommodation

For the design points selected, unanticipated system growth can be most economically accommodated by stretching the HO tank, while retaining the design point booster. The capability of the competing systems to do this is summarized in Figure 5-6 for the large Orbiter payload cases. For this analysis, it was assumed that the Booster and Orbiter would grow at the same percentage rate. For 10% HO tank stretch from the nominal design, the twin SRM parallel burn configuration can accept system growth of 2.8% beyond the 10% built into the nominal design; the pressure fed series burn case can accept 3.6%; and the F-1 series burn case 3.8%. The parallel burn SRM case gains most by the use of higher staging velocities. At low staging velocities, growth accommodation is limited primarily by the Orbiter design thrust levels.

5.4 Cost Summary

The costs of the primary emphasis systems are summarized in Figure 5-7. Of the three large payload candidates, the F-1 Series System has the lowest total program cost, while the Parallel SRM case has the lowest RDT&E and cost per flight, even though the SRM's are expendable. The small payload Orbiter saves about \$526M in total program cost, and \$660 K per flight. Total program cost for the 120" SRM systems are the greatest. Although RDT&E costs are low, cost per flight is higher than other systems evaluated.

15 MARCH 1972

FIGURE 5-5

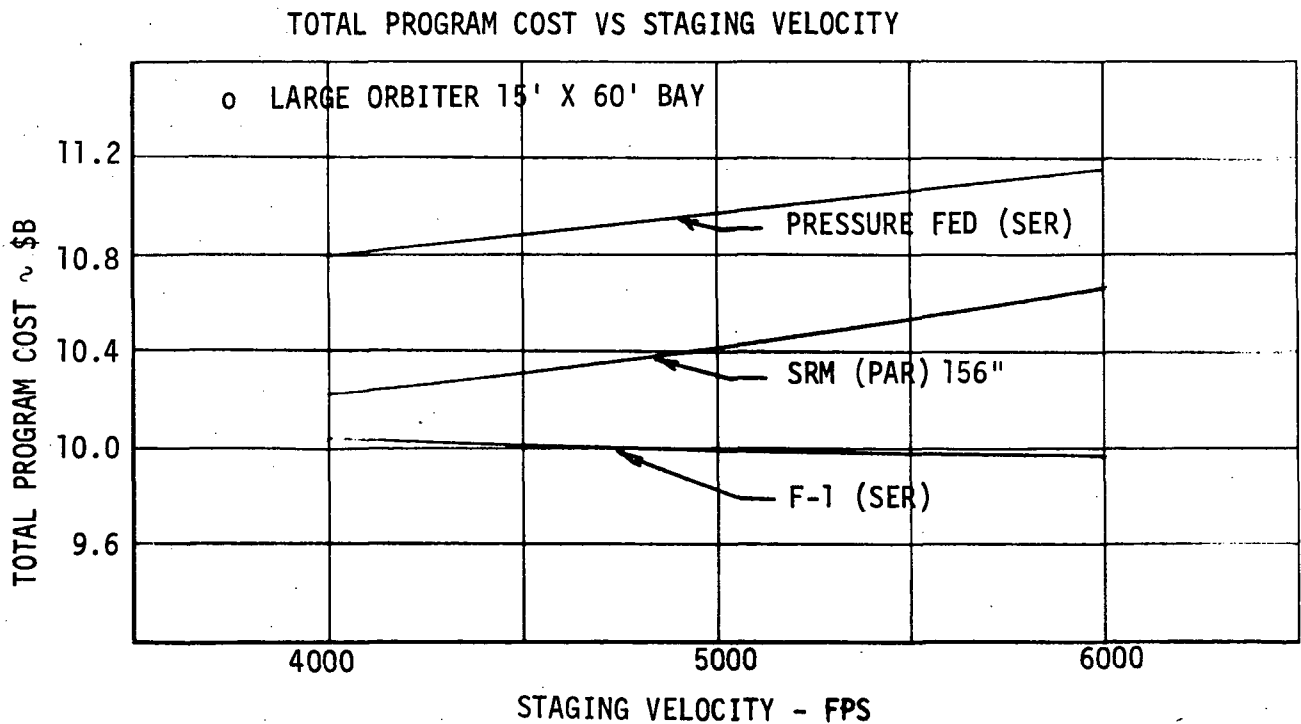
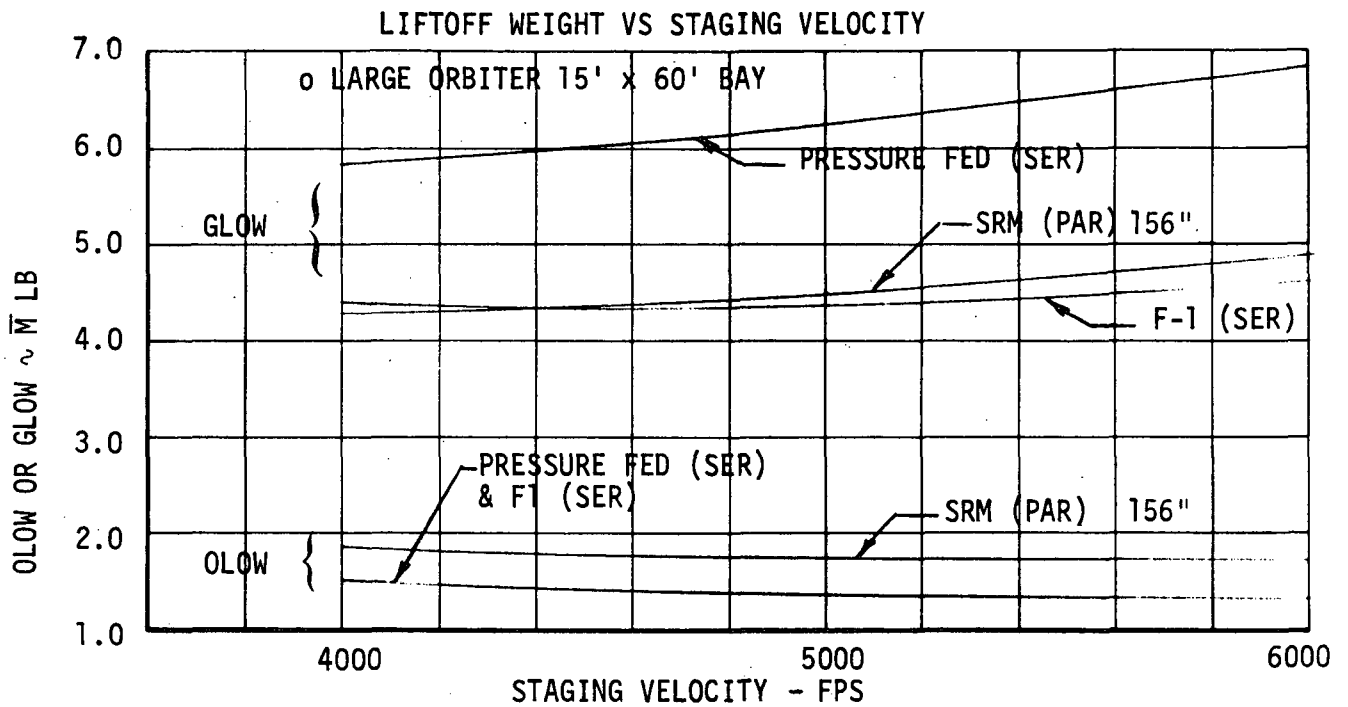


FIGURE 5-6

GROWTH ACCOMMODATION

• LARGE ORBITER 15' x 60' BAY

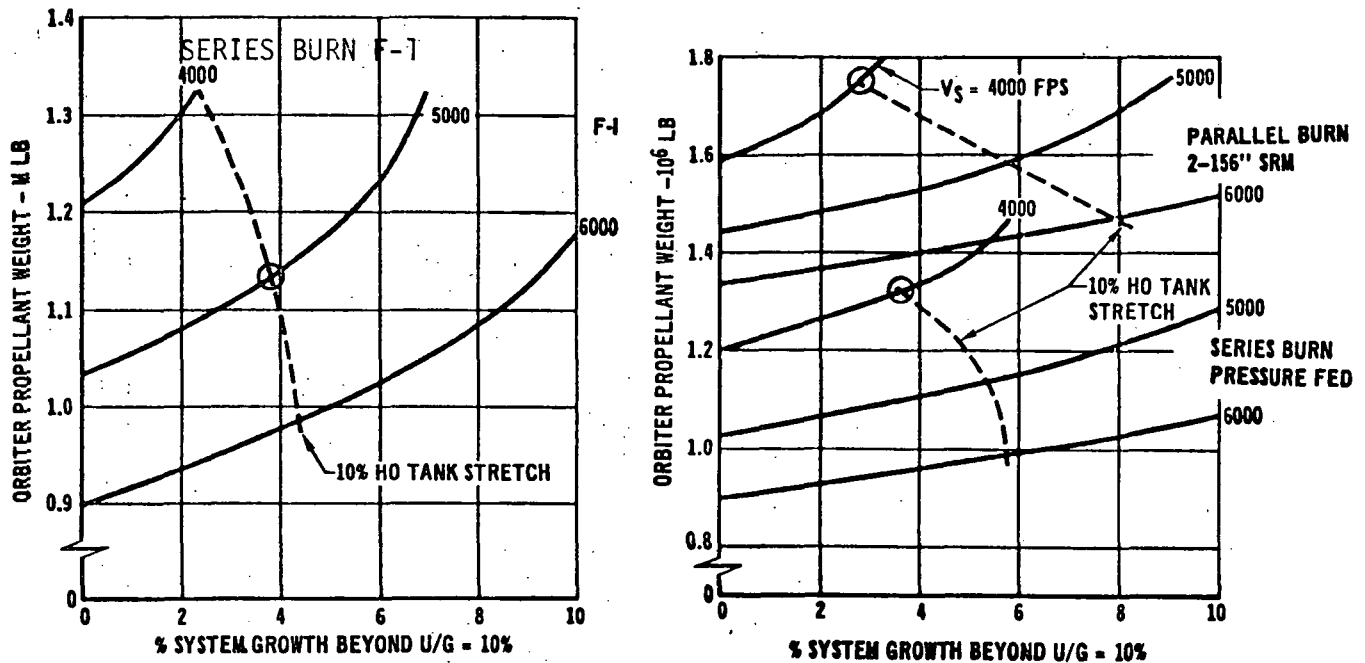


FIGURE 5-7

COMPARISON OF MAIN EMPHASIS SYSTEMS

	BURN MODE	SERIES		PARALLEL			
	ENGINE	PRESS. FED	F-1	2-156'' SRM	2-156'' SRM	4-120'' SRM	4-120'' SRM
	BAY SIZE	15 x 60	15 x 60	15 x 60	14 x 45	15 x 60	14 x 45
STAGING VELOCITY (FPS)		4000	5000	4000	4000	3692	4837
WEIGHTS (K LB)							
PAYLOAD (EAST)		65.0	65.0	65.0	45.0	65.0	45.0
GLOW		5777.6	4458.3	4315.2	3617.4	4662.4	4382.2
OLOW		1512.2	1341.7	1898.6	1691.2	1853.1	1572.4
USABLE PROPELLANT		1196.6	1031.2	1577.9	1407.7	1533.8	1292.3
ORBITER DRY		148.8	148.8	148.8	139.1	148.8	139.1
TANK DRY		58.1	53.5	64.8	60.3	63.5	57.2
BLOW		4265.4	3116.5	2416.5	1926.2	2809.8	2809.8
USABLE PROPELLANT		3436.9	2687.3	2 x 1064.5	2 x 848.5	4 x 607.4	4 x 607.4
BOOSTER DRY		734.9	372.3	(2 x 143.8)	(2 x 114.6)	(4 x 95.0)	(4 x 95.0)
(T/W) ₁ /(T/W) ₂		1.35/0.934	1.35/1.053	1.405/0.927	1.405/1.071	1.425/0.944	1.516/1.162
COSTS (\$M)							
TOTAL PROGRAM		10,798	10,002	10,207	9,681	11826	11628
RDT&E		5,676	5,018	4,126	4,006	4006	3906
PEAK FUNDING (YEAR)		1,263(76)	1,135 (76)	946 (76)	917 (76)	922(76)	887(76)
PER FLIGHT (TOTAL)		7.50	7.56	10.49	9.83	14.11	13.94
OPERATIONS		4.65	5.17	3.80	3.78	4.06	4.01
TANKS		1.37	1.28	1.49	1.41	1.47	1.35
EXPENDED BOOSTER HARDWARE		1.48	1.11	5.20	4.64	8.58	8.58

ALL CONFIGURATIONS HAVE 3-470 K HIPc ORBITER ENGINES

15 MARCH 1972

6. Space Shuttle Systems Descriptions

The configuration characteristics, ascent trajectories, and the performance and control requirements of the systems selected for main emphasis are discussed below.

6.1 Configuration Characteristics

The characteristics of two series burn and four parallel burn systems selected for detailed analysis are summarized in Figure 6.1-1. The parallel burn twin 156" solid rocket motor vehicles are the lightest and the smallest systems but require large HO tanks. Parallel burn system weights are based on expendable SRM's. The ascent navigation and guidance functions for both series and parallel burn configurations are controlled by the Orbiter.

Figures 6.1-2 through 6.1-7 illustrate each mated configuration.

FIGURE 6.1-1
CONFIGURATION COMPARISON

BOOSTER TYPE CARGO BAY SIZE	SERIES BURN		PARALLEL BURN			
	PRESSURE FED 15x60	F-1 15x60	156" TSRM 14x45	156" TSRM 15x60	(4) 120" SRM 14x45	(4) 120" SRM 15x60
MATED CONFIGURATION	TEL	TEL	SIDE BY SIDE	SIDE BY SIDE	SIDE BY SIDE	SIDE BY SIDE
GLOW (K LB)	5777	4458	3617	4315	4382	4662
LENGTH (FT)	303	258.6	150	155	150	160
EASTERLY PAYLOAD (K LB)	65	65	45	65	45	65
STAGING VELOCITY (FPS)	4000	5000	4000	4000	4837	3692
ORBITER						
OLW (K LB)	1512	1342	1691	1899	1572	1853
NO. ENG/THRUST PER ENG (K LB)	3/470	3/470	3/470	3/470	3/470	3/470
HO TANK						
TANK ARRANGEMENT	SEP BLKHD	SEP BLKHD	COMMON BLKHD	COMMON BLKHD	COMMON BLKHD	COMMON BLKHD
SIZE (FT)	26.2x140	23x136.5	29.5x126	28.5x142	28.3x129	28.3x140
DRY WEIGHT (K LB)	58.1	53.5	60.3	64.8	57.2	63.5
BOOSTER						
NO. ENG/SL THRUST PER ENG (K LB)	6/1300	4/1505	2/2068	2/2575	4/1397	4/1397
TVC	GIMBAL-6 DEG	GIMBAL-6 DEG	NONE	NONE	LITVC-2 DEG AV	LITVC-2 DEG AV
CANT	3 DEG	ZERO	15 DEG	15 DEG	6 DEG	6 DEG
TANK	COMMON BLKHD	COMMON BLKHD	SEGMENTED	6 SEGMENT	7 SEGMENT	7 SEGMENT
RECOVERY *	6 SPEED BRAKES	4 SPEED BRAKES	BALLUTE	BALLUTE	NONE	NONE
	6 RETRO ROCKETS	4 RETRO ROCKETS	SUSTAINER ROCKETS	2 SUSTAINER ROCKETS		
	IMPACT VEL:100 FPS	IMPACT VEL:35 FPS	IMPACT VEL:35 FPS	IMPACT VEL:35 FPS		
BLOW	4265	3117	1926	2417	2810	2810

* 156" TSRM SYSTEM WEIGHTS ARE BASED ON EXPENDABLE MOTORS. BLOW WILL INCREASE APPROXIMATELY 125 K LB FOR RECOVERABLE SYSTEM

15 MARCH 1972

FIGURE 6.1-2
SERIES-BURN PRESSURE-FED BOOSTER
15 x 60 Payload Bay

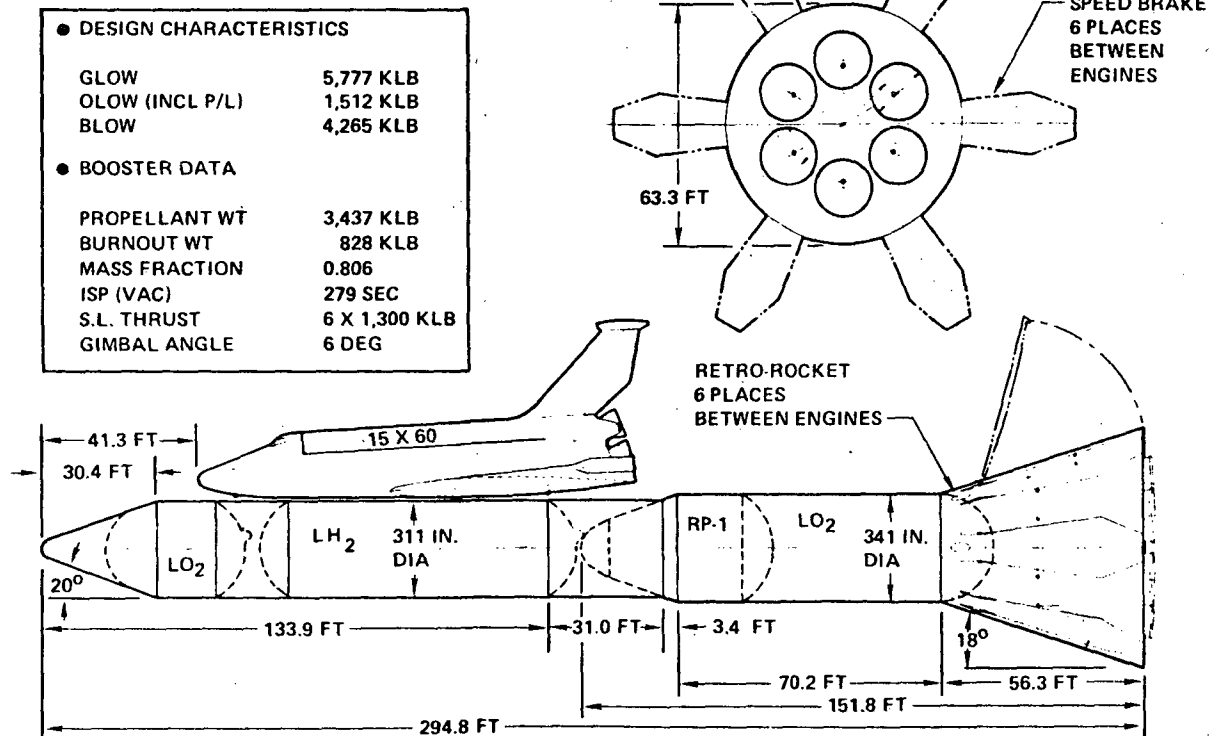
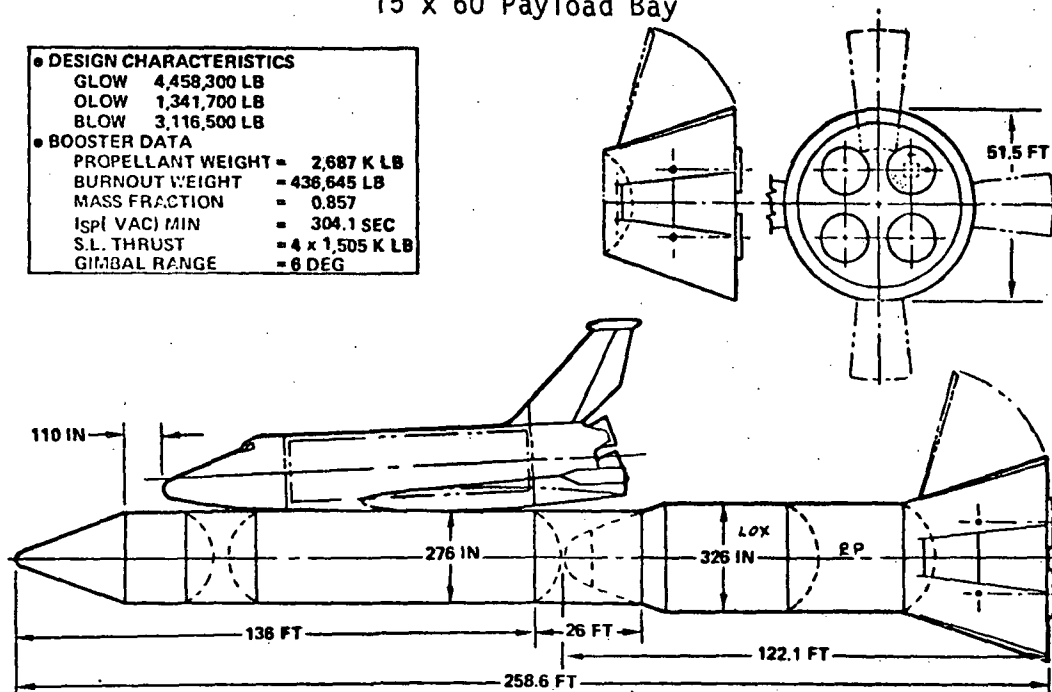


FIGURE 6.1-3

SERIES-BURN F-1 BOOSTER
15 x 60 Payload Bay

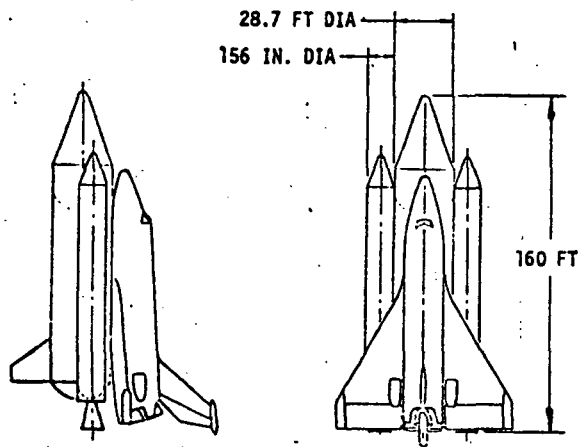


15 MARCH 1972

FIGURE 6.1-4

PARALLEL BURN TWIN 156" SRM BOOSTER

15 x 60 Payload Bay



• DESIGN CHARACTERISTICS

GLOW	4,315 K LB
OLW	1,899 K LB
BLOW	2,417 K LB

• BOOSTER DATA (PER SRM)

PROPELLANT WT	1,065 K LB
BURNOUT WT	144 K LB
MASS FRACTION	0.881
I _{sp} (VAC)	264 SEC
S.L. THRUST	2,575 K LB
GIMBAL ANGLE	NONE

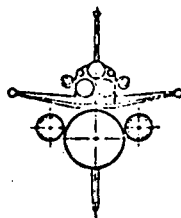
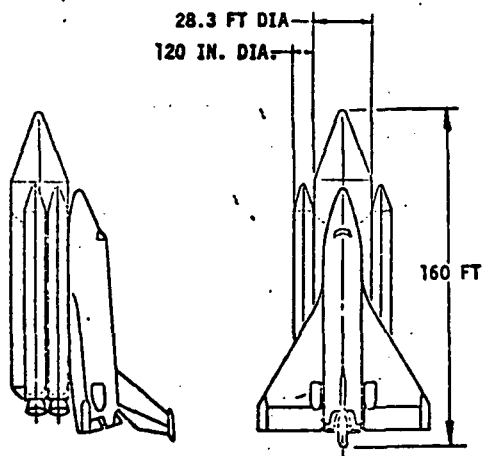


FIGURE 6.1-5

PARALLEL BURN 4-120" 7 SEGMENT SRM BOOSTER

15 X 60 Payload Bay



• DESIGN CHARACTERISTICS

GLOW	4663 K LB
OLW	1853 K LB
BLOW	2810 K LB

• BOOSTER DATA (PER SRM)

PROPELLANT WT	607 K LB
BURNOUT WT	95 K LB
MASS FRACTION	.8647
I _{sp} (VAC)	263 SEC
S.L. THRUST	1397 K LB
TVC-LIQUID INJECT	2 DEG Av

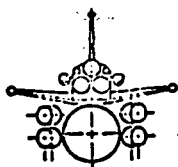
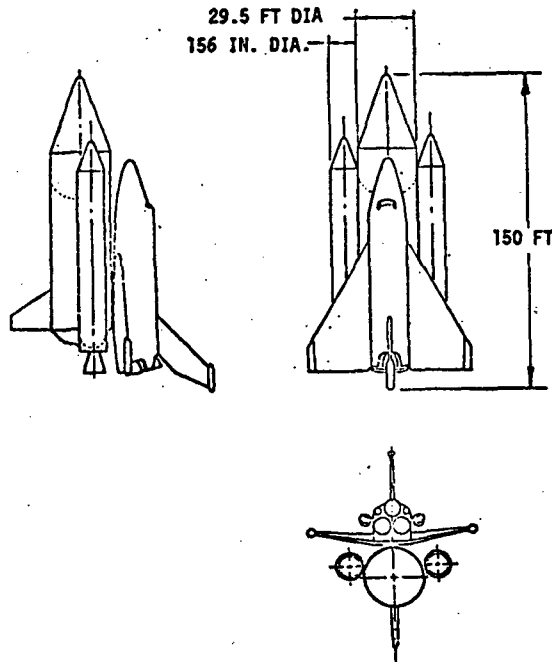


FIGURE 6.1-6
PARALLEL BURN TWIN 156" SRM BOOSTER
14 x 45 Payload Bay

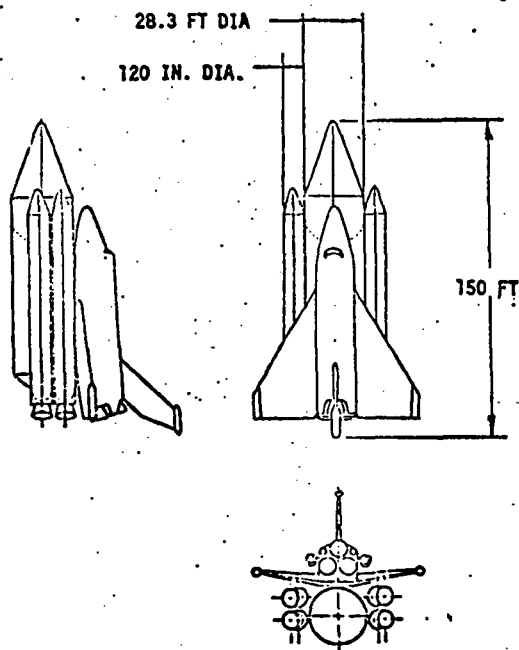


- DESIGN CHARACTERISTICS

GLOW	3617 K LB
OLW	1691 K LB
BLOW	1926 K LB
- BOOSTER DATA (PER SRM)

PROPELLANT WT	848 K LB
BURNOUT WT	115 K LB
MASS FRACTION	0.881
I _{sp} (VAC)	264 SEC
S.L. THRUST	2068 K LB
GIMBAL ANGLE	NONE

FIGURE 6.1-7
PARALLEL BURN 4-120" 7 SEGMENT SRM BOOSTER
14 x 45 Payload Bay



- DESIGN CHARACTERISTICS

GLOW	4382 K LB
OLW	1572 K LB
BLOW	2810 K LB
- BOOSTER DATA (PER SRM)

PROPELLANT WT	607 K LB
BURNOUT WT	95 K LB
MASS FRACTION	.8647
I _{sp} (VAC)	263 SEC
S.L. THRUST	1397 K LB
TVC-LIQUID INJECT	2 DEG Av

6.2 Ascent Trajectories

Extensive ascent trajectory analyses have been performed for the Shuttle system concepts examined during this study phase. These analyses provided performance verification of the sizing studies; insight into thrust shaping techniques required to satisfy q and g constraints; reference trajectories for control, separation, abort, and Booster entry analyses; and data for mission planning purposes.

A trajectory model has been used to simulate plane flight over a 3-D rotating earth. Aerodynamic characteristics used for the trimmed mated flight phases reflect wind tunnel test data where available; see Volume IV, Booster Aerodynamics Design Data Book. A three dimensional propulsion model has been used to simulate engine gimballing required to satisfy moment balance. During the COV (Calculus of Variation) optimization phase (following Booster Separation) a point mass simulation was used along with lift and drag forces.

6.2.1 Ascent Flight Sequence - Figures 6.2-1 and -2 show altitude as a function of downrange distance and relative velocity for the Parallel Burn Twin SRM 156" and Series Burn Pressure Fed Booster Configurations. The flight sequence is initiated at lift-off with Booster and Orbiter main propulsion systems operating. At 8 seconds the vertical flight phase terminates and a constant pitch-over rate is flown to $t = 20$ seconds. A trimmed gravity turn is flown for the balance of the mated flight. Max q (650 psf) occurs at approximately 70 seconds. The Booster thrust profile is tailored to satisfy the 3g maximum axial load factor constraint and max q constraint with only a modest performance penalty. For parallel burn, the Orbiter engine thrust is held constant (no throttling) to maintain adequate control authority. SRM case impact is approximately 96 nm downrange of the launch site, based on an atmospheric reentry simulation for a non-recoverable mode. The impact range of a recoverable SRM case (assuming a ballute plus retro rocket system) is within 5 nm of the non-recoverable case. The recoverable pressure fed series burn booster impacts 100 nm from the launch site.

Guidance for the balance of the Orbiter burn is based on an optimal thrust vectoring profile. The Orbiter engines are throttled to about 70% of nominal during the last 60 seconds of flight to maintain a 3g axial load factor. Orbit insertion (50 x 100 nm) occurs at 515.7 seconds after lift-off, with the Orbiter engines throttled to approximately 70% of the full thrust level.

15 MARCH 1972

6.2.2 Staging Conditions - Figure 6.2-3 summarizes staging conditions and Booster impact range for the Parallel Burn SRM, Series Burn Pump Fed F-1, and Series Burn Pressure Fed Configurations sized for a 15 x 60 ft payload bay and for the Parallel Burn SRM with a 14 x 45 ft payload bay. All configurations were sized for 4000 fps staging velocity, except for the series Burn Pump Fed F-1, which was sized for 5000 fps. Lift-off T/W is 1.35 for the Series Burn configurations and 1.40 for the Parallel Burn. A nominal 650 psf max q constraint was maintained in each case. The lower staging q and increased Booster impact range for the Series Pump Fed F-1 is consistent with the higher staging velocity.

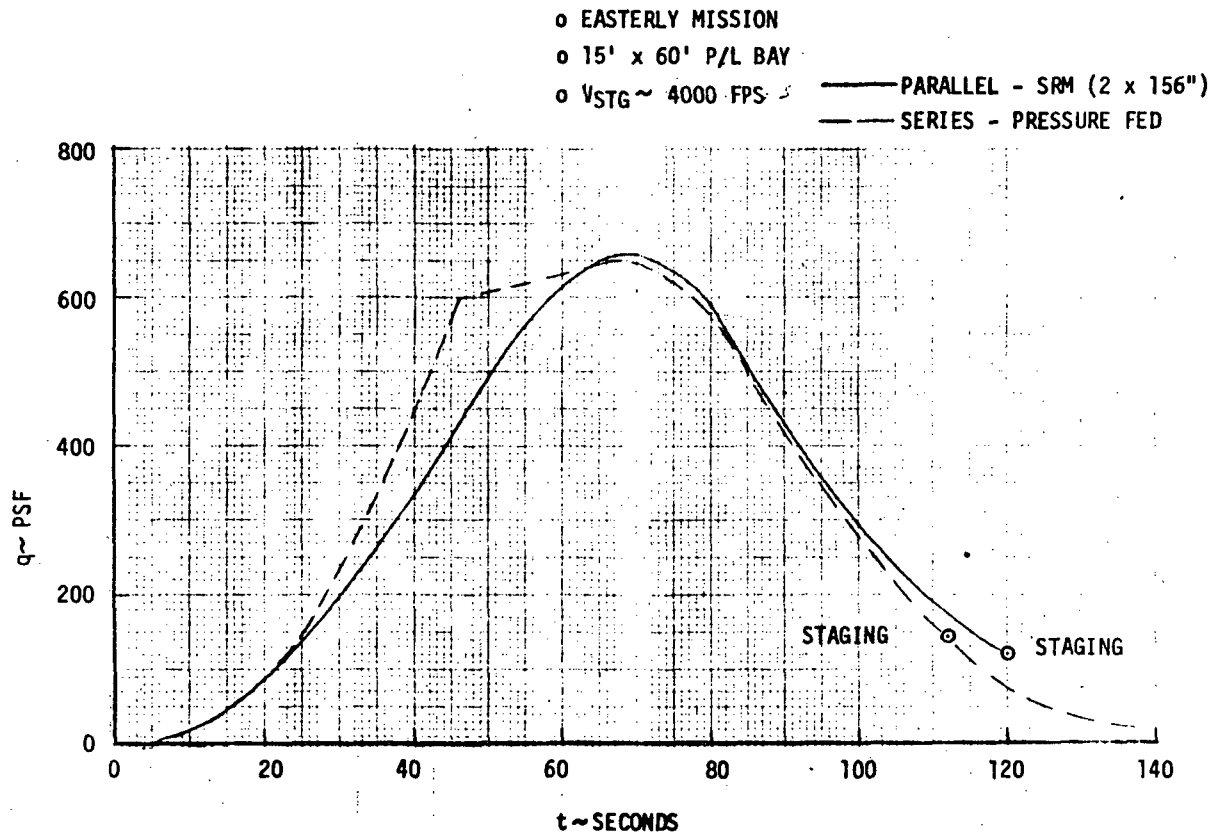
6.2.3 Max q Control - Sizing of the main contenders reflects the Booster thrust shaping required to satisfy the 650 psf max q constraint. The greatest latitude in thrust shaping is provided by the SRM Booster and the least by the Pump Fed F-1 Booster. Alternate max q control techniques examined for the F-1 Booster were (1) trajectory shaping without thrust shaping, (2) shutdown of one engine @ approximately 50 seconds and (3) two position orifice control of thrust level, which is probably the best approach.

A comparison of the SRM and Pressure Fed Booster thrust shaping techniques are illustrated in Figure 6.2-4. The resultant q-time histories are shown in Figure 6.2-5. A sharp rise on the front side of the q curve and an abrupt discontinuity associated with the 20% thrust reduction identifies the Pressure Fed Booster. Very little difference exists between the two time histories past max q. For optimal performance, staging q is not strongly driven by the thrust shaping technique. Staging velocity, however, can significantly affect staging q.

6.2.4 Staging Velocity - Because cost considerations tend to size the Shuttle with small Boosters and hence low staging velocities, it is of interest to examine the effect of staging velocity on key ascent trajectory parameters. Two parameters of particular interest are staging q and Booster downrange impact distance. A high staging q may impose high post-separation structural loads ($\alpha \cdot q$) on the Orbiter. Downrange impact distance affects range safety planning and recovery operations. Staging q and Booster downrange impact distance are plotted in Figure 6.2-6 for staging velocities varying from approximately 3,000 to 6,000 fps. Thrust or trajectory shaping can provide a substantial reduction in staging q if desired, but will be accompanied by a performance penalty.

15 MARCH 1972

FIGURE 6.2-5
ASCENT Q-HISTORIES



6.2.5 Payload Off-Load - The effect of launch without payload on ascent trajectory characteristics and subsequent systems impact has been examined by simulating a nominal 65K easterly launch trajectory with a zero payload trajectory. Results are compared in Figure 6.2-7.

It is desirable to avoid propellant off-loading, minimize impact on staging environment, and maintain nominal tank residuals at Orbiter burnout. The technique selected to satisfy these objectives was to throttle the Orbiter engines after staging to a level which ensures that all of the propellant loading is consumed. In the simulation, Orbiter engines were throttled to 2/3 nominal thrust level, causing burn time to increase approximately 190 seconds. This additional burn time must be examined in light of engine requirements, life time, etc. Additionally, extended operation at a 67% throttle position requires further evaluation.

Although the Series Burn Pressure Fed Concept was evaluated, the results are applicable for a Parallel Burn Configuration. Staging conditions and Booster impact range are not changed significantly.

FIGURE 6.2-6
EFFECTS OF STAGING VELOCITY

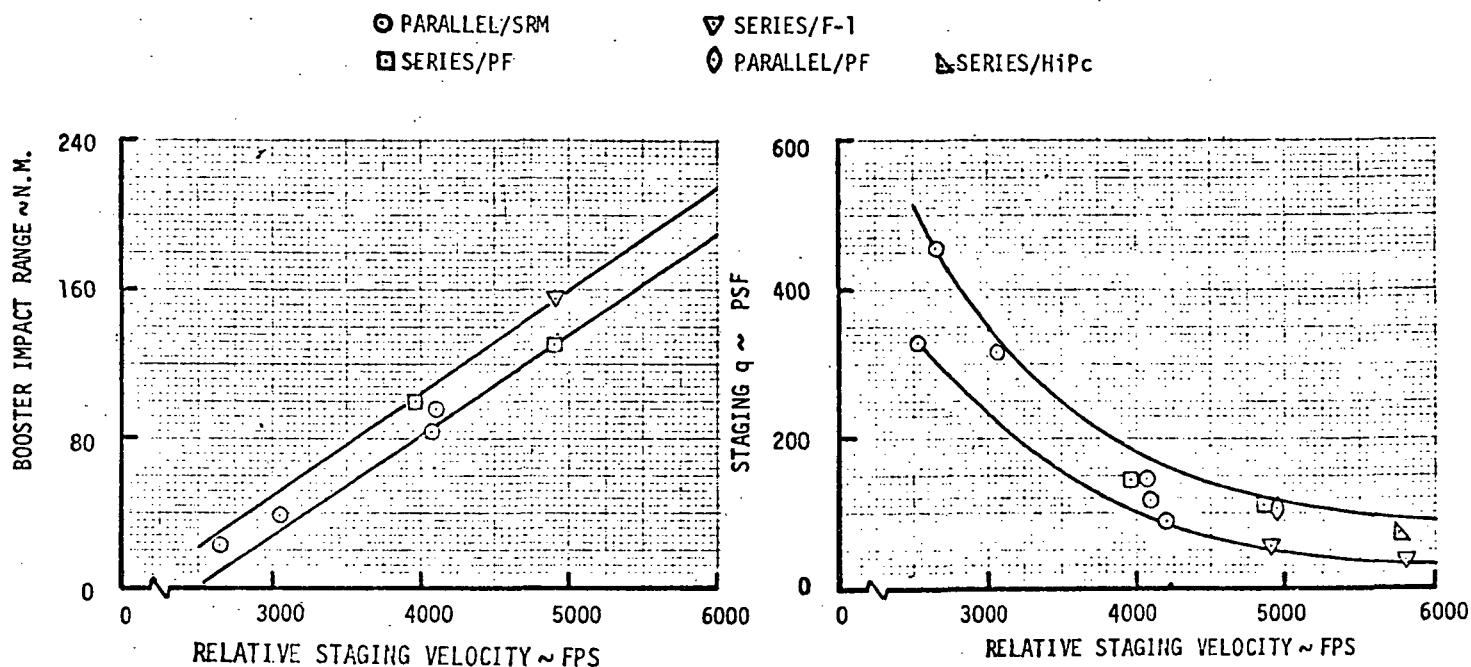


FIGURE 6.2-7 EFFECT OF PAYLOAD OFF-LOAD; SERIES BURN/PRESSURE FED

	65K PAYLOAD	0 PAYLOAD
STAGING		
h - FT	112,798	123,901
V _R - FPS	3,963	4,038
γ _R - DEG	32.2	37.1
q - PSF	143.	90.
BOOSTER IMPACT		
R ~ NM	100	108
ORBITER BURN		
t ~ SEC	505.6	696.7

15 MARCH 1972

6.3 Ascent Control

Response analyses of the ascent control systems have been performed for the parallel burn configuration with two 156" SRM's and for two series burn configurations; the pressure fed Booster and the F-1 pump fed Booster. Sketches of the three configurations analyzed are presented in Figures 6.3-1 through 6.3-3.

Control of the parallel burn system is derived from Orbiter thrust vector control (TVC), augmented by Orbiter aero surface control during the high dynamic pressure flight regime. This control authority was found to be adequate without resorting to the additional cost of TVC in the twin SRM Booster elements. Mated series burn control authority is provided by Booster TVC and Orbiter aero surfaces.

6.3.1 Configuration Characteristics - The pitch plane and yaw plane aero moment gradient required to control moment gradient ratios are presented in Figures 6.3-4 and 6.3-5. The parallel burn configuration is aerodynamically stable in pitch and yaw throughout burn. Both series burn configurations are aerodynamically unstable in pitch. The pump fed Booster configuration is also unstable in yaw while the pressure fed Booster configuration is essentially neutrally stable in yaw.

The roll control characteristics of the three configurations are compared in Figure 6.3-6. The roll control problem is more severe with the parallel burn configuration because the Orbiter TVC roll moment capability is less than that available from series burn Booster TVC.

The gimbal patterns for the series burn configurations are shown in Figures 6.3-2 and 6.3-3. In both cases the corner of the pattern (i.e., the point of maximum engine deflection) is located to give a thrust vector deflection perpendicular to a radius from the vehicle centerline. This maximizes the roll moment capability available for a given actuator deflection capability.

6.3.2 Control System Description - For the parallel burn configuration, ascent control is obtained by gimbaling the three Orbiter main propulsion system engines to provide pitch, yaw and roll control moments. In the high dynamic pressure region, roll control moments are also provided by the Orbiter rudder and aileron.

Two modes of roll control have been investigated: roll attitude control throughout the mission (within the control moment capability of the vehicle) and free roll (i.e., no roll attitude control in the high dynamic pressure region). the case for the roll attitude control mode, a 200 sq ft ventral fin located at the aft end of the H0 tank was required to obtain satisfactory system response characteristics.

For the first 30 seconds and again from about 80 seconds after liftoff until Booster burnout, roll control is obtained by differential deflection of the bottom two engines in pitch and the top and bottom engines in yaw. Between 30 and 80

ASCENT CONTROL CONFIGURATIONS

FIGURE 6.3-1

PARALLEL BURN TWIN 156" SRM BOOSTER

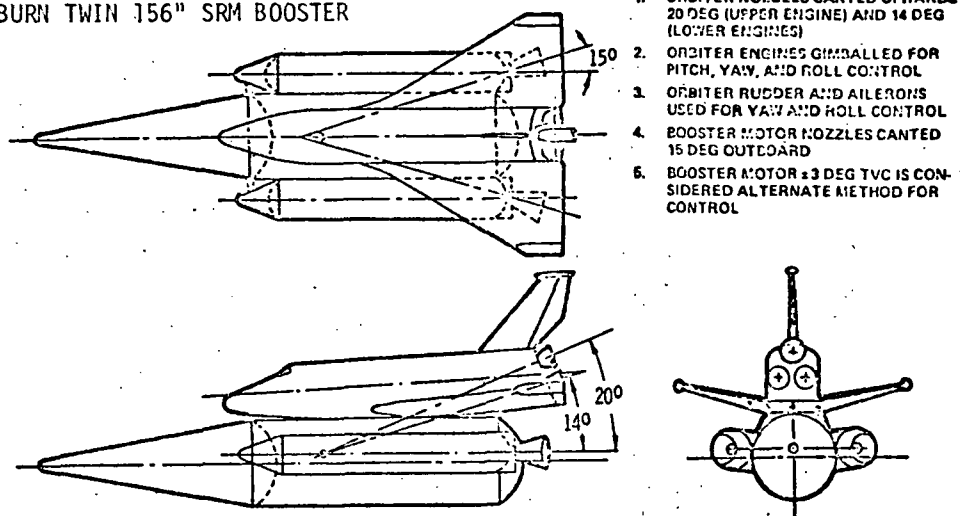


FIGURE 6.3-2

SERIES BURN PRESSURE - FED BOOSTER

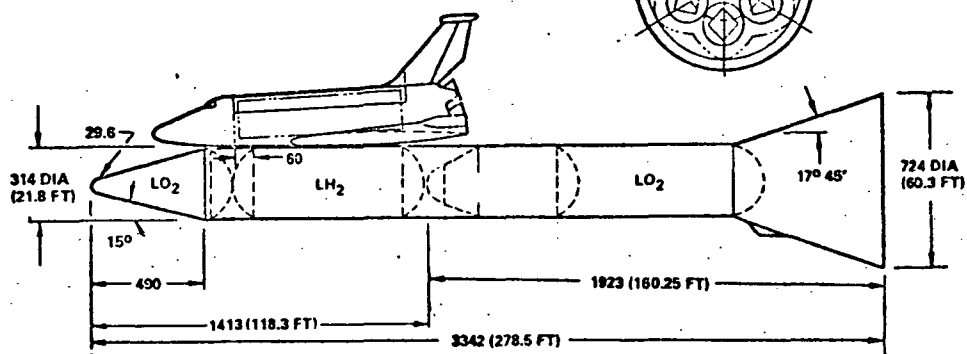


FIGURE 6.3-3

SERIES BURN F-1 BOOSTER

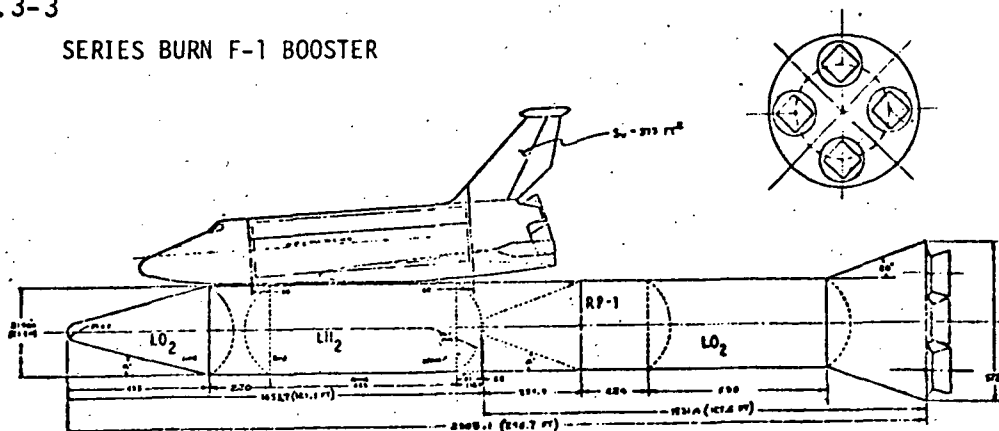


FIGURE 6.3-4

PITCH AERO TO CONTROL MOMENT GRADIENT RATIO

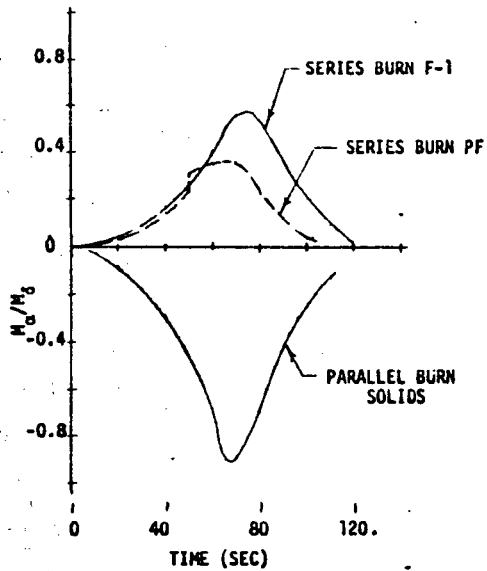


FIGURE 6.3-5

YAW AERO TO CONTROL MOMENT GRADIENT RATIO

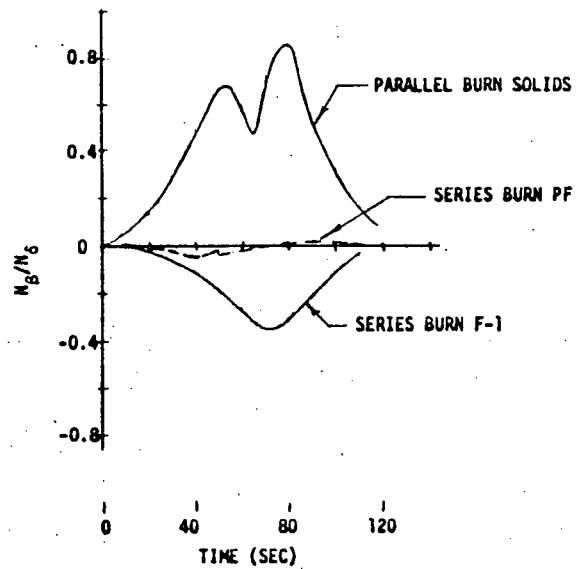
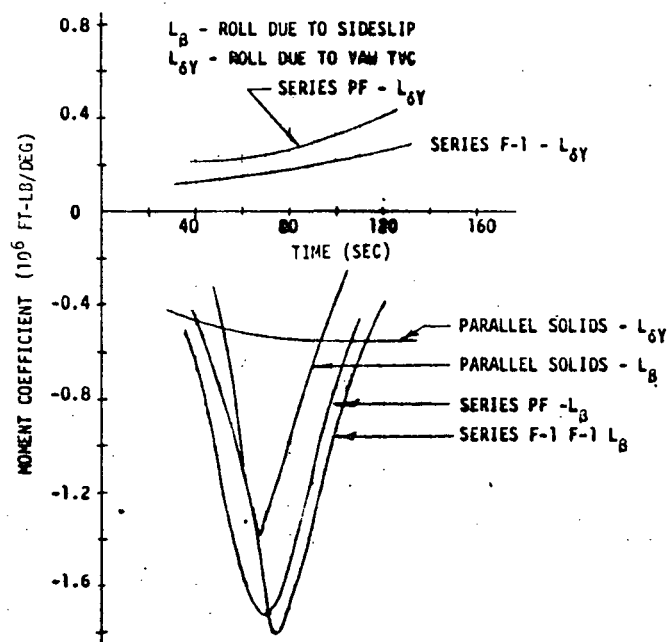


FIGURE 6.3-6

ROLL MOMENT COEFFICIENT COMPARISON
PARALLEL BURN VERSUS SERIES BURN



15 MARCH 1972

seconds after liftoff roll control is maintained with the Orbiter rudder and aileron, within the hinge moment capabilities imposed by reentry control requirements.

The free roll configuration is identical to that described above except that roll attitude error is not used on the rudder or aileron. However, both the rudder and aileron are required for roll rate control in the high dynamic pressure region. It should be noted that use of a roll attitude error limit would be a more practical method of obtaining free roll (as compared to zero gains). Use of the limit would result in free roll for severe winds (i.e., large roll disturbances would saturate the attitude loop) while roll attitude control could be maintained for less severe winds.

Both series burn configurations use Booster TVC as the prime source of ascent control. It was found that the actuator deflection requirement could be reduced significantly by supplementing roll control with the Orbiter aileron and rudder. Because of this both Orbiter aero control surfaces were used for roll control during ascent. The free roll mode is also a feasible solution to any roll control problems associated with the series burn configurations but has not been investigated except for the parallel burn, as discussed above.

The baseline ascent control scheme for pitch and yaw is drift minimum control. This control law can be achieved with attitude-only feedback for the parallel burn configuration because of the stable airframe. In the case of the two series burn configurations either an angle of attack (or sideslip) or a normal acceleration feedback signal is required to achieve drift minimum control. It should be noted that the drift minimum control law was subsequently modified depending on the requirements of the individual configuration. For example, load relief control in yaw was required to reduce the roll control requirements for both series burn configurations.

6.3.3 Control Moment Requirements

6.3.3.1 Parallel Burn Configuration - Thrust unbalance between the SRM's and unequal thrust tailoff characteristics dictate a requirement for a yaw cant angle on the SRM thrust vectors. The trade off between Orbiter gimbal angle required to trim in the presence of thrust unbalance and Booster I_{sp} loss is shown in Figure 6.3-7, as a function of SRM yaw cant angle. [REDACTED]

15 MARCH 1972

The Orbiter pitch and yaw actuator deflections required to trim in the presence of SRM misalignment, thrust eccentricity, unbalance, etc., are summarized in Figure 6.3-8. The data presented includes the roll component and the roll due to yaw gimbal angle. The most severe requirement is that for pitch actuator deflection of the bottom engine.

The total pitch gimbal angle requirement as a function of burn time is presented in Figure 6.3-9. The requirement includes the no wind trim for zero net lift, the off-nominal thrust and CG effects (from Figure 6.3-8), and the wind requirement. An allowance of one degree for attitude maneuvering is also included at lift-off and Booster burnout. [REDACTED]

[REDACTED] This saturation is not severe, however, and will not impact trajectory dispersions significantly.

[REDACTED] Note that since the ventral fin is located at the aft end of the H0 tank, it increases yaw plane stability as the roll moment is decreased.

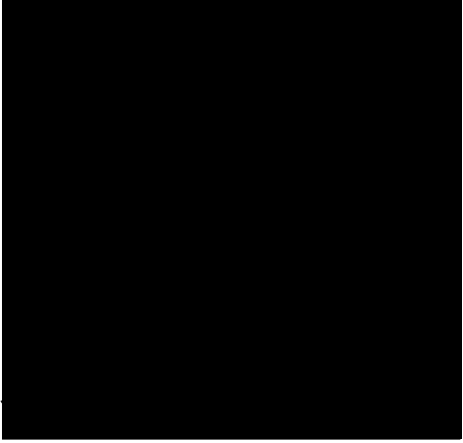
As can be seen in the figure the control requirement is greater than the control capability. [REDACTED]

[REDACTED] Under this condition control system saturation will occur but, as was the case in pitch, the resulting trajectory dispersions do not degrade performance significantly.

6.3.3.2 Series Burn Pressure-Fed Configuration - Since the Booster thrust vector is gimballed, the misalignments, unbalances, etc., are not as critical and have been considered as part of the actuator deflection margin requirement (30 percent on top of the wind gust requirement).

The pitch gimbal angle requirement due to head and tail winds is shown in Figure 6.3-11 for a series of wind gust altitudes. Pitch TVC gimbal angle requirements due to winds are within ± 3.75 degrees. TVC trim angle for a nominal no wind zero lift trajectory is also shown. A Booster precant angle is not used to reduce the trim angle since it produces no significant decrease in the TVC gimbal angle

PITCH TVC ANGLE (DEG)



ACTUATOR DEFLECTIONS (DEG)



envelope.

The pitch control requirement is presented in terms of gimbal angle which represents an average thrust vector deflection in the pitch plane. The actuator pattern (see Figure 6.3-2) is such that the pitch requirement will be distributed among all the actuators and, in terms of actuator deflection, will be slightly less (by cosine of 15 deg) than the gimbal angle requirement.

Since the configuration is only slightly unstable in the pitch plane, the control requirement can be maintained within a 6 degree actuator capability without adding fins. Also this requirement can be maintained with a drift minimum control law.

The yaw/roll control requirement in terms of the worst actuator (i.e., maximum deflection) is presented in Figure 6.3-12 for a series of side wind gust inputs. The actuator deflection requirement is based on control frequencies of 0.15 and 0.3 cps in yaw and roll respectively and a 40 percent load relief scheme (i.e., sideslip feedback gain increased 40 percent above drift minimum). Use of the Orbiter aerodynamic control surfaces reduces the Booster TVC actuator deflection requirements approximately 1.2 degrees.

The use of a 40 percent load relief control scheme reduces the actuator deflection requirements from 8 to 5.4 degrees but imposes a penalty of 70 feet per second on the Flight Performance Reserve (FPR). Additional reduction in actuator

15 MARCH 1972

deflection requirements is also possible, as shown in Figure 6.3-12. For example, 60 percent load relief reduces the maximum roll/yaw actuator deflection to 4.5 degrees but results in an additional 70 feet per second FPR penalty.

These results indicate that an actuator deflection requirement of ± 6.0 degrees can be achieved at only a nominal FPR penalty.

[REDACTED]

[REDACTED]

[REDACTED]

[REDACTED]

[REDACTED]

6.3.3.3 Series Burn F-1 Configuration - Actuator deflection requirements were determined for the F-1 configuration based on the vehicle response at the worst gust altitudes (30,000 and 35,000 feet). Since the F-1 engine is presently capable of a gimbal angle of only ± 6 degrees analyses were directed at defining a control scheme or aerodynamic modifications that would achieve this requirement.

The TVC actuator deflection requirements in a 95 percent wind speed profile environment were determined, using a closed loop control frequency of 0.15 cps and a damping ratio of 0.3. With drift minimum gains, the maximum pitch actuator deflection requirement was reduced to 5.0 degrees by increasing the angle of attack feedback by 20 percent while maintaining the closed loop control frequency at 0.15 cps. This is the maximum load relief possible without introducing positive feedback in the attitude loop. When the 30% margin for misalignments, unbalances, etc. is included, the overall gimbal angle requirement exceeds 6.0 degrees. Flight performance reserve penalties associated with the 20 percent increase are negligible.

[REDACTED]

[REDACTED]

[REDACTED]

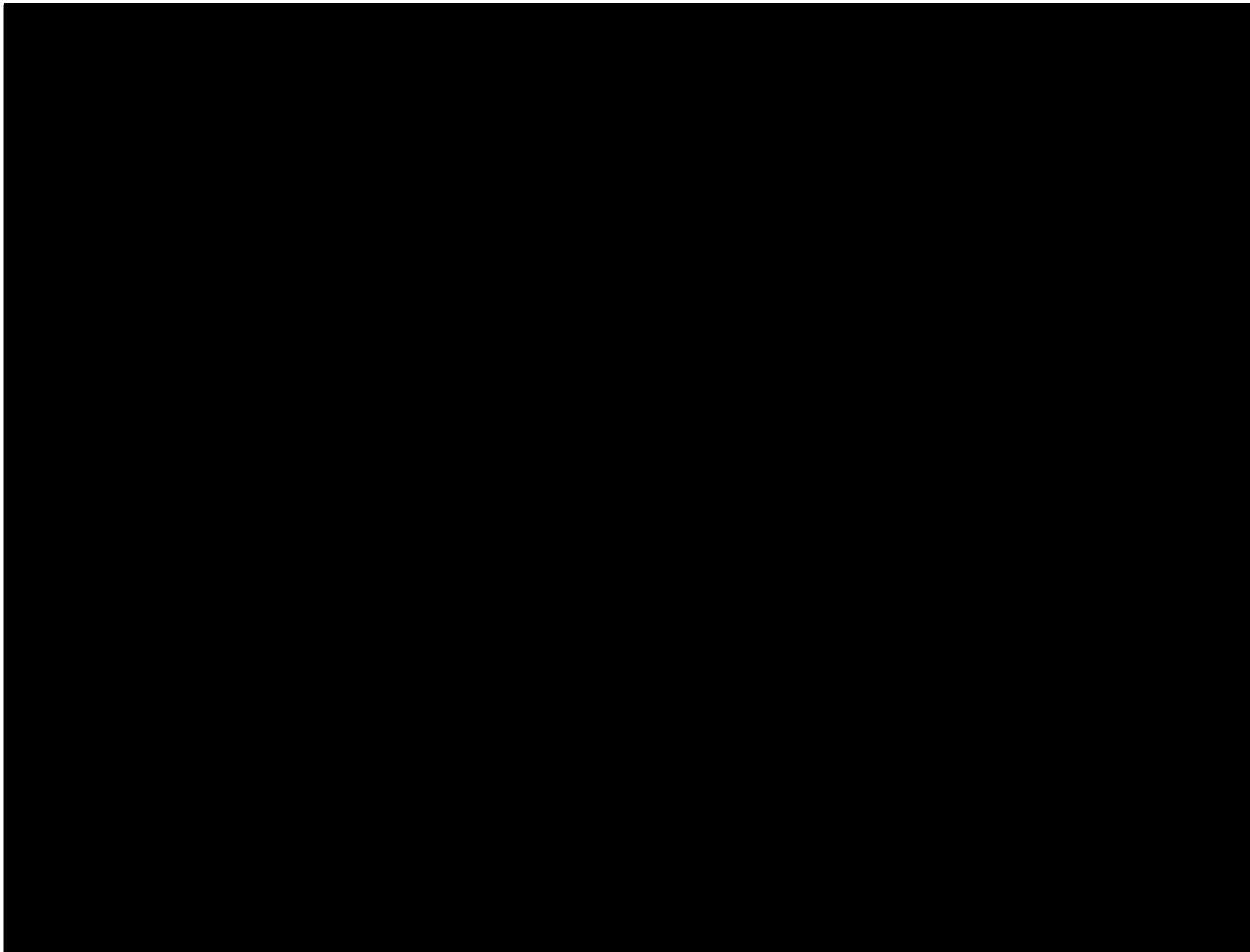
The use of fins to move the AC ten feet aft results in an actuator deflection requirement of 4.5 degrees. Even with the addition of misalignment, cg offsets, and the other error sources the actuator deflection requirement should remain within the ± 6.0 degrees capability of the F-1 engine when fins are incorporated.

The TVC actuator deflection requirements are the most severe in yaw/roll primarily due to the high aerodynamic moment required to control moment gradient ratio in the roll axis. Both differential gimbaling of the Booster engine and the Orbiter control surfaces were used for roll control. The yaw/roll actuator

15 MARCH 1972

deflections are based on the current engine location and actuator geometry, which allows maximum engine travel perpendicular to the vehicle radius to result in maximum roll control moment capability.

The sensitivities of TVC gimbal requirement to the closed loop frequency of the yaw control system are shown in Figure 6.3-13 for drift minimum gains. The closed loop roll control frequency was held constant at 0.3 cps. The figure shows high sensitivity to yaw frequencies below 0.15 cps. A minimum control frequency of 0.2 cps will probably be required to maintain loading and actuator deflection requirements to reasonable values. Note, however, that the higher control frequency may introduce higher frequency stability problems. Stability problems associated with yaw/roll coupling are also increased as the natural frequencies of the two axes come closer together.



15 MARCH 1972

The significant reduction in aerodynamic loads and associated TVC actuator deflection that results from increasing the angle of sideslip feedback by 20 percent is also shown in Figure 6.3-13.

Figure 6.3-14 shows the sensitivity of directional AC shift (aft) and roll moment coefficient to TVC actuator deflection requirements. Significant reduction in the roll moment coefficient is required to reduce the actuator deflection requirement to within the 6 degree engine capability.

The present F-1 configuration results in high TVC actuator deflection requirements on the order of 7.5 to 8.0 degrees. Reduction of the requirement to within the 6 degree capability of the F-1 engine will require changes in the vehicle aerodynamic characteristics through the use of fins or other devices. Although elimination of roll attitude control (free roll) is considered a feasible method of reducing the gimbale angle requirement, in this case fins are still required for the pitch axis, and the free roll approach is not recommended.

6.3.4 Dynamic Response to Winds - Detailed dynamic response results have been obtained for the parallel burn solids and the series burn pressure-fed. As mentioned previously, series burn F-1 configuration was not evaluated for all wind gust altitudes (the F-1 analysis concentrated on gimbale angle requirement with gusts of 30,000 and 35,000 feet).

6.3.4.1 Parallel Burn Configuration - The peak αq product profile is presented in Figure 6.3-15 as a function of Mach number for wind gust altitude of from 10,000 to 50,000 feet. Data is also presented for several misalignment and engine out conditions for both the 10,000 feet and the 30,000 feet gust altitude. Nominally the maximum αq reaches a value of about 7400 deg-lb/ft². Under the most severe off-nominal conditions the αq reaches a value of 8500 deg-lb/ft². Passive load relief control (i.e., no angle of attack or acceleration feedback) was investigated and it was found that approximately a 20 percent reduction in αq could be achieved. In order to achieve a more significant reduction, active load relief will be required. This technique will be investigated if the high αq imposes significant structural weight penalties. This is not expected since recent wind tunnel tests indicate that the Orbiter lift is reduced 50 percent due to the interference effects of the Booster.

The peak βq profile in response to side wind gust altitudes of 10,000 feet through 40,000 feet is presented in Figure 6.3-16. The data is given for a

15 MARCH 1972

condition with roll attitude control and with a [REDACTED]. As was the case for pitch, the βq profile shown is based on drift minimum yaw attitude gains.

Both passive and active load relief were investigated in order to reduce the maximum βq . Passive load relief did not result in a significant reduction in βq . A βq reduction to 3000 deg-lb/ft² was achieved using active angle-of-sideslip feedback. Further loads analyses are required to better define the impact of the high βq associated with the drift minimum control scheme before the need for active load relief can be established.

Roll system response under conditions of free roll showed a significant reduction in Orbiter aero surface requirements and eliminated the need for the 200 ft² ventral fin. Actual load relief in yaw due to the vehicle rolling to a headwind condition cannot be guaranteed because of misalignments which may counter-act roll due to winds.

6.3.4.2 Series Burn Configurations - The maximum αq profile for the series burn pressure fed configuration is presented in Figure 6.3-17 for a series of headwind and tailwind gust altitudes. The data is presented for drift minimum control gains and shows a maximum of 5500 deg-lb/ft². It is felt that significant reduction in αq can be achieved using load relief control if structural loads analyses shows a weight penalty associated with drift minimum control. [REDACTED]

[REDACTED]

[REDACTED]

The peak βq profile is presented in Figure 7.3-18 for the series pressure fed configuration. [REDACTED]

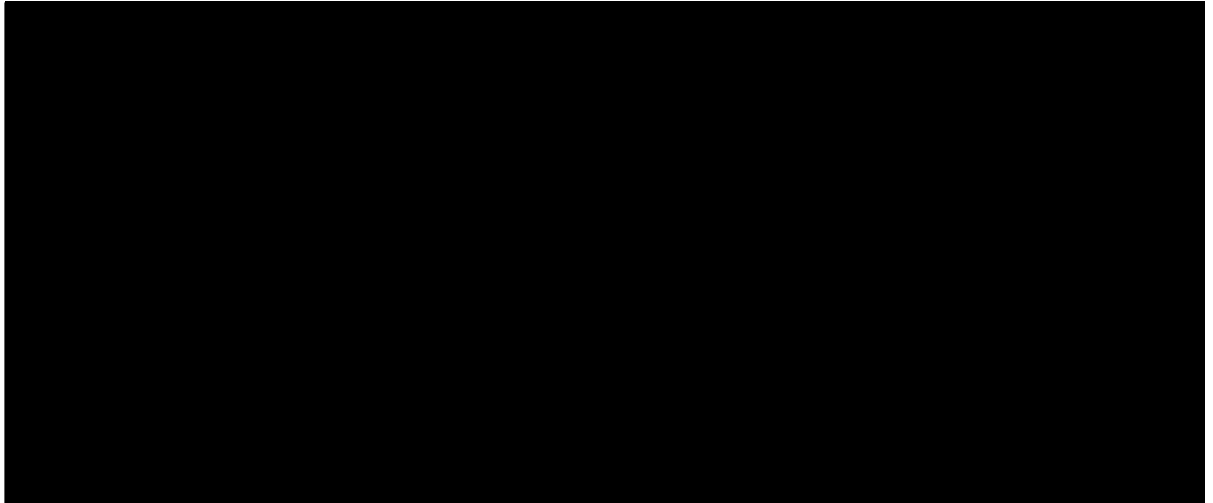
[REDACTED]

[REDACTED]

6.3.5 Conclusion - A summary comparison of the parallel burn configuration versus the series burn configuration is presented in Figure 6.3-19 and a detailed comparison of the significant control parameters is presented in Figure 6.3-20.

The most significant conclusion is that the control authority associated with the parallel burn configuration, though adequate, is limited compared to that for series burn. (It should be noted that the use of Booster TVC would eliminate this shortcoming, but at a significant increase in program cost.) The impact of the limited control authority is most significant near liftoff and imposes severe

FIGURE 6.3-20



tolerances on SRM misalignment, SRM thrust unbalance and vehicle CG offset. Since the parallel burn configuration is aerodynamically stable, loss of control authority in the high dynamic pressure region is not significant and results only in small FPR penalties.

6.4 Separation

At Booster propellant depletion separation of the two SRM Boosters (or one liquid series Booster) is performed. In parallel burn concept, the SRM's are separated when their thrust has dropped to the point where the Booster will fall behind the Orbiter and not recontact it. For the series burn Booster concept the staging sequence includes Orbiter engine startup and Booster engine shutdown.

6.4.1 Parallel Burn Configuration - The parallel burn SRM Boosters are separated with an aft hinge mechanism which permits each Booster to rotate out from under the Orbiter wings following pyrotechnic release of the forward attach point.

The sequence of events for nominal separation of the parallel burn Booster concept is shown in Figure 6.4-1. Booster tail-off is sensed by total vehicle acceleration level, and the signal is given to release the forward attachment points. When the Boosters have rotated outward forty degrees, as measured at the aft hinge they are released. The Boosters then continue their swing out from under the wings.

The relative displacement history of the separation maneuver is shown in Figure 6.4-2. The Boosters are shown to clear the wing at about .6 seconds after

FIGURE 6.4-1

NOMINAL SEPARATION SEQUENCE
PARALLEL BURN TWIN SRM (156")

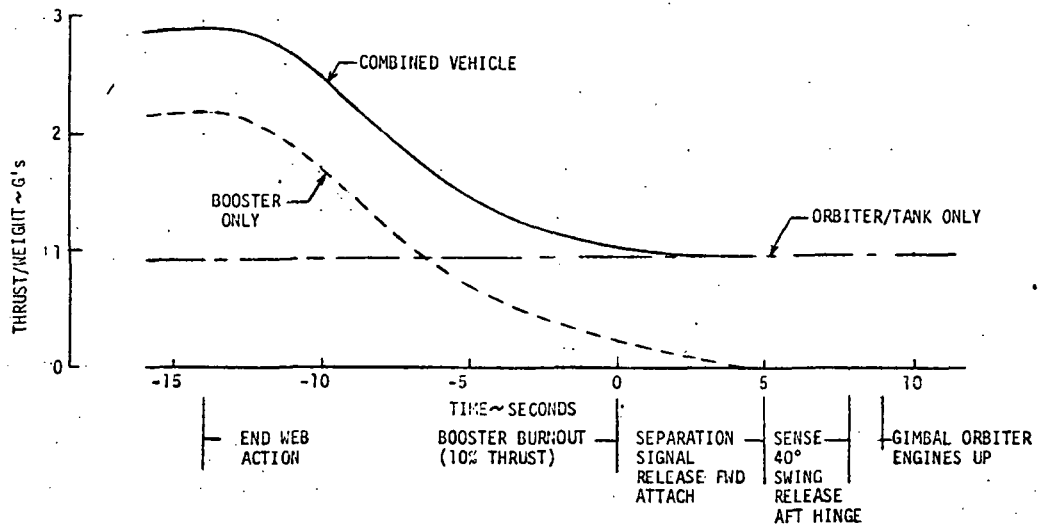
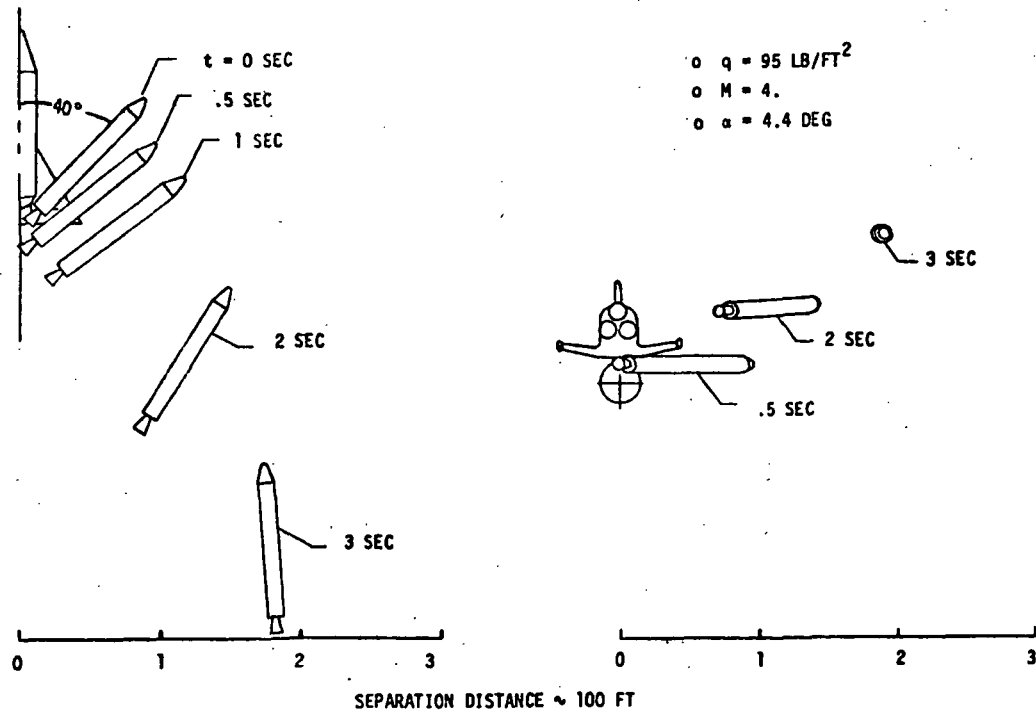


FIGURE 6.4-2

NOMINAL STAGING SEPARATION CHARACTERISTICS
AFT HINGE CONCEPT



15 MARCH 1972

release. The aerodynamic interference effects are such that the Booster is stable in yaw; when released its yaw angle tends to reduce with respect to the tank centerline. This yaw stability is shown to enhance Booster-wing clearance.

6.4.2 Series Burn Configuration - The nominal separation of a series burn Booster includes separation of two launch vehicle components: first, the Booster and secondly the interstage. Pyrotechnic separation devices are employed at both the Booster/interstage and interstage/tank interfaces.

The sequence of events for nominal separation is shown in Figure 6.4-3. [REDACTED]

[REDACTED]

[REDACTED]

[REDACTED] To insure separation clearance between the Booster and the Orbiter tail, the Orbiter engines are gimballed so that the thrust vector points through the Orbiter plus tank center of percussion with respect to the Orbiter tail. The Orbiter plus tank then "pitches" away from the nose of the Booster for clean separation. To minimize Booster rates at separation the Booster engines are nulled

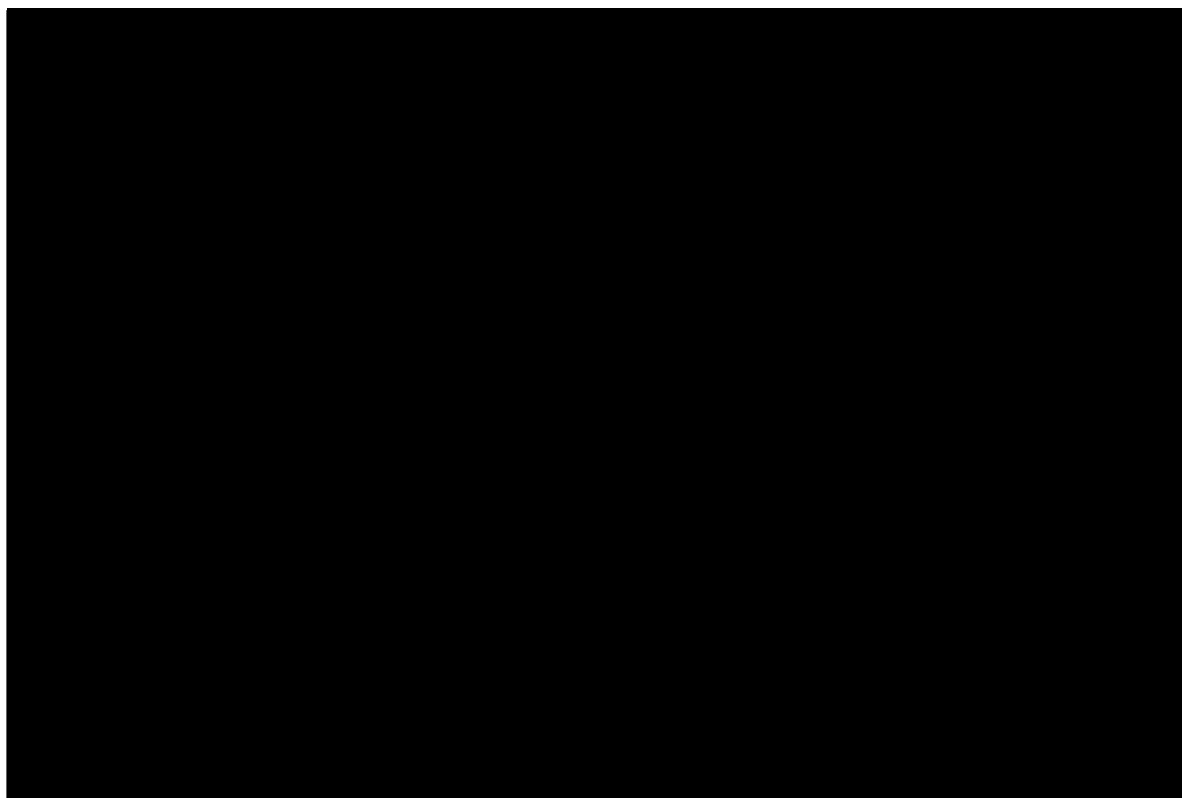


FIGURE 6.4-3

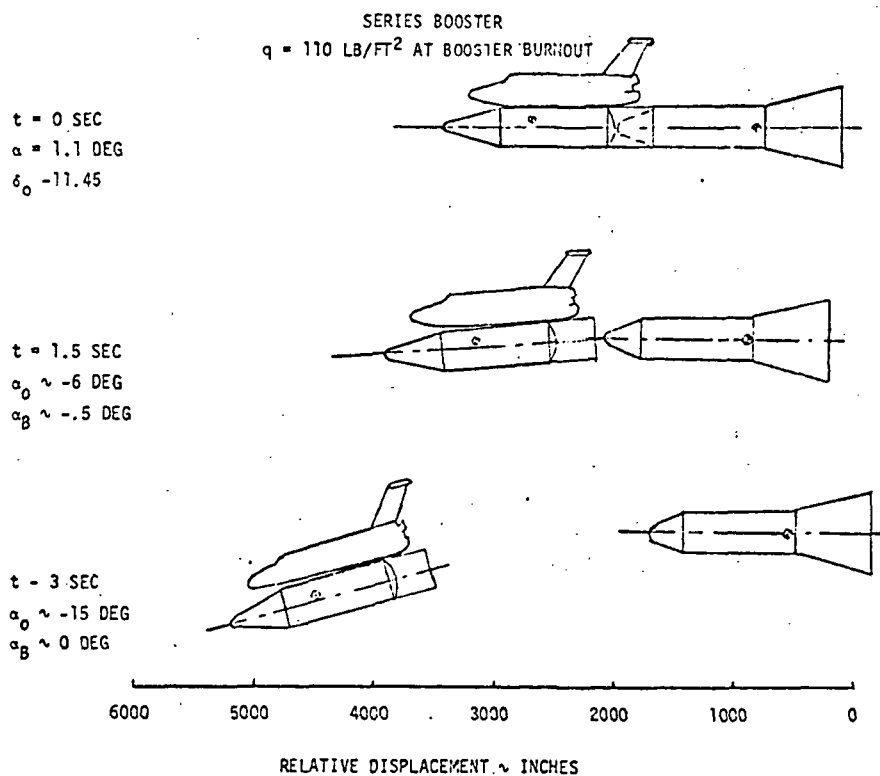
15 MARCH 1972

at cutoff. The Booster/tank interstage separation signal is given when the Orbiter plus tank acceleration exceeds the combined vehicle acceleration.

The relative displacement history of the separation maneuver is shown in Figure 6.4-4. Aerodynamic, control system and Orbiter plume impingement characteristics were included in the separation analysis. The time history shows that the Orbiter plus tank pitches down to -15 degrees angle of attack after three seconds. At this time however, the negative pitch rate has been arrested and the Orbiter plus tank can be pitched up to the desired angle of attack. The Booster pitch rate after 3 seconds is only -0.2 deg/sec and it is near its zero angle of attack trim point so that it can be captured on entry.

FIGURE 6.4-4

NOMINAL SEPARATION RELATIVE DISPLACEMENT HISTORY



15 MARCH 1972

7.0 Abort Analysis

Abort analyses have concentrated on intact recovery of the Orbiter following an abort during the ascent mission phase. The abort forcing event was assumed to be failure of the Booster or Orbiter main engine. However, aborts for other causes requiring immediate separation during the mated portion of the ascent phase were also considered. Methods for eliminating abort gaps have been determined.

The technique to be used depends on conditions at the time of abort as shown in Figure 7-1. These techniques are as follows:

- Mode 0 - A solid propellant abort rocket accelerates the Orbiter away from the HO tank and Booster; the Orbiter glides to a landing strip.
- Mode 1 - The Orbiter separates from the HO tank and the Booster and glides to the landing strip.
- Mode 2 - The Orbiter and HO tank separate from the Booster and maneuver toward the launch site; the HO tank is jettisoned and the Orbiter glides to landing.
- Mode 3 - Following a main engine failure after normal staging, the Orbiter uses the OMS system to supplement the remaining main engines and injects into a once-around trajectory, returning to the launch site.
- Mode 4 - Following a main engine failure after normal staging, the Orbiter injects into a 50 x 100 NM orbit with enough OMS capability remaining to perform some mission functions before deorbit. (DEGRADED MISSION)
- Mode 5 - (PARALLEL BURN ONLY) - Following the failure of an Orbiter main engine during mated ascent, the Booster is maneuvered to a point where the Orbiter can separate and perform a Mode 1 or 2 abort.

In each mode it is necessary to ensure that separation will occur, that the Orbiter can be flown to a satisfactory landing site, and that it will be in a suitable landing condition. Although the solid rocket system for Mode 0 and the separation system used for separation at high dynamic pressure in Mode 1 were not incorporated in the Orbiter baseline nor reflected in the Orbiter weights presented in Section 6, the characteristics of these systems are presented in the following section.

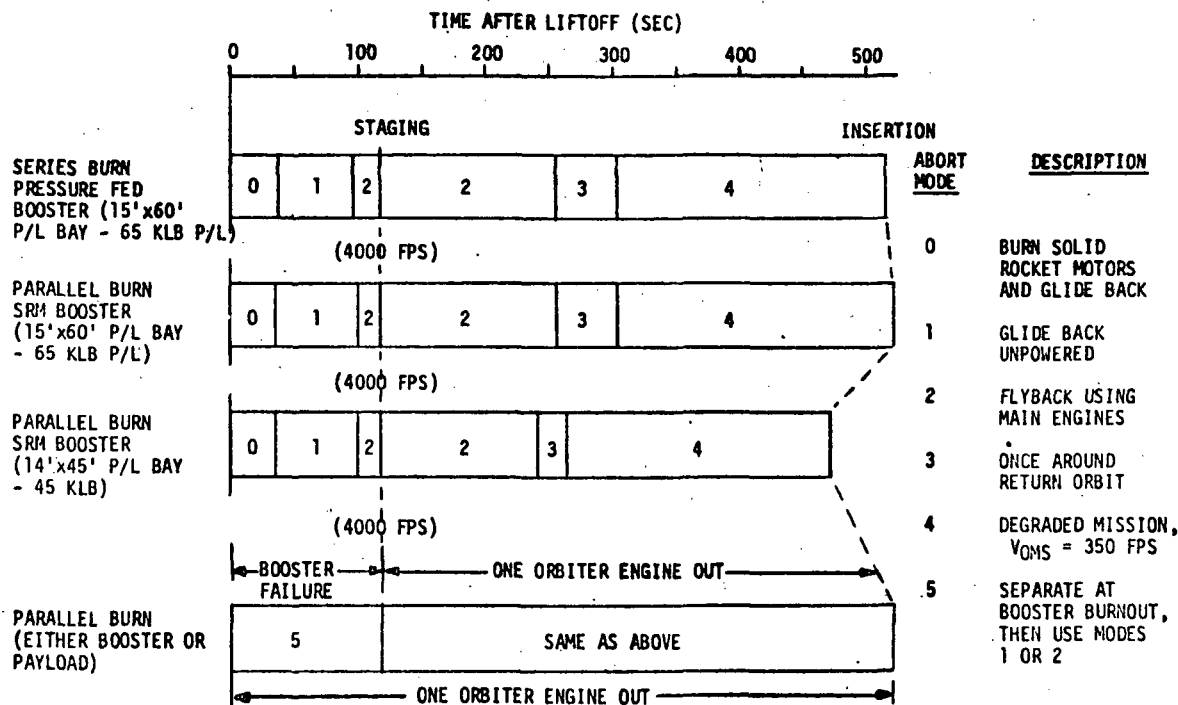
7.1 Abort Modes

The design and operational requirements for an abort capability are imposed primarily by the on-pad and ascent mission phases. An abort from orbit will involve near nominal maneuvers and performance. The abort modes and regions of application

15 MARCH 1972

FIGURE 7-1
ORBITER ABORT CAPABILITY SUMMARY

- o 3 H1Pc ENGINES - 1410 K LB TOTAL THRUST
- o EASTERLY MISSION
- o 105-109%



for the Parallel and Series Burn System are similar due to the similarity of the nominal ascent trajectories and acceleration time histories.

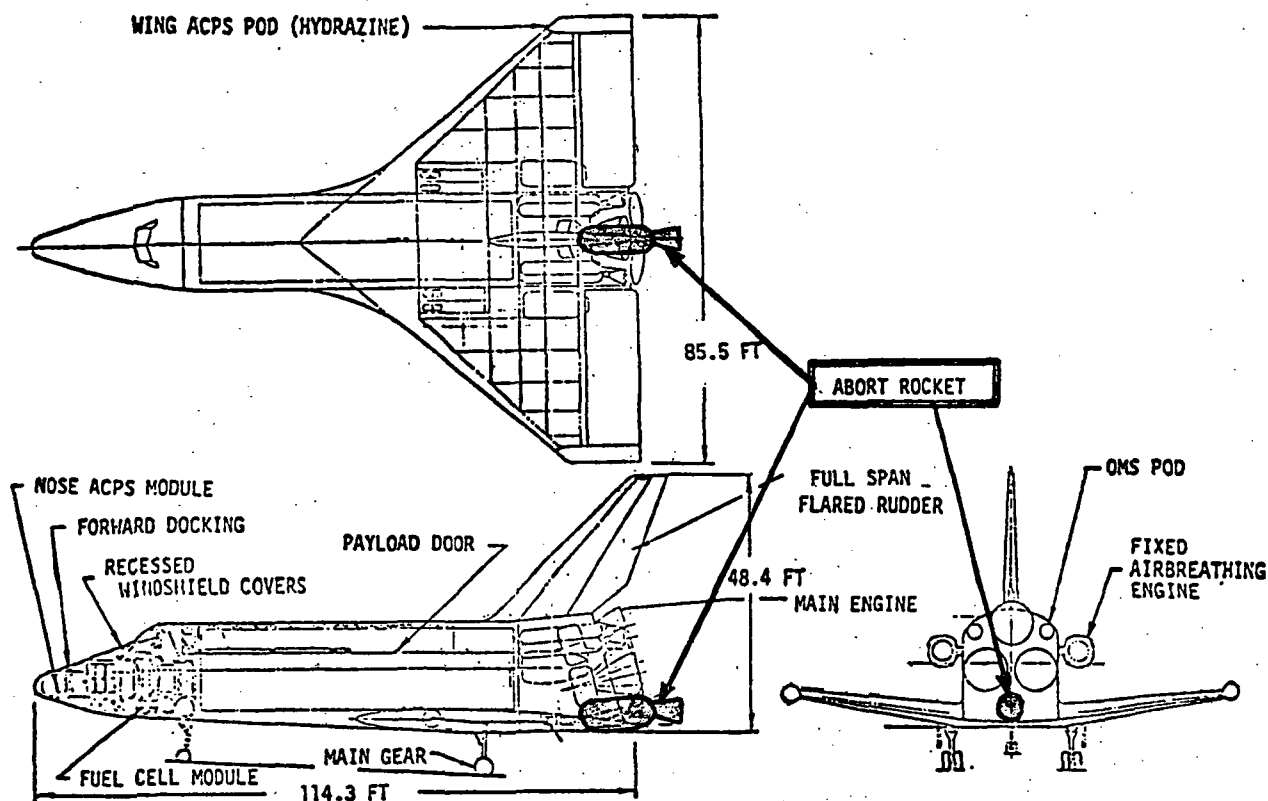
7.1.1 Mode 0

If an emergency arises while the vehicle is on the pad, the crew and passengers will nominally leave the Orbiter, move across the tower swing arm, and descend to a sheltered area via the existing high speed elevator or down slide wires or dive tubes. A lead time of 2 to 3 minutes is required, however.

If the emergency condition will not permit this lead time, an Orbiter with an abort rocket, as shown in Figure 7-2, can separate from the tank and Booster and fly away from the pad. Following rocket burnout and the coast to apogee, the Orbiter glides back to an emergency landing strip located about 1 mile inland from the pad. The abort rocket is sized to provide the Orbiter (configured for the easterly mission - 65 K payload) an initial T/W of approximately 3 g's and a total impulse of near 6 million lb-sec. Such a rocket will not provide the capability of escaping an unanticipated Booster and HO tank explosion, but will

ABORT ROCKET INSTALLATION
15 x 60 PAYLOAD

FIGURE 7-2



provide a relatively fast (several second lead time) intact abort mode for other potential catastrophic failures.

The abort rocket can provide intact abort capability across the initial 35 to 40 seconds of mated vehicle flight, thereby eliminating the "fall back" gap which would otherwise exist. Typical trajectories for the "Mode 0" abort at lift-off, 20 seconds and 40 seconds are shown in Figure 7-3.

The abort rocket is burned in the nominal mission when it is no longer needed for the Mode 0 abort, i.e. after Mode 1 becomes possible. The spent rocket casing is jettisoned shortly after burnout for the Parallel Burn Systems and at staging for the Series Burn System. It is impractical to jettison the casing prior to staging for the Series Burn System because the configuration will not permit acceptable clearance.

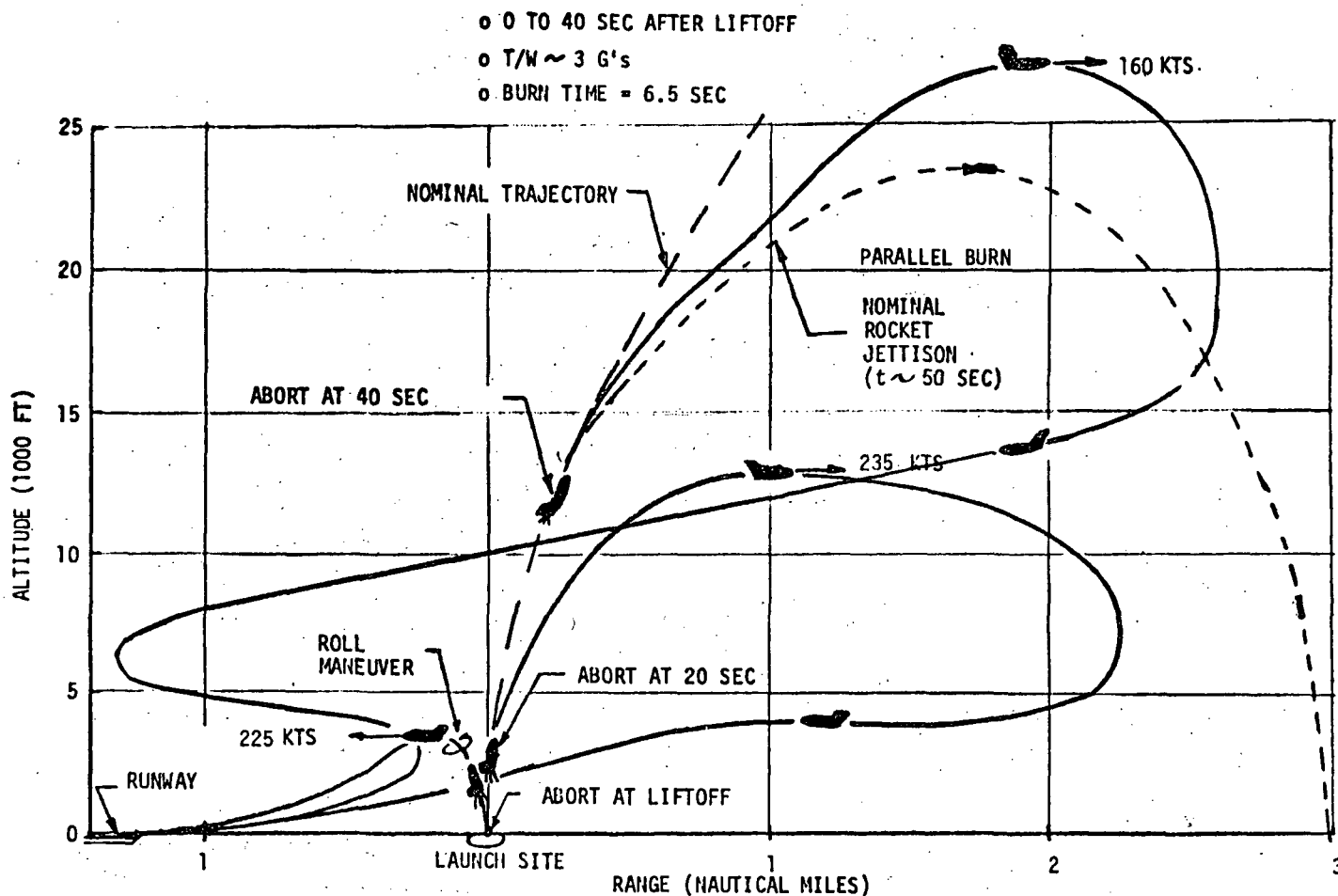
7.1.2 Mode 1

At approximately 40 seconds into the flight, the mated system achieves a velocity and altitude which permits the Orbiter to glide back (unpowered) to the launch site following separation from the HO tank and Booster. This abort mode,

ABORT USING ROCKETS

15 MARCH 1972

FIGURE 7-3



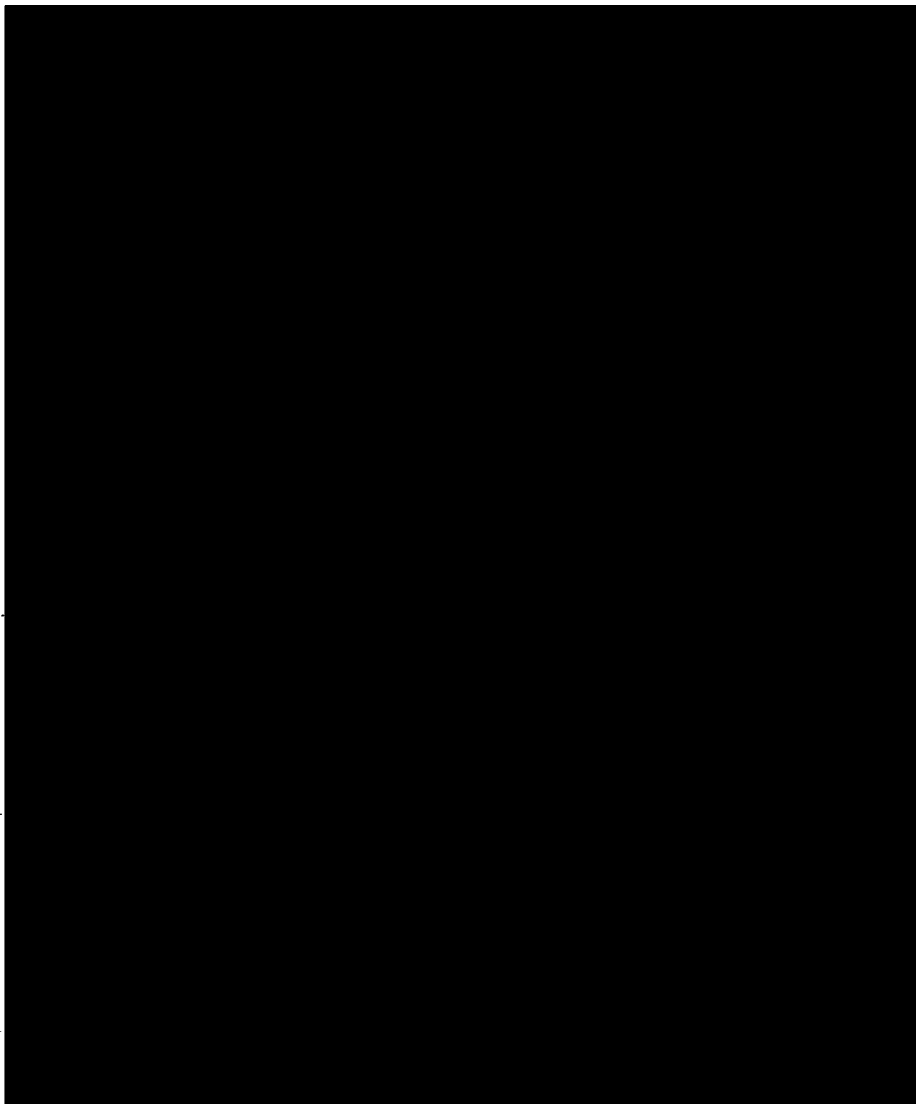
referred to as Mode 1, exists approximately the same period for both the Series and Parallel Burn systems. It involves separation at high dynamic pressure, and the incorporation of this capability will impose structural penalties on both the separation mechanism and the supporting structure. A typical trajectory for a Mode 1 abort at 70 seconds (max q) is shown for a Parallel Burn System in Figure 7-4.

7.1.3 Mode 2

The initial part of the Mode 2 abort region extends from the point in the nominal ascent trajectory (approx. 100 seconds) where the range and velocity of the mated vehicle are too great to permit an unpowered Mode 1 abort through nominal Booster staging. In the event of a Booster failure, the abort mode is to start the Orbiter main engines (series case only), separate the Orbiter plus H₂O tank from the Booster, and perform a powered maneuver to negate the horizontal velocity and propel the Orbiter back toward the launch site. It is assumed that the Orbiter main engines (series case) can be started before the thrust of the Booster main engine falls below the 2.5 g's necessary to provide adequate inlet pressures. When

15 MARCH 1972

DOWNRANGE - NM

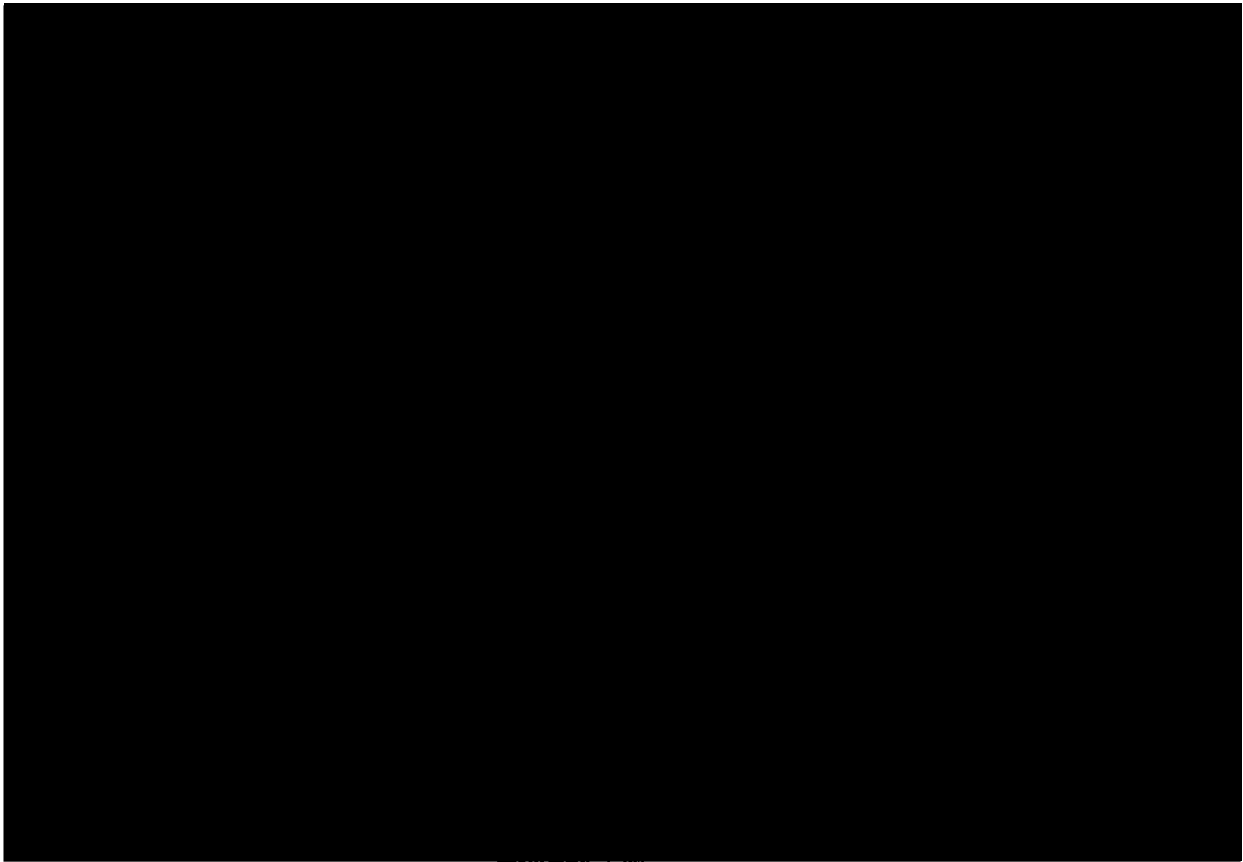


the point is reached where a normal unpowered glide back to the launch site can be accomplished, the Orbiter separates from the HO tank and makes a normal approach and landing. The initial separation of the Booster from the Orbiter plus tank can occur at relatively high dynamic pressure, but the environment is not as severe as is experienced in a Mode 1 separation and abort. However, the Booster thrust must be terminated to enable the separation.

A typical trajectory for a Parallel Burn System aborting at the start of the Mode 2 region is shown in Figure 7-5. It should be noted from the figure that the thrust termination and tank jettison point is only 35-40 miles from the launch site. The maneuver would be performed so as to jettison the tank at an altitude, velocity and range which will permit an entry and impact at a safe distance from the site. For an abort at this point, the tank will likely be partially filled when it is jettisoned and will break up under the buildup of entry and impact loads.

15 MARCH 1972

FIGURE 7-5

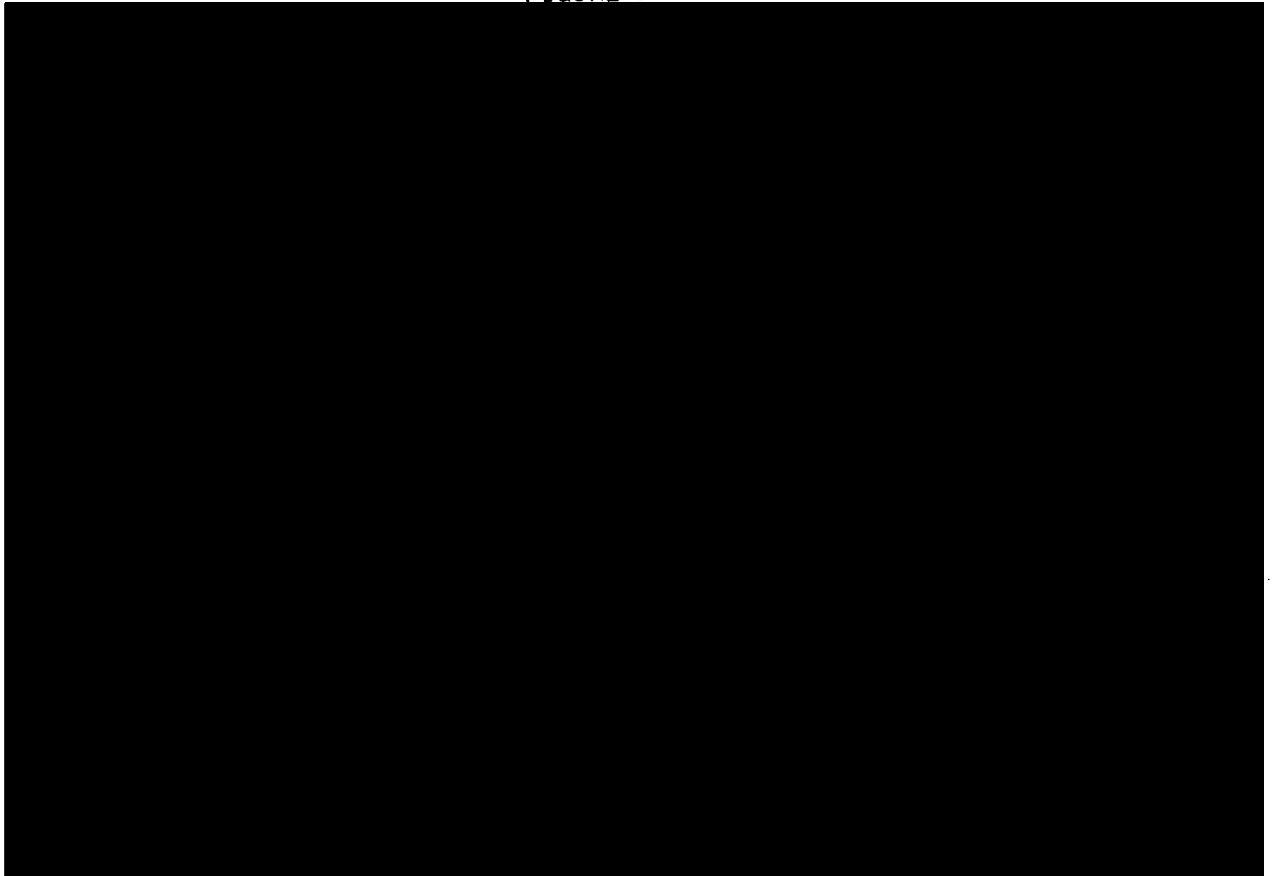


The second part of the Mode 2 region extends from staging to the point in the normal ascent trajectory where all the main propellant in the H0 tank is required to perform the retrograde and flyback maneuver. For a lift-off T/W of 1.4 and a staging velocity of 4000 fps, the Mode 2 region extends 120-135 seconds beyond staging. (This assumes that the abort forcing failure is an Orbiter Main engine out. The region would be larger if all Orbiter main engines were available.) A typical trajectory for a Parallel Burn System aborting approximately 100 seconds after staging is shown in Figure 7-6.

7.1.4 Mode 3

The Mode 3 region begins at the point in the Orbiter ascent phase where the Orbiter with a main engine out can inject into a once-around trajectory, returning to the launch site. This point depends on trajectory conditions, Orbiter T/W, main engine EPL, OMS thrust level, and OMS propellant available to supplement the main engine burn. The orbiters for both the Series and Parallel Burn systems employ three 470,000 lb thrust HiPc engines, which provide a T/W of .93 at staging. The engines have up to 109% EPL, depending on the mixture ratio swing that is required. The Orbiter OMS provides 15,000 lb of thrust and 900 fps of ΔV (easterly mission) for use in the abort maneuver.

FIGURE 7-6



Current analysis indicates the Mode 3 capability for these systems coincides with the upper limit of the Mode 2 regions. However, 109% EPL and all the OMS propellant is required, which means that the butting of these two regions is marginal. For the polar mission, only 650 fps of OMS ΔV is available for abort but this is somewhat offset by the higher T/W of the Orbiter. For either mission, however, there is a very high probability that a significant quantity of FPR propellant would be available for use in the abort, extending the Mode 3 region closer to nominal staging. With these considerations the initial point of the Mode 3 region is shown to coincide with upper limit of the Mode 2 region. If the number of Orbiter engines were increased from 3 to 4, or the T/W increased, the Mode 3 region would overlap the Mode 2 region.

7.1.5 Mode 4 (Degraded Mission)

When the Orbiter reaches the point in its ascent trajectory where with an engine out it can inject into 50 x 100 NM orbit with enough OMS propellant remaining to circularize at 100 NM and deorbit (350 fps for both maneuvers), the vehicle can abort to orbit or fly a "degraded mission". In this case, some of the mission objectives may be accomplished prior to the deorbit maneuver and nominal entry.

15 MARCH 1972

7.1.6 Mode 5 (Parallel Burn System Only)

If an Orbiter main engine fails in the Parallel Burn System during mated ascent, several options are available to the Orbiter. The remaining Orbiter engines and the Booster engine could be shut down and a Mode 0 or 1 abort mode could be flown. However, the preferred option would likely be to continue the mated ascent phase targeting to a Booster burnout condition that would permit a near nominal separation and a Mode 1 or 2 abort mode. The latter approach (continue to Booster burnout) is referred to as Mode 5.

7.2 Abort Separation

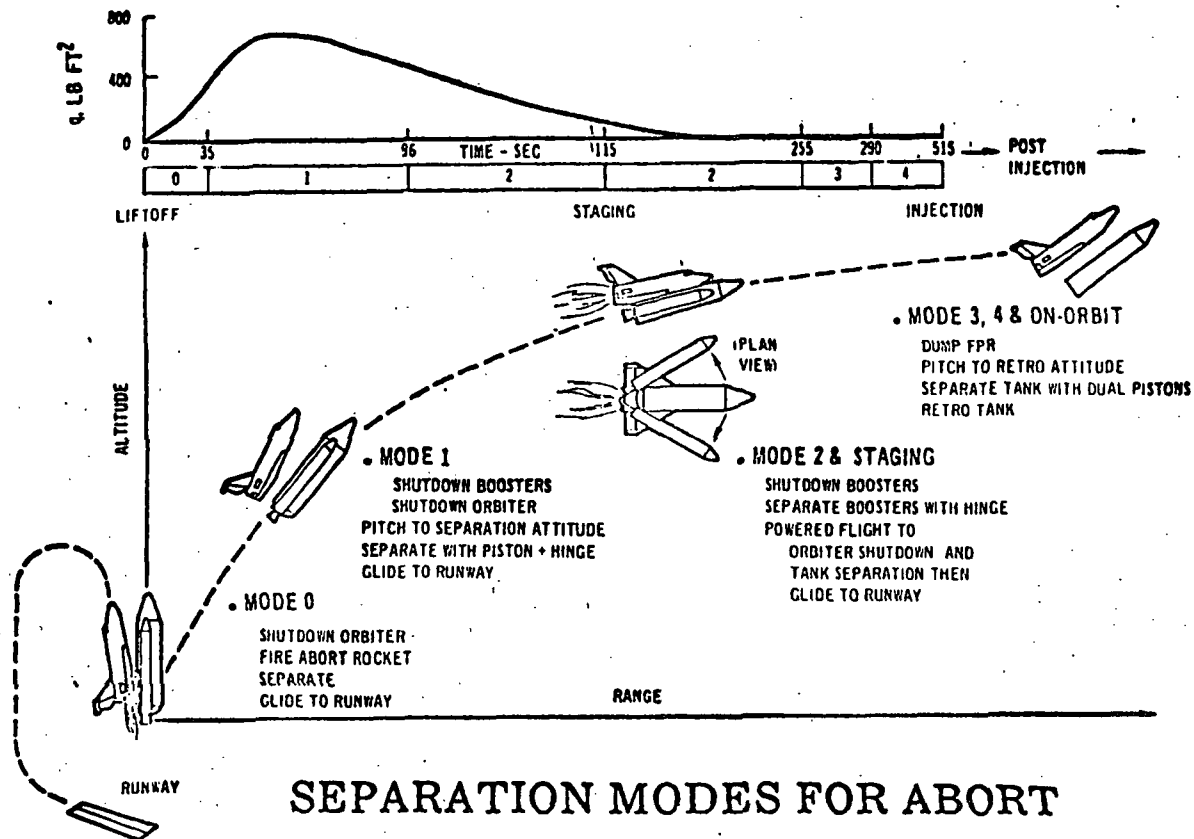
Separation of Space Shuttle vehicle components is essential to intact recovery of the Orbiter following an abort. For Mode 0 abort, a solid rocket is employed to provide the energy required for separation and return. Separation must be performed at the Orbiter-tank interface. From about 40 seconds into the launch trajectory, when enough energy has been acquired for a glide return, until the unpowered Orbiter has too much energy to turn and glide back, separation will also be performed at the Orbiter tank interface (Mode 1). Abort at a later point in the launch trajectory, prior to nominal staging, requires two separations; first the Booster from the Orbiter plus tank, and then jettison of the tank, following the powered retrograde and flyback maneuver, enabling the Orbiter to glide back to the launch site (Mode 2). Following nominal staging, the separation of the tank from the Orbiter for Modes 3 and 4 is near nominal.

The critical events of separation modes for abort for the Parallel and Series Burn Systems are shown in Figures 7-7 and 7-8, respectively. For off-the-pad abort (Mode 0) the separation mechanism is fired in conjunction with the abort rocket. For Modes 0, 1, and 2 a forward piston and an aft hinge rotate the Orbiter nose away from the tank plus Booster. The mechanism is sized to provide satisfactory separation characteristics under the high dynamic pressure of Mode 1 abort. In Mode 2, tank separation is performed after the vehicle has turned back toward the launch site and has acquired sufficient energy to return. For Modes 3 and 4 and nominal post-injection separation, a dual piston mechanism provides the separation impulses at the Orbiter tank interface. The dual pistons are sized for nominal separation to provide a tank velocity of 1.5 ft/sec on-orbit. The two act to balance the moments about the tank c.g. so that no tank rotation is induced by separation. At the tank-Booster interfaces, the separation mechanism is an aft

SEPARATION MODES FOR ABORT

Parallel Burn

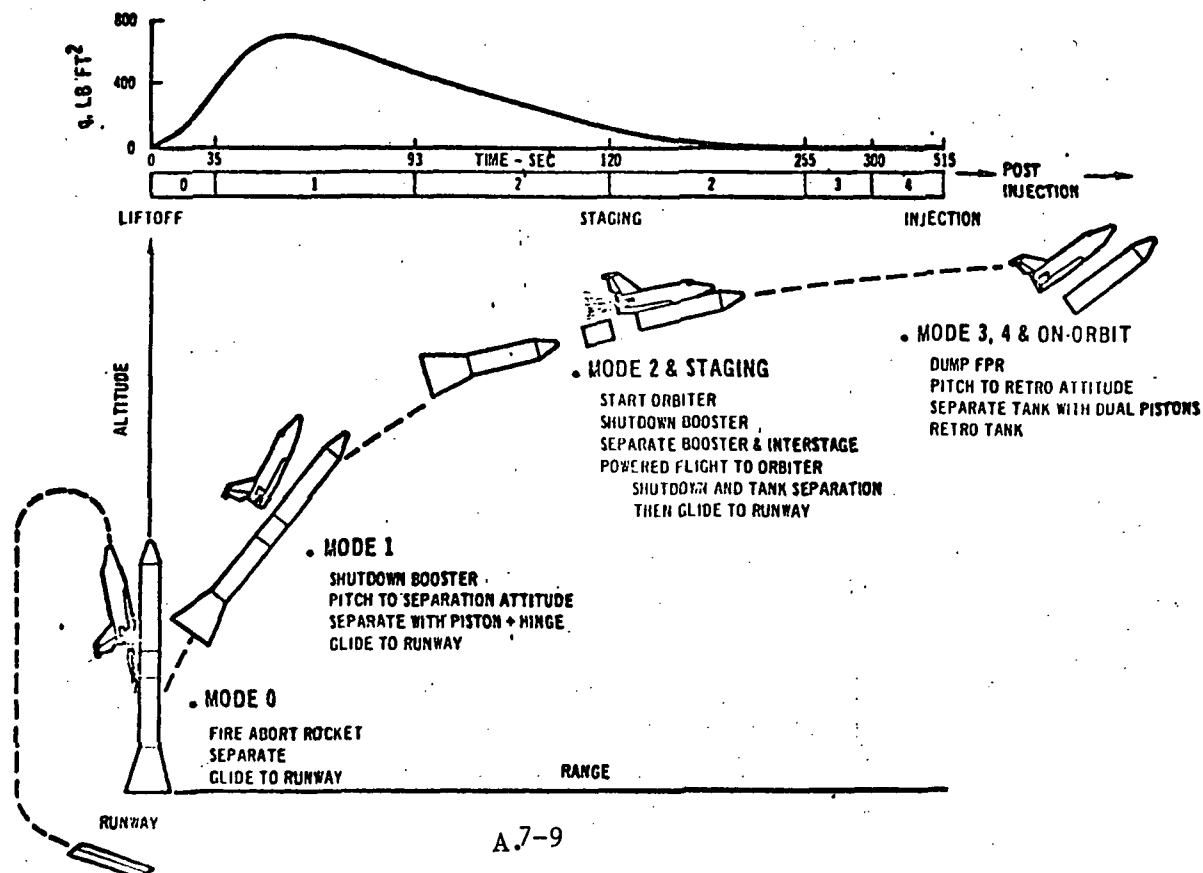
FIGURE 7-7



SEPARATION MODES FOR ABORT

Series Burn

FIGURE 7-8



15 MARCH 1972

hinge for the parallel burn SRM's and an interstage for series burn. These mechanisms provide satisfactory separation for Mode 2 abort (prior to staging) and nominal staging. The mechanisms are sized by the Mode 2 abort dynamic pressures.

For abort Modes 0 and 1, the parallel burn concept requires Orbiter engine shut down. Mode 2 abort, however, requires starting the Orbiter engines for the series burn case. Booster thrust termination is required for Mode 2 abort prior to staging and although not required for separation in Mode 0 or 1 abort, is performed for safety.

7.3 System Impact

The intact abort gaps typical of either the Series or Parallel Burn System, before any abort penalties to the system are imposed, are shown in the first bar of Figure 7-9. These gaps can be eliminated by incorporating the pad abort system (solid rocket) and a separation system which will enable Orbiter separation at high dynamic pressure. The weight and cost penalties associated with the incorporation of these abort capabilities are shown in Figure 7-10.

The program cost increase shown includes ground testing only. If the critical abort modes identified by the triangles in Figure 7-9 are flight demonstrated, the basic cost of approximately \$250 M could increase to approximately \$600 M, depending on the degree of demonstration required, as shown in Figure 7-11. If the vertical flight test program is manned, no abort flight demonstration is recommended. If unmanned flights are anticipated, it is recommended that only the abort demonstration that can be attained at minimum cost as an integrated part of the vertical flight testing be performed.

FIGURE 7-9
ABORT GAPS

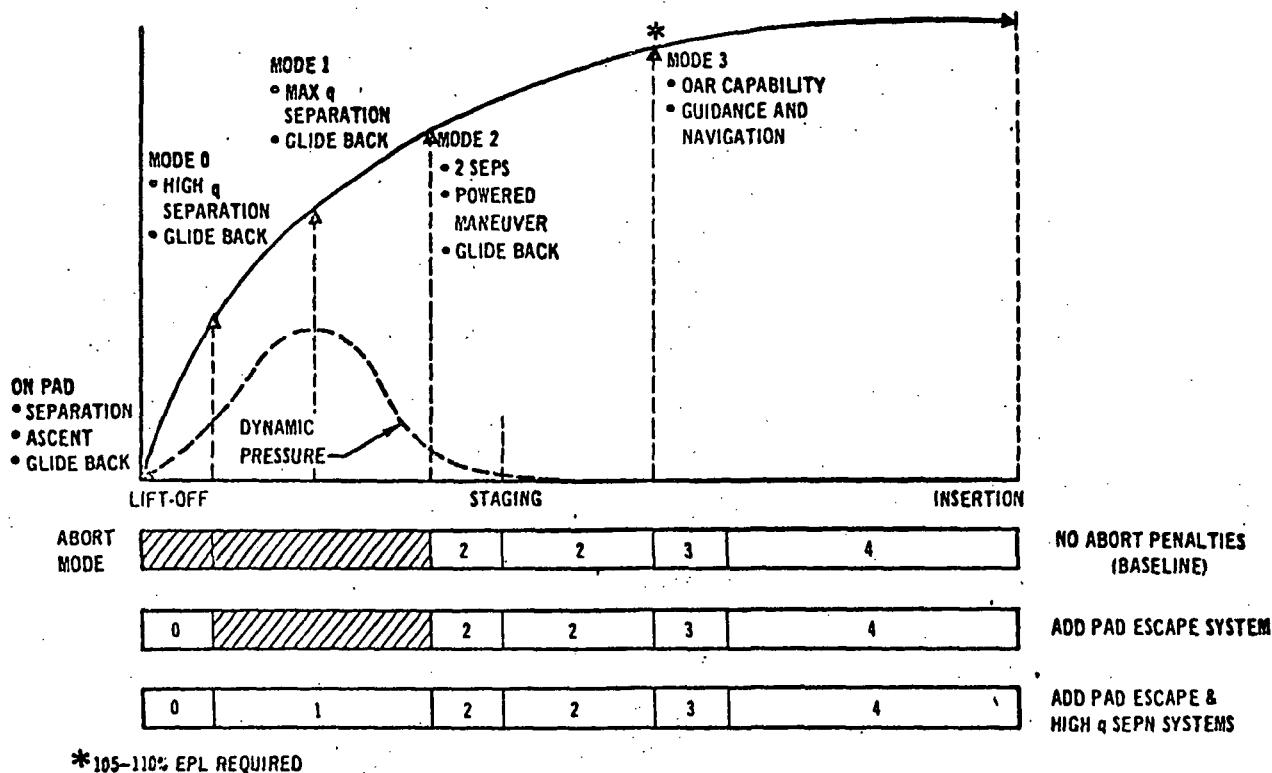
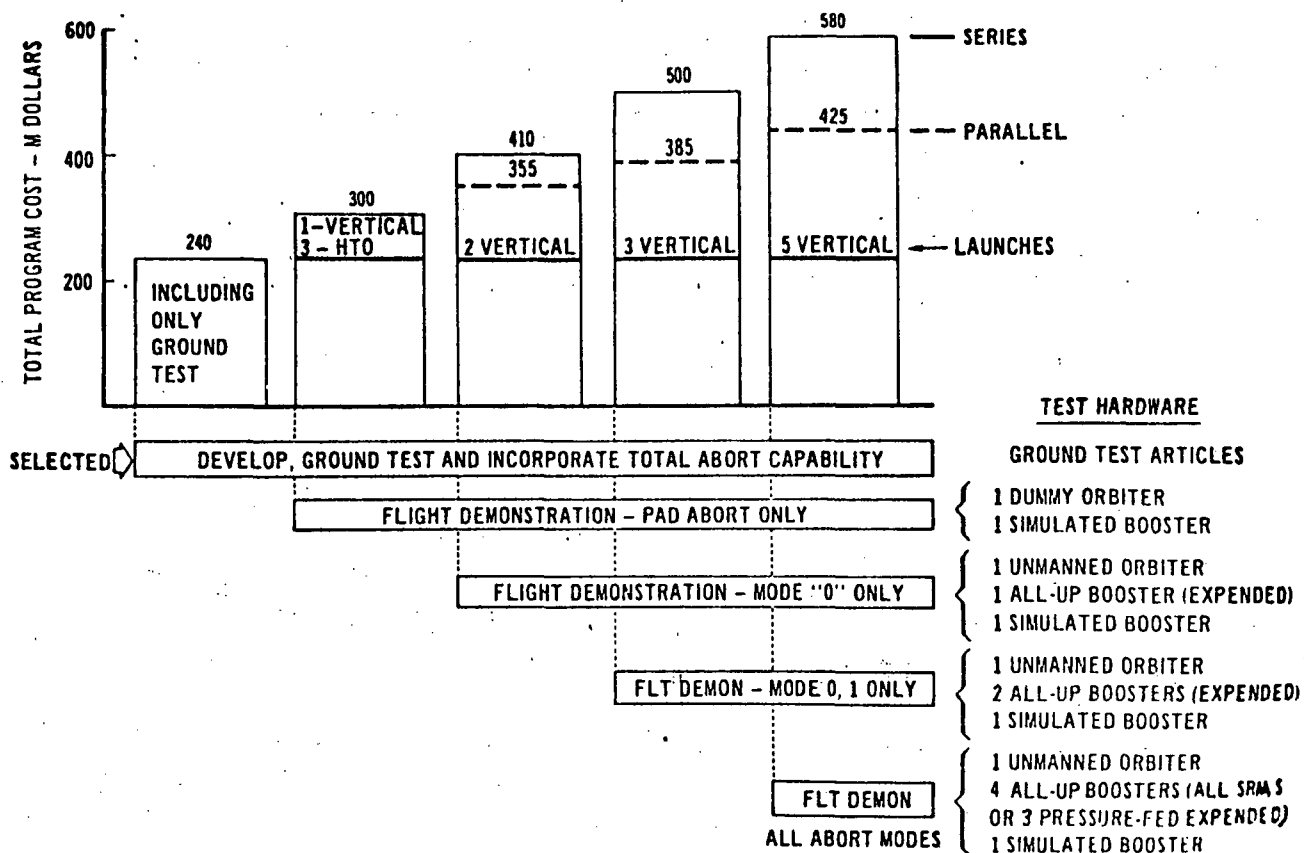


FIGURE 7-10
WEIGHT AND COST PENALTIES
TO ELIMINATE ABORT GAPS

ABORT CAPABILITY	ORBITER DRY WEIGHT INCREMENT (LB)	PROGRAM COST INCREASE FOR ABORT (GROUND TEST ONLY) (\$M)	
		SERIES BURN	PARALLEL BURN
BASELINE	REF	REF	REF
BASELINE + PAD ABORT (ROCKET)	+3300	+208	+213
BASELINE + PAD ABORT + HIGH Q ABORT SEPARATION	+5300	+240	+245

FIGURE 7-11

TOTAL COST TO ELIMINATE GAPS AND DEMONSTRATE CAPABILITY



8.0 Operations

Ground Operations for the Space Shuttle include post-landing activities, maintenance and refurbishment of Orbiters and Boosters, and the prelaunch activities: erection, mating and servicing. Real time support of flight operations is also provided from lift-off to landing. This includes communications, tracking, contingency mission planning, and technical consultation.

8.1 Ground Operations

Ground Operations activities are based on the concept of prepad integration and the use of existing facilities and systems.

Figure 8.1-1 presents the configuration and orientation of the Parallel Burn Twin SRM Shuttle on an existing (modified) mobile launcher. The physical integration takes place in the high bay area of the Vehicle Assembly Building (V.A.B.) at K.S.C. Prior to mating, the Orbiter progresses through postlanding, maintenance, and horizontal prelaunch activities in adjacent areas. The segmented solid (2-156") Boosters, after completing inspection and off-loading activities, are prepared for buildup on a mobile launcher unit, described below. After erection of the twin Boosters and subsystem checkout, the external hydrogen/oxygen tank is erected and connected to the Booster structural interfaces. Then the Orbiter, having completed ground turnaround activities (including cargo installation) is being into the high bay area for erection and mating. Figure 8.1-2 presents a timeline of Booster and Orbiter activities during a typical turnaround cycle for expendable SRM's.

The mobile launcher is a key facility integration unit. Three units will be available and will be used for Orbiter/tank vibration testing during early program operations. One unit will be modified for flight readiness firings (FRF) of the Orbiter (including H₂O tanks) and will remain in this configuration until completion of the fifth Orbiter. One other unit will require modification for Booster FRF if a liquid propellant system is selected.

Basically each mobile launcher will be modified by 1) removing existing equipment lines, swing arms etc., 2) reducing the height of the umbilical tower by 80 feet, 3) removing the structural hardpoints on the launcher deck, and 4) reconfiguring the engine chamber area.

Modifications are also required at both the V.A.B. facility and the pad(s). Typical examples include modifications to vertical doors, extendable platforms, interface services, and the propellant storage, emergency egress, and blast

15 MARCH 1972

FIGURE 8.1-1
PARALLEL BURN TSRM/MOBILE LAUNCHER

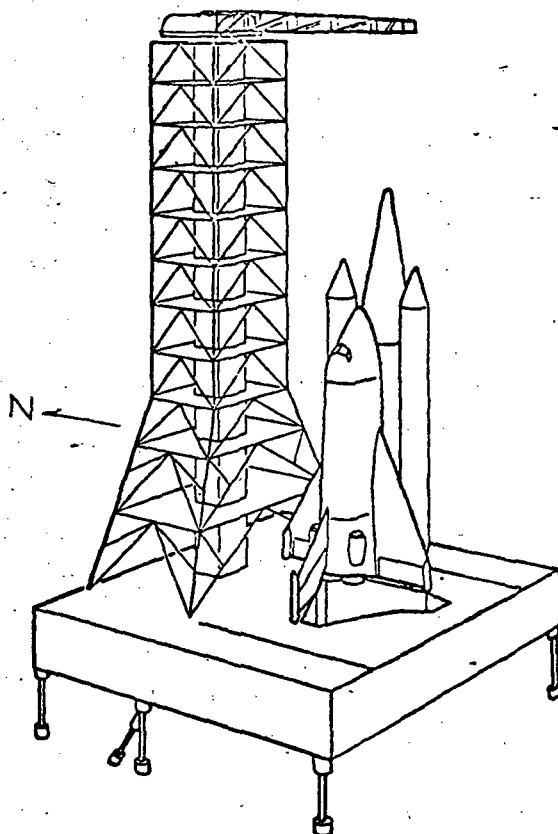
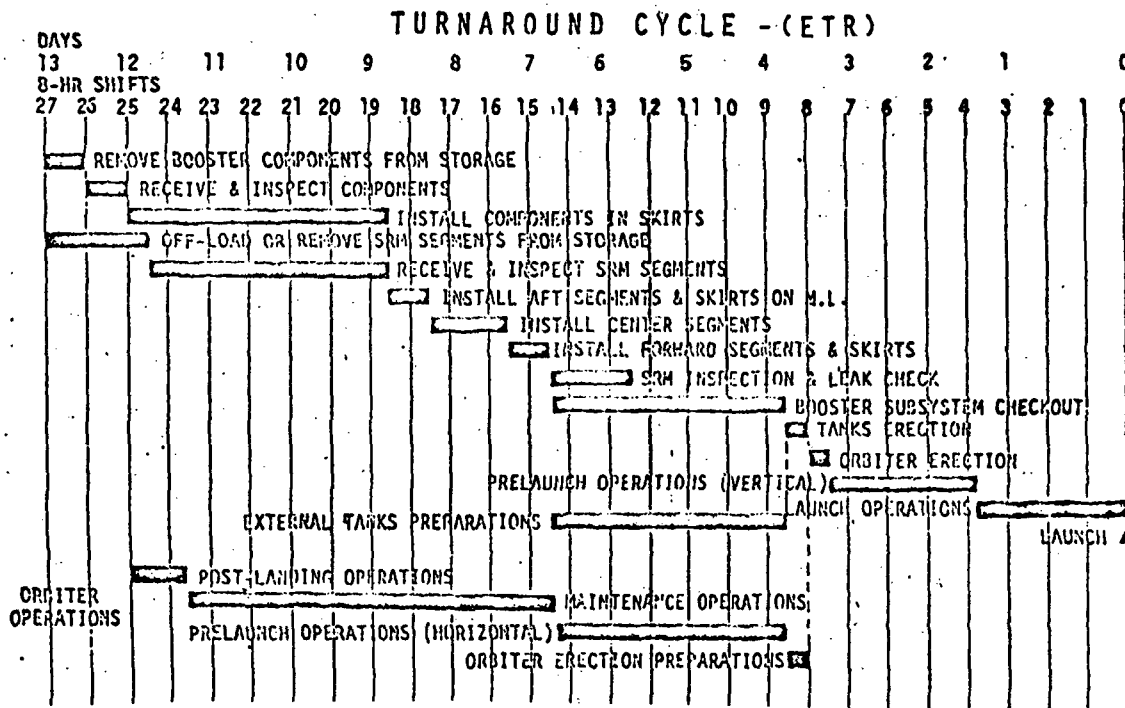


FIGURE 8.1-2
PARALLEL BURN TSRM



protection areas. The existing Mobile Service Structure will be modified for vertical "on pad" cargo module replacement. As an alternate, consideration is being given to the design and construction of a simplified mobile tower.

The Series Burn Liquid Booster configuration presented in Figure 8.1-3 follows the same prepad integration concept as the twin configuration. The integration process begins in the V.A.B. high bay area with Booster erection on the mobile launcher unit. The external hydrogen/oxygen tank is then erected and mated to the Booster, and the Orbiter is erected and mated to the tank.

This configuration requires a swing arm for umbilical interface connections to the main engine section of the Orbiter. A dampener arm will be provided from the mobile launcher umbilical tower to provide additional structural support during transfer and on pad operations. A rotating fixture for payload changeout will also be provided on the umbilical tower to allow removal and replacement of a complete cargo module.

The Orbiter is offset from the umbilical tower to provide clearance for launch and also for crew ejection.

Figure 8.1-4 presents the timeline estimate for a complete turnaround cycle. The nominal estimate for post water recovery operation is 2 1/2 days. However, due to the relative simplicity of this vehicle compared with the fully reusable system, the total turnaround time (less than 14 days) is not increased.

8.2 Flight Operations Support

The baseline support system shown in Figure 8.2-1 is essentially the same as was defined for the Phase B Shuttle. Figure 8.2-2 shows the proposed flight support system. During the first two years, communications and navigation support are provided by the Manned Space Flight Network (MSFN) (or its 1978 equivalent), together with whatever network (landlines or wideband) is available for communications between the remote sites and the operations center.

A Tracking and Data Relay Satellite System (TDRSS) (including satellites, ground station and ground network) is assumed to be operational by the third year and will provide all communications and navigation support. Even if a TDRSS is not available, MSFN support requirements for Shuttle are halved ("Limited" MSFN configuration), as soon as onboard autonomous navigation capability is verified. Console support in the Ground Mission Operations Center (GMOC), which is initially high for flight test and early operations flights, is also reduced for the remainder of the operations.

FIGURE 8.1-3

SERIES BURN LIQUID BOOSTER/MOBILE LAUNCHER

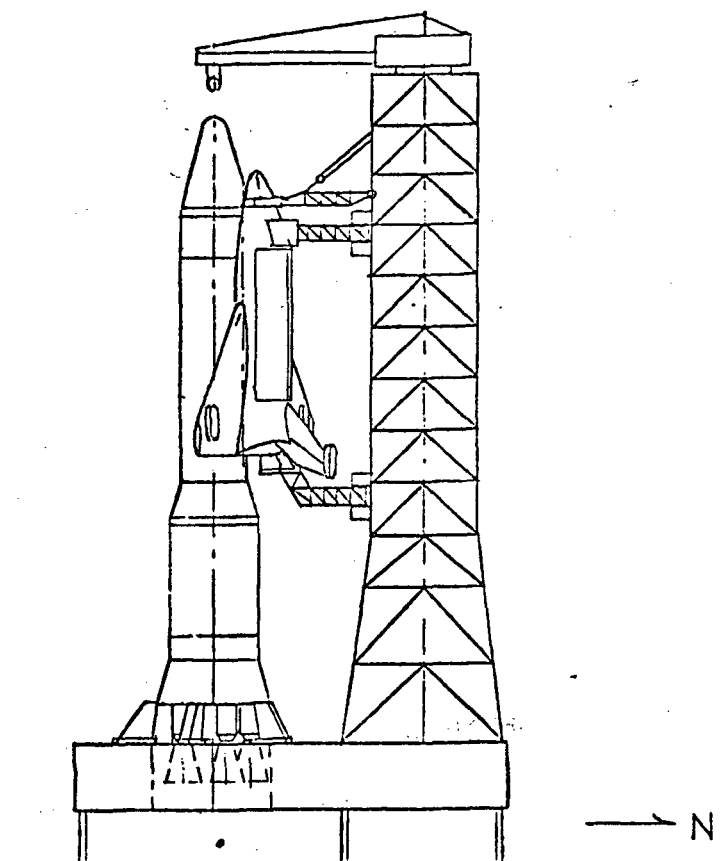


FIGURE 8.1-4

SERIES BURN LIQUID BOOSTER
TURNAROUND CYCLE

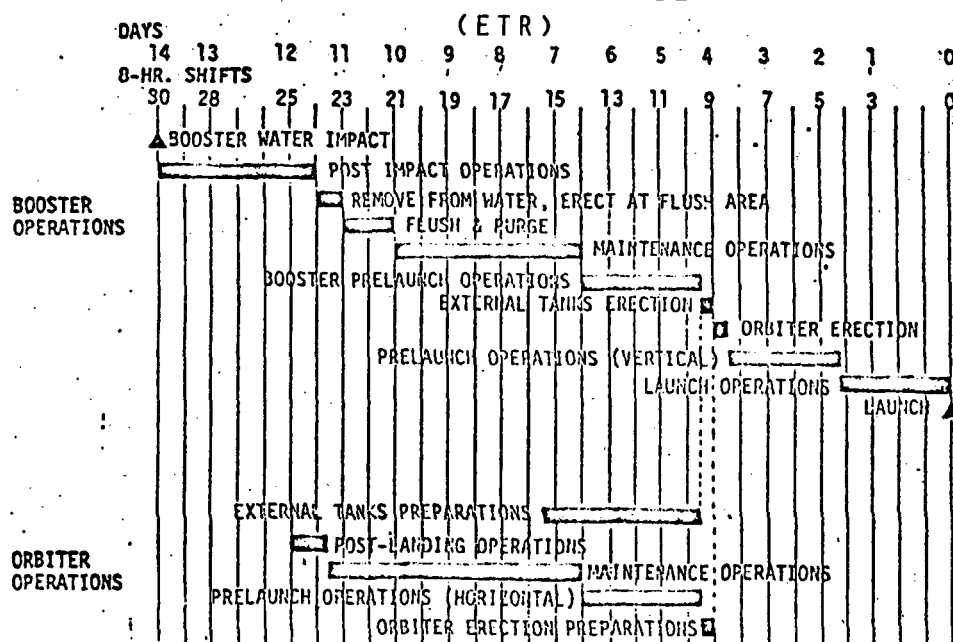


FIGURE 8.2-1

BASELINE MISSION OPERATIONS SUPPORT CONCEPT

	1978	1979	1980	1981 AND ON
FLIGHT PHASE	FMOF		OPERATIONAL TDRS	
	FLT TEST	EARLY OPERATIONAL	OPERATIONAL	
COMMUNICATIONS SUPPORT	MSFN	MSFN	MSFN TDRSS	TDRSS
NAVIGATION SUPPORT	H/S-S/T MSFN	H/S-S/T MSFN	H/S-S/T TDRSS MSFN	H/S-S/T TDRSS
GMOC CONSOLES	28	28	21	21
MANNING				
FULL TIME	28	28	12	12*
ON CALL	0	0	9	9

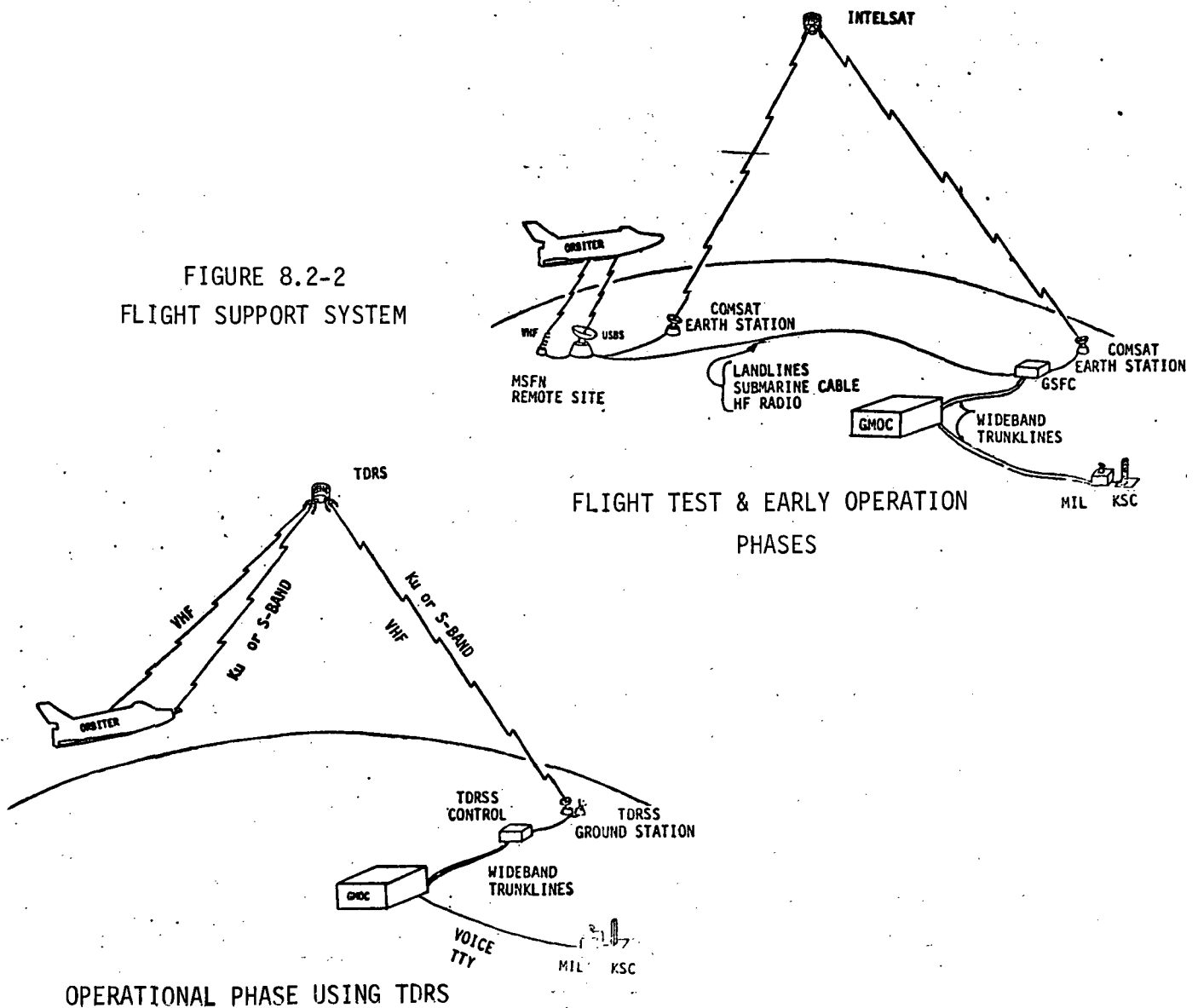
MSFN -- MANNED SPACE FLIGHT NETWORK OR ITS 1978 EQUIVALENT

TDRSS - TRACKING & DATA RELAY SATELLITE SYSTEM

H/S-S/T - HORIZON SENSORS & STAR TRACKERS

* - FIVE OF THESE POSITIONS COULD BE PART OF TDRSS SYSTEM & NOT REQUIRED AT GMOC

FIGURE 8.2-2
FLIGHT SUPPORT SYSTEM

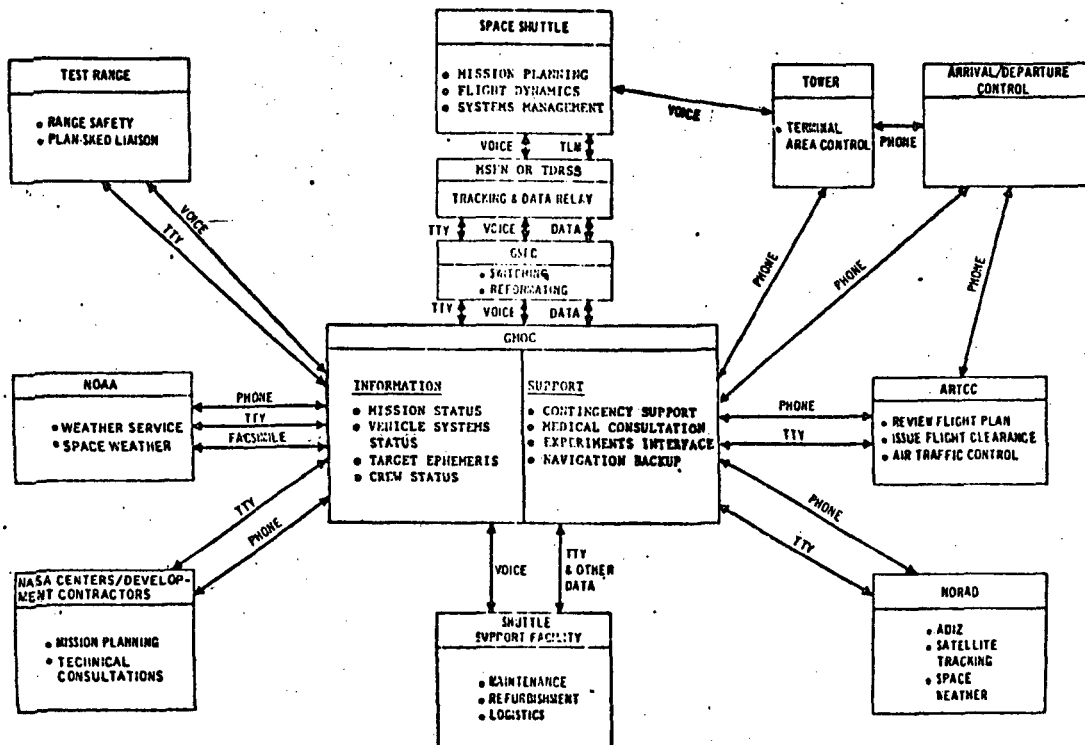


15 MARCH 1972

8.2.1 Ground Mission Operations Center

GMOC functions as shown in Figure 8.2-3 are to coordinate and control communications, to interface with all outside supporting agencies, and to provide consultation as requested by the flight crew.

FIGURE 8.2-3
GROUND MISSION OPERATIONS CENTER



The GMOC will provide flight support consistent with an overall program concept of autonomous space Shuttle operations, minimal ground support and turnaround time, and the lowest practical operating costs.

8.2.2 Manned Space Flight Network

It is expected that by 1978, the current MSFN and Space Tracking and Data Acquisition Network (STADAN) will be combined into a single unified network. This network will probably consist of about 15 remote sites. The communications circuits could either be a low data rate system similar to that in use today, or a wide band direct return system. The low data rate system would tie into a central data processing and switching center and thence to the GMOC. The wide band system could tie remote sites directly to the GMOC.

8.2.3 Tracking and Data Relay Satellite System - The baseline support system

15 MARCH 1972

concept assumes the use of a TDRSS. When this concept becomes operational, it will be used for all communications and navigation support. The elements of this system include:

- o TDRSS Satellites
- o TDRSS Operational Control Center
- o TDRSS Ground Stations
- o TDRSS Network Control Facilities
- o Ground Data Handling System

Since the intent of a TDRSS is to support many users, both manned and unmanned, a high degree of standardization will be required. This will impact the T/M formats, data handling techniques, and command formats and procedures used for the Orbiter, as well as the scheduling of TDRSS capabilities (e.g. high gain antennas), orbit determination procedures, etc. These are things that can only be resolved once the TDRSS is defined.

8.2.4 Landing Support

Ground support required by the Orbiter during aerocruise or landing will be provided by existing Federal Aeronautics Authority (FAA) or military aircraft cruise or landing aids. The baseline Orbiter uses TACAN for post-entry or aerocruise navigation updates, UHF radio for voice communications, and an Instrument Landing System (ILS) for final landing approach navigation and guidance data.

8.2.5 Mission Control Functions

During a Shuttle Mission, the functions required to insure crew/vehicle safety and mission success are divided between ground, and onboard systems and personnel as shown in Figure 8.2-4.

15 MARCH 1972

FIGURE 8.2-4

MISSION PLANNING

FLIGHT DYNAMICS

ONBOARD FUNCTIONS

- DETERMINE CURRENT MASS PROPERTIES FOR VARIOUS VEHICLE CONFIGURATIONS
- UPDATE RETURN PHASING & DEORBIT PLAN
- GENERATE CONTINGENCY DEORBIT DATA AS NECESSARY
- COMPUTE REQUIRED VEHICLE ATTITUDES
- MONITOR NOMINAL MISSIONS & EVALUATE EFFECTS OF CHANGES ON NOMINAL MISSIONS
- CALCULATE AND EVALUATE ORBIT SHAPING OR RENDEZVOUS MANEUVERS
- UPDATE LAUNCH GUIDANCE TARGETING

SHARED ONBOARD/GROUND FUNCTIONS

- MONITORING OF SOLAR ACTIVITY
- UPDATES TO CREW ACTIVITY TIMELINES

GROUND FUNCTIONS

- DETERMINE NOMINAL LAUNCH TRAJECTORY
- PROVIDE NOMINAL PRE-MISSION PLANS
- PROVIDE BACKUP MISSION PLANNING SUPPORT TO FLIGHT CREW AS NECESSARY
- LANDING SITE WEATHER MONITOR

ONBOARD FUNCTIONS

- UPDATE ORBITER STATE VECTOR USING HORIZON SENSOR AND STAR TRACKER DATA
- DETERMINE THE RELATIVE STATE VECTOR BETWEEN ORBITER AND TARGET VEHICLE
- COMPUTATION OF GUIDANCE TARGETING
- MONITORING EXECUTION OF MANEUVERS AND STATUS OF CURRENT TRAJECTORY
- PERFORMING ALIGNMENTS AND CALIBRATIONS OF THE VARIOUS ONBOARD SENSORS
- MONITORING TRAJECTORY DURING ENTRY
- DETERMINE TANK DISPOSAL TIME AND ATTITUDE

SHARED ONBOARD/GROUND FUNCTIONS

- MONITOR BOOST TRAJECTORY PARAMETERS TO DETERMINE DEVIATION FROM NOMINAL

GROUND FUNCTIONS

- PROVIDE TARGET TRAJECTORY DATA FOR ASCENT AND RENDEZVOUS TARGETING
- PROVIDE ORBITER STATE VECTOR UPDATES BASED ON MSPN OR TDPS TRACKING DATA
- PROVIDE PROCEDURAL OR COMPUTATIONAL SUPPORT UPON REQUEST
- PROVIDE GROUND AIDS FOR CRUISE AND LANDING NAVIGATION
- PROVIDE BACKUP FLIGHT DYNAMIC SUPPORT TO CREW AS NECESSARY
- PREDICT BOOSTER IMPACT POINT

SYSTEMS MANAGEMENT

BASELINE CREW FUNCTIONAL
DIFFERENTIATION

ONBOARD FUNCTIONS

- MONITOR EMERGENCY CAUTION AND WARNING SYSTEM
- MONITOR AND CORRELATE SYSTEMS PARAMETERS TO VERIFY NOMINAL OPERATION OR DETECT ABNORMAL SITUATIONS
- ISOLATE MALFUNCTIONS TO SUBSYSTEM(S) OR COMPONENT(S) LEVEL AND TAKE APPROPRIATE CORRECTIVE ACTION
- EXERCISE SELF CHECKS OF SYSTEMS/SUBSYSTEMS TO VERIFY PROPER OPERATION
- MONITOR AUTOMATICALLY SEQUENCED FUNCTIONS, INITIATE APPROPRIATE PRE-PROGRAMMED ALTERNATE SEQUENCES AND GENERATE ALTERNATE SEQUENCES
- MANAGE SYSTEMS CONFIGURATION TO UTILIZE SYSTEM RESOURCES
- MANAGE CONSUMABLES AND IDENTIFY OFF NOMINAL USAGE RATES

GROUND FUNCTIONS

- PROVIDE BACKUP SYSTEMS MANAGEMENT SUPPORT TO FLIGHT CREW AS NECESSARY
- PROVIDE SYSTEM CONFIGURATION, STATUS, AND PERFORMANCE DATA KEEPING

COMMAND PILOT

PILOT

- ASCENT MONITORING
- ON-ORBIT BURNS
- NOSE DOCKING
- BRAKING
- ENTRY MANUAL BACKUP
- APPROACH AND LAND
- BACKUP PILOT FUNCTIONS
- ON-ORBIT ATTITUDE CONTROL

- ASCENT NAVIGATION MONITOR
- RENDEZVOUS NAVIGATION
- ENTRY NAVIGATION
- COMMUNICATION FUNCTIONS
- RADIO AND FINAL APPROACH NAVIGATION
- BACKUP TO COMMAND PILOT
- MANIPULATOR ARM CONTROL
- ACPs COMMAND FOR PAYLOAD OPERATIONS

SYSTEM ENGINEER

PAYLOAD SPECIALIST

- SUBSYSTEM FUNCTION MONITOR
- NON-TIME CRITICAL FAILURE MONITOR
- FAULT ISOLATION
- CIRCUIT BREAKER CONTROL
- FUEL CELL CONTROL
- ECS CONTROL

- PAYLOAD BAY ENVIRONMENT MONITOR
- PAYLOAD MONITOR
- PAYLOAD ACTIVATION AND TEST
- PAYLOAD EVA/IVA ACTIVITY
- PAYLOAD ALIGNMENT (OPTICS SYSTEM)
- PAYLOAD COMMUNICATOR

15 MARCH 1972

9.0 Recommended System

The Shuttle Phase B Study Extensions were concluded with the following recommendations:

A. Abort:

- o Incorporate pad abort and high-q separation capability in baseline.
- o Conduct comprehensive ground testing of abort systems. In addition, if unmanned flight testing is incorporated as part of the development program, then include a limited abort demonstration as part of the vertical flight test program.

B. Orbiter Thrust Level:

- o Retain three engines at 470,000 lb vacuum thrust

C. Payload Size and Weight:

- o Retain 15 x 60 ft and 65,000 lb east payload

D. Booster System Selection:

- o Recommend Twin 156" SRM with thrust termination and water recovery as first choice, because of its lowest RDT&E cost and lowest technical risk.
- o Recommend F-1 Pump Fed System as second choice.

Figure 9-1 summarizes the weights and costs of the three main candidate systems with the recommended additional capabilities incorporated. Of these systems the Parallel Burn Twin 156" system is recommended as the initial baseline for the Space Shuttle Program.

9.1 Recommended System Description

The recommended Shuttle system is illustrated in Figure 9-2. It consists of two 156" solid rocket motors (SRM's) side-mounted on the Orbiter HO tank. Thrust of each SRM is 2,776 K lb. These in combination with the three Orbiter engines generate a lift-off thrust-to-weight $(T/W)_1$ of 1.405.

Each SRM has a consumed propellant weight of 1,149 K lb and a mass fraction of 0.867, including interstage structure and approximately 33.4 K lb of recovery systems provisions. The Orbiter HO tank contains 1,608 K lb of propellant (6:1 of LOX and LH_2 by weight) and has a mass fraction of 0.959. The two recoverable SRM's are separated from the Orbiter/HO tank at a relative staging velocity of 4,000 fps. They are each initially decelerated by a ballute and finally braked by solid rockets

FIGURE 9-1
MAIN CONTENDERS WITH UPDATED CAPABILITIES

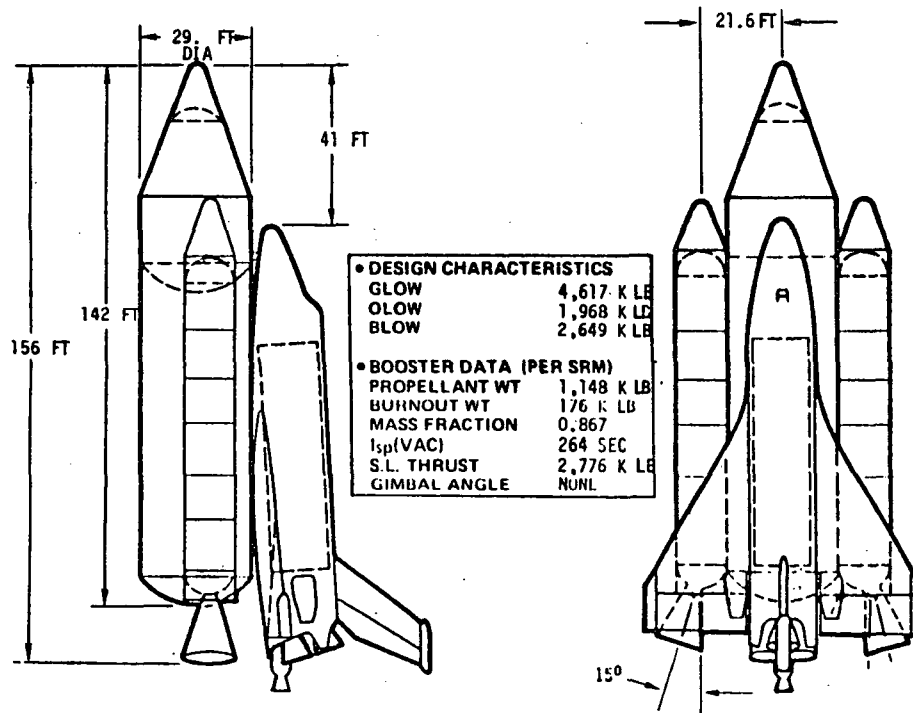
1-423A

- OFF-THE-PAD ABORT CAPABILITY
- RECOVERABLE BOOSTERS
- LARGE ORBITER, 15 x 60 BAY
- ORBITER ENGINES: 3-470 K LB

	SERIES BURN		PARALLEL BURN
	PRESSURE FED	PUMP FED F-1	TWIN 156 IN SRM
WEIGHTS (K LB)			
PAYLOAD (EAST)	65.0	65.0	65.0
GLOW	5930.9	4573.4	4617.1
OLOW	1578.3	1401.7	1967.7
USABLE PROP	1224.2	1052.6	1608.1
ORBITER DRY	153.4	153.4	153.5
TANK DRY	59.3	54.7	66.0
BLOW	4352.6	3171.8	2649.4
USABLE PROP	3512.9	2740.7	2 x 1148.5
BOOSTER DRY	744.5	374.0	2 x 176.2
REL STAGING VEL (FPS)	4000.0	5000.0	4000.0
(T.W)1 / (T / W)2	1.35/0.913	1.35/1.031	1.405/0.905
COSTS (\$ M)			
TOTAL PROGRAM	11,048	10,230	9,816
RD&E	5,803	5,135	4,313
PEAK FUNDING (YR)	1,299 (76)	1,165 (76)	988 (76)
PER FLIGHT (TOTAL)	7.53	7.58	9.49
OPERATIONS	4.65	5.17	3.80
TANKS	1.39	1.30	1.51
EXPND BOOSTER HDWRE	1.49	1.11	4.18

FIGURE 9-2
RECOMMENDED PARALLEL BURN SYSTEM

Twin-156 In. SRMs



15 MARCH 1972

having a terminal sustainer thrust mode.

The Orbiter has a single pad abort rocket (about 32.4 K lb propellant and case weight) which is fired and jettisoned during normal ascent. Installation of this abort rocket adds about 3.2 K lb scar dry weight to the Orbiter. High-q HO tank separation provisions have also been included which add about 1.5 K lb to the Orbiter and 0.5 K lb to the HO tank.

This twin SRM configuration is inherently stable during its ascent mated condition. No thrust vector control is provided in the SRM stage. All mated control is provided by the Orbiter main engines, supplemented by Orbiter aerodynamic surfaces.

9.2 Selection Rationale

Details of relative assessments which lead to the selection of the Twin 156" Parallel Burn System are contained in Section 4.2. Primary areas of concern associated with each of the three major contenders are summarized in Figure 9-3.

Parallel solids are better overall on near-term costs and technical risk considerations. Though SRM per flight costs are more than those of the series burn recoverable systems, many additional SRM flights can be bought with the dollars saved in RDT&E. Recovery of SRM's also appears feasible, and in many respects simpler than recovering the more vulnerable liquid Boosters. Recovery and refurbishment reduces SRM per flight costs by 11% at an increase in RDT&E of only 1%. Combustion products of the large SRM's are of concern relative to their impact on the local launch site environment. Technically this potential risk appears to be small and can be controlled by constraining launches during adverse weather conditions. Malfunction detection in the SRM stage is a concern that is partially obviated by the high reliability and simplicity of solid motor Systems. A considerable background of analysis and design experience has been accrued in the Titan program to enhance safety of flight under similar solid stage operations. This background provides confidence that risks can be avoided or eliminated in this area.

Separation of the Parallel burn SRM's introduces additional complexities; however, preliminary dynamic analyses indicate feasibility. Several alternate devices such as trapeze separation mechanisms and separation rockets can be resorted to if dictated by designs as they mature. Detracting from the Pressure

15 MARCH 1972

FIGURE 9-3

ISSUES OF PRIMARY CONCERN

1-343 A

SERIES BURN PRESSURE FED	SERIES BURN PUMP FED F-1	PARALLEL BURN 156 IN. SOLIDS
• RDT&E COST	• RDT&E COST	• PER FLIGHT COST
• WATER RECOVERY AND RECERTIFICATION	• WATER RECOVERY AND RECERTIFICATION	• POTENTIAL POLLUTION
• NEW ENGINE DEVELOPMENT (POTENTIAL STABILITY PROBLEMS)	• ENGINE WATER IMMERSION EFFECTS AND REFURBISHMENT	• MALFUNCTION DETECTION
	• ENGINE-OUT LOSS OF BOOSTER	• MORE COMPLEX SEPARATION PROVISIONS

Fed System is the need to carry two major engine development programs with their different potential problems. For Booster liquid engines in general, sealing against salt water ingress and protection against water impact damage are considered to be design problems involving considerable risk and potential cost. Added to this is the vulnerability of the four engine F-1 system to losses caused by a single engine failure early in flight.

15 MARCH 1972

1.0 Summary

Two orbiter configurations were used in the Phase B Extension studies for system analyses. The first evolved from the MSC 040A configuration, with modifications primarily to accommodate a change from four J2-S main engines to three 470K HiPc engines. This configuration has the large (15 x 60) payload capability and has received most of the detailed orbiter analysis during this study period. It is referred to as 068B. The second configuration (085A) is the small payload (14 x 45) orbiter, which is essentially the 068B reduced due to the smaller payload bay. It also uses three 470K HiPc engines. It has been examined only to support the system studies.

The discussion in this section is based on a 068B that is compatible with parallel burn twin 156" SRM booster unless otherwise stated. Size and weight comparisons are provided for the smaller orbiter. The minor changes required to be compatible with the series burn boosters are also listed.

Numerous other orbiter configurations and engine combinations have also been examined briefly to provide the parametric data used in system and orbiter trade studies.

In this report, the technical considerations pertinent to overall systems or orbiter configuration selection have been emphasized at the expense of discussing subsystem selections which were not primary factors in system or orbiter decisions during the Phase B Extension.

2.0 GENERAL CHARACTERISTICS

The physical and functional characteristics of the basic 068B orbiter are covered in this section. Brief comparison data are provided on the small orbiter. Issues affecting the overall configuration are discussed under aerodynamics and performance in this section and under the major subsystems in Section 3.

2.1 Physical Characteristics -

The Orbiter general arrangements for the large and small payload bay orbiters are presented in Figures 2.1-1 and 2.1-2, respectively. These evolved from the MSC 040A configuration. Key features of the 068B differing from 040A are listed below.

<u>FEATURE</u>	<u>REASON</u>
o LARGE AREA ELEVONS (.14 S _W)	- IMPROVED HYPERSONIC TRIM CAPABILITY
o FAIRED CANOPY AND INTERNAL MANIPULATOR STOWAGE	- REDUCED HYPERSONIC FLOW INTERFERENCE
o FLARED RUDDER BRAKE	- IMPROVED TRANSONIC/SUPersonic/APPROACH STABILITY AND CONTROL
o FAIRED WING TIP ACPS PODS	- REDUCED DRAG
o HIGH ASPECT RATIO FIN	- LIGHTER STRUCTURAL WEIGHT
o EXTERNAL AIRBREATHING ENGINE PODS	- REDUCED STRUCTURAL WEIGHT AND DEPLOYMENT DYNAMICS
o DIRECT X-AXIS DOCKING	- PILOT CONTROL AND REDUCED STRUCTURAL WEIGHT
o LARGE PRESSURIZED CABIN (2500 FT ³)	- MISSION VERSATILITY
o STRUCTURAL CARGO BAY DOOR	- STRUCTURAL STRAIN COMPATIBILITY AND REDUCED WEIGHT
o SEPARATE ARTICULATED RADIATOR	- MISSION VERSATILITY AND REDUCED STRUCTURAL WEIGHT
o STRAIN-ISOLATED THERMAL PROTECTION	- REDUCED PRODUCTION COSTS AND STRUCTURAL WEIGHT

15 MARCH 1972

FIGURE 2.1-1
ORBITER GENERAL ARRANGEMENT

15 x 60 Payload (068B)

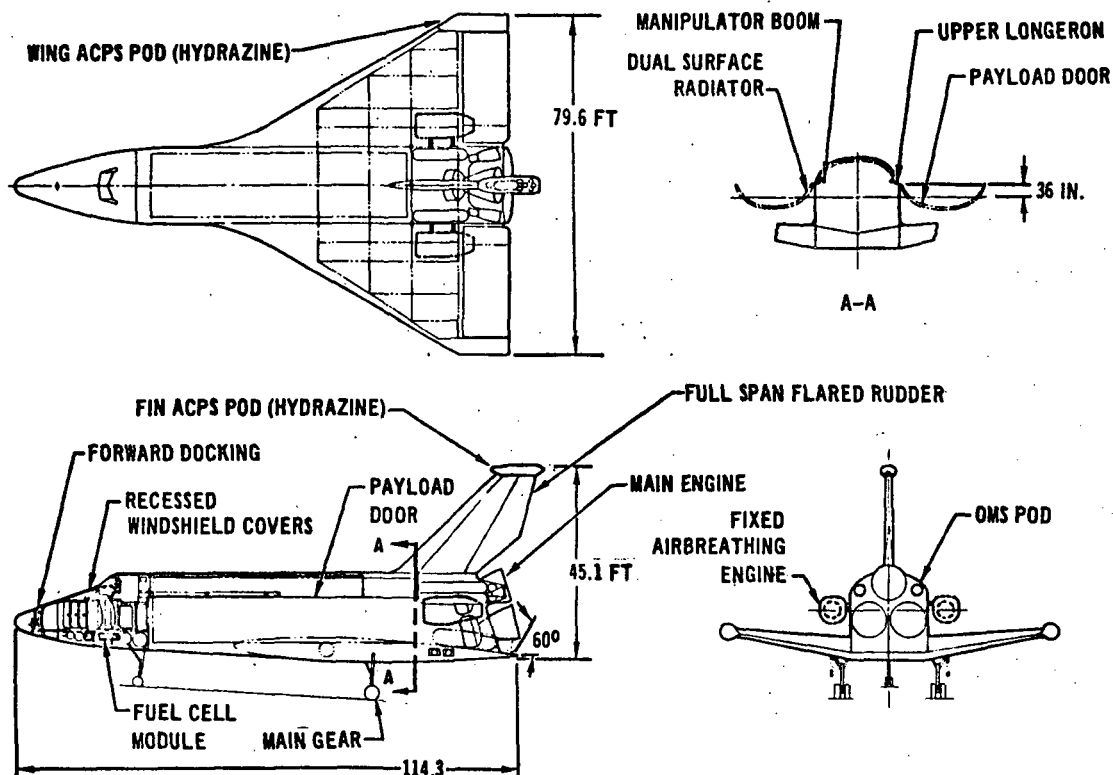
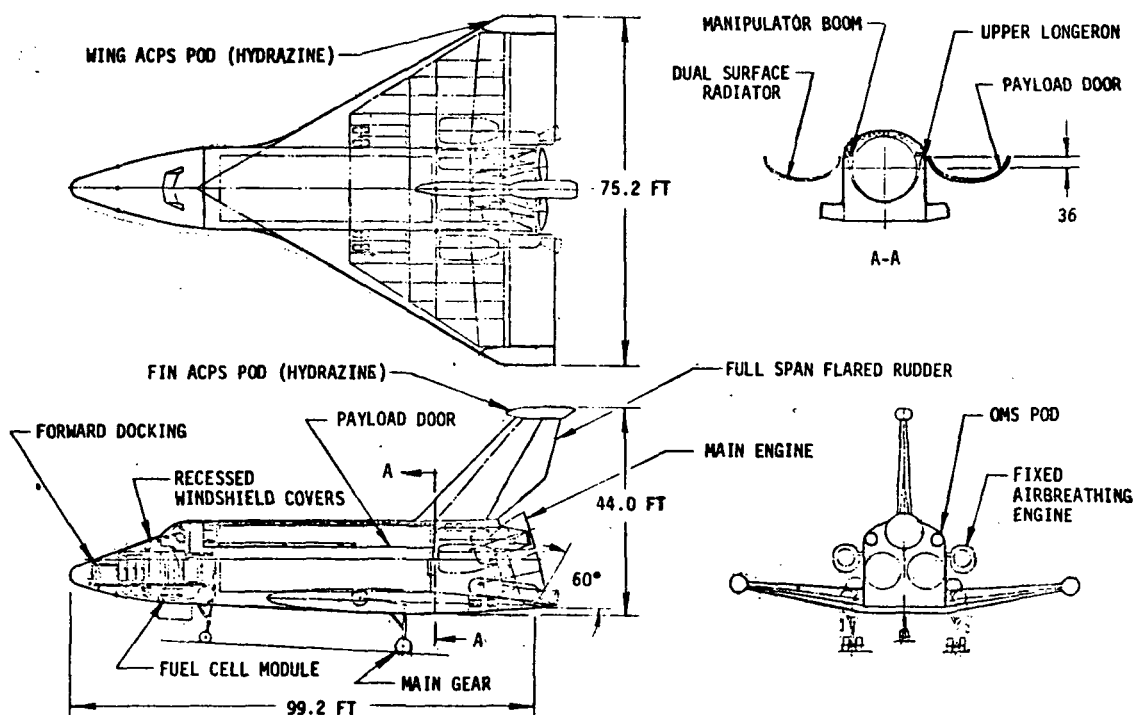


FIGURE 2.1-2
ORBITER GENERAL ARRANGEMENT
14 x 45 PAYLOAD (085A)



15 MARCH 1972

Orbiter geometric data are as follows:

	15 X 60 PL (068B)	14 X 45 PL (085A)
Overall Configuration		
Length - Ft.	123.0	107.8
Height (Less Landing Gear) Ft.	45.1	44.0
Angle, Max Tail Down - Deg.	17	17
Fuselage		
Length, to end of body - Ft.	114.3	99.2
Base Area - Ft ²	314	310
Wing		
Area, Theoretical - Ft ²	3720	3310
Span - Ft.	79.6	75.2
Aspect Ratio	1.7	1.7
Dihedral - Deg.	7	7
Incidence - Deg.	1.5	1.5
L.E. Sweep - Deg.	60	60
Airfoil Section - NACA	0008-64	0008-64
Vertical Tail		
Area, Theoretical - Ft ²	434	418
Aspect Ratio	1.43	1.43
L.E. Sweep - Deg.	45	45
Airfoil Section - NACA	0012-64	0012-64

15 MARCH 1972

2.2 Orbiter Weight Summary

The comparisons of Orbiter weights for the two payload bay sizes are shown below for parallel burn, easterly mission.

BAY SIZE	15 x 60 (3-470 K LB)	14 x 45 (3-470 K LB)	
PAYLOAD WEIGHT REQUIREMENT	65K	45K	ΔWT
WING GROUP	16,214	14,305	-1,909
TAIL GROUP	2,766	2,643	-123
BODY GROUP	28,620	26,120	-2,500
INDUCED ENVIRONMENT PROTECTION	25,602	22,771	-2,831
LANDING, RECOVERY, DOCKING	7,977	7,249	-728
PROPULSION-ASCENT	23,000	23,000	
PROPULSION-CRUISE	0	0	
PROPULSION-AUXILIARY	6,483	6,261	-222
PRIME POWER	4,020	4,020	
ELECT. CONVER. AND DISTRIBUTION	2,530	2,530	
HYDRAULIC CONVER. AND DISTRIBUTION	1,510	1,510	
SURFACE CONTROLS	4,997	4,525	-472
AVIONICS	5,730	5,730	
ENVIRONMENTAL CONTROL	6,550	6,550	
PERSONNEL PROVISIONS	800	800	
RANGE SAFETY AND ABORT	0	0	
BALLAST	0	0	
GROWTH UNCERTAINTY	12,008	11,128	-880
SUBTOTAL (DRY WEIGHT)	(148,807)	(139,142)	(-9,665)
PERSONNEL	1,200	1,200	
CARGO	65,000	45,000	-20,000
ORDNANCE	0	0	
RESIDUAL FLUIDS	2,156	2,156	
SUBTOTAL (INERT WEIGHT)	(217,163)	(187,498)	(-29,665)
RESERVE FLUIDS	13,915	12,472	-1,443
INFLIGHT LOSSES	5,443	5,443	
PROPELLANT-ASCENT	0	0	
PROPELLANT-CRUISE	0	0	
PROPELLANT-MANEUVERS/ACS	15,501	14,290	-1,211
TOTAL (GROSS-WEIGHT) LB	(252,022)	(219,703)	(-32,319)
ENTRY WEIGHT - LB	197,710	173,045	(-24,665)
LANDING WEIGHT - LB	192,659	167,994	(-24,665)
LANDED CARGO	40,000	25,000	(-15,000)

Orbiter modifications for the series burn configuration impose minor changes in system weight as listed below:

15 MARCH 1972

	<u>Lbs</u>
Propulsion Feed System	
Add Recirculation System	+750
Plumbing	+150
APU	
Shorter Ascent Time	-100
Base Heating	
Lower Aft End Heating	-120
Precant and Gimbal Angle Change	
Reduced Body Wetted Area	-490
Interconnect Backup	- 40
Thrust Structure	-100
Vertical Tail	
Reduced Rudder Loads	-400
Wing	
Increased Down Loads	+100
Fuselage	
Increased Down Bending	+300
Uncertainty/Growth Maintained at 10%	+ 5
Net Effect	+ 55

15 MARCH 1972

2.3 Aerodynamics and Performance

Data of the 040A configuration generic wind tunnel tests were used as the basis of estimating the aerodynamic characteristics and flight performance of the 040A, 068B, and 085A configurations.

Current aerodynamic design objectives are defined in Figure 2.3-1. The longitudinal stability and control summary for the 040A configuration is shown in Figure 2.3-2. The flared rudder schedule ($2.5 < M < 7$, $\pm 30^\circ$; $1.2 < M < 2.5$, $\pm 20^\circ$; $M < 1.2$, $\pm 5^\circ$) was selected to provide maximum longitudinal stability. The nominal angle of attack provides acceptable flight characteristics, including sonic boom and elevon hinge moments, through the angle of attack transition from the high angle required at hypersonic speeds to the airplane mode at transonic speeds. The $\pm 5^\circ$ off-nominal was considered as desirable for instrumentation, and manual control. The 2% static margin at landing speeds at the aft center-of-gravity is minimal to obtain Level III longitudinal short period handling qualities. The available aerodynamics capability for the 040A with the nominal angle of attack band is limited to a forward center of gravity of approximately 65% of the body length and to an aft center of gravity of 66%. As noted in the mass properties description, the center of gravity travel with 25,000 lb payload in/out exceeds the aerodynamic capability.

NASA directed changes to the payload landing weight and resulting engine, structure, etc. changes required configuration resizing, resulting in a larger wing and tail to maintain the 150 kt design speed and adequate directional stability. Additional configuration changes included;

- (a) smaller ACPS pods
- (b) incorporation of the OMS pods in the base, and
- (c) a longer, wider body.

The longitudinal characteristics of the resulting configuration, 068B, are shown in Figure 2.3-3. The trim lift and drag characteristics at subsonic and hypersonic speeds are given in Figure 2.3-4. The aerodynamic capability is deficient relative to the forward and aft center of gravity. Aerodynamic modifications which have been evaluated to provide an adequate aerodynamic corridor of approximately 3% l_B are:

- (a) nose change - broader planform, trapezoidal cross-section, more ramp angle
- (b) body flap - movable, trailing edge up.

CURRENT AERODYNAMIC DESIGN OBJECTIVES

HYPERSONIC ENTRY: L/D COMMENSURATE WITH ONCE AROUND CROSSRANGE REQUIREMENT
 α TRIM $\pm 5^\circ$ OR $\pm 20\%$ OF $(L/D)_{NOM}$
 $-30^\circ \leq \delta_e \leq 10^\circ$ TO PROVIDE AILERON CONTROL FOR OPERATING CG'S INCLUDING $\Delta y \leq 3''$
 SAFE AND CONTROLLABLE (AT LEAST $C_{\eta\beta DYN} > 0$, $C_{m\alpha} < 0$)

TRANSITION: ADEQUATE UNAUGMENTED HANDLING QUALITIES (AT LEAST $C_{\eta\beta DYN} > 0$ AND
 LONGITUDINAL STABILITY $> 1\% I_B$)

AIRPLANE MODE: LEVEL 3 HANDLING QUALITIES UNAUGMENTED (AT LEAST)

APPROACH AND LANDING* (GEAR DOWN, HALF SPEED BRAKES, 40K AT P/L CTR)	V KTS	α	GROUND EFFECTS
TOUCHDOWN SPEED	150	- STABLE & CONTROLLABLE - SCRAPE - BELOW STALL	NO
	175	- $\eta \geq 0.4g$	YES
APPROACH SPEED (AVERAGE)	210	- FOR L/D_{MAX}	NO

*MINIMUM $(L/D)_{MAX} = 3.0$ (FULL SPEED BRAKE, GEAR DOWN)

FIGURE 2.3-1

STABILITY AND CONTROL SUMMARY 040A

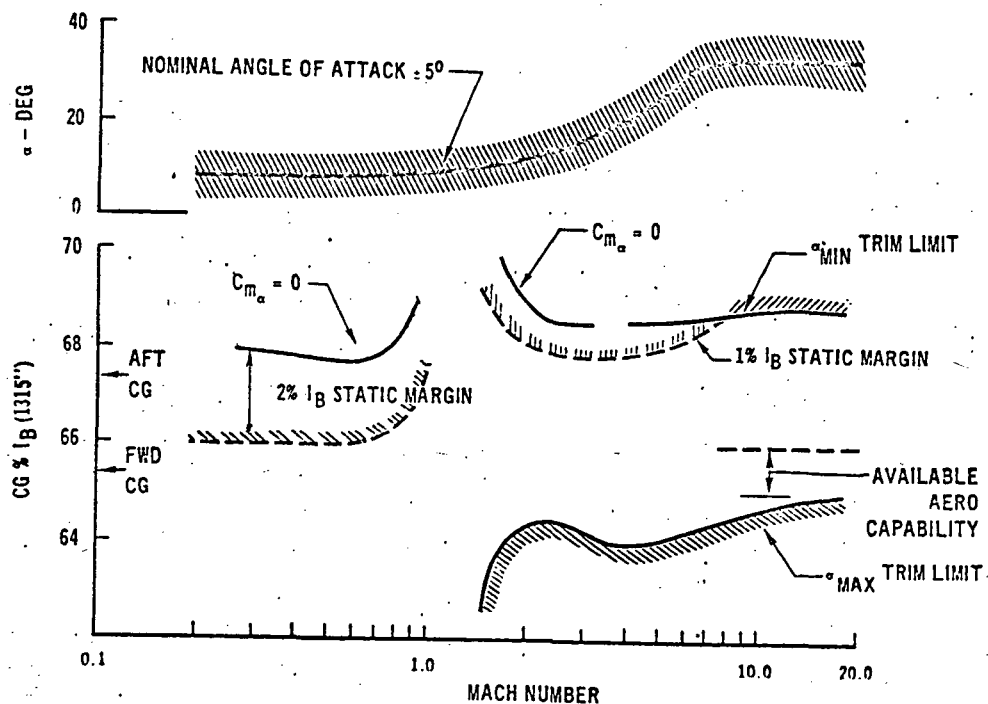


FIGURE 2.3-2

STABILITY AND CONTROL SUMMARY 068B

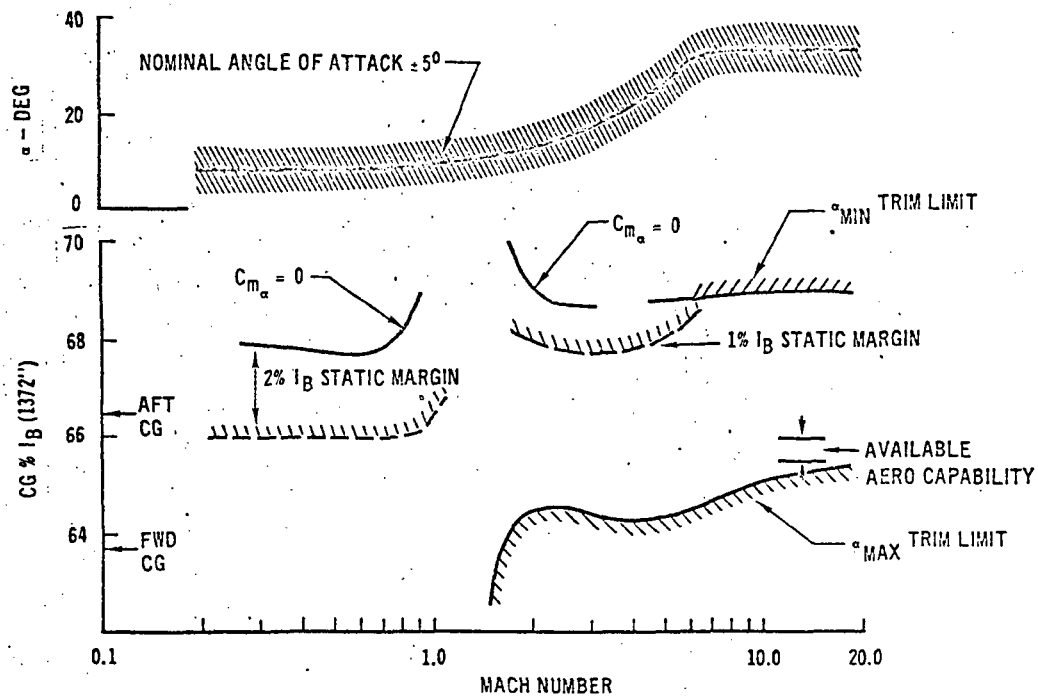


FIGURE 2.3-3

DELTA WING ORBITER 068B TRIM AERODYNAMIC CHARACTERISTICS

$X_{CG} = 65\% \bar{c}$
 $I_{CG} = 1372 \text{ in}^2$
 $\rho_{REF} = 3720 \text{ ft}^2$

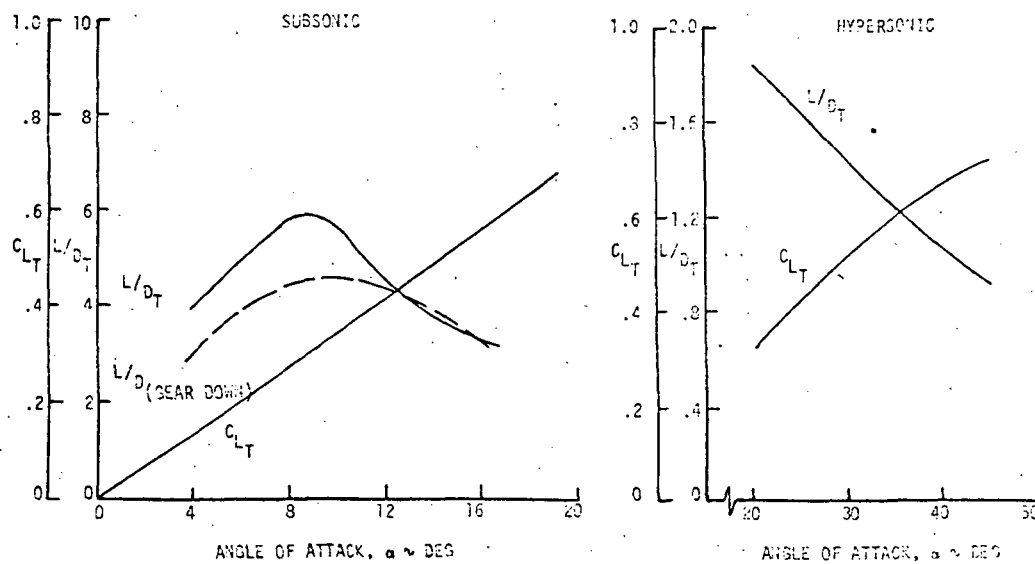


FIGURE 2.3-4

15 MARCH 1972

Depending on possible revision of criteria for handling qualities and landing speed, additional studies are required to define the resulting mass properties, aerodynamic characteristics and performance capability.

Aerodynamic analyses during this period also included evaluation of a 14 x 45 foot payload compartment design. The resultant configuration (085A) was sized in like manner to the baseline configuration. The foreshortened fuselage reduced the corridor available with the incorporated aerodynamic improvements from 3% l_B to 2.5% l_B . However, the shorter length and reduced payload results in a payload in/out center of gravity requirement of only 1.5% l_B , thereby resulting in an adequate corridor margin.

Lateral-directional characteristics of the 040A were determined through wind tunnel testing, Figure 2.3-5. At hypersonic speeds the requirement for positive $C_{n\beta}$ dynamic is met by the wing dihedral at the nominal high angle of attack. Control is obtained from the ACPS until the airplane mode is reached. At subsonic speeds the vertical tail was sized for 068B to obtain a $C_{n\beta}$ of .001. For this configuration the dutch roll characteristics will meet Level II handling qualities.

Performance characteristics of the Orbiter are summarized in Figures 2.3-6 and 2.3-7. The most severe entry, in terms of the heating environment, is the maximum crossrange entry from a 100 NM polar orbit with a 40,000 lb payload. The entry is initiated with a 250 ft/sec retro maneuver. During retro burn, the vehicle is held in a 49 degree nose down attitude for minimum range sensitivity. Initial entry is made with a 34 degree angle of attack and a 70 degree bank angle. After pullout, bank angle is modulated to suppress skip and to accomplish the required crossrange and downrange maneuvers while maintaining the angle of attack constant at 34 degrees. After decelerating to Mach 7, a gradual transition to low angle of attack is initiated. The velocity-altitude profile of this entry is shown in Figure 2.3-6.

Orbiter subsonic performance is summarized in Figure 2.3-7. Air breathing engines are not installed for the nominal polar missions and landings are performed utilizing the high energy approach technique. Speed control is provided by rudder flare modulation. Touchdown is made at an angle of attack of 12.2 degrees, slightly above that for maximum L/D with partial speed brakes. A maneuver load factor of over 0.4g is attainable at touchdown.

For the orbital missions having 2 ABES installed, the thrust is not sufficient to maintain a level flight cruise. A glide extension of 55 NM is attainable with

15 MARCH 1972

ORBITER LATERAL - DIRECTIONAL CHARACTERISTICS

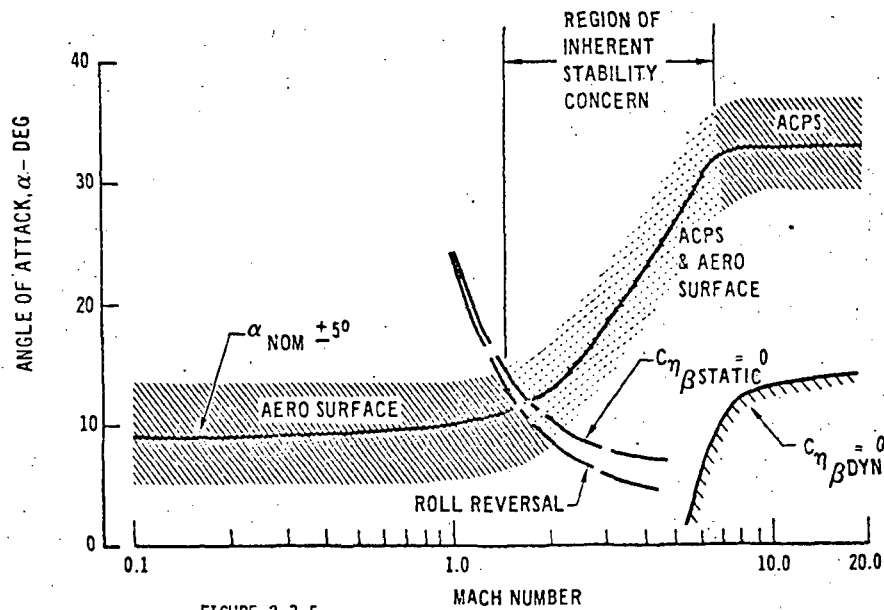


FIGURE 2.3-5

ORBITER ENTRY TRAJECTORY

068B CONFIGURATION

100 NM POLAR ORBIT
 V_0 INERTIAL = 25,655 FT/SEC
 γ_0 INERTIAL = -0.83 DEG
 h_0 = 400,000 FT
 γ = 90 DEG

1100 NM CROSS RANGE

$\alpha = 34^\circ$

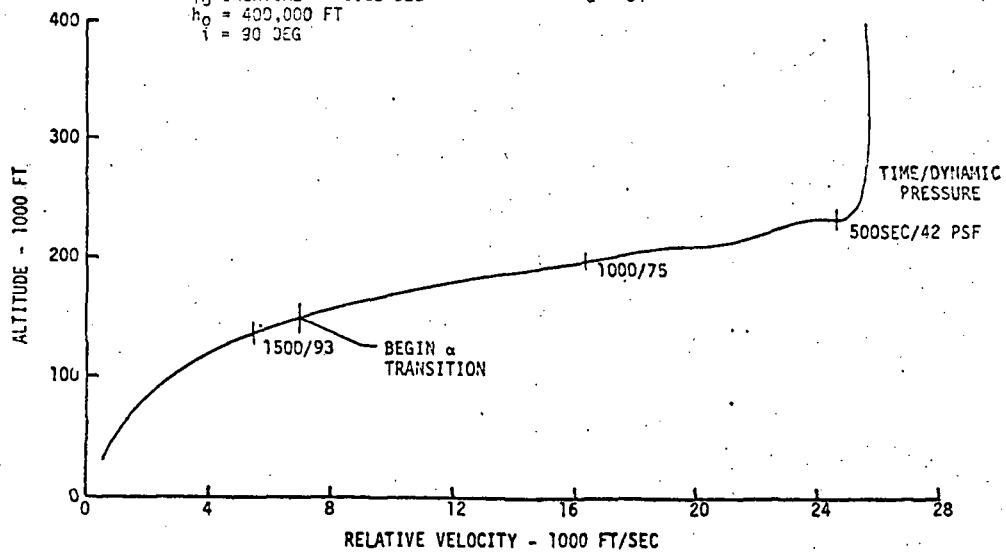


FIGURE 2.3-6

ORBITER PERFORMANCE CHARACTERISTICS

Entry (100 NM Polar Orbit)

Angle of Attack - Deg	34
Crossrange - NM	1100
Initial transition - Mach No.	7
Time - 400,000 ft to 50,000 ft, minutes	32

Subsonic - Unpowered

Design Landing Weight - lb	192659
Approach Speed - Kts	245
Landing speed - st'd day - sea level, kt	175
Touchdown angle of attack - deg	12.2
Field Length - FAA Dry - ft	9550

Subsonic - Powered (Hot Day)

Glide Extension capability - 2 Engines - NM	55
Fuel Required - lb	4500
Glide Extension capability - Engine out - NM	15
Fuel Required - lb	1100

Ferry Mission (Hot Day - 4 Engines and JATO)

Take-off weight - lb	208000
Field Elevation - ft	4000
Balanced field length - ft	13000
Initial cruise altitude - all engines - ft	15500
Initial cruise altitude - engine out - ft	6400
Average cruise speed - all engines - kts	316
Design cruise range - NM	440

FIGURE 2.3-7

15 MARCH 1972

this installation, however. About 4500 lb of fuel are required for maximum extension.

Ferry mission performance requires 4 airbreathing engines. In addition, RATO units are needed to meet the FAA balanced field requirements in case of an engine failure during take-off. With current ferry configuration weights, engine out cruise performance is marginal. A ferry kit providing wing tip extensions and a base fairing to increase the cruise L/D would provide improved ferry performance.

2.4 Flight Control Characteristics

Flight control characteristics are summarized in Figure 2.4-1. The following paragraph briefly discuss the way the control system is utilized along with the guidance and navigation systems to provide spacecraft mission flight control.

Ascent - The ascent guidance utilizes information from the inertial navigation system and explicit closed loop guidance equations to calculate a constant rate steering profile that will achieve the desired end conditions. The desired end conditions are defined in terms of final radius, velocity magnitude, radial velocity and out-of-plane velocity, and position coordinates. Attitude error commands from the guidance are converted to rate commands and summed with the corresponding rates to form control commands to the engine gimbal logic.

On-orbit - The guidance system calculates the required transfer maneuvers, rendezvous sequence maneuvers, landing site selection, and the turning for initiation of deorbit maneuvers. The guidance utilizes computer state vector information which is updated by the ground network and star tracker and horizon sensor measurements. The relative position and velocity of the Shuttle and the target are updated by S-band ranging. Attitude control is provided by ACPS thrusters in response to automatic and manual commands. ΔV maneuvers are performed with the OMS engines.

Reentry - Following the deorbit maneuver and prior to atmospheric encounter, the vehicle is controlled in an attitude hold mode utilizing the ACPS. Guidance and active bank and angle-of-attack control are initiated at .05 g's and provided throughout the atmospheric flight phase. The guidance is targeted for a point approximately 750 nm from the runway and utilizes closed form range predictions to obtain the required L/D. Navigation is provided by the onboard inertial system with vertical channel stabilization. Elevons and jets are employed for pitch axis control with the jets deactivated at 40 psf. All reaction control is utilized for the lateral-directional channel. Differential elevons provide trim control for

FLIGHT CONTROL CHARACTERISTICS														
MISSION PHASE	CONTROL DESCRIPTION	CONTROL CHARACTERISTICS												
ASCENT	- SPACECRAFT AND AIRCRAFT SYSTEMS - AUTOMATIC RATE COMMAND SYSTEM - RATE GYRO FEEDBACK IN SPACECRAFT FLIGHT CONTROL ELECTRONICS - ATTITUDE FEEDBACK THROUGH ASCENT STEERING EQUATIONS	- MAIN ENGINE GIMBAL REQUIREMENTS ± 8.0 DEG PITCH ± 6.5 DEG YAW - TORQUE CAPABILITY PITCH $7.18 \sqrt{\frac{FT-LB}{DEG}}$ (MIN) ROLL $1.25 \sqrt{\frac{FT-LB}{DEG}}$ (MIN) YAW $4.79 \sqrt{\frac{FT-LB}{DEG}}$ (MIN) - DIFFERENTIAL ELEVON AND RUDDER FOR ROLL RATE DAMPING												
ON-ORBIT	- SPACECRAFT SYSTEMS ONLY - REACTION JET CONTROL OF ATTITUDE AND TRANSLATION - THREE AXIS ATTITUDE AND TRANSLATION CONTROL HANDLES - ATTITUDE CONTROL MODES: MANUAL - ACCELERATION, PULSE, RATE COMMAND/ATTITUDE HOLD AUTOMATIC - ATTITUDE HOLD - TRANSLATION CONTROL MODES: MANUAL - ACCELERATION AUTOMATIC - ΔV INCREMENT - MAJOR ON-ORBIT TRANSLATION MANEUVERS PERFORMED USING OMS; ACPS THRUSTERS PROVIDE TRANSLATION CONTROL DURING DOCKING - DERIVED ATTITUDE RATES	32 ACPS THRUSTERS: 400 LBS. THRUST LEVEL 2 WING PODS WITH 13 THRUSTERS EACH 1 TAIL POD WITH 6 THRUSTERS - ACPS ANGULAR ACCELERATION (NOMINAL) AXIS ANG. ACCEL. (DEG/SEC²) ROLL ± 1.15 PITCH ± 0.20 YAW ± 0.23 - ACPS TRANSLATIONAL ACCELERATION (NOMINAL) DIRECTION ACCELERATION (FT/SEC²) $\pm X$ ± 0.38 $\pm Y$ ± 0.19 $\pm Z$ ± 0.22 - OMS CHARACTERISTICS - TWO LM ASCENT ENGINES - THRUST 3500 LBS. EACH - DEORBIT T/W ≈ 0.035												
ENTRY	- ACPS LATERAL-DIRECTIONAL CONTROL TO COMPLETION OF TRANSITION (M = 2) - DIFFERENTIAL ELEVONS PROVIDE TRIM CONTROL THROUGHOUT ENTRY - ACTIVE RUDDER AND AILERON CONTROL INITIATED AT TRANSITION - BLENDED ELEVON/REACTION JET PITCH AXIS CONTROL. JETS TURNED OFF AT 40 PSF DYNAMIC PRESSURE. - ACPS MANUAL CONTROL VIA THREE AXIS SIDEARM CONTROLLER - AERO MANUAL CONTROL VIA TWO AXIS SIDEARM CONTROLLER AND RUDDER PEDALS	- ACPS ANGULAR ACCELERATION CAPABILITY AXIS ANG. ACCEL. (DEG/SEC²) ROLL ± 1.15 PITCH ± 0.43 YAW ± 1.05 10 DEGREE DIFFERENTIAL ELEVON CAPABILITY FOR LATERAL - DIRECTIONAL TRIM CONTROL												
TERMINAL APPROACH TO LANDING	- ALL AERODYNAMIC CONTROL - AERODYNAMIC SURFACES USED: ELEVONS, RUDDER, FLARED RUDDER - MANUAL CONTROL VIA TWO AXIS SIDEARM CONTROLLER AND RUDDER PEDALS - RATE COMMAND - BANK COMMAND - FLIGHT PATH ANGLE COMMAND - AUTOMATIC CONTROL MODES - ANGLE OF ATTACK CONTROL - BANK ANGLE CONTROL - GLIDESLOPE AND LOCALIZER HOLD - ALTITUDE HOLD - HEADING HOLD	CONTROL SURFACE RATE REQUIREMENTS-DEG/SEC <table> <thead> <tr> <th></th><th>SUBSONIC</th><th>SUPERSONIC</th></tr> </thead> <tbody> <tr> <td>ELEVONS</td><td>30</td><td>15</td></tr> <tr> <td>RUDDER (CONTROL)</td><td>30</td><td>15</td></tr> <tr> <td>RUDDER (FLARE)</td><td>10</td><td>10</td></tr> </tbody> </table>		SUBSONIC	SUPERSONIC	ELEVONS	30	15	RUDDER (CONTROL)	30	15	RUDDER (FLARE)	10	10
	SUBSONIC	SUPERSONIC												
ELEVONS	30	15												
RUDDER (CONTROL)	30	15												
RUDDER (FLARE)	10	10												

FIGURE 2.4-1

15 MARCH 1972

offset c.g.'s. Bank angle orientation is achieved by commanding a maneuver about the yaw body axis.

Transition - During the transition phase, the vehicle is maneuvered from the back side to the front side of the L/D curve. The vehicle will respond to commands from the terminal guidance system during the pitch over maneuvers. Navigation updates from TACAN stations will start at Mach 7.0. The pitch control system is unchanged for the transition and subsequent flight phase. However, the lateral-directional control concept is changed just prior to transition from all reaction to blended aero/reaction control. Bank control is performed by commands to the roll axis.

Terminal Phase - Energy management during this phase is accomplished through angle-of-attack, bank angle, and speed brake modulation. The guidance is initiated at the beginning of transition and is divided into the following phases: (1) cruise or energy dissipation; (2) acquisition of the heading alignment circle; and (3) heading alignment. The guidance logic switches from phase to phase on the basis of energy. The onboard inertial navigation system will be augmented by mathematically filtering TACAN and barometric altitude data. The vehicle is controlled entirely with aerodynamic controls during this phase of flight. Manual pitch and roll commands are introduced through a two axis control stick and yaw commands with the rudder pedals. Automatic turn coordination is provided to minimize the requirement for primary control inputs into the yaw axis.

Automatic speed brake modulation provides the essential speed control for the autoland system. The autoland system utilizes the two step approach path which combines the high energy approach with the conventional final approach. This allows for standard obstacle clearance and a final flare which is quite similar to the final approach of present large transport aircraft.

15 MARCH 1972

3.0 SUBSYSTEMS

The subsystems are reported in depth here only where they have strongly influenced the system or orbiter decisions, or where key issues have occurred during the Phase B Extension. Some of the trade studies reported used data on MSC configuration 040A. However, the data are directly applicable or qualifications stated. Additional descriptive detail is in the data book on all subsystems.

3.1 Airframe

This section covers the basic airframe and miscellaneous mechanical systems. The structure and TPS are reported in some depth because they have changed considerably during Phase B Extension, primarily as a result of the new booster concepts.

3.1.1 Structures - This section describes the structural concepts selected for the Space Shuttle Orbiter (Ref. Dwg. SK-ET-0068B) as a result of studies conducted during the Phase B Extension period. Substantiating data for the selection of specific structural concepts and materials is presented in the form of trade studies which have been performed.

The overall philosophy was to achieve minimum total program costs commensurate with structural efficiency and minimum technical risk. Therefore, conventional aircraft design and construction techniques have been recommended for the most part.

The primary structure selected for the Orbiter is an all-aluminum stiffened shell airplane-type structure covered with a combination of non-metallic reusable surface insulation and ablative TPS to form the aerodynamic moldline surface (see Section 3.1.2 for TPS description). Structural arrangement and definition drawings are shown in Figures 3.1.1-1 and 3.1.1-2.

The selection of aluminum for the primary structure was based on a trade study which compared aluminum and titanium structure, and their associated TPS, on the basis of weight, cost, and performance. Then, the Aluminum Alloy Selection Study (Ref. Section 3.1.1.1) showed 2024 aluminum alloy to be the best choice on the bases of strength efficiency, formability, and stress corrosion resistance. This study also showed that the best alloys for forged and welded parts were 7075 and 2219 aluminums, respectively.

FIGURE 3.1.1-1
SPACE SHUTTLE
ORBITER STRUCTURAL ARRANGEMENT

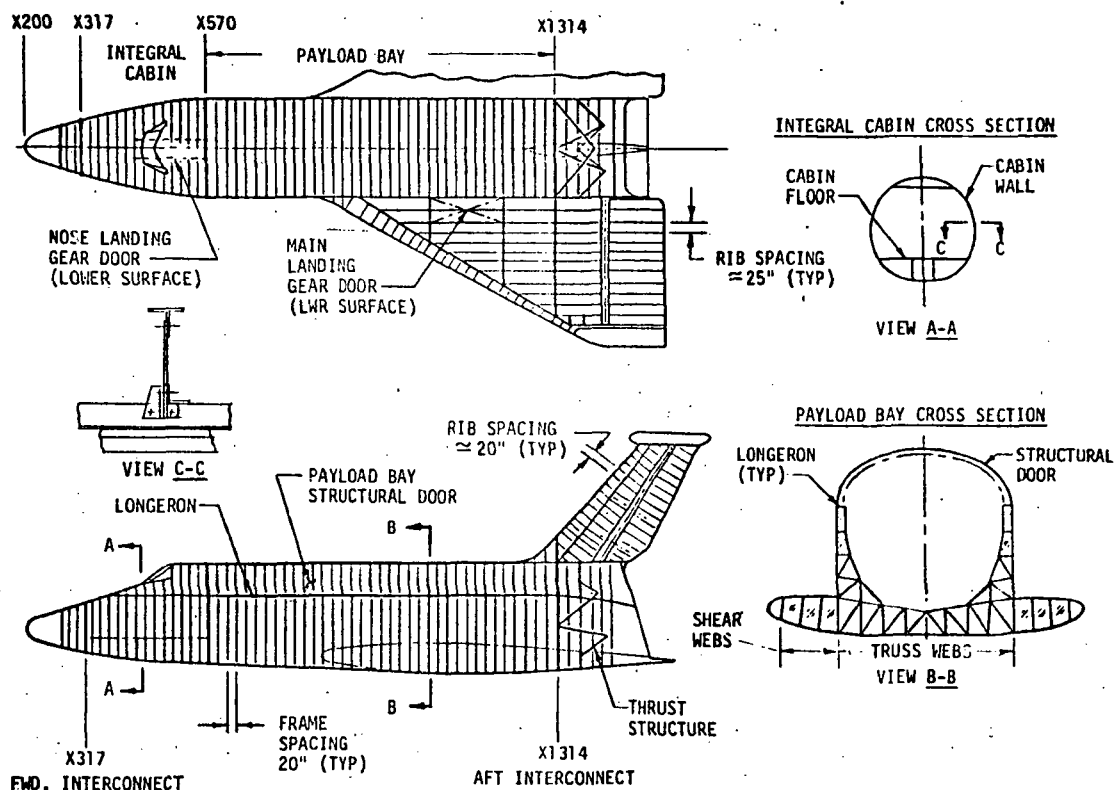
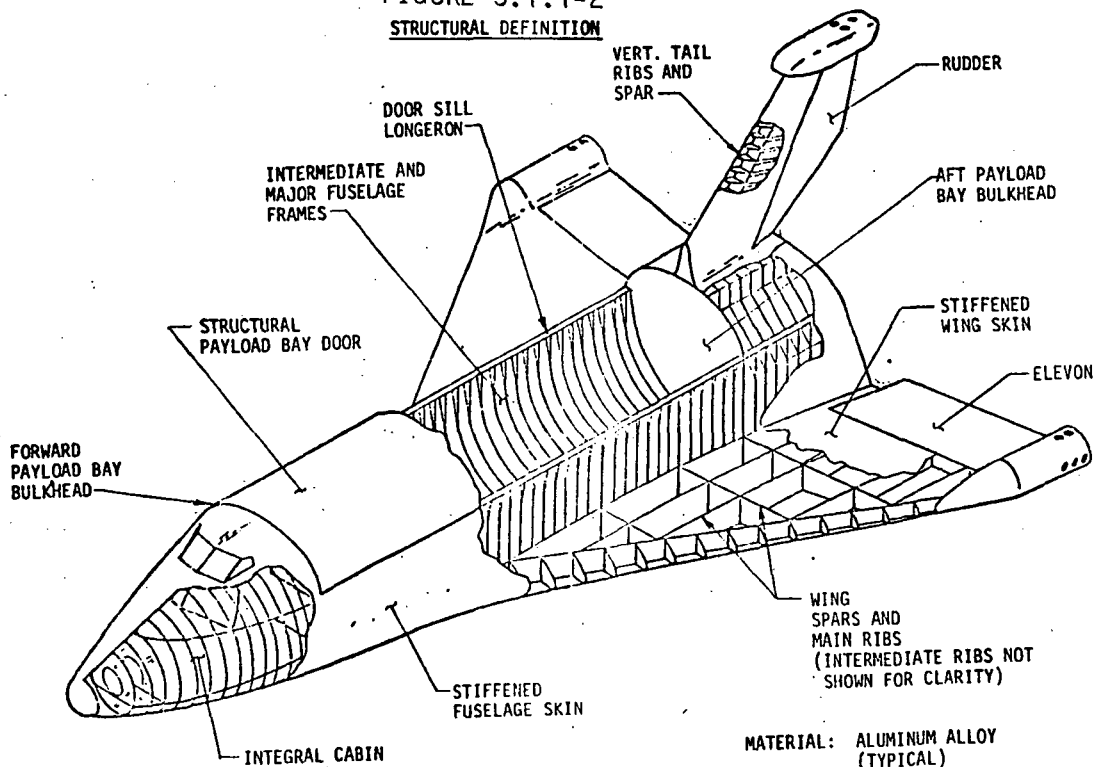


FIGURE 3.1.1-2
STRUCTURAL DEFINITION



15 MARCH 1972

The basic fuselage shell is formed by zee-stiffened skins, protected by TPS which forms the aerodynamic moldline. Fuselage frames, which react pressure loads from the skins and prevent general instability failure of the shell, are spaced 20 inches apart (based on the Frame Spacing Optimization Study, Ref. Section 3.1.1.2). The stiffened skins are permitted to buckle elastically under shear and axial compression loads, based on results of a study comparing the integrated structural and TPS weights of buckling and non-buckling skin concepts (Ref. Section 3.1.1.3). TPS is strain-isolated from the skin and protected from skin buckle deflections by a layer of silicon sponge rubber.

The forward fuselage consists of a pressurized integral cabin structure formed by the stiffened moldline fuselage shell and the cabin floor. Internal frames, spaced approximately 20 inches apart, are designed to prevent general instability failure of the stiffened shell and to carry moments due to pressure differential acting on the non-circular cabin enclosure. The moldline integral cabin concept is tentatively selected as the baseline. However, final concept selection will be determined from future studies comparing designs on the basis of minimum cost and minimum cabin gas leakage (Ref. Section 3.1.1.4). Other major items in the forward fuselage are the nose cap, docking port, forward interconnect, and nose landing gear.

Contained in the center fuselage are the payload bay, payload attach points and handling mechanisms, and wing carry-through structure. Overall vehicle loads during ascent are carried by the fuselage sides and bottom, acting together with a one-piece structural payload bay door. The fuselage is capable of carrying vehicle entry and post-entry loads without the door being latched, in the event of a door malfunction. The design is based on a study which evaluated both structural and non-structural payload doors (Ref. Section 3.1.1.5). The significant increase in overall stiffness offered by the structural door was a major influence in its selection.

The aft fuselage is an extension of the basic fuselage shell and contains the boost engine thrust structure. It consists of major frames and the payload bay aft bulkhead, which redistributes loads from the vertical tail, wing, thrust structure, aft interconnect, OMS pods, and ABES. An upper shelf beam distributes thrust loads from the upper engine into the fuselage shell structure.

15 MARCH 1972

The boost engine thrust structure consists of thrust struts and a support beam gridwork. The upper engine is supported by a tripod truss which transmits the thrust loads to the upper shelf beam and the vertical support beams. Lower engine loads are reacted by vertical support beams and a set of lateral stabilizing truss struts.

The exposed portion of the Orbiter wing is a four-spar torque-box structure with ribs at approximately 25 inch spacing and angle-stiffened cover skins. Aluminum angle stiffeners were selected over 'zees' because they resulted in a lower weight when forced crippling due to diagonal tension effects were considered. The rib spacing is based on the results of an earlier study but is considered applicable for this configuration. This conventional design concept, using distributed wing bending material was selected over a concept employing discrete spars and ribs with floating skin panels because of greater confidence in the conventional concept and the resulting weight estimation (Ref. Section 3.1.1.6).

Wing carry-through structure located under the payload compartment is integrated with the fuselage lower shell structure and major fuselage frames. The fuselage shell structure in this area (lower surface between spars X 1196 and X 1448) is composed of integrally stiffened (waffle) skin panels and was selected to satisfy the requirement for carrying both longitudinal fuselage loads and transverse wing carry-through loads. Wing carry-through spar webs are truss type and are integrated into the lower portion of major fuselage frames; wing carry-through upper spar caps are concentrated members which shear lag their loads into distributed bending material on the upper surface of the exposed wing.

The vertical tail is a one cell, two spar torque box employing zee-stiffened skin for distributed bending material. The load in the distributed bending material is shear lagged into two spars at the fin root which are aligned with major aft fuselage frames.

Interconnect structure between the Orbiter and the external tank is provided at two locations, one forward and one aft, each consisting of three attachment points. The forward interconnect consists of a center attach point which reacts lateral loads only, and two outboard attach points which react vertical and lateral loads from outboard attach struts. A major bulkhead at the front end of the cabin enclosure (X 317) distributes these concentrated loads into the forward fuselage.

15 MARCH 1972

The interconnect consists of a centerline attach point which reacts longitudinal and lateral loads, and two outboard attach points which react vertical and lateral loads from outboard attach struts. A major bulkhead at the aft end of the payload bay (X 1314) plus a lower drag beam introduce these loads into the aft fuselage.

3.1.1.1 Aluminum Alloy Selection - Candidate aluminum alloys and associated tempers were evaluated early in the extension study. Selected high weight efficient alloys were incorporated into the detailed structural sizings. Other alloys and/or tempers may be necessary for the solution of specific detail design considerations.

Alloy and Temper Selection - The following aluminum alloys and tempers are recommended for general usage on orbiter structure. All of these materials have good stress corrosion resistance and good strength characteristics at elevated temperatures.

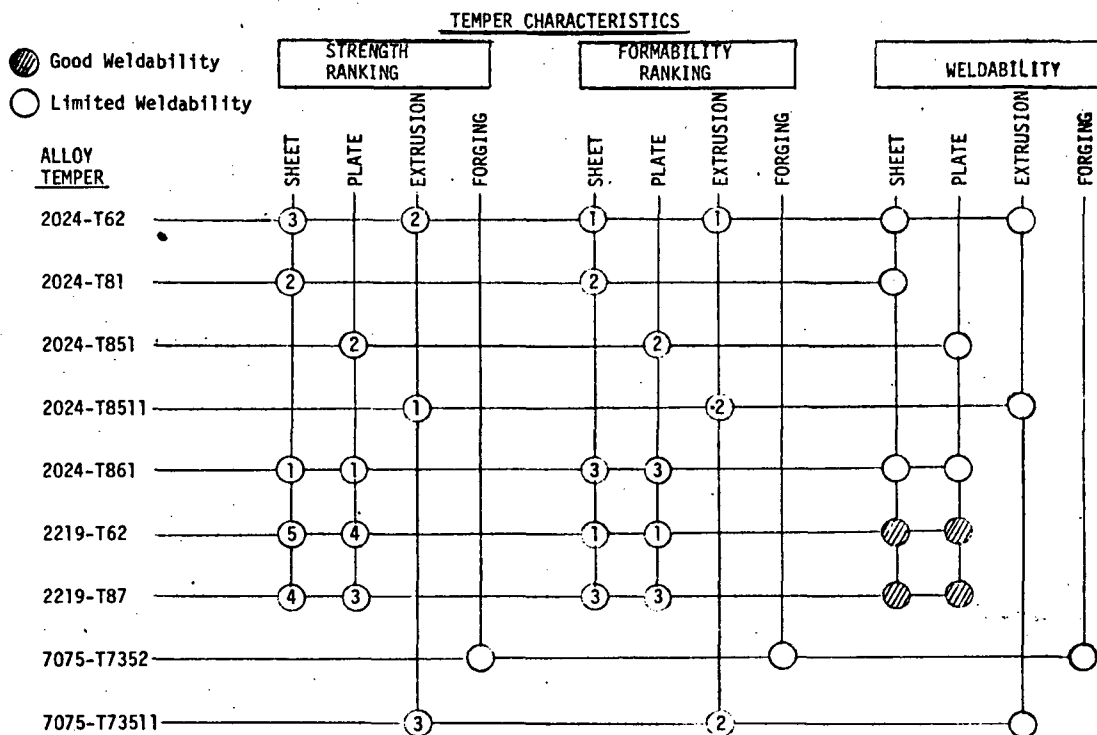
Mill Product	Alloy	Temper	Application
Sheet	2024	-T861	Flat or Slight Forming
		-T81	Moderate Forming
		-T62	Severe Forming
	2219	-T87	Welded Parts; Flat to Moderate Forming
		-T62	Welded Parts; Severe Forming
Extrusions	2024	-T8511	Slight to Moderate Forming
		-T62	Severe Forming
	7075	-T73511	Tension Members in Cool Regions
Plate	2024	-T851	Plate Elements
Forging	7075	-T7352	Forged Elements

Figure 3.1.1-3 rates the tempers of selected alloys with respect to strength, formability, and weldability. T8 tempers have the greatest strength while T6 tempers are easier to form.

The 2219 alloy is generally weldable by all commercial procedures and methods and exhibits good resistance to stress corrosion cracking at the weld. The 2024

15 MARCH 1972

FIGURE 3.1.1-3



and 7075 alloys have limited fusion weldability because of increased crack sensitivity, loss of resistance to corrosion, or degradation of mechanical properties.

Strength Efficiency Comparison at Elevated Temperatures - Strength efficiency curves for selected sheet, plate, and extruded shapes are shown in Figures 3.1.1-4 through 3.1.1-7. Fig 3.1.1-8 presents a comparison of the material crippling index efficiency for consideration in members designed by compression loading. At temperatures up to 300°F, the 2024-T861 material exhibits an efficiency advantage from 4.5% to 10.8% over other selected materials. The 2219 alloys become advantageous only at temperatures beyond 450°F, which is beyond its maximum use temperature. Strength data for the above figures were taken from ALCOA test results which classified tensile properties as being typical.

Temperature Use Limits - Major considerations for establishing temperature use limits for aluminum alloys are creep strength and reduction of mechanical properties at room temperature. The temperature range at which a 0.2 percent creep strain occurs (often used as a criteria for temperature use limits) under a 3000 psi stress level after 100 hours exposure is from 575°F to 625°F. But the severe degradation of mechanical properties at these temperature levels makes this criteria unacceptable for establishing upper limits.

Table 3.1.1-19 lists maximum-use temperatures for selected alloys arbitrarily based on either F_{tu} or F_{ty} being degraded by 10.0 percent.

15 MARCH 1972

FIGURE 3.1.1-4
ALUMINUM SHEET, PLATE

Based on ALCOA Test Data
Typical Tensile Props.

AT ROOM TEMPERATURE AFTER 100 HRS EXPOSURE

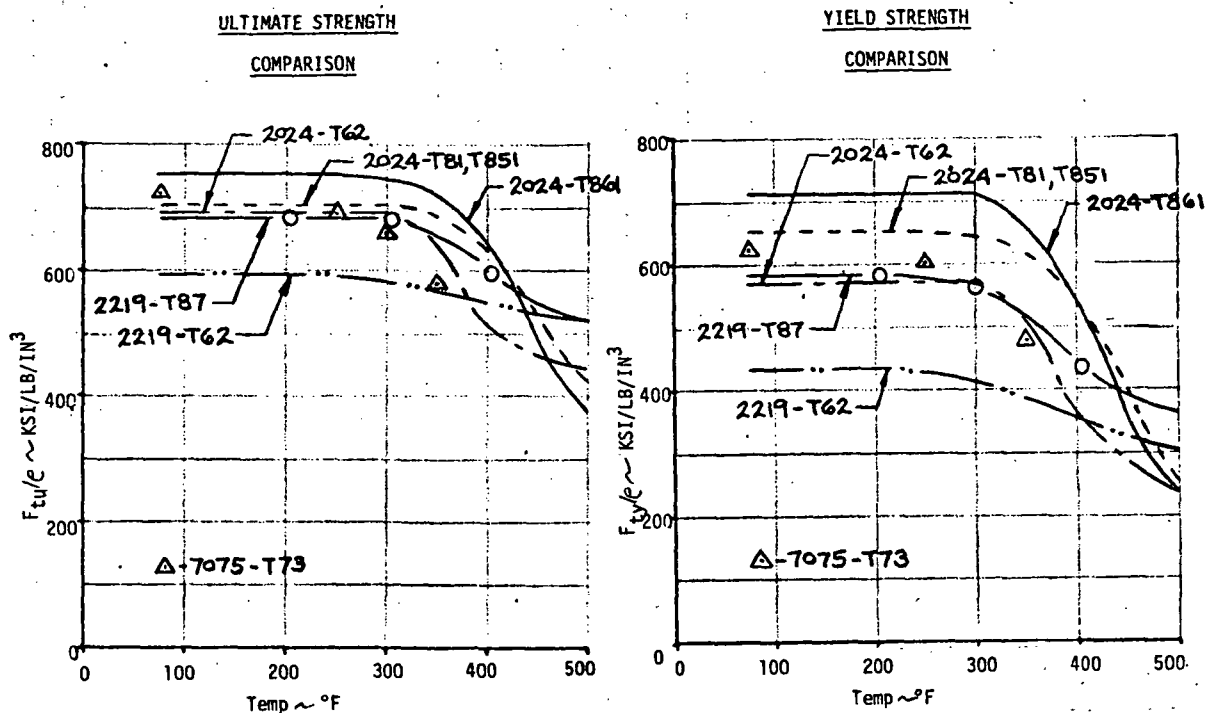
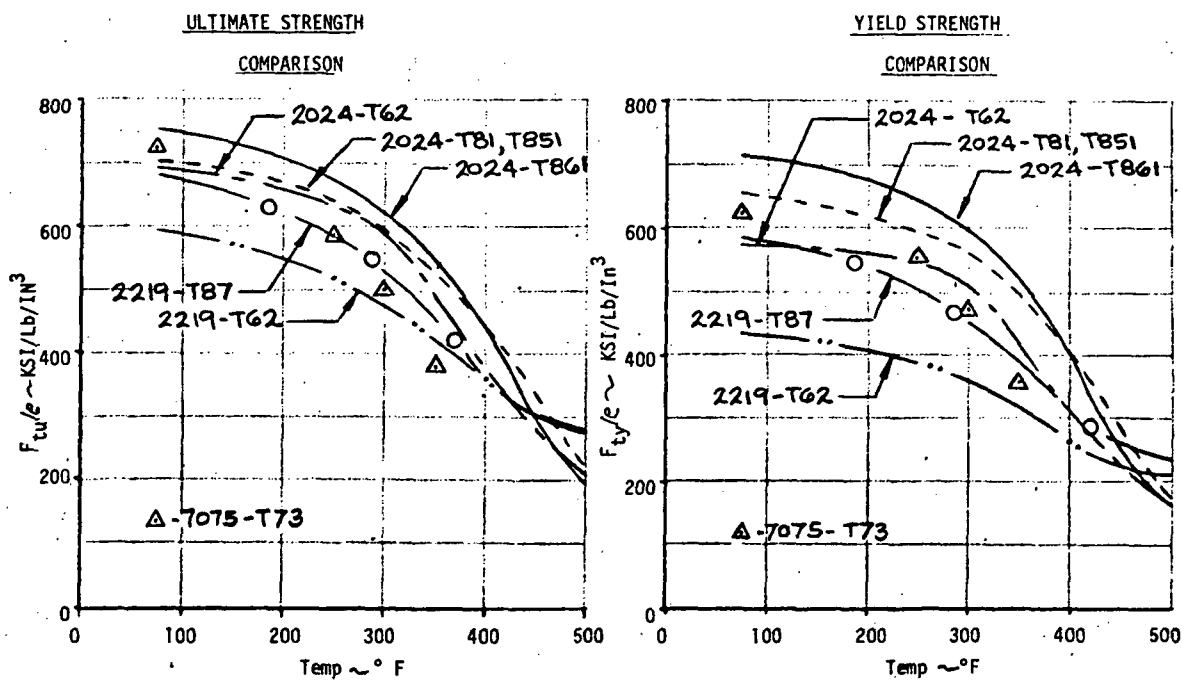


FIGURE 3.1.1-5

ALUMINUM SHEET PLATE

Based on ALCOA Test Data
Typical Tensile Props.

AT EXPOSURE TEMPERATURE AFTER 100 HRS EXPOSURE



15 MARCH 1972

FIGURE 3.1.1-6
ALUMINUM EXTRUDED SHAPES

Based on ALCOA Test Data
Typical Tensile Props.

AT ROOM TEMPERATURE AFTER 100 HOURS EXPOSURE

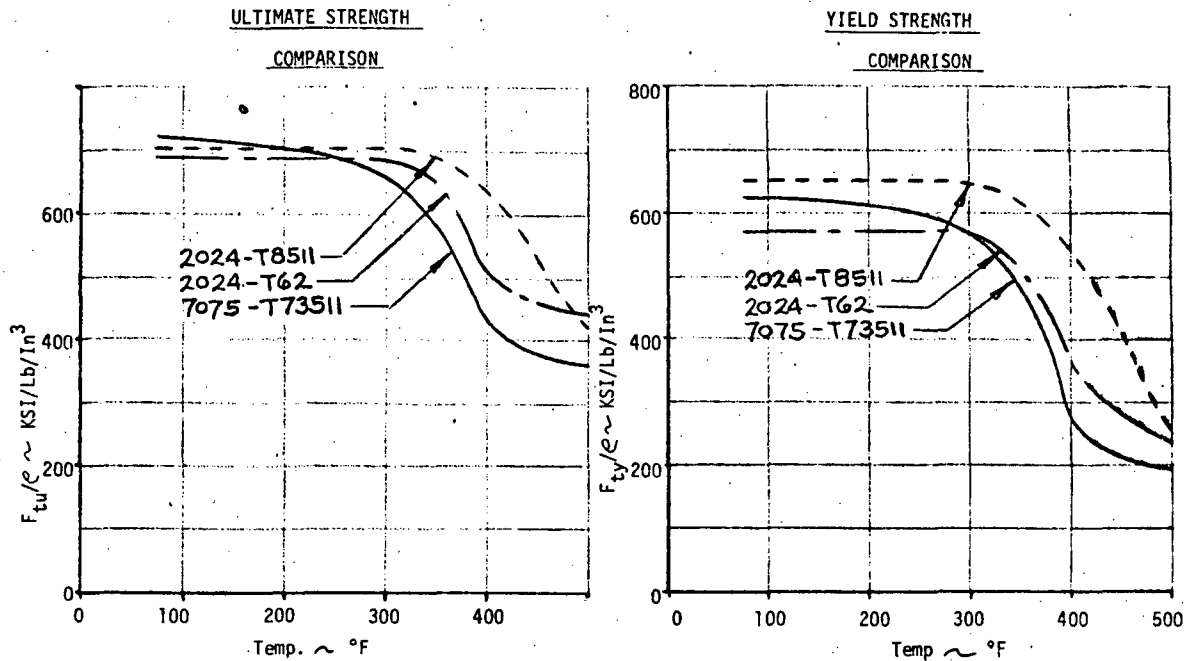
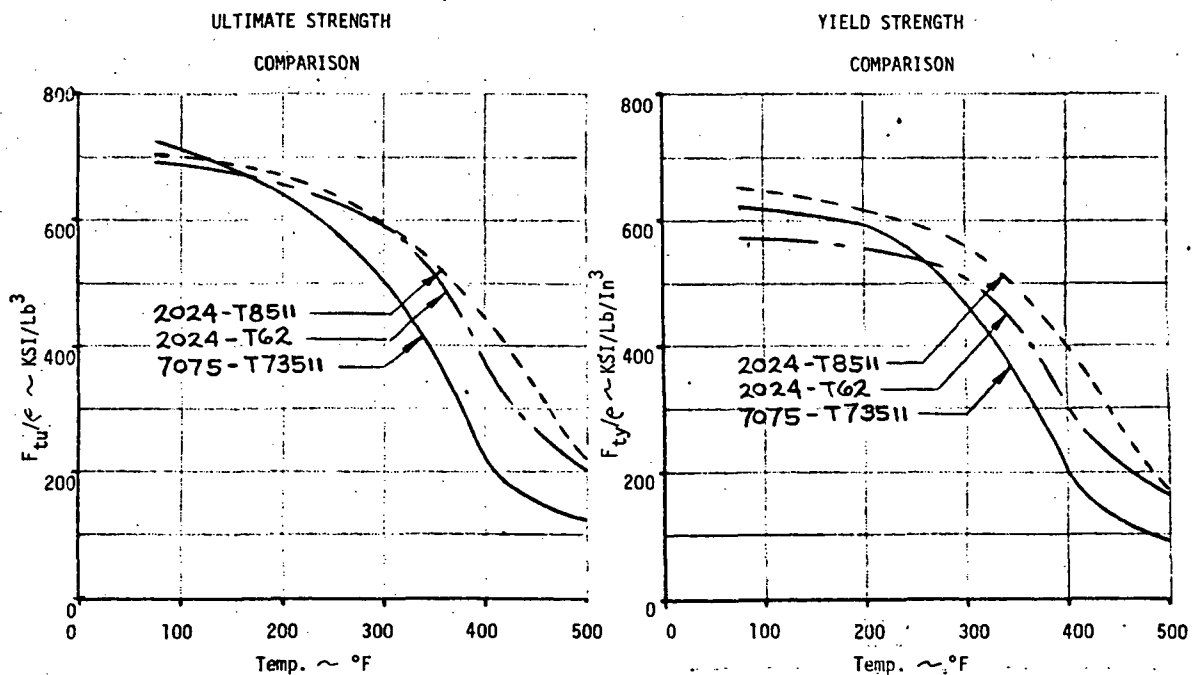


FIGURE 3.1.1-7
ALUMINUM EXTRUDED SHAPES

Based on ALCOA Test Data
Typical Tensile Props.

AT EXPOSURE TEMPERATURE AFTER 100 HRS EXPOSURE



15 MARCH 1972

FIGURE 3.1.1-8
MATERIAL CRIPPLING INDEX

Based on ALCOA Test Data
Typical Tensile Props.

100 HOURS EXPOSURE

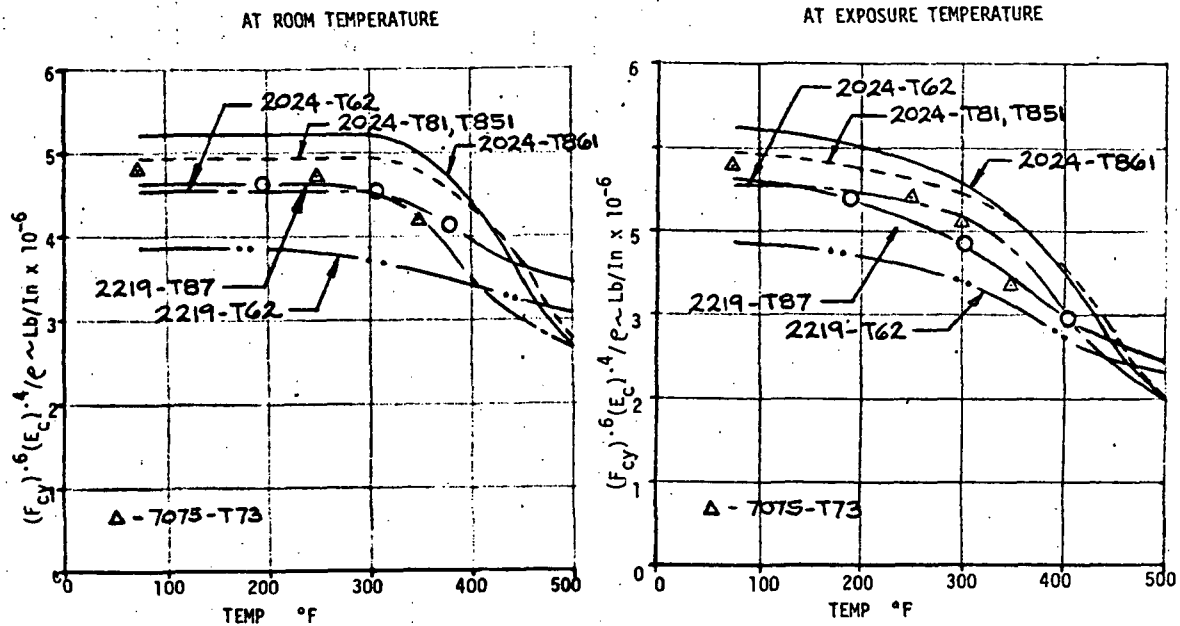


FIGURE 3.1.1-9

TEMPERATURE USE LIMIT

Alloy	Form	Temperature Use Limit (°)
2024-T62	Sheet Extrusion	350
2024-T81	Sheet	375
2024-T851	Plate	375
2024-T8511	Extrusion	375
2024-T861	Sheet, Plate	360
2219-T62	Sheet, Plate	345
2219-T87	Sheet, Plate	345
7075-T73511	Extrusion	310
7075-T7352	Forging	310

3.1.1.2 Frame Spacing Optimization Study - This study was performed to determine the spacing of frames which would yield the minimum fuselage structural weight for a typical running load of:

$$N = 1100 \text{ lb/in}$$

$$q = 300 \text{ lb/in}$$

Weights were determined for various frame spacings, using typical fuselage frames and for zee-stiffened shell structure. Material was assumed to be 2024 - T86 aluminum alloy at 70°F.

Results showed that minimum fuselage structural weight was achieved by maintaining a fuselage frame spacing of 20 inches.

Orbiter configuration used in study is not identical to that described in Section 3.1.1, however, the conclusions reached are considered valid.

3.1.1.3 Buckled vs Non-Buckled Mold Line Skin Panels - This study was conducted to determine the total system weight change for a non-buckling design approach for mold line skin panels. The current design approach is to permit the panels to buckle in shear and compression. The mold line RSI TPS is isolated from in-plane panel strains and buckle deflections by use of a foam rubber substrate (Figure 3.1.1-10).

The weights of the RSI TPS and the associated mold line structure were generated for both buckling and non-buckling design approaches. Typical RSI TPS unit weights which were derived are shown in Fig 3.1.1-10 and the weight variations for the total system are shown in Fig 3.1.1-11. The TPS weight increase for strain isolation pads required to isolate RSI from panel deflections is $.13 \text{ lb/ft}^2$ (see Figure 3.1.1-12).

The range of structure weight increase (3000 lb. to 4500 lb.) for a non-buckling design is based on skin-stringer riveted structures and included evaluation of the mold line structure of the fuselage, wing, tail and control surfaces. The weight range is due to uncertainties associated with the fact that not all loading conditions are defined and that compression and shear buckling interaction was not considered. The net weight penalty associated with the non-buckling design is from 1880 to 3380 lb.

The recommended design approach is to permit panel buckling, because of the significant weight advantage. A foam rubber substrate will be provided to isolate the RSI from buckle deflections. Panel element tests are being performed to substantiate the feasibility of this design approach.

FIGURE 3.1.1-10

BUCKLING OF MOLD LINE SKIN PANELS WEIGHT PENALTY FOR ISOLATION OF RSI FROM PANEL BUCKLING DEFLECTION

- o FACTORS INFLUENCING REQUIRED FOAM THICKNESS
 - o BUCKLE DEPTH (SHEAR AND COMPRESSION)
 - o FOUNDATION MODULUS OF FOAM
 - o BENDING STRENGTH OF RSI PANEL; ALLOWABLE STRESS AND PANEL THICKNESS
 - o STRUCTURE AND RSI IN-PLANE STRAINS
- o ESTIMATED FOAM THICKNESS TO ISOLATE RSI
 - o IN-PLANE STRAINS - $\sim .10$ IN. AVE.
 - o BUCKLE DEFLECTION - $\sim .25$ IN. AVE.
- o WEIGHT DELTA DUE TO ISOLATION FROM BUCKLE DEFLECTIONS
 - o AVERAGE $\sim + .13$ LB/FT² - INSULATING CAPABILITY OF FOAM REDUCES RSI REQUIREMENT

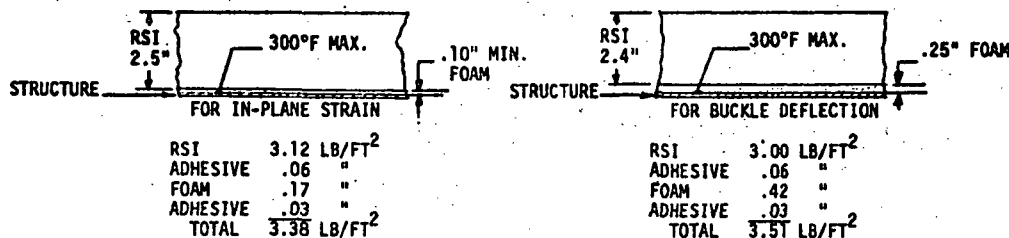


FIGURE 3.1.1-11

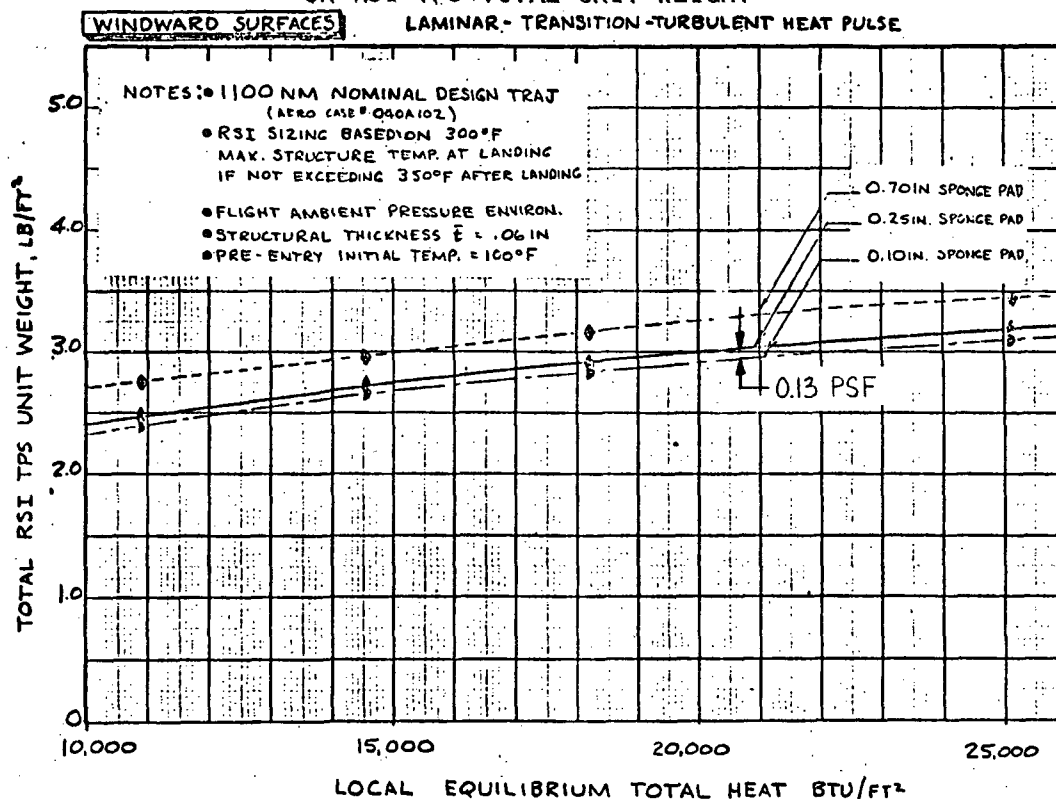
BUCKLING OF MOLD LINE SKIN PANELS TOTAL SYSTEM WEIGHT

COMPARISON OF BUCKLING AND NON BUCKLING SKIN DESIGN APPROACH

- o BASELINE DESIGN
 - o WEIGHT BASED ON BUCKLING SKIN
 - o WEIGHT INCLUDES STRAIN ISOLATOR FOR BUCKLING DEFLECTION ($t = .25$ IN. AVE.)
- o ALTERNATE DESIGN
 - o WEIGHT BASED ON NON BUCKLING SKIN
 - o WEIGHT INCLUDES STRAIN ISOLATOR FOR IN-PLANE STRAINS ($t = .10$ IN. AVG.)
- o DELTA WEIGHTS FROM BASELINE
 - o STRUCTURE + 3000 LB TO + 4500 LB
 - o STRAIN ISOLATOR - 1120 LB (8652 FT² @ .13 PSF)
 - o TOTAL + 1880 LB TO + 3380 LB.

15 MARCH 1972

FIGURE 3.1.1-12
EFFECT OF STRAIN ISOLATION - SPONGE PAD THICKNESS
ON RSI TPS TOTAL UNIT WEIGHT



3.1.1.4 Cabin Design Considerations - This section summarizes a trade study comparing three cabin concepts on the bases of weight, volumetric efficiency, and ease of inspection.

Although the Orbiter configuration is not identical to that described in Section 3.1.1, the information presented is considered valid. The moldline integral is presented as baseline, however, the study is continuing; final selection will be based on minimum cost and minimum gas leakage.

The three concepts were:

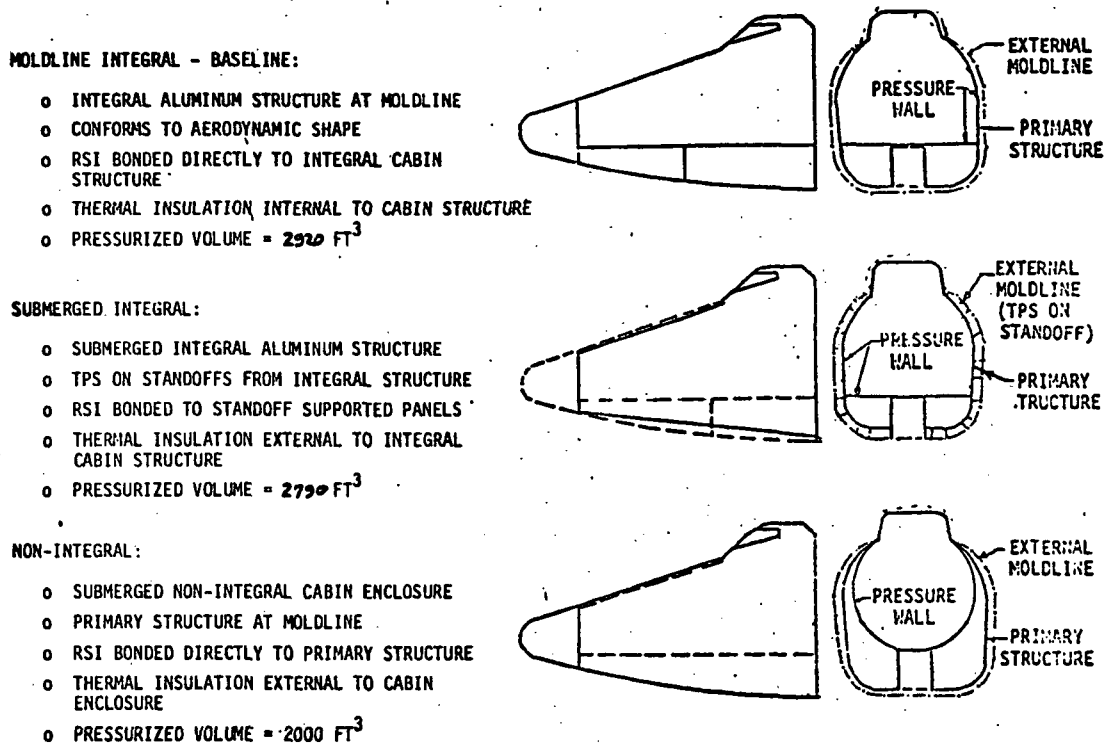
- o Moldline integral aluminum
- o Submerged integral with TPS on standoffs
- o Submerged non-integral structure with separate primary structure at moldline

The primary considerations were:

- o Minimum potential for cabin gas leakage
- o Moisture formation on cabin walls in orbit
- o Attenuation of external acoustic environment
- o Heat loss in orbit

15 MARCH 1972

FIGURE 3.1.1-13
CABIN DESIGN CONSIDERATIONS
CABIN DESIGN CONCEPTS STUDIED



The design concepts studied are shown in Figure 3.1.1-13

The preliminary conclusions were:

- o Weight is not a major discriminator
- o Non-integral and submerged integral cabins have greatest inherent acoustic attenuation
- o Reentry thermal protection system plus insulation provides adequate thermal control to prevent on-orbit moisture formation with non-integral and submerged integral cabins
- o Non-integral and submerged integral cabin pressure walls can be more easily fabricated from large integrally stiffened panels than integral moldline pressure walls with compound curvature.
- o Non-integral and submerged integral cabin provide inherent protection from damage (leaks) from TPS refurbishment
- o Integral and submerged integral cabin provide higher volumetric efficiency
- o Submerged integral is most inspectable
- o Further study required.

15 MARCH 1972

3.1.1.5 Payload Bay Door/Interstage Attach Method Trade Study - This trade study was performed to determine the combination of (1) payload bay door concept, and (2) forward interstage attach method which offers the greatest advantage in terms of weight and torsional stiffness.

Fig 3.1.1-14 defines eight configurations examined during this trade study, and also explains differences between structural and non-structural payload bay doors and between single point and three-point forward interstage attach methods (Figure 3.1.1-15). Configurations A, C, E, and F, formed by combinations of these concepts, are the basic configurations studied. Configurations A and C (with a structural door) were reexamined as configurations B and D, to determine what effects would result from inability of the door to carry overall vehicle loads after ascent. Configurations E and F (with non-structural door) were reexamined as Configurations G and H, to determine the advantages offered by a structural liner on the lower half of the payload bay.

Determination of the maximum fuselage twist θ and payload door enclosure axial deflection δ is important to this study, because extreme deflection would result in interference between the door and its enclosure. The result would be local damage to the thermal protection system (TPS) and possibly to the structure itself. Preliminary analysis has shown that the design clearance should be .12 - .25 inches. With a non-structural door, only configurations with three-point forward attachments have acceptable maximum δ values during ascent, and all configurations with non-structural doors are outside the acceptable range during unsymmetrical braking at landing (the critical post-ascent condition for fuselage twist).

Additional factors to be considered are:

- (1) Mechanisms for latching structural doors are not significantly more complex or heavier than required by non-structural doors.
- (2) Peak body bending moments during ascent will cause additional axial deflections of 1.0 - 3.0 inches in the door enclosure. For non-structural doors these values represent unacceptable gaps or interferences between the door and its enclosure.
- (3) These high axial deflections indicate that non-structural doors will act as load-carrying structure regardless of the door design concept.

In conclusion, the Orbiter configuration with structural payload bay door and three-point forward attachment offers the advantages of least structural weight and

FIGURE 3.1.1-14

DESCRIPTION OF CONFIGURATIONS STUDIED

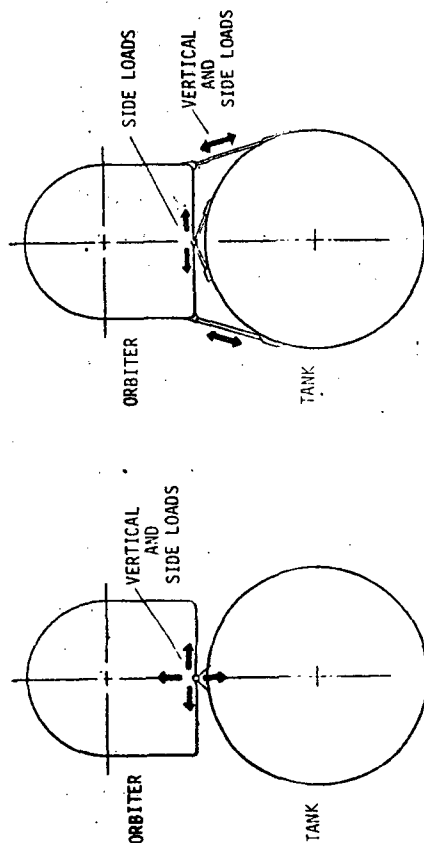
DOOR CONCEPT	PAYLOAD BAY LINER CONCEPT	FORWARD ATTACH METHOD	
		THREE-POINT FORWARD ATTACH	SINGLE-POINT FORWARD ATTACH
STRUCTURAL PAYLOAD BAY DOOR	NO STRUCTURAL LINER	A	C
STRUCTURAL DOOR DURING ASCENT; NON-STRUCTURAL FOR ALL OTHER MISSION PHASES	NO STRUCTURAL LINER	B	D
NON- STRUCTURAL PAYLOAD BAY DOOR	NO STRUCTURAL LINER	E	F
	STRUCTURAL LINER	G	H

DEFINITION OF CONCEPTS

- o PAYLOAD BAY DOOR
 - o STRUCTURAL - acts with rest of fuselage cross-section in carrying overall vehicle bending, axial, and torsion loads.
 - o NON-STRUCTURAL - designed to carry only its own aerodynamic and inertial loads.
- o STRUCTURAL PAYLOAD BAY LINER - surrounds the entire lower half of the payload envelope, acts with the rest of the fuselage cross-section in carrying overall vehicle bending, axial, and torsion loads.
- o FORWARD INTERSTAGE ATTACH METHOD
 - between ascent propellant tank and orbiter vehicle.
 - o SINGLE-POINT - y - and z - direction loads transmitted through single-point; no capability for transmitting fuselage torque.
 - o THREE-POINT - y - direction loads transmitted through center attach point; z- and y- direction loads transmitted through two outer points, thus, providing capability for transmitting torque.

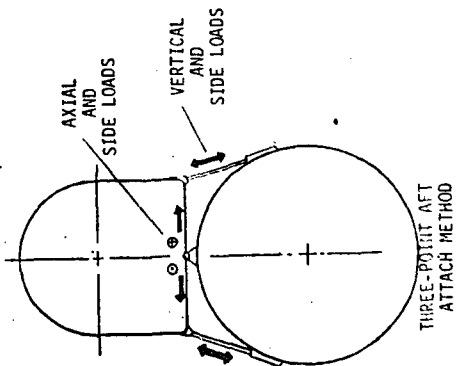
15 MARCH 1972

FIGURE 3.1.1-15
INTERSTAGE
ATTACH
METHODS



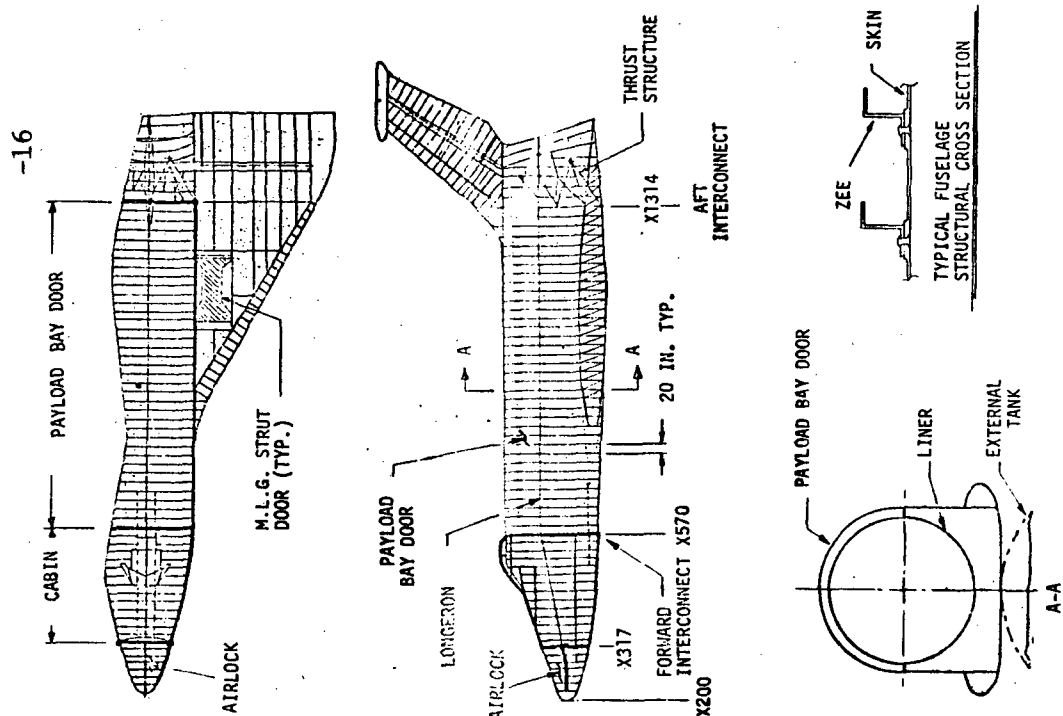
THREE-POINT FORWARD
ATTACH METHOD

SINGLE-POINT FORWARD
ATTACH METHOD



THREE-POINT AFT
ATTACH METHOD

FIGURE 3.1.1-16
STRUCTURAL ARRANGEMENT



15 MARCH 1972

highest torsional stiffness. Also, in this configuration improper door closure after ascent would be acceptable in terms of mission safety and program cost due to the low probability of occurrence.

A general arrangement of the baseline delta-wing Orbiter for this study is shown in Fig 3.1.1-16. This Orbiter has a single-piece payload bay door, and the interstage attach points are located at the bulkheads at the forward and aft ends of the 15 ft. dia. x 60 ft. payload bay (fuselage stations X570 and X1314, respectively). This configuration is somewhat different than the Orbiter described in Section 2.2.1.1, however, conclusions reached are still considered valid.

Fig 3.1.1-17 summarizes the overall vehicle loads at the three fuselage locations used in sizing the vehicle cross-sections, plus values of fuselage torque used to determine the maximum fuselage twist angles and door enclosure deflections

Basic conditions and assumptions for this study include the following:

o Material and Properties

- 2024-T86 aluminum alloy at 70°F after 200 hours exposure at 300°F
- "Guaranteed typical" properties used; defined as

$$\begin{array}{ccc} \text{GUARANTEED} & & \text{MIL-HDBK-5A} \\ \text{TYPICAL} & + (1.07) & \text{"A" VALUES} \\ & & \times \end{array}$$

(Valid for F_{tu} , F_{ty} , F_{cy} , and F_{su})

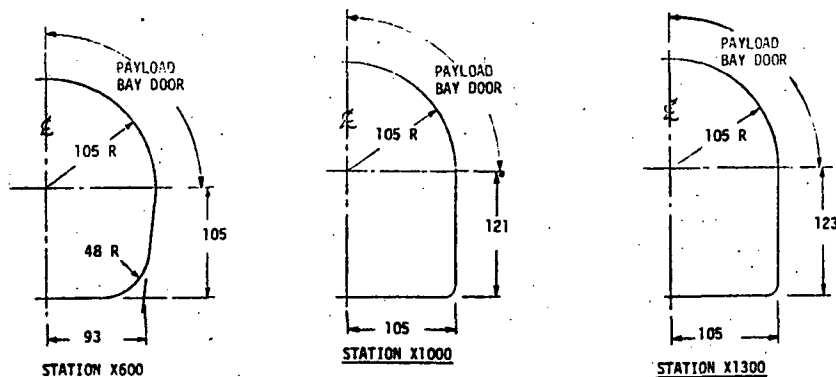
- o Ascent Propellant Tank - 300 in. dia. tank; 0.15 inch thick (average) aluminum wall.
- o Fuselage Sizing Criteria -
 - 20 inch frame spacing
 - Zee-stiffened panels carrying axial and shear loads
- o No contribution from wing for Orbiter fuselage bending and torsional stiffnesses.

Thermal stresses were neglected since the fuselage is basically sized by loading conditions which occur during ascent, when aerodynamic heating is negligible.

Results - Fig 3.1.1-18 summarizes the results of the analyses. They indicate that structural payload bay doors offer significant advantages in structural weight and in torsional stiffness.

The three-point attach method (Configuration C) permits the ascent propellant tank and Orbiter fuselage to act together like two closed cells in carrying Orbiter

FIGURE 3.1.1-17
FUSELAGE CROSS-SECTIONS



ULTIMATE LOADS

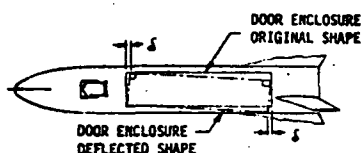
CROSS-SECTION LOCATION	LOAD	LOADING CONDITIONS						
		MAX. a_g (HV)	MAX. a_g (TV)	BOOSTER EJECT	ORBITER ENGINE IGNITION	MAIN GEAR LANDING	MAX. a_g (SV)	UNSYMMETRICAL MAIN GEAR LANDING
X600	M_y	-1.5	11.0	1.5	3.5	-5.0	-2.0	--
	M_x	0	0	0	0	0	± 26.0	--
	P_x	-.13	-.15	-.09	-.05	0	-.15	--
	M_z	--	--	--	--	--	22.0	5.0
X1000	M_y	-51.0	48.0	-18.0	65.0	-28.0	-33.0	--
	M_x	0	0	0	0	0	± 21.4	--
	P_x	-.32	-.35	-.27	-.17	0	-.35	--
	M_z	--	--	--	--	--	22.0	5.0
X1300	M_y	-65.0	23.0	-49.0	117.0	-7.0	-40.0	--
	M_x	0	0	0	0	0	± 24.0	--
	P_x	-.55	-.60	-.61	-.38	0	-.60	--
	M_z	--	--	--	--	--	22.0	5.0

- LOAD MAGNITUDES ARE IN 10^6 LB OR 10^6 IN-LB
- ORBITER NOSE LOCATED AT STATION X200
- PAYLOAD BAY CENTERLINE LOCATED AT 2400
- FACTOR OF SAFETY = 1.4
- "UNSYMMETRICAL MAIN GEAR BRAKING" REPRESENTS MOST CRITICAL M_z LOAD FOR POST-ASCENT MISSION PHASES
- LOADS REFERENCED TO 2135

FIGURE 3.1.1-18

STRUCTURAL vs NON-STRUCTURAL
PAYLOAD BAY DOOR

SUMMARY



CONF	STRUCTURAL CONFIGURATION		TWIST & DEFLECTION VALUES				WEIGHT PENALTIES (LB)
	ASCENT	POST-ASCENT	ASCENT		POST-ASCENT		
			θ	δ	θ	δ	
A			.13	NA	.03	NA	0
B		 "A" WITH DOOR MALFUNCTION	.13	NA	1.6	~0.7	0 *
C			.86	NA	.20	NA	0
D		 "C" WITH DOOR MALFUNCTION	.86	NA	1.5	~0.7	0 *
E			.14	~.06	1.4	~0.6	410
F			6.6	~3.0	0.8	~.36	2210
G			.14	~.06	1.1	~0.5	1240
H			2.8	~1.3	0.65	~0.3	1640

- * NO ADDITIONAL PENALTY REQ'D FOR POST-ORBITAL LOADINGS WITH DOOR MALFUNCTION.
- θ IN DEGREES, δ IN INCHES
- NA = NOT APPLICABLE

15 MARCH 1972

fuselage torques. This configuration has much more torsional stiffness than Configuration A, which has a single/point forward attach. The two configurations do not significantly differ in weight for the following reasons: (1) Fuselage torque did not affect the fuselage cross-section sizing for either configuration; (2) overall orbiter bending and axial loads are the same for either attachment method.

Configurations E and F, with non-structural payload bay doors, also differ structurally only in forward attach point method. Comparison of the values for fuselage twist θ and payload door enclosure axial deflection δ , (explained in Fig 3.1.1-18), plus fuselage weight, indicates the advantages in transmitting fuselage torques into the tank at the forward attach point. Sizing the fuselage cross-sections for overall Orbiter bending and axial loads results in a 410 lb increase for E and F, as compared with Configurations A and C, in which the door acted with the fuselage structure in carrying these loads. In addition, Configuration F requires a further increase of 1790 lb to account for the increased internal stresses in the Orbiter fuselage which result from carrying fuselage torques in differential bending.

Configurations B and D are identical to Configurations A and C, respectively, except that the door was considered to be not effective in taking loads after completion of ascent. Such a condition could result from inadequate latching. Results show that structural weight is not affected, and that values for θ and δ are comparable to those for Configurations E and F (with non-structural doors) for post-ascent mission phases.

Configurations G and H are identical structurally to Configurations E and F, respectively, except that a structural liner (capable of carrying shear and axial loads) was added to the lower half of the payload bay for its full length. The liner results in a significant weight gain rather than a weight savings, because much of the liner perimeter is near the neutral axes for M_y and M_z bending loads, and is thus inefficient in taking body bending loads.

15 MARCH 1972

3.1.1.6 Conventional Wing Structure vs. Discrete Spars and Ribs with Removable Panels - This section summarizes a trade study of structural sizing for a conventional wing utilizing distributed bending material and for a wing employing discrete spars and ribs with floating surface panels. The following weight summary shows essentially the same weight for either approach. However, the conventional design concept of distributed wing bending material was selected because of greater confidence in this concept and the resulting weight estimation. The wing configuration used in this trade study is not the same as that described in Section 3.1.1; however, conclusions reached are still considered valid.

<u>Component</u>	<u>Conventional Wing</u>	<u>Wing With Floating Surface Panels</u>
<u>Exposed Wing</u>		
Covers	3275 lb.	1900 lb.
Spars	1365	2450
Ribs	1710	1900
<u>Carrythrough</u>		
Spars	1000	1570
Cover (lower)	1074	686
Root Rib	<u>520</u>	<u>520</u>
Subtotal	8944	9026
10% Contingency	<u>894</u>	<u>903</u>
TOTAL	9838	9929

Notes:

1. Spar and rib weights include an increment for drag structure.
2. Material 2024-T81 aluminum alloy.
3. Weights include 28% non-optimums.

15 MARCH 1972

3.1.2 Thermal Protection System - The orbiter thermal protection system (TPS) is passive and consists of: reusable surface insulation (RSI) mullite HCF; SLA-561 ablator, and ESA-3560-IIA ablator. RSI is used where surface temperatures are between 750°F and 2500°F (maximum temperature for extensive reuse); SLA-561 is used where temperatures are below 750°F (no charring) and above 2500°F; ESA-3560-IIA is used on the nose cap and leading edges where protection from rain erosion is needed.

Figure 3.1.2-1 shows the orbiter TPS material distribution, unit weights and thicknesses. High density ablator (56 pcf) ESA-3560-IIA is used on the nose cap and leading edges. Low density ablator (14.5 pcf) SLA-561 is used: on the wing lower surface and just aft of the nose cap where the surface temperature is above 2500°F; on the aft upper wing surface where the surface temperature is below 750°F (no charring); on the vertical tail sides; on the aft upper fuselage; on the aft fuselage sides; and on the base heat shield. RSI (15 pcf) is used on all other surfaces. The thermal environment used to compute these TPS unit weights, Figure 3.1.2-2, was based upon Cornell wind tunnel test data.

The orbiter baseline TPS concept for RSI and ablator is shown in Figure 3.1.2-3. A layer of foam rubber is used between the structure and RSI for thermal and mechanical strain isolation. This strain isolation permits the structure to buckle, which saves weight, and conforms to manufactured surface irregularities. A reduced bond weight is needed to insure a continuous bond line.

The RSI/ablator is sized to limit the structural temperature to 300°F at the end of landing rollout and no greater than 350°F thereafter. A maximum structure temperature of 350°F after landing rollout was selected based on material degradation and reuse considerations.

The remainder of this section will describe six studies that helped to define and refine the selected TPS concepts described above. These studies are concerned with:

- o nose cap and wing/tail leading edges
- o fuselage, wing and tail surfaces
- o cabin windshield
- o fuselage base
- o influence of maximum structural temperature on combined weight of TPS, structure and ECLS insulation
- o effect of crossrange on TPS weight.

15 MARCH 1972

3.1.2.1 TPS Evaluations: Nose Cap, Wing and Tail Leading Edges - The design goal in choosing TPS materials and concepts is to have a fully reusable TPS on the entire orbiter surface. If the selected materials are not reusable, the design concept should permit easy replacement and/or refurbishment. The concept must provide minimum design risk but maximize the use of existing technology. The TPS material must be able to resist erosion without surface recession that decreases airfoil L/D. Thermal isolation from the aluminum primary structure must also be provided. Special TPS requirements exist for the nose cap, wing and tail leading edges because they are exposed to very high temperatures and an erosive environment.

A TPS of high density ablator, ESA-3560-IIA (56 pcf), was selected. A reusable TPS made of carbon/carbon has been selected as an alternate but will require development. The major design considerations that influence TPS concept selection and material choice for the nose cap and leading edges are listed in Figure 3.1.2.1-1.

Candidate TPS concepts are rated in Figure 3.1.2.1-2. Specific advantages and disadvantages are listed for each concept. High density ablators (above 40 pcf) were selected for the baseline TPS candidate and carbon/carbon was selected as an alternate candidate. Ablators will provide high confidence and low risk for early development flights while evolution to carbon/carbon should provide a TPS with greatly reduced refurbishment requirements.

The potential ablative candidates for the Orbiter are listed in Figure 3.1.2.1-3. ESA-3560-IIA appears the best choice; however, test data on the rain erosion and thermal behavior of several materials will be required before a final evaluation can be made. ESA-3560-IIA was selected over S-3 primarily because of its smoother char surface, although the densities are the same. AVCOAT-5026 has a lower density and was judged to have a corresponding lower resistance to rain erosion when charred.

Figures 3.1.2.1-4 and 3.1.2.1-5 present the wing leading edge selected TPS concept and reusable alternate concept, respectively. The nose cap TPS concept is presented in Figure 3.1.2.1-6. The rationale for selecting the baseline TPS for the nose cap, wing and tail leading edges is presented in Figure 3.1.2.1-7.

3.1.2.2 TPS Evaluations: Fuselage, Wing and Tail Surfaces - The primary TPS design considerations for the major surfaces of the Orbiter (fuselage, wing and tail surfaces) are listed in Figure 3.1.2.2-1. Minimum risk, refurbishment and development cost, and weight are prime design considerations. Major TPS candidate categories (ablator, RSI and metallic) are shown in Figure 3.1.2.2-2 with the specific advantages and disadvantages of each. Low density ablators and RSI were

15 MARCH 1972

FIGURE 3.1.2.1-1

ORBITER NOSE CAP, WING & TAIL LEADING EDGES MAJOR TPS DESIGN CONSIDERATIONS

- PROVIDE MINIMUM DESIGN RISK
- MAXIMIZE USE OF EXISTING TECHNOLOGY
- ENDURE RAIN AND WIND EROSION DURING LAUNCH AND FLYBACK
- AVOID SURFACE RECESSION AT MAXIMUM HEATING RATES
- MAINTAIN SMOOTH AERODYNAMIC SURFACE TO AVOID INFLUENCES ON LIFT AND DRAG
- RETAIN ABLATIVE CHAR LAYER INTACT THROUGH ENTRY AND FLYBACK
- PROVIDE THERMAL ISOLATION FROM STRUCTURE LIMITED TO 300°F AT LANDING ROLLOUT; LIMITED TO 350°F THEREAFTER
- PERMIT LOW COST REPLACEMENT OR REFURBISHMENT IF REQUIRED
- MINIMIZE WEIGHT AFTER ABOVE CONSIDERATIONS ARE SATISFIED

FIGURE 3.1.2.1-2

ORBITER TPS CANDIDATES

NOSE CAP; WING & TAIL LEADING EDGES

CANDIDATE	ADVANTAGES	DISADVANTAGES	RELATIVE RATING
ABLATOR ($\rho < 20$ PCF) (OR RSI)	• LOWEST WEIGHT • MATERIALS AVAILABLE • EXCELLENT THERMAL ISOLATION • AVOIDS RISK OF HEAT UNCERTAINTIES	• POOR RAIN EROSION RESISTANCE • SURFACE WILL RECEDE AT PEAK HEATING • REFURBISHMENT AFTER EACH FLIGHT	6
ABLATOR ($\rho \approx 30$ PCF)	• LOW WEIGHT • NO SURFACE RECESSION • MATERIALS AVAILABLE • GOOD THERMAL ISOLATION • AVOIDS RISK OF HEAT UNCERTAINTIES	• MARGINAL RAIN EROSION RESISTANCE • REFURBISHMENT AFTER EACH FLIGHT	8
 ABLATOR ($\rho > 40$ PCF)	• MODEST WEIGHT • GOOD RAIN EROSION RESISTANCE • NO SURFACE RECESSION OR SWELLING • MATERIALS AVAILABLE • GOOD THERMAL ISOLATION • AVOIDS RISK OF HEAT UNCERTAINTIES	• REFURBISHMENT AFTER EACH FLIGHT	10
 CARBON-CARBON	• MODEST WEIGHT • GOOD RAIN EROSION RESISTANCE • NO SURFACE RECESSION • NO/LOW REFURBISHMENT	• CURRENTLY DIFFICULT TO FABRICATE • OXIDATION COATING REQUIRED • POOR THERMAL ISOLATION	6
CERAMIC FOAMS Zr O ₂ Zr B ₂	• GOOD RAIN EROSION RESISTANCE • NO SURFACE RECESSION • NO REFURBISHMENT	• HIGH WEIGHT-POOR IN THERMAL SHOCK • POOR THERMAL ISOLATION • EMISSION COATINGS REQUIRED	4
COATED TANTALUM	• GOOD RAIN EROSION RESISTANCE • NO RECESSION WITHIN DESIGN LIMITS	• POOR THERMAL ISOLATION • REQUIRES RECOATING EACH FLIGHT • FAILURE SENSITIVE TO HEATING UNCERTAINTIES • HIGH WEIGHT	3

 SELECTED CANDIDATE FOR EARLY DEVELOPMENTAL FLIGHTS; MUST BE REPLACED AFTER EACH FLIGHT

 SELECTED ALTERNATE CANDIDATE FOR LATER DEVELOPMENTAL AND OPERATIONAL FLIGHTS; GREATLY REDUCED REFURBISHMENT EXPECTED

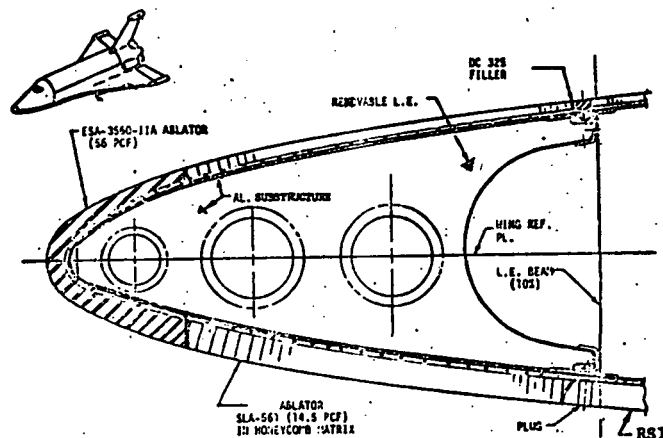
15 MARCH 1972

FIGURE 3.1.2.1-3
ORBITER NOSE CAP, WING & TAIL LEADING EDGES
POTENTIAL ABLATIVE CANDIDATES

CANDIDATE	DENSITY PCF	ADVANTAGES	DISADVANTAGES
▶ ESA-3560-11A (PRIME)	56	● FLIGHT PROVEN PERFORMANCE ● EASY FABRICATION & APPLICATION FOR WIDE RANGE OF DESIGN THICKNESS ● GOOD RAIN EROSION RESISTANCE	
S-3 (GEMINI)	56	● FLIGHT PROVEN PERFORMANCE ● 40% LOWER WEIGHT THAN 3560-11A HOWEVER APPLICATION AREA IS SMALL ● GOOD RAIN EROSION RESISTANCE	● SURFACE SWELLS TO ROUGH TEXTURE AT SHUTTLE HEATING RATES
AVCOAT-5026 (APOLLO)	32	● FLIGHT PROVEN PERFORMANCE	● PERFORMANCE DATA FOR SHUTTLE ENVIRONMENTS SCANT
PHENOLIC-FIBERGLASS (MERCURY)	115	● FLIGHT PROVEN PERFORMANCE ● EXCELLENT RAIN EROSION RESISTANCE	● VERY HEAVY ● SUBJECT TO DELAMINATION PROBLEMS ● HIGH COST TOOLING & FABRICATION
KOROTHERM	70	● BASELINE NOSECAP ON EXTERNAL TANKS ● SIMPLE APPLICATION IN THIN LAYERS NEEDED FOR ASCENT	● HEAVIER THAN OTHER AVAILABLE CANDIDATES

▶ SELECTED

FIGURE 3.1.2.1-4
THERMAL PROTECTION SELECTION RATIONALE
WING LEADING EDGE TPS
BASELINE CONCEPT



ADVANTAGES

- MATERIALS ARE AVAILABLE
- LOW RISK FOR MAX. HEATING AREAS
- AVOIDS GAPS IN MOST FORWARD AREAS
- EASY THERMAL ISOLATION FROM STRUCTURE

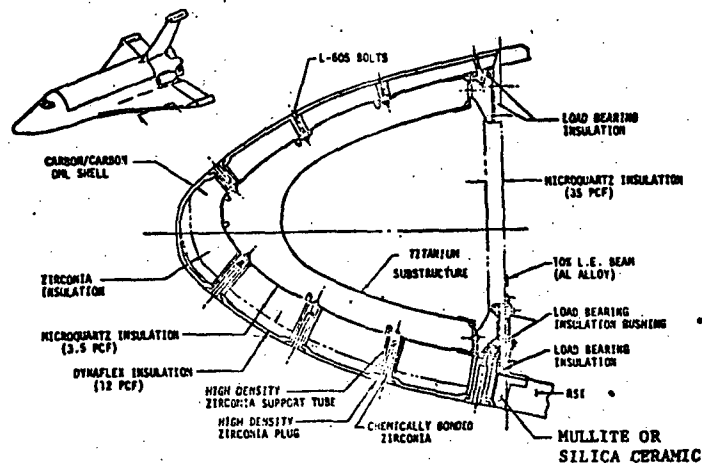
SHORTCOMINGS

- NOT REUSABLE
- REPLACEMENT AFTER EACH FLIGHT
- LACK OF EROSION RESISTANCE DATA FOR ABLATIVE CHAR

15 MARCH 1972

FIGURE 3.1.2.1-5

WING LEADING EDGE TPS.
REUSABLE ALTERNATE CONCEPT



ADVANTAGES

- REUSABLE TPS FOR MAX. HEATING AREAS
- EXPECTED TO PROVIDE REUSABLE EROSION RESISTANCE

SHORTCOMINGS

- REQUIRES DEVELOPMENT
- MORE COMPLEX THERMAL ISOLATION FROM ALUMINUM STRUCTURE
- IMPROVED STRUCTURAL INTEGRITY REQUIRED

FIGURE 3.1.2.1-6

NOSE CAP TPS

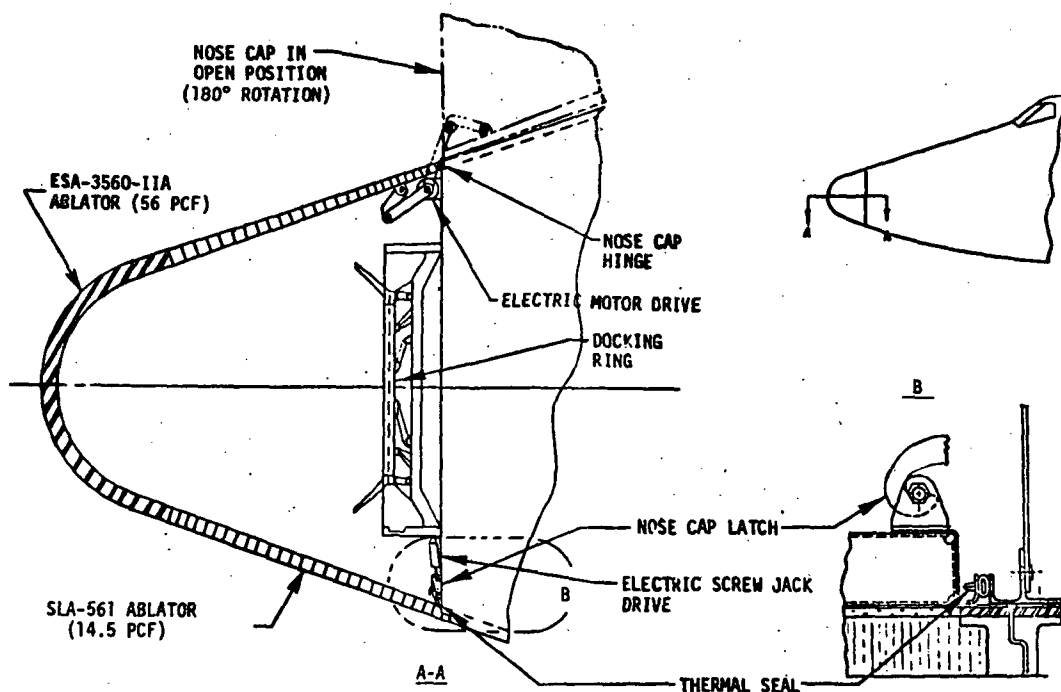


FIGURE 3.1.2.1-7

ORBITER NOSE CAP, WING & TAIL LEADING EDGES
SELECTED BASELINE CONCEPT & RATIONALE

CHOICE OF ABLATOR

- REQUIRES NO MATERIAL DEVELOPMENT PROGRAM
- PROVIDES SIMPLE THERMAL ISOLATION FROM STRUCTURE
- AVOIDS RISK DUE TO HEATING UNCERTAINTIES IN HIGH FLUX AREAS
- AVOIDS NEED FOR HIGH EMITTANCE OR OXIDATION RESISTANT COATINGS

CHOICE OF ESA-3560-11A ABLATOR (56 PCF)

- PROVIDES GOOD RAIN EROSION RESISTANCE
- MAINTAINS A SMOOTH SURFACE FOR AIRCRAFT FLIGHT PERFORMANCE
 - NO SURFACE RECESSION
 - NO SURFACE SWELLING
 - NO CHAR SPALLING
- PERMITS SIMPLE APPLICATION AND REFURBISHMENT
- RESULTS IN MODEST TPS WEIGHT FOR HIGH FLUX AREAS

FIGURE 3.1.2.2-1

ORBITER FUSELAGE, WING & TAIL SURFACES MAJOR TPS DESIGN CONSIDERATIONS

- PROVIDE PERFORMANCE INSENSITIVE TO HEATING UNCERTAINTIES;
MINIMIZE RISK
- MINIMIZE REFURBISHMENT COST AND TIME IF REQUIRED
- MINIMIZE WEIGHT BECAUSE OF LARGE SURFACE AREAS
- PROVIDE TPS STRAIN ISOLATION (IF REQUIRED) FROM STRUCTURE
PERMITTED TO BUCKLE AT LIMIT LOADS
- ACCOUNT FOR THERMAL & ACOUSTIC ENVIRONMENTS DURING ASCENT
& ENTRY
- MINIMIZE DEVELOPMENT OF NEW MATERIALS OR PROCESSES

FIGURE 3.1.2.2-2

ORBITER TPS CANDIDATES FUSELAGE, WING AND TAIL SURFACES

CANDIDATES	ADVANTAGES	DISADVANTAGES
▶ ABLATOR ($\rho = 15$ PCF) ◦ SLA-561 ◦ 1004X ◦ ULD-100-4 ◦ ULD-200 ◦ S20-T ◦ FIREX 250(35 PCF)	• MATERIALS AVAILABLE • INSENSITIVE TO HEATING UNCERTAINTIES • SIMPLE APPLICATION & REMOVAL • INSENSITIVE TO SEVERE ACOUSTIC ENVIRONMENT	• REFURBISHMENT AFTER EACH FLIGHT OVER MAJORITY OF SURFACE • BRITTLINESS OF ABLATOR & ADHESIVE FOR EXTREME ORBITAL COLD CONDITION MAY BE PROBLEM
▶ RSI ($\rho = 15$ PCF) ◦ MULLITE HCF ◦ LI-15 ◦ REI	• SIMPLE TRANSITION FROM ABLATORS TO RSI • REUSABLE - LITTLE REFURBISHMENT	• REQUIRES ADDITIONAL DEVELOPMENT • SOMEWHAT SENSITIVE TO LARGE HEATING UNCERTAINTIES • SENSITIVE TO SEVERE ACOUSTIC ENVIRONMENT • REQUIRES STRAIN ISOLATION FROM STRUCTURE PERMITTED TO BUCKLE • REQUIRES WATERPROOF COATING • ADHESIVE BRITTLINESS IN ORBITAL COLD
METALLIC SHINGLES ◦ NICKLE SUPERALLOYS ◦ COBALT SUPERALLOYS ◦ COATED COLUMBIUM	• REUSABLE - LITTLE REFURBISHMENT • MATERIALS AVAILABLE • WATERPROOFS INTERIOR OF VEHICLE • LOW SENSITIVITY TO SEVERE ACOUSTIC ENVIRONMENT	• DIFFICULT TRANSITION FROM ABLATORS TO RSI • USE WOULD REQUIRE RSI WEIGHT PENALTY FOR PANELS AND HEAT SHORT STAND-OFFS • WEIGHT & FAILURE SENSITIVE TO PRESSURE AND HEATING UNCERTAINTIES • COLUMBIUM COATINGS AND REPAIRS REQUIRE ADDITIONAL DEVELOPMENT • REQUIRES INSULATION PACKAGING DEVELOPMENT
▶ SELECTED CANDIDATES		

15 MARCH 1972

selected for additional evaluation; candidates in each category are compared in Figure 3.1.2.2-3. The selected orbiter TPS concepts are SLA-561 and RSI. SLA-561 ablator provides minimum risk to initial heating uncertainties and permits rapid, low risk transition to RSI. RSI mullite HCF reduces refurbishment costs, provides tolerance to modest heating uncertainties and is weight competitive with ablators.

The orbiter baseline TPS concept is shown in Figure 3.1.2-3. A layer of foam rubber is used between the structure and RSI for thermal and mechanical strain isolation. The RSI/ablator thickness is sized to limit the structural temperature to 300°F at the end of landing rollout and no greater than 350°F thereafter. A maximum structural temperature of 350°F after landing rollout was selected based on material degradation and reuse considerations. The distribution of SLA-561 and RSI on the fuselage, wing and tail surfaces is shown in Figure 3.1.2-1.

3.1.2.3 TPS Evaluations: Cabin Windshield - Thermal protection for the windshield must interface with several other orbiter systems, and has unique considerations that do not exist elsewhere on the orbiter. Major windshield design criteria/considerations are listed in Figure 3.1.2.3-1. The advantages of windshield covers listed in Figure 3.1.2.3-2 indicate a strong need for some form of external or internal cover. Three windshield TPS candidates are shown in Figure 3.1.2.3-3. External covers using RSI as TPS were selected because they best satisfy the design criteria and considerations.

3.1.2.4 TPS Evaluation: Fuselage Base - The fuselage base region has requirements that differ from the rest of the orbiter. Major design considerations are listed in Figure 3.1.2.4-1.

FIGURE 3.1.2.2-3
ORBITER FUSELAGE, WING AND TAIL SURFACES
ABLATOR & RSI TPS CANDIDATES

CANDIDATE	DENSITY PCF	ADVANTAGES	DISADVANTAGES
ABLATOR			
SLA-561	14.5	• LOWEST UNIT WEIGHT • DESIGN DATA AVAILABLE • EASY FABRICATION & APPLICATION	
1004X	15	• PERFORMANCE WELL KNOWN • EASY FABRICATION & APPLICATION	• UNIT WT 0.2 TO 0.3 PSF HIGHER THAN SLA-561 • WATERPROOF COATING NEEDED
ULD-100-4	16	• EASY FABRICATION & APPLICATION • DEMONSTRATED REUSE POTENTIAL (~10x) • LOWER REFURBISHMENT COST	• PERFORMANCE DATA IS MEAGER • HIGH ESTIMATED UNIT WEIGHT • THERMAL CONDUCTIVITY IS TWICE SLA-561
ULD-200	16	• BETTER PERFORMANCE THAN ULD-100	• PERFORMANCE DATA IS SCANT
S20-T	20	• DESIGN DATA AVAILABLE	• MODEST WEIGHT PENALTY • CONSISTENT FABRICATION IS DIFFICULT
FIREX 250	35	• BASELINE ON EXTERNAL TANKS	• HIGH UNIT WEIGHT FOR LONG ENTRY HEATING DURATION
RSI			
MULLITE HCF	15	• HIGHER TEMP. CAPABILITY THAN LI-15 • HIGHEST STRENGTH OF RSIS • LOWEST THERMAL CONDUCTIVITY	• 10 TO 15% HIGHER THERMAL CONDUCTIVITY DUE TO LARGER FIBER DIAMETER • LOWER STRENGTH DUE TO SMALLER FIBERS
LI-15	15		
MULLITE REI	12	• THERMAL CONDUCTIVITY APPEARS SIMILAR TO LI-1500(AT LOW TEMP)	• LOWER STRENGTH DUE TO SMALLER FIBERS

 SELECTED FOR AREAS BELOW 750°F (NONCHARRING) AND ABOVE 2500°F
 SELECTED FOR AREAS GREATER THAN 750°F, BUT LESS THAN 2500°F

FIGURE 3.1.2.3-1

ORBITER CABIN WINDSHIELD MAJOR DESIGN CRITERIA/CONSIDERATIONS

CRITERIA

- ANY ONE PANE CAN FAIL
- CABIN PRESSURE 14.7 PSIA
- OUTER PANE IS VENTED, 8 PSI ON OUTER GLASS
- GLASS SAFETY FACTOR OF 3.0
- TEMPERED GLASS CANNOT BE USED ON OUTER PANES DUE TO POSSIBLE METEORITE PENETRATION

CONSIDERATIONS

- PROVIDE MINIMUM DESIGN RISK
- MAXIMIZE USE OF EXISTING TECHNOLOGY
- PROVIDE STRUCTURAL CONCEPT TO SEAL-IN CABIN ATMOSPHERE
- AVOID DEPOSITS ON WINDOWS THAT WOULD OBSCURE VISION
- PROVIDE ADEQUATE VISIBILITY WHEN REQUIRED; ORBIT AND AIRPLANE FLIGHT
- AVOID HEAT LOAD INTO CABIN DURING LAUNCH AND ENTRY
- PROVIDE BIRD PROOFING

15 MARCH 1972

FIGURE 3.1.2.3-2
FORWARD WINDSHIELD

WITHOUT COVERS

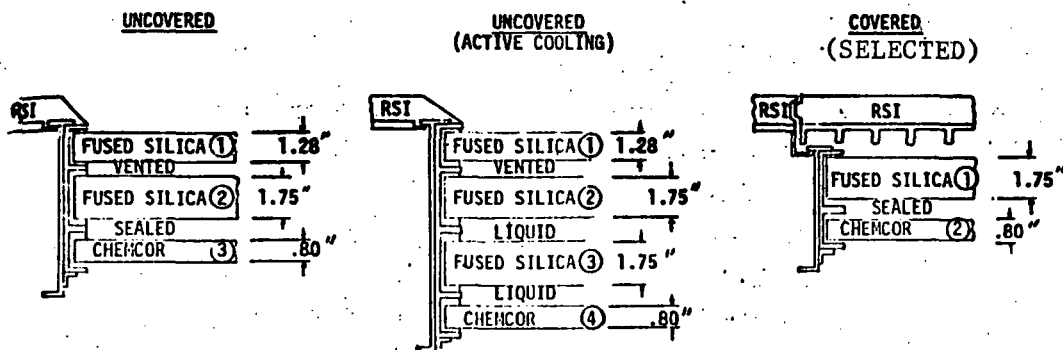
- o AREA FWD OF WINDOWS CANNOT BE ABLATOR DUE TO VISION IMPAIRMENT; MUST BE METAL, GLASS COATED RSI, CARBON-CARBON NOSE CAP, ETC.
- o UNCERTAINTY IN HEATING RATES CAN PRODUCE TEMPERATURES FROM 1070°F TO 1800°F ON OUTER PANE. FAILURE OF THE OUTER PANE CREATES A SEAL DESIGN PROBLEM ON CENTER PANE
- o WINDOW COOLING SYSTEM REQUIRED OR CABIN ECLS LOAD INCREASED
- o INTERNAL INSULATED COVER REQUIRED

WITH COVERS

- o COVERS MAY BE ABLATIVE OR RSI; SAME AS OVERALL ORBITER
- o GLASS TEMPERATURE LIMITED TO 280°F-MAX.; COVERS CLOSED ON LAUNCH AND ENTRY, OPEN IN ORBIT AND SUBSONIC FLIGHT
- o FO/FS RELIABILITY ASSURED BY:
 - o SEPARATE MECHANISM FOR EACH WINDOW COVER
 - o REDUNDANT POWERED DRIVE WITH MANUAL BACKUP
 - o OUTER GLASS PANE DESIGNED ENTRY HEATING

FIGURE 3.1.2.3-3

WINDOW CONCEPTS



MAXIMUM WINDOW TEMPERATURES - °F

PANE	UNCOVERED	UNCOVERED (ACTIVE COOLING)	COVERED
①	1070	1070	280
②	365	350	130
③	184	100	---
④	---	100	---

FIGURE 3.1.2.4-1

ORBITER FUSELAGE BASE TPS
MAJOR TPS DESIGN CONSIDERATIONS

- o MINIMIZE RISK FROM HIGHLY UNCERTAIN THERMAL ENVIRONMENT
- o SURVIVE SEVERE ACOUSTIC ENVIRONMENT (166 DB OR GREATER WITH PARALLEL BURN)
- o PROVIDE EASY POST-FLIGHT REMOVAL FOR ENGINE INSPECTION
- o MINIMIZE REFURBISHMENT COSTS
- o PERMIT ENGINES TO GIMBAL - MINIMIZE HOT GAS LEAKAGE
- o PROVIDE TPS STRAIN ISOLATION (IF REQUIRED) FROM STRUCTURE PERMITTED TO BUCKLE AT LIMIT LOADS
- o MINIMIZE LONG DURATION FLOW IMPINGEMENT ON NOZZLES DURING ENTRY

15 MARCH 1972

Three potential TPS candidates for the fuselage base are evaluated in Figure 3.1.2.4-2; low density ablators, metallic shingles/fibrous insulation and RSI imbedded into honeycomb. Low density ablators were selected because they best met the design considerations. The requirement to remove the fuselage base heat shield after each flight for main engine inspection makes refurbishment of the ablative shield attractive. The used shield could be removed and a new ablative shield installed after engine inspection is completed. The shield could then be refurbished remote from the Orbiter and need not be completed prior to the next flight. Coated columbium and RSI are marginally near the upper reuse temperature of 2500°F when preliminary ascent heating rates with less-than-ideal radiation view factors are considered.

The selected fuselage base TPS concept is shown in Figure 3.1.2.4-3. Flow impingement on the nozzles is avoided for entry angles of attack up to 40°F. At 40° angle of attack the flow at the end of the fuselage bottom is expected to expand almost 60° at various entry Mach numbers. The rationale for selecting the illustrated concept and SLA-561 ablator is summarized in Figure 3.1.2.4-4.

3.1.2.5 Influence of Maximum Structural Temperature on Combined Weight of TPS, Structure and ECLS Insulation - A study was conducted to determine the effect of maximum allowable substructure temperature on the combined weight of TPS, structure and ECLS insulation. Maximum temperature limits of 300° at end of landing roll out and 350° thereafter were selected. The analysis was based on the 1100 NM nominal design trajectory and thermal environment and considered an all-ablative and an all RSI-TPS concept, Figure 3.1.2-3.

The ablative concept considered was SLA-561 ablator bonded directly to aluminum substructure with 20 mils of 80 lb/ft³ adhesive (.13 lb/ft²). The RSI is bonded to a .25 inch foam pad for thermal and mechanical strain isolation, and the back side of the foam pad is bonded to aluminum substructure. The ablator and RSI surfaces are waterproofed with an organic coating. Behind the substrate is ECLS insulation that is used where necessary to avoid exceeding 300°F on the structure. Additional insulation that may be required in the payload bay or the cabin to maintain the structure inner surface to temperatures below 300°F was not included in this study.

Figure 3.1.2.5-1 summarizes the results of the analysis. A significant amount of weight is saved for both the ablative and RSI TPS concepts when the maximum allowable substructure temperature is increased from 200°F to 300°F (7925 lb for RSI and 3023 lb for ablator). Increasing the allowable temperature from 300°F to 350°F saves approximately 10% additional weight (786 lb for RSI and 342 lb for

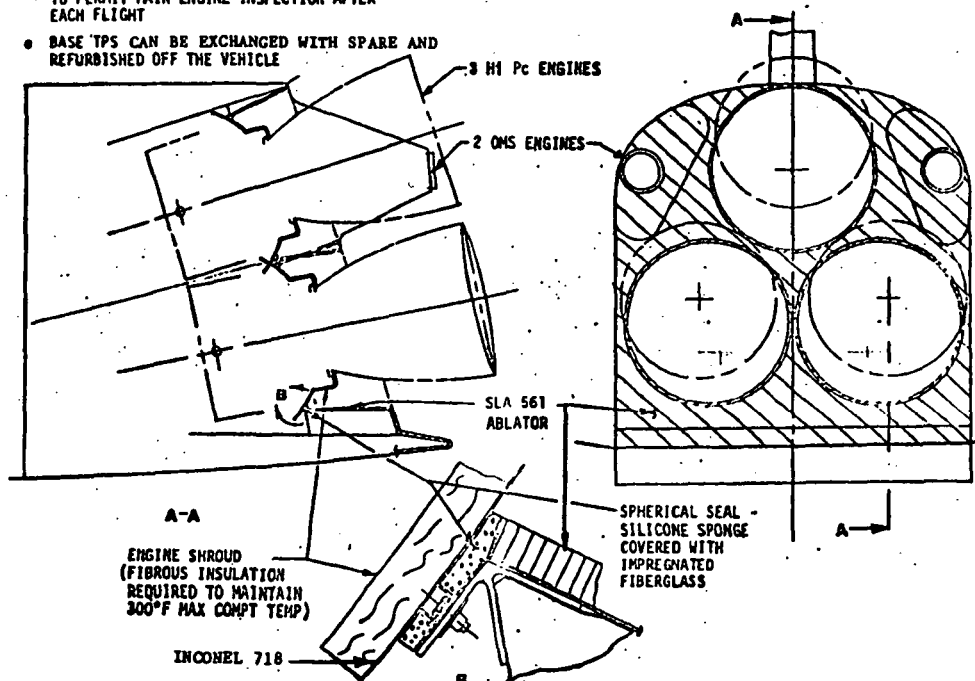
FIGURE 3.1.2.4-2
ORBITER TPS CANDIDATES
FUSELAGE BASE

CANDIDATE	ADVANTAGE	DISADVANTAGE
► LOW DENSITY ABLATORS	• MINIMUM RISK FROM HIGHLY UNCERTAIN HEATING • EXCELLENT TOLERANCE TO HIGH ACOUSTIC LEVELS • DESIGN CAN PERMIT SIMPLE POST-FLIGHT REMOVAL FOR ENGINE INSPECTION	• REFURBISHMENT AFTER EACH FLIGHT
METALLIC SHINGLES - FIBROUS INSULATION	• REUSABLE • EXCELLENT TOLERANCE TO HIGH ACOUSTIC LEVELS • EASY POST-FLIGHT REMOVAL FOR ENGINE INSPECTION	• SENSITIVE TO HEATING UNCERTAINTIES • COATED COLUMBIUM USAGE IS MARGINAL FOR PREDICTED HEATING ENVIRONMENTS
RSI IMBEDDED INTO HONEYCOMB	• SIMILAR CONCEPT USED ON SATURN-V • POTENTIALLY REUSABLE	• UNKNOWN ACOUSTIC CAPABILITIES FOR SHUTTLE ENVIRONMENTS • REQUIRES STRAIN ISOLATION FROM STRUCTURE • MARGINAL MAXIMUM TEMPERATURE CAPABILITIES FOR SHUTTLE (UNLESS ZIRCONIA HCF IS DEVELOPED)

► SELECTED

FIGURE 3.1.2.4-3
FUSELAGE BASE TPS

- MAIN ENGINE NOZZLES SHIELDED FROM ENTRY HEATING
- KEY FEATURE OF BASE TPS IS EASY REMOVAL TO PERMIT MAIN ENGINE INSPECTION AFTER EACH FLIGHT
- BASE TPS CAN BE EXCHANGED WITH SPARE AND REFURBISHED OFF THE VEHICLE



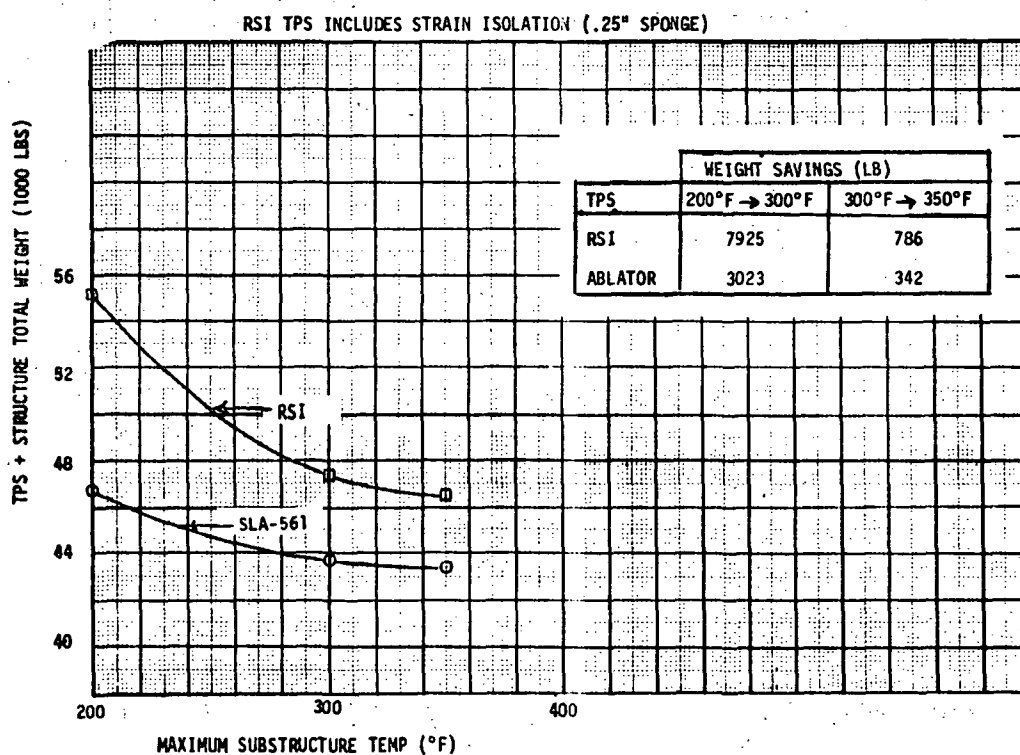
15 MARCH 1972

FIGURE 3.1.2.4-4
ORBITER FUSELAGE BASE TPS
SELECTED CONCEPT AND RATIONALE

CHOICE OF SLA-561 ABLATOR AND DESIGN CONCEPT

- MINIMIZES RISK FROM ASCENT AND ENTRY HEATING ENVIRONMENTS
- PROVIDES EXCELLENT TOLERANCE TO HIGH ACOUSTIC LEVELS
- PERMITS SIMPLE POST-FLIGHT REMOVAL FOR ENGINE INSPECTION
- REDUCES REFURBISHMENT COSTS WITH SIMPLE SHIELD REMOVAL
- PERMITS ENGINES TO GIMBAL
- PROVIDES HOT GAS SEALS ON GIMBALS TO MINIMIZE LEAKAGE
- AVOIDS STRAIN ISOLATION REQUIREMENT
- AVOIDS FLOW IMPINGEMENT ON NOZZLES DURING ENTRY

FIGURE 3.1.2.5-1
INFLUENCE OF MAXIMUM BONDLINE TEMP ON TPS WEIGHT
○ ALUMINUM SUBSTRUCTURE
○ 1100 N.M. NOMINAL DESIGN TRAJECTORY



15 MARCH 1972

ablator). The ablator and RSI weights include the effect of the heat sink of local structure.

Figure 3.1.2.5-2 graphically illustrates the structural temperature trade-off. Increasing the maximum allowable substructure temperature from 300°F to 350°F requires an increase in structural weight due to a decrease in the aluminum mechanical properties with time at temperature, thus reducing the net total weight saved.

3.1.2.6 Effect of Crossrange on TPS Weight - A study was conducted to determine the sensitivity of orbiter ablator and RSI TPS weight to reducing crossrange from 1200 NM. The study defines TPS weight savings only (ablator or RSI water-proof coating and adhesive bonds). Results of this study (Figure 3.1.2.6-1) show that approximately 5700 lb of all-ablative TPS would be saved when the crossrange requirement is reduced from 1200 NM to 430 NM. Approximately 4600 lb of TPS would be saved if the entire surface were covered with RSI.

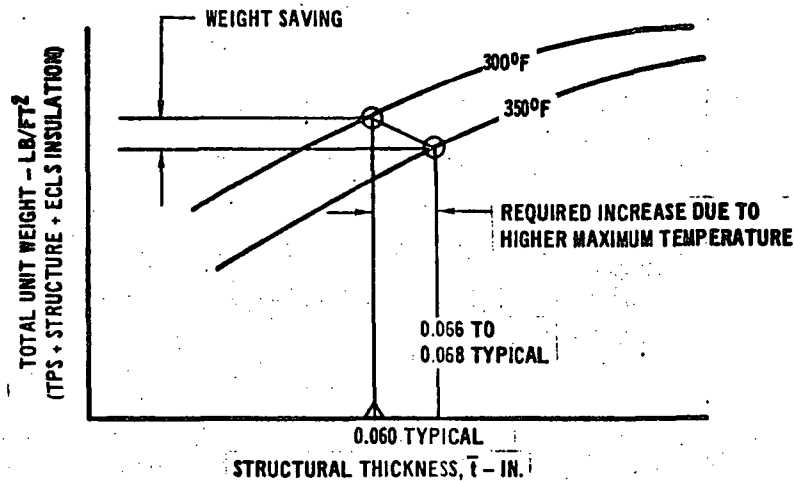
The all-ablative concept was SLA-561 bonded directly to the aluminum structure with 20 mils of 80 lb/ft² adhesive (.13 lb/ft²). The RSI concept utilized RSI (mullite HCF) bonded directly to the aluminum substructure with .17 lb/ft² of adhesive. RSI and ablator surfaces are waterproof protected with an organic coating. Strain isolation foam for RSI was not included in this analysis since only weight differences from a reference 1200 NM trajectory were of interest. Total TPS units weights with or without strain isolation have been found to be very similar (approximately 0.10 psf) so that this consideration would have only a minor effect.

15 MARCH 1972

FIGURE 3.1.2.5-2

MAXIMUM STRUCTURAL TEMPERATURE

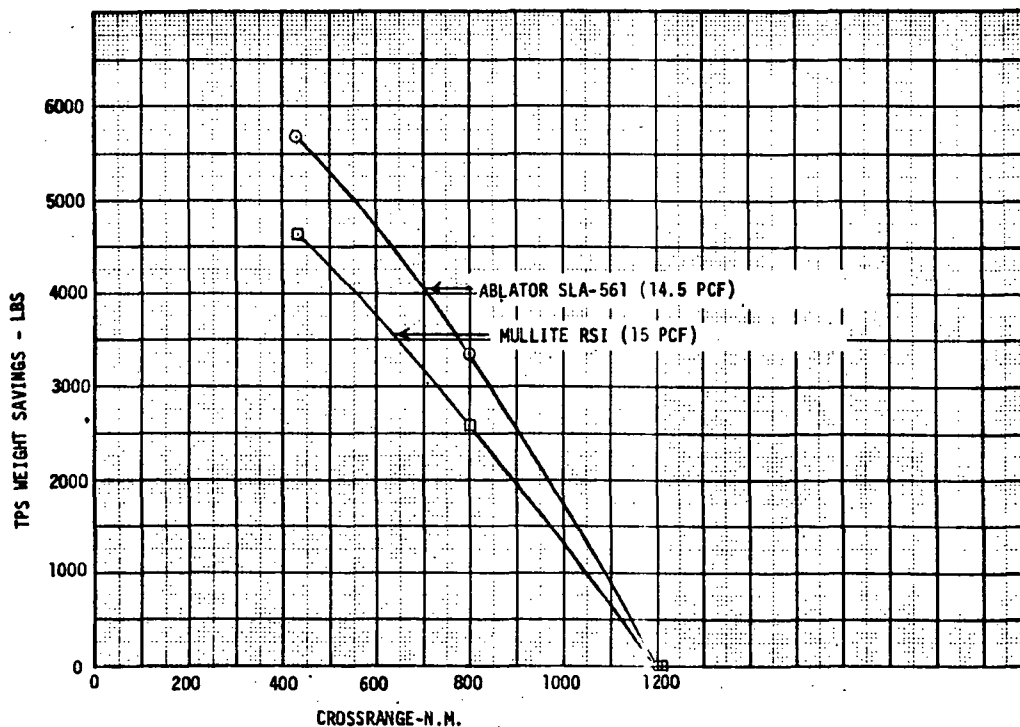
- SLA-561 ABLATOR OR RSI TPS
- 2024 ALUMINUM STRUCTURE
- 3.5 PCF MICROQUARTZ ECLS INSULATION (FUSELAGE ONLY)



CONCLUSION: MODEST WEIGHT SAVING FOR ALUMINUM STRUCTURAL TEMPERATURE ABOVE 300°F

FIGURE 3.1.2.6-1

TPS WEIGHT SAVED FOR REDUCED CROSSRANGE



15 MARCH 1972

3.1.3 Mechanical Systems

3.1.3.1 External Tank Separation - The interstage attachment system includes provisions for structural mating of the Orbiter and the HO tank with subsequent staging release and separation.

Interstage tie loads are transferred at two locations: a forward location for vertical and lateral loads only and an aft location for vertical, lateral, and drag loads.

Separation is effected from the Orbiter side only. GN_2 pressure is applied against pistons of the forward and aft load fittings, followed by pyrotechnic release of the forward and aft vertical load struts, causing separation to occur.

The design criteria are summarized as follows:

- o 100 ft or greater distance between tank and orbiter prior to retrograde
- o Low force, short time separation system, for minimum deorbit attitude error
- o Active release/separation components on Orbiter; minimum components on tank side of interface
- o No attitude control system on tank

The design will feature:

- o 1 1/2 - ft/sec tank separation velocity
- o Propellant residual pump for c.g. control
- o Sealing of propulsion lines sealed to prevent disturbance torques
- o Redundant, parallel control valves
- o Separation cylinder fail-safe sensors
- o Time delay retrograde signal

The design will provide operational redundancies as follows:

Function	Control	Power Source		
		Operating System	2nd System (Fail Operational)	Emergency System (Fail Safe)
Release	Hard Wire	Pyro Pin Puller	Redundant Pyro Pin Puller	Dual Igniters
Separation	Hard Wire	GN_2 Pneumatic System	Parallel Control Valves	ACPS System

3.1.3.2 Landing and Deceleration - The Landing/Deceleration System consists of conventional single chamber air-oil tricycle struts and pneumatic rubber tired wheels, with a steerable nose wheel assembly and anti-skid brakes on the main wheels.

15 MARCH 1972

Dual wheel arrangements are used on both the main and nose gears. This system must provide for horizontal landing and stopping, safe reliable ground control, and ferry flight takeoff.

The main gear is located in the wing just forward of the forward center spar and the nose gear at the forward payload bay bulkhead. All gears have free fall capability. The extension braking and steering systems are electrically and/or mechanically controlled and hydraulically operated. The anti-skid brake control is supplied by an analog system. All subsystems basically have FO/FS redundancy.

The design criteria are summarized as follows:

- o Landing vertical load factor - 2.5g
- o Free fall gear geometry
- o Brake and tire design life - 5 normal landings and one rejected take-off
(normal landing 40K payload, V_{TD} 175 knots)
- o Wheel wells thermally controlled and unpressurized
- o Struts and tires initially pressurized only
- o Gear extension time - 5 seconds
- o Subsystems basic design - pilot in loop
- o Airline type checkout (minimum instrumentation)
- o Abort landing: 65,000 lb payload, 190 knot V_{TD} , sea level hot day,
10,000 ft dry runway stop
- o Drag chute only if required to meet stopping distance

The design will provide operational redundancies as follows:

Function	Control	Power Source		
		Operating System	2nd System (Fail Operational)	Emergency System (Fail Safe)
Unlock latches & Doors	*Hardwire & Manual Cable	Hyd. System 2	Hyd. System 4	Manual Cable
Lower Gears	Hardwire	Hyd. System 2	Free Fall	Free Fall
Raise Gears	Hardware	Hyd. System 2	Gear Down	Gear Down
Brakes	*Mechanical & Electronic	Hyd. System 2 Hyd. System 4	Hyd. System 4 Hyd. System 2	Hyd. System 3 Hyd. System 3
Steering	*Mechanical	Hyd. System 2 Hyd. System 4	Hyd. System 4 (Red. Cap.) Hyd. System 2 (Red. Cap.)	Differential Braking

* Unmanned operation will require an avionic controlled power source added to system

15 MARCH 1972

3.1.3.3 Docking Mechanism - The docking system provides the Orbiter with capability to capture mating spacecraft and absorb the relative energy.

Docking maneuvers are accomplished with the Orbiter ACPS. The docking mechanism, located in the Orbiter nose, provides on-axis docking/undocking and direct visibility from the cockpit. Dock/undock functions are controlled from the Orbiter side of the docking interface. Docking aids include optical alignment with the target and illumination.

The docking mechanism is androgynous and consists of a four spade alignment ring mounted on eight extendable shock attenuators. The attenuator/actuators are fluid damped and are extended and retracted by redundant nitrogen sources. Extension pressure is regulated to the mass to be docked. After initial docking capture, the shock attenuators are retracted to permit engagement of twelve latches. A seal is provided for pressurized crew/cargo transfer.

The design criteria are summarized as follows:

- o Capability to capture mating spacecraft and absorb relative energy
- o Damped connection capability between mated spacecraft
- o Rigidized connection capability between mated spacecraft
- o Shirt sleeve transfer capability between mated spacecraft
- o Androgynous mechanism
- o Visual docking control and determination
- o Capability to redock after a misdock
- o Automatic interface attachment and release
- o Capability for removal/replacement of docking assembly

The design will feature:

- o Shock absorber/damper gas pressure, regulated to mass to be docked
- o GN_2 for extension of the shock absorber/actuator and GN_2 with metered fluid for damping
- o Redundant valving system for GN_2 with mechanical override for vent valves
- o No rigidizing possible until three series capture indication switches are activated
- o Collimated reticle optical sight for target alignment
- o Docking assist light located on docking adapter
- o Indicator lights for "capture," "rigidized," and "nose cap latched" and docking assist light on.

15 MARCH 1972

The design will provide operational redundancies as follows:

Function	Control	Power Source		
		Operating System	2nd System (Fail Operational)	Emergency System (Fail Safe)
Latch/Unlatch Nose Section	Hard Wire	Electrical/Mechanical Actuator	Electrical/Mechanical Actuator	Manual Latch
Actuate Nose Section	Hard Wire	Electrical/Mechanical Actuator	Electrical/Mechanical Actuator	Manual Closure
Docking Frame Extension/Retraction	Hard Wire	N ₂ Actuator (Capable of Operating with 5 of 8 Actuators)	N ₂ Actuator	N ₂ Actuator (Nose Section can close and latch with frame extended)
Capture-Latch Release	Hard Wire	Electrical/Mechanical Actuator	Electrical/Mechanical Actuator	Manual
Rigidize Latch/Unlatch	Hard Wire	Electrical/Mechanical Actuator	Electrical/Mechanical Actuator	Manual Latch

3.1.3.4 Payload Handling - Orbiter payload handling accommodations include payload bay door and radiator actuation and latching; payload attach and release mechanisms; payload deployment/dock/retrieval actuation; and Orbiter to payload IVA and EVA access.

Opening the payload bay door and radiator exposes the full length and diameter of the payload. The payload is secured to the Orbiter at five of six load carrying locations, arranged in a statically determinate pattern.

Payload deployment/dock/retrieval is accomplished through the use of twin manipulator arms (controlled manually with computer assist) that stow on the drive links of the bay door and radiator. Opening the door and radiator rotates the manipulators clear of the stowed payload to the position from which they are deployed.

Direct vision is provided into the payload bay. Payload handling aids include bay illumination, directable lights for payload capture, and a TV camera and light at the manipulator end effector.

The design criteria are summarized as follows:

- o Payload Access
 - o Access panels for umbilicals and ground servicing
 - o Payload bay door open on-pad vertical in 40 sec.
 - o Standardized effluent vents

15 MARCH 1972

- o Payload Attachment
 - o Payload envelope clearances - 6 inches side, 3 inches top/bottom
 - o Standard attachments
 - o Non-redundant attachment load paths
 - o Remote actuation of latch system
- o Payload Handling
 - o Design weight goal for manipulator system $\leq 2,000$ lb
 - o Manipulator tip - Rate 0 - 0.2 ft/sec
 - Position - 4 in. radius sphere (± 2 in.)
 - Deflection (with 65,000 lb. payload) ≤ 2.00 in.
 - o Tilt table pallet mounted required for certain payloads

The design will also feature:

- o Hatch into compartment (via airlock) for manned access to payloads
- o Manual backup of payload latches
- o Manual backup of door and radiator rotation

The mechanical subsystem, as designed, provides the operating redundancies that are delineated in Figure 3.1.3-1.

FIGURE 3.1.3-1

OPERATIONAL REDUNDANCIES

Function	Control	Power Source		
		Operating System	2nd System (Fail Operational)	Emergency System (Fail Safe)
Cargo Bay Door Latch/Unlatch (18 Latches)	Hard Wire	Electro/Mechanical Actuator	Electro/Mechanical Actuator	Manual Handcrank
Cargo Door Rotation	Hard Wire	Electro/Mechanical Actuator	Electro/Mechanical Actuator	Manual Handcrank
Radiator Rotation	Hard Wire	Electro/Mechanical Actuator	Electro/Mechanical Actuator	Manual Handcrank
Radiator Latch/Unlatch	Latched/Unlatched by Cargo Door - No System Required			
Payload Structural Latch/Unlatch	Hard Wire	Electro/Mechanical Actuators (3 places)	Electro/Mechanical Actuators (3 places)	EVA Handcrank
Payload Deployment	Hard Wire	Manipulator #1	Manipulator #2	None
Manipulator Unlatch/Latch	Hard Wire	Electro/Mechanical Actuators	Electro/Mechanical Actuators	EVA Manual
Manipulator Deployment & (Shoulder) Stowage	Hard Wire	Electro/Mechanical Actuators	Electro/Mechanical Actuators	IVA Manual
*Manipulator Working Motions; 6 Rotating Joints	Hard Wire	Electro/Mechanical o Motor o Gears o Clutch o Feedback Pots	Redundant Motors, Clutches, & Pots	Jettison at Shoulder Joint

* Note: Entire system redundant - Twin Boom System

3.2 Propulsion

The Orbiter propulsion group consists of the main propulsion, reaction control, orbit maneuver, airbreathing, and solid propellant abort systems. These systems provide the propulsive energy required for orbiter ascent, on-orbit and entry reaction control, landing assist, and off the pad abort.

3.2.1 Main Propulsion - The Orbiter main propulsion system consists of three 472K (vac) thrust high Pc engines, an external tank containing liquid oxygen and liquid hydrogen propellants, and supporting propellant control subsystems, including feed, propellant conditioning, pressurization/vent, pneumatic control, fill/drain, and propellant management. In the study extension, trade studies were performed to define the characteristics of the main propulsion system for both the series burn and parallel burn orbiter configurations, wherever there are significant differences from the Phase B configuration.

The 472K engine selected for both configurations is the result of a series of engine trade studies documented in Section 4.4 of Part A, Shuttle system. External tank configuration studies leading to the selection of a separate bulkhead tank for series burn and a common bulkhead tank for parallel burn are summarized in Sections 4 and 5 of this volume. Main propulsion topics related to the tank configuration development, such as propellant utilization, residuals, pressurization and vent, are covered in the tank section.

The subsequent paragraphs summarize LOX feedline and propellant conditioning studies for the Orbiter configurations.

3.2.1.1 Orbiter Feed System Dynamics - Series Burn - The dynamic characteristics of the Orbiter LOX feed system were investigated to determine the minimum pressure available to meet the pump NPSP requirements during the engine start transient. This analysis was conducted using a MDC computer program. This program utilizes the method of characteristics, together with feedline elasticity and fluid compressibility, to solve for the pressures and flowrates throughout the feed system as a function of time. The program has been used previously to simulate the Thor and the S-IC booster start transients with good agreement with test data.

The computer model includes the propellant tank, sump, isolation valves and all line segments between the tank and the pump inlet. Since bellows and expansion joints which absorb flow energy and reduce the magnitude of the pressure oscillations were neglected, the model results are conservative. The frictional resistance

15 MARCH 1972

of the feedlines and flow resistance associated with bends, tees, and valves were lumped at discrete locations in the feed system. The pump flowrate history and vehicle acceleration history which were input to the program were determined from the engine thrust history, except that the flowrate history between the 15% and 100% power levels was specified by a constant propellant acceleration rate of 1350 lb/sec². Additional data input to the program included propellant properties, feedline lengths, diameters and wall thicknesses, propellant tank description and component datum height.

The results of the simulation are presented in Figure 3.2.1-1, which shows the pump inlet pressure transient for each of the three engines. [REDACTED]

[REDACTED]

[REDACTED]

[REDACTED]

The predominant frequency exhibited by the results is 3.5 Hz, the first organ pipe mode between the tank and the pump. Cross engine modes are essentially non-existent due to feed system symmetry and the fact that the engines were started simultaneously.

3.2.1.2 Orbiter Feed System Recirculation and Geyser Suppression--Series Burn -

The major configuration differences affecting chilldown between the current series-burn Orbiter vehicle and the Orbiter as defined in the Phase B Final Report include: (1) disposable propellant tanks; (2) three HiPc engines in place of two; and (3) revised line sizes and routing. (Engine heat loads (BTU/sec) and feedline/manifold heating rates (BTU/hr ft²) have been maintained at the Phase B values.) The main impact of the configuration differences is in the flow rates and temperature profiles for the recirculation system. There is also at least one significant hardware change: the LH₂ recirculation pumps move onto the Orbiter side of the feed system disconnects. This approach permits reuse of expensive hardware that has readily attainable high lifetimes.

The trade studies and selections made during Phase B with respect to natural recirculation through the engine for chilldown, LOX feedline insulation, and chill-down flow direction remain valid for the current version, because the heating rates have not been changed.

The Murphy geyser correlation confirms that there is no geysering potential in the LH₂ feed system or in the lower LOX feed system below the crossover manifold.

15 MARCH 1972

However, the twin LOX feedlines which run the length of the LH_2 tank in a relatively high heat input environment will require geysering protection. The proposed method of suppression remains the same as in the Phase B report; namely circulation from the tank into the downcomer line, through the crossover manifold, up the riser line, and back into the LOX tank sump. The circulation is initiated by helium injection and maintained by the LOX density gradient. No propellant in either line exceeds $168^\circ R$, so start conditions for the engine inlets are automatically maintained. The topping flow to replace boiloff must be conditioned to this temperature or lower, and flows to the tank through the riser line. The schematic of the LOX system is shown on Figure 3.2.1-2 and Table 3.2.1-1 contains the significant performance characteristics. Chillover flow rate to provide the required propellant temperature at the engine inlets is 17.6 lbs/sec, which is tapped off the main flow.

The LH_2 chillover takes propellant directly to the recirculation pump inlets from the main fuel tank. It is then passed through the engines to accomplish chillover, and returns to the tank through the feedlines. The maximum temperature of the LH_2 in the feedline is $40.5^\circ R$, which provides a minimum $1.3^\circ R$ performance margin. The system is schematically shown in Figure 3.2.1-3 and Table 3.2.1-1 contains the significant performance characteristics.

3.2.1.3 Orbiter Feed System Dynamics - Parallel Burn - The dynamic characteristics of the LOX feed system for the parallel burn Orbiter configuration may be inferred from the results of the series burn configuration. As only one downcomer is used, the propellant in the downcomer would experience double the drop in pump inlet pressure of the series burn system if a simultaneous start were employed. To limit the drop in pump inlet pressure to acceptable levels and maintain a reasonable line size, the engines are sequence started. The amount of area increase required was determined from the incompressible equation, which relates change in pressure to change in flowrate. For the parallel burn configuration, this results in a feedline diameter of approximately 17 inches.

3.2.1.4 Orbiter Chillover and Geyser Suppression - Parallel Burn - In the parallel burn configuration, the main oxidizer feedline passes through the fuel tank. Since the temperature differential between the LOX and LH_2 favors heat transfer away from the LOX, zero heat input to that portion of the LOX feedline was baselined as a conservative assumption. [REDACTED]

[REDACTED]

[REDACTED]

15 MARCH 1972

[REDACTED]

The LH₂ chilldown system for parallel burn is similar to the LOX system with one important difference. [REDACTED]

[REDACTED]

In both the LOX and LH₂ feed systems, geysering will not exist, but the topping flow must again be conditioned to assure start quality propellant at all times once the chilldown sequence begins. The path for propellant conditioning flow runs down the main feed system, through the engine passages into the chilldown collection manifold, and overboard through a base liftoff disconnect. An adequate pressure head exists to drive this flow without the use of recirculation pumps. Conditioned topping flow maintains the propellant level in the tank.

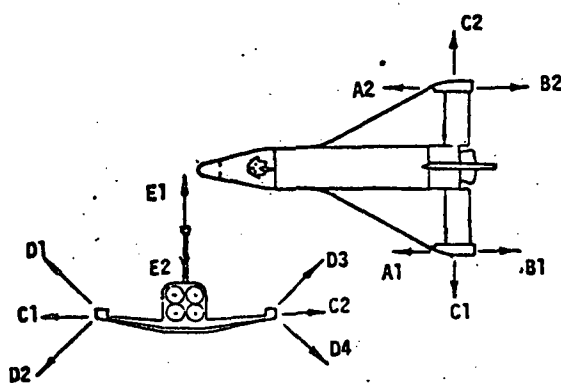
[REDACTED]

15 MARCH 1972

3.2.2 Reaction Control System (RCS) - The RCS controls attitude of the Orbiter and also provides three-axis vernier translation velocity. The system is three independent modules, each containing a group of engines, propellant, pressurant and associated controls. One module is located at the tip of the vertical tail, the others mounted on each wing tip. The module location, number of engines, engine arrangement, and function are shown in Figure 3.2.2-1.

The RCS is pressure fed using monopropellant hydrazine decomposed by spontaneous catalyst. Figure 3.2.2-2 is a schematic of the system. Ambient helium stored at high pressure is used to pressurize a positive expulsion propellant storage tank in each module. The propellant is distributed to the engines on demand as required to satisfy the mission requirements. The capability of the RCS is summarized in Figure 3.2.2-3.

FIGURE 3.2.2-1
BASELINE RCS POD & ENGINE ARRANGEMENT

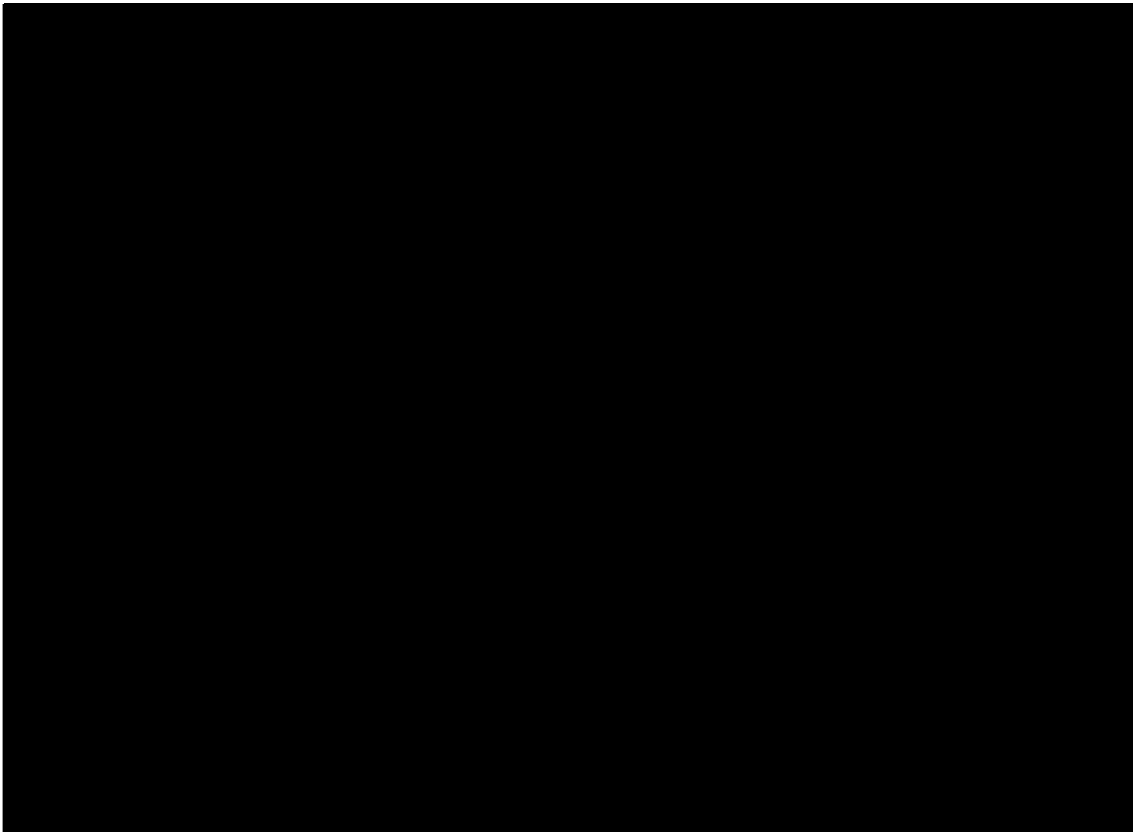


ARRANGEMENT		ENGINE NO. PER LOCATION				
THRUST	NUMBER	A	B	C	D	E
400	32	6	6	6	8	6

CONTROL AXIS		ON ORBIT	REENTRY
PITCH	UP	E1	E1
	DOWN	E2	E2
YAW	RIGHT	A2, B1	A2, B1, C2
	LEFT	A1, B2	A1, B2, C1
ROLL	CW	D2, D3	D2, D3
	CCW	D1, D4	D1, D4
AXIAL TRANSLATION	FWD	B1, D2	-
	AFT	A1, A2	-
LATERAL TRANSLATION	RIGHT	C1	-
	LEFT	C2	-
VERTICAL TRANSLATION	UP	D2, D4	-
	DOWN	D1, D3	-

15 MARCH 1972

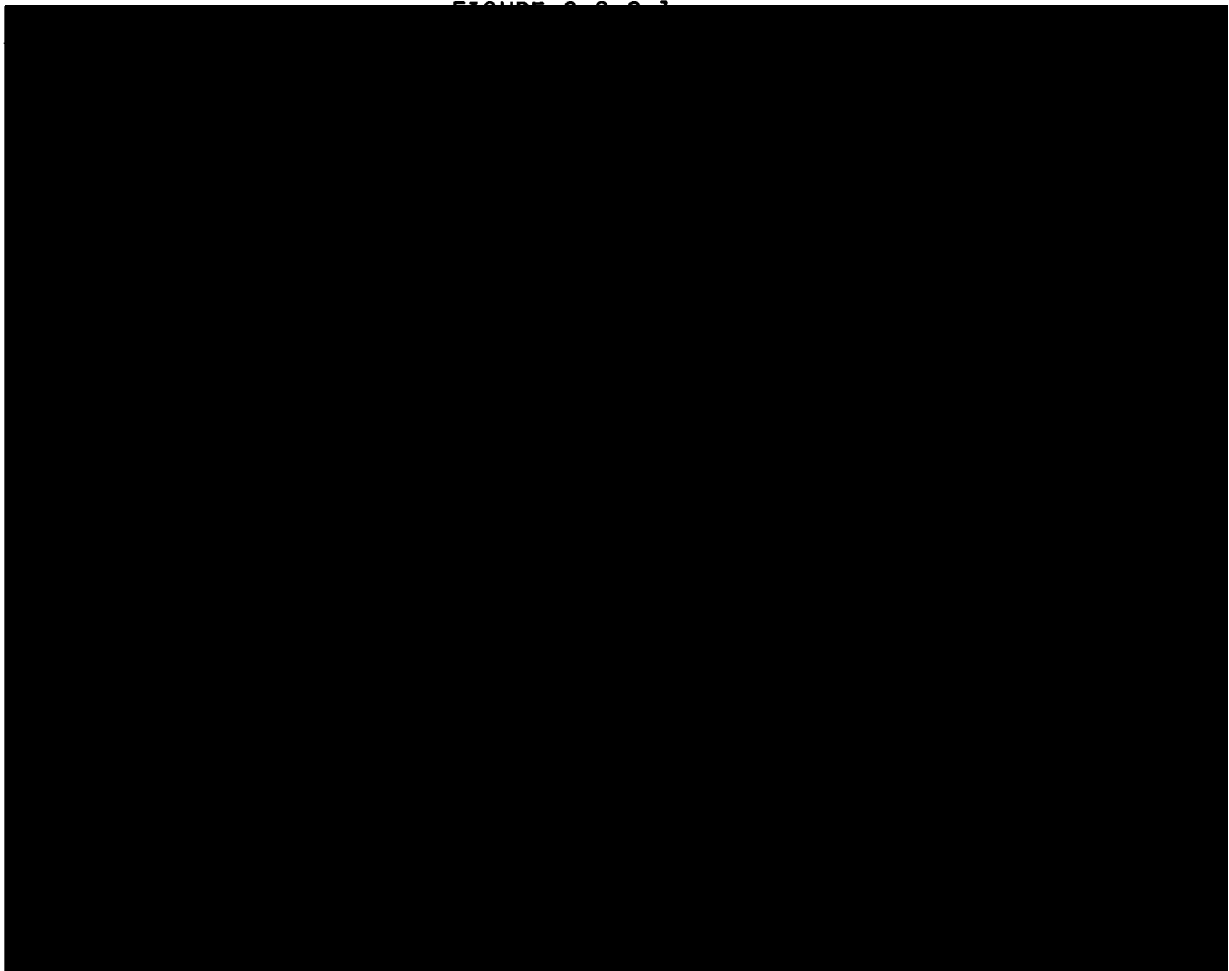
The module location for the podded system was provided in the follow-on baseline. Using this arrangement, engine thrust levels were analyzed to define the most nearly optimum system. Figure 3.2.2-4 is a summary of engine requirements and maneuver capability. The impulse requirements are lowest at the minimum thrust level which meets the acceleration requirements. The 400 lb level matches the acceleration requirement approximately. This level is somewhat marginal for translations; however, these requirements are least critical. The impulse savings warrant the slight undersized thrust for translation. The number of engines was selected from redundancy considerations. The baseline arrangement allows the total loss of any two RCS engines without the loss of the capability to provide entry attitude control.



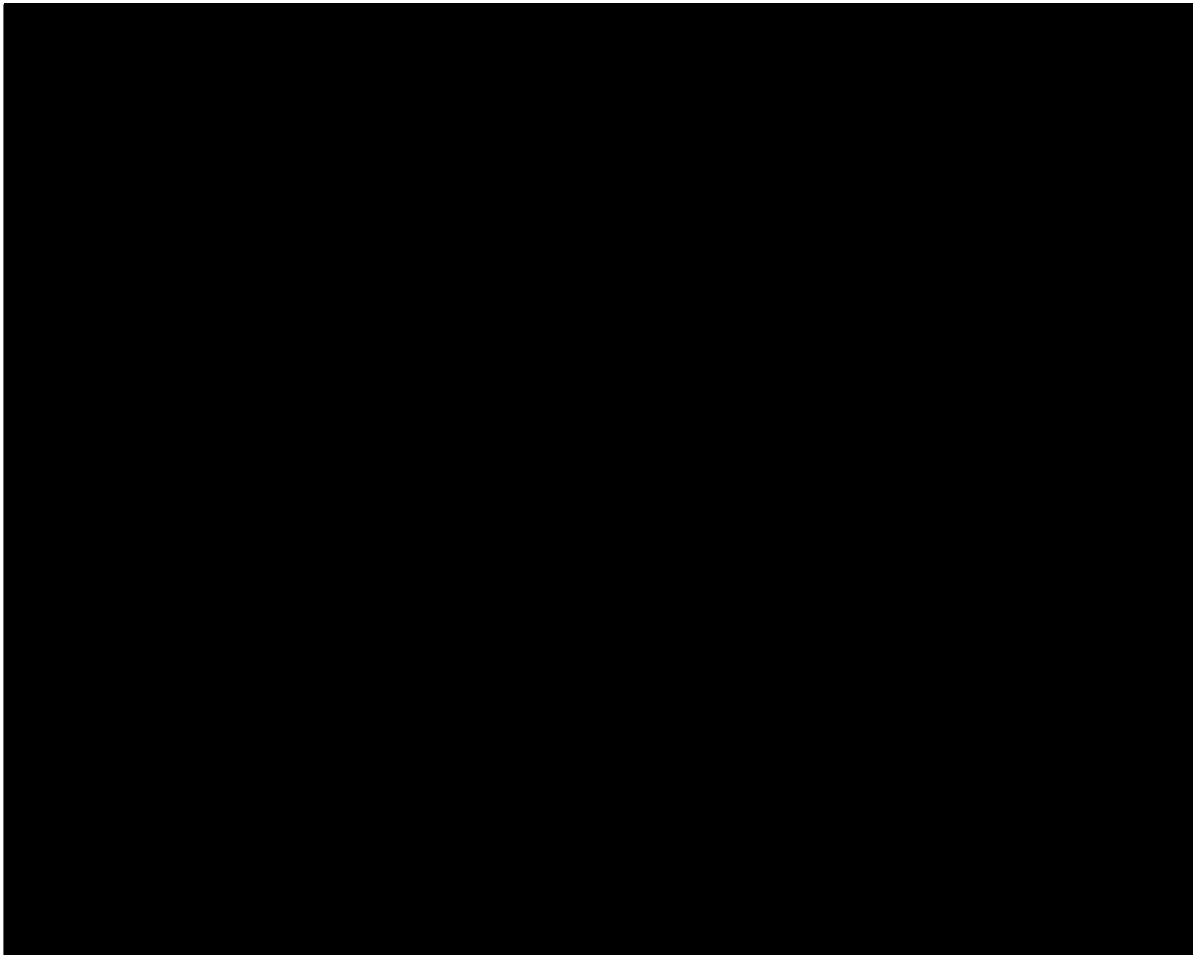
15 MARCH 1972

3.2.3 Orbit Maneuvering System - The major velocity changes of the Space Shuttle Orbiter from orbit insertion to retrograde, are performed by an orbit maneuvering system (OMS). The OMS consists of two independent modules. Each module contains pressurant and propellant supplies, controls and plumbing, and a single OMS engine. The modules are located within the mold line of the Orbiter's base area, one on each side of the vehicle's centerline, reference Figure 3.2.3-1. The system uses storable bipropellants, nitrogen tetroxide and Aerozene-50, to provide the total impulse required for the Orbiter maneuvers. Two 3500 lb thrust modified Lunar Module Ascent Engines (LMAE), one in each module, are used. Figure 3.2.3-2 provides a schematic of a single module; the second module is identical. As shown, separate pressurization supplies are used for the fuel and oxidizer propellants to eliminate potential vapor mixing problems. Redundant components are provided to satisfy Space Shuttle failure criteria.

The module tanks are sized to provide a 1000 ft/sec orbiter velocity change; i.e., 500 ft/sec from each module. The phantom lines shown in the schematic of Figure 3.2.3-2 shows the components which would be added to provide for additional ΔV capability.



15 MARCH 1972



3.2.3.1 Propellant Trade Study - A trade study was conducted to select the best propellants for use in the Orbiter OMS and RCS. During the Phase B studies, cryogenically stored oxygen and hydrogen propellants were selected for the baseline. Later a smaller Orbiter was selected, with lower design requirements, and the Phase B system was reevaluated. System schematics were generated to meet the new design requirements and component weight models were developed.

Using optimized design conditions, the system weights and volumes were defined and costs for systems operations, maintainability, RDT&E and investments were estimated.

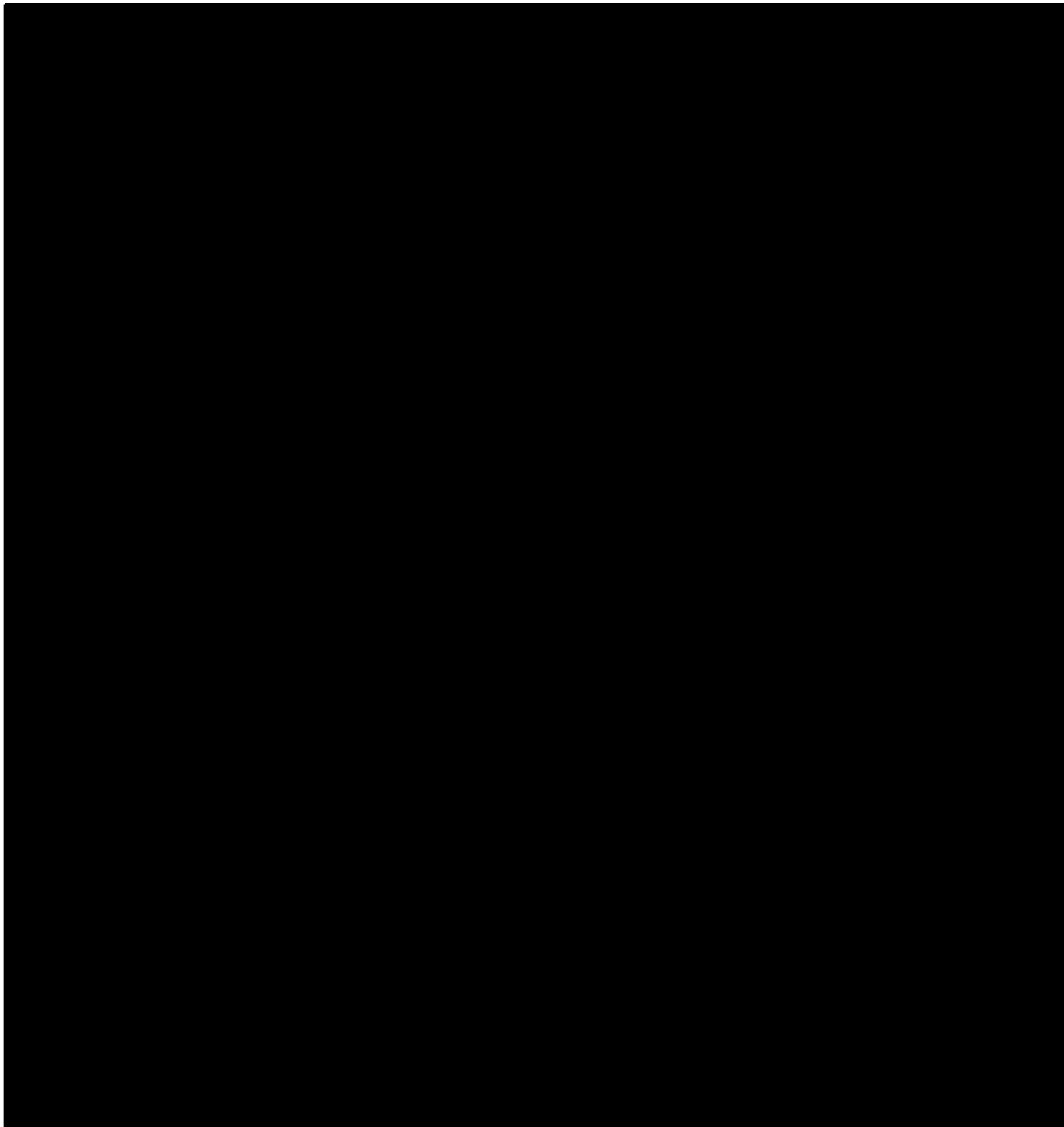
The propellants selected for this trade study consisted of: (1) cryogenically stored oxygen and hydrogen, (2) nitrogen tetroxide and either monomethyl or aerazine-50, and (3) monopropellant hydrazine. A matrix of system combinations was developed. In addition, since the Phase B design integrated the RCS and OMS with the APU, ECS, and fuel cells, each RCS and OMS propellant combination included these items.

15 MARCH 1972

The resulting system weights for each combination are presented in Figure 3.2.3-3. In this analysis, the Phase B integrated O_2/H_2 OMS/RCS was used as a reference. The O_2/H_2 and storable bipropellant OMS weights shown were based upon the performance of the RL-10 and LMAE engines, respectively.

The integrated O_2/H_2 system proved to be approximately 1100 lb lighter than the next best candidate.

However, to completely assess the relative weight of the candidates, it is necessary to define the impact of the installed systems on vehicle structure, electrical power and thermal control. In addition, assuming the use of common propellants for the Booster and Orbiter RCS, the effect of the Booster RCS weight must be included to bring the integrated systems to a launch vehicle level. The impact of



15 MARCH 1972

these considerations is presented in Figure 3.2.3-4. As shown, the integrated cryo system is still the lightest. The storable configuration shows approximately a 4000 lb weight disadvantage, and monopropellant shows approximately a 2000 lb weight disadvantage.

A payload cost of \$5000/lb was used to define the cost of resizing the launch vehicle for an equal payload. The costs for system development, investment and operations, were also estimated. As shown in Figure 3.2.3-5, the integrated O_2/H_2 system is not cost competitive with the other candidates. With the same RCS propellant, the cryogenic and bipropellant OMS have essentially the same overall cost. The bipropellant OMS system, however, includes a modified LMAE engine, incorporating a replaceable ablative "cartridge" and modifying the valves to allow reuse without an overhaul. Without this modification, the operational cost (estimated @\$104 M) for refurbishment of the LMAE engine results in a significant program cost increase.

The storable system shows significantly lower costs during the initial RDT&E and investment phases. In addition the large propellant tank volumes inherent with an O_2/H_2 OMS will affect OMS pod design, installation, and cost. For these reasons the bipropellant OMS with a modified LMAE engine was selected as the baseline design.

As shown in Figure 3.2.3-5, a monopropellant hydrazine RCS is approximately \$20M lower than the bipropellant RCS. Although this cost analysis was performed for a nonmodular RCS, the factors would be of the same magnitude for a modular RCS. Thus, monopropellant hydrazine was selected for the baseline modular RCS design.

FIGURE 3.2.3-4
HO ORBITER
INTEGRATED SYSTEM/VEHICLE WEIGHT COMPARISON

SYSTEM - OMS - ACPS - APU ITEM	H_2/O_2	H_2/O_2		NTO/A-50	
	H_2/O_2	NTO/A-50 N_2H_4	N_2H_4 N_2H_4	NTO/A-50 N_2H_4	N_2H_4 N_2H_4
SYSTEM WEIGHT	33,352	34,399	36,982	38,592	41,030
WING POD WEIGHT	70	120	120	120	120
ELEC. POWER WEIGHT	-----	512	171	512	171
PROP. THERMAL CONT.	-----	100	50	100	50
BOOSTER ACPS - NET	-----	-374	-320	-354	-320
PAYLOAD IMPACT					
TOT. INT. SYSTEM	33,422	34,821	37,003	38,970	41,051

FIGURE 3.2.3-5
ACPS/OMS/APU COST COMPARISON - ORBITER ($W_0 = 230,000$ LBS)
MODIFIED LEM-A ENGINES

SYSTEM - ITEM	OMS ACPS APU	H ₂ /O ₂	H ₂ /O ₂	H ₂ /O ₂	NTO/A-50	N ₂ H ₄
		H ₂ /O ₂	NTO/184H	H ₂ H ₄	NTO/A-50	N ₂ H ₄
		H ₂ /O ₂	N ₂ H ₄	N ₂ H ₄	N ₂ H ₄	N ₂ H ₄
SUBSYSTEM DEVELOPMENT - Δ\$M		(REFERENCE)	38	38	-20 ⁽¹⁾	-2 ⁽¹⁾
OMS	-141		-160	-127	-161	
ACPS	-18		-18	-18	-18	
APU	14		14	14 ⁽²⁾	14 ⁽²⁾	
ECLS/FUEL CELLS	-35		-45	-45 ⁽²⁾	-54 ⁽²⁾	
SUBSYSTEM SUPPORT	-142		-171	-196	-221	
TOTAL SUBSYSTEM RDT&E - Δ\$M						
INVESTMENT (4 ORBITERS) - Δ\$M	-18		-23	-21 ⁽³⁾	-27 ⁽³⁾	
PAYLOAD EFFECT @ 5000 \$/LB	7		18	28 ⁽⁴⁾	38 ⁽⁴⁾	
NET Δ COST - \$/M	-153		-176	-189	-210	
OPERATIONS PHASE - Δ\$M						
ACPS MAINTENANCE	8		12	8	12	
OMS ENGINE REFURBISHMENT				17 ⁽⁵⁾	17 ⁽⁵⁾	
OPERATIONS						
PROPELLANT	6	5	18	18		
GSE	9	4	9	9		
TOTAL NET Δ COST - Δ\$M	-130	-155	-137	-154		

- (1) \$7M INCREASE FOR DEVELOPMENT AND QUAL FOR MODIFIED ENGINE
(2) \$2M INCREASE FOR SUBSYSTEM SUPPORT OF MODIFIED ENGINE PROGRAM
(3) CHANGE IN CHAMBER PRICE NEGLECTED
(4) IMPACT ON VEHICLE SIZE TO MAINTAIN PAYLOAD; PRIMARILY RDT&E COST
(5) CHAMBER REPLACED EACH FLIGHT; INJECTOR EACH FIFTH FLIGHT; ENGINE EACH TENTH FLIGHT

3.2.3.2 Alternate OMS Engines - The storable propellant baseline included two existing Lunar Module Ascent Engines. This ablatively cooled engine is life limited and has been demonstrated only to 1000 second continuous burn times. This total time, with a number of burns and subsequent soak-out, is required for one 1000 fps design mission. Therefore, the engine is not capable of completing some of the potential missions. Thus, a group of alternates were examined for an OMS engine, as shown in Figure 3.2.3-6. Candidates include existing, new high thrust, and new low thrust clustered engines.

As shown, payload effects are relatively small. Otherwise, the most significant factors are low thrust-to-weight with failed engines for the LMAE and RCS engines. Figure 3.2.3-7 is a vendor cost comparison. (No program and system integration increments of cost have been added.) The LMAE and LMDE development and maintenance costs were derived from Rocketdyne and TRW data. Other engine data was derived to be consistent with these Lunar Module engines. The Agena engine is not man-rated; therefore, a cost increment was added with contingency for occurrence of stability problems (technical risk). The results show the LMAE and LMDE to be equally cost effective and lowest cost of the alternates. Installation of the

15 MARCH 1972

FIGURE 3.2.3-6

ALTERNATE STORABLE OMS ENGINES

SYSTEM CHARACTERISTICS	LMAE (BASELINE)	LMDE	DELTA	AGENA	NEW OMS	ACPS ENGINE
OMS PROPELLANT	NTO/A-50	NTO/A-50	NTO/A-50	IRFNA/UDMH	NTO/N ₂ H ₄	NTO/MMH
ENGINE THRUST	3500	10,000	8,000	16,000	6,000	400
NUMBER ENGINES	2	2	2	2	2	12
THRUST/WEIGHT	.029	.083	.067	.133	.05	.02
OMS WEIGHT	23,212	23,967	23,895	24,298	22,902	23,967
ACPS PROPELLANT	N ₂ H ₄	N ₂ H ₄	N ₂ H ₄	N ₂ H ₄	N ₂ H ₄	NTO/MMH
ACPS WEIGHT (1)	10,370	10,370	10,370	10,370	10,370	9,271
COMBINED WEIGHT	33,582	34,337	34,265	34,668	33,272	33,238
Δ WT. FROM B.L.	(REF.)	755	683	1,086	- 310	- 344

(1) BASELINE WING/TAIL POD ACPS

FIGURE 3.2.3-7

VENDOR LEVEL COST COMPARISON

ITEM	LMAE	LMDE	DELTA	AGENA	NEW ENGINE	ACPS ENGINE
SHUTTLE DEVELOPMENT	(REFERENCE)	- 3.4	- 3.4	14.6 (1)	44.6	-3.4
CONCEPT IMPACT						27 (2)
MAINTENANCE COST		- 5.9	18.3	- 16	- 27.2	- 16.2
INVESTMENT COST		.7	- .4	- 1.1	----	- .1
PAYLOAD RESIZING COST		9.4	8.5	13.6	- 3.9	- 4.3
TOTAL Δ COST \$M		.8	23	11.1	13.5	3.0

(1) AGENA ENGINE REQUIRES MAN RATING AND MULTI-START MODIFICATION

(2) PROPELLANT TRADEOFFS HAVE SHOWN \$ 27M COST Δ BETWEEN N₂H₄/BIPROP ACPS

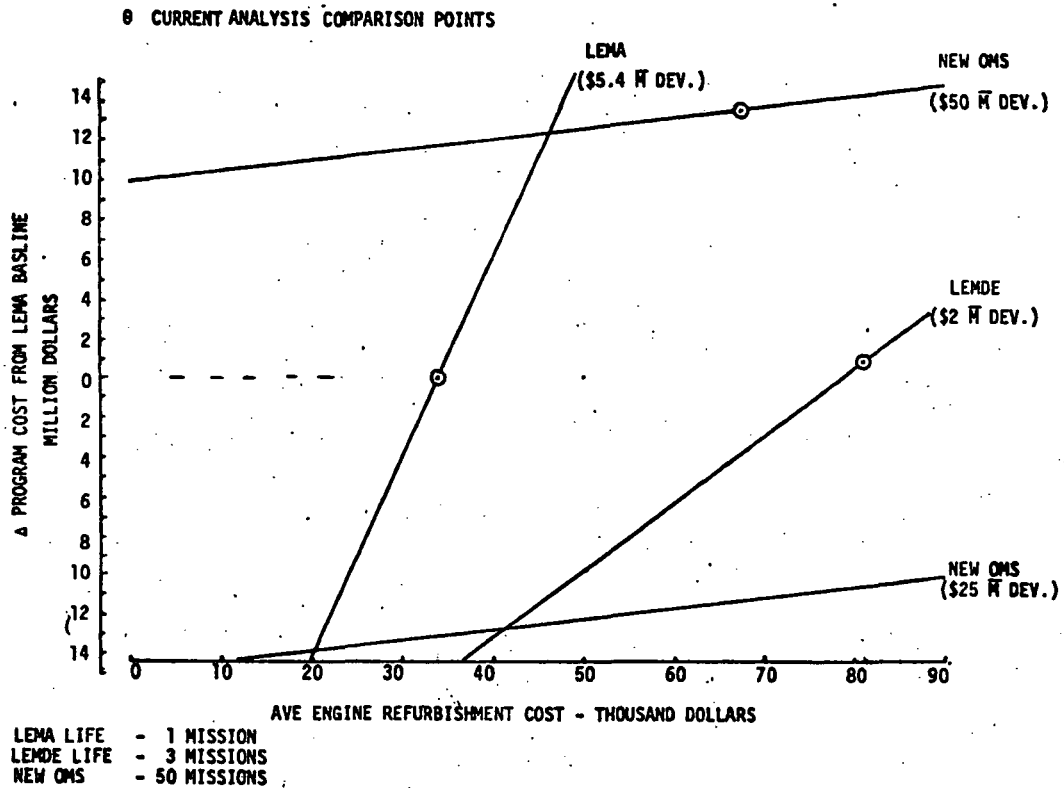
(3) PAYLOAD SIZING COST BASE = 12,500 \$/LB

larger LMDE, however, has not been included in the evaluation. The results include cost estimates of engine refurbishment and maintenance frequency. Figure 3.2.3-8 illustrates the sensitivity of program costs to these factors. Increased maintenance cost can quickly erase the advantage of an existing but life-limited engine. Therefore, the new engine is favored.

The decision then is based on program incentives. If early funding rate must be minimum, the existing engine should be used, accepting increased operational cost. If an increment of approximately 20 million dollars/year for 2-3 years can be accepted into the RDT&E phase spending, then a new OMS engine should be selected.

15 MARCH 1972

FIGURE 3.2.3-8
OMS COST SENSITIVITY ENGINE REFURBISHMENT COST



3.2.4 Airbreathing Propulsion - The airbreathing engine system is provided on the initial orbital flights and on the resupply orbital missions for the purpose of glide range extension after deorbit. With 2000 lb of JP-4 the glide range can be extended about 24 nm. Airbreathing engines are also used during the horizontal flight tests and for ferry missions. With 40,000 lb of JP-4 the nominal ferry range is about 440 nm. with a 25% fuel reserve. Ferry mission take-off is assisted with solid rockets to attain balanced field capability for hot day conditions.

For orbital missions, two engine pods are mounted externally on the aft fuselage as shown in Figure 3.2.4-1. Each pod contains one engine. The pods have ablative material on the external surfaces, a protective inlet cover (which is blown off prior to engine starting at about 40,000 ft), a nozzle space radiation cover, insulation, and internal thermal control radiators. See Figure 3.2.4-2.

The ferry/flight test engine pods are attached to the Orbiter aft fuselage in the same location as the orbital pods. In this case, however, each pod contains two airbreathing engines as shown in Figure 3.2.4-3. These pods are similar to conventional aircraft nacelles with inlet anti-icing, ground starting capability, fire extinguishing, etc.

15 MARCH 1972

FIGURE 3.2.4-1
AIRBREATHING ENGINE SYSTEM INSTALLATION
ORBITAL CONFIGURATION

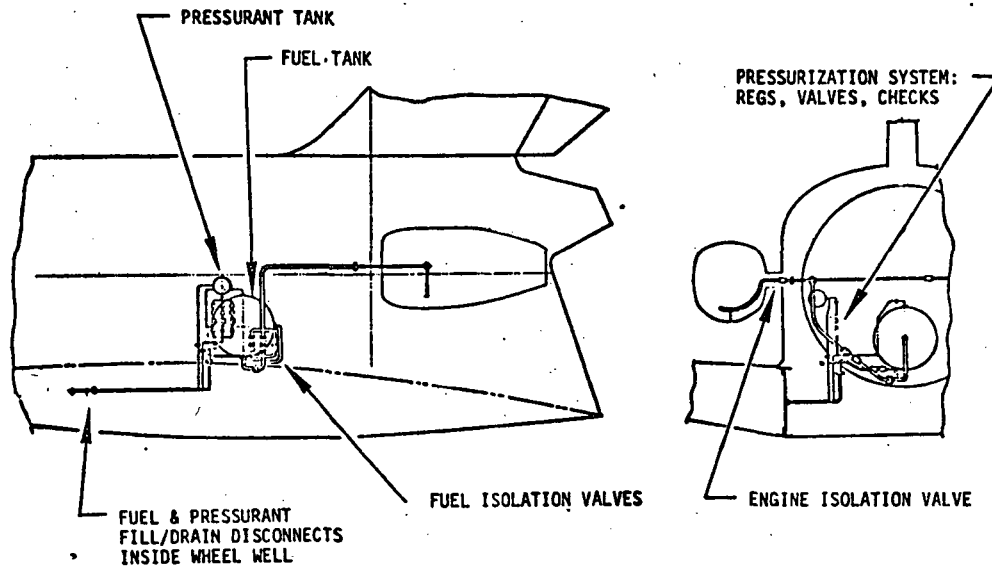


FIGURE 3.2.4-2
AIRBREATHING ENGINE INSTALLATION

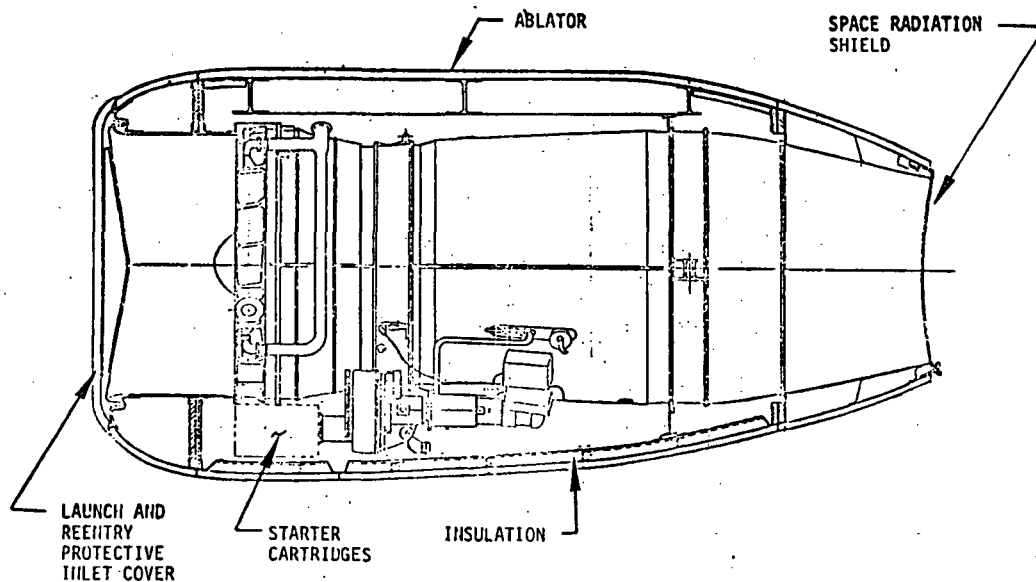
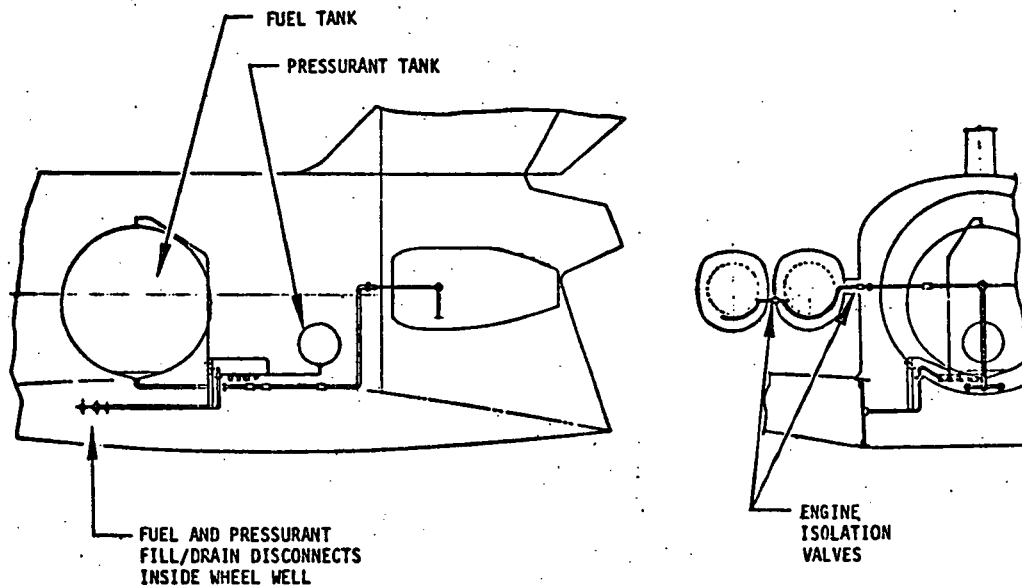


FIGURE 3.2.4-3
AIRBREATHING ENGINE SYSTEM INSTALLATION
FERRY CONFIGURATION

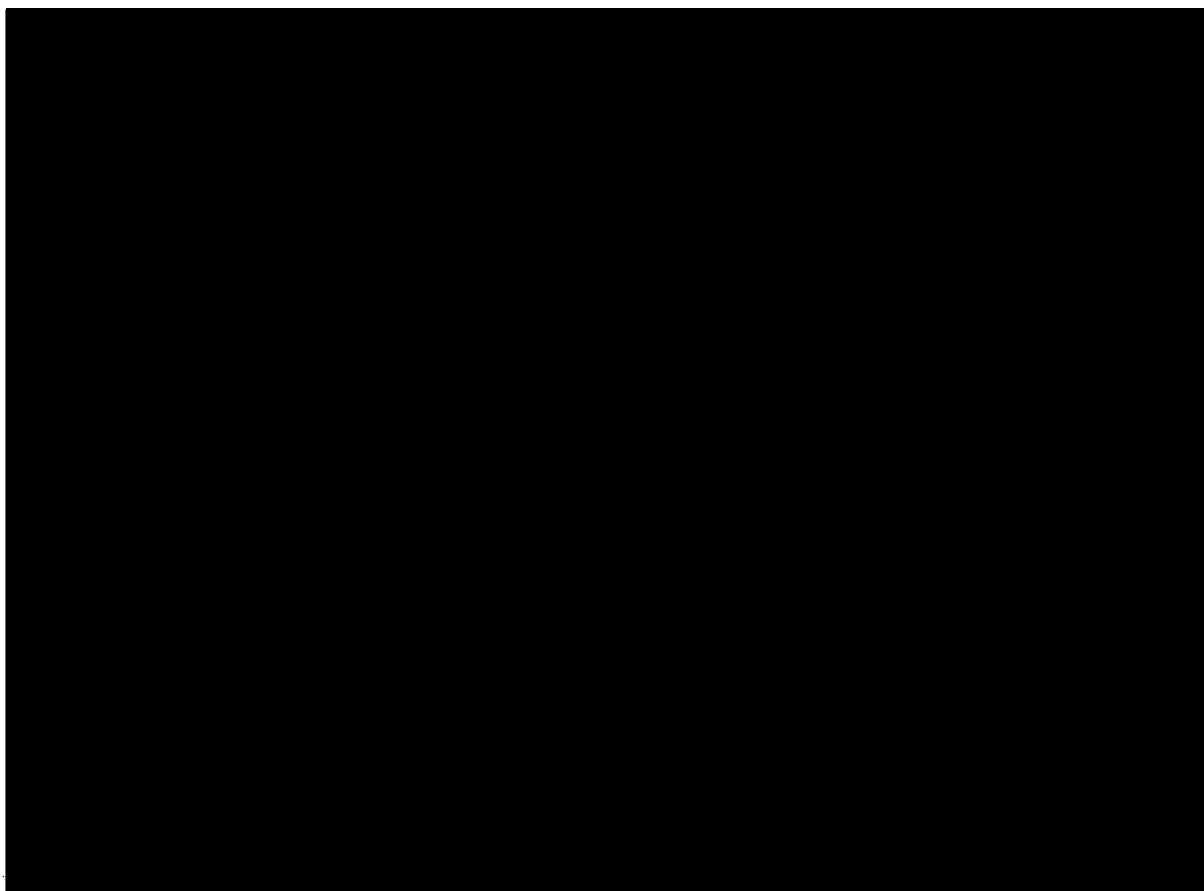


Both the orbital and ferry/flight test configurations use the F101/F12A3 turbofan engines. The engines will require sufficient development modifications to withstand the space environment, including separate oil storage, dry fuel controller, vacuum compatible seals and gaskets, etc.

The fuel system for both configurations utilizes a GN_2 pressure feed with fail operational/fail safe redundancy where practical.

The system schematic is presented in Figure 3.2.4-4.

3.2.5 Solid Propellant Abort Subsystem - For the 068B orbiter, the abort subsystem consists of two solid propellant rocket motors mounted over the wings. The motors were sized by the requirement to thrust the Orbiter away from the Booster and the external H_2O tank in case of an off-the-pad abort. The motors produce a total of 800,000 lb sea level thrust and burn for 6.5 seconds to loft the Orbiter to about 4000 feet. Aerodynamic controls are then used to land on a runway located about one mile from the launch pad. A similar procedure is followed if an abort is necessary during the initial 40 seconds of ascent. After that time, the additional energy from the motors is not required for an intact abort. Therefore



the motors are burned to furnish additional ascent velocity. The motor cases are retained until staging and then jettisoned to avoid carrying the extra weight to orbit. Key features of the selected motor are presented in Figure 3.2.5-1.

FIGURE 3.2.5-1

SOLID PROPELLANT ABORT MOTOR FEATURES

Thrust	400,000 (S.L.)
Burn Time	6.5 Sec
ϵ	5
I_{sp}	235 (S.L.)
Total Impulse	2.6×10^6 lb-sec per motor
Mass Fraction	.82
Propellant Weight	11,000 lb/motor
Inert Weight	2,400 lb/motor
Total Motor Weight	13,400 lb/motor

3.3 Avionics

The Orbiter avionics system baseline design was guided by requirements in NASA directive TD-78. Detailed system development and evaluations were accomplished for the avionics baseline requirements under Contract NAS 8-26016, a separately funded avionics study. The key system features and recommended baseline configuration resulting from this study effort are summarized in this report. Detailed results of subsystem studies and evaluations are presented in the Avionics Final Report MDC E0568 dated 15 March 1972.

The baseline avionics system architecture is summarized in Figure 3.3-1. Redundancies are noted in the lower right area of each equipment block. Components are grouped generally with the following subsystems:

- o Aeroflight Control
- o Spaceflight Guidance, Navigation and Control
- o Communications and Tracking
- o Operational Instrumentation
- o Displays and Controls, Caution, and Warning
- o Payload Handling

The avionics system implements each of the functions with separate dedicated subsystems with minimal interfaces. Each subsystem design is multiply redundant, with hardwired functional strings, and management at the subsystem level.

3.3.1 Aeroflight Control - The MDAC aeroflight evaluation concentrated on four aeroflight control system configurations which cover the tradeoffs of analog vs digital implementation and dedicated aircraft flight control vs integration with spacecraft guidance, navigation and control. Candidates 1 and 2 represent a digital implementation of the vehicle attitude rate stabilized inner loops with Candidate 2 also including the autopilot and auto-land functions. Both candidates include a minimal backup analog stabilization implementation that will satisfy level 3 handling characteristics. Candidate 3 is an analog AFCS for only the inner rate loops, with all autopilot functions performed in the spacecraft computer. Candidate 4 integrates the AFCS functions into the spacecraft GN&C computer. A block diagram of Candidate 2, the recommended design, is shown in Figure 3.3-2.

15 MARCH 1972

The recommended configuration features the stabilization and autopilot functions contained within a dedicated digital AFCS computer. A manual backup mode, using analog attitude rate control and providing Level 3 handling characteristics, is also included, since it reduces flight development risk and provides an alternate control mode in case of multiple hardware or software failures. Triple redundancy is used for the primary and backup AFCS components. The aero actuator electronics are quad-redundant. They provide voting of the AFCS outputs and switching and fade-in from the primary or backup AFCS or the spaceflight computer steering inputs to drive the surfaces during ascent. The aeroflight computer interface with the spaceflight computers is a digital link and contains attitude steering, navigation and aileron trim for surface control during entry. A separate analog interface from the spaceflight computer drives the control surface actuator electronics during the mated boost phase.

The triple redundancy in both the primary and backup channels allows voting for absolute failure detection and switchover. The goal to be flyable after two failures, including software, results in using three backup channels, to allow voting of the second failure.

The secondary servos and amplifiers are quad-redundant with force summing on the output. If one channel has a hardover failure, the other channels will overcome it and a delta pressure feedback will detect and shut off the bad channel. This also protects against electronic voter failures. The use of the voter in the backup channels also minimizes the tracking accuracy required between channels.

The minimal backup electronics needed to achieve Level 3 handling characteristics depends upon the level of vehicle aerodynamic stability. Rate feedback and augmentation networks are indicated, but may not be required, and the backup electronics would then consist of shaping controller pickoff signals, signal and gain switching, fading and amplification.

The key factors leading to the selection of our recommended design are that while the initial cost of analog implementation is less (\$11.9M vs \$22.8M) the digital is more flexible and better able to cope with changes in vehicle aero and structural characteristics, desired flight control modes, etc. Results of an AFCS change impact study indicate the total AFCS cost, including changes, would be nearly equal for an analog or digital implementation; however, the longer time required to incorporate changes in the analog AFCS could impact the schedule.

15 MARCH 1972

Although certain changes could be phased in within the normal flight test schedule, the flexibility of the digital design permits commitment to a high confidence, low risk development program within schedule and cost constraints. We have selected a dedicated digital AFCS because it allows flexibility for change and permits an independent decoupled development of the aero system with minimal organizational complexity. It also facilitates the earlier delivery with less risk of the AFCS system delaying horizontal flight test. Our recommended design incorporates the autopilot functions and autoland mechanization into the digital AFCS so that these capabilities can be adequately exercised in the horizontal flight test. Entry steering, and energy management computations are done in the spaceflight GN&C computer.

3.3.2 Spaceflight Guidance, Navigation and Control - The subsystem constitutes a triply redundant digital inertial guidance, navigation and flight control subsystem which provides steering, engine, sequencing, and trim commands to the vehicle. It operates as a pure inertial navigation system during ascent and as a radio aided inertial navigator on orbit and during descent. During descent, the functions are modified as the vehicle transitions from the spacecraft to the aircraft flight regime. A system block diagram is shown in figure 3.3-3.

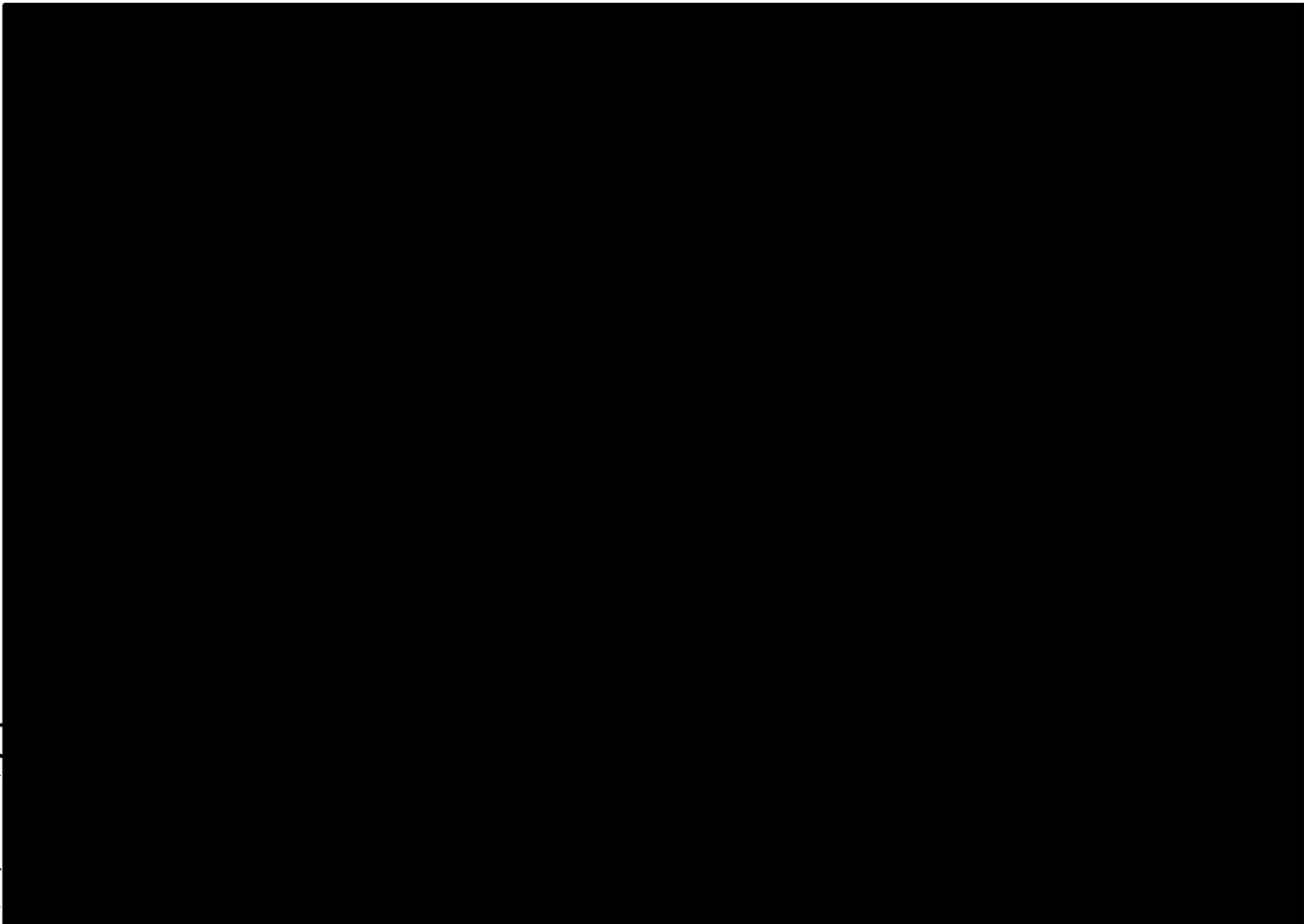
Each computer and I/O has a dedicated rate gyro set, keyboard, CRT, and a semi-dedicated IMU. The optical sensor, air data, and navigation radio signals are shared by all three computers. All computers operate simultaneously with one computer selected to be in control. The IMU dedication is unique in that alignment and calibration control are dedicated but outputs from each IMU may be manually selected for use by any computer. During time critical periods -- boost, reentry, landing, etc. -- indication of a malfunction would call for the GN&C to be switched as a set. During quiet periods, the sets could be reconfigured via the keyboard to direct selection of substitute IMU data in case of IMU failure.

The computer I/O is subdivided into dedicated and power switched sections. The dedicated sections operate continuously accepting inputs and providing outputs to display. The outputs which constitute commands go through the power switched section. The computer equipment set to be in control is selected by applying power to the power switched section of the I/O.

The total memory requirements identified tentatively permit sizing the computer memory at 38,000 words with a 50% margin for growth and flexibility in

15 MARCH 1972

FIGURE 3.3-3



programming. Likewise, a machine speed of 1.5-2 micro-seconds provides a 100% margin in execution time for the processing identified. Availability of machines with characteristics meeting those requirements was verified in a tour/survey of computer manufacturers.

In order to establish the required characteristics for inertial navigation hardware in terms of accuracy, drift, alignment, etc., a series of performance analyses have been conducted. The numbers derived were utilized in making error models for the inertial components. These error models were used in seeking current availability of IMU hardware for implementing the subsystem. A tour/survey of IMU availability including gimballed, and strapdown IRU's was conducted.

The conclusion drawn is that any of a number of current equipments would meet the requirements identified; no clear cut preference for a particular class or make of equipment emerges, and the costs of the various approaches is comparable within the uncertainties of the cost estimates used. Consequently, it

15 MARCH 1972

is concluded that the GN&C subsystem should be configured to accept any of the alternate IMU configurations. This conclusion and the results of reliability analysis have dictated that IMU cross-strapping be provided.

The GN&C subsystem provides attitude control outputs to booster and orbiter main engine TVC during ascent, to ACPS on orbit and during entry, and to the aeroflight subsystem during descent and landing.

The TVC electronics is quadruply redundant and interfaces channel for channel with the secondary actuators. The control outputs from the GN&C Computer Power switched I/O also are quadruply redundant. They are transmitted as four each Pitch Yaw and Roll gimbal commands which are independantly combined in each of the redundant control electronic channels to produce the appropriate gimbal deflection for each each engine. In addition, aerosurface control outputs directly to aerosurface actuators are provided during ascents.

The valve driver electronics is triply redundant corresponding to the valve redundancy. The signal path from the Power Switched I/O of the GN&C Computer is dual redundant and each of the two control circuits interfaces with all three valve drivers on each engine. The manual direct electronics utilizes the same dual-redundant signal paths as the computer derived commands.

During descent, guidance and trim commands generated in the Spaceflight GN&C are transmitted to the Aeroflight Computer for shaping and aerosurface command generation.

During landing, pitch and roll attitude angles and inertia velocity referenced to the runway are transmitted for attitude stabilization and for complementary filtering with MLS signals in generating autoland steering commands.

Studies have been conducted to determine the time criticality of faults to establish the time available for switching from a failed set. For attitude control functions, the worst case situations involve failures caused by I/O power loss, rate gyro hardover, or attitude reference loss. For sequencing functions, the inverse approach has been taken by providing a sequence control and arming panel by which critical discrete commands can be manually inhibited until the time in the flight sequence when they nominally occur. Design alternatives, providing for automatic switching, have been examined for use in the event that the time criticality precludes this remaining a manual function.

The primary method of subsystem fault detection is manual, using the operational displays to compare the behavior of the three systems with each other and

15 MARCH 1972

with nominal for the appropriate flight situation. This process is aided by Built-In-Test (BIT) in each Line Replaceable Unit (LUR) which detects and indicates malfunctions. In addition, a form of data validation, comparison and reasonableness testing is carried out in each computer for selected sensor inputs. Critical malfunctions are indicated on the caution and warning system.

3.3.3 Communication and Tracking System

3.3.3.1 Aeroflight Communications and Tracking Subsystem - This subsystem provides for all communications tracking and radio navigation functions required after entry with the exception that the space flight unified S-Band system is also used when landing at sites with S-Band capability (e.g., KSC).

Triple redundancy is used for critical flight functions required for landing and for the audio center equipment which is required for orbital flight as well as aeroflight.

The TACAN sets measure bearing and slant range to the selected ground stations. Raw bearing and range are displayed on the horizontal situation indicator (HSI) and are used to update horizontal position within the space flight guidance, navigation and control computer after entry.

The microwave landing system navigation outputs are elevation, azimuth, and slant range, all relative to the landing site. These outputs are directly displayed on the HSI and are processed in the aeroflight control computer to form the flight director commands displayed on the flight director altitude indicators (FDAI). The distance and elevation outputs are processed to provide a precise altitude reading used for final flare and landing. Morse coded ground station identification signals received by the microwave landing system and TACAN sets are sent to the audio control centers so that the crew can aurally verify that the proper ground stations are being used for Orbiter navigation.

The selected military UHF transceivers have 3600 channels within the 225 to 400 MHz band. Any of the channels are selectable by the crew. Both sets can be operated simultaneously for two channel capability.

A single voice actuated memo pad recorder is included onboard. The unit provides 10 hours of voice recording.

The audio equipment provides for voice communications among the crew throughout the vehicle, manned space flight network (MSFN), and docked vehicles. The audio

15 MARCH 1972

equipment also disseminates aural identification tones and caution and warning tones generated by nav aids and the C&W subsystem.

Two key issues shaping our recommended system include the question of autoland capability and landing aid selection. Studies and simulations indicate that autoland capability is not required to accomplish powered or unpowered landings. However, autoland capability will significantly improve the probability of mission success. Commercial airline statistics indicate that only one-sixth of the go arounds are due to ground or onboard equipment failures and one-third of the go arounds are due to pilot technique errors. Therefore, automatic landing systems would greatly reduce the largest cause of errors (pilot technique errors) that could cause unsuccessful landings. Also, the higher Shuttle electronic equipment redundancy levels will reduce (over airline practice) the probability of unsuccessful landings due to equipment failure. The Shuttle schedule indicates that the most cost effective time for the development of autoland capability is during the horizontal test phase. This allows for development of the autoland capability concurrently with meeting other flight test goals. However, it is believed that the first vertical flights do not require autoland capability. Therefore, autoland development can be extended (if necessary) without schedule impact on Shuttle.

Factors which have led to the recommendation of the FAA microwave landing system with ILS for early horizontal flight test only include:

- (1) Existing ground radar landing systems (SPN-42 shipborne) are very costly and landing site radars are/will not be deployed for general use.
- (2) Precision ranging systems (PRS) have never been used for operational landing systems thus much development is required for autoland. PRS ground stations will not be deployed for general use.
- (3) ILS is available and is deployed, but it requires a good site and control of vehicular traffic in the vicinity of the landing site for Category III autoland. ILS antennas are larger than those required for other systems.
- (4) The MLS is currently under development by the FAA. It is being designed for autoland and will be free of normal sighting effects and will not be adversely affected by local vehicular traffic. The MLS will be widely deployed and will become the national landing system. No interim system is required for vertical flights with this approach providing current FAA MLS development schedules are met.

15 MARCH 1972

3.3.3.2 Spaceflight Communications and Tracking Subsystem - The communications and tracking subsystem is utilized in the space flight mode to provide voice, data, and television communication, as well as rendezvous and orbit navigation data for operation of the vehicle and payload. This system operates at S-Band in a configuration compatible with the Apollo MSFN stations and formats. Two equipment sets operate at multiple frequencies to provide protection from single failure loss of capability and, without failure, to provide simultaneous operation in several operating modes.

In accordance with the NASA directive, utilization of the Apollo VHF rendezvous scheme has been examined in detail. However, a comparison tradeoff study indicates use of the S-Band equipment for rendezvous provides increased capability at lower cost.

The Orbiter-MSFN interface is utilized during on-orbit operations to provide timing and navigation update for reinitializing the onboard state vector from MSFN tracking data. It also provides the two-way voice link for communication between the Orbiter and ground as well as a wide band downlink for transmitting Orbiter TV signals to the ground. A narrow band (72 KBPS) data downlink is utilized for telemetry from the operational instrumentation system. Stored data from the operational instrumentation recorder is dumped on command at 1 MBPS using the MSFN range code turnaround channel. Range tracking data is interrupted during the data dump interval or the stored data may be dumped via the TV FM transmitter without interrupting range tracking.

The uplink also provides ground control of selected onboard functions, including data dump and antenna selection. In addition, it provides for communicating rendezvous target orbit data and landing site coordinates whenever these are required.

The Orbiter to space station or Orbiter to Orbiter interface is utilized during rendezvous by the active vehicle to extract rendezvous range and range-rate data. In addition, a two-way voice link is provided for coordinating operations and the relative alignment between the two vehicles. A two-way data link is provided for data exchange. The inactive vehicle utilizes its MSFN transponder while the active vehicle uses the inverse transponder frequency and performs the ranging function.

The design of the C&T subsystem is based entirely on existing equipment. The antennas proposed are those used on the Apollo command module, the television is

15 MARCH 1972

baselined as the color camera and monitor used in the Apollo Lunar Module. The S-Band equipment is derived from that used with Apollo.

Direct S-Band communications is recommended for early flights because the ground facilities exist and network performance has been demonstrated. The ground network to be used for the early flights will be the Space Tracking and Data Network (STDN) which is being formed from the existing Manned Space Flight Network (MSFN) and the existing Space Tracking and Data Acquisition Network (STADAN).

When available, the Synchronous Orbit Tracking and Data Relay Satellite (TDRS) system will be used for orbiter communications. The use of TDRS will allow a reduction in STDN stations (to reduce costs) and will provide for communications coverage nearly 100% of the time. Conventional VHF/UHF air traffic control communications is used for voice communications to the landing sites.

3.3.4 Operational Instrumentation System - The Operational Instrumentation System (OI) includes hardware and software required for data acquisition, processing, analysis, storage and distribution of information to the crew and other using subsystems. In addition, it provides backup capability to the caution and warning system, backup computational support for the Payload Manipulator System, and command and stimuli generation prelaunch checkout and servicing operations. A system block diagram is shown in figure 3.3-4.

The OI is controlled through crew commands via the OI panel switches and keyboard. The computational requirements, data acquisition functions, etc., are performed in two redundant computers which are identical to the GN&C computers. Data is collected by redundant Remote Multiplexer Unit (RMU) strings. Signal conditioners are incorporated to provide excitation and conditioning for remotely located transducers and isolation/buffering for other subsystem measurement pickoff points.

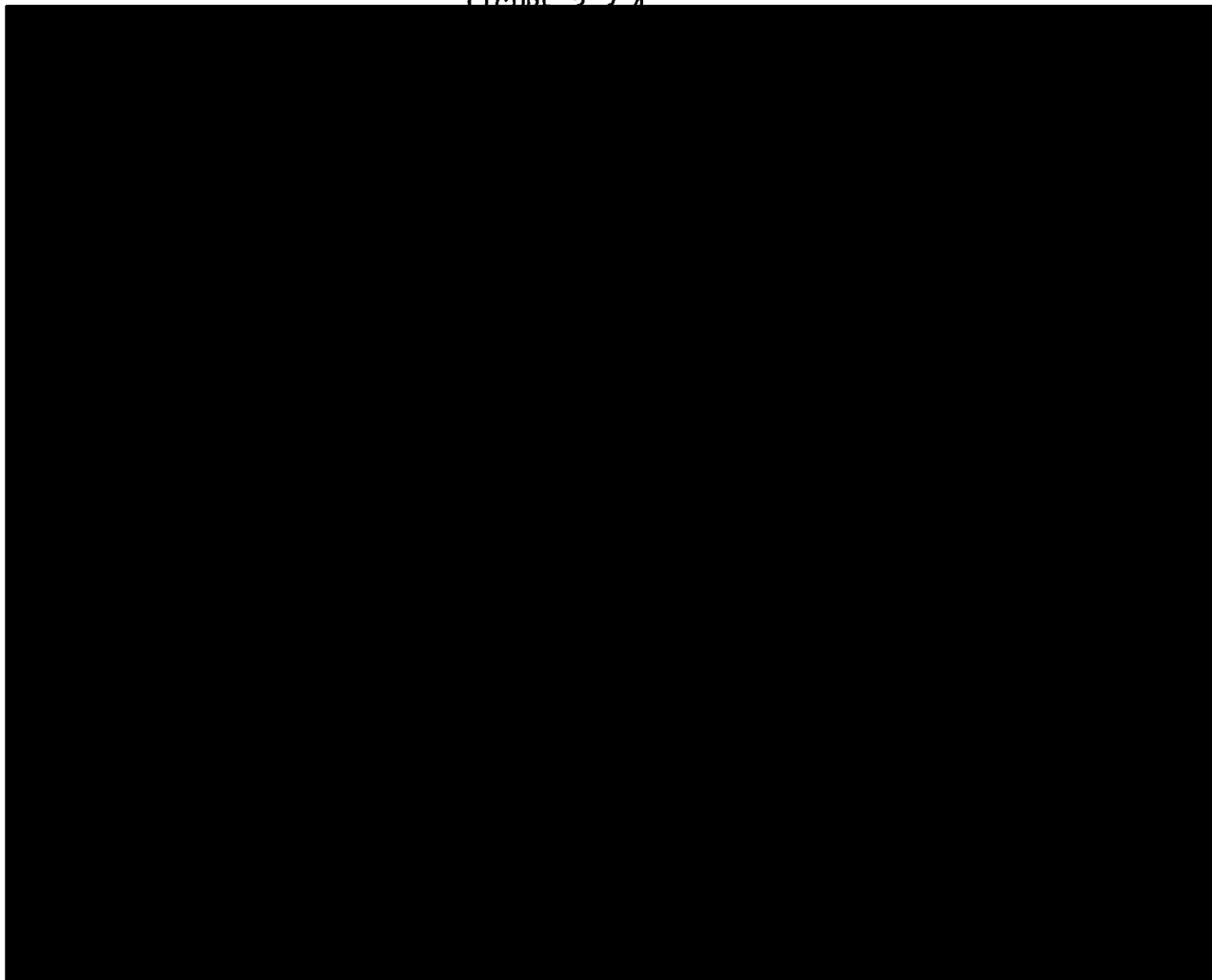
Onboard recording for post-flight maintenance and anomaly investigations is accomplished via redundant permanent storage maintenance recorders and temporary storage loop recorders.

The telemetry downlink is via the S-Band transmitters, upon command from the crew or the ground. The real time bit rate is 72K BPS and the delayed time bit rate (high speed dump from the maintenance recorder) is 1.024 M BPS.

The OI computer can be loaded from the mass memory with the payload manipulator system (PMS) software program requirements and can serve as a PMS computer in case of a PMS computer failure.

15 MARCH 1972

FIGURE 2-2-4



The systems monitor station and the center console keyboard are used by the crew to call up measurement displays on the CRT's, to enter constants into the OI computer, and in general to communicate with the OI computers. CRT background data is called up from the mass memory, and current measurement or other data is overlayed on the CRT background by the OI computer.

The Ground Command Decoder (GCD) provides onboard command distribution and stimulus generation used during prelaunch checkout and servicing operations (e.g. propellant valve commands and rate gyro analogue torquing signals). The ground complex sends these signals in digital form via a multiplexed line and the redundant GCD's generate the individual outputs. The GCD's are powered off in a fail-safe manner at lift-off.

3.3.5 Displays and Controls, Caution and Warning - The command pilot has the displays/controls to exercise executive control over all flight critical mission phases. He has prime responsibility for all time critical ascent functions and is the only crewman having access to the abort handle. He assumes the lead role in

15 MARCH 1972

system configuration and control/monitor for orbital burn executions, nose-to-nose docking, and reentry and landing. He is also qualified for the payload manipulation station operations.

The pilot functions as the navigator, communicator, and subsystems manager to relieve the command pilot of these tasks. However, in the conventional side-by-side arranged flight deck, as shown in figure 3.3-5, he has the displays/controls to be a backup to the command pilot. He is also qualified for the payload manipulation station operations.

The systems monitor station serves a dual function. It contains displays/controls that interface with the vehicle subsystems and with the payload. The systems monitor configures the instrumentation and communication subsystems to acquire the vehicle maintenance and fault data. The CRT/keyboard terminal at this station is used also to monitor this data on board. This data is used to guide the crew in systems reconfiguration and pre-phase checklists. The same CRT/keyboard/computer combination is used also for payload control/monitor functions. This arrangement allows several missions to be flown without the dedicated payload monitor and station.

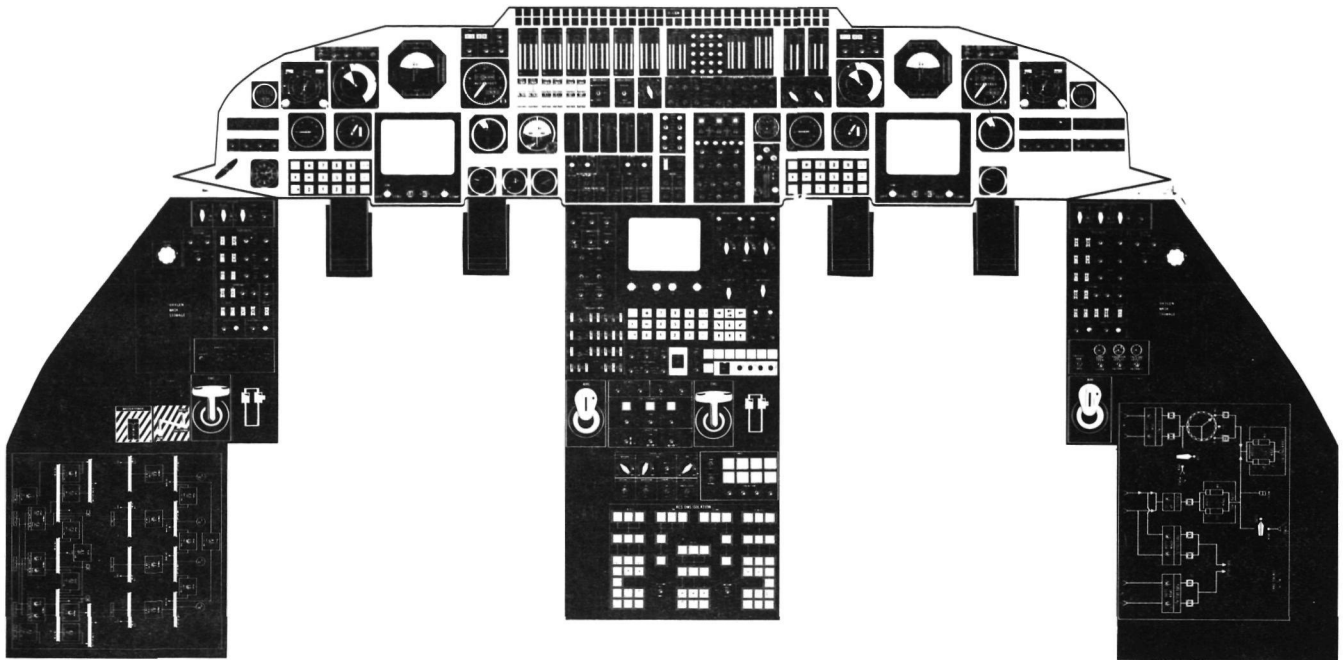
The payload monitor station is used to supplement the capability provided at the systems monitor station. The payload monitor will be responsible for any EVA activity associated with a payload.

The payload manipulator station is unmanned except during payload handling periods on orbit. It is manned by one of the pilots, since flight critical vehicle control must be executed in the process of station keeping or deploying/retrieving a payload module.

The primary flight control display set for each pilot consists of the conventional electromechanical "BASIC SIX" plus a CRT/keyboard terminal. The "BASIC SIX" includes the flight director attitude indicator (FDAI), and the horizontal situation indicator (HSI), plus the four air data dials indicating Mach/airspeed, angle of attack, barometric altitude, and vertical rate of descent. The CRT display is used for support of flight situation graphics and data pad formats.

The crew/computer communications link consists of a CRT, keyboard, symbol generator, and magnetic tape set. The keyboard is used to call up displays and enter update data into computer memory. The magnetic tape unit performs a dual

FIGURE 3.3-5 COCKPIT ARRANGEMENT



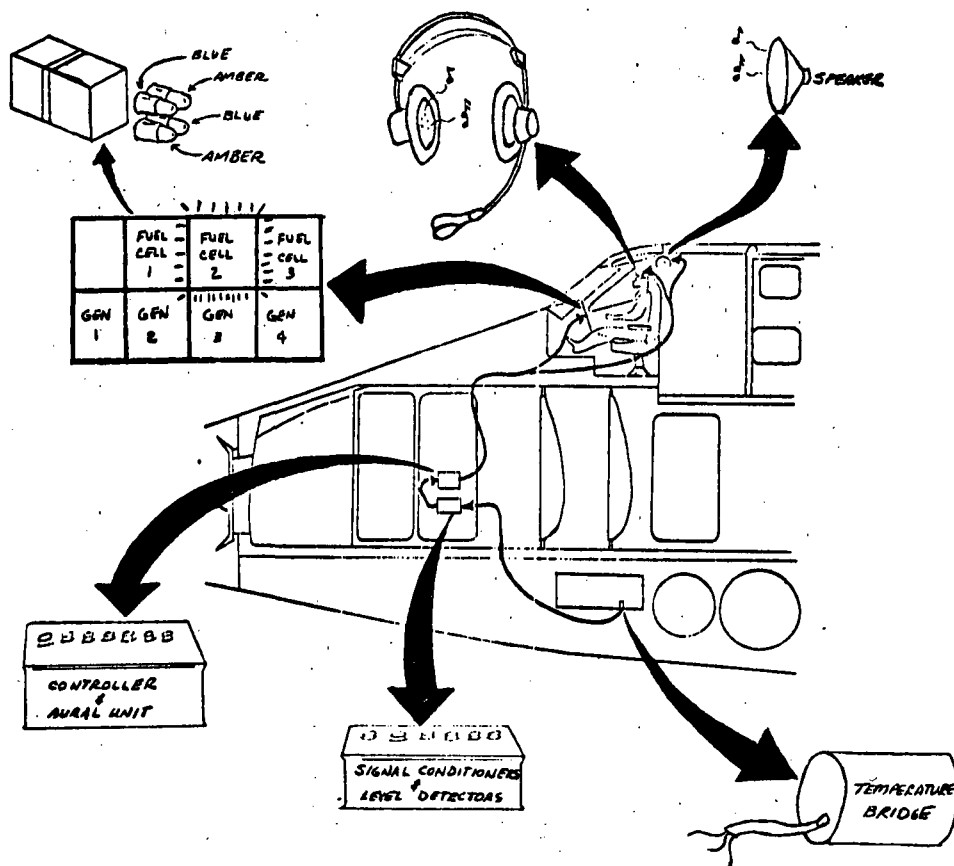
function by storing all the background display data and formatting instructions for the CRT display presentations and by storing computer mass memory data to provide for computer reprogramming on the ground or in flight.

The basic vehicle attitude controller is the new design dual function sidearm controller. The operation includes a crossfeed position tracking capability in the aero mode. This master/slave mode provides for transient free takeover by the backup pilot.

The Caution and Warning (C&W) system alerts the crew, via audible and visual outputs, to impending vehicle/equipment failures or hazardous conditions. The system provides for 10 aural alert tones and 220 dedicated function annunciator lights. There are 2 caution level tones, 1 common warning tone, and 7 unique warning condition/situation tones. Most of the annunciator lights are grouped in a matrix on the main flight deck overhead panel. The rest are distributed at other stations and on subsystem management panels. For example, 32 lights are schematically arranged on the center console to show failed and/or isolated ACPS jets. A functional diagram of the caution and warning system is shown in figure 3.3-6.

15 MARCH 1972

FIGURE 3.3-6
CAUTION AND WARNING SYSTEM



The implementation provides for dual redundancy of the caution and warning function. A hardwired system, with dedicated circuits for signal conditioning, limit detection, lamp and tone driver functions, provides the primary system. This system drives one of the dual lamps in each annunciator light. The other lamp is activated due to similar logic implemented in the Operation Instrumentation (OI) computer, using data derived from the data acquisition system. Each tone and annunciator light can be inhibited to override annoying outputs. Blue lamps indicate which system has been inhibited. The design provides for electrically alterable limits for the analog input channels. The OI computer software can be changed accordingly. For the primary hardwired system, new limits can be injected with ground support equipment. The circuitry design provides for a limit reset capability over the entire input range. A built in memory function is provided to store past annunciator bulb commands. The crew can call this memory function display upon command and can also reset the memory to initiate a new memory period.

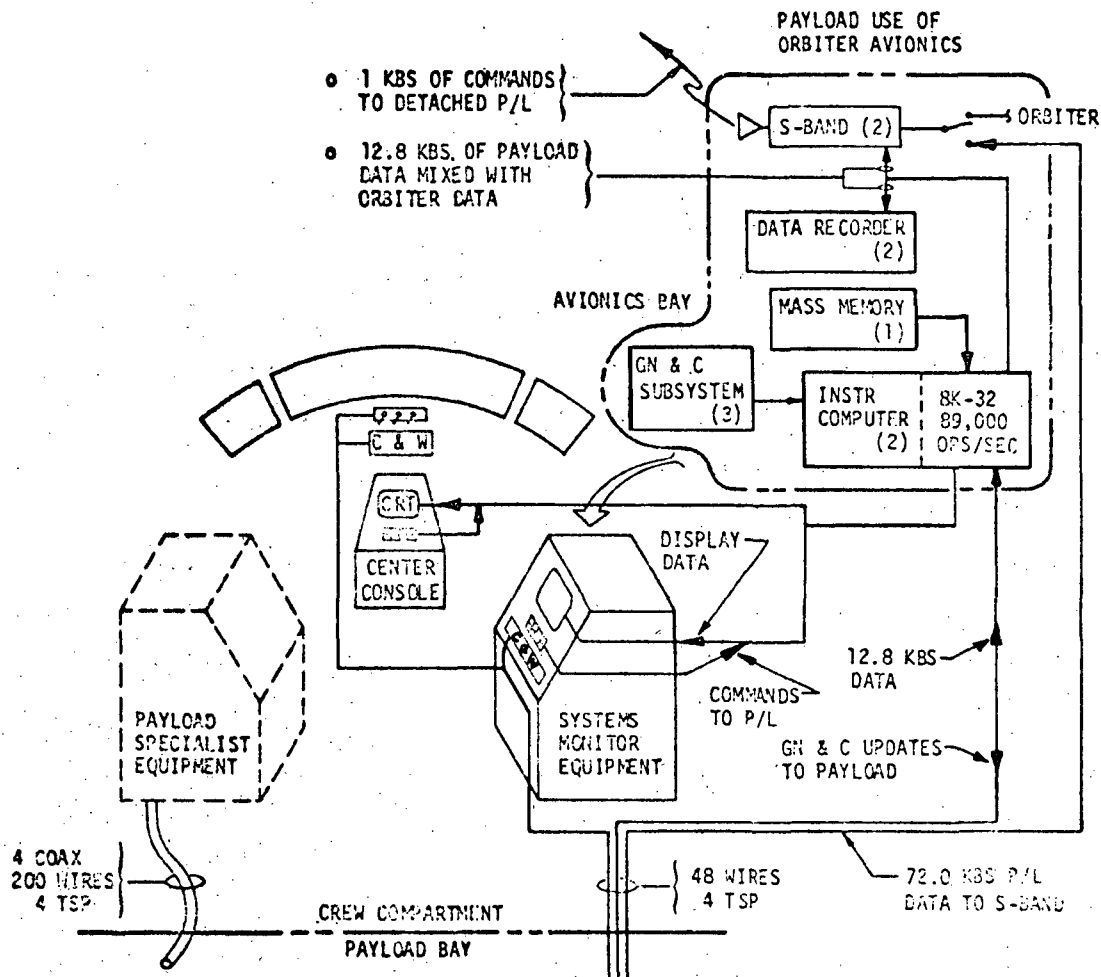
15 MARCH 1972

3.3.6 Payload - Serial digital payload data interfaces with the instrumentation computers at a rate of 12.8 KBS (400 words of 32 bits each second). The interface shown in figure 3.3-7 is dual redundant with dual payload data flowing into both instrumentation computers. This data flows into 8000 word memory module in each computer which is dedicated to payload use. The programming requirements for the 8K module are generated by the payload contractor and supplied to the Orbiter contractor for coding. Two more 8K payload modules can be stored on the mass memory and loaded at anytime via keyboard initialization. The input data processing section of the 8K module will typically be programmed to read the 400 words/sec from both payload data channels, check limits on health data and, compare the data from the two input channels for out of tolerance parameters. There are three basic sources of data available for transfer from the instrumentation computer to the payload over the 3.2 KBS data line (100 words of 32 bits per second). The S-Band uplink goes to the instrumentation computer. This data can be designated to go into the 8K payload module where it can be relayed as required to the payload. GN&C data goes into the instrumentation computer and is accessible to the payload through the 8K payload module. The third source of data available to the payload is the keyboard at the center console and systems monitor station.

Handling station electronics include the servo amplifiers required to apply the proper force feel in the manipulator hand controllers. These electronics also route controls either through the manipulator computer or directly to the arm electronics, as is the case for the backup mode. The present approach is to use a dedicated computer for the manipulator where the unit is the same as the aeroflight unit (12K - 16 bit memory). The manipulator computer receives manipulator joint angle commands from the hand controller and compares them to the actual manipulator joint angles. The angle differences are used to send commands to the manipulator and also to send force feel commands back to the hand controller. The manipulator computer will determine when the manipulator can not handle the dynamics of the payload and then send pitch, roll, and yaw commands to the ACPS. The instrumentation computers are used to back up the manipulator computers.

15 MARCH 1972

FIGURE 3.3-7
PAYLOAD AVIONICS INTERFACE



3.4 Crew Station

3.4.1 Environmental Control and Life Support -

Cabin Heat Transport and Heat Rejection - Dual redundant loops, using water as a coolant, provide for heat transport in the cabin. The cabin heat load is rejected to the radiator coolant loops by interface heat exchangers. If necessary, two pumps can be used during peak loads.

Radiator Heat Transport and Heat Rejection - Dual redundant loops, using Freon 21 as a coolant, provide for heat transport external to the cabin. The total vehicle heat load is rejected by a deployable space radiator during normal on-orbit operation. A GSE heat exchanger and expendable heat sink are also available for heat rejection in these coolant loops. The expendable heat sink utilizes water as an expendable to supplement the radiator during periods of peak on-orbit heat loads. From the time that the space radiator is retracted and stowed in the cargo bay until an entry altitude of 100,000 ft. is reached, heat from the radiator cooling loops will be rejected into the heat sink, utilizing water as the expendable. From 100,000 ft. altitude to landing, Freon 22 is utilized as the expendable. If necessary, two pumps in the coolant loops may be operated during peak on-orbit loads as well as for higher reentry heat loads.

Cabin Temperature Control - The cabin temperature is maintained within the required limits by a cabin heat exchanger, utilizing the cabin water coolant loops for removing the electrical, chemical and metabolic heat loads from the cabin atmosphere. The cabin temperature is controlled to any point between 65°F and 75°F (within the system capability) by bypassing air around the heat exchanger as the cabin sensible heat load is reduced. Heating is provided, if required, by an electrical heater in the air bypass line.

Cabin Humidity and Contaminant Control - Humidity control is provided by the same cabin heat exchanger which condenses moisture from the cabin atmosphere. A minimum air flow rate is maintained through the heat exchanger to assure that humidity requirements are met under any sensible heat load condition. Condensate is collected in an inertial device downstream of the heat exchanger and pumped to the waste water system by a rotary phase separator. Lithium hydroxide is utilized for CO₂ removal. The LiOH is stored in replaceable canisters, each capable of four man-days of operation. Manual replacement of a canister is required once per day.

15 MARCH 1972

Trace gas contaminants are removed from the cabin atmosphere by activated charcoal.

Cabin Ventilation - Ventilation is provided by process flow fans that supply cabin air to the parallel arranged cabin heat exchanger, LiOH canisters and charcoal canister. If additional air flow is necessary, it will be supplied by separate ventilation fans.

Atmosphere and Fuel Cell Reactants Storage - Hydrogen and oxygen reactants for the fuel cell are stored in supercritical cryogenic tankage. Oxygen required for cabin atmosphere pressure and composition control is also stored in the fuel cell oxygen tankage. Both the hydrogen and oxygen are warmed to acceptable temperature levels by cryogenic heat exchangers in the radiator coolant loops prior to delivery to the fuel cell or cabin pressure and composition subsystem. Oxygen is also supplied for air lock pressurization, cabin emergency oxygen purge, emergency oxygen masks, and for pressurization of potable water, wash water and atmosphere condensate storage tanks.

Nitrogen is stored as a high pressure gas. In addition to cabin atmosphere pressure and composition control, it is used for emergency cabin purge and for the pressurization of fuel cell product water and the expendable Freon storage tanks.

Cabin Pressure and Composition Control - Cabin pressure and composition control is provided by mechanical pressure regulators that deliver either oxygen or nitrogen to the cabin to maintain the cabin pressure at 14.7 psia. The pressure regulators are supplied with oxygen or nitrogen depending on the oxygen partial pressure in the cabin. This is accomplished by a two-gas controller that receives signals from oxygen partial pressure sensors. The controller then activates solenoid valves that allow delivery of either oxygen or nitrogen to the cabin pressure regulators. Cabin relief valves are provided to limit the maximum cabin pressure in the event of a failed open oxygen or nitrogen inflow valve. The relief valves can also be overridden to provide for dumping the cabin atmosphere to space in the event of an emergency.

Airlock Pressurization - Oxygen is supplied to the airlock for repressurization and EVA PLSS recharge. A relief valve limits the maximum airlock pressure and can be overridden to provide atmosphere dumping. Pressure equalization valves are also provided between the airlock and cabin and between the airlock and tunnel.

15 MARCH 1972

Potable and Wash Water Storage and Use - The source of all water used in the Orbiter is the fuel cell. The fuel cell delivers potable water to cold water, hot water, and wash water storage tanks. All three tanks are pressurized by the oxygen supply subsystem. Water in the tanks is maintained at a warm or cool temperature level by heat transfer from the cabin water coolant loops. The hot and cold potable water is delivered to the food management subsystem for drinking and food preparation. Warm water from the wash water tank is delivered to the waste management subsystem and to a sink through a water control module and cleaner dispenser. Air is drawn through the sink by a vacuum power module. The water entrained in the air stream is removed by rotary phase separators and delivered to a waste water tank for storage. The tank is pressurized by cabin pressure.

Waste Water and Expendable Freon Storage and Use - Atmosphere condensate, excess fuel cell water, and Freon are all used as expendables for heat rejection by the expendable heat sink in the radiator coolant loops. Condensate is stored in a tank that is normally pressurized by cabin pressure. When the tank is full, oxygen from the oxygen supply subsystem forces the condensate from the tank through the expendable heat sink. A small accumulator stores condensate received from the phase separators while the main condensate tank is being emptied. The water is stored in a tank pressurized from the nitrogen supply subsystem. The water is kept from freezing by heat transfer from the radiator coolant loops. Freon is stored in tanks pressurized from the nitrogen supply subsystem.

Waste Management - Waste collection is provided by an integral urine and feces collector. Urine and flush water are entrained in an air stream drawn through the urinal portion of the collector. Rotary phase separators remove the liquid from the air stream and deliver it to a urine storage tank. The tank contains a disposable bladder and is pressurized by cabin pressure. Flush water from the wash water tank and biocide from the biocide storage tank (hand pumped) are delivered to the urinal in the proper quantities by a urine flush module. The feces waste collector also uses an air stream to entrain the feces and carry them to a rotating slinger that distributes the feces on the walls. The air stream from both the urinal and feces collector are returned to the cabin through a bacteria and odor filter. When the feces collector is not in use, the feces are vacuum dried by exposing the storage volume to space.

15 MARCH 1972

3.4.2 Crew Station Arrangement

Flight Deck Arrangement - The flight deck contains all displays and controls for operating the Orbiter and for monitoring, deploying, and retrieving the payload.

The flight station provides for two pilots in a conventional side-by-side arrangement in the forward part of the flight deck. The geometry of the forward instrument panel, center console, side consoles and overhead panel is essentially the same as the DC-10. Side stick controllers are used in lieu of a center stick. The rudder pedals are retained for yaw control, braking, and nose wheel steering. Seat adjustments and floor spacing beneath the seats meet launch and orbital operations requirements.

Pressurized volume of the flight deck is 500 cubic feet.

Lower Cabin Arrangement - The lower cabin contains the crew living quarters, avionics and photographic equipment, and provisions for inter-vehicular transfer and EVA. All equipment is located at the sides of the cabin, leaving a clear passageway for access to hatches and sleep stations. The camera station is in the cabin ceiling between the avionics bays. The two-man airlock is located between the food preparation and personal hygiene areas adjacent to the payload bay, and has two hatches into the payload bay. The main hatch is used to connect to a payload that requires manning or maintenance. The upper hatch is to allow EVA with the payload door being open when the payload is so full of cargo that the center hatch is unusable. Air ducts, water storage and other ECLS equipment will use the areas beside the airlock. Pressurized volume of the lower cabin is 1760 cubic feet.

3.4.3 Flight Control System - The selected aerodynamic and engine control system is a "fly-by-wire" configuration. Electrical signals are generated by force transducers located on the pilot controls. Hand controllers and conventional rudder pedals are used for aerodynamic control. Hand controllers are used for engine controls. Electrical command signals generated by the pilot are transmitted to the appropriate electronic control system. These commands are quadruplexed to a secondary actuator which controls the master control valve input, thereby positioning the surface actuators. Either electro-mechanical or the electro-hydraulic secondary actuators developed on the 680J program are suitable; however, the presently baselined system utilizes the electro-hydraulic mechanization. All four hydraulic systems supply power for each aerodynamic control surface to provide a FO/FO capability. Engine control actuators utilize four hydraulic systems to each engine to provide the necessary redundancy.

15 MARCH 1972

3.5 Power Supply

This section summarizes the electrical and hydraulics subsystems. Additional detail is provided on the APU module selection since it resulted from a significant trade study.

3.5.1 Electrical - Three 28 volt fuel cell modules and four AC generators are used to supply prime power for the Orbiter. Two batteries provide power for restart of fuel cell modules and AC generators.

The fuses are normally isolated from one another. However, Bus tie capability is provided to permit more than one bus to be operated from a single source in the event of a source failure.

Static inverters provide AC power during periods when the AC generators are not running, and transformer-rectifier units provide DC power from the AC generators.

Power is transmitted fore and aft in the vehicle as 115/200-volt, 3- ϕ , 400-Hertz AC to reduce the weight of the transmission lines.

3.5.2 Hydraulics - The selected vehicle hydraulic system consists of four completely independent systems operating at 3000 psig nominal. Each system is powered by a hydrazine auxiliary power unit which drives the system pump. All four hydraulic systems are distributed to flight critical subsystems to provide the vehicle with a FO/FO design. Each system is capable of 150 peak horsepower, which represents 50% of the total vehicle requirements. The system is so designed as to provide continual operation even after two system failures without requiring pilot action. The flight control actuators are controlled by a fly-by-wire type system which is similar to the one presently being developed by MDAC on an F-4 aircraft. The hydraulic system includes power generation, distribution, aerodynamic flight control actuators, and utility system conversion elements.

3.5.3 Auxiliary Power Unit - Four individual APU modules are provided to satisfy the Orbiter hydraulic and electrical power requirements. The modules are not interconnected in any way. All four modules operate continuously; a failure in any one module results in shutdown of that module. Two operating modules are required for vehicle flight control - a FO/FS system.

The APU module schematic, Figure 3.5-1, shows the service panel with fill and drain and test ports for module checkout and servicing. The fluid disconnect permits removal of the tank and pressurant module for remote servicing. The helium

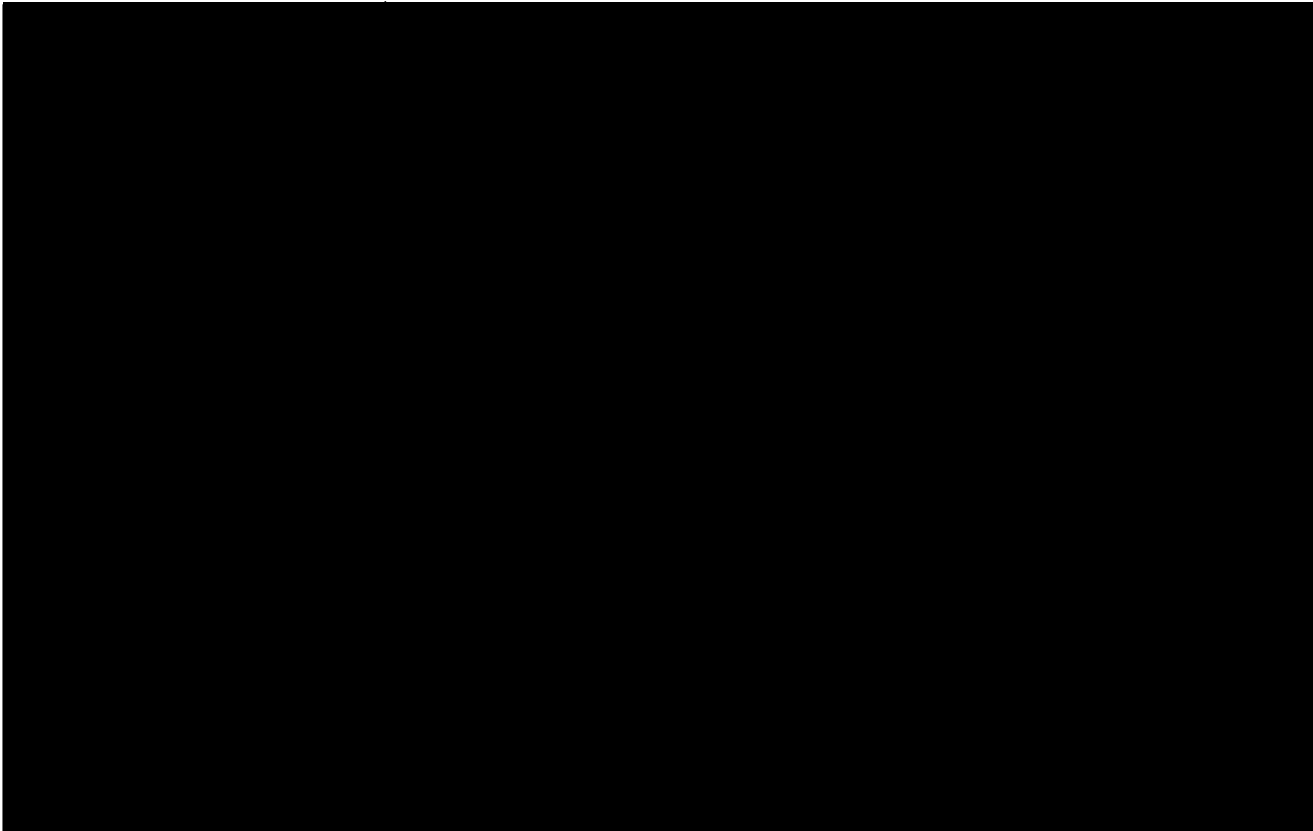
15 MARCH 1972

pressurant gas is regulated to the operating pressure of the hydrazine tank by the primary regulator. The secondary regulator provides regulation in the event of a fail open primary regulator. The hydrazine tank is pressurized by opening the pressurization control valve, which provides pressurized hydrazine to the 3-way start valve. An automatic purge of the reactor, turbine, and exhaust line is provided prior to each start and during each shutdown sequence.

APU start is initiated by opening the start valve. During operation the turbine power unit controller maintains constant turbine speed by modulating hydrazine mass flow via the control valve. The controller monitors selected turbine and gearbox parameters, detects malfunctions and automatically shuts down the APU in the event of a malfunction.

An external gearbox cooling loop is provided to remove heat from the gearbox and A.C. generator. Heat is exchanged with the hydraulic pump case drain loop, from which it is removed via a flash evaporator, with water as the cooling medium. A turbine ground test line is provided to operate the turbine at part load on heated nitrogen in the maintenance areas. End-to-end checks of the hydraulic and electrical systems are thus possible.

ORBITER APU MODULE SCHEMATIC



15 MARCH 1972

The APU module installation drawing, Figure 3.5-2, shows two modules mounted on each side of the vehicle centerline. A common line vents APU exhaust through the rear engine bay surface via a nonpropulsive vent.

APU MODULE INSTALLATION

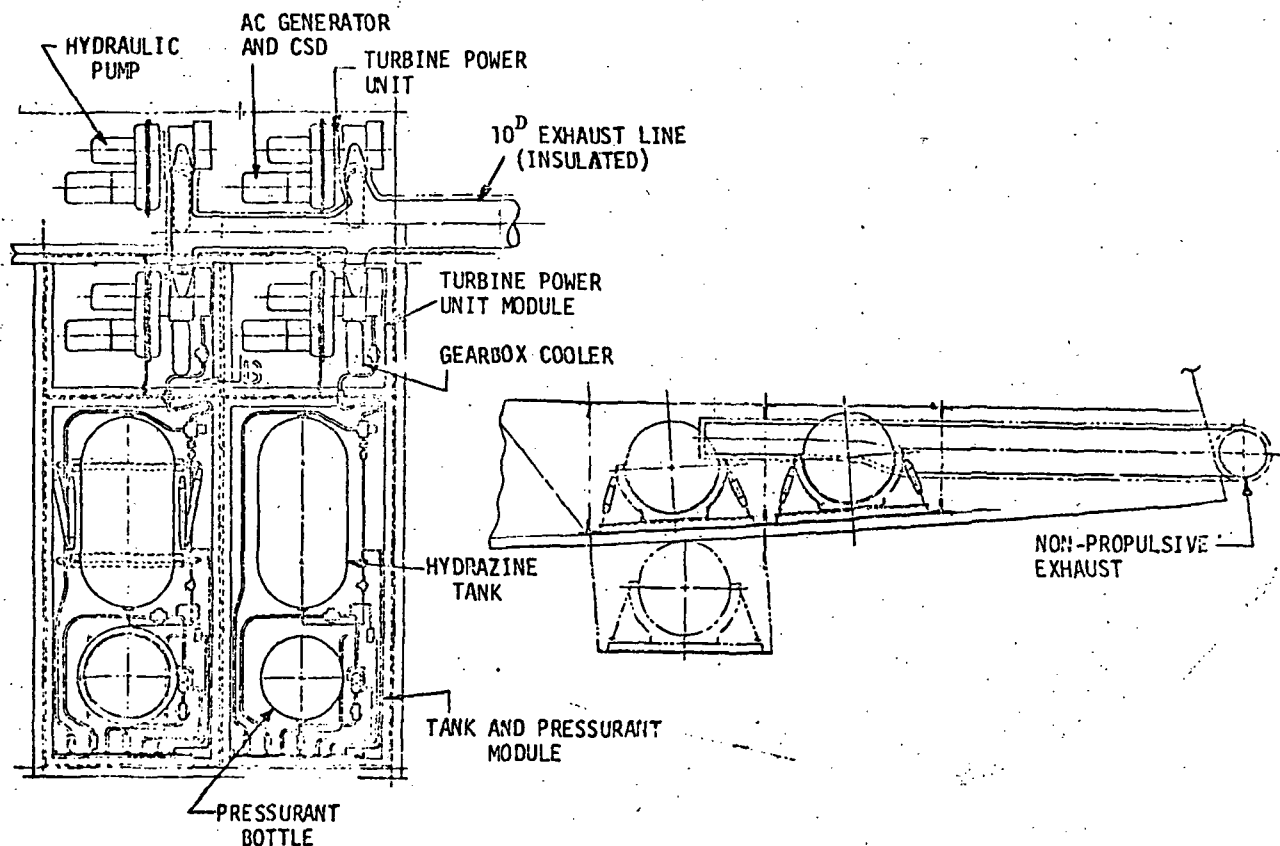


FIGURE 3.5-2

The characteristics of the candidate APU systems are listed below:

<u>APU Type</u>	<u>Fuel Cell Storage and Supply</u>	<u>ECLS/Hydraulic Cooling</u>
GO ₂ /GH ₂	Integrated	Integrated Hydrogen
Phase B Baseline		
Hydrazine	Independent	Water/Freon
Integrated Pumped	Integrated	Integrated Hydrogen
Liquid O ₂ /H ₂		
Dedicated Pumped	Independent	ECLS - Water/Freon
Liquid O ₂ /H ₂		Hydraulic-APU Hydrogen

15 MARCH 1972

The results of the weight/cost study are summarized in Figure 3.5-3. The monopropellant hydrazine APU compares most favorably with the Phase B baseline. A large portion of the cost penalty associated with the weight of the APU subsystems is offset by the lower RDT&E and investment costs of the hydrazine system.

If one assumes the abandonment of the cryogenic ACPS system, the monopropellant hydrazine APU is the best alternative for the following reasons:

- o Lower cost than pumped oxygen/hydrogen systems
- o Least risk

FIGURE 3.5-3

AUXILIARY POWER UNIT PROPELLANT SELECTION
(COMBINED APU/ECLS/HYDRAULIC COOLING/FUEL CELL STUDY)

	PROPELLANT OPTIONS			
	GO ₂ /GH ₂	N ₂ H ₄	INTEGRATED PUMPED O ₂ /H ₂	DEDICATED PUMPED O ₂ /H ₂
TOTAL INTEGRATED WEIGHT (LBS)	4251	6273	4565	5305
COSTS - RTD&E (APU) \$M	42 *	24.5	71.5	68.0
- RTD&E (MECHANICAL SYSTEMS) \$M	63.2 *	79.6	60.2	76.6
- APU INVESTMENT, OPERATIONS AND MAINTENANCE \$M	4.8	3.9	7.2	6.9
PAYLOAD EFFECT @ \$5000/LB	REF	10.1	1.6	5.3
	110.0 *	118.1	140.5	156.8

* DOES NOT INCLUDE APU/MECHANICAL SYSTEMS COST SHARE OF O₂/H₂ TANKAGE OF APPROXIMATELY \$20 M.

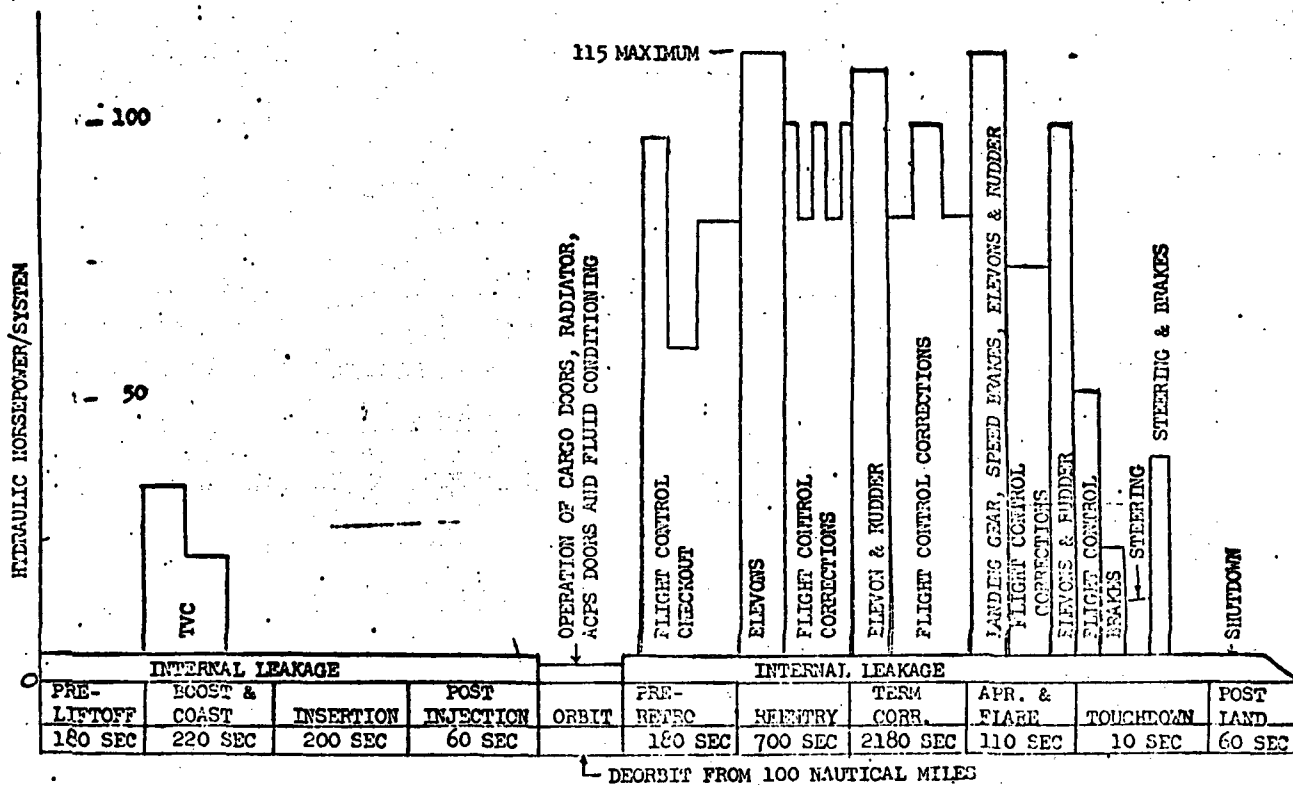
The current hydraulic power profile, which each APU module must meet, is shown in Figure 3.5-4. The maximum requirement is 115 hp. This power profile was expanded into some 50 discrete elements in order to calculate propellant quantity.

15 MARCH 1972

FIGURE 3.5-4

O-40A ORBITER HYDRAULIC POWER PROFILE

(4) 115 HP SYSTEMS)



15 MARCH 1972

A summary of the APU system is given in the following table:

Design Conditions

Number of APU Modules	4
Propellant	Hydrazine
Horsepower per Module	115 (Hydraulic) 175 (Gearbox)
Number of Propellant Tanks	4 (1 per module)
Number of Pressurant Tanks	4 (1 per module)

Weight - Per Module

Helium and Helium Tank	44 Pounds
Turbine Power Unit	127
Propellant Tank	35
Lines, Valves, Misc.	45
Modular Structure	60
Hydrazine	<u>255</u>
	566 Pounds

System Weight

4 Modules @ 566 Pounds	2264 Pounds
Exhaust Line @ 100 Pounds	<u>100</u>
	2364 Pounds

15 MARCH 1972

Auxiliary Power Unit Trade Study - The phase B baseline integrated cryogenic propellant storage and supply system provided propellants, reactants, or coolant to six principal subsystems. In addition to liquid or gaseous oxygen and hydrogen supply for the OMS and ACPS engines, gaseous oxygen and hydrogen reactants were supplied to both the APU and the fuel cells, liquid hydrogen coolant was supplied to the environmental control subsystem and hydraulic subsystem heat exchangers, and gaseous oxygen was supplied for crew station atmospheric make up.

The advent of smaller orbiters, with external hydrogen/oxygen tanks, and smaller attitude control total impulse requirements, renewed interest in storable propellant systems. The potential selection of storable propellants for the ACPS required the restudy of fluid storage and supply for all ancillary subsystems.

A study of the systems weight data resulted in the selection of a pressurized monopropellant hydrazine system for the final storable propellant APU candidate for comparison with the O_2/H_2 systems. Bipropellant systems were discarded for the following reasons:

- o No weight advantage
- o Increased system complexity
- o Increased logistic and maintenance problems for N_2O_4
- o Increased control complexity
- o Decreased reliability
- o Increased development cost

In parallel with the APU propellant trade study, a similar study was conducted for two systems to satisfy the ECLS, fuel cell and hydraulic subsystems requirements. These systems were: (1) a system in which all of the cryogenic oxygen and hydrogen required by these subsystems was integrated with the APU liquid oxygen and hydrogen tanks, and (2) a system in which these fluids were stored and supplied independently of the ACPS, OMS, or APU subsystems.

Two options for ECLS and hydraulic cooling (liquid hydrogen and water/freon) were evaluated for the storable systems to establish the desired mechanical subsystems fluid supplies to be incorporated in the final comparison.

The liquid hydrogen cooling systems are some 200 pounds lighter than the water/freon systems. However, safety considerations, related to continuous hydrogen gas venting during the 30 minute post-landing period while ECLS cooling continues, outweighed the potential weight savings. The water/freon system was selected.

15 MARCH 1972

4.0 Orbiter H₂O Tank (Series Burn-Pressure Fed)

An H₂O tank is expended every orbiter flight. This requires that the configuration design and design details be considered conscientiously in order to minimize cost, in light of the 453 units to be constructed. However, the innovations incorporated to reduce cost must not result in undue weight penalty (reduction in mass fraction). Unique designs, processes and manufacturing methods must be exploited to achieve the desired results in the finely balanced trade studies between weight, cost and schedule. One of the major difficulties facing the tank designer will be to "Keep it simple" and to prevent the H₂O tank from becoming the equivalent of another stage.

4.1 Phase B Extension Final Design

The impact of the final sizing to a propellant weight of 1,195,600 lb is shown in Figure 4.1. The only significant changes from the analysis baseline, to be discussed later, are the tank dimensions shown, an increase in tank inert weight to 62,131 pounds, and a change in mass fraction to 0.9506.

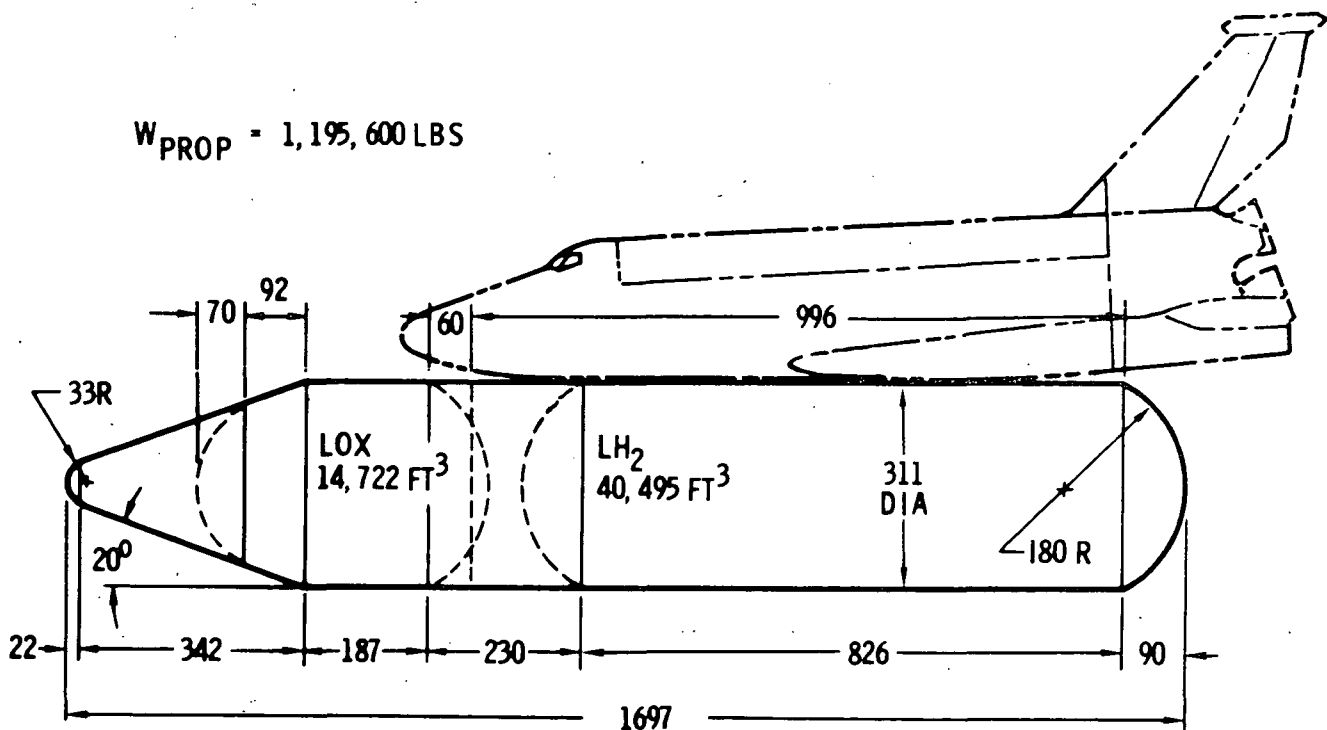


Figure 4.1 Orbiter H₂O Tank Final Phase B Configuration Series Burn - Pressure Fed Booster

4.2 Analysis Baseline

The Orbiter H0 tank (propellant weight = 942,100 lb) has been analyzed and designed for the series burn configuration using the pressure fed booster. The ascent configuration for the vehicle is shown on Figure 4.2.

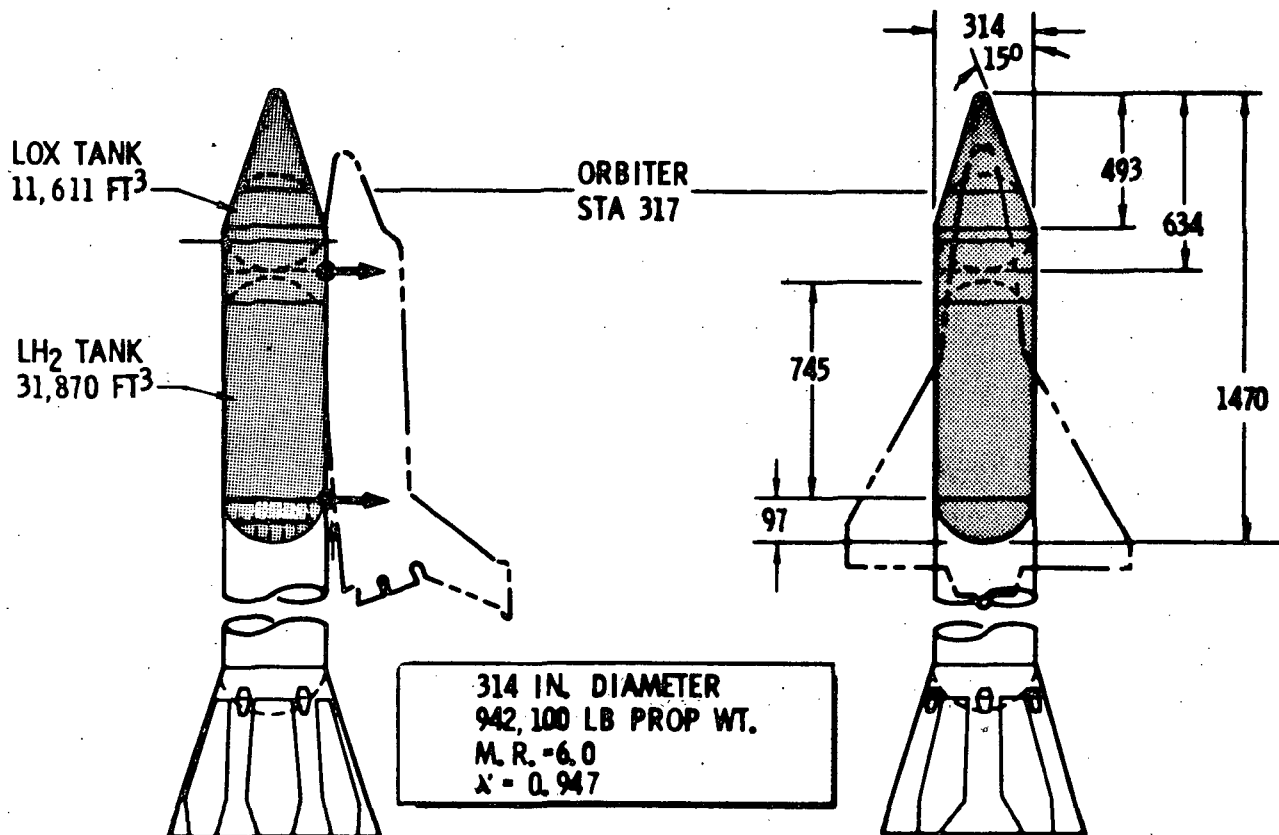


Figure 4.2 Orbiter H0 Tank Series Burn-Pressure Fed Booster

4.2.1 Configuration Definition

The analysis baseline used for the H0 tanks is shown on Figure 4.2.1-1. An inboard profile of this tank is shown on Figure 4.2.1-2. All of the major subsystems are shown in this figure, while significant configuration characteristics relating to it are shown on Table 4.2.1-1. Mass properties are shown on Table 4.2.1-2. This table shows two weight statements: the preliminary parametric weights, which are consistent with the mass properties status report, and the completed analysis weights, which represent a later iteration in design and analysis. Included in these weights is a contingency allowance of 5% for the body group, 10% for the induced environmental protection, and a 2% growth and uncertainty allowance on the total dry weight.

15 MARCH 1972

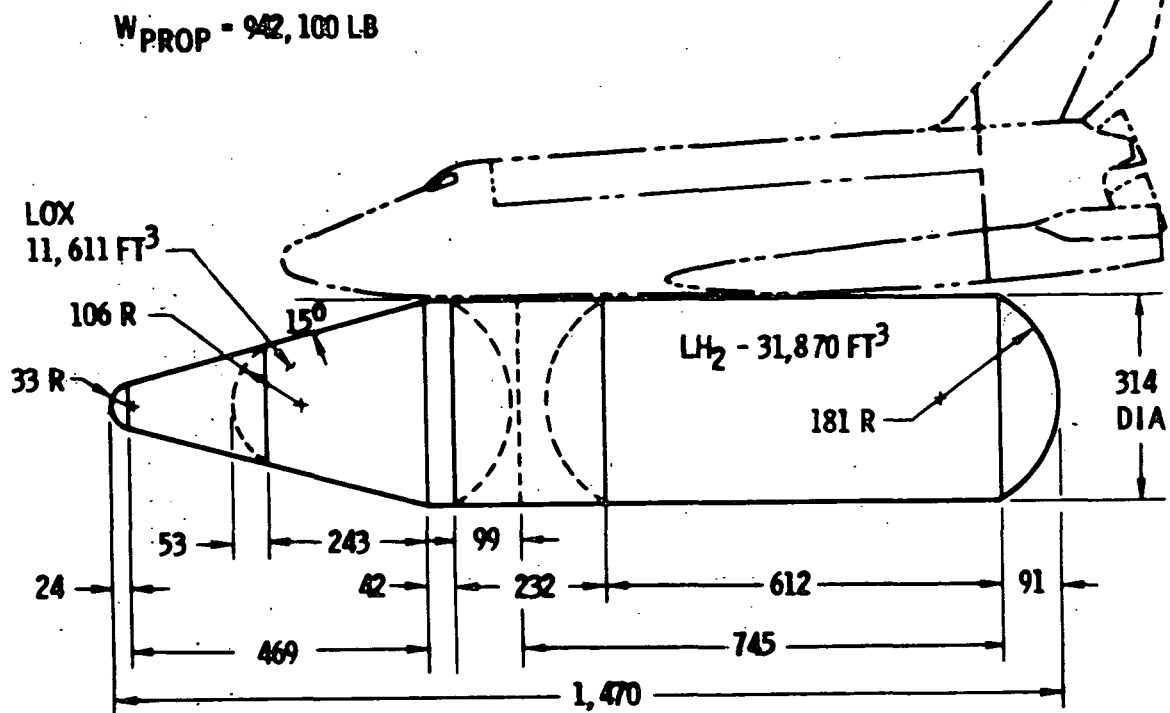


Figure 4.2.1-1 Tank Geometry - Series Burn

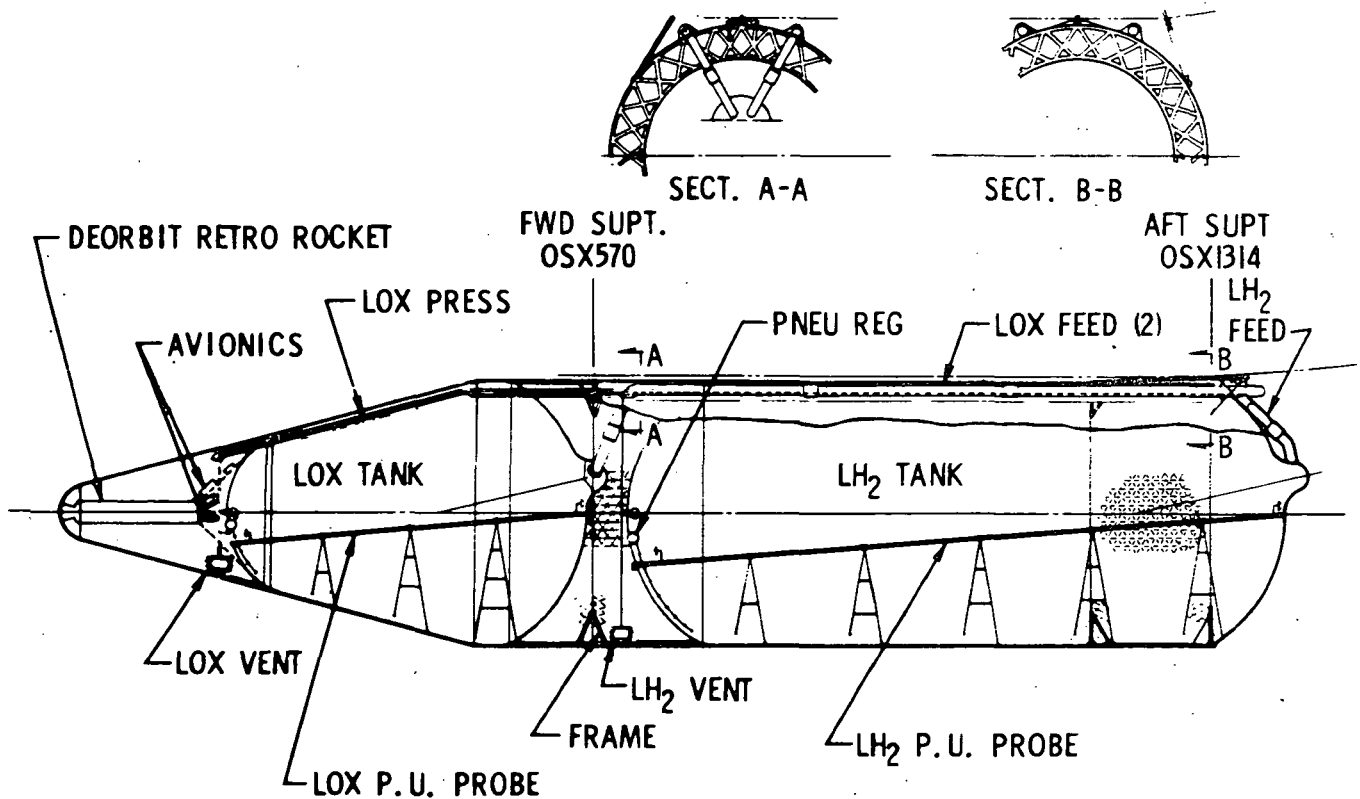


Figure 4.2.1-2 Inboard Profile - Series Burn

TABLE 4.2.1-1 ORBITER HO TANK DESCRIPTION

GROSS WEIGHT	994,820 LB
USABLE PROPELLANT WEIGHT (INCLUDES FPR)	942,100 LB
MIXTURE RATIO	6.0/1
PROPELLANT MASS FRACTION λ'	0.947
PROPELLANT TANKS	
○ DIAMETER	314 INCHES
○ LENGTH	1,470 INCHES (L/D = 4.7)
○ MATERIAL	2219 ALUMINUM
○ PROPELLANT LOCATION	LOX FORWARD
○ BULKHEADS	SEPARATE TANK
○ BOOSTER/ORBITER ATTACH	BOOSTER END LOAD/ORBITER AFT
FEED LINES	LOX - 2 X 12 INCHES DIAMETER
	LH ₂ - 1 X 16 INCHES DIAMETER
	1 X 6 INCHES DIAMETER
VENTS	TANK LOCATED
PROPELLANT UTILIZATION	CLOSED LOOP
INSULATION/TPS	EXTERNAL FOAM WITH FIREX - 250 ABLATOR
DEORBIT	SINGLE SRM IN NOSE CONE

Table 4.2.1-2 Weight Summary - Series Burn

	PRELIMINARY PARAMETRIC WTS.	COMPLETED ANALYSIS WTS.		PRELIMINARY PARAMETRIC WTS.	COMPLETED ANALYSIS WTS.
BODY GROUP	[36,782]	[36,646]	ELECT/INSTRUMENTATION	[345]	[277]
FUEL TANK	(19,810)	(18,959)	BATTERIES		20
FWD BULKHEAD	2,170	2,004	POWER DISTRIBUTION		20
SIDEWALL	15,310	15,112	EBH FIRING UNITS		29
AFT BULKHEAD	2,330	2,187	PU ELECTRONIC		10
OXIDIZER TANK	(8,037)	(7,017)	RANGE SAFETY		30
FWD BULKHEAD	432	201	INSTRUMENTATION SYSTEM		17
SIDEWALL	4,650	4,231	COAX AND WIRING		151
BAFFLES	105	105	DEORBIT SYSTEM	[2,520]	[1,740]
AFT BULKHEAD	2,850	2,480	GROWTH/UNCERTAINTY (6.2%)	[2,906](2%)	[1,010]
INTERTANK STRUCTURE	(4,410)	(3,880)	DRY WEIGHT	49,527	47,747
BOOSTER/ORB/TANK ATTACH	(3,370)	(5,371)	RESIDUAL PROPELLANTS	[3,193]	[3,185]
NOSE FAIRING	(409)	(329)	TANK UNDRAINABLES	400	400
UMBILICAL PANEL	(300)	(300)	FEED LINE TRAPPED	681	671
TUNNEL	(446)	(446)	PRESSURANT	1,772	1,774
INDUCED ENVIRONMENT PROTECT.	[3,440]	[3,583]	PU BIAS	336	336
ABLATOR	2,950	2,737	CHILLDOWN	4	4
FOAM	360	387	INERT WEIGHT	52,720	50,932
PRIMER, PAINT & SEALER	130	459	START PROPELLANT	0	1,388
PROPELLANT SYSTEMS	[3,534]	[4,491]	USABLE PROPELLANT	[942,100]	[942,100]
FEED SYSTEM	2,329	2,286	TOTAL GROSS WEIGHT	994,820	994,418
PRESS AND VENT	690	1,588	$\lambda' = \frac{W_{PROP}}{W_{GROSS}}$ (USEABLE)	0.947	0.949
SUIPS & VORTEX CONTROL	220	220	$\frac{W_{GROSS}}{W_{GROSS}}$ (LESS START PROP)		
PU SYSTEMS	185	185			
PNEUMATICS	42	44			
RECIRCULATION	68	168			

15 MARCH 1972

4.2.2 Vehicle Analysis

The structural loads on the vehicle were calculated for a range of conditions and the most critical loads were determined for use in the tank design. Figures 4.2.2-1 and 4.2.2-2 depict the maximum loads imposed on the tank structure from both ground and flight conditions.

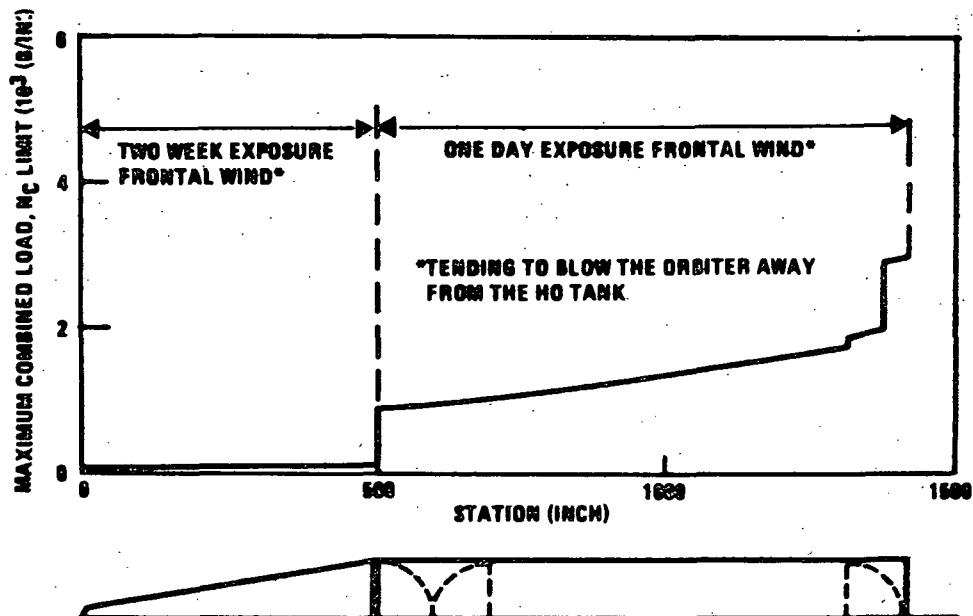


Figure 4.2.2-1 Maximum Load Envelope, Ground Wind Conditions - Series Burn

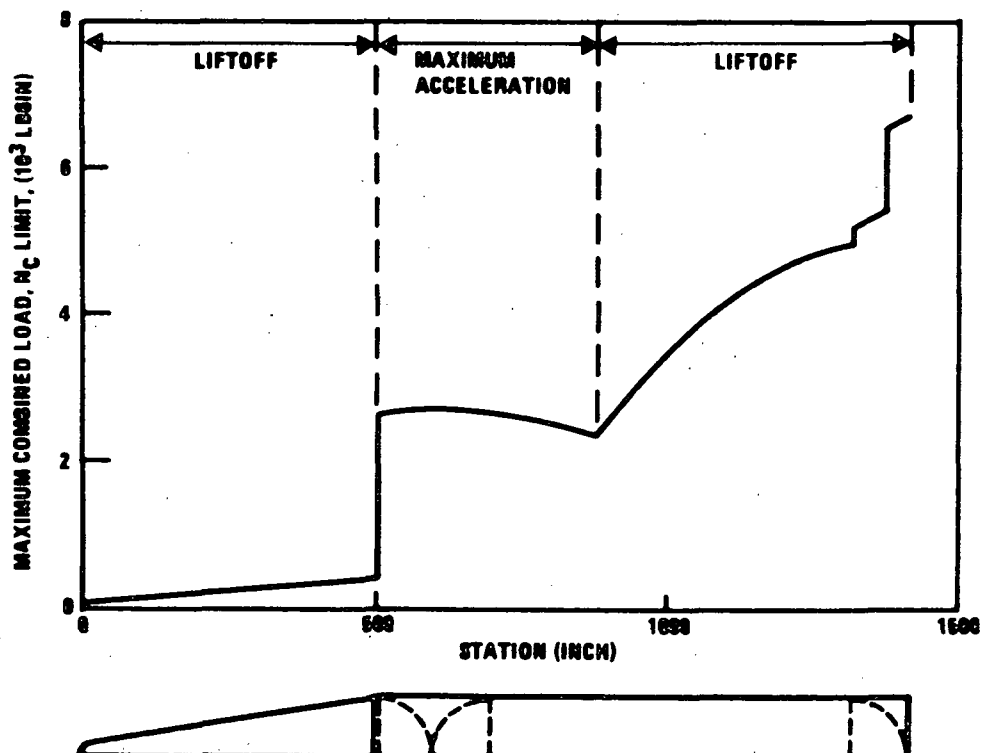


Figure 4.2.2-2 Maximum Load Envelope, Ascent Conditions - Series Burn

15 MARCH 1972

Ground Wind Loads -- Loads were determined for the following four ground wind conditions:

1. One day exposure frontal winds (tending to blow the orbiter away from the H0 tank), with vehicle tanks full.
2. One day exposure side winds, with full vehicle tanks
3. Two week exposure frontal winds (tending to blow the orbiter away from the H0 tank), with vehicle tanks empty
4. Two week exposure side winds, with vehicle tanks empty.

A summary of vehicle parameters associated with the conditions analyzed is shown in Table 4.2.2-1.

TABLE 4.2.2-1
SUMMARY OF VEHICLE PARAMETERS
GROUND WIND CONDITIONS

LOAD CONDITIONS	WIND DIRECTION	TANKS	VEHICLE WEIGHT (LB)
ONE DAY EXPOSURE WINDS	FRONTAL	FULL	5,296,460
ONE DAY EXPOSURE WINDS	SIDE	FULL	5,296,460
TWO WEEK EXPOSURE WINDS	FRONTAL	EMPTY	1,038,460
TWO WEEK EXPOSURE WINDS	SIDE	EMPTY	1,038,460

Ascent Loads -- Structural loads were determined for the following six ascent conditions:

- 1) Liftoff; 2) Max β_q /sidewind; 3) Max α_q /headwind; 4) Max α_q /tailwind;
- 5) Max. acceleration and 6) Post-staging. A summary of vehicle parameters associated with the conditions analyzed is shown in Table 4.2.2-2.

15 MARCH 1972

TABLE 4.2.2-2
SUMMARY OF VEHICLE PARAMETERS, ASCENT CONDITIONS

LOAD CONDITION	TIME (SEC)	MACH NO.	q (PSF)	α (β) (DEG)	Nx (g's)	Ny (g's)	Nz (g's)
LIFTOFF	0	0	0	0	1.245	0	0.308
MAX β q/SIDEWIND	55	0.90	559.9	3.6 (6.48)	1.5	0.186	-.025
MAX α q/HEADWIND	61	1.13	712.5	7.87	1.55	0	-.286
MAX α q/TAILWIND	70	1.18	524.7	-4.2	1.78	0	0.264
MAX ACCELERATION	118	3.83	211	0	3.0	0	-.06
POST-STAGING	131.5	4.65	110	0	1.1	0	0.311

The envelope of maximum rigid body limit loads associated with these ascent conditions is described in Figure 4.2.2-2. The loads shown are combined axial and moment compression, with unpressurized tanks. Two conditions generate the maximum loading: liftoff and maximum acceleration. These loads may not be design loads, however, when internal pressure, ambient pressure, temperature and dynamic amplification factors are considered in the sizing program that is used to obtain material gages. This is discussed in Section 4.2.3.

4.2.3 Vehicle Design

This section contains a description of the design tasks performed on the structural, thermal protection, propulsion, and avionic subsystems.

4.2.3.1 Structural Design

The HO tank consists of five basic structural systems: the nose fairing, LOX tank, intertank, LH₂ tank, and orbiter attach structure, as shown on Figure 4.2.3-1.

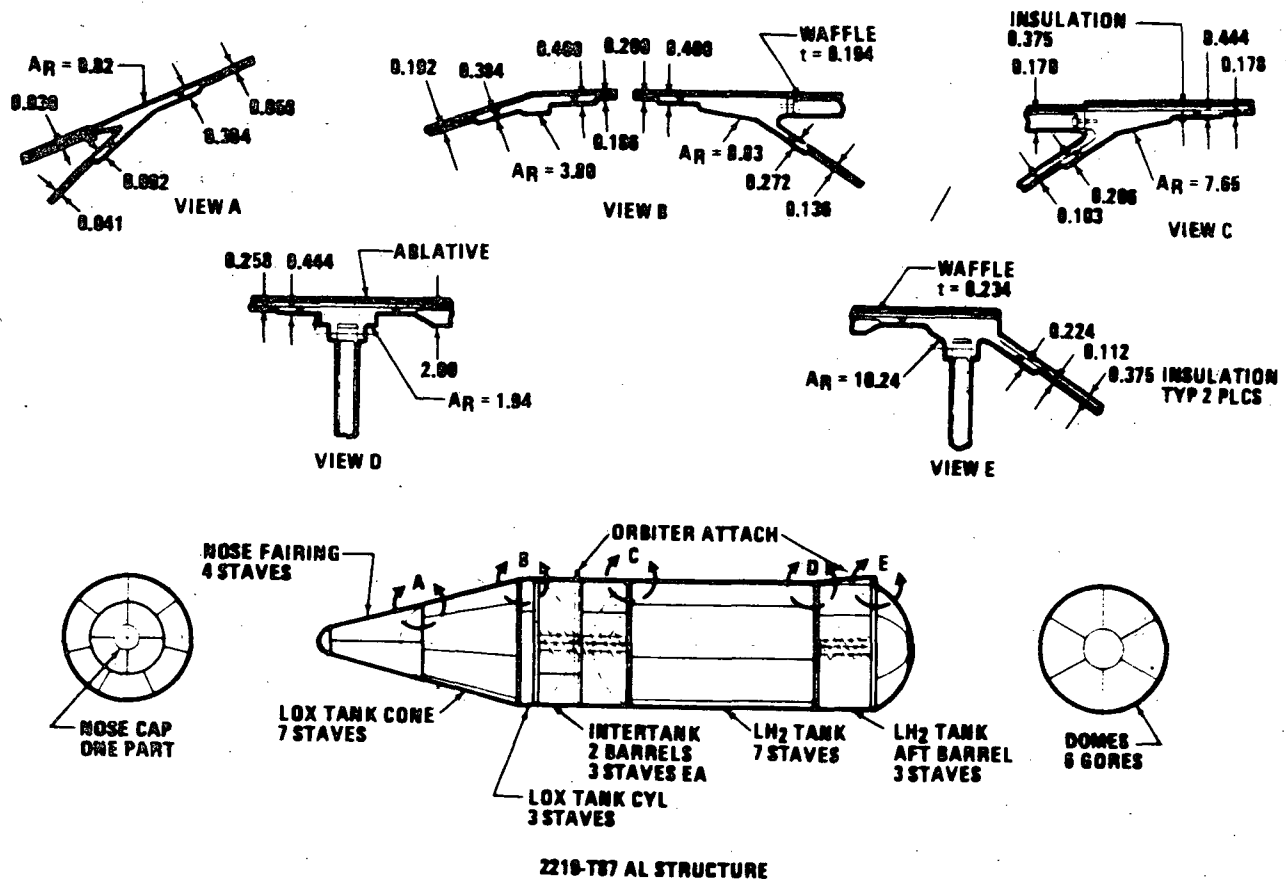


Figure 4.2.3-1 Basic Structure Series Burn

Structure Description -- The nose fairing, which houses the deorbit rocket, consists of a cone and a nose cap. The cone is designed to resist flight loads, and the nose cap is designed to resist local aerodynamic pressure. These are mechanically attached to each other and the assembly is mechanically attached to the LOX tank. The LOX tank is made up of a forward dome, a seven-stave cone, a cylinder having 3 staves, and an end dome made of 6 gores and one polar cap. Each component is of welded monocoque construction and is welded to adjacent components through Y-rings. The staves

15 MARCH 1972

in the cone and cylinder are machine tapered skins surrounded by constant thickness weld lands. Dome gores are constant thickness machined skins and constant thickness weld lands. Two integrally stiffened machined waffle (isogrid) barrels are used to form the intertank structure. Three staves are bolted together to form each barrel. The barrels are then bolted together and to the LOX and LH₂ tank Y-rings. The LH₂ tank cylinder has a long forward monocoque section made up of 7 welded machine tapered staves with constant thickness weld lands and a short aft isogrid barrel made up of three welded staves. The end domes consist of 6 gores welded together and to a polar cap. They are constant thickness machined skins with weld lands. Each component is welded to adjacent components through machined rings. Orbiter attach structure is considered in two parts, forward and aft. The forward attach structure consists of an external link hinged to an internal frame which is bolted to the intertank waffle structure. The aft attach structure consists of an external fitting which is bolted to the aft isogrid barrel and two frames at each end of the barrel. The inner caps and diagonal truss members of the frames are machined as one part from plate stock and mechanically attached to the outer caps which are welded to the LH₂ cylinder and aft dome as shown. The outer frame cap is also the Y-ring.

Attach Structure -- Sizing of the H₂O tank forward and aft attachment structure to the orbiter is based on the most critical condition experienced. Conditions investigated were liftoff, maximum βq (20.3K sidewind), maximum αq (25.7K headwind), maximum αq (35.7K tailwind), and maximum acceleration. Results of this investigation for the forward and aft attachments are shown in Tables 4.2.3-1 and 4.2.3-2. The attach structure is shown in Figure 4.2.1-2.

Table 4.2.3-1 Forward Attach Structure Design

ITEM	DESIGN CONDITION	STRESS	ALLOWABLE	M.S.	WEIGHT (LBS)	
Sway Brace Tube (2) Fitting (4)	Max. αq (HW)	$f_t = 63,000$ $f_{br} = 100,000$	$F_{tu} = 63,000$ $F_{br} = 100,000$	0 0	$2 \times 87 = 174$ $4 \times 11 = 44$	218
Side Link	Liftoff	$f_c = 8,300$	$F_c = 11,200^*$	.35		5
Frame Outside Cap Inside Cap Diagonals	Max. αq (HW)	$f_t = 94,500$ $f_t = 94,500$ $f_c = 66,800$	$F_{tu} = 94,500$ $F_{tu} = 94,500$ $F_{cy} = 66,800$	0 0 0	238 178 757	1173
					TOTAL	1396

Table 4.2.3-2 Aft Attach Structure Design

ITEM	DESIGN CONDITION	STRESS	ALLOWABLE	M.S.	WEIGHT (LBS)	
Frame Outside Cap Inside Cap Diagonals	Max. αq (TW)	$f_t = 94,500$ $f_c = 66,800$ $f_c = 66,800$	$F_{tu} = 94,500$ $F_{cy} = 66,800$ $F_{cy} = 66,800$	0 0 0	142 553 1535	2230
Kick Frame Caps Web	Max. Acc.	$f_t = 94,500$ $f_s = 38,000$	$F_{tu} = 94,500$ $F_{su} = 38,000$	0 0	341 200	541
Drag Ld. Fitting Web Flange Tank Flange	Max. Acc.	$f_s = 38,000$ $f_c = 50,000$ $f_c = 50,000$	$F_{su} = 38,000$ $F_{cy} = 50,000$ $F_{cy} = 50,000$	0 0 0	43 275 15	333
Skin Doubler	Max. Acc.	$f_c = 50,000$	$F_{cy} = 50,000$	0		227
Orb. Load Links	Max αq (TW)	$f_c = 50,000$	$F_{cy} = 50,000$	0		123
Side Load Straps	Liftoff	$f_t = 63,000$	$F_{tu} = 63,000$	0		32
					TOTAL	3486

15 MARCH 1972

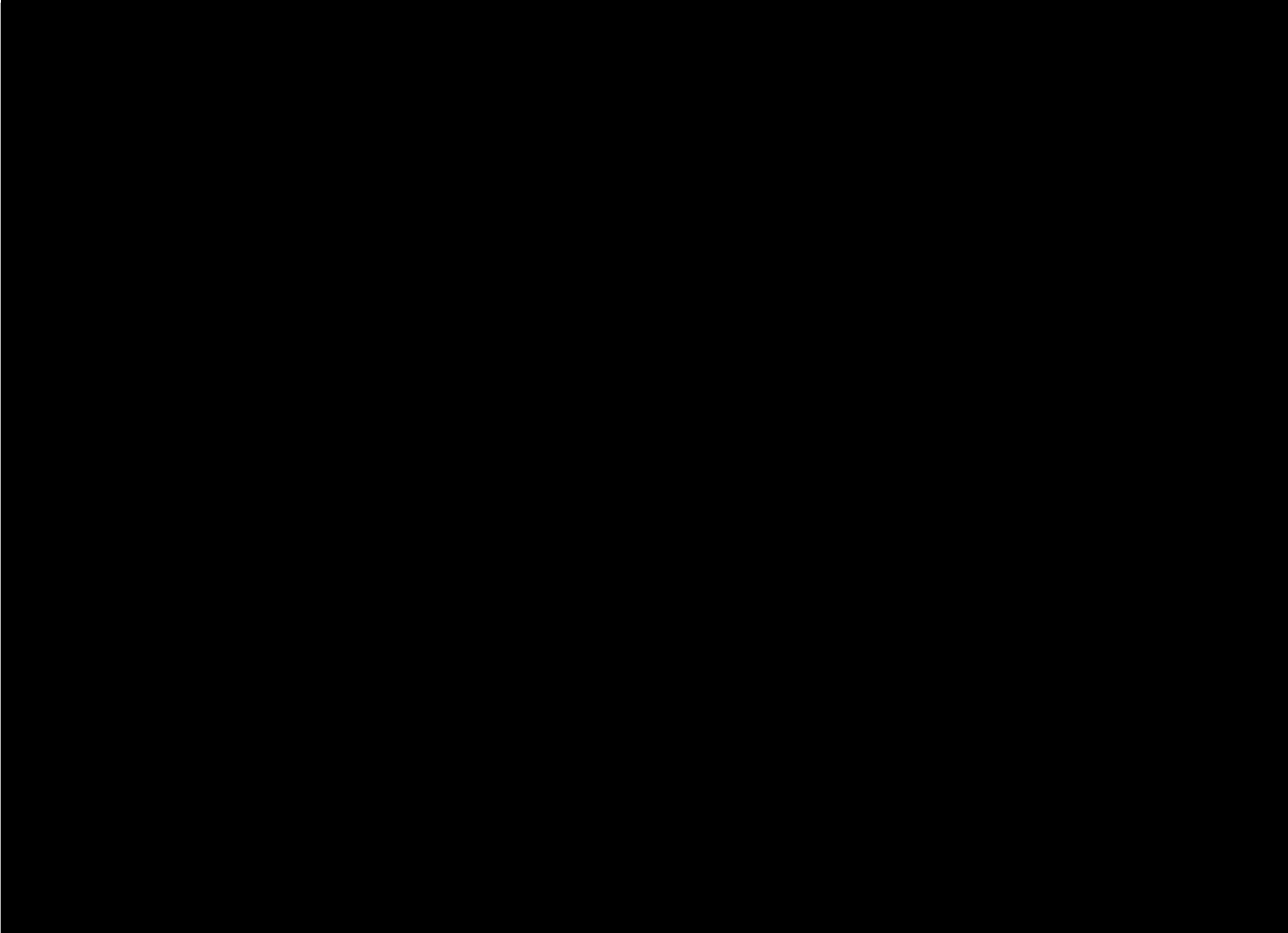
LOX Tank -- The LOX tank is designed as a pressure vessel capable of withstanding 30 psi ullage pressure and LOX hydrostatic head. The maximum head pressure in the tank results from the 3 g flight condition with the tank full. Wall temperatures used in the analysis are -298°F in the liquid area and $+45^{\circ}\text{F}$ in the ullage area. The all welded monocoque construction requires the staves and gores to be machined from plate, to provide a skin sized to resist pressure loading, with a weld land around the periphery. The weld land is constant thickness. The forward dome is designed for ullage pressure only, which resulted in a constant minimum skin thickness requirement of 0.033. To obtain this thickness chem milling after machining is required. This dictates tolerances of ± 0.008 and a nominal gage of 0.041. LOX tank conical sections taper from a nominal size of 0.058 inches at the forward end to 0.192 inches at the aft station. The forward end is sized to resist ullage pressure only, while the aft end is sized for ullage plus head pressure. Chem milling is not required; so tolerances of ± 0.005 were used to derive nominal sizes. The cylindrical section tapers from a nominal size of 0.186 inches at the forward section to 0.200 inches at the aft station. The entire cylinder is sized from ullage and head pressure. Tolerances of ± 0.005 were used. The aft dome was designed to resist ullage and the maximum head pressure at the most aft point. The result is a constant nominal skin thickness of 0.136 inches over the entire dome, using tolerances of ± 0.005 . Tapering of the gores was not considered because constant thickness is required for stretch forming, and the cost of chem mill tapering after forming is not warranted by the small weight saving. Y-rings are provided at the interfaces of all sections to resist the radial component of membrane load at these planes. The ring areas derived are indicated in Figure 4.2.3-1.

LH₂ Tank and Intertank -- The LH₂ tank and intertank structure are designed using a computer program developed to determine minimum weight designs for H₂ tanks. The program determines the thickness requirements

15 MARCH 1972

for the cylindrical sidewall and uses predetermined weight equations for end domes, welds, and propellant residuals. The program determines the most critical of seven failure modes: hoop stress (burst, yield, proof), longitudinal tension (yield, ultimate), and longitudinal compression (yield, buckling). For buckling, the monocoque structure was analyzed using the method described in NASA SP-8007, "Buckling of Thin-Walled Circular Cylinders." The waffle (isogrid) structure was analyzed using the methods of MDC E0375, "Space Shuttle Data," Part III, Booster, Airframe, Appendix B.

The final design for the LH_2 tank is selected after determining the total LH_2 tank weights for numerous operating pressure levels. These weights include the contributions from the basic sidewall, end domes, Y-Rings, welds, and propellant residuals. Figure 4.2.3-2 presents the weight summary of these analyses for both the baseline design and an alternate all-isogrid sidewall design. Both of these curves are respective loci of all minimum weight designs and include the adverse effects of a pressure tolerance of ± 1.0 psia. As can be seen in the figure, the optimum pressures are 65 psia for the baseline design and 40 psia for the all-isogrid design.



15 MARCH 1972

The critical design conditions for the baseline tank are shown in Figure 4.2.3-3. It is seen that the intertank region is designed by buckling at the maximum acceleration condition. Within the tank, between Stations 694 and 1000, the maximum acceleration condition is still the governing design condition, but the mode of failure changes to yield hoop stress. The tank between Stations 1000 and 1180 is buckling-critical at liftoff, and the isogrid section between Stations 1180 and 1320 is designed by both buckling at liftoff and yield hoop stress at maximum acceleration. Nominal dimensions for the resultant structure are depicted in Figure 4.2.3-1.

Additional work was also done to investigate the ability of the shell to withstand ground wind loads on the pad. Of the four conditions analyzed (see Table 4.2.2-1), the critical case proved to be frontal winds with one day exposure and both tanks full. [REDACTED]

[REDACTED] However, for an all-isogrid design, no internal pressure would be required to withstand ground wind loads. This later design was not chosen because of its considerably higher cost.

Fracture Control Methods for HO Tank -- A room temperature proof pressure test will provide an assessment of manufacturing workmanship, material quality, and permit a definition of flaw sizes remaining after this test. The proof test enhances flaw detection and will aid the 100% weld inspection planned to be conducted after the proof test. Strict process control and in-process inspection will be employed for parent metal and welds including the use of single pass welds to guard against lack of weld penetration defects.

A review was made of problems encountered by MDAC and others in the design, fabrication and testing of large cryogenic aluminum tanks. This review indicated that problems were primarily associated with welds and not the parent material. On rare occasions, cracking has been encountered during the forming of integrally stiffened ribs. Such cracks are readily detectable

15 MARCH 1972

within current nondestructive testing (NDT) capability, however, although strict process control is required to ensure that cracks are not inadvertently overlooked. Two types of weld defects have caused proof test failures. The first is a lack-of-weld-penetration defect which is considered undetectable by current NDT methods.

To prevent such a defect from being accepted, process control techniques were successfully developed for the Saturn S-IVB. Acceptance of the defect was prevented by using single pass welds, chamfering the root side of the weld, and visually examining the root side of the weld prior to shaving to ensure that the chamfer had been melted away. The other critical defect was associated with multiple hand repair welds which resulted in a low strength weld. The use of mechanized repairs and limiting the number of repairs has prevented this problem from reoccurring. These process controls will be adopted for fracture control of HO tanks.

A review of fracture mechanics test data indicated that a room temperature proof test would screen out critical flaws in parent material at LOX temperature, as shown in Figure 4.2.3-4. A room temperature proof test will also screen to flaw sizes that are only slightly larger than those critical at -423°F, but the toughness does not increase sufficiently compared to the increase in strength to screen out critical flaws. In-process control and 100% NDT inspection will be depended upon to detect critical flaws in the parent material. Data for toughness of welds at cryogenic temperatures indicates that room temperature proof tests will detect large flaws in welds, approximately 3-1/2" and larger. However, critical flaws at operating conditions could be as small as approximately 2-1/2". It is difficult to conceive of defects of this length arising except from a lack of penetration weld. Any other type of critical defect is expected to be open to the surface and consequently be readily detectable by conventional NDT techniques, particularly after a proof test. It is planned to remove detected cracks even though they may be less than a critical size.

15 MARCH 1972

The technology required to implement a reliable fracture control plan is considered adequate, although selected data acquisition is recommended, along with the development of techniques to ensure that the process control requirements are realized. The latter is necessary to minimize the chance of human error going undetected. Wide panel fracture toughness tests are considered desirable in the thicknesses and temperatures of interest, including effects of a room temperature proof test and subsequent cryogenic toughness determination.

4.2.3.2 Thermal Protection System

The baseline external cryogenic insulation is BX-250 polyurethane foam, coated with a low temperature ablator Firex 250, as shown in Figure 4.2.3-5. The prime factors considered in selecting the insulation system for the orbiter drop tanks were: reliability, thermal efficiency (weight) and cost. The foam was selected because of its light weight, 2 lb/ft³, and proven success on cryogenic flight hardware. The Firex 250 was selected because of its low ablation temperature, around 270°F, and its relatively low cost. Another low temperature ablator, Thermolag has several shortcomings, such as high water absorption, instability at ambient temperatures, contamination, and cost. No other ablatives have been identified with an ablation temperature as low as 270°F. The low ablation temperature helps minimize the thickness of ablator required to keep the foam from exceeding 200°F, where it becomes chemically unstable. The thickness of ablator specified has been adjusted throughout the vehicle to provide a maximum backside temperature of 200°F, as shown in Figure 4.2.3-5.



15 MARCH 1972

Thermal Environment -- Figure 4.2.3-6 shows the aerodynamic heating history used to size the insulation system. Base insulation requirements will be very slight for this configuration, as the base is protected from the high radiation intensity of the Booster plume, and the level of radiation intensity from the plume from the Orbiter engines is very low.

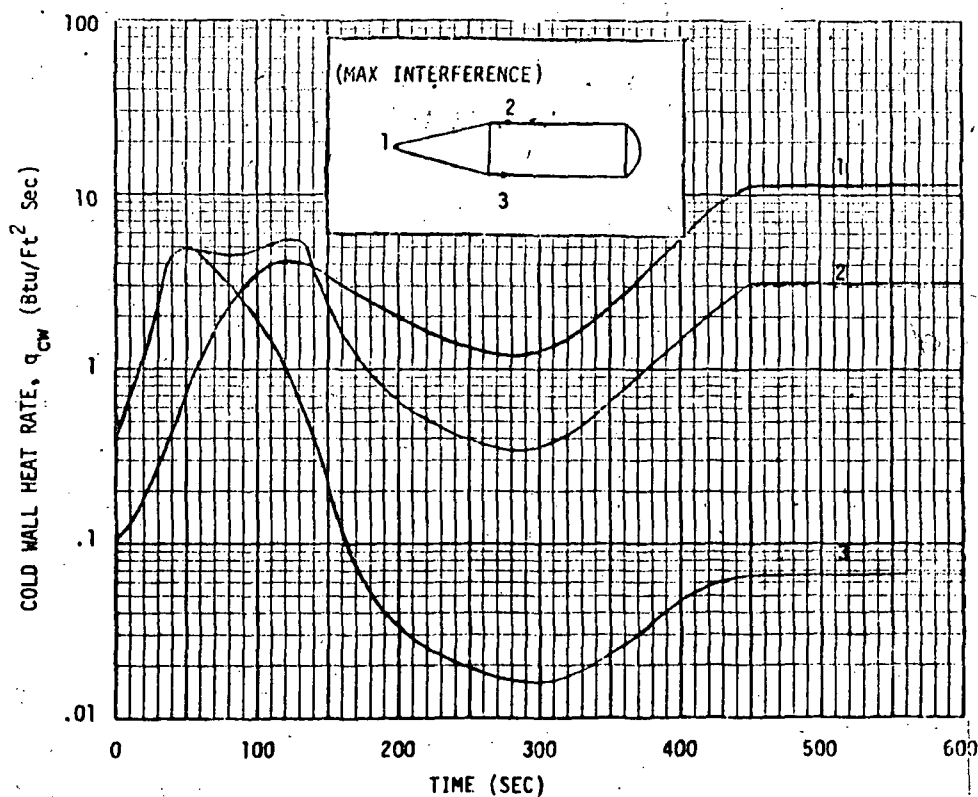


Figure 4.2.3-6 Aerodynamic Heating - Series Burn

15 MARCH 1972

TPS Materials -- A cross-section of the TPS and their unit weights are shown in Figure 4.2.3-7. Even though this system was selected as the baseline, further development is required.

Problems with the baseline system are primarily related to the areas of interference heating where ablative materials are needed. One problem is that the polyurethane foam cannot resist the high temperature encountered during boost. Above 200°F the foam begins to outgas, resulting in entrapped gases under the ablator. This gas pressure is sufficiently high to blow off the ablator. The other major problems associated with this system, are separation of the ablator from the foam during cryogenic cooldown and cryopumping. At lower temperatures, the tank and foam contract to a smaller diameter. The ablator, being a rigid ring (plastic or cork) around the foam resists the loads developed during contraction of the tank. Failure in the foam or the foam/ablator interface bond may occur if thermal stress loads exceed the strength of the foam or bond. This problem was experienced during the S-II insulation development program. Cork/foam separation occurred, especially at the edges. The problem was solved by using honeycomb to increase the bond strength of cork to insulation. This system is expensive and results in high weight penalties, and has not been used directly on the cryogenic tank. This is another reason why Firex 250 was selected as the ablator. Firex is a polyurethane base ablator that has good adhesion to the polyurethane foam. The material is sprayed continuously over the foam surface with no joints or edges where cryogenic strain failure can initiate. Cryopumping is another potential problem. It can be minimized by reducing the number of joints in the cryogenic insulation, ablator, and barrier coating. The use of counter-rotating spray nozzles for applying the polyurethane foam to the tank can eliminate this joint close-out problem. The spraying operation is uninterrupted: new foam is not sprayed over existing foam. The starting joint and completion joints are sprayed simultaneously by starting the spray nozzles together, traveling in opposite direction and coming together on the opposite side. The ablator and barrier coatings are sprayed as continuous coatings eliminating joints which usually provide the paths for cryopumping.

15 MARCH 1972

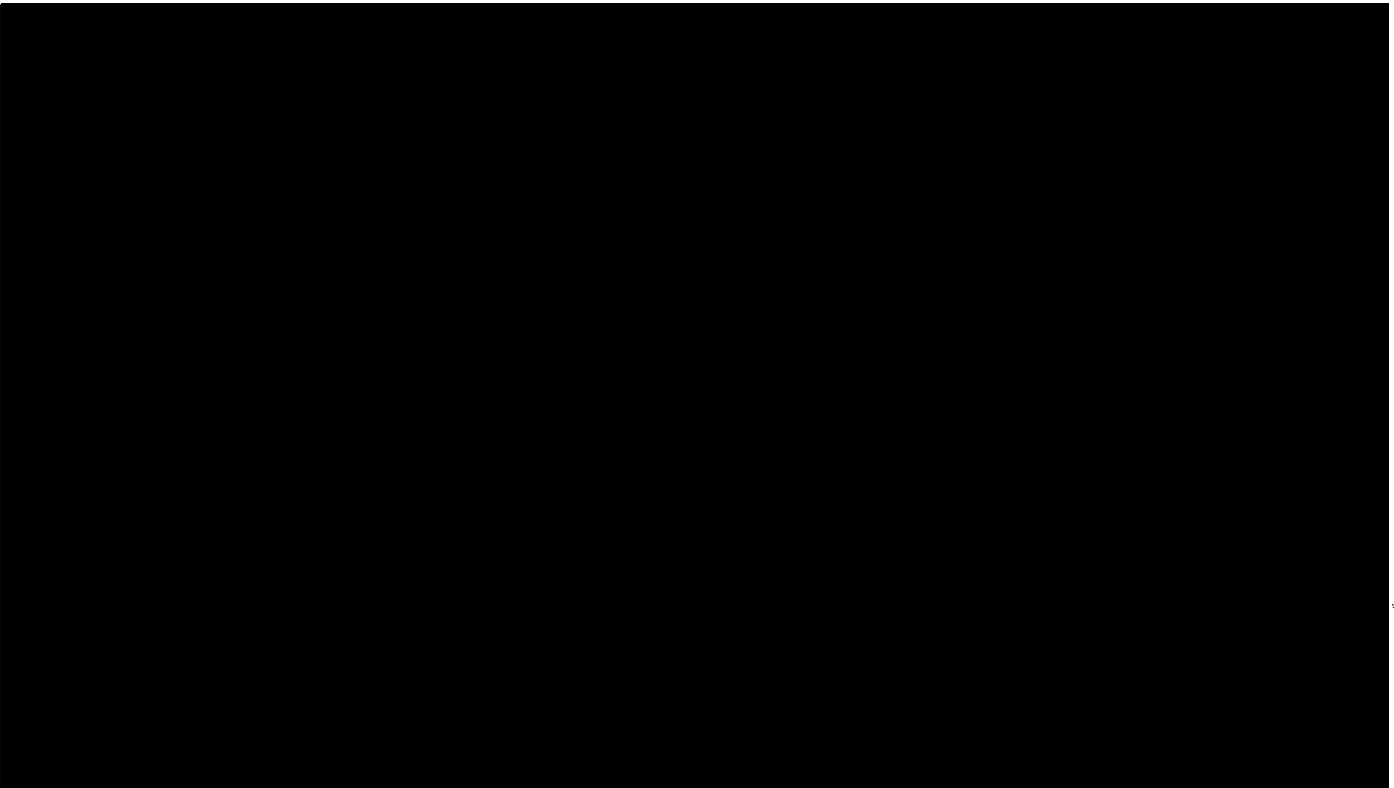


Figure 4.2.3-7 Thermal Protection System Series Burn

These problems have led us to investigate other insulation concepts designed to overcome the problems associated with cryogenic external insulations. These concepts are shown in Figure 4.2.3-8. Preliminary test results on materials which have been selected for the various construction shown appear very promising. Table 4.2.3-3 lists results of work accomplished to date. In addition to superior performance in terms of reliability and efficiency, these new insulation constructions should prove less expensive to install on the tank.

4.2.3.3 Propulsion Subsystem

The Orbiter HO tank for the series burn configuration consists of the LOX and LH₂ feed systems, LH₂ recirculation system, LOX and LH₂ vent systems, LOX and LH₂ pressurization systems, a pneumatic system, and sump and vortex control systems. Figure 4.2.3-9 shows a schematic of the basic tank main propulsion subsystems.

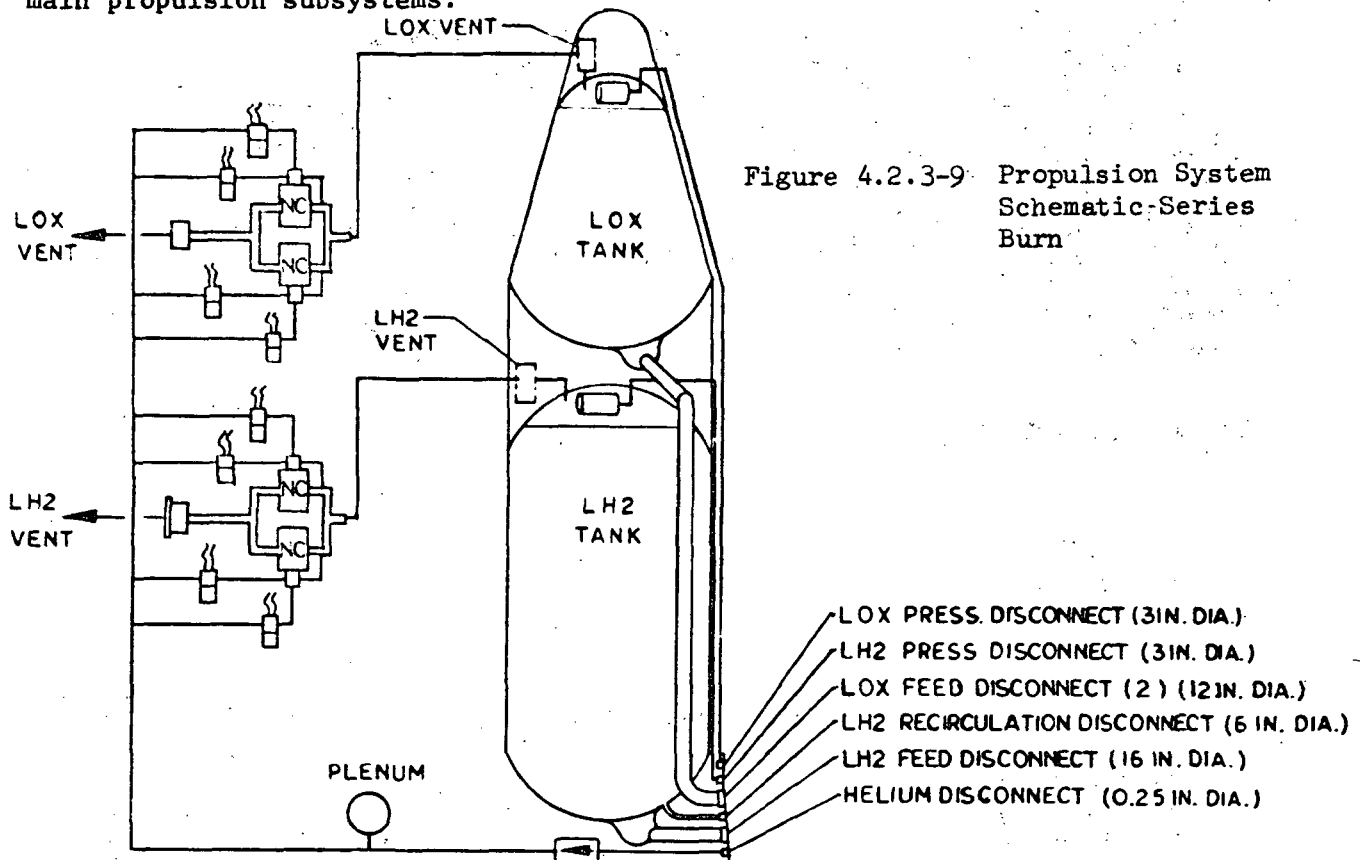


Figure 4.2.3-9 Propulsion System Schematic-Series Burn

LOX is supplied to the three 472K (vac) thrust H₁ P_c engines from the tank through two 12-inch diameter feed lines which are routed down the side of the HO tank and enter the orbiter in the engine area. Hydrogen is supplied through a single 16-inch diameter line. Figures 4.2.3-10 and 4.2.3-11 show these feed systems. A chilldown line also supplies LH₂ to the engines for pre-conditioning. The pneumatics for operating the vent valves are obtained from the orbiter, with a plenum added on the HO tank to minimize pressure drop and surge effects. The oxygen and hydrogen tanks are pressurized with gaseous oxygen and hydrogen, respectively, which are supplied from the orbiter (tapped off from the engines). Figure 4.2.3-12 shows the interface between the HO tank and Orbiter.

15 MARCH 1972

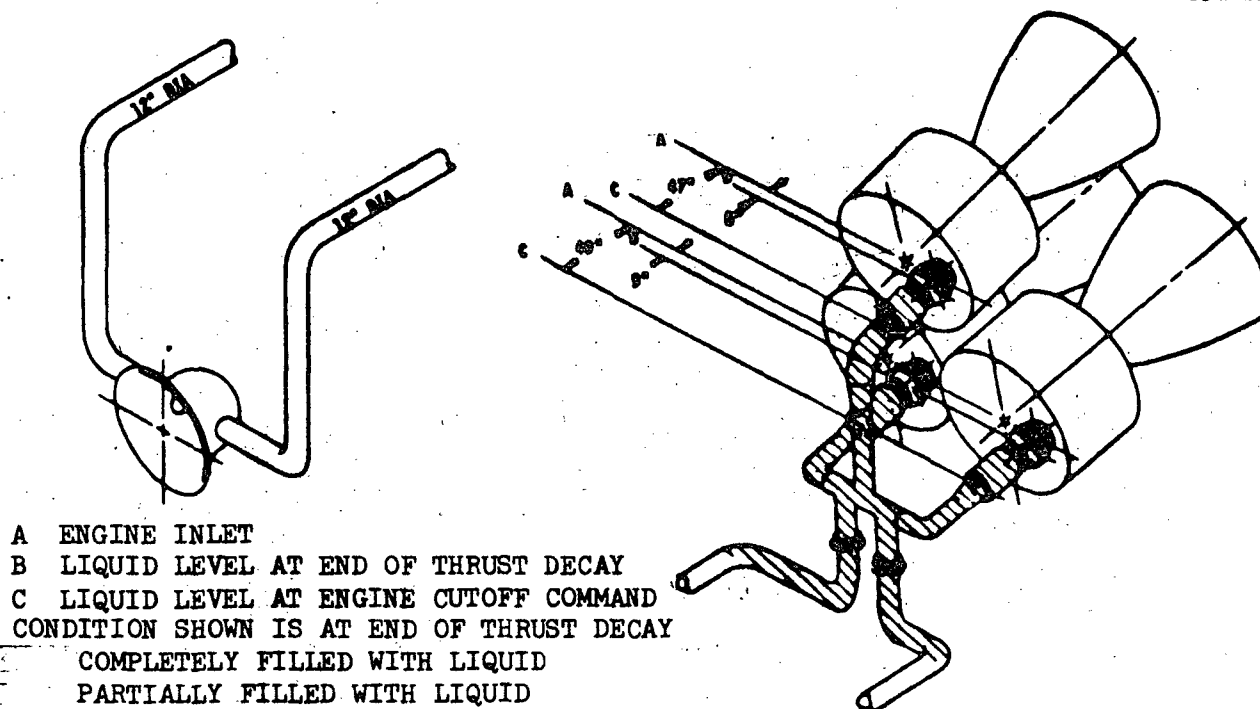


Figure 4.2.3-10 LO₂ Feed System - Series Burn

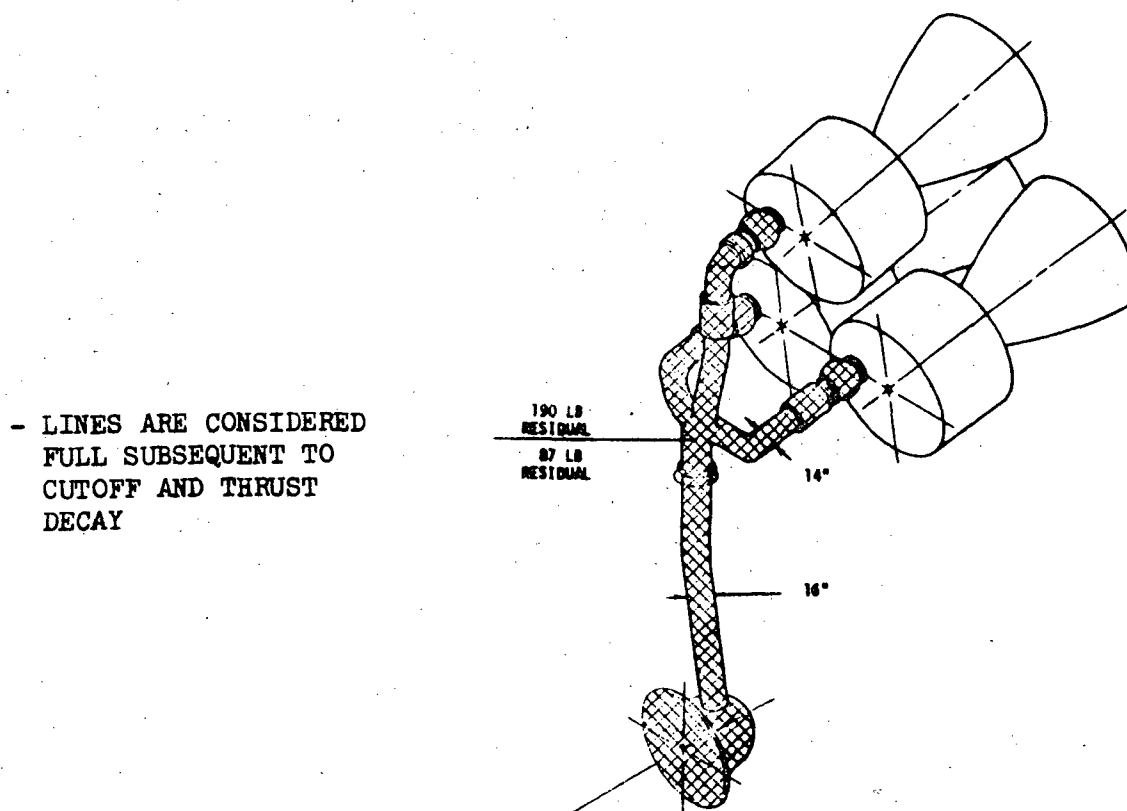


Figure 4.2.3-11 LH₂ Feed System - Series Burn

15 MARCH 1972

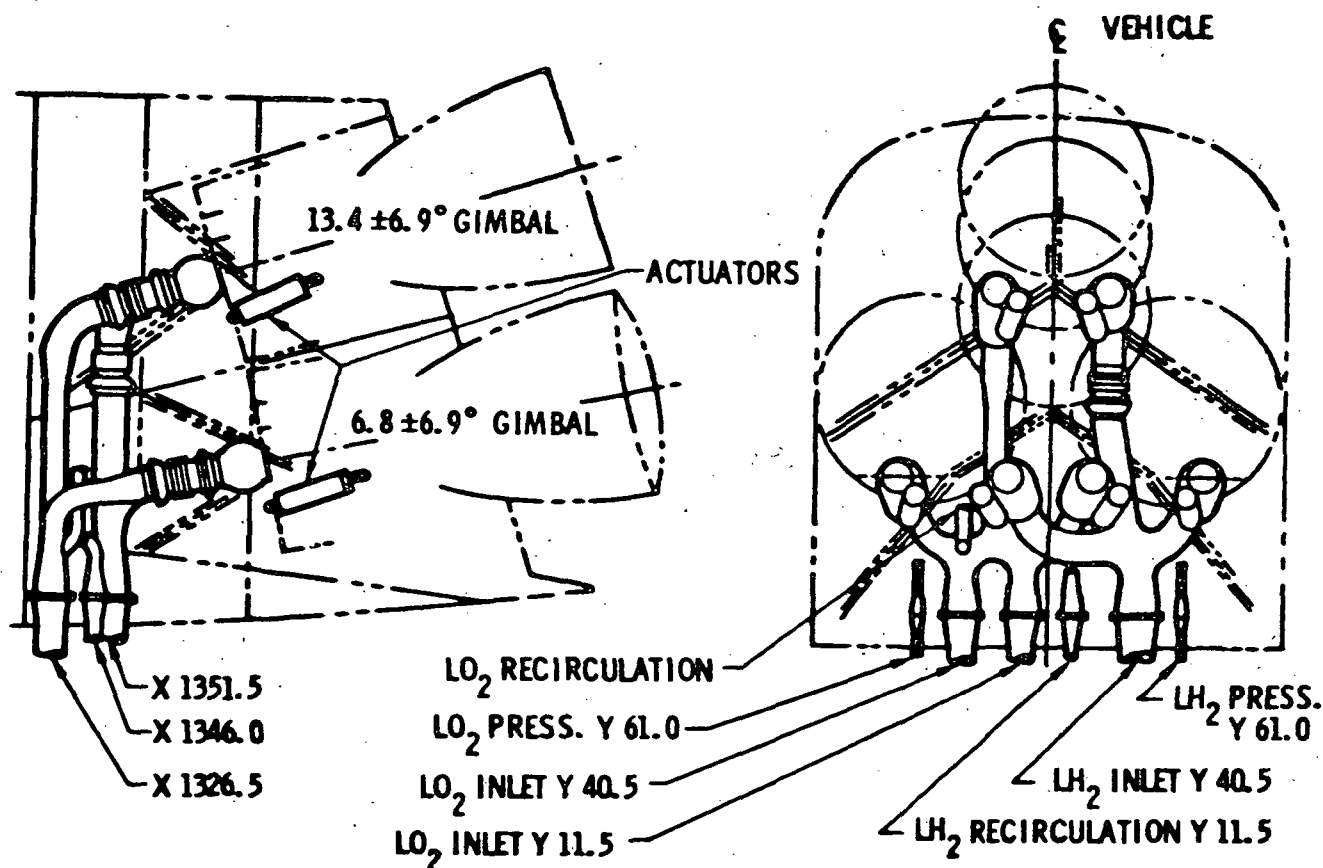


Figure 4.2.3-12 Main Propulsion Interface - Series Burn

Orbiter H₀ Tank Vent Systems -- The LOX and LH₂ vent and relief valves are located on the H₀ tank in the space available just above each tank. Both tanks utilize two normally closed valves in parallel. The vent valves are required for ground venting and possible relieving during boost and burn. The worst case condition for sizing is the ground venting case for the LOX tank. The LOX tank is uninsulated and a fairly high boiloff rate results for this external tank under windy conditions. The LOX boiloff rate was determined from the total heat input rate. The required line and valve size was then determined from Fanno flow analysis, allowing a maximum pressure drop of 0.5 psi from tank to ambient to maintain a low tank bulk saturation pressure. The resulting required vent size was 11-inches in diameter. The LH₂ tank sizing was less critical, allowing a smaller vent size. However, it is cost effective to have a common vent size. Therefore, the LH₂ vent size was also selected to be 11 inches in diameter.

15 MARCH 1972

Propellant Utilization System -- The propellant utilization system selected for the series burn configuration is a closed loop capacitance probe coupled to the orbiter computer which controls the main engine mixture ratio. The propellant in the tank is measured with a cylindrical capacitor, utilizing the liquid propellant as the dielectric. The probe selected is of constant diameter, giving a linear output with liquid level. Software is employed to reflect the tank geometry with respect to propellant mass remaining. The output is then used to obtain a null seeking error signal to control the engine mixture ratio. Cost trades have continued to show this system to be cost effective. The cost advantage of the closed loop system is \$52M (based upon 1.3M lb ascent load) over an open loop system.

Propellant Inventory -- Table 4.2.3-4 shows the propellant and residual inventory. There are five types of LOX and LH₂ unusable residuals. These consist of pressurant, tank undrainable, feedline, chilldown, and engine residual. For a nominal cutoff (velocity cutoff) there is also a PU bias and flight performance reserve (FPR) residual, both of which are usable propellants and could be burned in an off-nominal flight.

Table 4.2.3-4 Propellant Inventory

	PROPELLANT WEIGHT - LB			
	TOTAL		TANK RESIDUALS	
	LOX	LH ₂	LOX	LH ₂
ASCENT	807,514	134,586		
START ADJUSTMENT	1,014	372		
SHUTDOWN ADJUSTMENT	150	87		
FEEDLINE RESIDUAL	1,522	277	584	87
CHILLDOWN RESIDUAL	273	23	0	4
ENGINE RESIDUAL	1,072	75	0	0
PU BIAS	0	336	0	336
TANK UNDRAINABLE	300	100	300	100
PRESSURANT	1,068*	478*	1,178**	596**
NOMINAL LOAD	812,913	136,334		
LOADING ALLOWANCE	8,129	1,363		
MAXIMUM LOAD	821,042	137,697		
PROPELLANT BELOW TANK	-13,852	- 375		
MAXIMUM LOAD IN TANK	808,190	137,322		
	VOLUME FT ³			
PROPELLANT VOLUME IN TANK	11,383	31,245		
TANK VOLUME (2% MINIMUM ULLAGE)	11,611	31,870		

** INCLUDES PREPRESSURIZATION GASES
LOX TANK: 110 LB
LH₂ TANK: 118 LB

* INCLUDES ONLY PRESSURANT FROM LIQUID

The above propellants and the additional start and shutdown propellants which are required due to lower specific impulse during the transients must be loaded in addition to the ascent propellants.

After accounting for all the propellants required, a one percent overload contingency is added to provide for the maximum load case (1% loading accuracy). The propellants below the tanks are subtracted. The tanks are then sized to provide a minimum 2% ullage volume for the maximum load case. The maximum load case is the impulsive requirement associated with the minimum single-engine I_{sp} .

Deorbit Propulsion -- The SRM shown on Figure 4.2.3-13 provides the thrust necessary to deorbit the HO tank. As shown, the SRM is attached to the forward LOX dome via a thrust structure, and is protected during ascent by the nose cone fairing. Pertinent characteristics of the SRM and thrust structure weights are also shown on this figure.

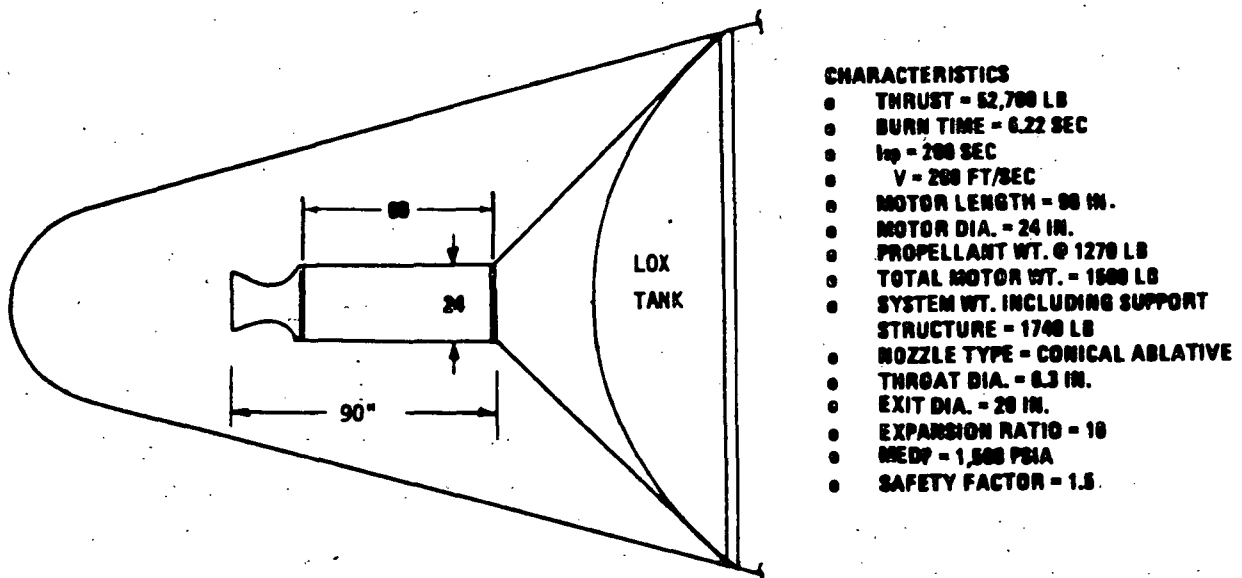
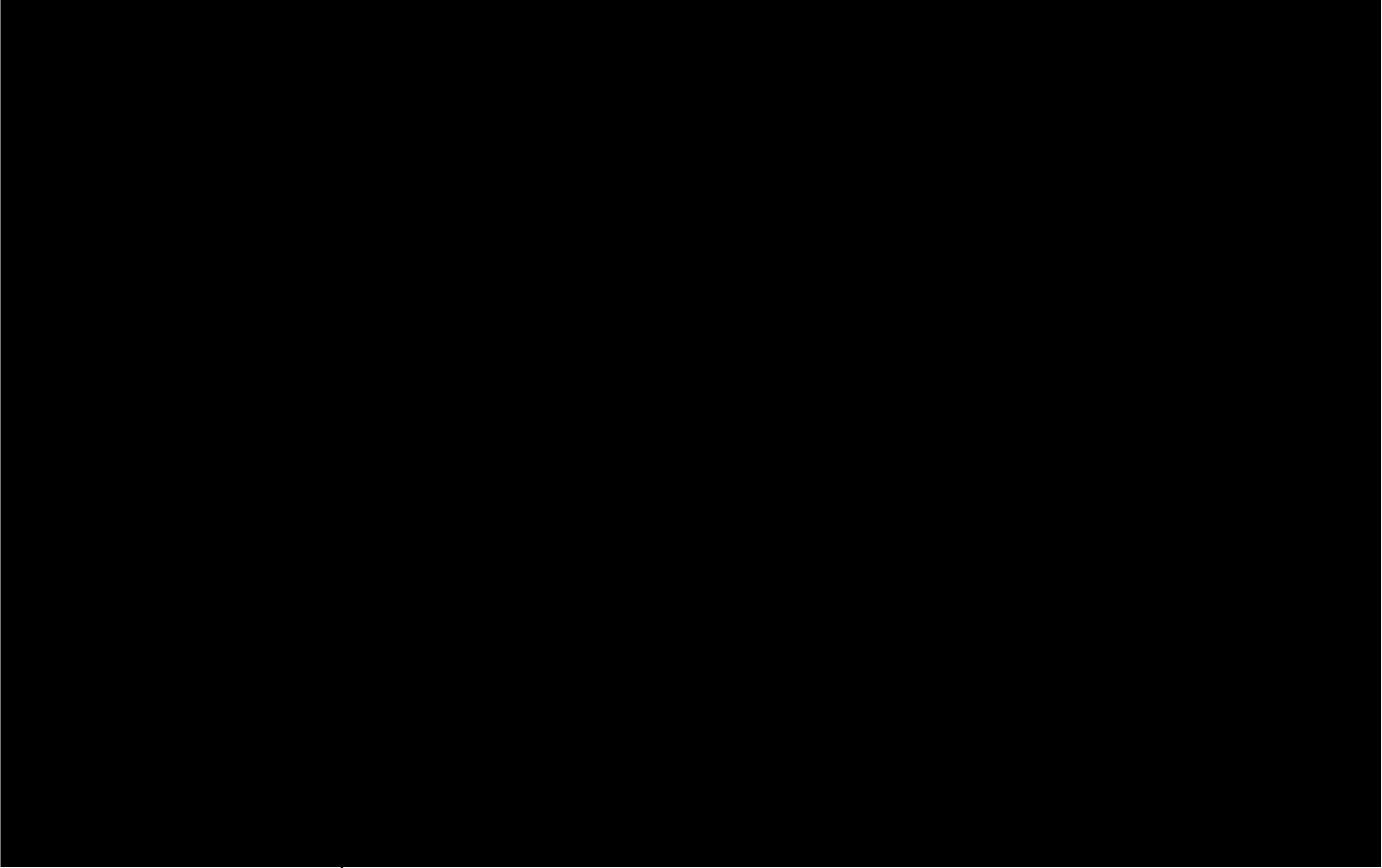


Figure 4.2.3-13 Range Deorbit System - Series Burn

15 MARCH 1972

4.2.3.4 Avionics

The minimum expendable avionics for the HO tank consists of: (1) propellant utilization probes and capacitance-to-frequency converters for propellant loading and residual control; (2) time delays and control circuitry for retrorocket ignition; (3) power distribution, external/internal switching, and dual 28 vdc batteries; (4) signal conditioning/multiplexing of HO tank data for the orbiter telemetry system; and (5) range safety command/destruct system. Tank vent controls will be hardwired from the orbiter. All other equipment is an integral part of the orbiter avionics and as such is reusable. Figure 4.2.3-14 is a block diagram of the avionics system.



Propellant Utilization System -- The orbiter propellant utilization (PU) system is designed to accomplish the following. First, the LOX and LH₂ tanks must be loaded to an accuracy of 0.75 percent. Provision for detection of overfill as a safety backup is provided. The loading function includes constant monitoring of boiloff during ground hold and control of propellant topping during this time. During flight, propellant usage is monitored and controlled to assure simultaneous depletion within 0.25% of total propellant load. A second inflight capability of this subsystem is to provide

15 MARCH 1972

a variable engine mixture ratio for programmed thrust/ I_{sp} profiles. At the bottom of each tank, separate sensors are located to provide backup depletion signals. These are used to shutdown engines in case of malfunction of the primary mass detector.

The propellant in each tank is measured with a cylindrical capacitor utilizing the liquid propellant as the dielectric. The probe is constant diameter giving a linear output with liquid level. This output is also proportional to density of the liquid. This capacitance is brought out of the tank with a cryogenic feed-through and is input to an oscillator. The period of oscillation is proportional to the capacitance and therefore to the liquid level height. The signal is then sent to the orbiter in the form of frequency. The overfill and depletion sensors are of the capacitance point level type. Three sensors are mounted at the same height and are fed to a control box that votes 2 out of 3 and provides a bilevel signal to be sent with the orbiter flight control computer.

A frequency-to-digital-word converter prepares the capacitance probe signals for use in the flight control computer. The computer is programmed to apply volume versus height curves to the probe signals. This converts the signals to mass remaining in the tank. The slosh dynamics is then filtered out, resulting in the corrected mass remaining. This is scaled and summed together to obtain a null seeking error signal, which is converted from digital to analog and sent to the valve motor servo amplifier. The valve motor servo amplifier drives all four valves in parallel, adjusting the engine mixture ratio. A block diagram of the PU system is shown in Figure 4.2.3-15.

Deorbit Controls -- After the orbiter/tank has achieved the desired velocity and depleted the fuel and oxidizer in the H0 tank, the orbiter positions the H0 tank at the desired attitude and initiates separation. This is done in such a manner as to minimize any rotational acceleration. After a short time delay, which permits the orbiter to move away from the tank, the H0 tank nose cone is separated and the retrorocket fired. This causes the tank to deorbit and reenter the atmosphere.

15 MARCH 1972

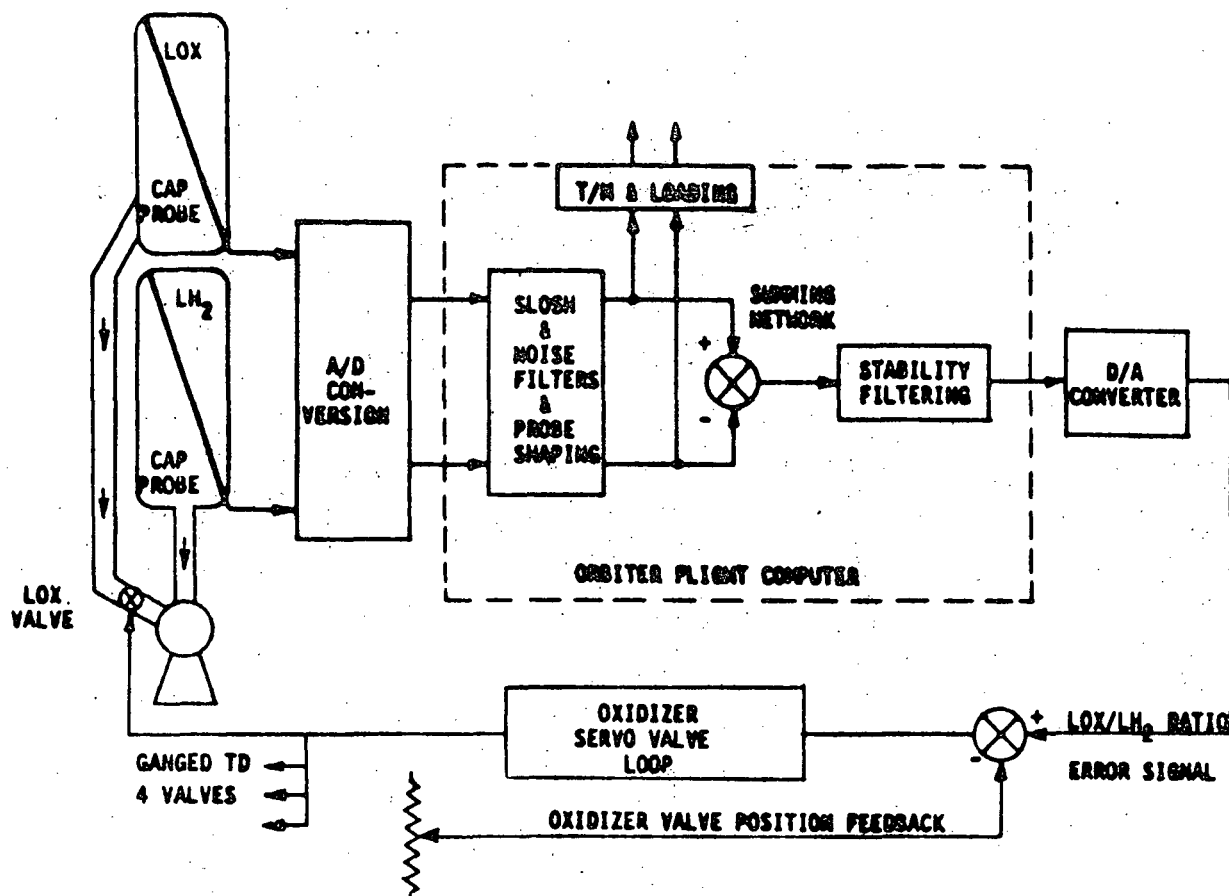
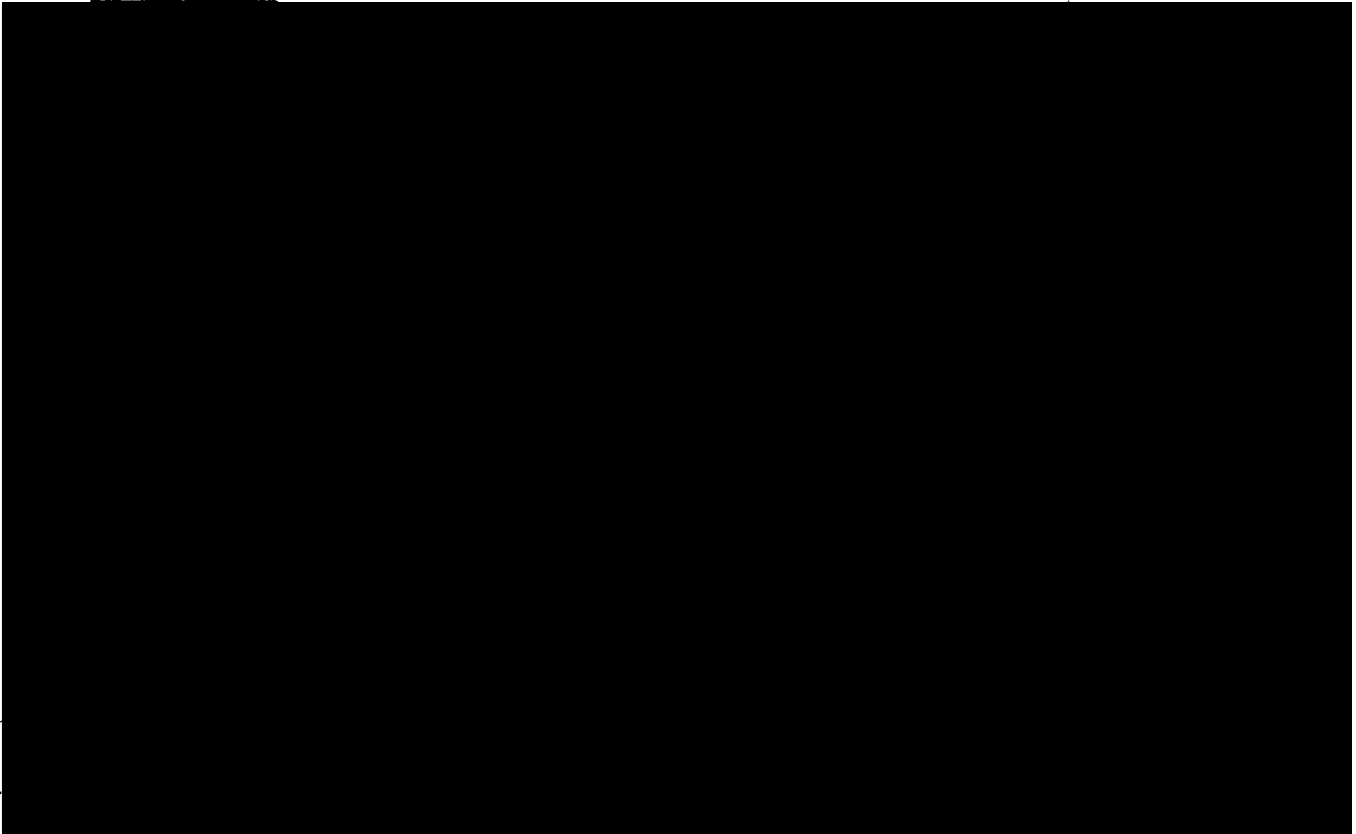


Figure 4.2.3-15 Propellant Utilization System Block Diagram

Electrical Power System -- The electrical power system consists of dual silver-zinc batteries and a simple distribution system. The primary purpose of redundancy is to insure operation of the range safety system and dispersion of the propellant in the event of an abort.

Instrumentation System -- Critical functions in the HO tank are sensed and hardwired to the orbiter caution and warning system. Less critical parameters are multiplexed and monitored in the orbiter via the orbiter computers. The orbiter telemetry system is used for ground monitoring.

Range Safety -- A range safety command/destroy system is provided for dispersal of the fuel and oxidizer in case of an abort. This system is a standard GFE receiver/decoder system and is dual redundant. Prior to vehicle first motion, the S&A device is remotely switched to the armed position. The destruct system is made inoperative in flight by disabling the secure command receiver command circuits at a time when the vehicle reaches a point in its trajectory where the vehicle is no longer dangerous to ground installations. Figure 4.2.3-16 shows a block diagram of this system.



Equipment Installations -- Figure 4.2.1-2 illustrates the PU probe installations and the location of the avionic equipment in the forward portion of the vehicle. Assembly and installation of the PU probes is accomplished inside the tanks. Probe wiring terminates in capacitance-to-frequency converters as close to the tank bulkhead feedthrough as possible to minimize cable capacitance effects.

Other avionics equipments are grouped together near the retrorocket in the nose of the vehicle. Orbiter interface wiring is routed through a tunnel aft to the orbiter/tank interface. It should be noted that the booster/orbiter interface wiring must also pass through the H0 tank.

5.0 Orbiter H0 Tank Parallel Burn Two 156" SRM

An H0 tank is expended every orbiter flight. This requires that the configuration design and design details be considered conscientiously in order to minimize cost, in light of the 453 units to be constructed. However, the innovations incorporated to reduce cost must not result in undue weight penalty (reduction in mass fraction). Unique designs, processes and manufacturing methods must be exploited to achieve the desired results in the finely balanced trade studies between weight, cost and schedule. One of the major difficulties facing the tank designer will be to "Keep it simple" and to prevent the H0 tank from becoming the equivalent of another stage.

5.1 Phase B Extension Final Design

The impact of the final sizing to a propellant weight of 1,577,900 pounds is shown in Figure 5.1-1. The only significant changes from the analysis baseline, to be discussed later, are the tank dimensions shown and an increase in tank inert weight to 68,625 pounds and a change in mass fraction to 0.9583.

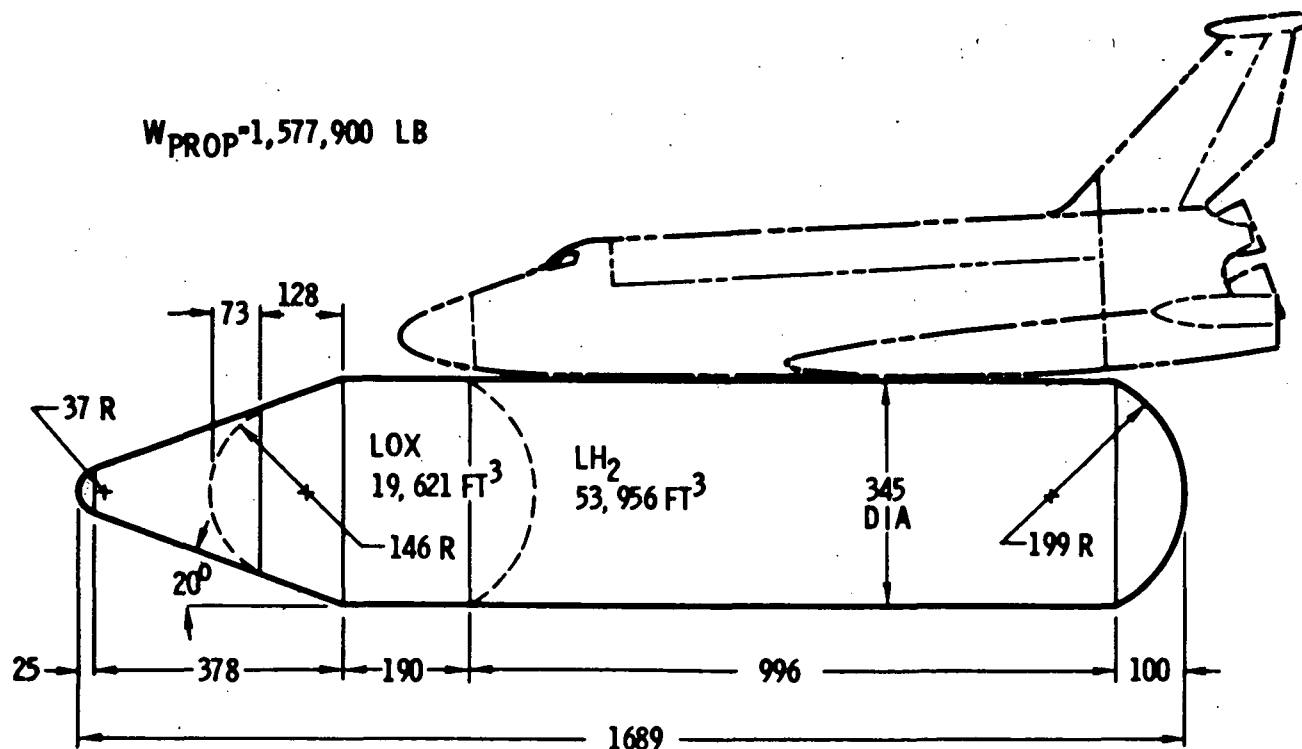


Figure 5.1-1 Tank Geometry - Parallel Burn

15 MARCH 1972

5.2 Analysis Baseline

The orbiter HO tank (propellant weight of 1,530,800 pounds) has been analyzed and designed for the parallel burn-two 156" SRM configuration. The ascent configuration for the vehicle is shown in Figure 5.2-1.

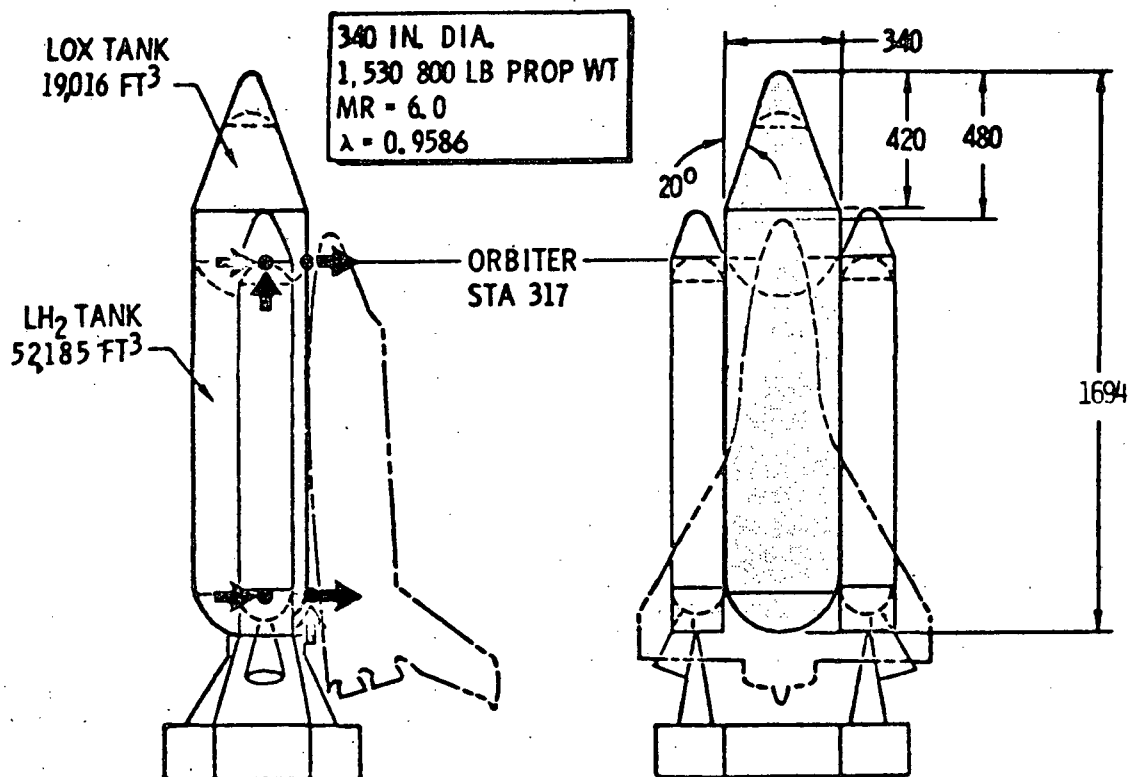


Figure 5.2-1 Orbiter HO Tank Parallel Burn-Solid (Two 156") Booster

5.2.1 Configuration Definition

The analysis baseline used for the HO tanks is shown on Figure 5.2.1-1. An inboard profile of this tank is shown in Figure 5.2.1-2. All of the major subsystems are shown in the figure, while significant configuration characteristics relating to it are shown in Table 5.2.1-1. Mass properties are shown on Table 5.2.1-2. This table shows two weight statements: the preliminary parametric weights, which are consistent with the mass properties status report, and the completed analysis weights which represent a later iteration in design and analysis. Included in the weights is a contingency allowance of 5% for the body group, 10% for the induced environmental protection, and a 2% growth and uncertainty allowance on the total dry weight.

15 MARCH 1972

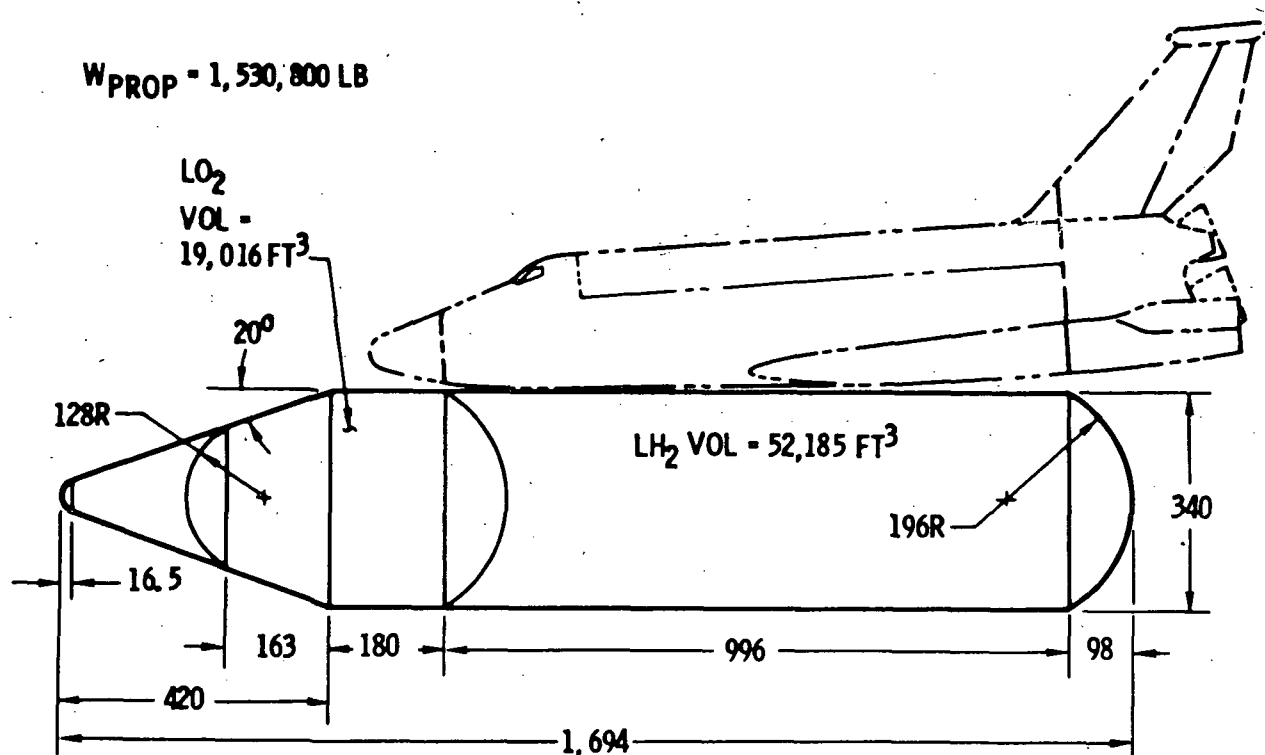


Figure 5.2.1-1 Tank Geometry - Parallel Burn

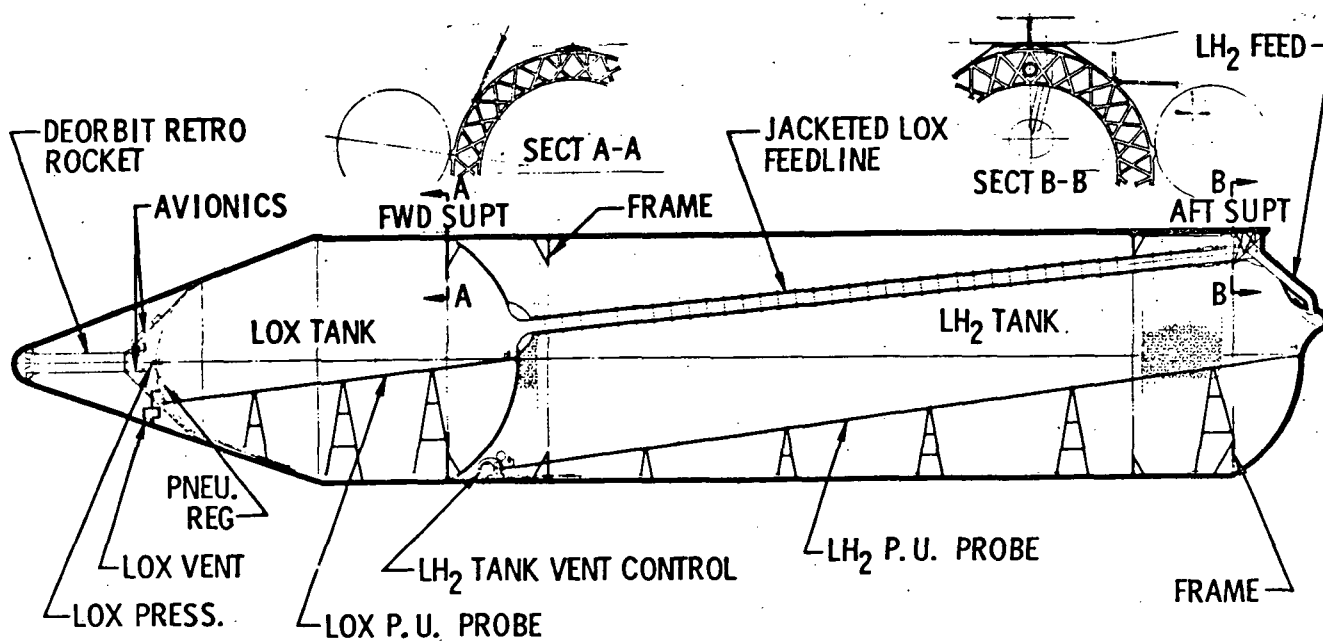


Figure 5.2.1-2 Inboard Profile - Parallel Burn

15 MARCH 1972

Table 5.2.1-1 Configuration Characteristics - Parallel Burn

GROSS WEIGHT	1,596,911 LB
USABLE PROPELLANT WEIGHT (INCLUDES FPR)	1,530,800 LB
MIXTURE RATIO	6.0:1
PROPELLANT MASS FRACTION χ	0.9586
PROPELLANT TANKS	
• DIAMETER	340 INCHES
• LENGTH	1,694 INCHES (L/D - 5.0)
• MATERIAL	2219 ALUMINUM
• PROPELLANT LOCATION	LOX FORWARD
• BULKHEADS	COMMON BULKHEAD
• BOOSTER/ORBITER ATTACH	FORWARD BOOSTER ATTACH/ORBITER AFT
FEED LINES	LOX - 1 X 17 INCHES DIAMETER
	LH ₂ - 1 X 16 INCHES DIAMETER
VENTS	TANK LOCATED
PROPELLANT UTILIZATION	CLOSED LOOP
INSULATION	EXTERNAL FOAM WITH FIREX - 250 ABLATOR
DEORBIT	SINGLE SRM IN NOSE CONE

Table 5.2.1-2 Weight Summary - Parallel Burn

	PRELIMINARY PARAMETRIC WTS.	COMPLETED ANALYSIS WTS.		PRELIMINARY PARAMETRIC WTS.	COMPLETED ANALYSIS WTS.
BODY GROUP	[41,760]	[49,030]	ELECT/INSTRUMENTATION	[420]	[274]
FUEL TANK	(19,560)	(22,603)	BATTERIES		20
AFT BULKHEAD	2,010	3,559	POWER DISTRIBUTION		20
CYLINDRICAL SIDEWALL	17,550	19,044	EBW FIRING UNITS		29
OXIDIZER TANK	(7,120)	(8,157)	PU ELECTRONICS		10
FWD BULKHEAD	460	337	RANGE SAFETY		30
SIDEWALL	6,320	7,480	INSTRUMENTATION		17
BAFFLES	340	340	COAX AND WIRING		148
COMMON BULKHEAD	(3,930)	(4,340)	DEORBIT SYSTEM	[3,630]	[2,170]
BOOSTER/ORB/TANK ATTACH	(9,500)	(12,966)	GROWTH/UNCERTAINTY (3,42)	[2,047](27)	(1,265)
NOSE FAIRING	(650)	(464)	DRY WEIGHT	62,377	64,550
UNBILICAL PANEL	(300)	(300)	RESIDUAL PROPELLANTS	[3,733]	[3,854]
TUNNEL	(200)	(200)	TANK UNDRAINABLES	100	100
INDUCED ENVIRONMENT PROTECT.	[9,650]	[6,546]	FEED LINE TRAPPED	191	317
ABLATOR	8,400	5,344	PRESSURANT	2,895	2,890
FOAM	650	612	PU BIAS	547	547
PRIMER, PAINT AND SEALER	600	590	INERT WEIGHT	66,110	68,404
PROPELLANT SYSTEMS	[5,370]	(5,265)	USABLE PROPELLANT	[1,530,800]	[1,530,800]
FEED SYSTEM	3,500	3,438	TOTAL GROSS WEIGHT	1,596,911	1,599,204
PRESS. AND VENT	1,400	1,366	$\chi = \frac{W_{PROP}}{W_{GROSS}} =$	0.9586	0.9572
SUAPS & VORTEX CONTROL	200	220			
PU SYSTEMS	190	185			
PNEUMATICS	60	56			

5.2.2 Vehicle Analysis

The structural loads on the vehicles were calculated for a range of conditions and the most critical loads were determined for use in tank design. Figures 5.2.2-1 and 5.2.2-2 depict the maximum loads imposed on the tank structure from both ground and flight conditions.

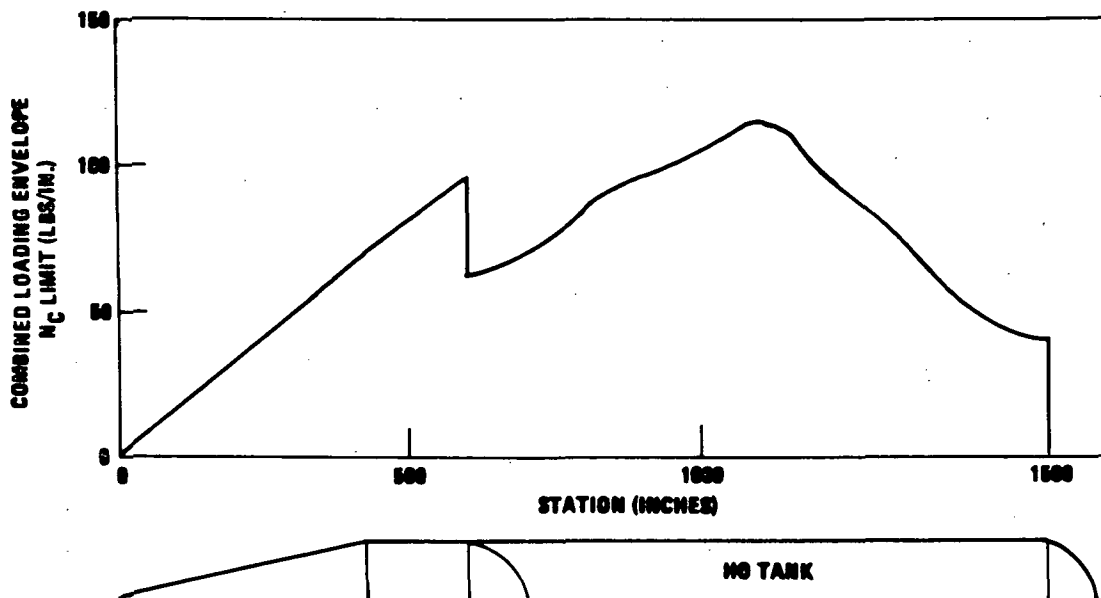


Figure 5.2.2-1 Ground Wind Loading Envelope - Parallel Burn

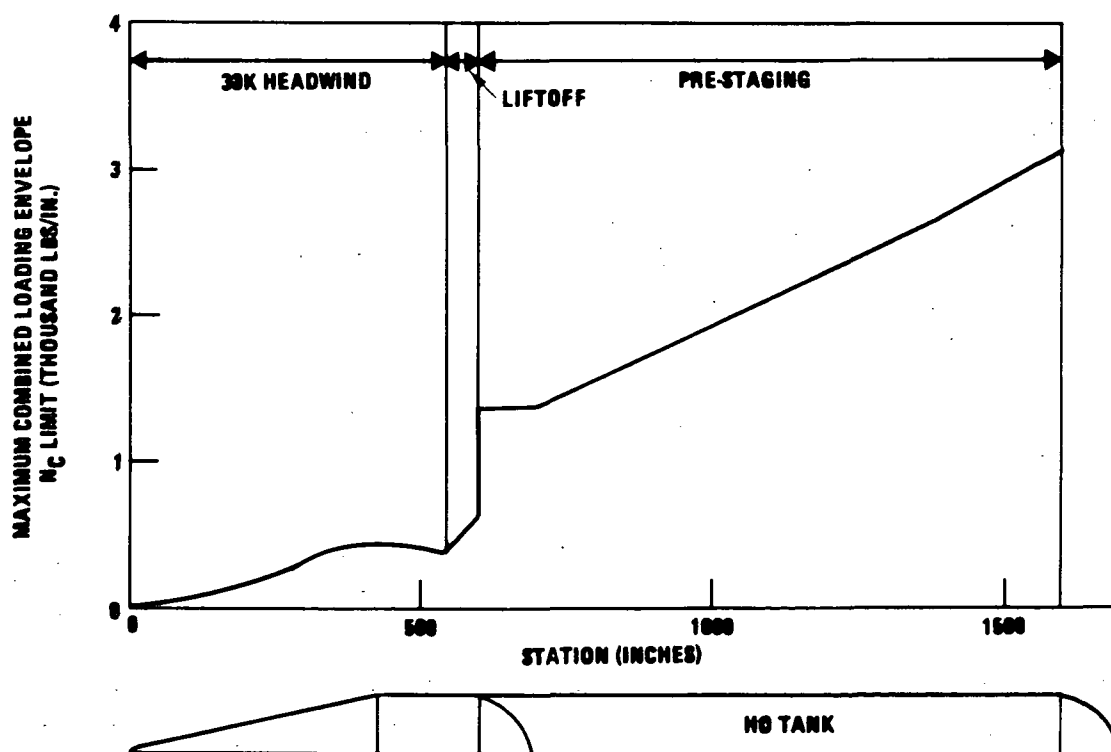


Figure 5.2.2-2 Maximum Ascent Loading Condition - Parallel Burn

15 MARCH 1972

Ground Wind Loads -- The maximum ground wind load condition on the H0 tank was for tanks fully loaded and a one day exposure wind. The wind was applied in the direction tending to separate the orbiter from the H0 tank. The height above ground to the base was assumed to be 97 feet. A drag coefficient of 1.2 and a design factor of 1.5 were used in the computation of airloads. The maximum combined axial and moment compression loads with unpressurized tanks is shown in Figure 5.2.2-1. Combined loading is quite small in compression, since the H0 tank is supported in the axial direction at the forward end.

Ascent Loads -- Ten load conditions were analyzed on the H0 tank. These are shown in Table 5.2.2-1. The maximum combined axial and moment compression loads with unpressurized tanks is shown in Figure 5.2.2-2 for three load conditions. These may not be the design loads, however, when internal pressure, ambient pressure, temperature, and dynamic amplification factors are considered in the sizing program.

Table 5.2.2-1
LOAD CONDITION SUMMARY

LOAD CONDITION	TIME (SEC)	MACH NO.	q (PSF)	α (B) (DEG)	Nx	Ny	Nz
SRM's TIED DOWN; ORBITER THRUSTING	0	0	0	-	1.0	0	0
LIFTOFF	0	0	0	-	1.0	0	-.308
LIFTOFF	0	0	0	-	1.0	0	.308
25K TAILWIND	57	.95	508	-4.72	1.76	0	.322
30K HEADWIND	62	1.28	751	11.04	1.73	0	-.254
30K SIDEWIND	62	1.18	640	$\alpha = 7.00$ (B = -7.03)	1.72	-.181	-.024
MAX ACCELERATION	124.9	4.10	84	6.56	2.58	0	.230
PRE-STAGING	124.9	4.10	84	6.56	.78	0	.234
POST-STAGING	125	4.10	84	6.56	.93	0	.262
ORBITER BURNOUT	500	-	0	0	3.00	0	.547

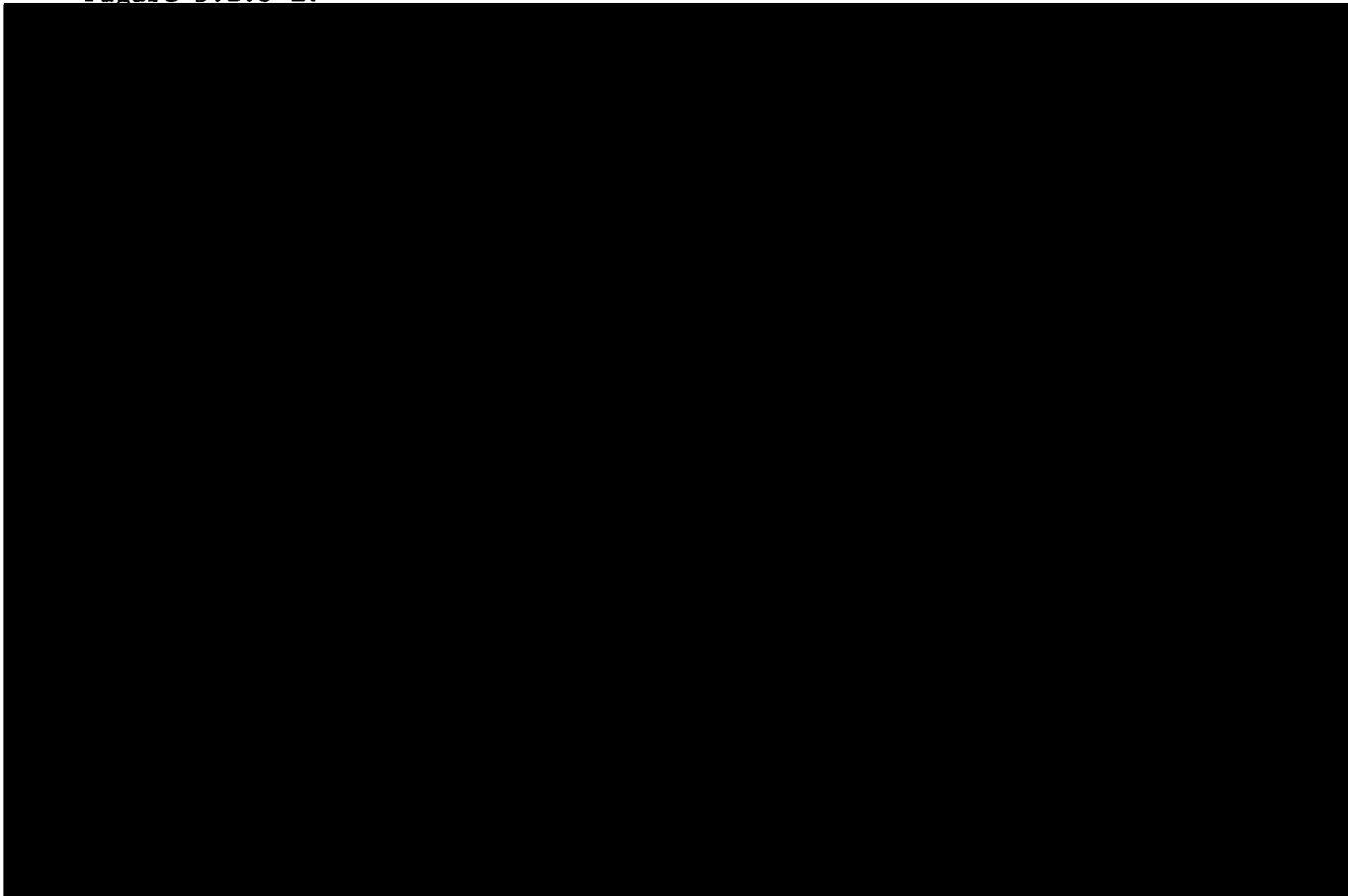
15 MARCH 1972

5.2.3 Vehicle Design

This section contains the description of the design tasks performed in structural/thermal protection, propulsion, and avionics.

5.2.3.1 Structural Design

The five basic structural systems of this orbiter H0 tank are a nose fairing, LOX tank, common bulkhead, LH₂ tank, and attach structure, as shown on Figure 5.2.3-1.



Structure Description -- The nose fairing is similar to that on the series burn vehicle except for its shape. The LOX tank is made up of a forward dome, a cone and cylinder, consisting of 8 staves each, and the common bulkhead. The forward dome, cone, and cylinder are welded monocoque structure, and each is welded to adjacent components through Y-rings. The staves in the cone and cylinder are machine tapered skins surrounded by constant thickness weld lands. The forward dome gores are constant thickness machined skins, bordered by constant thickness weld lands. The common bulkhead consists of eight integrally machined waffle (isogrid) gores, welded to a polar cap of

15 MARCH 1972

similar construction. The dome assembly is welded to the cylinder through a Y-ring as shown. The LOX tank is designed to withstand 30 psi ullage pressure and LOX hydrostatic head. Lift-off and 3 g flight conditions were considered in determining the maximum head pressure at various stations in the tank. The wall temperature used is -298°F in the liquid area and $+45^{\circ}\text{F}$ in the ullage area. The all welded monocoque construction requires the staves and gores to be machined from plate, to provide a skin sized to resist pressure loading, with an integral weld land around the periphery. The basic skin may or may not be tapered, but the weld lands are of constant thickness. Reduced material allowables due to welding require the land to be at least twice the maximum basic skin thickness. The forward dome is designed for ullage pressure, which resulted in a constant nominal skin thickness of 0.048. This gage is thin enough to require chem milling after machining, which dictates a tolerance of $\pm .008$. The conical section of the LOX tank tapers from a nominal size of .080 forward to .139 aft. The forward plane resists ullage pressure only, while the aft plane is sized for ullage and head pressure. Chem milling is not required; so tolerances of $\pm .005$ were used to derive nominal sizes. The cylindrical section of the tank tapers from a nominal size of 0.131 inches forward to 0.176 inches aft. The entire cylinder is sized for ullage and head pressure. Tolerances of $\pm .005$ were used. The common bulkhead is designed to resist both burst caused by LOX ullage and head pressure and a reverse buckling condition caused by LH_2 ullage pressure combined with a loss of pressure in the LOX tank, a negative $\Delta p = 40$ psig.

The LH_2 tank consists of a cylinder, aft dome, and the common bulkhead described above. The cylinder has a long monocoque center section, made up of 8 welded machine-tapered staves with constant thickness weld lands, and two short end barrels, each consisting of 3 welded isogrid staves. The aft dome consists of 6 gores welded together and to a polar cap. They are constant thickness machined skins with constant thickness weld lands. Each component is welded to adjacent components through machined rings. A computer program is used to determine the most critical of seven failure modes: hoop stress (burst, yield, proof), longitudinal tension (yield,

ultimate), and longitudinal compression (yield, buckling). For buckling, the monocoque structure was analyzed using the method described in NASA SP-8007, "Buckling of Thin-Walled Circular Cylinders." The waffle (isogrid) structure was analyzed using the methods of MDC E0375, Space Shuttle Data, Part III, Booster, Airframe, Appendix B. Figure 5.2.3-2 presents a weight summary of the tank analyses. This curve is the locus of all minimum-weight designs, and includes the effect of a pressure tolerance of ± 1.0 psia. The absolute optimum design is seen to occur at 37 psia. A summary of the critical design conditions is presented in Figure 5.2.3-3. The orbiter burnout condition is critical over most of the tank, because the temperature at this time (500 seconds after liftoff) has increased from -423°F to -66°F , thereby reducing the ultimate tensile stress from 94,500 psi to 66,500 psi. The remaining part of the tank sidewall is buckling critical during the pre-staging condition. The aft dome is critical during the orbiter burnout condition. Nominal dimensions determined by the sizing program are presented in Figure 5.2.3-1.

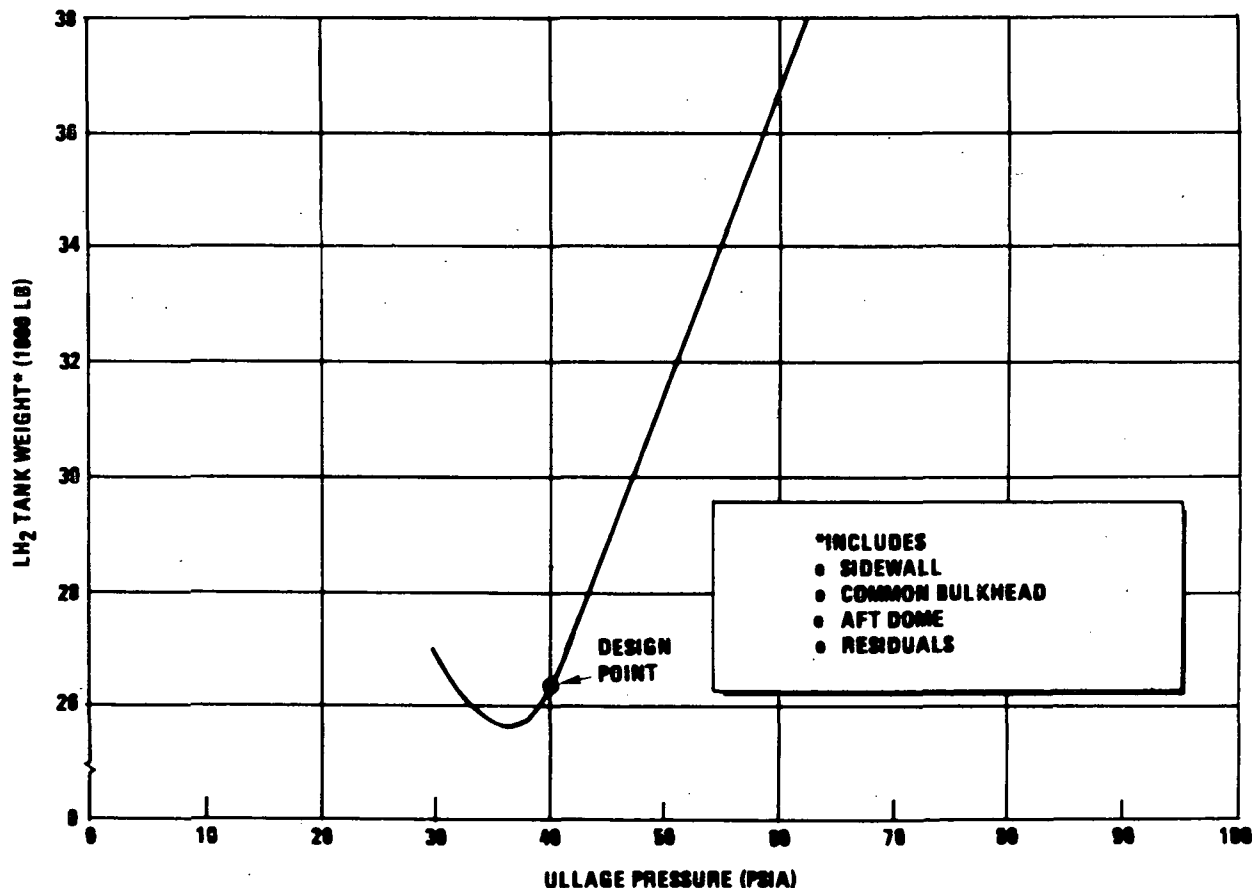
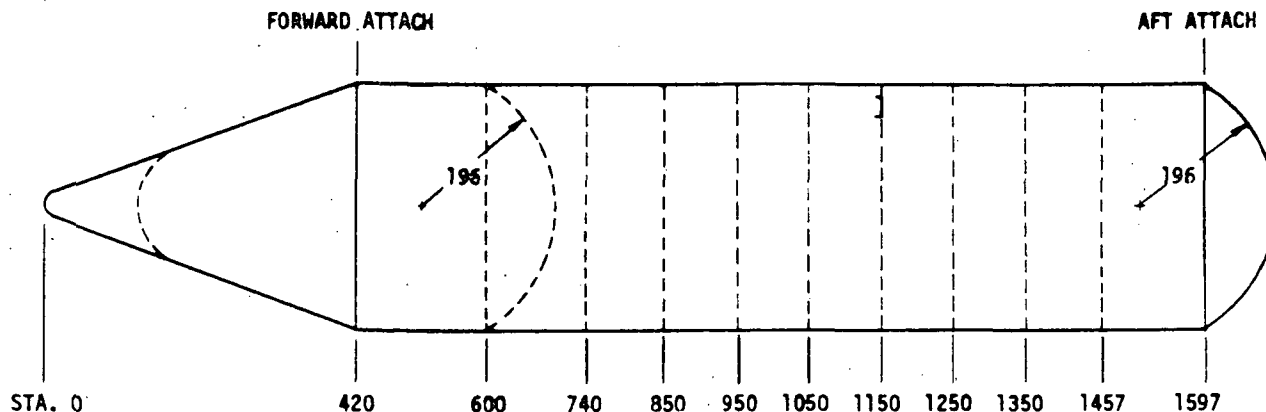


Figure 5.2.3-2 LH₂ Tank Weight Vs. Ullage Pressure - Parallel Burn

15 MARCH 1972



	LOX TANK SIDEWALL	LH ₂ TANK SIDEWALL		
OPERATING PRESSURE (PSIA)	30	40		
BASILINE CONSTRUCTION	MONOCOQUE	WAFFLE	MONOCOQUE	WAFFLE
DOMINATING LOAD CONDITIONS	LIFTOFF & MAX ACCELERATION	ORBITER BURNOUT		PRE- STAG- ING PRE- STAG. & ORB. BURNOUT
MODE OF FAILURE	BURST HOOP STRESS	BURST HOOP STRESS		BUCKLING & BURST HOOP STRESS

Figure 5.2.3-3 Structural Design Conditions - Parallel Burn

The forward and aft attach structure in this design resists both booster and orbiter loads. Forward attach structure consists of external fittings connected to the Y-ring at the common bulkhead, the booster attach doubler which is bolted to the Y-ring and down the length of the forward LH₂ tank waffle barrel, and the two frames at each end of this barrel. The aft attach structure consists of an external fitting which is bolted to the aft waffle barrel and the two frames at each end of the barrel. The inner caps and diagonal truss members of all frames are machined as one part from plate stock and mechanically attached to outer caps which are welded to the cylinder and domes as shown. The most forward and most aft frame caps are also dome Y-rings. Sizing of the HO tank attachment structure between the orbiter and the two SRM boosters was based on the most critical condition experienced. Conditions investigated were liftoff, maximum α q (30K headwind), maximum acceleration, and pre-staging. Results of this investigation for the forward and aft attachments are shown in Tables 5.2.3-1 and 5.2.3-2. The material used for the attach structure is 2219-T87 aluminum, except for the socket fitting, which is 200 grade maraging steel. Added structural weight of the HO tank barrel at the forward booster attachment to distribute longitudinal booster attach loads consists of 1818 pounds. The attach structure is shown in Figure 5.2.1-2.

Table 5.2.3-1 Forward Tank Attach Design

ITEM	DESIGN CONDITION	STRESS (PSI.)	ALLOWABLE (PSI.)	M.S.	WEIGHT (LBS.)	
Sway Brace Tube (2) Fitting (2)	Pre-Staging Pre-Staging	$f_c = 11,050$ $f_{br} = 87,400$	$F_c = 12,000^*$ $F_{br} = 100,000$	.08 .14	$2 \times 168 = 336$ $4 \times 10.2 = 40.8$	377
Side Link	Liftoff	$f_c = 11,200$	$F_c = 11,200^*$	0		41
Frame Outside Cap Inside Cap Diagonals Cross Brace	Liftoff Liftoff Liftoff Liftoff	$f_c = 94,500$ $f_c = 66,800$ $f_c = 35,500$ $f_c = 25,000$	$F_{tu} = 94,500$ $F_{cy} = 66,800$ $F_c = 35,500^*$ $F_c = 25,000^*$	0 0 0 0	585 228 1575 600	2988
Socket Fitting (2)	Max. Acc.	$f_b = 180,000$	$F_b = 252,000$	.40	2×169	338
Doubler (2)	Max. Acc.	$f_t = 62,000$	$F_{tu} = 62,000$	0	2×292	584
*Buckling					TOTAL =	4328

Table 5.2.3-2 Aft Tank Attach Design

ITEM	DESIGN CONDITION	STRESS	ALLOWABLE	M.S.	WEIGHT (LBS.)	
Frame, Sta. 1597 Outer Cap Inner Cap Diagonals	Liftoff Liftoff Liftoff	$f_c = 66,800$ $f_c = 66,800$ $f_c = 28,100$	$F_{cy} = 66,800$ $F_{cy} = 66,800$ $F_c = 28,100^*$	0 0 0	1315 1295 955	3565
Frame, Sta. 1457 Caps Web	Pre-Stage Pre-Stage	$f_c = 66,800$ $f_s = 11,000$	$F_{cy} = 66,800$ $F_s = 11,000^*$	0 0	415 1000	1415
Drag Load Fitting Web Flange Tank Flange	Pre-Stage Pre-Stage Pre-Stage	$f_s = 38,000$ $f_c = 50,000$ $f_c = 50,000$	$F_s = 38,000$ $F_{cy} = 50,000$ $F_{cy} = 50,000$	0 0 0	57 364 17	438
Skin Doubler	Pre-Stage	$f_c = 50,000$	$F_{cy} = 50,000$	0		300
Orb Load Links	Liftoff	$f_c = 50,000$	$F_c = 50,000^*$	0	2×35	70
Side Load Straps	Liftoff	$f_t = 63,000$	$F_{tu} = 63,000$	0	2×16	32
Booster Side Load Fitting					2×25	50
Booster Load Links	Liftoff	$f_t = 63,000$	$F_{tu} = 63,000$	0	4×160.5	642
					TOTAL =	6512

* Buckling

Notes: (1) Booster load links not included with HO tank weight
 (2) Booster side load fitting not analyzed

15 MARCH 1972

Fracture Control -- The summary write up on fracture control in Paragraph 4.2.3.1 is equally applicable to the parallel burn configuration.

5.2.3.2 Thermal Protection System

The baseline external cryogenic thermal protection system is shown in Figure 5.2.3-4. The features of this insulation system are described in Paragraph 4.2.3.2.

MAXIMUM



Thermal Environment - Figure 5.2.3-5 shows the aerodynamic heating used to size the H0 tank insulation system for the parallel burn - solid (two 156") configuration. The insulation thickness distribution is shown in Figure 5.2.3-4 for the 200°F case. Ablator thicknesses required to protect the foam on the base from the radiation from the booster plumes are shown in Figure 5.2.3-6. As base heating rates are much higher than the aero heating rates shown in Figure 5.2.3-5, the temperature gradients through the ablator are much higher.

15 MARCH 1972

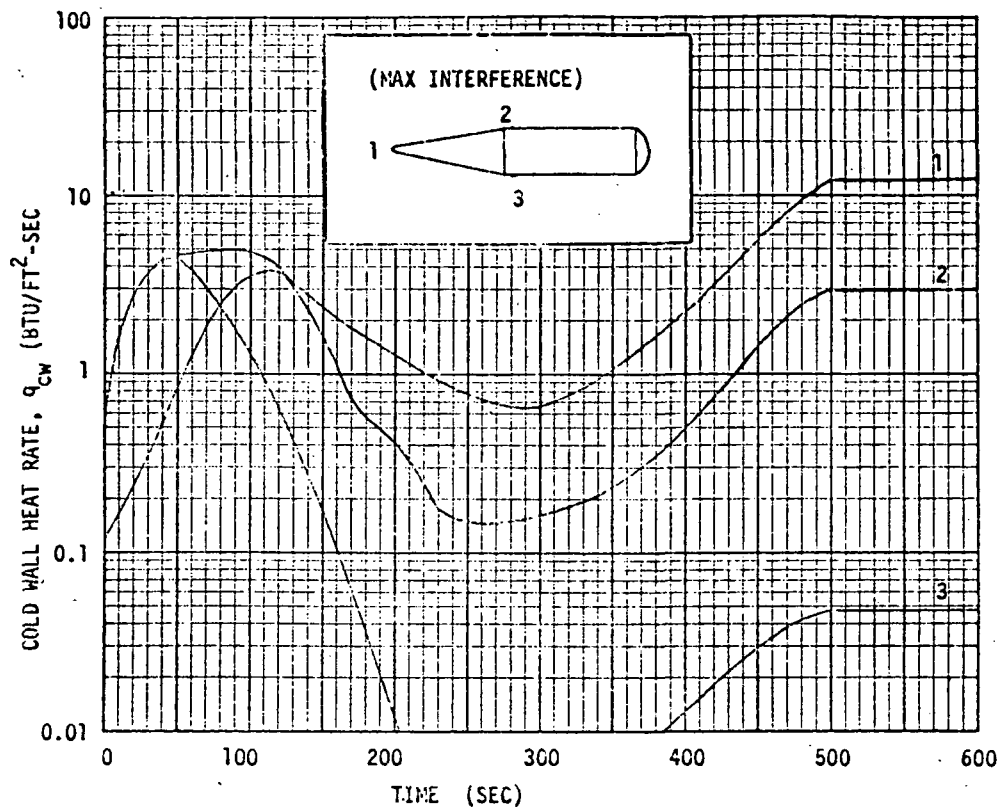


Figure 5.2.3-5 Aero Heating Environment - Parallel Burn

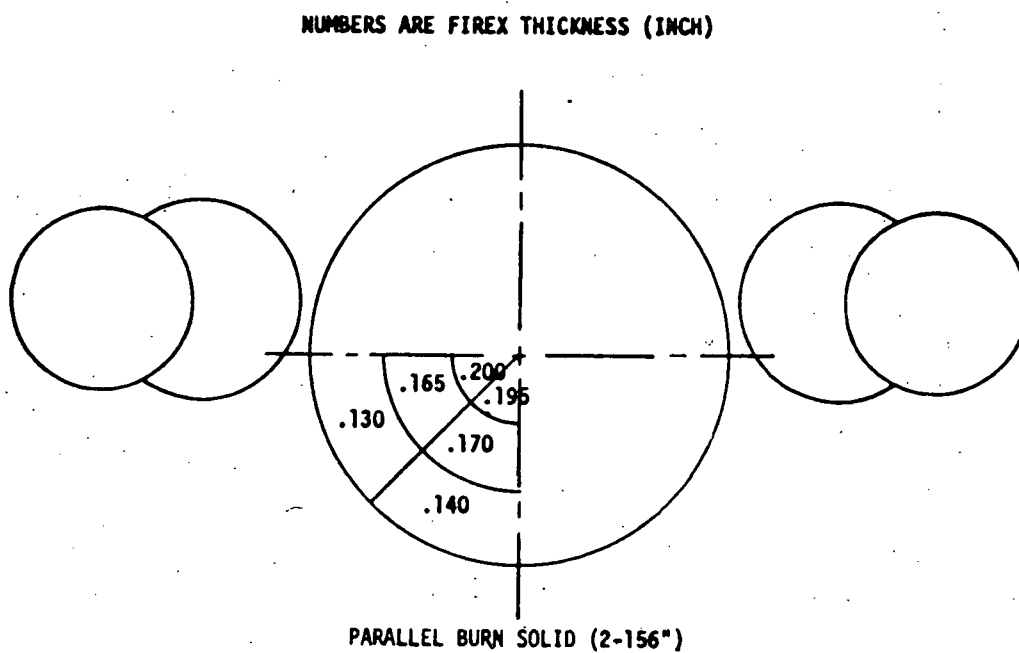


Figure 5.2.3-6 End Dome Ablator Thickness

TPS Materials -- Paragraph 4.2.3.2, including Figures 4.2.3-6, 4.2.3-7, and 4.2.3-3, is equally applicable to the parallel burn configuration.

5.2.3.3 Propulsion Subsystem

The parallel burn HO tank propulsion system consists of the LOX and LH₂ feed systems, LOX and LH₂ vent systems, LOX and LH₂ pressurization systems, pneumatic system, and sump and vortex control systems.

This configuration, which must support an orbiter start under sea level conditions, employs a single LOX feed line penetrating the hydrogen tank. The LH₂ feed system also employs a single feed line from the tank to the orbiter. Geyser control, engine preconditioning and feed system preconditioning are affected by ground overboard bleed. A schematic diagram of this system is shown on Figure 5.2.3-7.

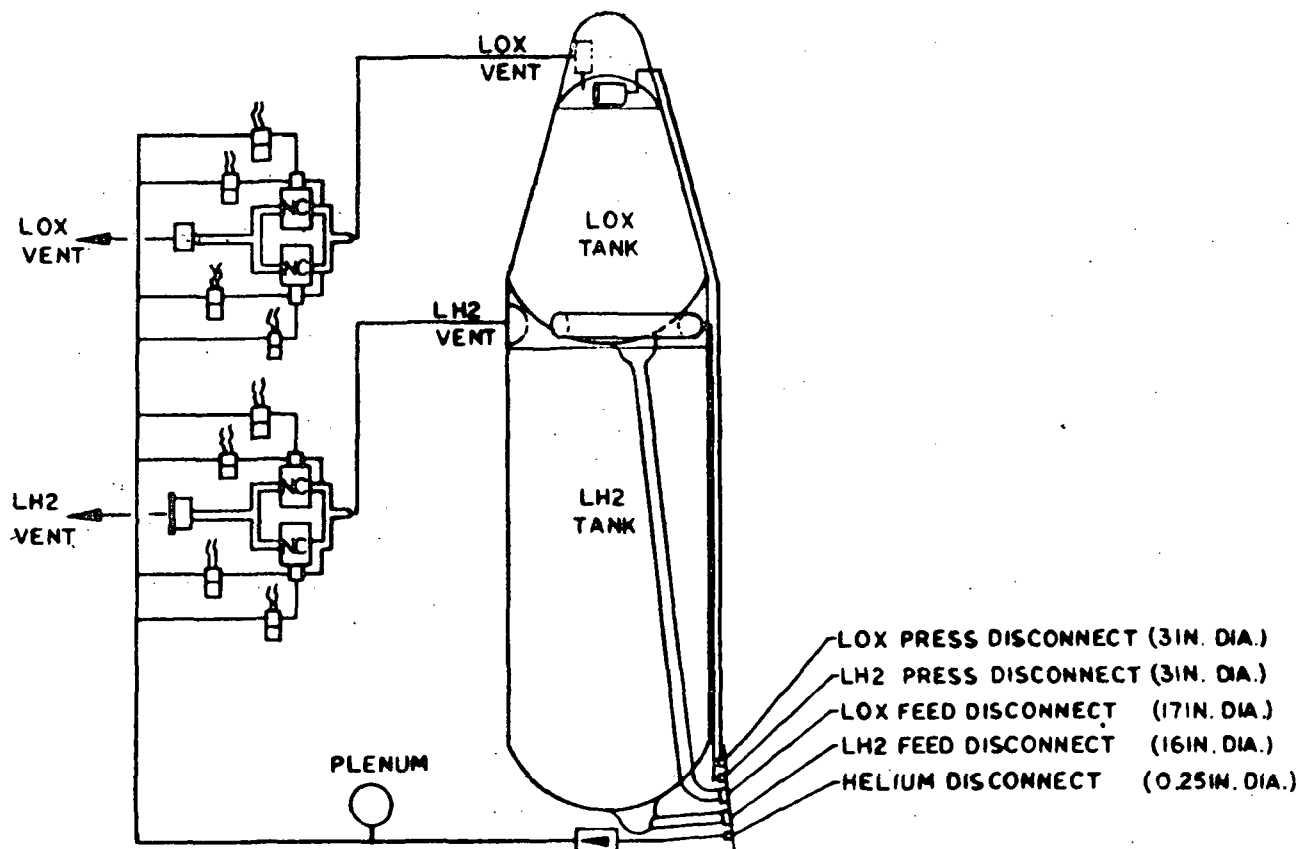


Figure 5.2.3-7 Propellant System Schematic - Parallel Burn

15 MARCH 1972

Orbiter HO Tank Vent Systems -- The vent system configuration and method of sizing for the parallel burn configuration is similar to that of the series burn configuration. A separate vent sizing for the parallel burn case was not made. Instead the same size vent valves (11-inch diameter) were used for the parallel configuration as were used for the series configuration. The rationale involved was that although the parallel burn LOX tank has a larger surface area, it is located closer to the ground and is more sheltered from the wind. Therefore, the heat transfer per unit area would be reduced. These compensating factors would combine to give a similar steady state standby boiloff rate to that of the series burn configuration.

Propellant Utilization System --The propellant utilization system selected for the parallel burn configuration is identical to the series burn system. A closed loop capacitance probe utilizing software probe shaping is baselined. As in the series burn case, cost effectiveness dictates the use of closed loop rather than the open loop approach; a cost advantage of approximately \$50M results.

Closed loop residuals are considered to be 0.25% of the ascent load whereas the S-IVB worst case demonstrated residual was 0.17% of the ascent load. Current technology and the 2% EMR controlled SSME should result in much improved performance.

Nominal Residuals -- Table 5.2.3-3 shows the nominal residual inventory. Since the parallel burn configuration has only a single LOX feedline, feedline size (and overall tank weight) can be minimized by staggering the engine start times. Since the engines are started on the ground, no velocity losses result from the staggered start. The staggered start does result in the start propellant (propellant burned prior to liftoff) being greater than the start losses for the series burn.

The single LOX feedline draining from the bottom of the LOX tank allows the design of a true zero drop-out sump for the parallel burn configuration, which reduces the LOX tank undrainable to zero.

15 MARCH 1972

Table 5.2.3-3 Propellant Inventory - Parallel Burn

	PROPELLANT WEIGHT - LB			
	TOTAL		TANK RESIDUALS	
	LOX	LH ₂	LOX	LH ₂
ASCENT	1,312,114	218,686		
START PROPELLANT	7,414	1,441		
SHUTDOWN ADJUSTMENT	150	87		
FEEDLINE RESIDUAL	918	285	222	95
CHILLOWDOWN RESIDUAL	134	6	0	0
ENGINE RESIDUAL	1,072	75	0	0
PU BIAS	0	547	0	547
TANK UNDRAINABLE	0	100	0	100
PRESSURANT	1,740*	795*	1,920**	970**
NOMINAL LOAD	1,323,544	222,022		
LOADING ALLOWANCE	13,235	2,220		
MAXIMUM LOAD	1,336,779	224,242		
PROPELLANT BELOW TANK	-13,104	-366		
MAXIMUM LOAD IN TANK	1,323,675	223,876		
	VOLUME FT ³			
PROPELLANT VOLUME IN TANK	18,643	50,939		
TANK VOL FOR FLUIDS (2% MIN ULLAGE)	19,016	51,958		
VOL. DISPLACED BY INTERNAL LOX FEEDLINE	0	227		
TOTAL TANK VOLUME	19,016	52,185		

* INCLUDES ONLY PRESSURANT FROM LIQUID.

** INCLUDES PREPRESSURIZATION GASES
LOX TANK: 180 LBS - LH₂ TANK: 175 LBS

Chilldown is accomplished for the parallel burn configuration by dumping propellants overboard prior to liftoff. The line size and routing for this case results in lower childdown residuals than for the recirculation system used for the series burn configuration.

Deorbit Propulsion - The SMR shown on Figure 5.2.3-10 provides the thrust necessary to deorbit the HO tank. The SRM is attached to the forward LOX dome with a thrust structure and is protected during ascent by the nose cone. Characteristics of the SRM and thrust structure are shown in the figure.

15 MARCH 1972

Feed System - LOX is supplied to the three 470K (vac) thrust $H_1 P_c$ engines from the tank through 17-inch diameter feed lines which are routed through the LH_2 tanks. Figure 5.2.3-8 shows the system. Hydrogen is supplied through a single 16 inch diameter line as shown on Figure 5.2.3-9.

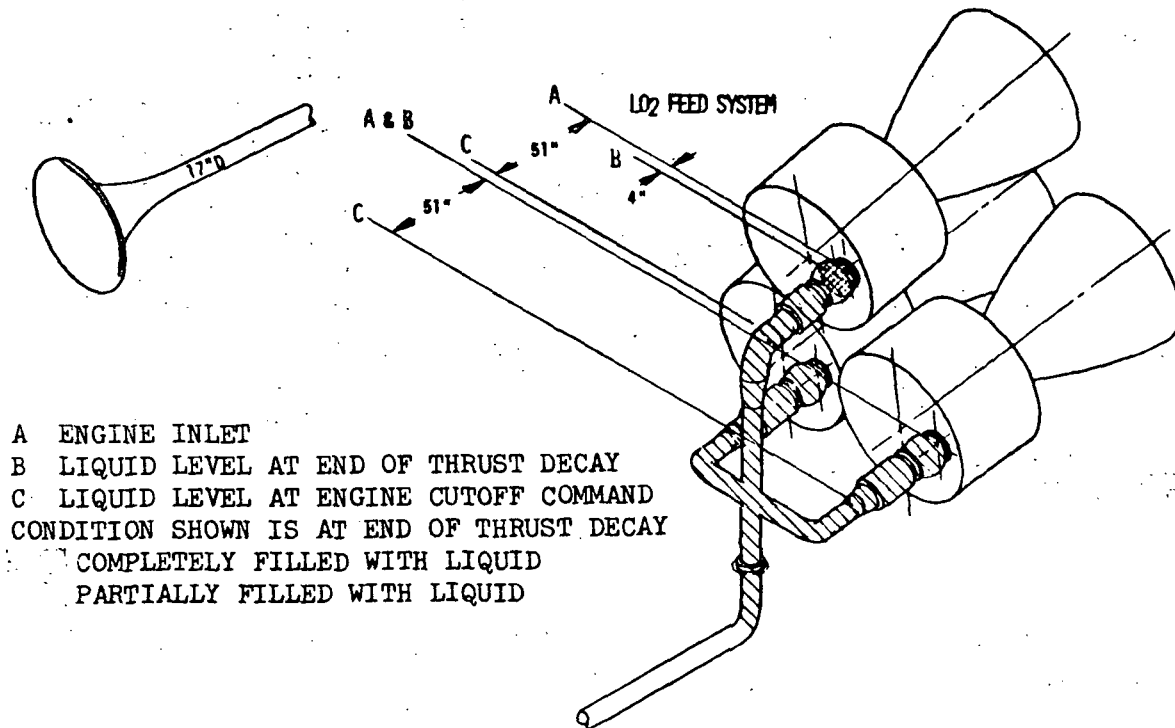


Figure 5.2.3-8 LOX Feed System

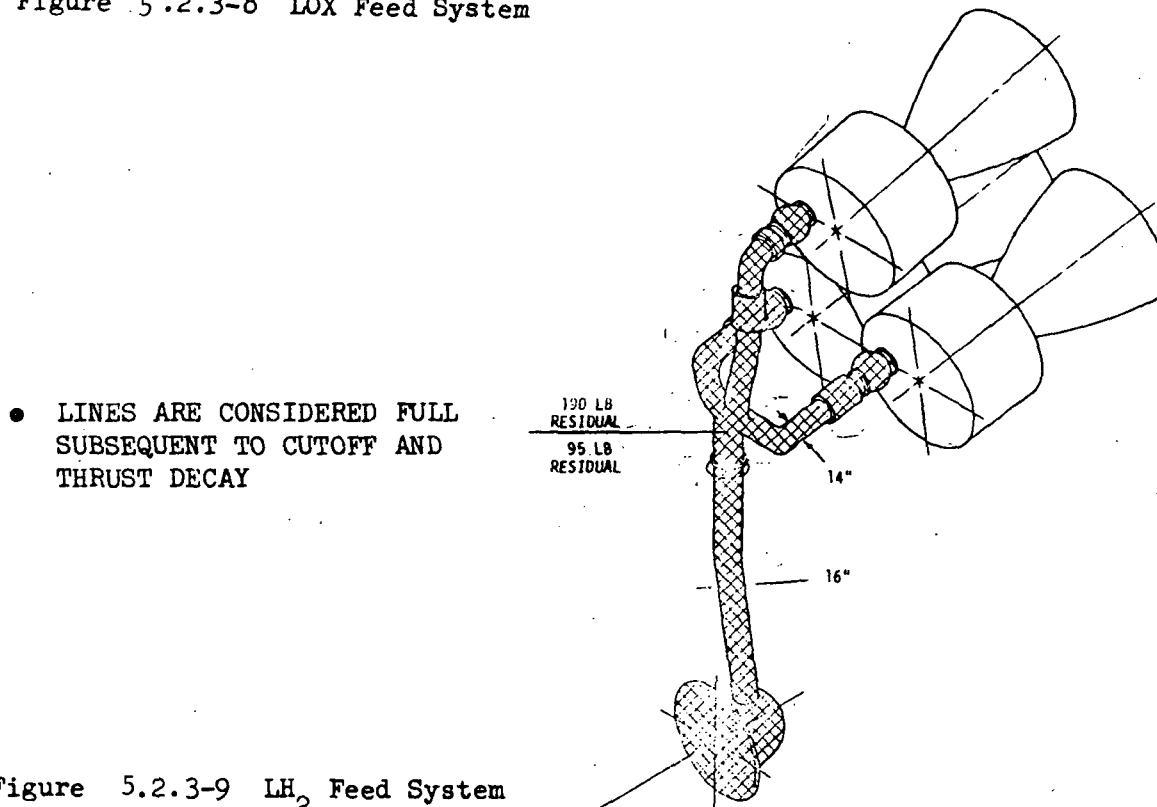


Figure 5.2.3-9 LH_2 Feed System

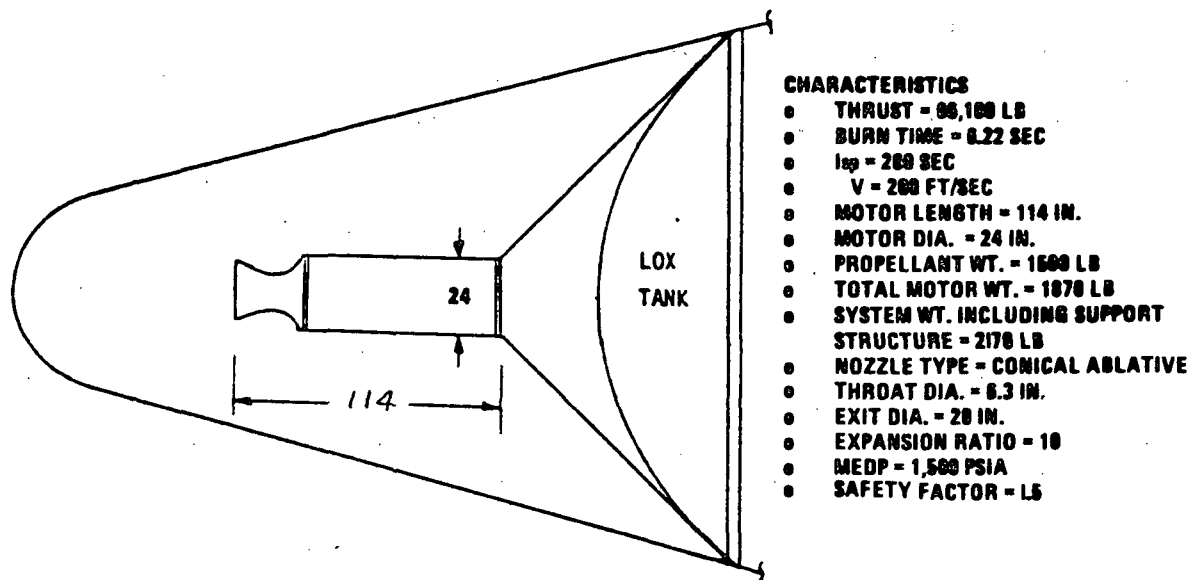


Figure 5.2.3-10 Deorbit System - Parallel Burn

5.2.3.4 Avionics

The avionics summary in Paragraph 4.2.3.4 is equally applicable for the parallel burn configuration.

15 MARCH 1972

6.0 Test and Manufacturing

6.1 Development Testing

Test planning for the Space Shuttle is guided by the groundrules that a baseline concept configuration is available at the program outset, that the configuration is frozen early in the engineering design phase, that horizontal flight testing will be used to optimize landing techniques, and that five orbital flights are adequate proof of vehicle capability.

The test program includes ground tests of models, test units, and simulators. These are used to evolve the flight test vehicles. Two flight vehicles are required to verify horizontal aero-flight capability. Use of two vehicles assures flight test flexibility and the capacity to meet and handle all contingencies.

6.1.1 Ground Testing - The ground testing is a progressive process that verifies portions of the system in discrete steps. The configuration, the functional subsystem, and the structure are each proven so that the Orbiter vehicle can be certified for horizontal then vertical flight as seen in Figure 6-1 and explained below.

Some test units are retained in working condition beyond the First Mated Orbital Flight (FMOF) for possible flight test support.

6.1.2 Flight Testing - Flight testing of the Space Shuttle Orbiter will be conducted in two segments: 1) Orbiter horizontal takeoff flight tests in which an individual Orbiter makes a rolling takeoff from a runway using its airbreathing propulsion system; and 2) vertical takeoff flight tests in which the Orbiter is mated with a Booster and launched vertically into orbit.

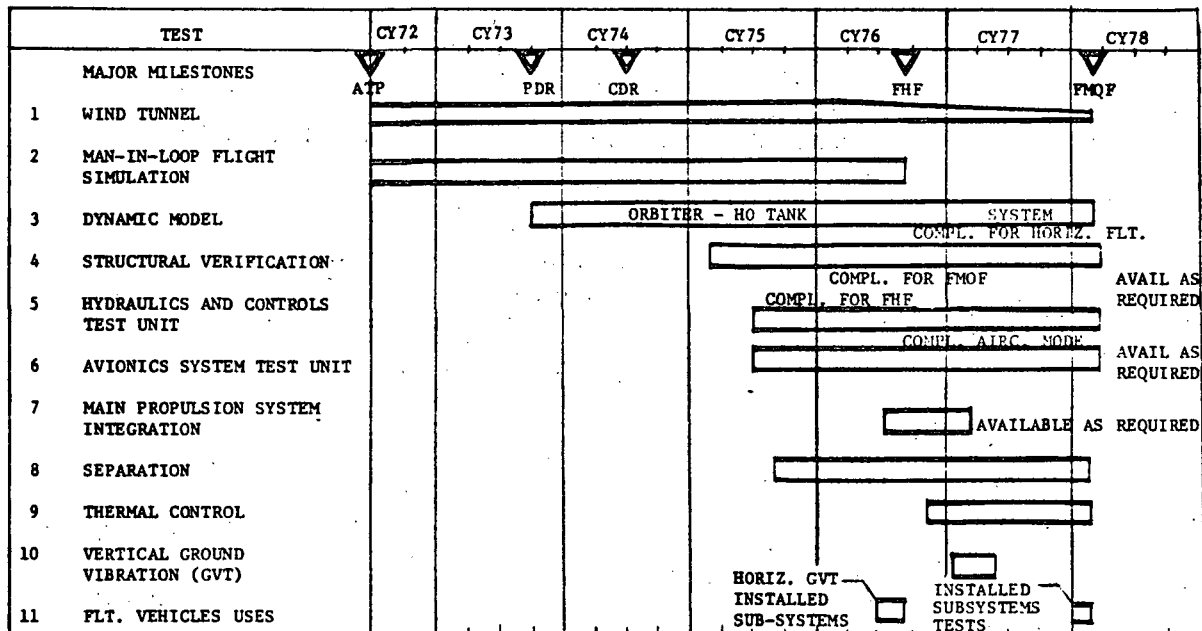
Horizontal takeoff tests will utilize two prototype Orbiters to verify aero-flight and landing capability. A total of 402 flight hours of testing is required. The first horizontal flight is scheduled for 1 September 1976. (Figure 6-2) Initial testing will be performed at Edwards AFB with the final portion of this phase flown from Kennedy Space Center.

Five vertical flight tests will be conducted with these same two orbiters. All five of these flights are planned to be manned, orbital flights from Kennedy Space Center, with the first occurring 1 March 1978. These flights will be heavily instrumented but actual payloads are to be carried, and all mission phases are

15 MARCH 1972

FIGURE 6-1

ORBITER/SYSTEM* LEVEL - SUMMARY GROUND TEST SCHEDULE

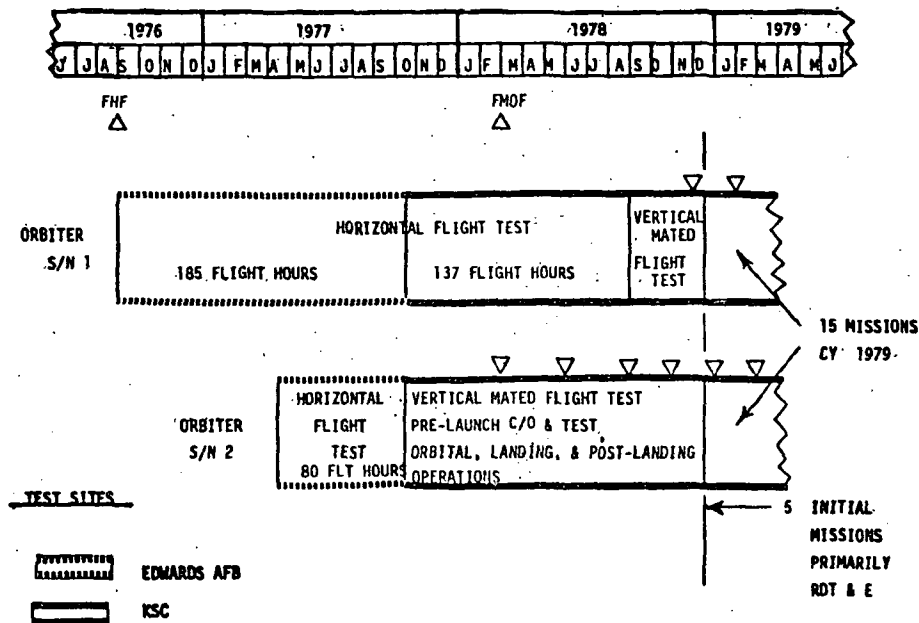


*REFERENCE: SERIES PRESSURE FED ORBITER

1. Wind tunnel testing acquires the data needed to analyze aerodynamics, thermodynamics, loads, structural dynamics, and the propulsion technologies. Tunnel occupancy hours is estimated to be 29,000 hours for Orbiter testing and 6,000 hours for mated and proximity testing.
2. Pilot interfaces, handling qualities and flight procedures are evaluated with man-in-the-loop flight simulation.
3. Dynamically scaled models of the Orbiter, HO Tank, and Booster are tested to acquire structural dynamic characteristics data to enhance analytical modeling.
4. Structural verification encompasses strength, safe-life, and functionality (e.g., landing gear deployment). The Structural Test Article utilization is supplemented with testing of other multi-purpose structural test sections.
5. Hydraulic subsystem and flight control system integration testing is accomplished using the Hydraulics and Controls Test Unit.
6. The Avionics System Test Unit, supplemented with commercial computational equipment, is used for hardware integration testing and software validation.
7. Cluster firing for integration testing of the main propulsion system employs a 'boilerplate' HO Tank which allows simulation of flight 'g' effect on the propellant column above the engines by increasing tank ullage pressure.
8. Separation testing employs planar mass-inertia models and flight hardware interconnects. Booster-Orbiter staging is accomplished along with various modes of HO Tank and Orbiter separation.
9. Thermal control verification data is acquired from heat balance tests employing portions of the Structural Test Article supplemented by structural test sections.
10. Low energy dynamic responses and modal damping characteristics are acquired by use of the Orbiter and HO Tank structural test articles. These data verify hardware characteristics in the ascent configuration.
11. A horizontal ground vibration test (GVT) is performed on Orbiter No. 1 prior to First Horizontal Flight. Installed subsystems tests are performed on both development vehicles.

15 MARCH 1972

Figure 6-2.
FLIGHT TEST PLAN
ORBITER



verified. Subsequent to the five vertical, developmental tests, all flights are categorized as operational flights but some developmental testing may also be performed.

6.2 Manufacturing

Manufacturing activity during the Phase B Extension Study period has been directed at supporting design engineering in performance of conceptual design trade studies, and in analyzing newly selected design concept baselines to determine their impact on the previously established manufacturing approach. The primary conclusion of this manufacturing analysis is that the latest selected Orbiter design concept is easier to manufacture because of the:

- 1) Reduction of the size of the Orbiter from 174.7 ft long by 63.7 ft high with a 107.5 ft wing span, to 123 ft long by 45.1 ft high with a 79.6 ft wing span.
- 2) Removal of the internal, structurally integral, main propulsion system tank.
- 3) Change in material for the vehicle structure from titanium with boron aluminum stiffeners to an all aluminum structure.
- 4) Change of the orbital maneuvering, attitude control, and air breathing

15 MARCH 1972

propulsion systems from internal, structurally integral systems to separable, self-contained modules.

5) Change in design of the Orbiter vehicle to a more conventional aircraft monocoque type of structure.

The reduction in size of the Orbiter vehicle reduces manufacturing parts and assembly sizes. These are more compatible with existing machine tool capabilities and capacities, allow greater application of standard raw material stock sizes, and permit consideration of a greater number of manufacturing locations.

Removal of the internal, structurally-integral main propulsion tank, coupled with the change to a more conventional monocoque type of vehicle structure, permits a manufacturing breakdown of the Orbiter body into more conventional aircraft fuselage section assemblies. This allows a greater subdivision of the manufacturing effort, facilitates performing the manufacturing operations in parallel, and results in a shorter manufacturing schedule time span.

Removal of the internal tank and the reduction in the vehicle size allows a manufacturing breakdown of the vehicle into sections that are small enough to be air transportable. This removes all requirements for barge transportation (except for the external main propulsion tank).

An all-aluminum Orbiter structure eliminates the cost of titanium and boron-aluminum materials. It also results in cost and time savings by eliminating the hot-forming operations and the titanium piece parts. The more complex fabrication of boron-aluminum parts is also eliminated.

The design change to separable, self-contained propulsion modules for the orbital maneuvering, attitude control, and air breathing propulsion system promotes manufacturing replaceability and potential interchangeability. It makes possible complete manufacturing and acceptance test of the modules at single locations. It also reduces manufacturing schedule span time by permitting more work to be performed in parallel.