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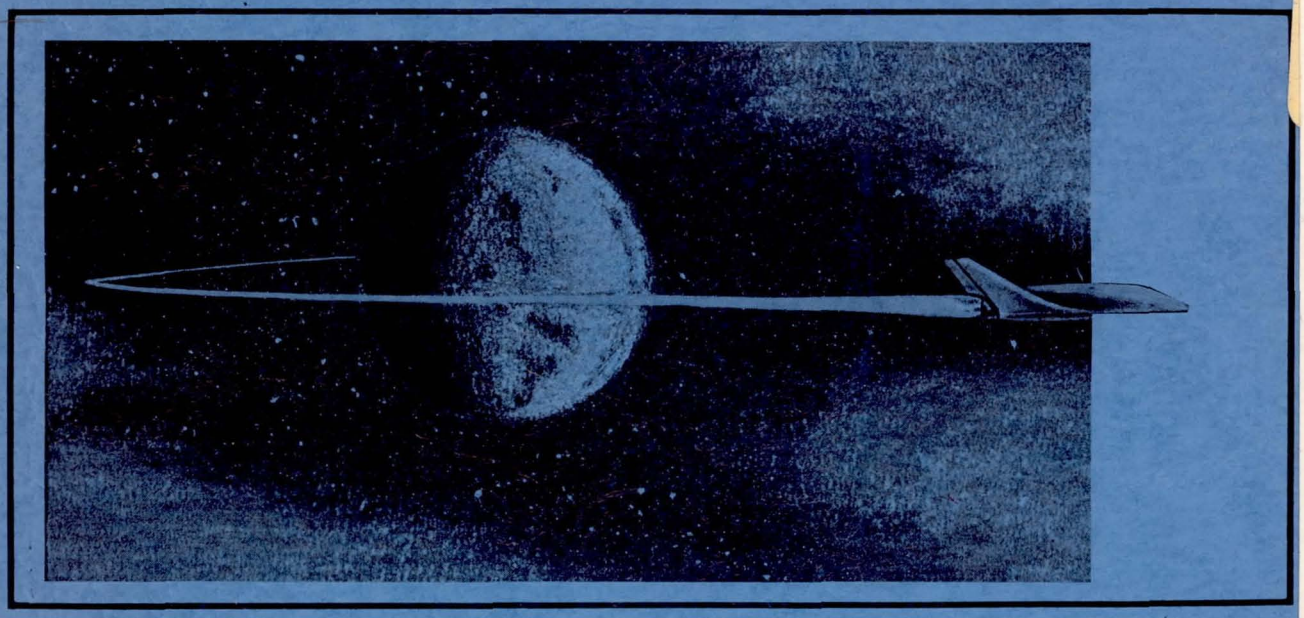

FINAL REPORT

STUDY OF ALTERNATE SPACE SHUTTLE CONCEPTS

VOLUME II — PART II
CONCEPT ANALYSIS AND DEFINITION

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Contract NAS 8-26362

Prepared for George C. Marshall Space Flight Center by
Manned Space Programs, Space Systems Division

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

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FOREWORD

This is the final report of a Phase A Study of Alternate Space Shuttle Concepts by the Lockheed Missiles & Space Company (LMSC) for the National Aeronautics and Space Administration George C. Marshall Space Flight Center (MSFC). The eleven-month study, which began on 30 June 1970, is to examine the stage-and-one-half and other Space Shuttle configurations and to establish feasibility, performance, cost, and schedules for the selected concepts. This final report consists of four volumes as follows:

Volume I	–	Executive Summary
Volume II	–	Concept Analysis and Definition
Volume III	–	Program Planning Data
Volume IV	–	Cost Data

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L2-01-06-M1-1B	Electrical Power System Preliminary Baseline For Stage-and-One-Half Including Redundancy
L2-01-03-M1-1B	Data Management System Preliminary Baseline For Stage-and-One-Half Including Redundancy
L2-01-03-M1-5	Incremental Two-Stage Avionics Development Analysis

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2.14 MASS PROPERTIES

The mass properties data included herein represent a culmination of the design effort for a stage-and-one-half vehicle. The present design, designated LS 200-10, is essentially the same basic vehicle as the LS 200-7 reported in the Eighth Monthly Progress Report except for the following changes to achieve hypersonic stability.

(1) Cruise engines were moved from a fixed emplacement in the base to a movable pod in the bottom mid section.

(2) The payload deployment mechanism was shifted to the front of the payload compartment.

(3) Cruise fuel tank was shifted forward.

The ensuing weight summaries reflect the three basic missions, with major emphasis applied to the mission requiring the highest ΔV capability, that is the 55 deg by 270 nm. This becomes the basic reference mission for the stage-and-one-half design, since its nine main stage engine application renders it insensitive to the abort mode criteria. All detailed veh. wts. are shown for this case; summary statements which reflect vehicle weight for 40K payload south polar launch and 65K payload due east launch are also shown. Exact mission payloads are shown and are consistent with abort criteria discussed in Section 3.3.3. In addition, Table 2.14-1 defines potential payload to south polar and due east of 54,600 lb and 95,100 lb resulting from filling droptanks with propellant to capacity and limiting GLOW to 3,816,420 lb (T/W = 1.25).

The weights, geometric parameters, mass properties and sequential weights for the reference mission are summarized in Table 2.14-1 through 2.14-8, while the Due East mission weights are summarized in Tables 2.14-9 and 2.14-10, and the South Polar mission in Tables 2.14-11 and 2.14-12.

2.14.1 Droptanks

The droptanks were sized for the stage-and-one-half design mission (55 deg/ 270 nm) which accommodates 31,122 lb of startup transient propellant losses, 3,098,495 lb of impulse propellant and 10,555 lb of tank residuals at separation. The dry weight of 112,162 lb was developed using a tank $\lambda' = 0.9615$. This value was calculated from detailed tank drawings and data as presented in the Third and Fifth Monthly Progress Reports. Structural materials for weight evaluation were AL 2219-T87 for LO_2 tank, intertank, and LH_2 tank and skirt, and titanium for the aft support cone and aft thrust structure attach box beam. The droptank insulation consists of a combination of cork, foam, and bonding material weighing 7,144 lb. Cork insulation is used in the nose-cap and oxygen tank cone section for protection from ascent aerodynamic heating. Two pounds per cubic foot of spray polyurethane foam is used on the hydrogen tank and is sized to prevent formation of liquid air during ground hold. An additional 2,400 lb of cork insulation is required over the entire surface of the tanks to protect the tank structure for intact entry. The design and thermal discussion are presented in EM L2-02-01-M1-5, "Droptank Dispersion Study-Thermostructural Analysis," Appendix A. Droptank plumbing weights were estimated from data for associated line diameters, temperatures, and line lengths presented in the Third Monthly Report of the Cyrogenic Optimization Study, LMSC-A981648 (NAS 9-11330), pages 3-64 to 3-68.

The electrical conversion and distribution weight of 326 lb is an estimate for sensors and instrumentation.

The primary landing gear bulkheads and aft payload bay bulkhead weights were obtained by statistical comparison to the C-141 primary bulkhead unit weights. The C-141 was chosen because its landing speed, sink rate and landing weight condition most closely approximated the LS 200-10 design. A 1,200 lb weight penalty to the payload bay trough section is included for the due east mission with the 65K payload. The crew cabin weights are based upon a 14.7 psia

shirtsleeve environment for a four-man capability but two-man normal occupancy. The primary engine thrust structure of titanium was sized initially for 11 main engines of 420Klb thrust each (EM L2-01-04-M6) and the present values for the LS 200-10 were obtained by application of these weights to the present 9 engine design of 530Klb each. A NOF of 50 percent was applied to all truss members and 25 percent was applied to shear panels. This resulted in a factor of .0033 lb of structure per pound of thrust, which was used for thrust structure evaluation for both One-and-One-Half and Two-Stage orbiter weight estimates.

The remaining secondary structural items, such as docking penalties, landing gear door penalties, ABPS engine supports and actuation structure, airlock tunnels, equipment supports, etc., reflect estimated or statistically derived values only. Major frames or sill structures have been provided for in the weights for sections such as the payload bay door, the two landing gear doors, and the lower surface ABPS doors. It is felt that these weight allocations are generous, but definitive values must be delayed until allowable deflections for adequate door closure and primary TPS sealing are determined.

2.14.2.3 Induced Environmental Protection. The TPS weights are based upon a passive, fully reusable insulation backed by a titanium zee-stiffened panel operating at 600°F for the vehicle lower surface. For the vehicle upper surface operating at temperatures below 1000°F, the TPS consists of a titanium panel with dynaflex insulation beneath. The titanium panels are sized upon airload external pressure and permissible deflection only and a minimum face sheet of 0.015 in. leading to a composite thickness of 0.037 in. is employed.

A growth and contingency allowance of 10 percent of dry weight was used. This is consistent with NASA orbiter groundrules and previous LMSC droptank weight estimates.

2.14.2 Orbiter

2.14.2.1 Aero Surfaces. The aero surfaces weights are in part based upon statistical observations. This is true for the upper elevons and vertical fins. The lower elevon was structurally sized on the basis of the analytically determined hinge moments (EM L-1-02-14-M5) for an earlier design and the unit weights obtained were applied to the LS 200-10 areas. The upper elevons, and rudders, being in a more benign environment, reflect a reduction in this unit weight based upon an estimated hinge moment reduction. The vertical fin unit weights were obtained by comparison to a family of aircraft fins operating structurally in approximately the same thermal environment. Actuation system weights for the elevons and rudders are based upon considerations of the hinge moments and duty cycles. A NOF of 20 percent was applied to the lower elevon analytically determined values and are reflected by scaling in the upper elevons and rudders.

2.14.2.2 Body Group. Body shell and frames, including the payload bay trough section were sized, using the finite element computer program "FAST" (See EM L2-01-01-M1-3). This program calculates section properties and load intensities for any geometrical section having at least one axis of summetry. The program outputs from 9 to 17 discrete elements per half fuselage cross-section.

The basic panel sizes are a zee-stiffened section with the zees oriented inwardly and external zee frames spaced 50 in. apart for support of the TPS panels. Longerons and post supports spaced 100 in. apart are employed to break up the long unsupported lengths of the flat-sided frame elements with the post supports being used most effectively in the payload bay area. An NOF of 25 percent was applied to these analytically determined weights.

here as well as for the basic shell of the body structure. Since the loading is rather straightforward and the analysis uncomplicated, an NOF of 5 percent only is applied to these weights. This is justified on the basis that the weights were determined by a detailed analysis of a representative design and generous NOFs were applied for fillets and fastening devices (Third Monthly Report for Vehicle LS-200-2).

2.14.2.4 Landing Recovery/Docking. The landing gear weights are predicated upon a statistical factor of 3.7 percent of the landed weight and are fairly representative of a wide range of aircraft designs. This percentage has been developed by using the Liebermann techniques as presented in S.A.W.E. Paper No. 210; Title "Rolling Type Alighting Gear Weight Estimating" by C. R. Liebermann, dated 1959, Revised 1965.

2.14.2.5 Ascent Propulsion. The main ascent engines are ICD 13M15000B types as of 1 March 1971. The LS 200-10 system reflects 9 of these engines, 4 fixed and 5 gimballed. The expansion ratio is 53 to 1. The feed and drain system weights reflect a detailed analysis (Fifth Monthly Progress Report) with scaling laws applied to the new geometries associated with the LS 200-10 configuration. The internal tankage membrane weights were determined by the methods documented in EM No. L2-02-01-M2. A 35 percent NOF is applied to these membrane weights to account for gage tolerances weld penalties, access provisions, and local discontinuities. In addition to the above, estimates were added to account for auxiliary fluid systems, tankage, and line supports.

2.14.2.6 Cruise Propulsion. The cruise propulsion consists of 6 PW JTF22A-4 engines packaged in an extensible pod located at the vehicle lower surface midship (See Section 2.5.2). Weight allocations have been made for nacelle packages, pylon extensions for shear transfer to the door skin, as well as penalties to the door and actuation linkages that transfer the thrust load to the bulkhead at Station 1272. In addition, engine accessories, plumbing, and tankage weights have been accounted for. Primary framing around the pod cavity as well as the cavity skin and insulation have been provided in the body structure group.

2.14.2.7 Auxiliary Propulsion. The auxiliary propulsion system consists of the OMS plus the ACS system. The OMS consists of two RL10A-3-3A engines plus associated plumbing, tankage, and supports while the ACS consists of 40 thrusters operating with GO_2 and GH_2 at 1,500 lb thrust each plus associated valves, plumbing, accumulators, and pumps required.

2.14.2.8 Prime Power. The prime power group weights are based upon a package consisting of 3X5KW fuel cells, 3X40KVA alternators, 3X200HP APUs, and a 12,000 watt-hour standby battery capability.

2.14.2.9 Electrical. The electrical package consists of power conversion devices, power control units, distribution in the form of busses and wiring and supports for all equipment.

2.14.2.10 Hydrualic Conversion and Surface Controls. The hydraulics and surface control package is a 14,000 psi triple redundant FO/FO/FS unit and consists of 12X55 HP pumps and actuators along with their associated supports, plumbing lines, valves, and accumulators, etc. The actuator weights were determined as a function of hinge moment, stroke and required duty cycle.

2.14.2.11 Avionics. The weights for all avionics qquipment are based upon the units described in the series EM L2-01-03 that were presented in the Fourth and Fifth Monthly Status Reports

2.14.2.12 Environmental Control and Personnel Provisions. The weights for these systems are based upon an $O_2 - N_2$ minimum system sized on a two-man 7-day shirtsleeve environment for a 14.7 psia capability and a 25,000 Btu/hr cooling capacity.

2.14.2.13 Growth/Uncertainty. Based upon 10 percent of all dry weight items less the ICD engine weights.

2.14.2.14 Personnel. These weights are based upon two men, their helments, garments, and pressure suits.

2.14.2.15 Reserves and Residuals. For definition of these weights, see Items 24 and 25 of the accompanying Design Data Summary. For a more thorough definition of these and the other remaining fluid items, see the discussions on Performance (Section 2.4) and Propulsion (Section 2.9) of this report.

Table 2.14-1

STAGE-AND-ONE-HALF
WEIGHT SUMMARY

	55 Deg/270 nm		Due East		South Polar	
	Droptank	Orbiter	Droptank	Orbiter	Droptank	Orbiter
GLOW - System	3816420		3450832		3587902	
Propellant Ascent Imp.	3063218	239209	2713668	239662	2877017	239889
Pre-Separation	122716	630158	122716	614448	122716	587841
Orbiter Injection Wt		376030		360071		333711
ABPS Propellant Cruise		8430		0		0
Maneuver/ACS Imp.		39824		18226		16981
Other, Reserve Residuals Losses		22899		20963		20702
Personnel		725		725		725
Cargo*		25000		65000		40000
Dry Weight		294399		269872		269872
Main Engine Weight		14490		14490		14490
Growth and Contingency		20008		17794		17794
Payload:						
Required		25000		65000		40000
Potential**		25000		95143		54653

*Cargo achieved by off-loading droptank impulse.
**Payload potential obtained by filling droptanks to capacity limiting GLOW at 3816420 lb.

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Table 2.14-2

55 Deg/270nm MISSION WEIGHT SUMMARY

CONFIGURATION: STAGE-AND-ONE-HALF DROPTANK | BY: Broadhead | DATE: 3May71

CODE	SYSTEM	A	B	C	D	E	F	G	H
1	WING GROUP	N/A							
2	TAIL GROUP	N/A							
3	BODY GROUP	88374							
4	INDUCED ENVIR PROTECTION	7144							
5	LANDING, RECOVERY, DOCKING	N/A							
6	PROPULSION - ASCENT	6102							
7	PROPULSION - CRUISE	N/A							
8	PROPULSION - AUXILIARY	N/A							
9	PRIME POWER	N/A							
10	ELECT CONVER & DISTR	326							
11	HYDRA CONVER & DISTR	N/A							
12	SURFACE CONTROLS	N/A							
13	AVIONICS	N/A							
14	ENVIRONMENTAL CONTROL	N/A							
15	PERSONNEL PROVISIONS	N/A							
16	RANGE SAFETY & ABORT	N/A							
17	BALLAST	N/A							
18	GROWTH/UNCERTAINTY	10196							
19									
	SUBTOTAL (DRY WEIGHT)	112162							
20	PERSONNEL	0							
21	CARGO	0							
22	ORDNANCE	0							
23	RESIDUAL FLUIDS	10554							
24									
	SUBTOTAL(INERT WEIGHT)	122716							
25	RESERVE FLUIDS	N/A							
26	IN FLIGHT LOSSES	N/A							
27	PROPELLANT - ASCENT	3063218							
28	PROPELLANT - CRUISE	0							
29	PROPELLANT -- MANEUV/ACS	0							
30									
	TOTAL(GROSS-WEIGHT)LB.	3185934							

DESIGNATORS:

EVENTS
A LAUNCH WEIGHT
B SEPARATION
C
D
E
F
G
H

NOTES & SKETCHES:
DROPTANK λ' - 0.9615

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Table 2.14-3

55 DEG/270 NM	GENERAL DATA SUMMARY				
CONFIGURATION	STAGE-AND-ONE-HALF LS-200-10	BY	BROAD-HEAD	DATE	3May71
ITEM	ORBITER		DROPTANKS		TOTAL
GEOMETRIC DATA			N/A		
Length (Base to Nose) - Ft	134.7				
Wing Span - Ft	N/A				
Wing Area (Theoretical) - Sq Ft	N/A				
Wing Area (Exposed) - Sq Ft	N/A				
Vehicle Planform Area - Sq Ft	6,846				
Body Wetted Area - Sq Ft	13,529				
Vehicle Wetted Area - Sq Ft	18,944				
Body Outer Mold Line Volume - Cu Ft	-				
Vehicle Outer Mold Line Volume - Cu Ft	97,600				
Ascent Propellant Tank Volume - Cu Ft					
PROPULSION DATA			N/A		
Ascent Engine Thrust - lb					
Sea level	350 K				
Vacuum	612 K				
Ascent Engine Expansion Ratio	35:1				
Gimballed	5				
Fixed	4				
Number of Ascent Engines	9				
Cruise Engine S.L. Thrust - lb	Classified				
Number of Cruise Engines	6				
Cruise Fuel Type	JP-4				
AERODYANMIC DATA			N/A		
Entry Angle-of-Attack - Deg					
Hypersonic L/D Max. (Trimmed)	1.87				
Angle-of-Attack (Subsonic L/D Max) - Deg	15		15		
Subsonic L/D Max (Trimmed)	(1)5.85	to	(2) 5.45		
Cruise L/D (Average)	(1)5.85	to	(2) 5.45		
Cruise Range (No Wind) - nm					
cg Limits Fwd/Aft -					
Landing Speed - knots					
@ α L/D_{max} = 15°	(2) 182 to (1) 195				
@ α Tail Scrape = 22°	(2) 146 to (1) 164				

(1) 40 KLB Cargo In/Airbreather Engines Out

(2) Cargo Out/Airbreather Engines In

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Table 2.14-4

RESUPPLY MISSION (55°)

55 DEG/270 NM PERFORMANCE DATA NUMBER					
CONFIGURATION	STAGE-AND-ONE-HALF	By	Broadhead	Date	3May71

ITEM	UNITS	ORBITER	DROPTANKS
WEIGHTS DATA:			
Burnout Weight	-Lb	385,755	127,986
Nominal Propellant Load	-Lb	239,281	3,065,094
Payload	-Lb	25,000	
Gross Weight	-Lb	625,036	3,193,080
		3,818,116	
VELOCITY DATA:			
Nominal Ascent Velocity	Ft/sec	6,870	23,165
Flight Performance Reserve	Ft/sec	300	-
Total Ascent Velocity	Ft/sec	30,335	
Ascent Specific Impulse	Sec	445.0	445.0
On Orbit Maneuver Velocity	Ft/sec	650	-
Ascent Thrust/Weight (initial)	-	2.02	1.25
On Orbit Maneuver I_{sp}	Sec	439.0	-
PAYLOAD SENSITIVITY DATA			
Specific Impulse	Lb/sec	1,890	
Inert Weight	Lb/lb	-1	- .457
Propellant Load	Lb/lb	.170	.046
On Orbit Maneuver Velocity	Lb/ft/sec	23.1	-
Ascent Velocity	Lb/fps	28.0	-
Thrust/Weight - (initial)	Lb/(0.1 T/W)		
Orbit Inclination	Lb/deg	-630	
Launch Site Altitude	Lb/ft	1.75	
Gross Weight	Lb/lb	.0370	
INJECTION ORBIT CHARACTERISTICS			
Apogee	nm	100	
Perigee	nm	50	
Inclination	Deg	55	
Launch Site Latitude	Deg	28.5	
Launch Site Altitude	Ft above SL	0	

NOTE: Sensitivities are estimates for ~~early~~ resupply mission with 550K engines, and assume a fixed vehicle, except for gross weight, which assumes variable-size droptanks.

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Table 2.14-5

55 DEG/270 NM DESIGN DATA SUMMARY			PAGE 1 of 8
CONFIGURATION	STAGE-AND-ONE-HALF LS 200-10	BY BROADHEAD	DATE 3May 71

MAX DYNAMIC PRESSURE, LB/SQ.FT _____
MAX q α PSF DEGREE 2000
ENTRY VELOCITY FT/SEC 25,100

ENTRY ANGLE DEGREES α - 35° δ -1.7°

ULTIMATE LOAD FACTOR	Nx	Ny	Nz	
Ascent	2.1	±1.4	±1.4	
Separation	4.2	- .84	- .84	
Entry	-0.75	-3.0	-3.0	
Cruise	-0.75	±1.5	±1.5 -3.75	
Landing	-1.95	-1.9	-4.0	

1. WING GROUP "N/A"

	INSIDE FUSELAGE	EXPOSED	
Gross Area SQ FT			
Torque Box			
Leading Edge (Fixed)			
Trailing Edge (Fixed)			
Movable Surfaces			
Volume - CU.FT			

	MAC	ROOT THEORETICAL	BODY JUNCTION	PLANFORM BREAK	TIP
CHORD LENGTH (FT.)					
CHORD THICKNESS (FT.)					

SPAN BETWEEN CHORDS (FT)
DIHEDRAL ANGLE (DEG) ____ SPAN BETWEEN DIHEDRAL BASES FT _____
SWEEP BACK ANGLE (MEAN CHORD) _____ CHORD _____
AIRFOIL DESCRIPTION _____
TAPER RATIO _____
THICKNESS/CHORD: ROOT _____ TIP _____
DESIGN LOAD _____
CRITICAL LOAD CONDITION _____

CONTROL SURFACES	TYPE	AREA — SQ FT RETRACT	EXTEND
L. E. Flaps			
Spoilers			
Speed Brakes			

PRIMARY STRUCTURAL MATERIALS	MATERIAL	DESIGN TEMP °F
Torque Box		
Leading Edge		
Trailing Edge (Fixed)		
Movable Surfaces		

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Table 2.14-5 (Cont'd)

55 DEG/270 NM	DESIGN DATA SUMMARY			PAGE 2 of 8
CONFIGURATION	Stage-and-one-half - LS 200-10	BY	BROADHEAD	DATE 3 May 71

2. TAIL GROUP	AUX SURFACE	FIN/ RUDDER	FLAP(UPR) HORIZONTAL	FLAP(LWR)	
EXPOSED AREA FT^2 TOTAL	(225)	(1248)	(693)	(1086)	(3252)
Torque Box					( )
Leading Edge (Fixed)					( )
Trailing Edge (Fixed)					( )
Movable Surfaces	225	374	693	1086	(2378)
CARRY THROUGH AREA - FT^2					
EXPOSED SEMI-SPAN-FT^2					
CARRY THROUGH SPAN-FT					
NO. OF SURFACES/VEHICLE	2	2	2	1	
VOLUME – CU FT	*	4330		1330	
PRIMARY STRUCT.MATERIAL	Ti	Ti	Ti	Ti	
CHORD LENGTH					
Root (Theoretical) Ft.					–
Mac Ft					
Body, Junction					
Tip, Ft					
CHORD MAX. THICKNESS					
Root Ft					
Tip, Ft					
DIHEDRAL ANGLE, DEGREE					
SWEEPBACK, 25% CHORD					
CRITICAL LOAD CONDITION					

3. BODY GROUP	FWD BULKHEAD	BARREL	AFT BULKHEAD	COMMON BULKHEAD	
INTEGRAL TANK WETTED AREA - FT^2					
Oxidizer Tank					( )
Fuel Tank					( )
Inter-Tank Structure					( )
ULLAGE PRESSURE – PSI	OXIDIZER		FUEL		
BASIC STRUCTURE WETTED AREA – FT^2	FWD (1187)	CTR (7412)	AFT (1371)	SKIRT (3113)	13083
Sidwalls					
Bulkheads					
Partitions					
Thrust Structure (Main Ascent Engine)					
Body Volume - Cu Ft (Total)					( )
PRIMARY STRUCTURAL MATERIAL	Al	Al/Ti	Al	Al/Ti	

*Incl in Fin/Rudder Volume

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Table 2.14-5 (Cont'd)

55 DEG/270NM	DESIGN DATA SUMMARY		PAGE 3 of 8
CONFIGURATION	Stage-and-one-half - LS 200-10	BY Broadhead	DATE 3 May 71

3. BODY GROUP (Continued)

MISCELLANEOUS	WETTED AREA-SQ.FT.	VOLUME CU.FT.	ULT DES.PRESS. DIFF. - PSI
Crew Compartment	1026	645	
Equipment Compartment		1185	
Radome Antennas			
Speed Brakes			
Doors PAYLOAD	1300		
Tanks - Oxidizer			
Tanks - Fuel			

4. INDUCED ENVIRONMENT PROTECTION

	WING (Flap(upr) (Lwr)	HOR.TAIL	VERT.TAIL	BODY	
TOTAL VEH.WETTED AREA-FT^2	N/A	(2190)	(1996)	(13529)	18944 *
NOSE CAP AREA	( )			(203)	203
- SQ FT					
Material Ta/Ti/Dynaflex				203	
Material					
Material					
SURFACE PROT. AREA-SQ.FT.	( )	(1086)	(964)	(11376)	13426
Material LI-1500		1086	964	7642	
Material Ti/Dynaflex				3734	
Material					
Material					
Material					
Material					
Material					
Material					
UNPROTECTED AREA-SQ FT	( )	1104	1032	(1950)	4086
BASE: MATERIAL LI-1500 / RSE **				1229	1229
TOTAL VEHICLE VOLUME - CU. FT. (OUTER MOLDLINE)					97600
VOLUME INSIDE PRIMARY Structure - Cu Ft					
TPS VOLUME CU FT					
LEAD. EDGE/NOSE CAP RAD. FT				4	
LEAD. EDGE/NOSE CAP MAX TEMP ^{0}F			2280^{0}F	2810^{0}F	
LOWER SURFACE MAX TEMP ^{0}F		2300^{0}F		2210^{0}F	
SIDE SURFACE MAX TEMP ^{0}F			1600^{0}F	1600^{0}F	- 800^{0}F
UPPER SURFACE MAX TEMP ^{0}F				600^{0}F	- 800^{0}F

* 693 Ft^2 of Upr Flap Incl in Body

**REINFORCED SILICON ELASTOMER

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Table 2.14-5 (Cont'd)

55 DEG/270 NM	DESIGN DATA SUMMARY			PAGE 4 of 8	
CONFIGURATION	Stage-and-one-Half LS 200-10	BY	Broadhead	DATE	3 May 71

5. LANDING, RECOVERY, DOCKING

ALIGHTING GEAR	NO./VEH	DESIGN LOAD - LB	EXTENDED STRUT LGTH -	STROKE	PRIMARY MATL	BRAKE MATL
Main Gear	2	468200	178 IN	18 IN	STL	BE
Nose Gear	1	82600	178 IN	18 IN	STL	BE

Max. Design Landing Wt - Lb = 324,000
Landing Speed - Knots = 168
Angle of Attack @ Landing - Deg = 20

Limit Landing Sink Speed
10 Ft/Sec @ 324,000 LB. Landing Weight
10 Ft/Sec @ 324,000 LB. Max Design Landing Weight

SEPARATION SYSTEM
Design "q" @ Separation – psf = 4.0
Max. Axial Acceleration - g's = 3.0
Max. Design Separation - Wt.Lb. = 625,642

DECELERATION CHUTE
Diameter - Ft = N/A
No./Vehicle = N/A

6. PROPULSION – MAIN ASCENT

	THRUST-SL	THRUST-VAC	EXP RATION	I_{sp} VAC	CHAMB PRES PSI
ENGINE	530 K	612 K	53:1	445	3000

PROPELLANT SYSTEM	FUEL	OXIDIZER	
Ullage Pres. - psi-Ascent/Empty	21/26	210*/28	OXIDIZER LINE
Propel Type	LH_2	LO_2	
Ullage Vol - Ft^3	243	0	( )
Pressurant	LH_2	LO_2	
Total Tank Vol - Ft^3	8358	2565	
Usable Prop Vol - Ft^3	8115	2565	
Total Vol of Feedlines - Ft^3		595	

TANKAGE – NONINTEGRAL	NO.	SHAPE	WET AREA SQ FT	VOLUME CU FT
Oxidizer	2	Sphere	571 ea.	1284 ea
Oxidizer				
Oxidizer				
Fuel	2	FRUST.	1812 ea.	4390 ea
Fuel				
Fuel				

*Cascaded from droptanks 3g max accel.

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Table 2.14-5 (Cont'd)

55 DEG/270 NM	DESIGN DATA SUMMARY			PAGE 5 of 8
CONFIGURATION	Stage-and-One-Half - LS 200-10	BY	Broadhead	DATE 3 May 71

7. PROPULSION - CRUISEBACK

No. of Engines	6
Engine Thrust - S.L. Static - Lb	*
Specific Fuel Consumption Lb/Lb - Thrust Per Hr @ Nominal Cruise Altitude - Ft	
Nominal Cruise Altitude - Ft	
Nominal Cruise Speed - Knots	
Cruise Altitude Engine Out - Ft	
Cruise Speed, Engine Out - Knots	
Cruise Range (Actual Req) - NAMI	
Cruise Range (Max Avail - No Headwind, All Engines Up)	
Cruise Lift Drag Ratio	
Engine Thrust Sized By	
Lift Coefficient for Critical Thrust Condition	

	TYPE	TANK VOL CU FT	TANK MATL	TANK PRES-psi	BURST FACTOR	NO. OF TANKS
FUEL SYSTEM	JP-4	196	Al			1
PRESSURIZATION SYSTEM						

8. PROPULSION - AUXILIARY

THRUSTERS Thrust (Vac) - Lb	I_{sp} - Sec	ACS	RL-10 QUANTITY REQ MANEUVER	COMBIN	
1500	352	39			( )
RL-10 15000	439		2		( )
					( )
					( )
					( )

PROPELLANT SYSTEM	TYPE	TANK VOLUME	TANK MATL	TANK PRES-psi	BURST FACTOR	NO. OF TANKS
Fuel	LH_2	2358	Al	21		1
Oxidizer	LO_2	640	Al	19		1
Fuel Pressurant	GH_2					
Oxidizer Press.	GO_2					

**Net Usable Plus Ullage

9. PRIME POWER

	SPECIFIC POWER		TOTAL POWER		TYPE
Batteries	60	Watt-Hr/Lb	12000	Watt-Hrs	Ag-Z_n
Engine/Turbine	158	HP-Hr/Lb Fuel	120 K	HP-Hr	
	2.0	HP/Lb of Engine (	800	HP	
Fuel Cell	1235	Watt-Hrs/Lb Fuel	864	Watt-Hrs	H_2 O_2
	30	Watts/Lb of Fuel Cell	16 K	Watts	

*Classified

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Table 2.14-5 (Cont'd)

55 DEG/270 NM DESIGN DATA SUMMARY				PAGE 6 of 8
CONFIGURATION	Stage-and-One-Half - LS 200-10	BY	Broadhead	DATE 3 May 71

10. ELECTRICAL POWER CONVERSION/DISTRIBUTION

System Voltage - 28 VOLTS
Peak Power - 6900 WATTS
Average Power - 4800 WATTS

11. HYDRAULIC POWER CONVERSION/DISTRIBUTION

System Nominal Oper Pressure - 4000 PSI
Peak Power - 342 HORSEPOWER
Average Power - 115 HORSEPOWER
Total Volume of Fluid - ____ FT^3
Fluid Type MIL-H-5606 Max Oper Temp - °F ____

*12. SURFACE CONTROLS

SURFACE	AREA FT^2	MAX DEFL RATE DEG/SEC	MAX DEFL DEG	DESIGN HINGE MOM. FT/LB	NO/ VEHICLE
Upr Flap	693	15	** -30	788 K	2
Rudder	374	15	+10 -30/+15	364K/121K	2
Aux. Surf	225	2.5	-20 - 40	367 K	2
Flap Lwr	543	1	+10 - 25	966 K	1
Elevon Lwr	543	15	±20	201 K	2

13. AVIONICS (LIQUID COOLED, 120 VDC POWER, QUAD REDUNDANT, FO/FO/FS)

14. ENVIRONMENTAL CONTROL

	TOT. STOR VOL - FT^3	STORAGE PRES.	TANK MATL	NO. OF TANKS
Gas Supply System	(10)			
Primary Oxygen & Cooling H_2			STL.	
Second Oxygen (Super Critical)		900 psi	STL.	1
Diluent N_2 (" ")		900 psi	Ti	1

Gas Requirement Average Rates
Metabolic - 11.6 Lb Man-Day
Leakage - 2.0 Lb Day
Repressurize - N_2 = 29.4; O_2 = 8.9 Lb Repressurize (CABIN)
Repressurize - ____ Lb/Repressurize (AIRLOCK)

Heat Transport System Capacity - 35000 Btu Hr (PEAK) Radiator
- 10000 Btu Hr (PEAK) Total System

Radiator Area - 640 Sq Ft α/ϵ .05/.8

Water Management System Capacity

Drinking Water	6*	Lb Man-Day	14	Man Days
Washing	2*	Lb Man-Day	14	Man Days
Other	5*	Lb Btu		BTU's

15. PERSONNEL PROVISIONS

16. RANGE SAFETY AND ABORT

*Supplied by fuel cell.
** Rudder Bias/Control Deflection

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Table 2.14-5 (Cont'd)

55 DEG/270 NM	DESIGN DATA SUMMARY			PAGE 7 of 8	
CONFIGURATION	Stage-and-One-Half - LS 200-10	BY	Broadhead	DATE	3 May 71

17.

Design C.G. FWD = 70.5 %L AFT = 76 %L
Nominal C.G. = 73 - 78 %L ENTRY
Nominal C.G. with 25000 Lb Payload = 73 %L ENTRY

L = 1800 inches

18. GROWTH/UNCERTAINTY

Current Allowance = 20008 LB
Contractors Est of Allowance Needed to guarantee
Current Payload = 25000 LB*
Remaining Growth Allowance for
Customer Changes = LB

19. OPEN

20. PERSONNEL

No. of Crew = 2 ; ave percentile man = 37 *
No. of Personnel = 0 ; ave percentile man =
TOTAL

*REF: SAWE HANDBK

21. CARGO CARGO BAY VOLUME = 10,770 Cu Ft

Bay Dia = 15 FT
Bay Lgth = 60 FT

22. ORDNANCE

23. RESIDUAL FLUIDS - DEFINE WEIGHT ESTIMATING RATIONALE

24 TANKS/LINES (Asct/) - MAINTAIN TANK PRESSURE

AB Fuel - 3% of Tank Content
ACPS - Accumulator Gasses
SERVICE - Cabin Radiator Freon plus coolants
Hydraulic fluids and fuel cell resid.

25. RESERVE

Ascent - 1 Percent of V ideal for 50 x 100 nm
Maneuver - Incl. in on-orbit ΔV reqt
A/B Fuel - 14 percent impulse fuel
ACPS - 10 percent impulse
Service -
ECS - 1 day supply
~~EPS - 10 percent on APU and 20 percent on fuel cell~~

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Table 2.14-5 (Cont'd)

55 DEG/270 NM	DESIGN DATA SUMMARY			PAGE 8 of 8
CONFIGURATION	Stage-and-One-Half LS 200-10	BY	Broadhead	DATE 3May71

27 - 29 PROPELLANTS EXPENDED

	ASCENT	CRUISE	MANEUVER	ATTITUDE
Oxidizer/Fuel Ratio (By Weight)	6	N/A	* 5 or 3.52	3.52
Oxidizer Ullage Volume - Percent	3		N/A	N/A
Fuel Ullage Volume - Percent	3			
Fuel Density - pcf	4.274		4.274	4.274
Oxidizer Density - pcf	70.2		70.2	70.2
Fuel Bias - Percent				
Incremental Velocity -fps				
Inertial				
Maneuver Losses				
Gravity Losses				
Drag Losses				
Back Pressure Losses				
Engine Cant				
Earth Rotation				
FLT. Performance Reserve	8079			
		( )		

*Using Either RL-10 or ACPS System

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Table 2.14-6

55 Deg/270 nm	GROUP WEIGHT STATEMENT			Page 1 of 6
CONFIGURATION	STAGE-AND-ONE-HALF LS 200-1	BY		DATE 3 May 1971

1. WING GROUP					N/A
		CARRY THROUGH	EXPOSED SURFACE		
Basic Structure					
Torque Box					
Leading Edge					
Trailing Edge					
Secondary Structures					
Variable Geometry (incl ____ lbs mechanism)					
Doors Insulation Fairings					
Control Surfaces		SURFACE	SUPT/MECH		
Elevon — (incl. bal wt. ____ lb)					
T.E. Flaps					
L.E. Flaps					
Spoilers					
Speed Brakes					
2. TAIL GROUP		VERTICAL	HORIZ		18050
Basic Structure		5242		5242	
Torque Box					
Carry Through					
Leading Edge					
Trailing Edge					
Secondary Structure					
Control Surfaces		SURFACE	SUPT/MECH	12808	
Rudder (incl. bal wt 0 lb)		2246			
Body Flap and Elevon		5104	1786		
Upper Trim Flap		2772			
Aux Cont Surf		900			
3. BODY GROUP					58386
Integral Tankage					
Fuel Tank					
Oxidizer Tank					
Between tanks (cmn blkhd)					
Insulation					
Basic Structure	FWD. (1857)	CTR. (20315)	AFT. (6384)	SKIRT (23476)	52032
Sidewalls	1317	14284	6384	5220	
Bulkheads	540	1100			
Partitions		4931			
Thrust Structure (main ascent engine)				18256	
Secondary Structure					6354
Crew Compartment				2370	
Equipment Compartments				710	
Payload Attach & Deploy				1000	
Speed Brakes					
Engine Heat Protection - Structure				1438	
Interstage (incl. mech. ____ lbs)					
Doors/fairings				836	
Gear/wing provisions/ABES					
Contingency					

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Table 2.14-6 (Cont'd)

55 Deg/270 nm	GROUP WEIGHT STATEMENT				Page 2 of 6
CONFIGURATION	Stage & One-Half LS 200-10	By Broadhead		Date	3May71

4. INDUCED ENVIRONMENTAL PROTECTION						35716
Thermal Protection		HOR.TAIL	VER. TAIL	BODY	1147	
Nose Cap - SKT						
Ta/Ti/Dynaflex				1147		
Surface Protection					32353	
Body Lower (LI-1500)			13811			
Body L.E. (LI-1500)			8104			
Body Upr T>1000 (LI-1500)			2512			
Body Upr T<1000 (Ti Dynaflex)			3340			
Fin L.E. (LI-1500)			403			
Fin Side (LI-1500)			1715			
Lower Trim Surf (LI-1500)			2468			
Base Heat Protection					2216	
Base (LI-1500)				1439		
Flame Curtain (RSC)				777		
Sound Protection						
Meteorite / Radiation Protection						
Contingency						

Table 2.14-6 (Cont'd)

55-Deg/270 nm	GROUP WEIGHT STATEMENT			Page 3 of 6
CONFIGURATION	STAGE-AND-ONE-HALF - LS 200-10	By	Broadhead	Date 3May71

5. LANDING, DOCKING							11,988
Alighting Gear	ROLLING*	STRUCTURE	CONTROLS		11988		
Main	5406	4326	858				
Nose	268	824	306				
Docking	* Incl. Brakes						
Auxiliary Systems							
Deceleration chutes			N/A				
Flotation gear							
Handling gear							
Contingency							
6. PROPULSION – MAIN ASCENT							105858
Engine & Accessories					74949		
Engine			31436				
Gimbal System			42880				
Ignition and Control System							
Propellant System			633				
Accessories							
Installation, Ducts, Shrouds		LH_2	LO_2				
Propellant System		9344	9495		19684		
Purge							
Pressurization							
Fill & Vent							
PCV System							
Pre-Valves							
Feed Systems							
Vortex, Flow Control System							
Supports and Install		300	545				
Tankage – Nonintegral	TANK	INSUL	SUPPORT		11225		
Fuel	2692	741	704				
Oxidizer	5378	0	1710				
Contingency							
7. PROPULSION – CRUISE BACK							20603
Engine & Accessories							
Engine			14490		15510		
Ignition and Control Sy							
Lubrication Sy (dry)							
Accessories			1020				
Installation, Ducts, Shroud					4435		
Air Induction							
Engine Mounting Extension			2610				
Nacelles Pylons (incl [illegible])			1825				
Exhaust System							
Propellant System					360		
Fill Drain			360				
Pressurization (dry)							
Vent System							
Pump							
Feed System							
Transfer System							
Dump System							
Supports/Installation							
Tankage – Non Integral	TANK	INSUL	SUPPORT		298		
Fuel-JP-4	240	Incl	58				
Contingency							

Table 2.14-6 (Cont'd)

55 Deg/270 nm	GROUP WEIGHT STATEMENT					Page 4 of 6
CONFIGURATION	STAGE-AND-ONE-HALF LS 200		By Broadhead			Date 3 May 1971

8. PROPULSION – AUXILIARY			Att. Control (4606)		Maneuv. (2295)	6901
Thruster Installation			1130		700	
Thruster		1130		700		
Accessory						
Propellant System			2655		422	
Fill/Drain/Vent Lines & Valves				147		
Pressurization						
Heat Exchanger		265				
Feed System/Accumulators		2390		275		
Conditioning						
Supports		Incl.		Incl.		
Tankage		LO_2	821	LH_2	1173	
Tanks		446		852		
Insulation		175		151		
Supports		200		170		
Contingency						
9. PRIME POWER	POWER UNIT	MTG INSTAL	PROPEL TANK/SYS	TEMP CONTROL	CON-TROLS	TOTAL 1620
Batteries	198	36				234
Engine Turbine	368	80			24	472
Fuel Cells	480	86	348			914
Contingency						
10. ELECTRICAL		SUPPLY	CONVER-SION	CONTROL UNITS		912 3704
Equipment		120	192	600		
Distribution and Control Circuitry						2227
Utility Systems						
Supports/Installation						565
Contingency						
11. HYDRAULIC		Primary (2073)			2073	2073
Power Supply		144				
Distribution Control Ctr		1725				
Temperature Control Sy		16				
Auxiliary Systems						
Supports/Installation		188				
Contingency						
12. SURFACE CONTROLS						4246
Cockpit Controls						120
Flight Control System						
System Actuation	CONTROLS	POWER XMISSION	ACTU-ATOR	FEEL SY	SUPTS INSTALL	(SEE AVIONICS)
Upr. Elevon	(200)	(150)	(3390)		(386)	4126
			720			
Flap/Elevon Lwr		150	1600			
Rudder			720			
A/B Extension			200			
Misc.	200		150			
Contingency						
13. AVIONICS	UNITS (2816)	CIRCUITRY (321)	COOLING	ANTENNAS (58)	INSTALL. (567)	3762
Guid/Nav	1174	35			35	
Flight Control	909	91			180	
Data Management	186	118			235	
Communicate	107	33		58	30	
Config. Seq.	440	44		Incl.	87	
Instrumentation	Incl.					
Displays	Incl.					
Contingency						

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Table 2.14-6 (Cont'd)

55 Deg/270 nm	GROUP WEIGHT STATEMENT			Page	5 of 6
CONFIGURATION	STAGE-AND-ONE-HALF LS 200-1	By	Broadhead	Date	3 May 1971

14. ENVIRONMENTAL CONTROL			1274
Gas Supply System (dry)		62	
Gas Management, Processing (dry)		177	
Heat Transport System (dry)		996	
Water Management System (dry)		39	
Purge System			
Insulation		Incl.	
Contingency			
15. PERSONNEL PROVISIONS			210
Seats/Restraint Sys (No -)		84	
Fixed Life Support Equipment		112	
Emergency Equipment			
Cargo Handling			
Furnishings		14	
16. RANGE SAFETY AND ABORT			0
17. BALLAST			0
18. GROWTH/UNCERTAINTY			20008
19. OPEN			
SUBTOTAL (Dry Weight)			294399
20. PERSONNEL			725

NO	MAN	GARMENTS	HELMET	PRESS. SUIT	ACCESSORIES	
Crew (2)	330	6	14	70	22	442
Passenger( )						126
Personal Gear/Accessories						157
Life Support						
Food					102	
Equipment - Portable					55	

21. CARGO			25000
22. ORDNANCE			
23. RESIDUAL & UNUSABLE FLUIDS			3330
Ascent		482	
Cruise		286	
Maneuver (Incl. All Aux. Tk. Residuals)		183	
Attitude Control		94	
ECS		450	
EPS		35	
Hydraulic		1800	
Misc (Shock struts, etc.)		Incl.	
24. OPEN			
SUBTOTAL(Inert Weight)			323454

Table 2.14-6 (Cont'd)

55 Deg/270 nm	GROUP WEIGHT STATEMENT			Page 6 of 6
CONFIGURATION	STAGE-AND-ONE-HALF LS 200 By10		Broadhead	Date 3 May 1971
25. RESERVE FLUIDS				10072
Ascent			8079	
Cruise			1205	
Maneuver			0	
Attitude Control			562	
ECS			10	
EPS			140	
			76	
26. INFLIGHT LOSSES				9497
Ascent			7428	
Cruise			0	
Maneuver (Incl. All Aux. Tk Boiloff)			1224	
Attitude Control			0	
ECS			145	
EPS			700	
27. PROPELLANT - ASCENT				239209
28. PROPELLANT - CRUISE				8430
	MANEUV.	ACS		
29. PROPELLANT - MANEUVER/ACS	34207	5617		39824
TOTAL (Gross Weight)				630486

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Table 2.14-7

55 Deg/270 nm MISSION WEIGHT SUMMARY

CONFIGURATION: Stage-and-One-Half LS-200-10 | BY: Broadhead | DATE: 3May71

CODE	SYSTEM NM	A	B	C	D	E	F	G	H
1	WING GROUP	N/A							
2	TAIL GROUP	18050							
3	BODY GROUP	58386							
4	INDUCED ENVIR PROTECTION	35716							
5	LANDING, RECOVERY, DOCKING	11988							
6	PROPULSION - ASCENT	105858							
7	PROPULSION - CRUISE	20603							
8	PROPULSION - AUXILIARY	6901							
9	PRIME POWER	1620							
10	ELECT CONVER & DISTR	3704							
11	HYDRA CONVER & DISTR	2073							
12	SURFACE CONTROLS	4246							
13	AVIONICS	3762							
14	ENVIRONMENTAL CONTROL	1274							
15	PERSONNEL PROVISIONS	210							
16	RANGE SAFETY & ABORT								
17	BALLAST								
18	GROWTH/UNCERTAINTY	20008							
19									
	SUBTOTAL (DRY WEIGHT)	294399							
20	PERSONNEL	725							
21	CARGO	25000							
22	ORDNANCE								
23	RESIDUAL FLUIDS	3330							
24									
	SUBTOTAL(INERT WEIGHT)	323454	323454	323454	323454	323454	323454	323454	323454
25	RESERVE FLUIDS	10072	10072	10072	1993	1993	1993	1993	1993
26	IN FLIGHT LOSSES	9497	9169	4653	2329	2254	492	467	0
27	PROPELLANT - ASCENT	239209	239209	239209	0	0	0	0	0
28	PROPELLANT - CRUISE	8430	8430	8430	8430	8430	8430	8430	0
29	PROPELLANT—MANEUV/ACS	39824	39824	39824	39824	13268	12522	464	0
30	SUBTOTAL - ORBITER GROSS	630486	630156	625642	376030	349399	346891	334808	325447
	DROPTANKS - DRY	112486	112162						
	SUBTOTAL(DROPTANK INERT)	122716	122716						
	DROPTANK - RESIDUAL	10554	10554						
	DROPTANK PROP—ASCENT	3063218	0						
	SUBTOTAL-DROPTANK GROSS	3185934	122716						
	TOTAL-COMPOSITE VEHICLE	3816420	752874						

DESIGNATORS:

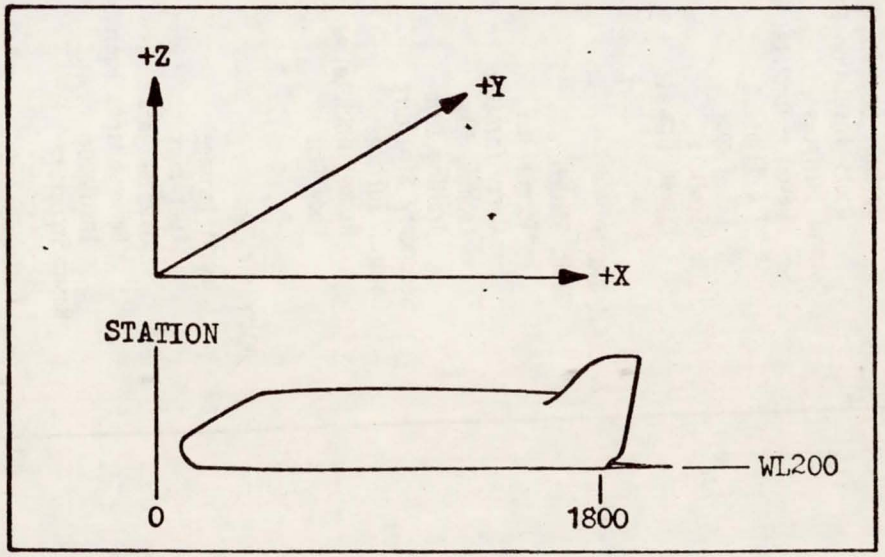
EVENTS
- A LAUNCH WEIGHT
- B BEFORE TANK SEPARATION
- C AFTER TANK SEPARATION
- D INJECTION
- E ON-ORBIT - DOCKING
- F PRE-RETRO
- G ENTRY
- H LANDING

NOTES & SKETCHES:

Table 2.14-8

Stage-and-One-Half LS 200-10 (55 deg, 270 nm)		Center of Gravity Inches			Moment of Inertia Million Slug Ft^2/1000			Product of Inertia Thousand Slug Ft^2/1000		
25K Payload	Weight (lb)	X	Y	Z	Roll	Pitch	Yaw	Roll-Pitch	Pitch-Yaw	Yaw-Roll
Launch	3816420	138		336	31498	347308	374618	0	0	+3662
Before Tank Separation	752874	1170	-2	326	6685	40125	43876	0	0	+1664
After Tank Separation	625642	1291	-3	316	4389	17438	20726	0	0	+638
Orbiter - 26% Ascent Burn	538297	1324	-3	320	3949	14064	17088	0	0	+1078
Orbiter - 52% Ascent Burn	484607	1324	-4	318	3571	13790	16462	0	0	+487
Orbiter - 78% Ascent Burn	430913	1318	-4	316	3180	13574	15881	0	0	+529
Orbiter - 100% Ascent Burn	386433	1298	-4	312	2820	13088	15048	0	0	-5
Injection	376030	1300	-5	312	2820	13032	14993	0	0	-6
Orbit - Docking	349399	1308	-2	308	2820	11274	14551	0	0	0
Pre-Retro	346891	1308	-2	308	2820	11272	14449	0	0	-15
Entry	334808	1310	0	310	2799	11230	14408	-	-	-
Landing	325447	1332	0	316	2857	10203	13322	-	-	-

Table 2.14-8



X = Roll Axis
Y = Pitch Axis
Z = Yaw Axis

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Table 2.14-9

DUE EAST	GROUP WEIGHT STATEMENT			Page 1 of 6	
CONFIGURATION	STAGE-AND-ONE-HALF LS 200	BY	SGB	DATE	3 May 1971

1. WING GROUP					N/A
		CARRY THROUGH	EXPOSED SURFACE		
Basic Structure					
Torque Box					
Leading Edge					
Trailing Edge					
Secondary Structures					
Variable Geometry (incl ____ lbs mechanism)					
Doors Insulation Fairings					
Control Surfaces		SURFACE	SUPT/MECH		
Elevon — (incl. bal wt. ____ lb)					
T.E. Flaps					
L.E. Flaps					
Spoilers					
Speed Brakes					
2. TAIL GROUP		VERTICAL	HORIZ		18050
Basic Structure					
Torque Box					
Carry Through					
Leading Edge					
Trailing Edge					
Secondary Structure					
Control Surfaces		SURFACE	SUPT/MECH		
Rudder (incl. bal wt 0 lb)					
Body Flap					
3. BODY GROUP					56676
Integral Tankage					
Fuel Tank					
Oxidizer Tank					
Between tanks (cmn blkhd)					
Insulation					
Basic Structure	FWD. (1857)	CTR. (18605)	AFT (6384)	SKIRT (23476)	50322
Sidewalls					
Bulkheads					
Partitions					
Thrust Structure (main ascent engine)					
Secondary Structure					6354
Crew Compartment					
Equipment Compartments					
Payload Attach & Deploy					
Speed Brakes					
Engine Heat Protection					
Interstage (incl. mech. ____ lbs)					
Doors/fairings					
Gear/wing provisions/ABES					
Contingency					

Table 2.14-9 (Cont'd)

DUE EAST	GROUP WEIGHT STATEMENT					Page 2 of 6	
CONFIGURATION	STAGE-AND-ONE-HALF LS 200-10			By	SGB	Date	3 May 1971
							35716
4. INDUCED ENVIRONMENTAL PROTECTION							
Thermal Protection	WING	HOR.TAIL	VER. TAIL	BODY			
Leading Edge/Nose Cap							
Surface Protection							
Base Heat Protection							
Sound Protection							
Meteorite / Radiation Protection							
Contingency							

TABLE 2.14-9 (Cont'd)

DUE EAST	GROUP WEIGHT STATEMENT		Page 3 of 6
CONFIGURATION	STAGE-AND-ONE-HALF LS 200-1B	BY SGB	Date 3 May 1971

	ROLLING	STRUCTURE	CONTROLS		
5. LANDING, DOCKING					11988
Alighting Gear					
Main					
Nose					
Docking					
Auxiliary Systems					
Deceleration chutes					
Flotation gear					
Handling gear					
Contingency					
6. PROPULSION – MAIN ASCENT					105858
Engine & Accessories					
Engine (as supplied)					
Gimbal System					
Ignition and Control System					
Propellant Utilization System					
Accessories					
Installation, Ducts, Shrouds					
Propellant System					
Purge					
Pressurization					
Fill & Vent					
PCV System					
Pre-Valves					
Feed Systems					
Vortex, Flow Control System					
Supports and Install					
Tankage – Nonintegral					
Fuel					
Oxidizer					
Contingency					
7. PROPULSION – CRUISE BACK					-0-
Engine & Accessories					
Engine					
Ignition and Control Sy					
Lubrication Sy (dry)					
Accessories					
Installation, Ducts, Shroud					
Air Induction					
Engine Mounting					
Nacelles Pylons (incl ____ lb mech)					
Exhaust System					
Propellant System					
Fill Drain					
Pressurization (dry)					
Vent System					
Pump					
Feed System					
Transfer System					
Dump System					
Supports/Installation					
Tankage – Non Integral					
Fuel					
Contingency					

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Table 2.14-9 (Cont'd)

DUE EAST	GROUP WEIGHT STATEMENT						Page	4 of 6
CONFIGURATION	STAGE-AND-ONE-HALF LS 200 By 10						Date	3 May 1971
8. PROPULSION – AUXILIARY								6901
Thruster Installation								
Thruster								
Accessory								
Propellant System								
Fill/Drain/Vent Lines & Valves								
Pressurization								
Feed System/Accumulators								
Conditioning								
Supports								
Tankage								
Tanks								
Insulation								
Supports								
Contingency								
9. PRIME POWER	POWER UNIT	MTG INSTAL	PROPEL TANK/SYS	TEMP CONTROL	CON-TROLS	TOTAL		1620
Batteries								
Engine Turbine								
Fuel Cells								
Contingency								
10. ELECTRICAL	SUPPLY	CONVER-SION	CONTROL UNITS					3704
Equipment								
Distribution and Control Circuitry								
Utility Systems								
Supports/Installation								
Contingency								
11. HYDRAULIC								2073
Power Supply								
Distribution Control Ctr								
Temperature Control Sy								
Auxiliary Systems								
Supports/Installation								
Contingency								
12. SURFACE CONTROLS								4246
Cockpit Controls								
Flight Control System								
System Actuation	CONTROLS	POWER XMISSION	ACTU-ATOR	FEEL SY	SUPTS INSTALL	(SEE AVIONICS)		
Aileron								
Elevator								
Rudder								
T.E. Flap								
Speed Brake								
Contingency								
13. AVIONICS	UNITS	CIRCUITRY	COOLING	ANTENNAS	INSTALL.			3762
Guid/Nav								
Flight Control								
Data Management								
Communicate								
Config. Seq.								
Instrumentation								
Displays								
Contingency								

Table 2.14-9 (Cont'd)

DUE EAST	GROUP WEIGHT STATEMENT						Page 5 of 6
CONFIGURATION	STAGE-AND-ONE-HALF LS 200				By SGB		Date 3 May 1971
14. ENVIRONMENTAL CONTROL							1274
Gas Supply System (dry)							
Gas Management, Processing (dry)							
Heat Transport System (dry)							
Water Management System (dry)							
Purge System							
Insulation							
Contingency							
15. PERSONNEL PROVISIONS							210
Seats/Restraint Sys (No =)							
Fixed Life Support Equipment							
Emergency Equipment							
Cargo Handling							
Furnishings							
16. RANGE SAFETY AND ABORT							0
17. BALLAST							0
18. GROWTH/UNCERTAINTY							17794
19. OPEN							
SUBTOTAL (Dry Weight)							269872
20. PERSONNEL	NO	MAN	GARMENTS	HELMET	PRESS. SUIT	ACCESSORIES	725
Crew ( )							
Passenger( )							
Personal Gear/Accessories							
Life Support							
Food							
Equipment - Portable							
21. CARGO - (65K TO ORBIT 40K RETURN)							65000
22. ORDNANCE							0
23. RESIDUAL & UNUSABLE FLUIDS							3045
Ascent						483	
Cruise						0	
Maneuver (Incl. All Aux. Tk. Residuals)						183	
Attitude Control						94	
ECS						120	
EPS						365	
Hydraulic						1800	
Misc (Shock struts, etc.)							
24. OPEN							
SUBTOTAL(Inert Weight)							338642

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Table 2.14-9 (Cont'd)

DUE EAST	GROUP WEIGHT STATEMENT		Page 6 of 6
CONFIGURATION	STAGE-AND-ONE-HALF LS 200-10 By	SGB	Date 3 May 1971
25. RESERVE FLUIDS			8414
Ascent		7674	
Cruise		0	
Maneuver		0	
Attitude Control		564	
ECS		10	
EPS		140	
Hydraulic		76	
26. INFLIGHT LOSSES			9504
Ascent		7435	
Cruise		0	
Maneuver (Incl. All Aux. Tk Boiloff)		1224	
Attitude Control		0	
ECS		145	
EPS		700	
Hydraulic			
27. PROPELLANT - ASCENT			239662
28. PROPELLANT - CRUISE			-0-
	MANEUV.	ACS	
29. PROPELLANT - MANEUVER/ACS	12589	5637	18226
TOTAL (Gross Weight)			614448

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Table 2.14-10

DUE EAST — MISSION WEIGHT SUMMARY

CONFIGURATION: STAGE-AND-ONE-HALF LS 200-10 | BY: Broadhead | DATE: 3 May '71

CODE	SYSTEM	A	B	C	D	E	F	G	H
1	WING GROUP								
2	TAIL GROUP	18050							
3	BODY GROUP	56676							
4	INDUCED ENVIR PROTECTION	35716							
5	LANDING, RECOVERY, DOCKING	11988							
6	PROPULSION - ASCENT	105858							
7	PROPULSION - CRUISE	-0-							
8	PROPULSION - AUXILIARY	6901							
9	PRIME POWER	1620							
10	ELECT CONVER & DISTR	3704							
11	HYDRA CONVER & DISTR	2073							
12	SURFACE CONTROLS	4246							
13	AVIONICS	3762							
14	ENVIRONMENTAL CONTROL	1274							
15	PERSONNEL PROVISIONS	210							
16	RANGE SAFETY & ABORT	-0-							
17	BALLAST	-0-							
18	GROWTH/UNCERTAINTY	17794							
19									
	SUBTOTAL (DRY WEIGHT)	269872	269872	269872	269872	269872	269872		
20	PERSONNEL	725	725	725	725	725	725		
21	CARGO	65000	65000	65000	65000	65000	40000		
22	ORDNANCE	-0-							
23	RESIDUAL FLUIDS	3045	3045	3045	3045	3045	3045		
24									
	SUBTOTAL(INERT WEIGHT)	338642	338642	338642	338642	338642	313642	313642	313642
25	RESERVE FLUIDS	8414	8414	8414	790	790	790	790	790
26	IN FLIGHT LOSSES	9504	9176	3908	2413	2338	574	549	0
27	PROPELLANT - ASCENT	239,662	239662	239662	-0-	-0-	-0-	-0-	-0-
28	PROPELLANT - CRUISE	-0-	-0-	-0-	-0-	-0-	-0-	-0-	-0-
29	PROPELLANT — MANEUV/ACS	18226	18226	18226	18226	8529	7783	464	-0-
30	SUBTOTAL-ORBITER GROSS	614448	614120	608852	360071	350299	322789	315445	314432
	DROPTANKS - DRY	112162	112162						
	DROPTANKS - RESIDUAL	10554	10554						
	SUBTOTAL-DROPTANK INERT	122716	122716						
	DROPTANK PROP - ASCENT	2713668	-0-						
	SUBTOTAL DROPTANK GROSS	2836384	122716						
	TOTAL COMPOSITE VEHICLE	3450832	736836						

DESIGNATORS:

EVENTS

A LAUNCH WEIGHT
B BEFORE TANK SEPARATION
C AFTER TANK SEPARATION
D INJECTION
E ON-ORBIT - DOCKING
F PRE-RETRO
G ENTRY
H LANDING

NOTES & SKETCHES:

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Table 2.14-11

SOUTH POLAR	GROUP WEIGHT STATEMENT			Page 1 of 6	
CONFIGURATION	STAGE-AND-ONE-HALF LS 200-10	BY	SGB	DATE	3 May 1971

1. WING GROUP					N/A
		CARRY THROUGH	EXPOSED SURFACE		
Basic Structure					
Torque Box					
Leading Edge					
Trailing Edge					
Secondary Structures					
Variable Geometry (incl ____ lbs mechanism)					
Doors Insulation Fairings					
Control Surfaces		SURFACE	SUPT/MECH		
Elevon — (incl. bal wt. ____ lb)					
T.E. Flaps					
L.E. Flaps					
Spoilers					
Speed Brakes					
2. TAIL GROUP		VERTICAL	HORIZ		18050
Basic Structure					
Torque Box					
Carry Through					
Leading Edge					
Trailing Edge					
Secondary Structure					
Control Surfaces		SURFACE	SUPT/MECH		
Rudder (incl. bal wt 0 lb)					
Body Flap					
3. BODY GROUP					56676
Integral Tankage					
Fuel Tank					
Oxidizer Tank					
Between tanks (cmn blkhd)					
Insulation					
Basic Structure	FWD.	CTR.	AFT.	SKIRT	
	(1857)	18605	6384	23476	50322
Sidewalls		12574			
Bulkheads		1100			
Partitions		4931			
Thrust Structure (main ascent engine)					
Secondary Structure					6354
Crew Compartment					
Equipment Compartments					
Payload Attach & Deploy					
Speed Brakes					
Engine Heat Protection					
Interstage (incl. mech. ____ lbs)					
Doors/fairings					
Gear/wing provisions/ABES					
Contingency					

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Table 2.14-11 (Cont'd)

SOUTH POLAR	GROUP WEIGHT STATEMENT			Page	2 of 6
CONFIGURATION	STAGE-AND-ONE-HALF LS 200-10	By	SGB	Date	3 May 1971

35716

4. INDUCED ENVIRONMENTAL PROTECTION

	WING	HOR. TAIL	VER. TAIL	BODY	
Thermal Protection					
Leading Edge/Nose Cap*					
Surface Protection*					
Base Heat Protection*					
Sound Protection					
Meteorite / Radiation Protection					
Contingency					

*Break out by type of material

Table 2.14-11 (Cont'd)

SOUTH POLAR	GROUP WEIGHT STATEMENT		Page 3 of 6
CONFIGURATION	STAGE-AND-ONE-HALF LS 200-10	By SGB	Date 3 May 1971

	ROLLING	STRUCTURE	CONTROLS		
5. LANDING, DOCKING					11988
Alighting Gear					
Main					
Nose					
Docking					
Auxiliary Systems					
Deceleration chutes					
Flotation gear					
Handling gear					
Contingency					
6. PROPULSION – MAIN ASCENT					105858
Engine & Accessories					
Engine (as supplied)					
Gimbal System					
Ignition and Control System					
Propellant Utilization System					
Accessories					
Installation, Ducts, Shrouds					
Propellant System					
Purge					
Pressurization					
Fill & Vent					
PCV System					
Pre-Valves					
Feed Systems					
Vortex, Flow Control System					
Supports and Install					
Tankage – Nonintegral					
Fuel					
Oxidizer					
Contingency					
7. PROPULSION – CRUISE BACK					0
Engine & Accessories					
Engine					
Ignition and Control Sy					
Lubrication Sy (dry)					
Accessories					
Installation, Ducts, Shroud					
Air Induction					
Engine Mounting					
Nacelles Pylons (incl ____ lb mech)					
Exhaust System					
Propellant System					
Fill Drain					
Pressurization (dry)					
Vent System					
Pump					
Feed System					
Transfer System					
Dump System					
Supports/Installation					
Tankage – Non Integral					
Fuel					
Contingency					

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Table 2.14-11 (Cont'd)

SOUTH POLAR GROUP WEIGHT STATEMENT						Page 4 of 6	
CONFIGURATION	STAGE-AND-ONE-HALF LS 200-100				SGB	Date	3 May 1971
8. PROPULSION – AUXILIARY							6901
Thruster Installation							
Thruster							
Accessory							
Propellant System							
Fill/Drain/Vent Lines & Valves							
Pressurization							
Feed System/Accumulators							
Conditioning							
Supports							
Tankage							
Tanks							
Insulation							
Supports							
Contingency							
9. PRIME POWER	POWER UNIT	MTG INSTAL	PROPEL TANK/SYS	TEMP CONTROL	CON-TROLS	TOTAL	1620
Batteries							
Engine Turbine							
Fuel Cells							
Contingency							
10. ELECTRICAL		SUPPLY	CONVER-SION	CONTROL UNITS			3704
Equipment							
Distribution and Control Circuitry							
Utility Systems							
Supports/Installation							
Contingency							
11. HYDRAULIC							2073
Power Supply							
Distribution Control Ctr							
Temperature Control Sy							
Auxiliary Systems							
Supports/Installation							
Contingency							
12. SURFACE CONTROLS							4246
Cockpit Controls							
Flight Control System							
System Actuation	CONTROLS	POWER XMISSION	ACTU-ATOR	FEEL SY	SUPTS INSTALL	(SEE AVIONICS)	
Aileron							
Elevator							
Rudder							
T.E. Flap							
Speed Brake							
Contingency							
13. AVIONICS	UNITS	CIRCUITRY	COOLING	ANTENNAS	INSTALL.		3762
Guid/Nav							
Flight Control							
Data Management							
Communicate							
Config. Seq.							
Instrumentation							
Displays							
Contingency							

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Table 2.14-11 (Cont'd)

SOUTH POLAR	GROUP WEIGHT STATEMENT							Page	5 of 6
CONFIGURATION	STAGE-AND-ONE-HALF LS 200-10					By	SGB	Date	3 May 1971

14. ENVIRONMENTAL CONTROL								1274
Gas Supply System (dry)							____	
Gas Management, Processing (dry)							____	
Heat Transport System (dry)							____	
Water Management System (dry)							____	
Purge System							____	
Insulation							____	
Contingency							____	
15. PERSONNEL PROVISIONS								210
Seats/Restraint Sys (No =)							____	
Fixed Life Support Equipment							____	
Emergency Equipment							____	
Cargo Handling							____	
Furnishings							____	
16. RANGE SAFETY AND ABORT								0
17. BALLAST								0
18. GROWTH/UNCERTAINTY								17794
19. OPEN								
SUBTOTAL (Dry Weight)								269872
20. PERSONNEL	NO	MAN	GARMENTS	HELMET	PRESS. SUIT	ACCESSORIES		725
Crew ( )	____	____	____	____	____	____	____	
Passenger( )	____	____	____	____	____	____	____	
Personal Gear/Accessories							____	
Life Support							____	
Food						____		
Equipment - Portable						____		
21. CARGO								40000
22. ORDNANCE								0
23. RESIDUAL & UNUSABLE FLUIDS								3045
Ascent							____	
Cruise							____	
Maneuver (Incl. All Aux. Tk. Residuals)							____	
Attitude Control							____	
ECS							____	
EPS							____	
Hydraulic							____	
Misc (Shock struts, etc.)							____	
24. OPEN								
SUBTOTAL(Inert Weight)								313642

Table 2.14-11 (Cont'd)

SOUTH POLAR	GROUP WEIGHT STATEMENT		Page 6 of 6
CONFIGURATION	STAGE-AND-ONE-HALF LS 200-BWO	SGB	Date 3 May 71

25. RESERVE FLUIDS		8153
Ascent		
Cruise		
Maneuver		
Attitude Control		
ECS		
EPS		
Hydraulic		
26. INFLIGHT LOSSES		9504
Ascent		
Cruise		
Maneuver (Incl. All Aux. Tk Boiloff)		
Attitude Control		
ECS		
EPS		
Hydraulic		
27. PROPELLANT - ASCENT		239889
28. PROPELLANT - CRUISE		0
	MANEUV. / ACS	
29. PROPELLANT - MANEUVER/ACS		16981
TOTAL (Gross Weight)		588169

LOCKHEED

Table 2.14-12

SOUTH POLAR — MISSION WEIGHT SUMMARY

CONFIGURATION: STAGE-AND-ONE-HALF LS 200-10 | BY: Broadhead | DATE: 3 May 71

CODE	SYSTEM	A	B	C	D	E	F	G	H
1	WING GROUP								
2	TAIL GROUP	18050							
3	BODY GROUP	56676							
4	INDUCED ENVIR PROTECTION	35716							
5	LANDING, RECOVERY, DOCKING	11988							
6	PROPULSION - ASCENT	105858							
7	PROPULSION - CRUISE	0							
8	PROPULSION - AUXILIARY	6901							
9	PRIME POWER	1620							
10	ELECT CONVER & DISTR	3704							
11	HYDRA CONVER & DISTR	2073							
12	SURFACE CONTROLS	4246							
13	AVIONICS	3762							
14	ENVIRONMENTAL CONTROL	1274							
15	PERSONNEL PROVISIONS	210							
16	RANGE SAFETY & ABORT								
17	BALLAST								
18	GROWTH/UNCERTAINTY	17794							
19									
	SUBTOTAL (DRY WEIGHT)	269872							
20	PERSONNEL	725							
21	CARGO	40000							
22	ORDNANCE								
23	RESIDUAL FLUIDS	3045							
24									
	SUBTOTAL(INERT WEIGHT)	313642	313642	313642	313642	313642	313642	313642	313642
25	RESERVE FLUIDS	8153	8153	8153	757	757	757	757	757
26	IN FLIGHT LOSSES	9504	9176	3908	2331	2256	492	467	0
27	PROPELLANT - ASCENT	239889	239889	239889	0	0	0	0	0
28	PROPELLANT - CRUISE	0	0	0	0	0	0	0	0
29	PROPELLANT — MANEUV/ACS	16981	16981	16981	16981	7993	7247	464	0
30	SUBTOTAL-ORBITER GROSS	588169	587841	582573	333711	324648	322138	315330	314399
	DROPTANKS - DRY	112162	112162						
	DROPTANKS - RESIDUALS	10554	10554						
	SUBTOTAL-DROPTANK INERT	122716	122716						
	DROPTANK PROP-ASCENT	2877017	0						
	SUBTOTAL DROPTANK GROSS	2999733	122716						
	TOTAL COMPOSITE VEHICLE	3587902	710557						

DESIGNATORS:

EVENTS
- A LAUNCH WEIGHT
- B BEFORE TANK SEPARATION
- C AFTER TANK SEPARATION
- D INJECTION
- E ON-ORBIT - DOCKING
- F PRE-RETRO
- G ENTRY
- H LANDING

NOTES & SKETCHES:

2.15 DROPTANK

The unique feature of the stage-and-one-half system is the use of expendable droptanks rather than a reusable booster vehicle. Questions of intense concern regarding those droptanks are: (1) What do they cost to develop and to produce? (2) How much do they weigh? (3) Should they be expended or recovered and reused? and (4) What is the risk of having the droptanks impact land masses, ships, or aircraft and thereby cause undesired damage and/or casualties. These factors were all examined during the study and results favorable to the stage-and-one-half concept were obtained in all cases. The results of these analyses are presented in the following sections.

2.15.1 Droptank Recovery Analysis

In examining droptank recovery, the tank configuration and nominal staging conditions associated with the LS 200-1 high crossrange baseline are used to establish a recovery concept and to define system cost and weight trades. The approach to the analysis is based on the following steps:

(1) Define general requirements for tank recovery based on the assumed trajectory and staging conditions.

(2) Determine options for droptank recovery configuration and support operations.

(3) Select a baseline recovery concept, including definition of operational, technical, and programmatic aspects.

(4) Determine total costs for this baseline, including operations, recovery equipment, and costs resulting from additional tank and vehicle weight.

(5) Make a cost comparison of expendable and recoverable tanks as a function of number of flights, program years, percent of damage, number of tank uses and basic tank cost ($/lb).

(6) Evaluate the desirability of tank recovery and finalize conclusions.

2.15.1.1 Droptank Recovery Options. Figure 2.15-1 presents some of the mission, tank configuration, and recovery equipment options considered in selecting the baseline concept. The tank recovery configurations are shown in more detail in Fig. 2.15-2.

Potential concepts for effecting tank recovery include:

(1) Vee tanks with aero surfaces.

(2) Vee tanks with aero deceleration.

(3) Separate tanks with rotors.

(4) Separate tanks with aero deceleration (baseline concept).

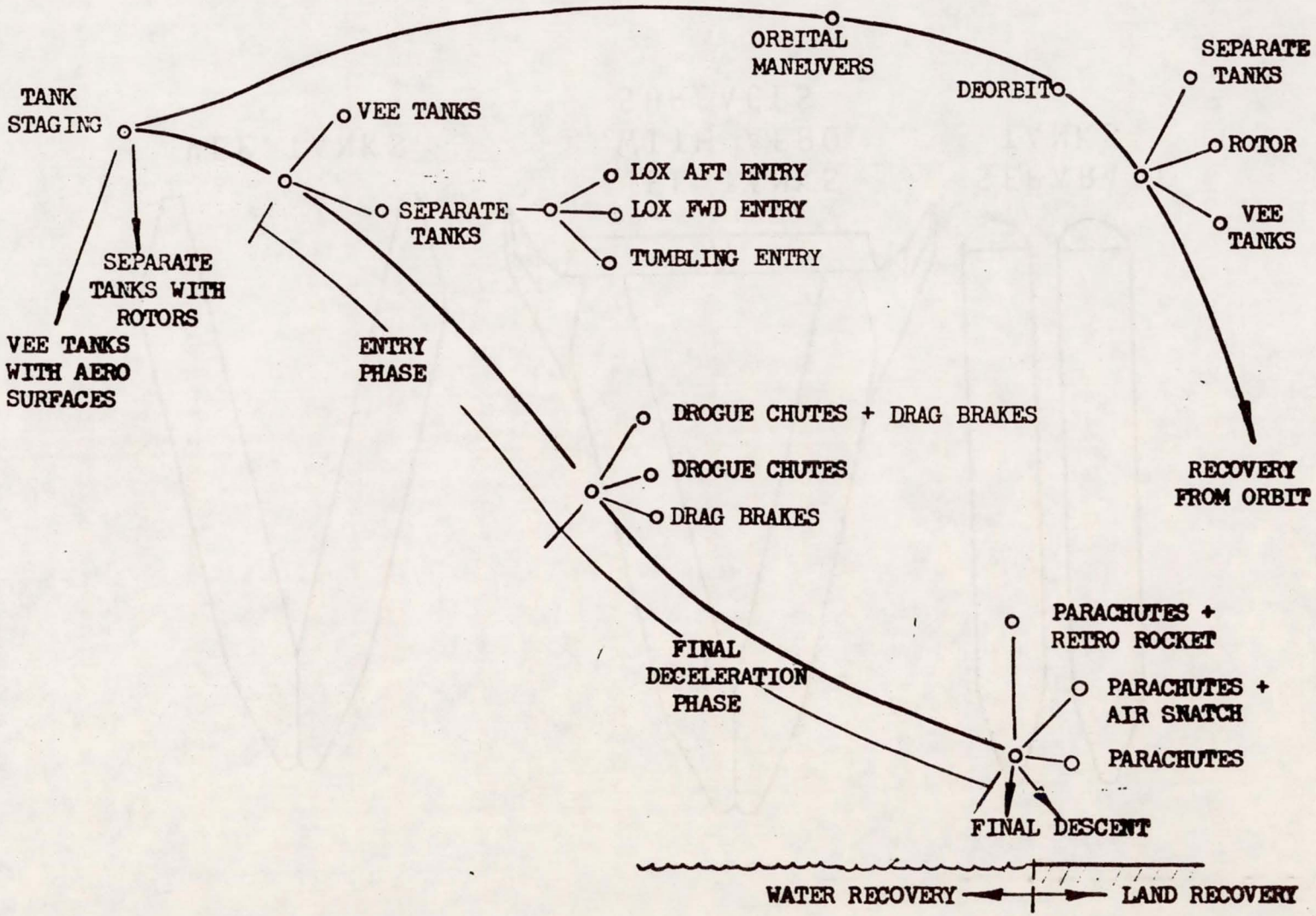


Fig. 2.15-1 Droptank Recovery Options

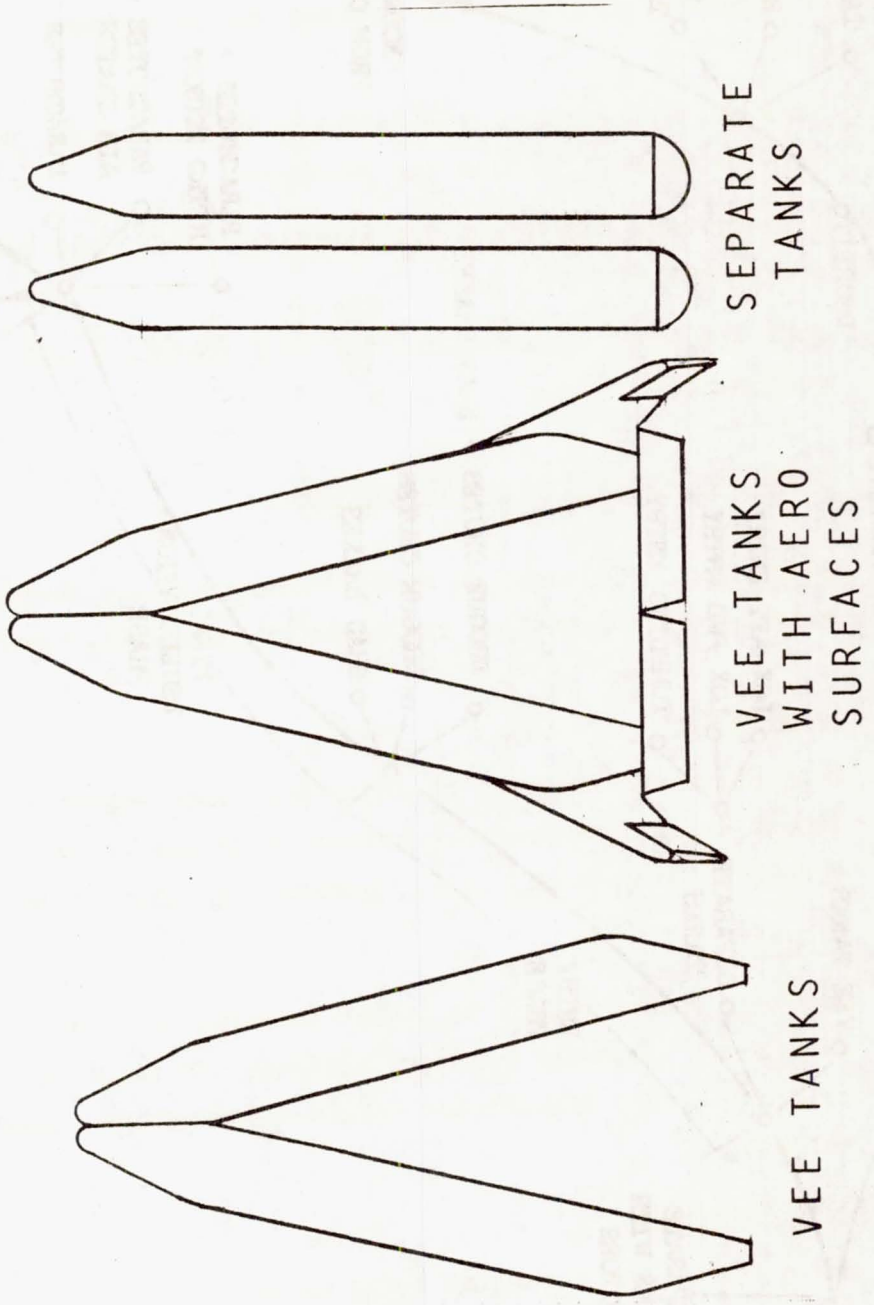


Fig. 2.15-2 Tank Recovery Configurations

FIG. 2.15-2

Tank recovery can be initiated at the staging point or can be accomplished from orbit. Recovery from orbit introduces the following additional system requirements:

(1) Tank main propulsion capability to attain orbital velocity and to provide deorbit impulse for recovery.

(2) Primary tank subsystems for power, guidance, attitude control and communications.

(3) More stringent environments for tank structure and recovery equipment.

2.15.1.2 Baseline Derivation and Selection.

Baseline Selection. The baseline tank recovery concept (separate tanks with aero deceleration) is selected on the basis of qualitative evaluation.

- Complexity

  Separation and recovery of individual tank halves allows lower landed weight per recovery system, thus smaller and simpler aero deceleration devices; i.e., parachutes, drag brakes, etc.

- Weight

  Recovery system weight is lowest compared to other concepts examined. Separate recovery equipment is required for each tank half but is not significant weight disadvantage compared to other methods.

- Cost Reduction

  Has a high reuse potential being exceeded only by the flyback tank concept (Vee tanks with aero surfaces). Lower landed weight per tank half and tank landing geometry tend to reduce the probability of landing impact damage.

- Ease of Recovery

  Does require separate recovery operations for each tank half, but reduced weight and tank shape provides easier handling and transportation for ground or water recovery.

- Development

  Considerable previous development on attitude control, parachutes, thermal protection, etc., provides minimum risk and cost of all candidates examined.

Staging and Entry Environments. Because of the additional requirements associated with recovery from orbit, the baseline is predicated on recovery from the staging point. Previous work (Ref. 2.15-1) on tank staging dynamics indicates typical pitch rates of 8-10 deg per second will be imparted to the tanks at separation. Yaw plane rates of 2-3 deg per second will also result when the vee tanks are separated at the apex for separate recovery. Heating and loads resulting from these initial conditions are expected to cause tank breakup during entry (Ref. 2.15-1). In an effort to achieve acceptable entry conditions, two tank entry modes are examined: (1) LOX tank forward, (2) LOX tank aft. In each case, a symmetrical tank (body of revolution) at α = 0 deg with zero yaw and pitch rates is assumed. For the LOX aft case, the LH_2 tank bulkhead is assumed to be ellipsoidal ($\sqrt{2}$:1). Temperature and heat rate histories for these cases are shown in Figs. 2.15-3 and 2.15-4. Basic requirements are drawn from these data.

Requirements. Basic requirements for the baseline concept are:

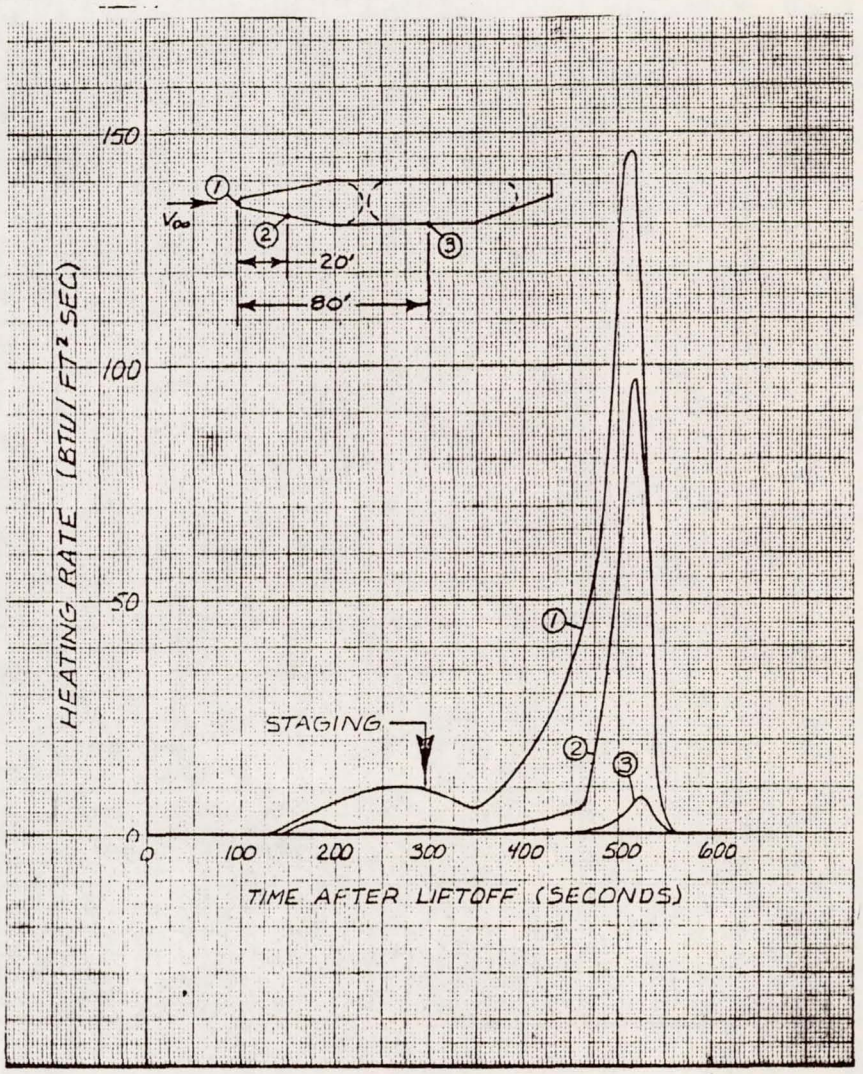
(1) Droptank aero stability during entry to limit structural loads without the addition of excessive structural weight.

(2) Capability for removing tank separation rates resulting from staging.

(3) Adequate tank thermal protection to limit entry heating.

(4) Capability for tank entry and landing deceleration from normal staging velocities.

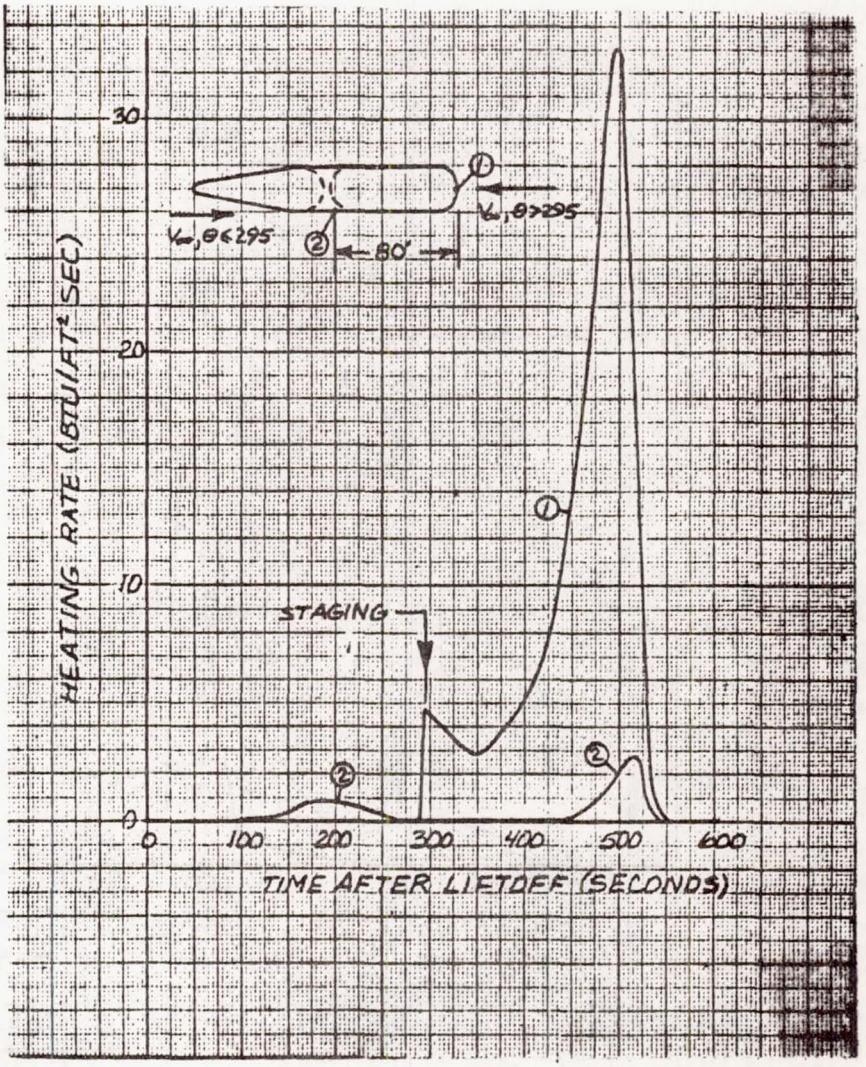
HEATING RATES TO VARIOUS LOCATIONS



REF: 2.15-2

Fig. 2.15-3 Droptank Recovery

HEATING RATES TO VARIOUS LOCATIONS



REF: 2.15-2

Fig. 2.15-4 Droptank Recovery

(5) Maintenance of tank structural integrity during entry and landing impact.

(6) Capability for tank disassembly for refurbishment, repair and replacement of major structural assemblies and equipment.

2.15.1.3 Recovery System Baseline Description.

Flight Operations. Figure 2.15-5 presents the flight sequence of events for droptank recovery from the staging point.

Following droptank separation from the spacecraft, the vee tanks are separated at the apex. The stabilization and control system on each tank half operates to remove all pitch and yaw rates produced by separation. At 250,000 ft, 130 sec after staging, the tanks are oriented for ballistic entry (α = 0 deg) with the blunt end forward (LOX tank aft). At 62,000 ft ($\bar{q}$ = 338 psf), the drag brakes are deployed to reduce the tank inertial velocity to 815 fps at 37,000 ft where two drogue chutes are also deployed to further reduce the tank velocity to approximately 125 fps at 25,000 ft, (Fig. 2.15-4). At this altitude, the drogue chutes are separated, drag brakes are closed, and the main chutes are deployed for final descent (Fig. 2.15-6). At an altitude of 20 ft above ground level, the radar altimeter activates the retro rocket firing circuit to fire the solid rocket motor, providing zero velocity at touchdown. For land recovery, the baseline concept provides impact attenuation by means of inflatable bags.

Recovery System Design.

Tank Thermal Protection. For entry thermal protection, a cork ablator bonded directly to the tank surface is assumed (Ref. 2.15-2). Thermal protection weights are based on a cork density of 30 pcf.

Tank Stabilization and Control. During entry, the following equipment provides tank stability and control: (1) Inertial reference package, (2) flight control electronics package, and (3) attitude control system. For attitude control,

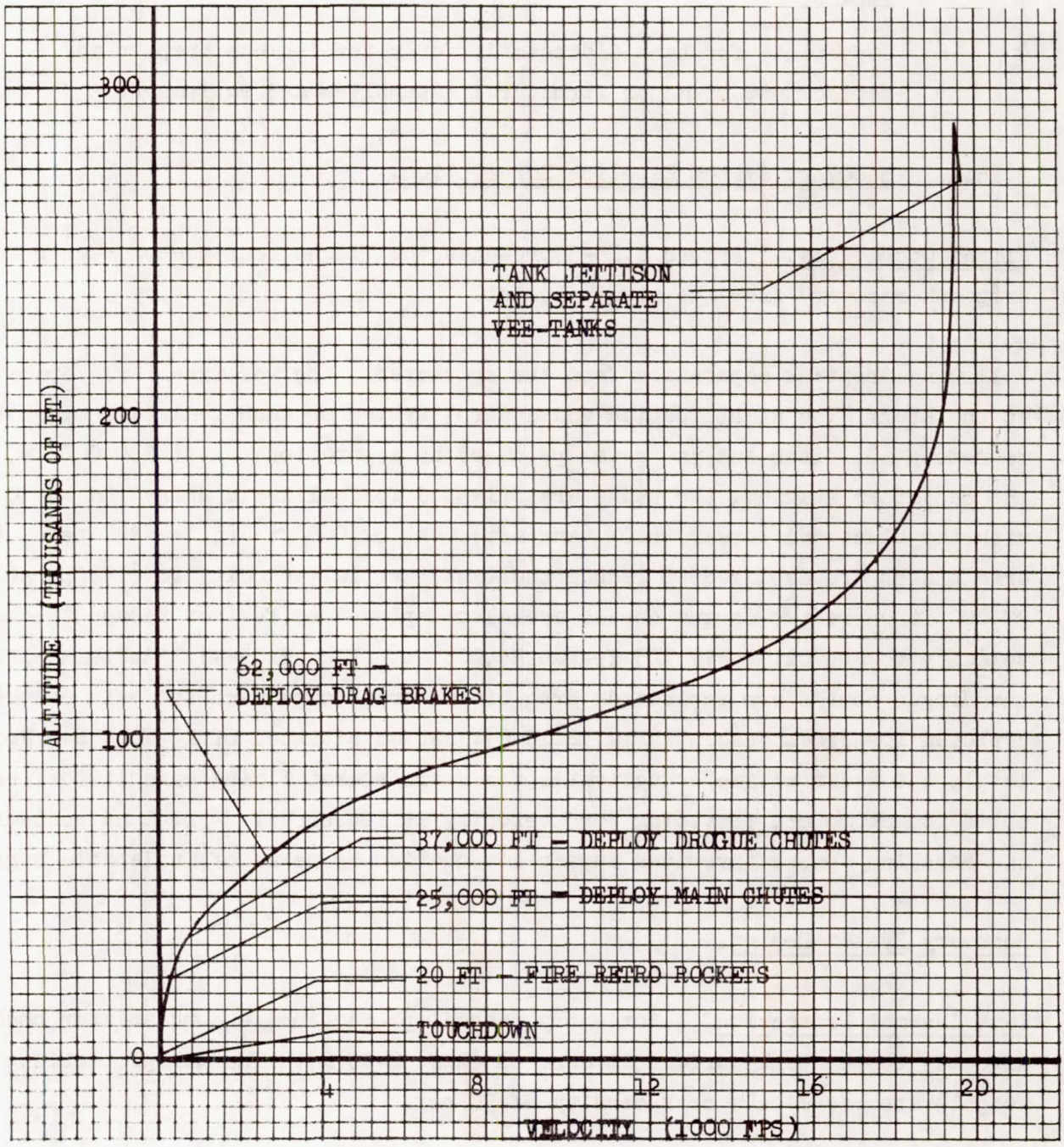


Fig. 2.15-5 Droptank Recovery Sequence

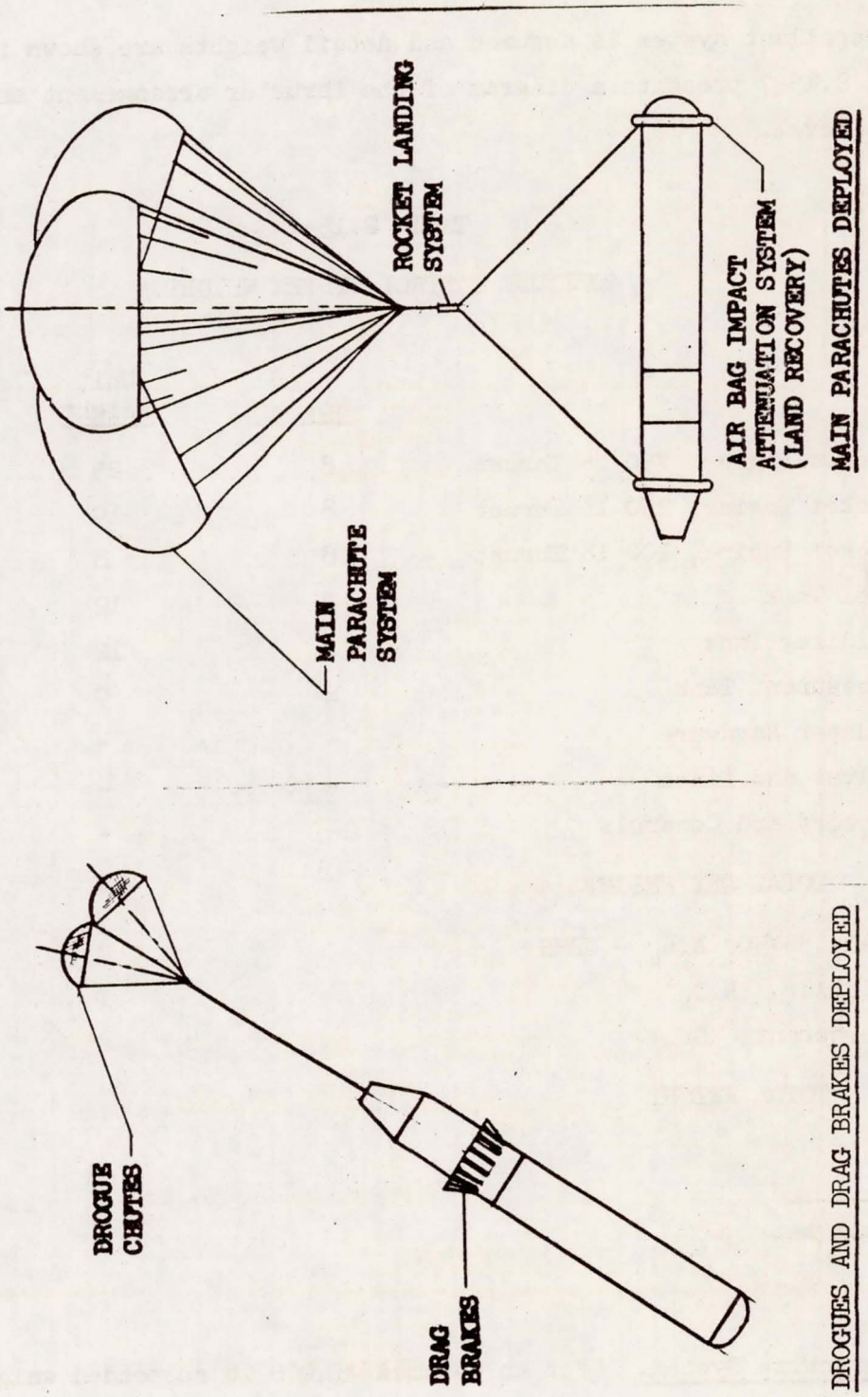


Fig. 2.15-6 Tank Recovery Configurations

FIG. 2.15-6

a bipropellant system is assumed and detail weights are shown in Table 2.15-1. Figure 2.15-7 presents a diagram of the thruster arrangement and required thrust level.

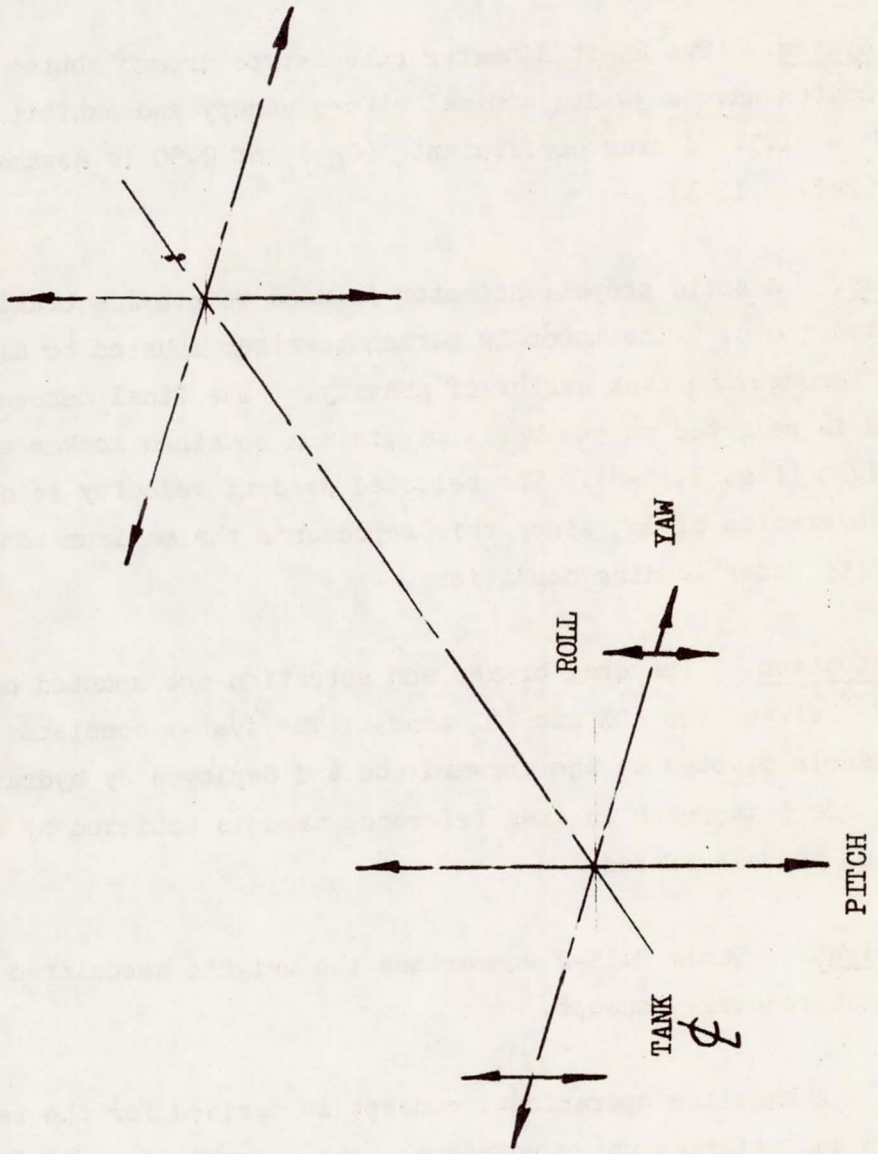
Table 2.15-1

ATTITUDE CONTROL SYSTEM WEIGHTS*

	No.	Unit Weight	Total Weight
1. Rocket Engine, 750 lb Thrust	8	25	200
2. Rocket Engine, 250 lb Thrust	8	19	152
3. Rocket Engine, 100 lb Thrust	8	8	64
4. Fuel Tank	2	12	24
5. Oxidizer Tank	2	12	24
6. Pressurant Tank	2	9	18
7. Cluster Hardware	-	-	200
8. Valves and Lines	-	-	125
9. Sensors and Controls	-	-	15
TOTAL DRY WEIGHT			822
Fuel, 50/50 N_2H_4 - UDMH			286
Oxidizer, N_2O_4			563
Pressurant, GN_2			55
TOTAL WEIGHT			1,726

*Per tank set

Main Parachute System. For an assumed 75,000 lb suspended weight, canopy area becomes extremely large for descent velocities <40 fps. To minimize parachute development and deployment problems, canopy diameter is limited to <150 ft.



	NO. THRUSTERS	THRUST LEVEL (LB)
PITCH	4	750
YAW	4	250
ROLL	4	100

Fig. 2.15-7 Attitude Control System

FIG. 2.15-7

Extended skirt-type chutes are assumed for tank recovery, since this type of chute is in current use for air drop applications and has demonstrated adequate opening reliability for canopy diameters up to 100 ft.

Drogue Parachute System. Two 25-ft diameter ribbon-type drogue chutes are assumed. These chutes have a 30 deg conical ribbed canopy and exhibit good stability at $<M = 1.3$. A drag coefficient (C_{D_C}) of 0.50 is assumed for this application (Ref. 2.15-1).

Retro-Rocket System. A solid propellant motor is used to provide braking thrust for final touchdown. The motor is parachute-riser mounted to align the thrust vector through the tank center of gravity. The final descent velocity of 60 fps is selected on the basis of minimum combined rocket and main parachute weight (Fig. 2.15-8). The selected descent velocity is coupled with a landing deceleration of 6g, since this represents the maximum tank structural capability under landing conditions.

Drag Brakes and Actuation. The drag brakes and actuation are mounted on the intertank structure between the LOX and LH_2 tanks. The system consists of 25 three-ft wide panels pivoted at the forward end and deployed by hydraulic actuation. A 65 percent increase in drag reference area is achieved by a 30 deg deployment of these surfaces.

Recovery System Weight. Table 2.15-2 summarizes the weights associated with the baseline droptank recovery concept.

Ground Operations. A baseline operational concept is derived for the recoverable tank subsystem and utilizes water retrieval with a combination of "flood deck" loading ships and coastal barges, and tank repair at the Integrated Main Base.

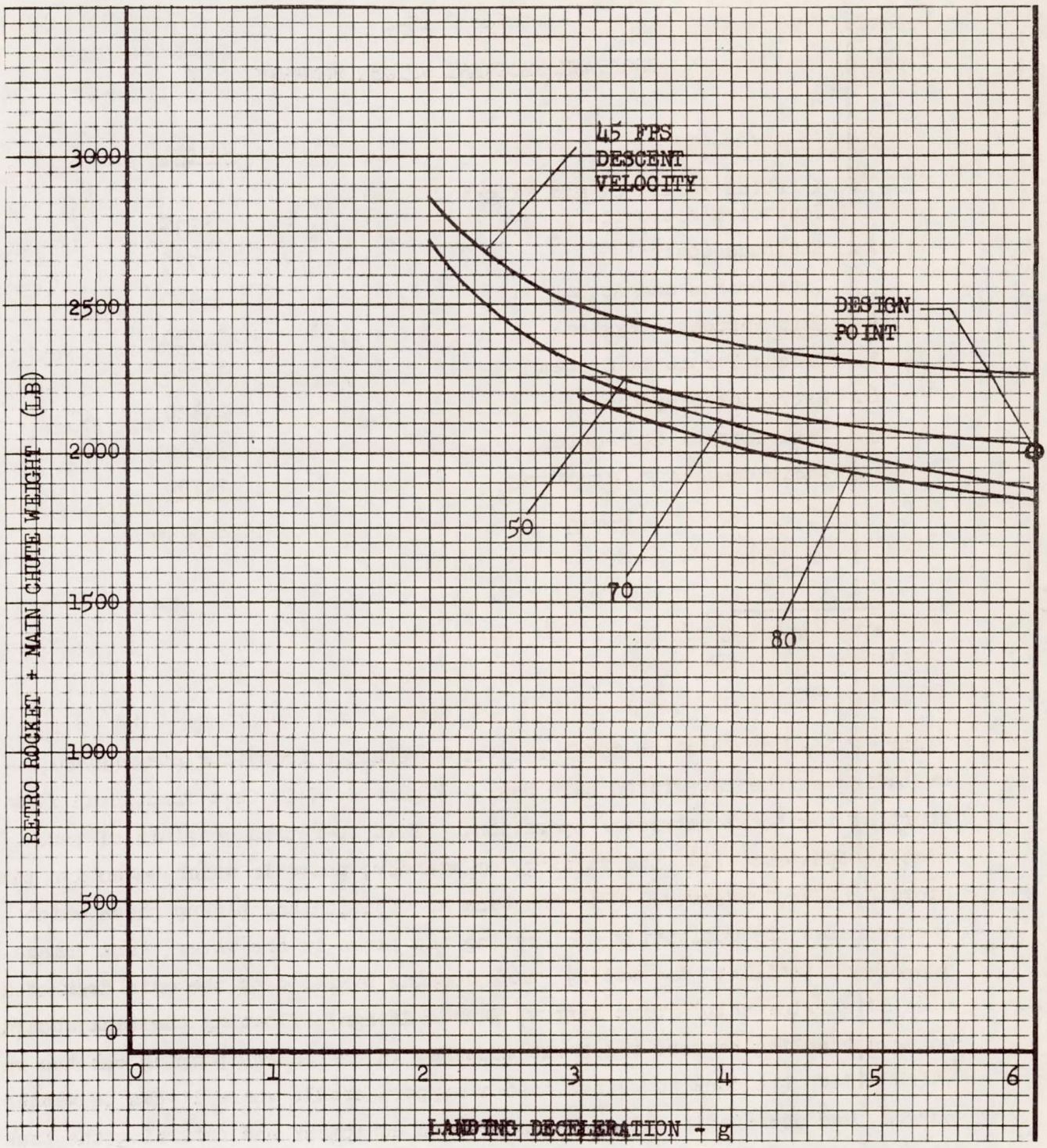


Fig. 2.15-8 Variation of Retro Rocket and Main Parachute Weight With Landing Deceleration

Table 2.15-2

DROPTANK RECOVERY WEIGHT SUMMARY

		Weight
1.	Inertial Reference and Flight Control Electronics	274
2.	Attitude Control	1,726
3.	Drag Brakes and Actuation	5,400
4.	Main Parachute System	1,930
5.	Drogue Parachute System	1,788
6.	Retrorocket System	2,420
7.	Tank Thermal Protection	13,600
8.	Additional Tank Structure	2,790
	TOTAL	29,928*

*Per tank set

Figure 2.15-9 depicts the total recovery operation from tank staging to unloading at the integrated Main Base. Figure 2.15-10 is a functional flow chart of the total retrieval operation. This operation consists of three distinct phases: (1) the waiting, or standby, phase in which no active tank operations are underway; (2) the tank acquisition and pickup phase which incorporates the activities of locating the tanks in the water, tank damage assessment (still in the water), tank loading by flooding the main deck of a special carrier ship (Point Barrow, Taurus design), floating the tank aboard using a handling harness which was attached in the water, pumping out the water which allows the tank to settle onto a support cradle where it can be washed, purged, and decontaminated, using shipboard equipment and returning the tank to a predeterminated transhipping port for transferring the tank to a coastal barge; and (3) the barge transfer and delivery phase which consists of

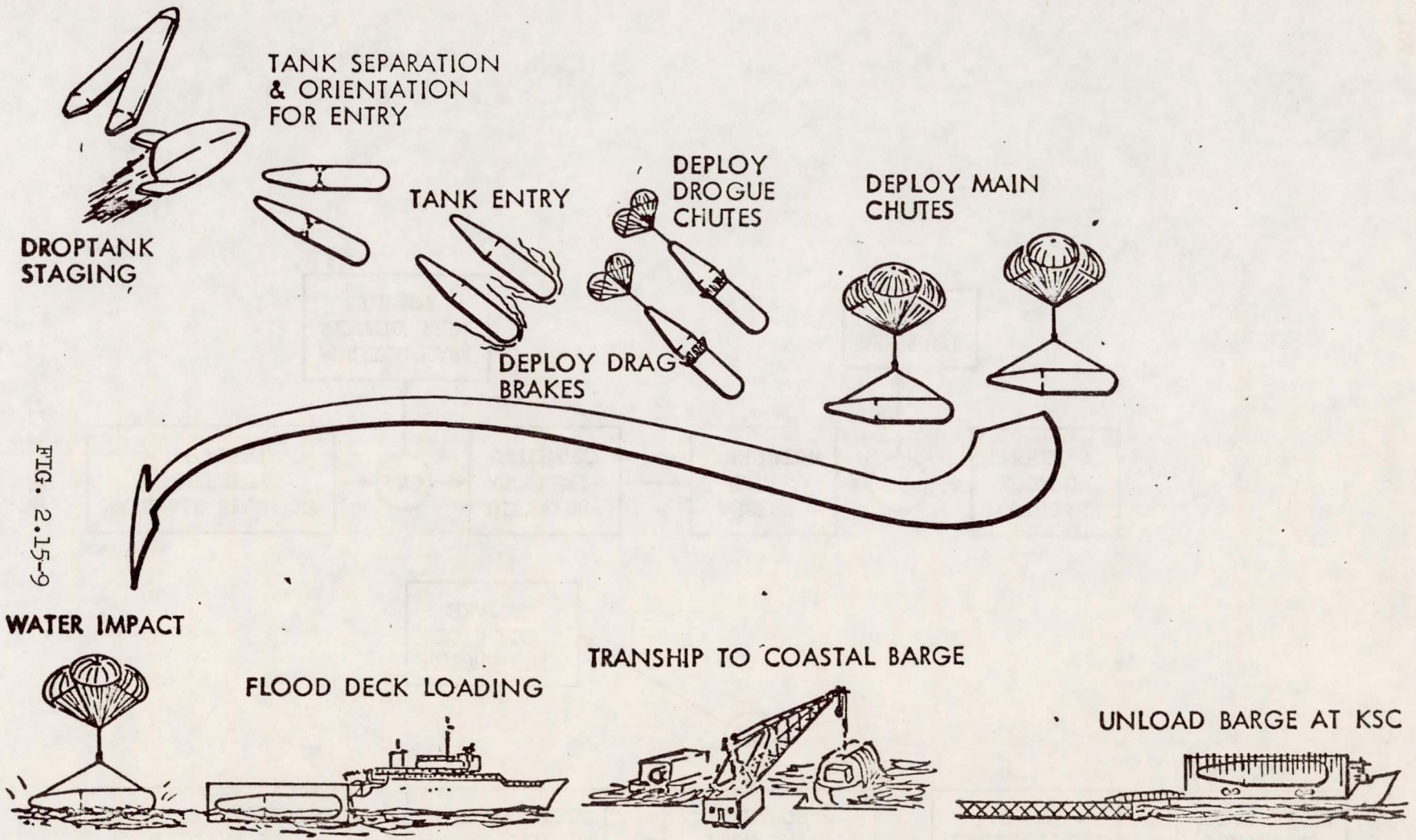


FIG. 2.15-9

Fig. 2.15-9 Droptank Water Recovery Operations

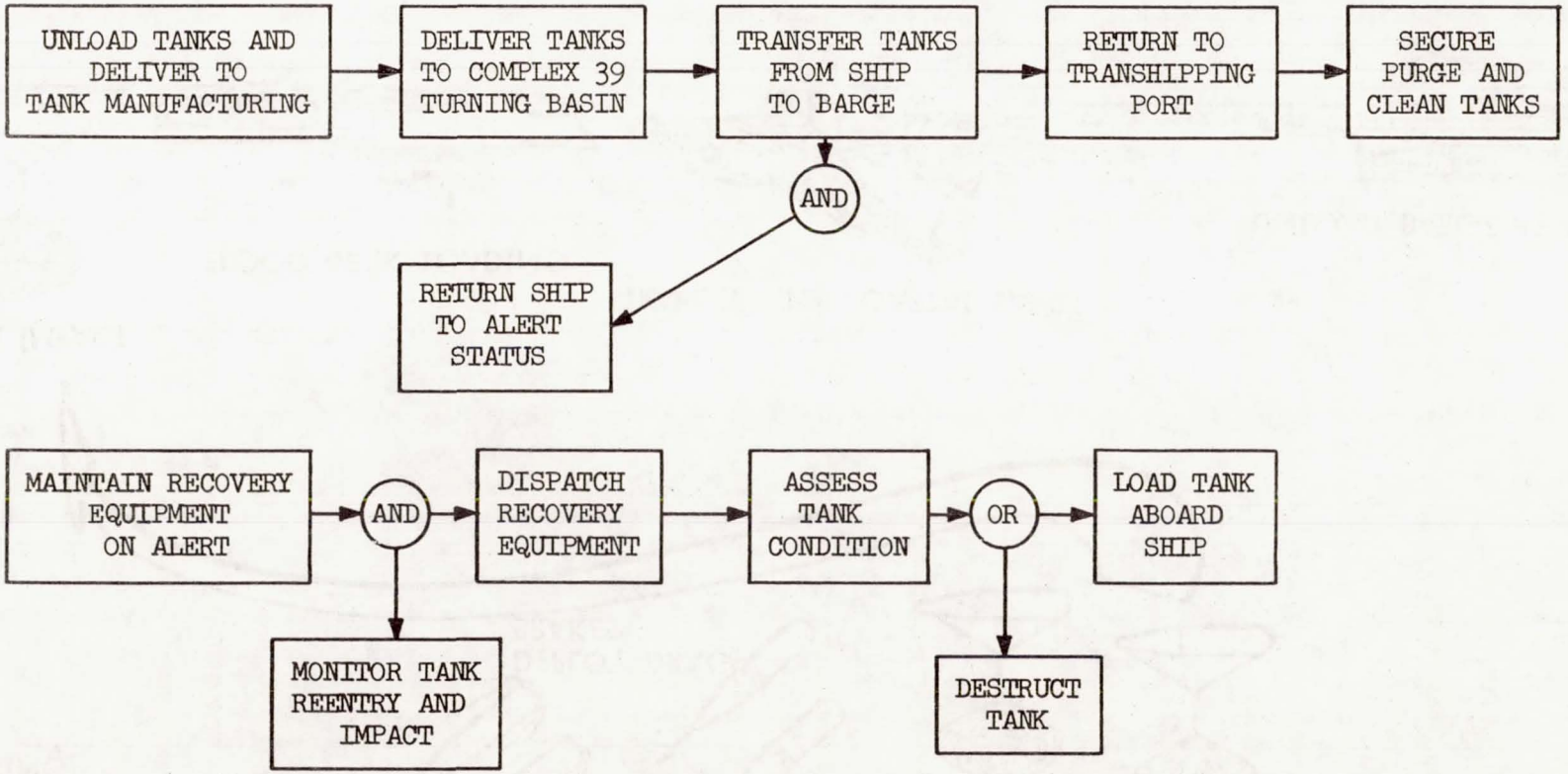


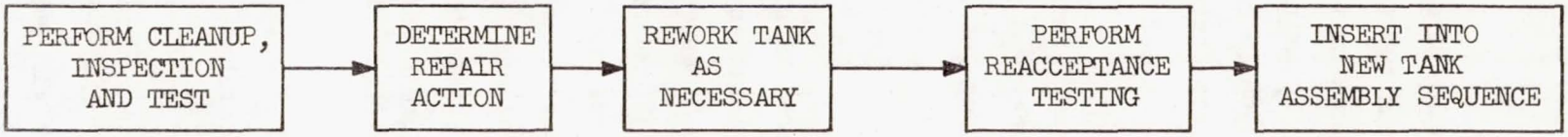
Fig. 2.15-10 Droptank Water Recovery Flow Diagram

offloading the tank from the pickup ship and onto a towable transporter located on the barge, returning the loaded barge by ocean tug to KSC Complex 39 turning basin (which is at the baseline Main Base), unloading the barge and delivering the tank to its transporter to the Tank Manufacturing Facility. For tank impact in areas which are due East from the CONUS land mass, the tanks would be delivered directly to the KSC area without coastal barge transhipping operation.

Two different types of recovery vessels are used because of the higher acquisition and operation costs of the special flood-deck type ship. The concept uses a minimum of this type of vessel and provides the long haul return with the relatively cheaper and more available barge and tug combination, even though this involves a transhipping operation.

Figure 2.15-11 is a functional flow chart of the recovered tank repair operations which also shows some of the major repair and checkout tasks involved. The tank repair concept involves three phases: (1) Damage assessment and repair action determination, which includes initial stripping of external components (harnesses, recovery equipment, lines, valves, etc.), test and inspections of welds and individual tanks and preparation of a repair plan; (2) tank rework which returns the tank to an acceptable condition; and (3) reacceptance testing which certifies the tank is ready for flight. These operations are analyzed on the basis of identical or similar work performed during initial tank fabrication.

2.15.1.4 Recovery and Reuse Costs. Additional costs associated with tank recovery and reuse stem from three sources: (1) the operational costs (manpower, equipment and facilities) required to retrieve and repair tanks; (2) additional tank equipment costs for recovery items (thermal protection, stabilization and control, parachutes, etc.); and (3) the vehicle weight penalty associated with heavier droptanks. These costs when added to the cost of the basic tank provide a total reusable tank program cost which can be compared with an expendable tank program to assess the desirability of tank recovery and reuse.



Remove Piping, Valves and Insulation

Clean Tank Exterior

Flush and Purge Tank Interior

Visually Inspect

Hydrostatic Test

Weld X-Ray

Dye Penetration

Remove and Replace Damaged Panels

Grind and Reweld

Replace Components

Weld X-ray

Dye Penetration

Pressure Test

Leak Check

FIG. 2.15-11

Fig. 2.15-11 Droptank Repair Flow Diagram

In the approach to the analysis, a baseline tank reuse cost is determined, based on fixed groundrules. Variations of these groundrules are then examined to determine the effect of specific parameters on total tank reuse cost.

The baseline assumptions are:

(1) Twenty (20) percent tank damage
(2) Ninety (90) percent learning
(3) Two hundred and fifty (250) total launches
(4) Ten (10) year program
(5) The 115,000 lb basic tank set weight
(6) All azimuth launches
(7) Use of Gemini and Apollo hardware development and fabrication cost history for on-tank recovery equipment
(8) Use retrieval ship and barge costs, based upon Apollo program history
(9) Consider only water landing and retrieval
(10) Assume vehicle weight increases only in the droptank and does not affect the basic core vehicle (Spacecraft).

Parameters subjected to variation:

(1) Launch density – 25/year and 75/year
(2) Launch azimuth – 55 deg only and all azimuth
(3) Basic tank set first unit cost – $5M ($44/lb), $10M ($87/lb) and $20M ($174/lb)
(4) Expected tank damage
(5) Expected tank uses

The initial analysis determined basic tank costs for first unit tank set fabrication estimates of $5M, $10M, and $20M, for various expected tank uses. Figure 2.15-12 shows a plot of these data.

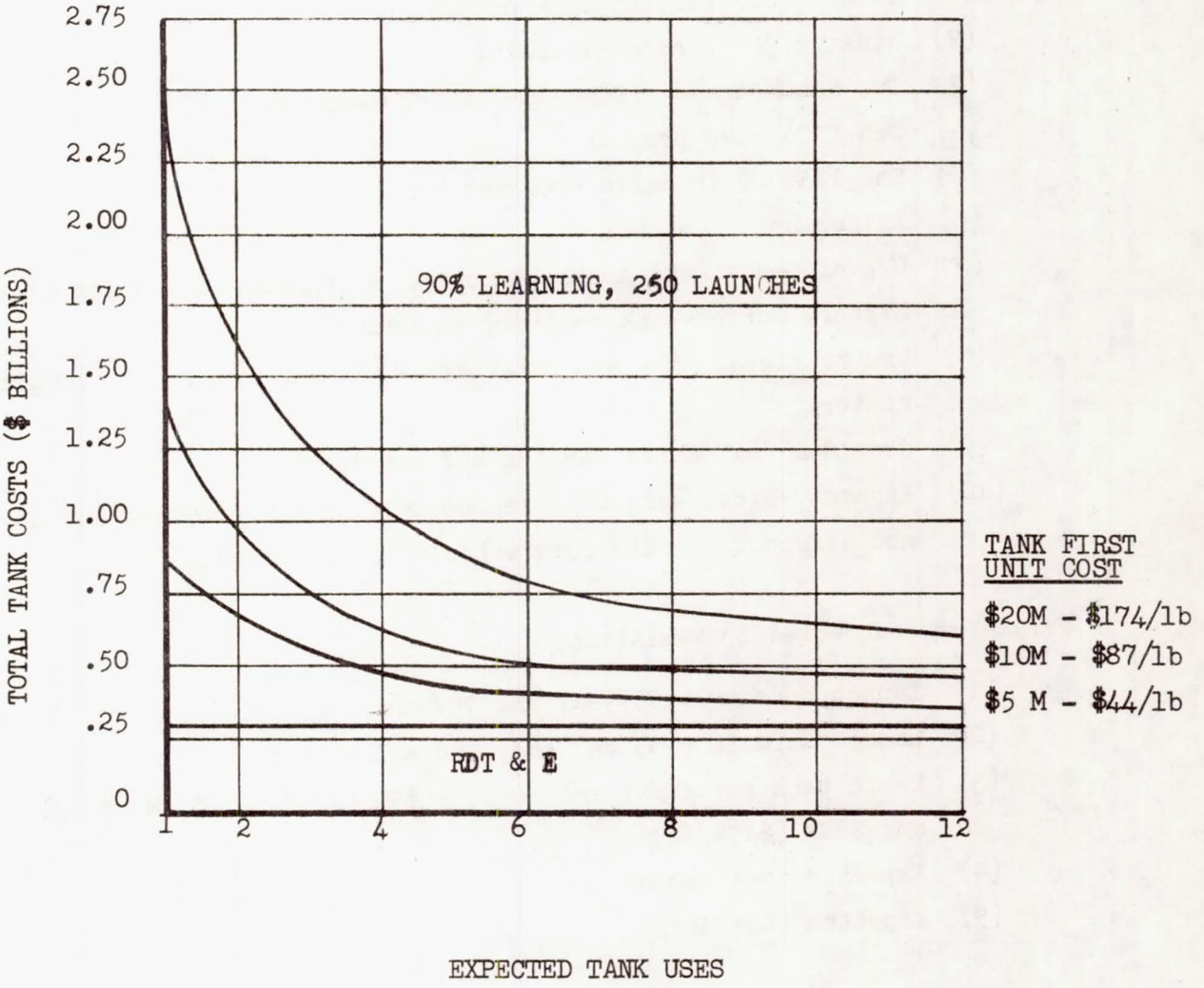


Fig. 2.15-12 Basic Droptank Cost – 115,000 Lb Tank Set

The RDT&E (nonrecurring) cost estimate for the droptank set is assumed constant at $316M for all three fabrication cost estimates. The expected tank use numbers determine the quantity of total tank sets required during the 10-yr program. By applying a learning factor of 90 percent, the total tank fabrication costs are then calculated and added to the RDT&E value to provide total basic tank program costs. Learning factors associated with the various tank quantities are:

Number of Uses	Total Tank Halves Built	Equivalent Quantity At 90 Percent Learning
1	500	228.8
2	250	126.9
3	125	70.3
4	84	50.1
5	63	39.2

The first additional cost determined for a reusable tank system is that associated with the weight growth of the total launch vehicle caused by the additional tank installed recovery equipment. This equipment consists of the tank thermal protection system, the stabilization and control hardware, the drag brake system, the retrorocket system, the parachute system, and the additional structural items. The total tank set weight increase for this equipment is 30,000 lb. Figures 2.15-13 and 2.15-14 show the method of assessing the total vehicle liftoff penalty associated with changes of tank mass fraction (λ'). By plotting the vehicle liftoff weight as a function of staging velocity for tank mass fractions around the values for the basic tank ($\lambda' = .96$), the vehicle liftoff weights at optimum staging velocities can be determined as a function of tank mass fractions and the vehicle weight increase caused by the tank installed equipment can be determined.

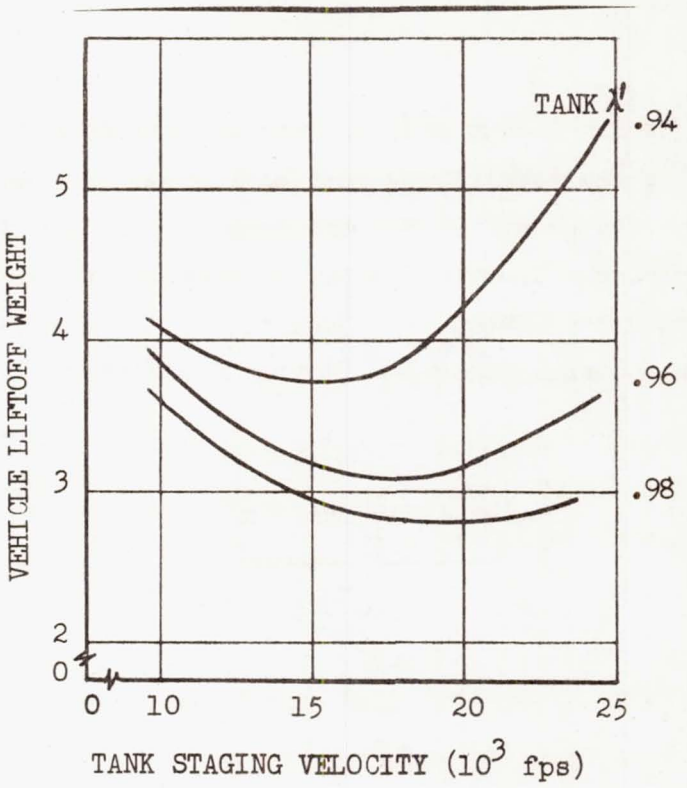


Fig. 2.15-13 Recoverable Tank Vehicle Weight Penalty

$$\text{TANK } \lambda' = \frac{\text{TANK PROPELLANT WT.}}{\text{TOTAL TANK WT.}}$$

$$\text{BASIC TANK } \lambda' = \frac{2,852,700}{2,978,700} = .958$$

$$\text{RECOVERABLE TANK } \lambda' = \frac{2,852,700}{2,978,700 + 30,000} = .948$$

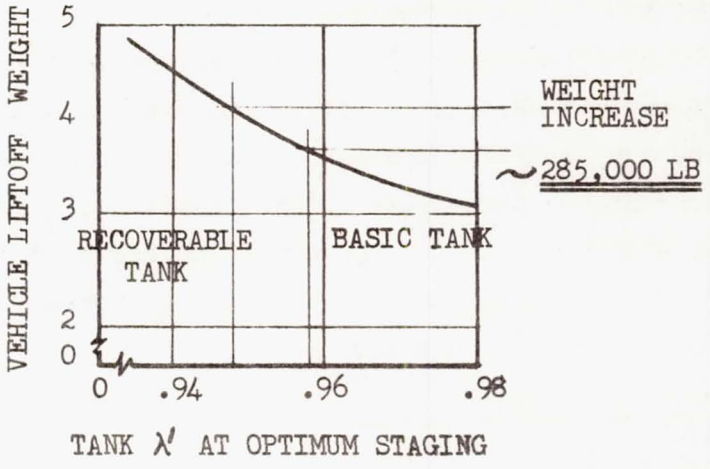


Fig. 2.15-14 Recoverable Tank Vehicle Weight Penalty

A tank weight increase of 30,000 lb results in a vehicle weight increase of 285,000 lb. Assuming this vehicle weight increase is absorbed entirely by droptank growth (no change in the spacecraft), the cost of this growth can be determined by adding the cost of the increased propellant load to the cost of the increased tank size:

For tank mass fraction (λ') = .95

Increased Propellant	=	242,000 lb
Increased Tank Size	=	13,000 lb
Additional Equipment	=	30,000 lb
		285,000 lb

Total increased propellant cost per flight = 242,000 ($.10/lb) = $.024M

Increased Tank Cost (First Unit) = 13,000 ($87/lb) = $1.13M

The next costs derived are those for the additional tank installed recovery equipment. These costs are based on historical cost curves for Mercury, Gemini, and Apollo equipment presented in Ref. 2.15-4. These equipment costs are of three types: (1) Fixed cost (nonrecurring); (2) single-use items (expended each flight); and (3) variable cost items (repaired and reused). These costs are summarized below:

Fixed Cost – Design and Development for:

Thermal Protection System	–	$ 35M
Landing Rocket System	–	4M
Stabilization and Control System	–	7M
Parachute System	–	11M
Drag Brake System	–	11M
TOTAL		$ 68M

Single-Use Items – First Unit Cost for:

Thermal Protection System	–	$ .746M
Landing Rocket System	–	.092M
Stabilization and Control (Guidance Package)	–	.100M
TOTAL		$ .938M

Variable Cost Items – First Unit Cost for:

Additional Structural Provisions	–	$ .07M
Drag Brake and Actuation	–	.27M
Parachute System	–	1.22M
Stabilization and Control (Hardware)	–	.50M
TOTAL		$2.06M

The total program cost for the single-use items is obtained by multiplying by the equivalent number of flights for 98 percent learning factor. This lower learning factor is justified because the hardware is essentially available, and therefore would not follow the 90 percent learning associated with a totally new product. The total program cost for the variable cost items is the product of the equivalent number of flights, using 90 percent learning and the number of equipment reuses.

Tank retrieval costs are based upon current bid prices for Landing Ship Dock (LSD) type vessels and the cost of Apollo retrieval and delivery operations recorded in Ref. 2.15-5. Retrieval costs are based on an assumed 25 launches per year equally spaced:

Ship Fleet Acquisition	–		$129M
Modifications on 2 ships	–	$ 9M	
Purchase of 6 ships	–	120M	
Ship Operating Costs (8 ships)	–		$118M
Personnel (800 men)	–	$103M	
Steaming ($1000/day)	–	8M	
On Station ($320/day)	–	7M	
Barge Operating Costs (2 Barges @ $1700/day)	–		$ 10M
Retrieval Equipment Maintenance	–		$ 20M
Dry Dock	–	$ 8M	
Port Costs	–	2M	
General Stores	–	8M	
Equipment Maintenance	–	2M	
TOTAL 10 YEAR COST			$277M

The above recovery costs are based upon ship coverage of four prime impact areas, each ship having its own port facility. These areas are: North Atlantic (basic 55 deg orbits), Atlantic (easterly launches), South Pacific (southerly launches), and Pacific (westerly launches). Only the North Atlantic utilizes a transhipping operation between recovery ship and coastal barge.

The final costs analyzed are those for tank repair. These estimates are based on initial cleaning, insulation removals, component removal, internal cleaning and inspection; and tank repair costs. These costs are summarized below:

• Cleaning, Inspection, and Preparation	$182,500
• Repair*	992,000
• Reacceptance Testing*	342,000
• Material*	234,000
• Burden and Profit*	226,200
Total First Unit Repair Cost	$1,976,700

These costs were based upon a bottom-up tank fabrication cost estimate included in Ref. 2.15-1, and are adjusted (increased 300 percent) to baseline the $10M ($87/lb) first units costs assumed.

Figure 2.15-15 shows the total program costs for reusable tanks as a function of number of tank uses for a $10M basic tank first unit cost and 20 percent tank damage. These data are a composite of all the costs discussed in the preceding paragraphs.

Figure 2.15-16 shows the total program costs for reusable tanks as a function of number of tank uses for all three of the basic tank first unit costs examined: $20M, $10M, and $5M. Also shown is the projection of tank total cost for one use (expendable tank costs) to indicate the break-even point between reusable and expendable tank programs. For 20 percent tank damage and a first

*These costs are based upon 20 percent of Tank First Cost ($10M)

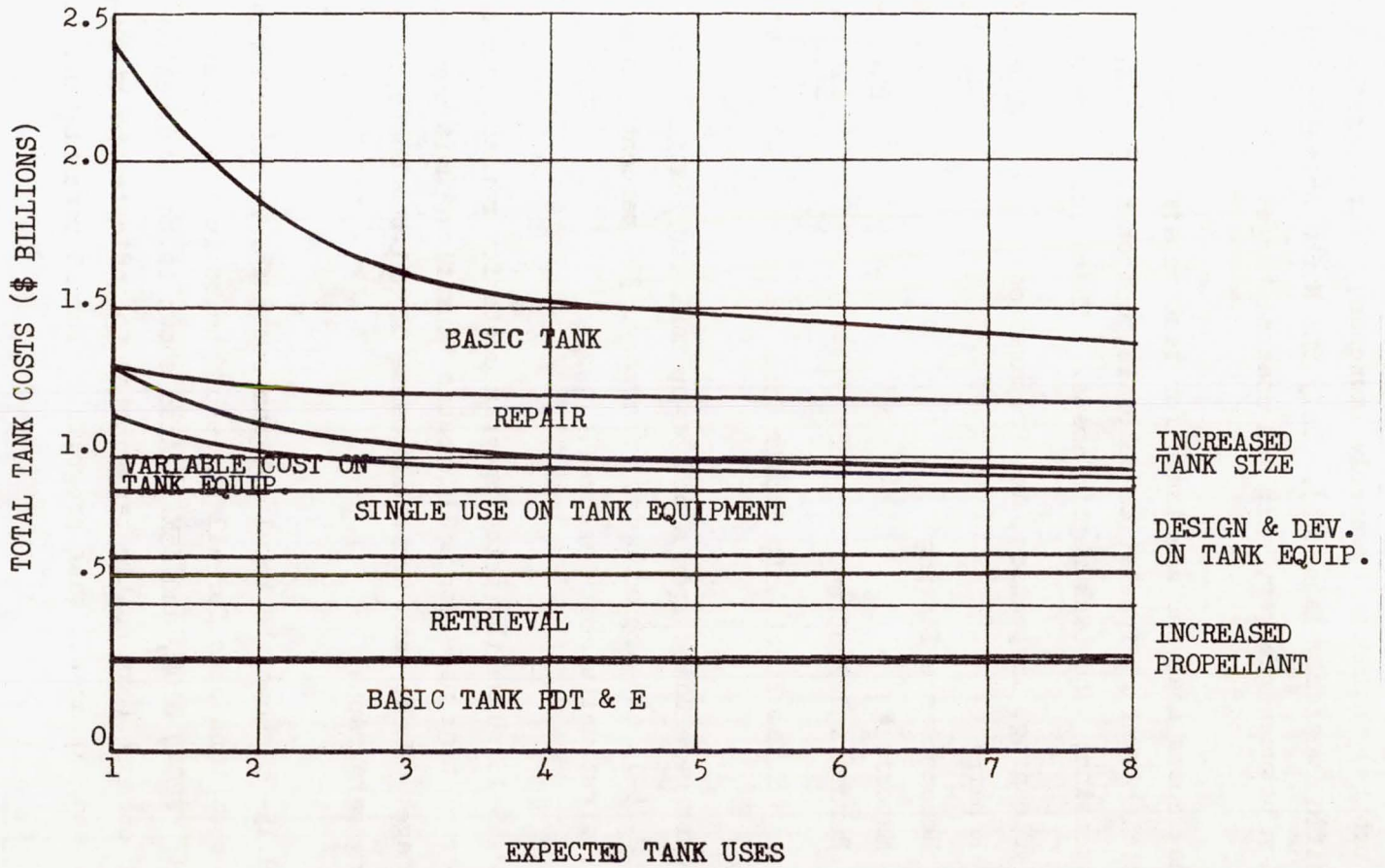


Fig. 2.15-15 Reusable Tank Costs – 250 Total Flights, $10M First Tank Cost, 20 Percent Tank Damage, 90 Percent Learning

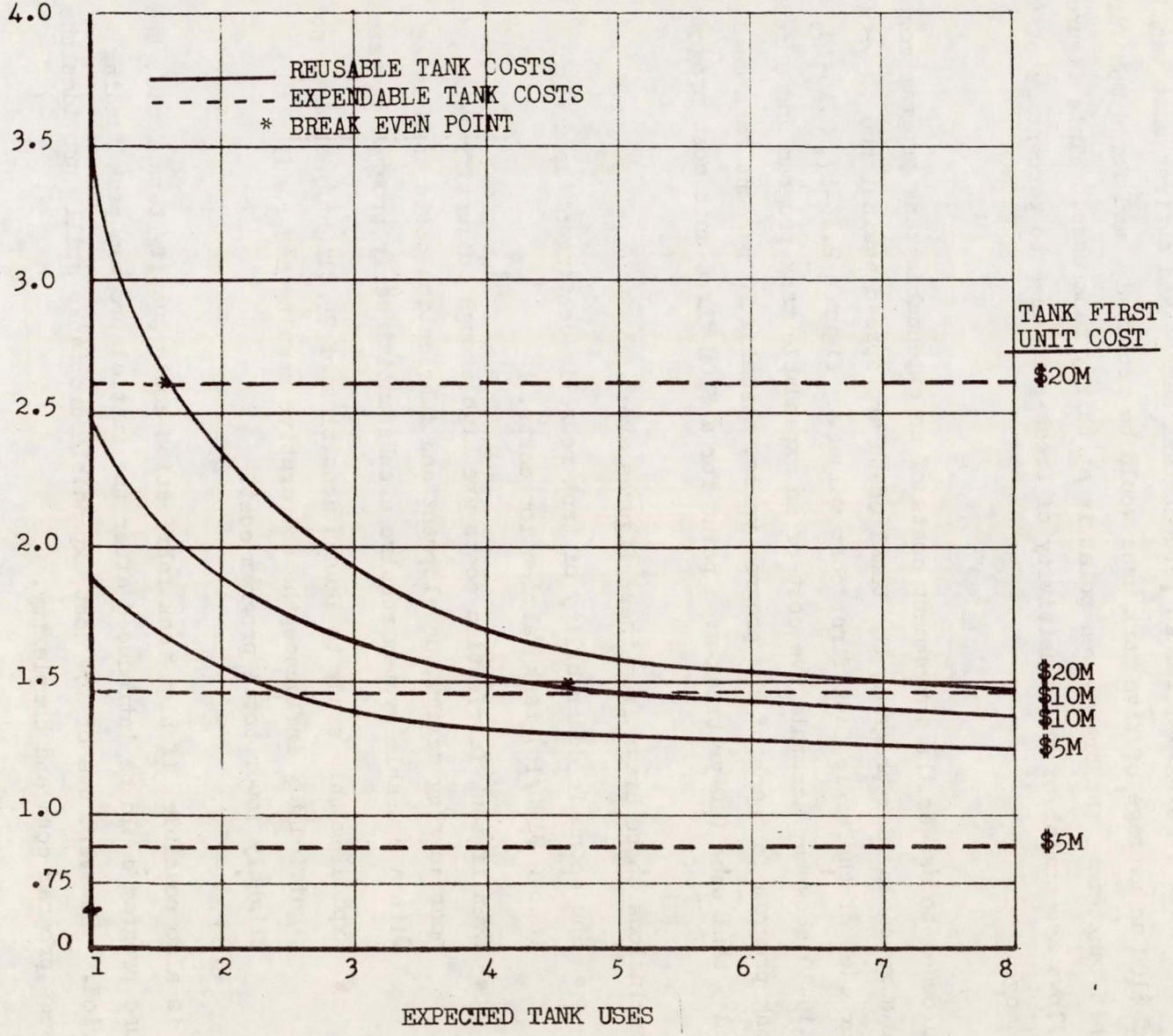


Fig. 2.15-16 Expendable Vs Reusable Droptank Costs – 250 Total Flights, 20 Percent Tank Damage, 90 Percent Learning

unit tank cost of $20M, an average of two tank uses would be required to break even in a 10-yr program; for 20 percent tank damage and a first unit tank cost of $10M an average of five tank uses would be required; and for a $5M first unit tank cost, the break-even point is at thirty tank uses. This figure allows assessment of the sensitivity of break-even uses to percentage of tank damage.

In order to assess the investment costs of an expendable tank program compared to a reusable tank program, the total costs are spread against the 10 yr program for each of the basic tank first cost values. Figures 2.15-17, 2.15-18, and 2.15-19 show the accumulative cost of an expendable tank program and a reusable tank program for each of the program years, assuming 20 percent tank damage and 5 tank uses (10-yr break-even point for a $10M first unit cost program).

Conclusions to be drawn from these figures are:

- The economic desirability of tank reuse is contingent upon high (above $100/lb) tank fabrication costs.
- Even if tank fabrication costs are high enough to justify reuse, the nonrecurring design, development and fabrication costs associated with a reusable system require a substantial early program investment.
- Expanding this study to launch densities of 75 flight/year does not significantly influence the comparative results and results in a slightly lower total program cost.

It is also possible, if high tank fabrication costs justify tank reuse, that a reuse program could be introduced after the initial program peak funding period. However, use of low cost expendable droptanks still provides the lowest program cost and complexity.

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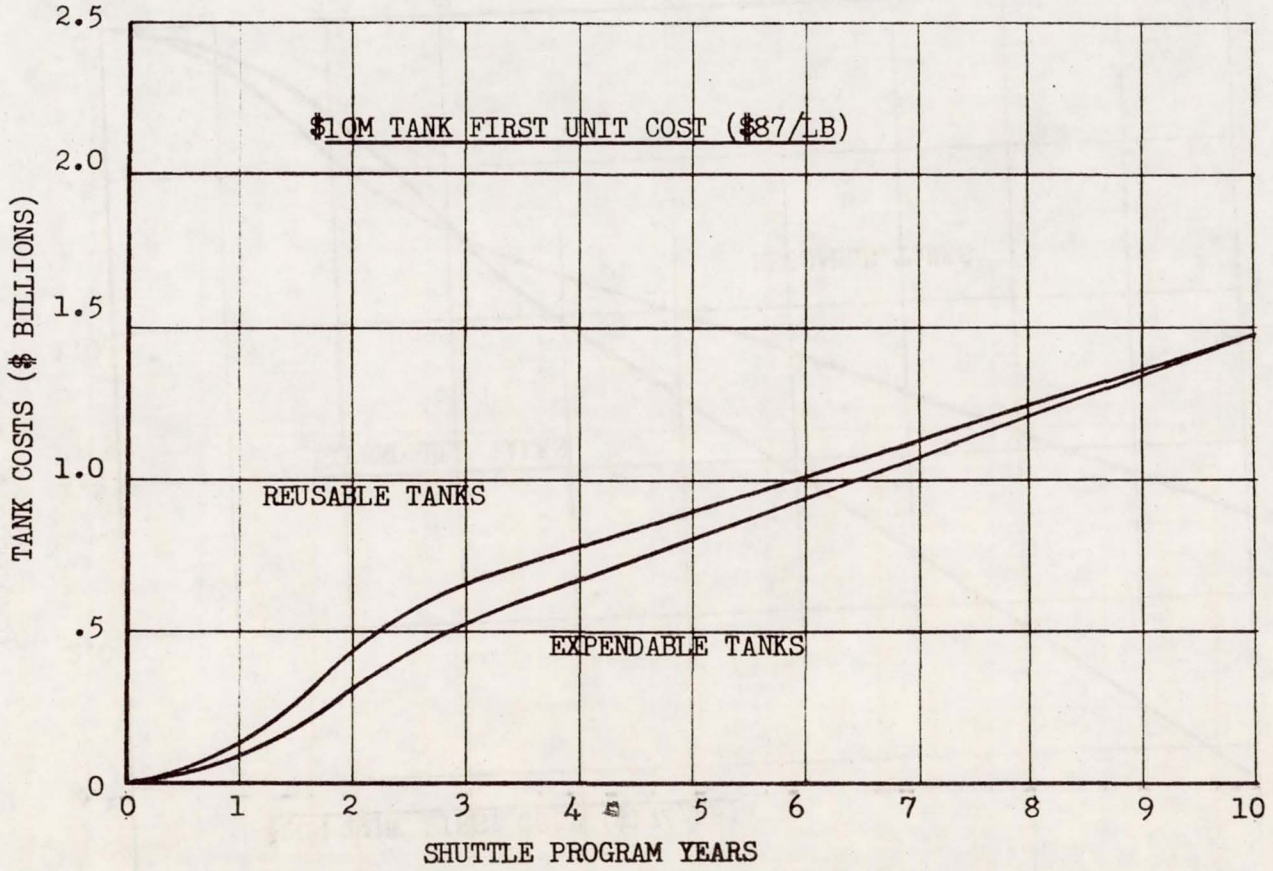

Fig. 2.15-17 Droptank Program Cumulative Costs – 90 Percent Learning, 20 Percent Damage, 5 Tank Uses, 250 Flights

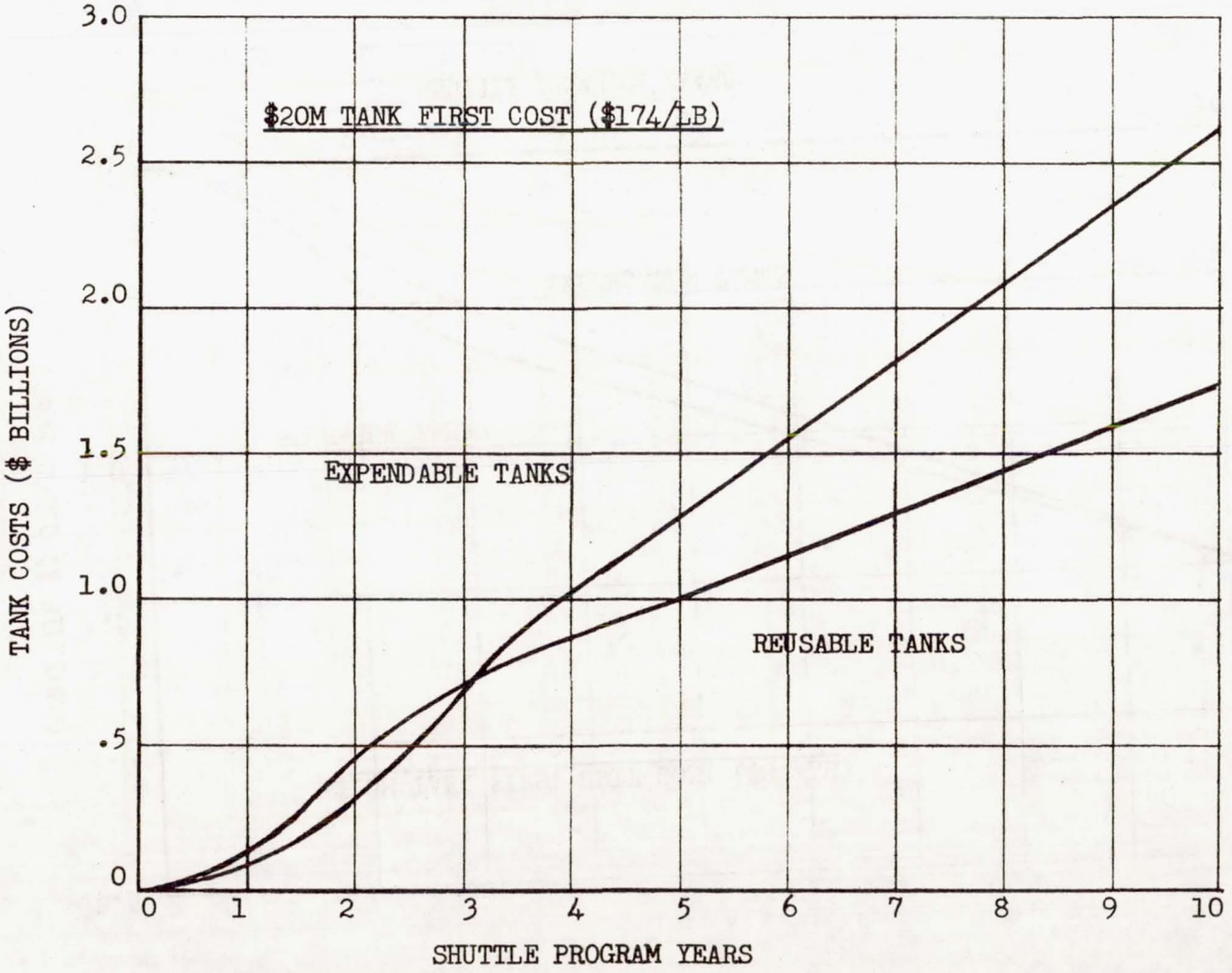

Fig. 2.15-18 Droptank Program Cumulative Costs – 90 Percent Learning, 20 Percent Damage, 5 Tank Uses, 250 Flights

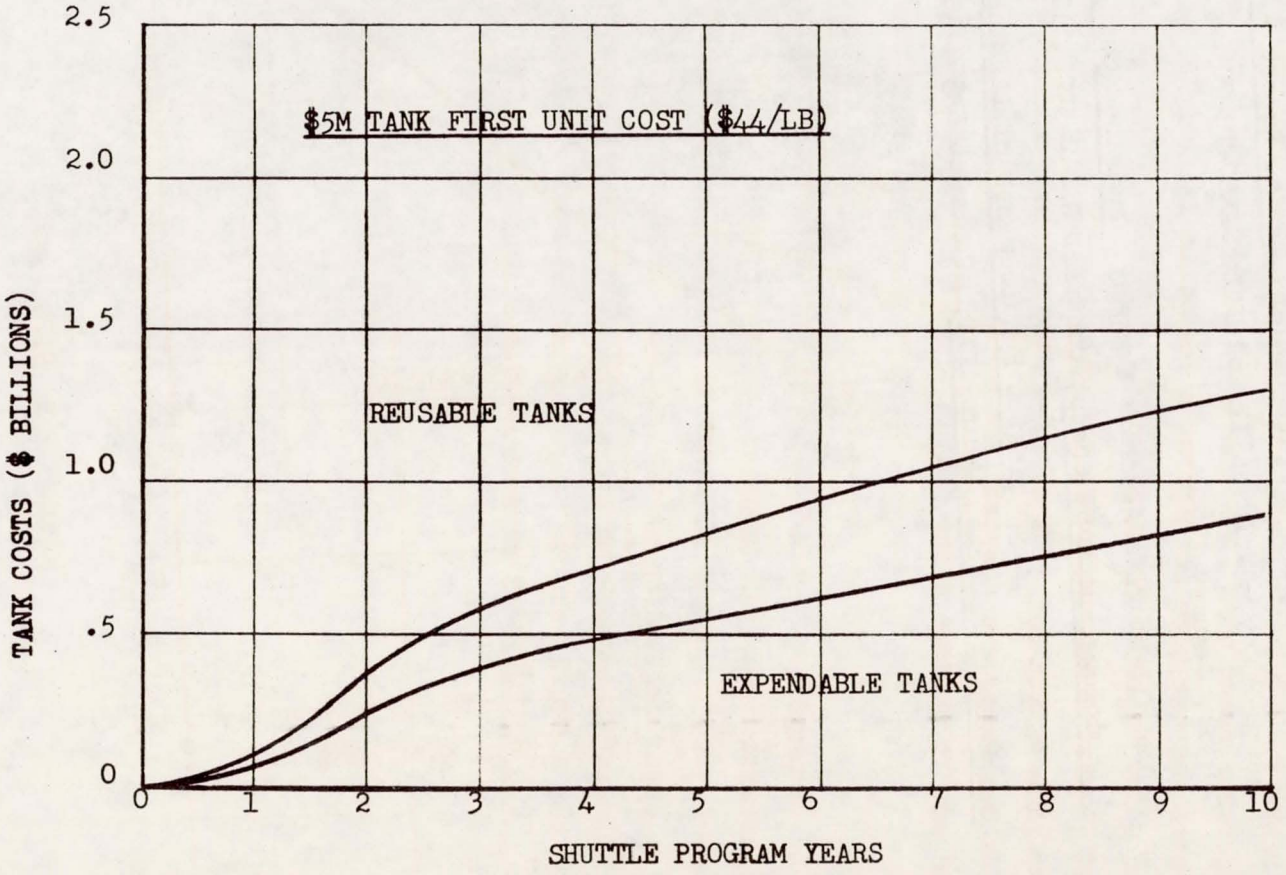

Fig. 2.15-19 Droptank Program Cumulative Costs – 90 Percent Learning, 20 Percent Damage, 5 Tank Uses, 250 Flights

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2.15-5 The Boeing Company, Mission Requirements of Lifting Systems – Operational Aspects, Volume III, Contract NAS 9-3522, August 1965.

2.15.2 Droptank Weight Analysis

During the first five months of the study, a detailed droptank weight and cost analysis was performed. In this analysis, detailed drawings, based on structures and load analyses, were made and, from these drawings and the associated structural analysis, a weight analysis was performed. The design data were used to assemble a "Droptank Bid Package" which included specifications, a delivery schedule, and other pertinent data. These bid packages were sent to a number of companies for droptank production cost estimates. The results of this droptank analysis are presented in detail in Section 2.10 of the Fifth Monthly Report and a summary of the cost results is presented in Section 2.15.3.

The weight analysis confirmed the Lockheed claim of a droptank λ' of slightly over 0.96. A summary weight statement for the droptanks for the LS 200-5 stage-and-one-half system is shown in Table 2.15-2. These weights are based on a sizing analysis based on the droptank line loads as shown in Figure 2.15-20. The droptanks for the later stage-and-one-half vehicles are of essentially the same design as those for the LS 200-5 and their structural and weight analyses are presented in Section 2.8 and 2.14, respectively, of this report.

Table 2.15-2

Droptank Weight Summary

	WEIGHT (LB)		NONOPTIMUM FACTOR	UNIT VALUES
STRUCTURE		83,198		
LO_2 TANK	13,228		1.23 AVG	
INTERTANK AND TIE STRUCTURE	15,570		1.10/1.00	
LH_2 TANK	40,724		1.24 AVG	
THRUST CONE AND ATTACH	13,676		1.15 AVG	
INSULATION		4,058	MAX THICKNESS	
LO_2 TANK	2,646		REQT APPLIED	
LH_2 TANK	1,412		TO TOTAL TANK	
PLUMBING		5,729		
LO_2	2,863			
LH_2	2,866			
INSTRUMENTATION		301		
CONTINGENCY (10%)		9,329		
TOTAL WEIGHT		102,615		.035 LB/LB PROP
λ'				.962

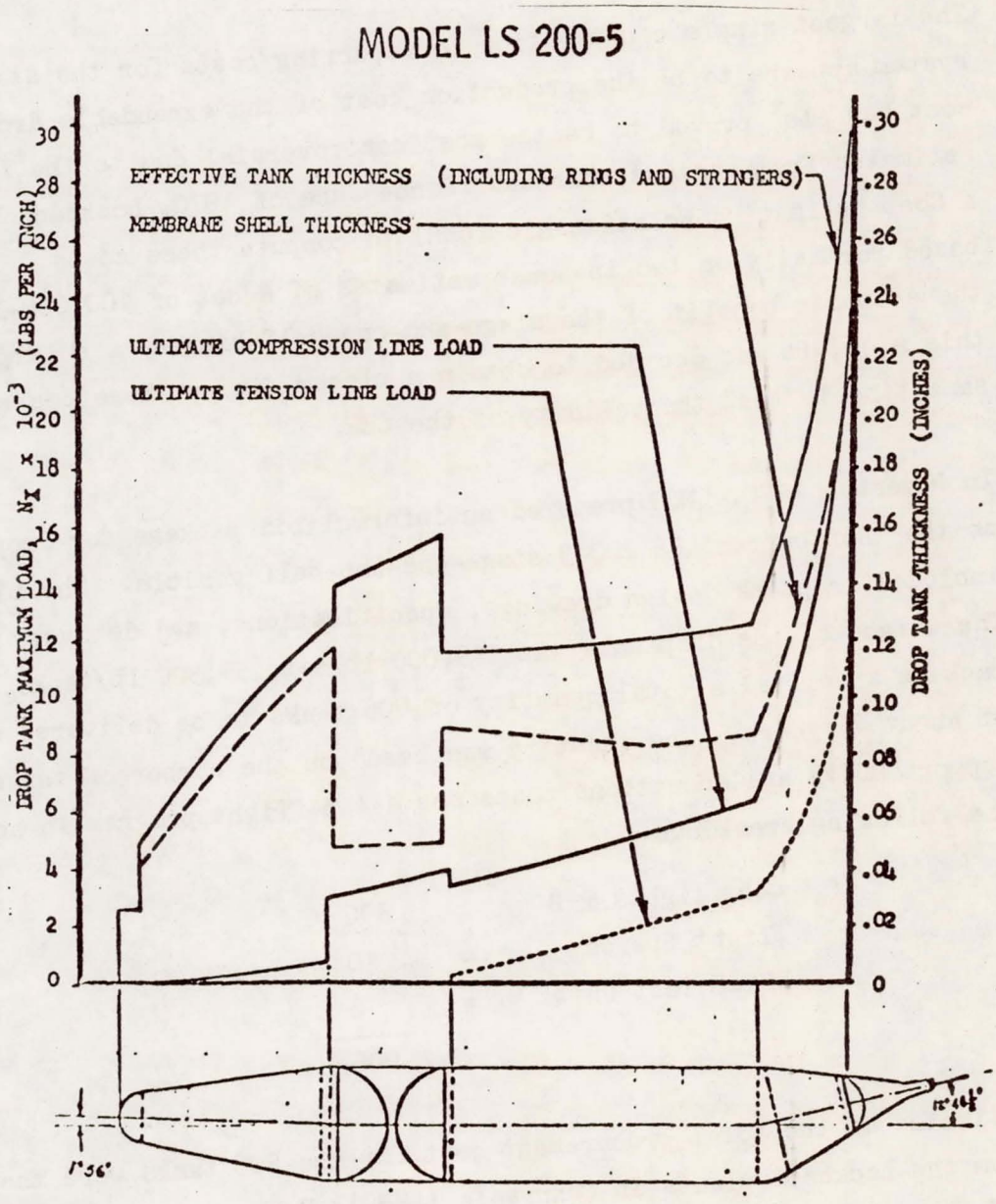

Fig. 2.15-20 Drop Tank Sizing

2.15.3 Droptank Cost Analysis

The largest single contribution to recurring costs for the stage-and-one-half system appears to be the production cost of the expendable droptanks. This cost has also proved to be the most controversial due to the large range of estimates from various sources. Since June of 1970, Lockheed has been using a Cost Estimating Relationship (CER) to compute these costs. The CER is based partially on two in-house estimates of a set of 100,000-lb tanks. Since the economic benefit of the stage-and-one-half hinges to a large degree on this area, it was decided to obtain a closer fix on these costs and at the same time to test the validity of the CER.

In November 1970, LMSC prepared an informal bid package for droptanks, based on the then current LS 200-3 stage-and-one-half vehicle. The bid package included droptank design drawings, specifications, and delivery schedules. The weight of the droptanks was 119,000 lb/set (59,500 lb/tank). The bid package specified a total quantity of 906 tanks to be delivered over a period of about 11 years. The quantity was based on the number of tanks required to support DDT&E and operations costs for a 445-flight program in accordance with the following breakdown:

445 flights x 2	=	890
Flight Spares	=	10
DDT&E Test Units	=	6
		906

Estimates of the total procurement cost for the 906 tanks were then solicited from the Lockheed plants in Sunnyvale (LMSC), Burbank (CALAC), and Marietta (GELAC), as well as from Chrysler Corporation Space Division at Michoud. The responses from the bidders, all of which included 10 percent fee, were as follows:

Chrysler	$ 1,672 M
LMSC	$ 2,014 M
CALAC	$ 2,532 M
GELAC	$ 3,389 M

An independent estimate of the design and development cost was also made by LMSC. This estimate of $65 million was for all DDT&E effort except test hardware and tooling, since these costs were included in the procurement estimates listed above. In order to isolate DDT&E from recurring costs in the estimates, all nonrecurring costs, where they could be identified, were subtracted out of the response estimates. In the case of Chrysler, nonrecurring costs would not be identified, and therefore, an amount proportionate to other companies' nonrecurring/total estimate ratio was subtracted from the Chrysler estimate. This resulted in the following breakdown:

	Chrysler	LMSC	CALAC	GELAC	MEAN
Estimate	1672	2014	2532	3389	
Nonrecurring	-130	-182	-216	-199	
Total Recurring	1542	1832	2316	3190	2220
Nonrecurring	130	182	216	199	
Design and Develop.	+65	+65	+65	+65	
Total DDT&E	195	247	281	264	247

In the above estimates, the DDT&E costs include the cost of 6 tanks to be used in development testing. The recurring costs, therefore, include the costs of the remaining 900 tanks to support operational flights and operational spares.

Corresponding costs for the same tanks derived by the Lockheed CER are:

Total DDT&E: $266 million
Total Recurring: $2174 million

The cost estimate data from all sources are summarized below:

	Detailed Estimates					
	Chrysler	LMSC	CALAC	GELAC	MEAN	LMSC CER
DDT&E Cost ($M)	195	247	281	264	247	266
Recurring Cost ($M)	1542	1832	2316	3190	2220	2174
Totals ($M)	1737	2079	2597	3454	2467	2440
Average Recurring ($/lb)	28.80	34.20	43.20	59.60	41.50	40.60

For this point estimate of a 119,000 lb/set droptank design weight, the CER is seen to track about 8 percent higher than the DDT&E mean detailed estimate and about 2 percent less than the mean recurring detailed estimate. Since droptank weight is not expected to exceed a range of $\pm$ 10 percent about the 119,000 lb design weight for stage-and-one-half configurations currently envisioned, the validity of the CERs was considered to be upheld, and they continue to be used in the cost estimating model.

2.15.3.1 Trajectory and Dispersion Analysis. The purpose of the studies in this catagory was to define the behavior of the droptanks during reentry, so that thermal and structural analyses could be carried out to determine survivability; establish the dispersion pattern of the droptanks, assuming the tanks do not fragment into pieces; for the case of a tank fragmenting into pieces, predict the dispersion pattern of the pieces; determine the performance penalty and dispersion pattern when the nominal impact location of the droptanks is changed to other locations; and establish the feasibility of a multi-azimuth launch from ETR and WTR. An overview report on these objectives is included as Ref. 2.15.3-1.

After staging, the droptanks are separated and begin tumbling end-over-end until dynamic pressure builds up so that the tumbling damps out. Then each tank begins a coning, precessing motion at a 40 deg angle-of-attack, aft and forward, rolling on its own longitudinal axis at rates up to 1.5 rev/sec. (see Fig. 2.15.3-1). The associated entry trajectories of such a motion have fairly short durations (225 sec to 100,000 ft), with peak stagnation-point heating rates of about 70 $Btu/ft^{3/2}$-sec and peak dynamic pressures of 200-400 psf.

The dispersion pattern associated with a structurally intact tank from the baseline staging condition is an ellipse 90-nm downrange (intrack) and 40-nm crossrange (out-of-track). This dispersion results from uncertainties in staging conditions, separation rates, tank orientation during reentry, winds, and atmospheric variations. The 90-nm downrange dispersion is a conservative estimate (three sigma). The dispersion variance (one sigma) is about 40-nm. Fig. 2.15.3-2 provides a summary of the dispersion study results. Both the point mass data and the six-degree-of-freedom cases are shown (See Refs. 2.15.3-2, -3 and -4).

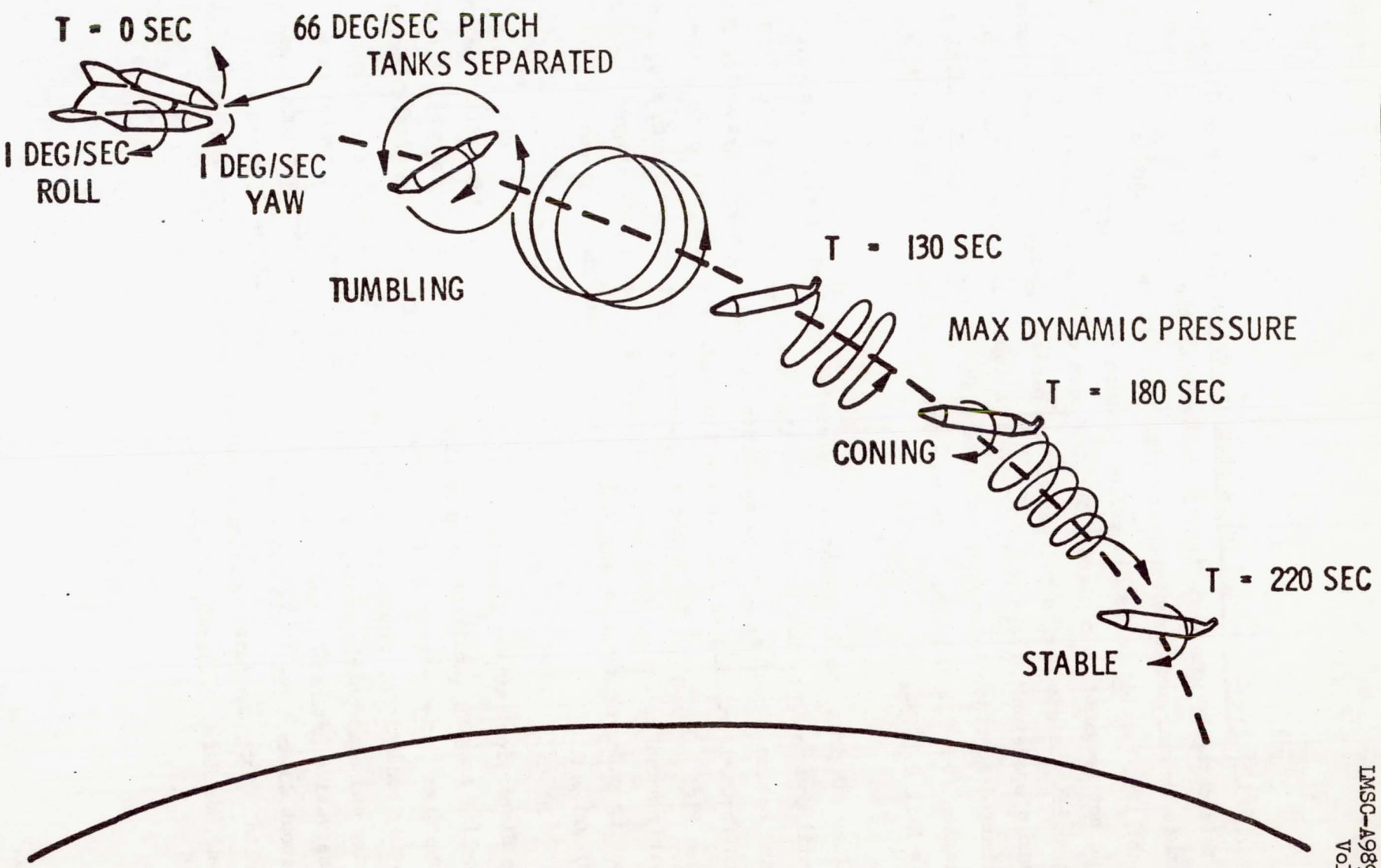

Fig. 2.15.3-1 Entry Mode of Droptanks

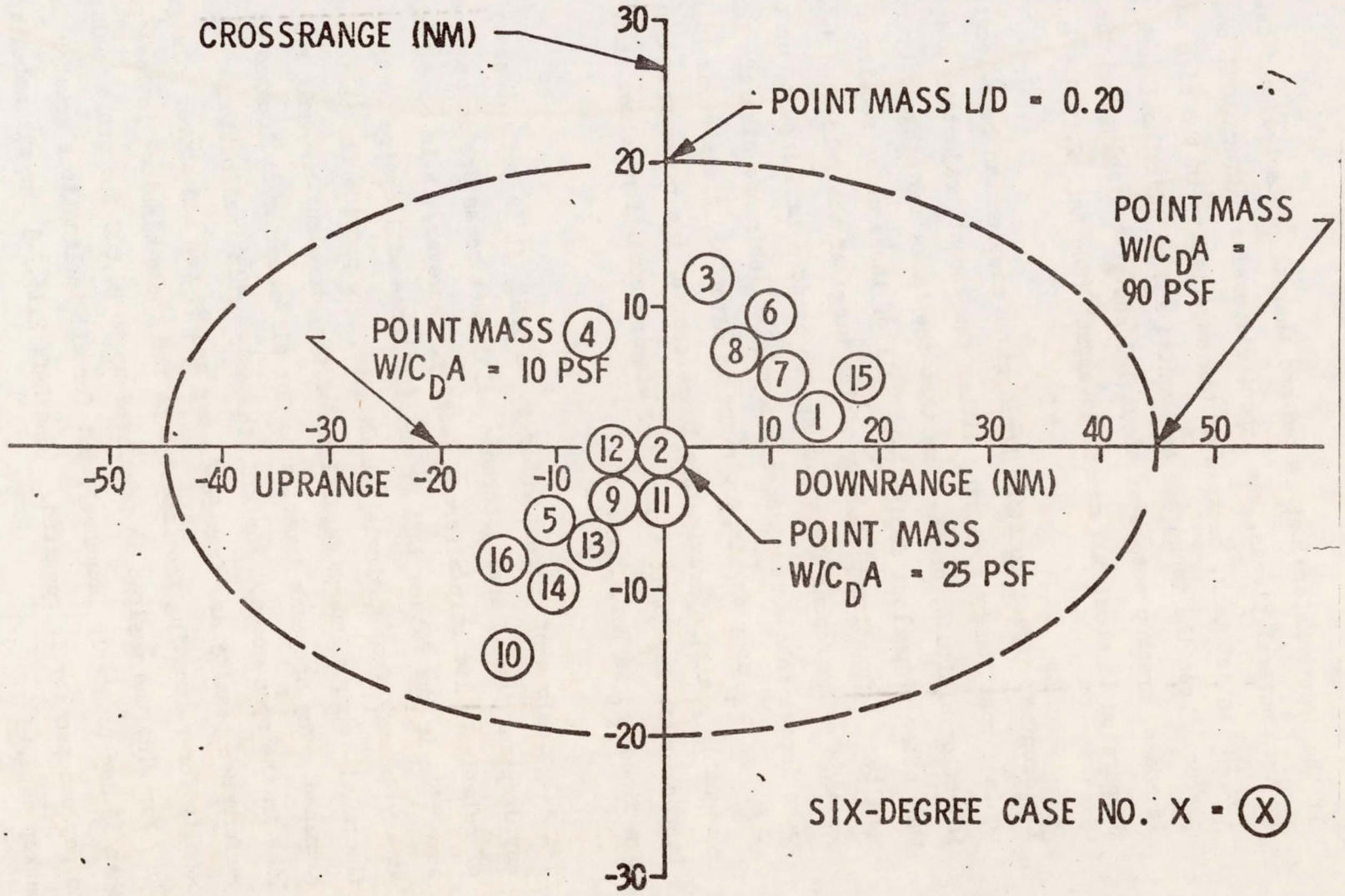

Fig. 2.15.3-2 Footprint Due to Variations In Droptank Orientation

If the droptanks are not vented and thermal protected, and the tanks break up into several pieces, the impact dispersion pattern will be larger than for the intact tank, because of the uncertainty in the size and shape of the pieces. When the tanks are not vented and no additional cork insulation is added, breakup occurs at about 120 sec past staging and the downrange dispersion is about 300 nm (three sigma) (See Ref. 2.15.3-5 and -6).

The droptank impact location for a given mission can be relocated from its nominal position by modifying either the ascent trajectory shape, the orbit injection altitude, or the time that the tanks are staged (by retaining them after propellant depletion). All of these modes require a payload penalty – a function primarily of the range of the relocated impact position – the resons for which are to avoid a land impact, or to move an ocean impact into a less densely populated (ship and aircraft lanes) ocean area. As the droptank impact range is extended, dispersions in range remain reasonably small, approximately 10 percent of the range from the launch site. Dispersion in impact locations for staging conditions other than nominal are discussed in Ref. 2.15.3-7.

For all missions of current interest to NASA, it is not necessary to relocate any droptank impact areas, because all impact zones are in open ocean areas of relatively low ship/aircraft density. However, it is conceivable that some time in the future NASA or the Air Force will desire to launch missions at any azimuth from a single launch site. To provide for that eventuality, the multi-azimuth launch capability of the stage-and-one-half system was examined. The droptank impact zone for all azimuths is located so that it lies in the open ocean. For ETR, the only orbit inclinations that incur any performance penalty are from 58-62 deg and 95 deg and above. The payload penalty for relocating the impact zone for these missions is less than 5,000 lb. For WTR, the payload is penalized up to 20,000 for orbit inclinations less than 55 deg (easterly launches), but for all inclinations greater than 55 deg, no payload penalty is required. (See Ref. 2.15.3-1 for all azimuth launch backup data.)

2.15.3.2 Thermo Structural Analysis. This part of the Droptank Impact and Dispersion Study was conducted to establish a conceptual droptank design for intact entry from which estimates of weight and cost could be prepared. The backup data for this section is included in Appendix A as Ref. 2.15.3-8.

By determining the reentry temperature histories for tumbling intact droptanks and comparing these with the thermo structural capability of the baseline tank design, it was found that the buildup in internal tank pressure would cause the tanks to burst shortly after separation. This analysis showed that the critical items of tank survivability are (1) control of internal tank pressure and (2) control of tank temperatures. Further study of actual values of pressure and temperature produced the following recommendations to provide intact tank impacts:

- Vent LH_2 tank to 8 psia after separation
- Vent LO_2 tank to 15 psia after separation
- Add 0.06 in. of cork insulation to LH_2 tank
- Add 0.09 in. of cork insulation to interstage

The above conditions result in a reduction of 1,320 lb of payload with constant GLOW, or an increase of 31,700 lb of GLOW with constant payload.

2.15.3.3 Operations and Safety. This area of Task 1 consisted of supplying a brief historical analysis of present launch vehicles expended, safety precautions, and policies; a stage-and-one-half shuttle droptank hazard analysis; and a cursory contamination study of the ecological effect of shuttle operation. The historical study included a determination of the quantities of ocean-impacting expended-stages, to date, to compare with projected shuttle droptank impacts; the ocean areas into which these expended-stages have impacted to compare with the projected droptank impact areas; and the standard launch and range safety procedures for both WTR and ETR launches.

The droptank hazard analysis consisted of producing ocean "population" density (ship and aircraft) maps and superimposing both the eight discrete launch azimuths defined for shuttle and the all-azimuth launch footprints onto these "population" density maps to determine ship and aircraft hit probability. The discrete azimuth case is shown in Fig. 2.15.3-3.

Impact probability was determined as a function of "population" density and droptank pieces and was used to compare droptank impact hazard with present standard transportation system hazards (ships, aircraft, automobiles, trains, etc.).

Details on the items discussed to this point are available in Ref. 2.15.3-9.

Another item of the hazard analysis was the consideration of droptank element collision during entry after separation. The analysis showed that design of the droptank subsystem could provide positive means of translating the separated tanks to ensure that collision will not occur. (See Ref. 2.15.3-10 for details.)

Also included in the hazard analysis was a study to determine the costs of reducing the tank dispersion area and the costs of prelaunch surveillance of the impact area. Tank dispersion can be reduced by preventing tank breakup and by adding guidance and control subsystems. Cost for both ship and aircraft surveillance were determined. (See Ref. 2.15.3-11 for details.)

The droptank ecological study considered two items: (1) ocean contamination by aluminum and titanium tanks and (2) polyurethane tank insulation contamination of the atmosphere. In both studies, no detrimental effects to the environment could be found. (See Refs. 2.15.3-12 and -13 for the complete studies.)

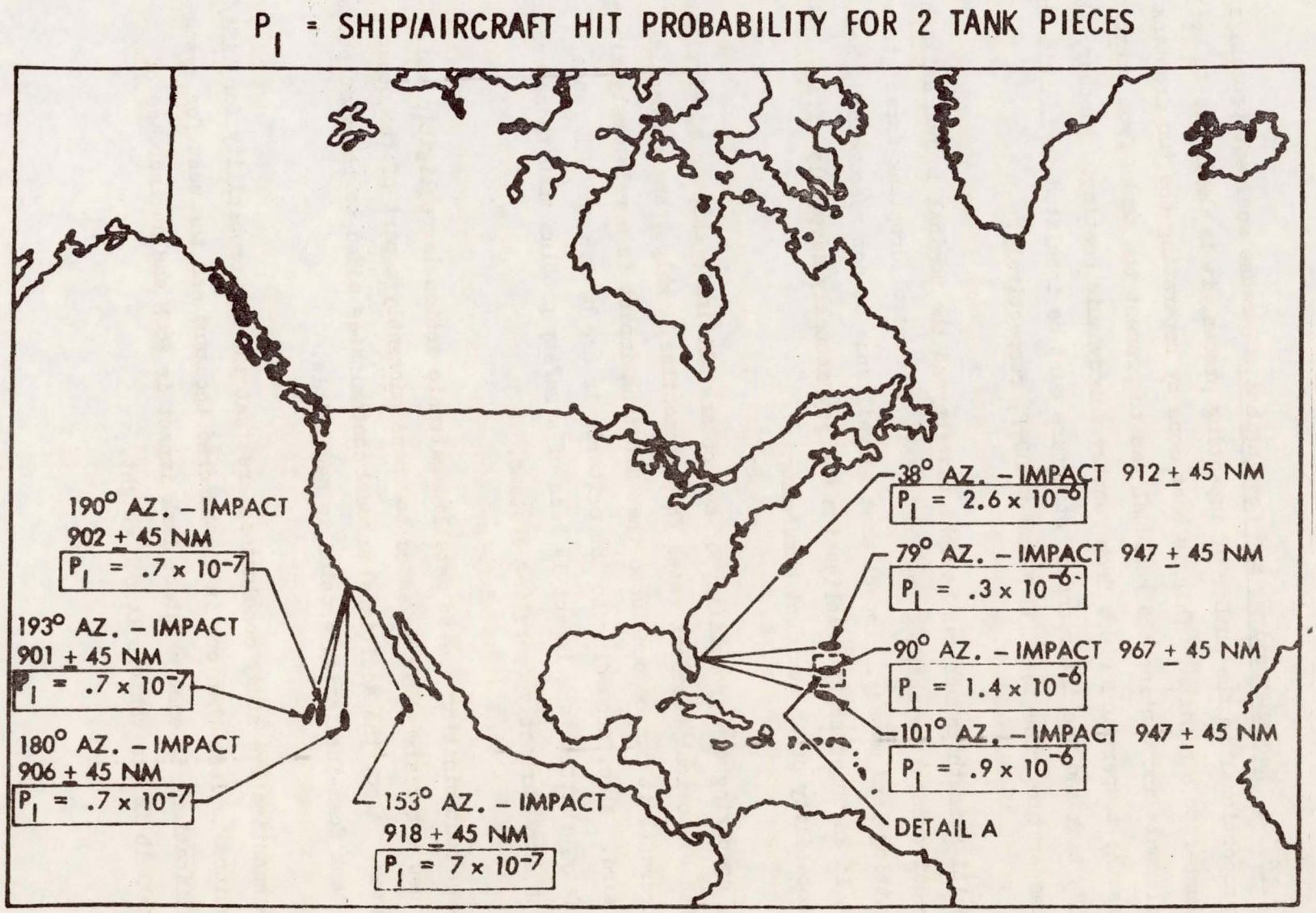

Fig. 2.15.3-3 Mission Model Tank Impact Probabilities

2.15.3.4 Recommendations and Conclusions. Because accident probability is proportional to the number of impacting pieces, it is desirable to keep this number to a minimum. This is best done by separating the two droptank elements and venting the residual gas to prevent the tanks from bursting as the temperature rises from reentry aerodynamic heating. Further, the LH_2 tank and the interconnect structure must be insulated to inhibit the temperature rise to + 430°F and + 600°F, respectively.

Multi-azimuth launch can be achieved (beyond the nominal launch azimuth requirements) by shaping the trajectory to control droptank impact to avoid hitting land masses. Impact area surveillance is not recommended, because it is an unnecessary complication and may actually adversely affect the probability of an accident occurring.

By comparing the probability of a droptank hitting a ship or an airplane with historical accident rates for automobiles, ships, and commercial aircraft, it can be concluded that droptank impact is a relatively minor hazard. Also, according to the criteria in use by ETR and WTR range safety, shuttle tank impact is less of a safety problem than those associated with many currently operating systems.

Ocean contamination due to metallic galvanic action is negligible and no new materials would be added to those currently a part of the ocean system. Also, the additional aerosol contaminates added to the atmosphere by tank foam-insulation erosion is negligible.

The quantitative study conclusions are that the hit probability for NASA/DOD missions is less than one in one-hundred thousand and the cost for system modifications to ensure intact tank impact is $60M and an increase of 32,000 lb in the vehicle launch weight.

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2.15.4 Droptank Impact and Dispersion

This section presents a summary of the Task 1 effort of the Alternate Concepts Study. The objective of this task was to investigate the problems associated with droptank impact and dispersion through predictions of the post-staging behavior of the expended droptanks, by determining cost and weight estimates for minimum dispersion design and operating concepts, and by evaluating the hazards associated with environment contamination and interference of air and sea traffic.

The LS 200-5 vehicle design for a total of 750 missions over 10 years of operations was used as a baseline for this study. The nominal mission launch azimuths used were:

ETR	WTR
30 deg	153 deg
79 deg	180 deg
90 deg	190 deg
101 deg	193 deg

Trajectory analyses, including tumbling tank six-degree-of-freedom trajectories, were conducted to provide basic input data to thermostructural, dispersion, and risk-hazard analyses. Also, trajectories were provided for a multi-azimuth launch study which shows stage-and-one-half launch capability from ETR and WTR at any azimuth.

The summary is divided into three study catagories and presents only the conclusions and highlights of each. The backup detail and in-depth analysis developed to support this summary is referenced throughout and is available in either Appendix A of this document or in previously published monthly status reports in Engineering Memorandum format.

2.16 SAFETY AND ABORT

Assessment of potential abort modes for the stage-and-one-half system was conducted in parallel with the system safety analysis; it recognized the hazard categories outlined in NASA Safety Program Directive 1A, dated 12 December 1969. The identified hazards are associated with conditions of environment, personnel error, design deficiency, procedural deficiency, or subsystem failure that threaten loss, damage, or degradation of the system; accordingly, they are classified as catastrophic, critical, or marginal hazards.

2.16.1 Guidelines

Pertinent guidelines are:

- Intact abort capability is required.
- Safe mission termination capabilities should allow for intact separation of the orbiter from the propellant tanks, in the case of the stage-and-one-half, following liftoff.

These requirements reflect a concern with presumed failure modes that dictate an atomospheric abort with return to base, as an alternative to the primary mode of abort to orbit. These abort mode concepts are depicted in Figs. 2.16-1 and 2.16-2.

2.16.2 Safety Philosophy

In discussing hazard and abort philosophy, it is necessary to distinguish between catastrophic and noncatastrophic conditions, and to consider essential differences between stage-and-one-half and 2-stage system characteristics. In the Space Shuttle design approach, about the only potential hazard area subject to rigorous engineering analysis is component and subsystem failure. Intuitive design provisions are

ABORT-TO-ORBIT OPTIONS

- RETURN WITH ORBITER INTACT
- REPAIR AND RETURN
- TRANSFER TO SPACE STATION
- EVA RESCUE

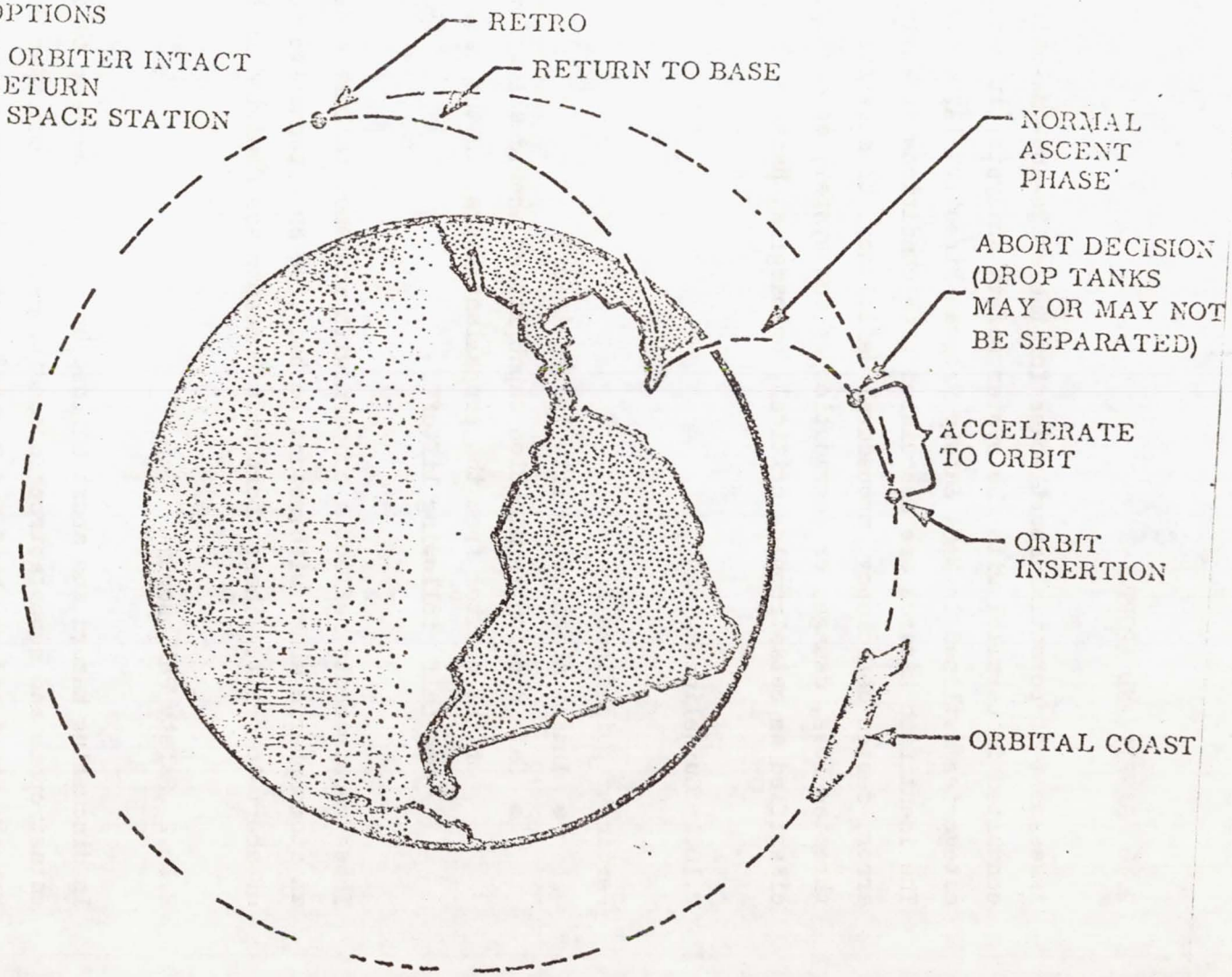


Fig. 2.16-1 Primary Abort Sequence

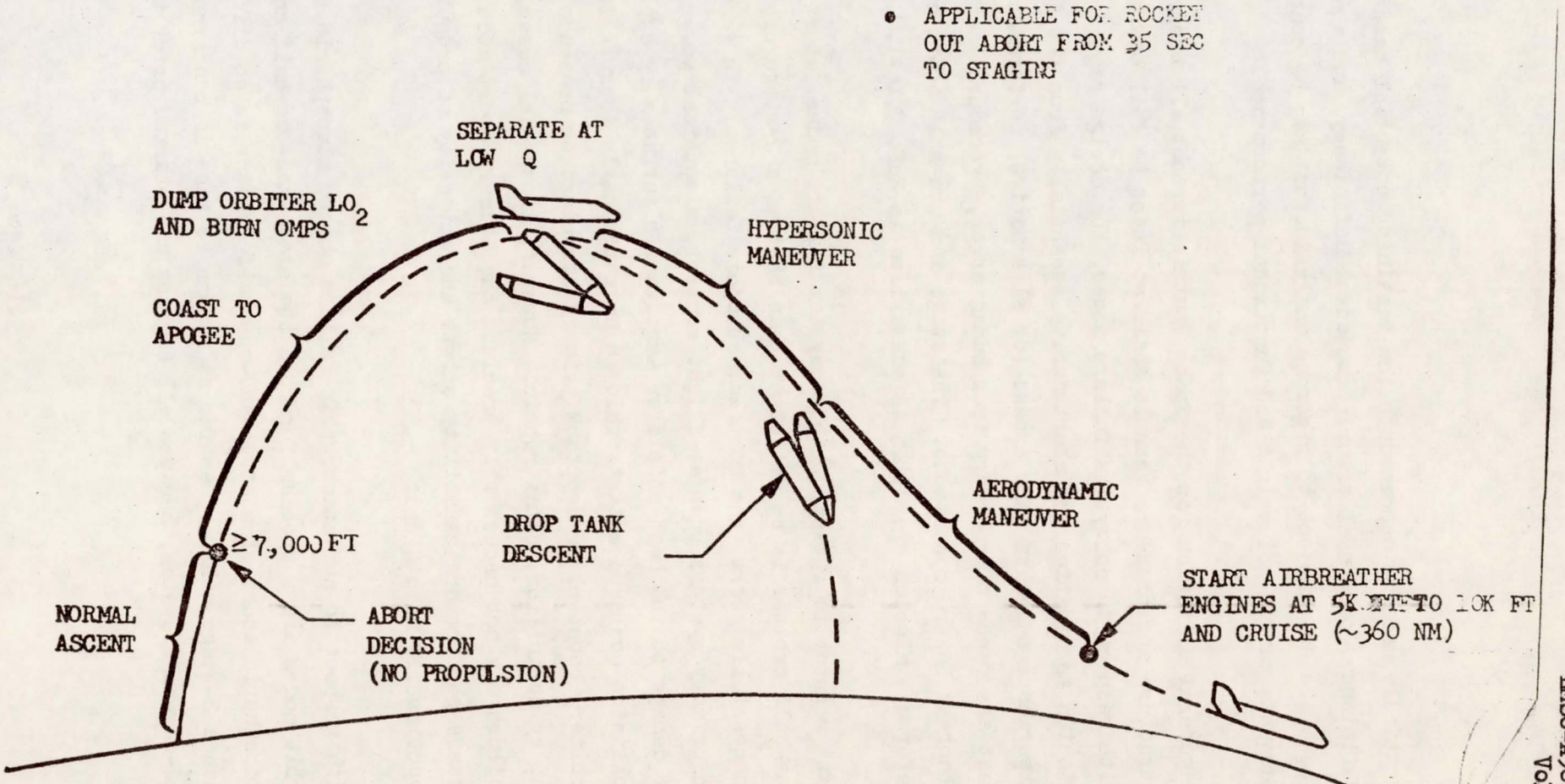


Fig. 2.16-2 Secondary Abort Sequence

routinely made to accommodate the remaining potential hazards due to design deficiency, personnel error, procedural deficiency, and abnormal environment. These are not subject to rigorous analysis, but may be compensated for by adequate margins of weight and functional performance.

A general criterion for the Space Shuttle is that all subsystems except primary structure and pressure vessels shall at least be fail operational-fail safe. With respect to subsystem failure modes, the design requirements for safe mission termination and abort derive essentially from the fail safe conditions. Experience suggests that excess (or alternative) performance capability is the best assurance of recovery from human error, procedural deficiency, and other abnormal hazard conditions. The stage-and-one-half abort capability for the Reference Mission operation is summarized in Ref. 2.16-1.

With respect to catastrophic subsystem failure modes, abort of the stage-and one-half orbiter is no different than that of a 2-stage orbiter; neither could successfully abort from most catastrophic failures. In general, total failures in primary systems such as flight control, electrical power, attitude control, or damage to the heat shield or aerodynamic surfaces by explosion, fire, or collision would be catastrophic in an atmospheric abort. Total failure of orbiter, booster, or droptank primary structure or pressure vessel containment is, by definition, catastrophic. However, in special cases, a single catastrophic failure may not necessarily preclude the primary abort mode through orbit if excess performance capability exists and integrity of propulsion and control functions is maintained.

With respect to noncatastrophic failure modes, atmospheric abort and abort to orbit modes are a problem for both the stage-and-one-half and 2-stage systems. Atmospheric abort of a stage-and-one-half orbiter is no different in principle than a 2-stage orbiter, as far as return to base is concerned. The stage-and-one-half system, because all engines are started prior to liftoff, is not

subject to the hazards of igniting orbiter engines in flight at staging, as is the 2-stage system, and droptank staging occurs at high altitude (272,000 feet) in nominal ascent trajectories. The 2-stage orbiter would necessarily be committed to an atmospheric abort maneuver if engines failed to ignite after staging. On the other hand, stage-and-one-half can afford multiple engine-out conditions which do not compromise its ability to safely achieve low orbit and, possibly, complete the mission. Furthermore, the FO-FS criterion is adequately satisfied by the stage-and-one-half orbiter primary rocket system at staging, but the same is not true of the 2-stage orbiter. For the latter, if one of the two engines fails to ignite, the 2-stage orbiter is automatically in a fail safe mode. If both engines fail to ignite, the vehicle will be lost without the availability of LH_2 airbreather engines.

The position established for this current study is that flight safety and abort considerations for mission phases beyond injection to the reference orbit are identical for stage-and-one-half and 2-stage concepts, and were not extensively analyzed.

2.16.3 Orbiter Post-abort Trajectory Analyses

Fundamental to any abort analysis for a flight vehicle is the return trajectory. The post-abort trajectory analyses for the stage-and-one-half system were conducted using the nominal ascent trajectory reported in Ref. 2.16-2. The following premises are pertinent to this analysis:

1. There is a noncatastrophic primary propulsion failure.
2. The droptanks are separated.
3. The orbiter liquid oxygen may be dumped, if necessary, although the analysis was done with and without oxygen onboard.

The ascent trajectory shape sets the entry conditions when abort occurs before staging. Although velocity at this point is suborbital, the operating environment is of concern since flight path angle cannot be controlled after abort and the vehicle is heavier than during a nominal reentry. The result is that care must be taken to avoid high surface temperatures, dynamic pressures, and accelerations.

One question to be answered is, at what point on the ascent trajectory does effect of flight path angle and velocity become critical? In addition, what is the effect of considering two reentry weights, simulating the case where there is no dumping and the case where oxygen alone is dumped? It is assumed that hydrogen is kept onboard for jet engine operation.

Trajectory control, in the event of thrust failure, is maintained by determining angle of attack and bank angle. In the analysis performed, it was assumed that these could be controlled to desired values, with sideslip angle maintained at zero. The first steps in the study involved a fixed angle of attack mode. A high angle of attack was selected because experience indicates that this results in lower temperature at pull-up.

In every case, the bank angle was held at zero (wings level) until pull-up. After pull-up, various bank programs were used. A phugoid oscillation results from continuing with zero bank. This motion is readily damped by modulating the bank angle.

The analysis showed that for an abort at 249 seconds after launch, the orbiter trajectory plot substantially penetrated the nominal 2300°F boundary. With oxygen removed, this was not so.

With abort at the later time point, (291 seconds after liftoff) there was penetration of the 2300°F line with or without oxygen dumping. A study of angle of attack modulation combined with oxygen dumping therefore was undertaken to alleviate the high temperature conditions existing during pull-up, as reported in Ref. 2.16-3. While the vehicle does plunge deeper into the atmosphere, α_o was modulated so that, for every combination of α, altitude and velocity, the 2300°F lower surface temperature is not exceeded. Dynamic pressures approached 830 psf, however, which is unacceptable. A compromise between a high temperature trajectory and the high $\bar{q}$ trajectory was indicated which could be achieved by employing a command angle-of-attack modulation.

The angle of attack, α, becomes a control parameter. As in the previous work, α was maintained at 55 deg until the temperature limit of 2300°F was reached. After this point α was changed by the rate

$$\dot{\alpha} = \frac{\alpha_c - \alpha}{\tau}$$

where τ is a constant input to the computer program and α_c is determined so that if at any time $\alpha = \alpha_c$ then the vehicle is exactly at the temperature limit for that α. Thus, if at any time $\dot{\alpha} = 0$, the maximum lower surface temperature is 2300°F. However the temperature increase with finite values of $\dot{\alpha}$ is small for values under five. A value of four (4) would result in a lower surface temperature of about 2340°F and a maximum dynamic pressure of about 300 psf. These values appear to be acceptable for an abort operation.

The results of the trajectory analysis established feasibility and indicated a probable need for an orbiter LO_2 dump capability.

2.16.4 Post-Abort Flight Operations

Having established the entry feasibility of the orbiter resulting from a prestaging mission termination, there were several systems analyses required to further establish the overall feasibility of the post-abort operations. Basically, these are divided into three distinct operations:

(a) The feasibility of atmospheric separation of the droptank from the unpowered orbiter.
(b) The establishment of feasibility and requirements for dumping the orbiter ascent liquid oxygen.
(c) The subsonic cruise capability of the orbiter to return to a safe landing.

These analyses are summarized in the following sections.

2.16.4.1 Droptank Separation for Atmospheric Abort. The trajectory analysis established the feasibility of returning the orbiter safely to low altitude subsonic flight, so that it was then necessary to investigate the feasibility of safely separating the droptanks from the orbiter during an atmospheric post-abort operation. This analysis is reported in Ref. 2.16-4.

The time of abort (due to a noncatastrophic propulsion shutdown) was selected at 100 sec after liftoff as representative of a relatively severe operating environment. The flight conditions at this time are:

(1) Mach No. $\approx$ 2
(2) Altitude $\approx$ 60,500 ft
(3) Dynamic Pressure $\approx$ 430 psf

The inherent stability of the composite vehicle, plus the activation of the ACPS, would allow the vehicle to coast to lower dynamic pressures which would alleviate the separation conditions. The composite vehicle was allowed to coast to a dynamic pressure of 100 psf at Mach 1.6 and an altitude of 80,500 ft. The abort dynamics, with and without droptank LO_2 onboard, were analyzed. The ACPS provided a 7.2×10^5 ft-lb maximum couple during the coast and separation. The analysis established the feasibility of accomplishing droptank separation for the stated conditions.

2.16.4.2 Orbiter Liquid Oxygen Dump. It is necessary to dump the orbiter liquid oxygen in order to lighten the vehicle for the flyback operation. A dump analysis was conducted and is reported in Ref. 2.16-5.

The LO_2 dump analysis showed that by using a blowdown LO_2 self-pressurization approach to oxygen dumping, coupled with the dynamic head (which is a comparatively minor contribution), the oxygen can be dumped satisfactorily during a post-abort operation. The dump time required is of the order of 40 to 50 sec. This short time is required for an abort early in the ascent (30 to 60 sec) and at later times ranging from abort 230 to 299 sec. The early phase is altitude critical in that the LO_2 must be dumped rapidly so that the orbiter can initiate aero-cruise operations before crashing. Rapid dump is required for later times to accomodate entry by reducing the orbiter wing loading. Two dump lines of 16 in. diameter (one for each tank) will allow the LO_2 to be dumped in 40 to 60 sec, depending on the atmospheric and flight conditions. This reduces the period during which recovery is not possible (the so-called "deadband") to about the first 35 sec after liftoff.

The installed weight of the oxygen dump hardware is estimated to be about 355 lb. This is equivalent to 390 lb of payload (including the contingency), or about 8,600 lb of the launch weight.

2.16.4.3 Post-Abort Orbiter Flyback. The post-abort flyback (cruise) capability of the orbiter was analyzed and reported in Ref. 2.16-6. The analysis determined the airbreather thrust, the flight altitude, and range capabilities of the LS 200-3 orbiter. The basic premise of this analysis was that the airbreather engines could operate with liquid hydrogen fuel and that the orbiter ascent of LH_2 was available for such an operation. It was further assumed that the orbiter ascent liquid oxygen had been dumped and that OMPS propellant had been expended prior to the cruise operation.

The analysis established that the orbiter (including a 25K payload), using the five airbreather engines required for go-around, can maintain level flight at altitudes above 2,500 ft, depending on which engine design is used. The level flight altitude for the lower thrust "A" engine modification is about 2,500 ft and is 4,000 ft for the "B" version with increased thrust. The corresponding cruise ranges are 350 nm and 330 nm.

Furthermore, most of the anticipated launches (about 95 percent) are predicted for four inclinations; 28.5, 35, 55, and 90 degs. For primary propulsion failures from about 35 sec to 200 sec, the orbiter can return to base regardless of the inclination. Beyond 200 sec, downrange or crossrange landing sites must be considered. For the 55 deg inclination, landings may be made along the eastern seaboard from North Carolina to New York. For inclinations from 28.5 deg to 35 deg, Bermuda Island is about the only possibility. For polar orbits out of WTR, Baja, California is a possibility.

A special reminder is in order at this point. This analysis was accomplished when there was a system requirement for go-around airbreather engines which operated on liquid hydrogen fuel. Since then the fuel has been changed to JP, which effectively eliminates the post-abort cruise capability. Consequently, a preorbital mission termination resulting from an orbiter primary propulsion failure will, a priori, result in the loss of the craft.

2.16.5 Reliability

The reliability analyses for this study were necessarily restricted by budget allocations because of the nature of the contract which was to establish system feasibility. These analyses were consequently exploratory in nature, tending to identify operating failure modes and to establish analytical concepts.

2.16.5.1 Failure Modes By Mission Phase. An analysis was conducted which postulated catastrophic failure probabilities by mission phase, from which were derived noncatastrophic mission phase reliability allocations. This analysis is reported in Ref. 2.16-7.

Design emphasis and attendant cost apportionment can be based on the relative probability of abort-forcing failures. This emphasis must be supported by simulation analysis which would be accomplished as systems design definition becomes available. Calculations for the catastrophic failures were based on time-dependent reliability estimates for the rocket engines and flight history of similar systems. The reliability apportionment per phase is based on the expected complexity of the operational subsystem.

The distribution of all the potential failure modes is the "fallout" from reliability apportionment and the catastrophic failures. The mix (distribution) of the type of failure modes would be determined from an operational simulation.

2.16.5.2 Simulation Model Example. Because of the importance of the primary propulsion system and the relative maturity of the design definition, this subsystem was selected as the most promising candidate for simulation modeling. This model is described in Ref. 2.16-8.

The study presents a sequence of events table, a functional subsystem schematic, the reliability functional block diagrams, and the related reliability equations; these are the basic inputs for an ascent propellant simulation model program. A data flow and computer program are presented and a preliminary sequence of events has been developed for the ascent propellant system, based on a preliminary functional schematic. Preliminary reliability block diagrams and equations for the preflight phase were also prepared. Nonfunctional elements such as ducts, gimbals, sliding joints, and manifolds would be included as part of the simulation model as the design disclosure develops.

2.16.5.3 Primary Propellant Subsystem Simulation. It was desired to assess the postulated technique for the optimization of reliability and safety considerations which would ultimately be subjected to the constraints of cost, weight, and volume. An applicable computer program (code name DRIVEW) was provided to LMSC by the Aerospace Corporation. The computer simulation was conducted to determine: (1) the frequency of catastrophic failures and intact abort modes, and (2) the effect of failure frequency for alternative configurations. This analysis is reported in Ref. 2.16-9.

The computer program determines, from among alternative candidate redundancies and competing design life increments, that combination of redundancy and limited lifetime expendables which minimizes the cost of ownership to carry out a specified mission with a system of given performance capability. It assumes that some weight margin is available over the minimum serial weight which can be allocated either to equipment redundancy or increased expendables.

The acceptable solutions are limited to those which do not exceed a maximum weight limit, as well as a maximum weight budget for expendables. It also can treat an additional constraint, namely "volume" (but was not actually so restricted) and will terminate the analysis if this constraint is violated.

Although the program has the capability to optimize cost and weight against reliability, these parameters were suppressed to perform a reliability trade analysis only. All devices were set to be one lb and one dollar with an optimum of maximum weight and cost equal to one times the number of units.

The resultant catastrophic failure rates for the propellant subsystem, per phase, which are indicated by this analysis show an order of magnitude greater than the earlier estimates reported in EM L2-06-01-M8-1. Alternative configurations in specific areas may be subsequently compared, which could result in recommendations for subsystem changes. The reliability gain through these changes may then further be assessed through the optimization characteristics of the computer program. When the program is expanded to include all subsystems, it would define an optimal compromise between weight, volume, and cost with regard to reliability.

2.16.6 Abort and Safety Results. It is analytically shown that the stage-and-one-half orbiter can be successfully returned to safe landing for the reference mission, either (1) by proceeding to abort through orbit (primary mode) if possible, or (2) coasting to conditions which are conducive to drop-tank separation, dumping oxygen, and returning to land by cruising the orbiter, operating the airbreather engines with the unused ascent liquid hydrogen. The analysis also shows the following:

- The orbiter can satisfactorily reenter with or without the ascent oxygen on board during abort from the worst condition, with the reservation that a compromise between temperature and loads is required for the worst condition.

- Engine(s) out or subnominal thrust
  (a) The mission may be completed with one engine out on the launch vehicle or one of three engines out during orbiter operations.
  (b) An abort through orbit can be accomplished with three engines out (after 20 secs and with 10 percent overthrust) on the launch vehicle or with two of three engines out on the orbiter.
  (c) In the event that the droptanks fail to separate after a nominal ascent, the orbiter has sufficient capability to accelerate the composite configuration into the 50 nm orbit.

REFERENCES

2.16-1 EM L2-06-01-M7-1, Reference Mission Abort Capability For The Stage-and-One-Half LS 200-2 Configuration, Third Letter Progress Report, LMSC-A977725, 15 October 1970.

2.16-2 EM L2-06-01-M2-10, Reentry From Abort During Ascent, Third Letter Progress Report, LMSC-A977725, 15 October 1970.

2.16-3 EM L2-06-01-M2-12, Reentry From Abort With Angle-Of-Attack Modulation, Fourth Letter Progress Report, LMSC-A980360, 15 November 1970.

2.16-4 EM L2-01-M1-6, Atmospheric Abort Staging Of The LS 200-5 Stage-and-One-Half Launch Vehicle, Sixth Letter Progress Report, LMSC-A984010, 15 January 1971.

2.16-5 EM L2-06-01-M7-3, Determination Of The Liquid Oxygen Dump Capability For The LS 200-3 Orbiter, Fifth Letter Progress Report, LMSC-A981489, 15 December 1970.

2.16-6 EM L2-06-01-M7-2, Secondary Abort Cruise Requirements and Capabilities For The LS 200-3 Orbiter, Fifth Letter Progress Report, LMSC-A981489, 15 December 1970.

2.16-7 EM L2-06-01-M8-1, Abort Occurrance Phased Probability Distributions, Third Letter Progress Report, LMSC-A977725, 15 October 1970.

2.16-8 EM L2-06-01-M8-2, Reliability/Safety Simulation Model-Ascent Propellant System, Fourth Letter Progress Report, 15 November 1970.

2.16-9 EM L2-06-01-M8-3, Propellant Subsystem-Optimum Allocation of Redundancy, Fifth Letter Progress Report, LMSC-A981489, 15 December 1970.

2.17 OPERATIONS

An integral part of the Alternate Concepts Study has been the consideration and evaluation of various methods of performing shuttle operations. Some operations analysis relating directly to the droptanks was performed in great detail and is included in other subsections of this report. This subsection presents the results of studies on the overall shuttle operating techniques examined to support a Stage-and-One-Half Space Shuttle Program and highlights those areas peculiar to stage-and-one-half operation. Most of the supporting analysis and backup data for this subsection can be found in Reference 2.17-1 and in appropriate subsections of Reference 2.17-2.

2.17.1 Ground Turnaround Operations

A baseline operations concept to support a stage-and-one-half shuttle was evolved on the basis of maximum use of existing facilities and equipment. This concept is conceived to utilize existing Saturn/Apollo equipment and facilities insofar as possible. The Main Launch Base is at KSC and includes use of the Vertical Assembly Building, Launch Pad 39, some existing support areas, and a new airfield, Tank Manufacturing Building, and Maintenance Annex. This concept is shown pictorially in Figure 2.17-1.

This subsection contains a description of all major ground operational support elements from landing to launch. It includes functional flow diagrams and timelines, a description of major facilities required for the operational base and a base manpower allocation to support the baseline operations concept.

2.17.1.1 Functional Flow. The operating methodology to support KSC as the Main Launch Base is shown by the Functional Flow Diagram in Figure 2.17-2. This figure shows operation phase 6.0 (Mate Droptanks to Vehicle and Checkout) highlighted, as it will be expanded in this report to illustrate the methodology followed in all the eleven phases shown. It was picked because it is the phase most peculiar to stage-and-one-half vehicle support operations.

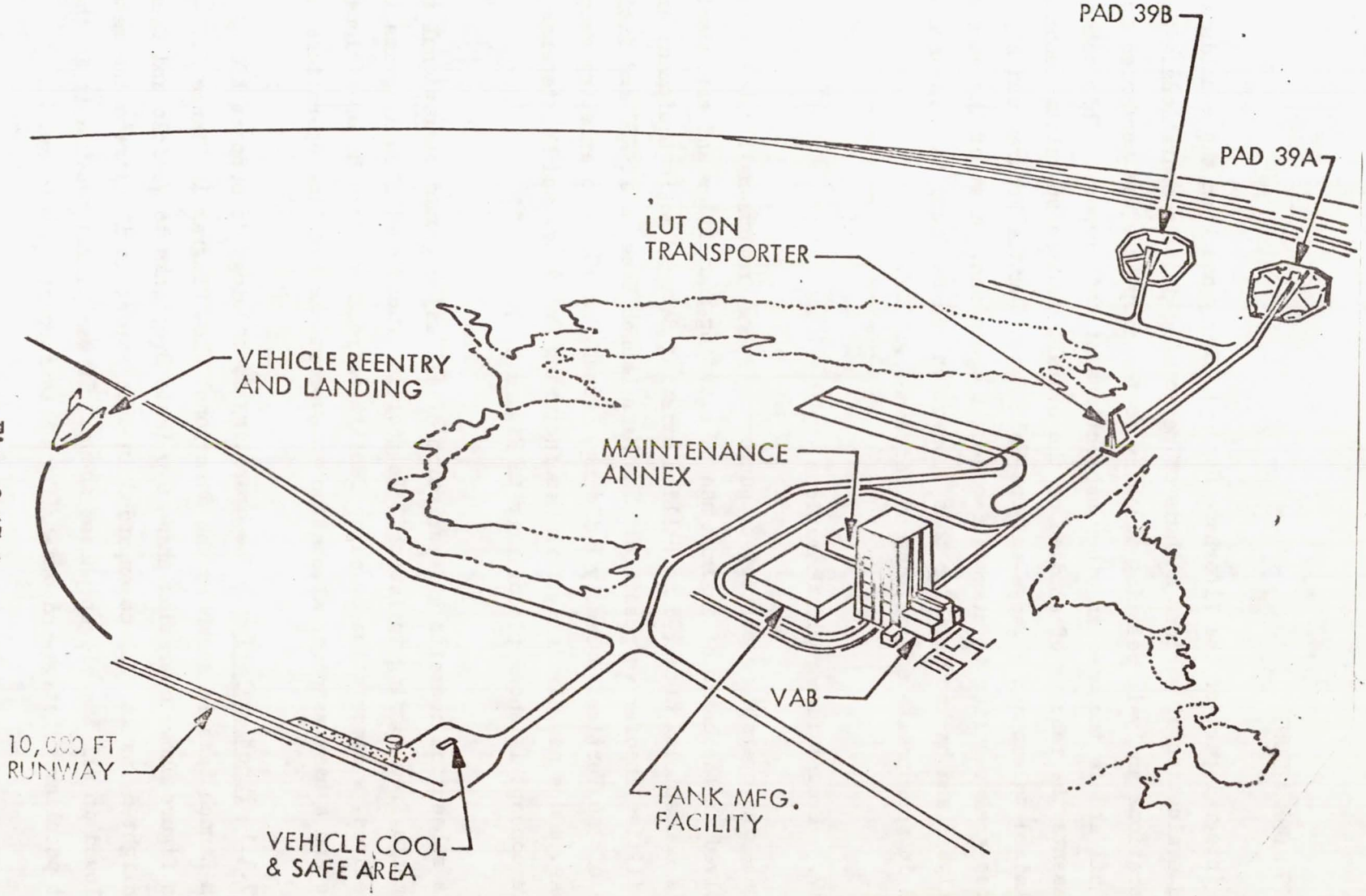


Fig. 2.17-1 Launch Site Facilities (KSC)

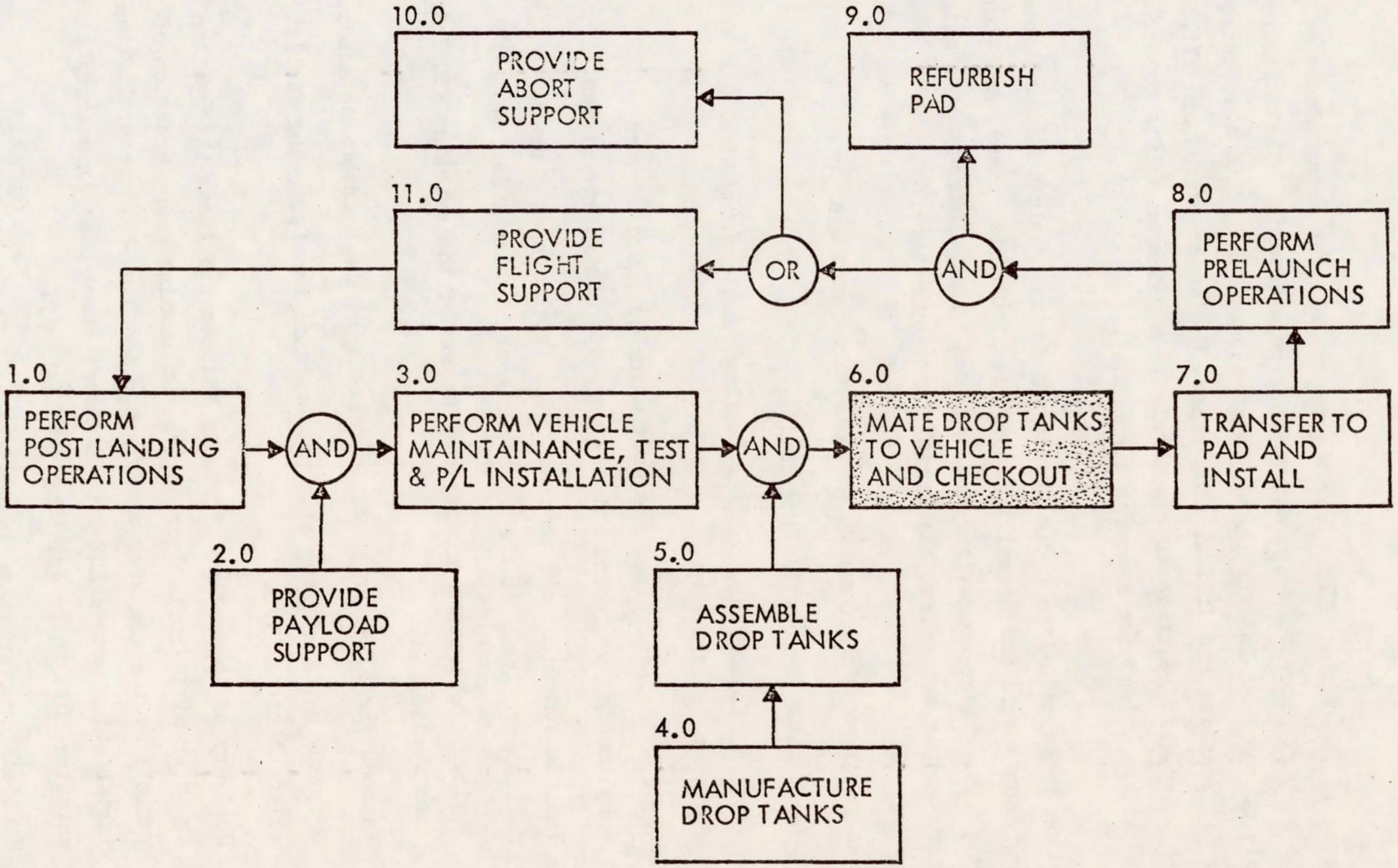


Fig. 2.17-2 Operations Flow

This overall functional flow diagram presents all the major phases of the shuttle support operations and shows their relationship. The mainline cycle consists of the Post-Landing phase, Maintenance phase, Mate and Checkout phase, Pad Transfer and Installation phase, Prelaunch phase, and Flight Support phase. It requires approximately 19 eight-hour shifts to accomplish the ground operations for a normal turnaround.

Supporting these main-line cycle operations is the Payload Support phase, Droptank Manufacture and Assembly phases, Pad Refurbish phase, and Abort Support phase. These operations are provided in parallel with the main-line operations and do not affect the normal turnaround time.

The tasks required for a complete vehicle turnaround are:

- Landing at the new landing field.
- Vehicle off-loading, purging, safing, and cooling at a new Purge and Safing Area.
- Vehicle inspection and checkout using on-board systems supplemented by ground stimuli and data management system.
- Vehicle scheduled and unscheduled maintenance at a new Maintenance Annex to the Vertical Assembly Building (VAB).
- Payload preparation and support in one of the existing KSC Industrial Area facilities.
- Payload installation into the vehicle in the Maintenance Annex.
- Droptank manufacturing at a new Droptank Manufacturing Facility adjacent to the VAB.
- Delivery of the vehicle from the Maintenance Annex through the VAB transfer aisle north door on its own landing gear, erection of the vehicle by the transfer aisle crane and high-bay crane, lift and installation onto a LUT-type launcher (same base dimensions as the existing LUT) in an existing VAB high bay.
- Vertical droptank mate in the VAB high-bay cell.

- Transfer to the pad using an existing Crawler/Transporter and installation at the pad on the existing pedestal supports; positioning of a new flame deflector in the existing flame trench; hookup of propellant and other ground support lines to the launcher; loading of propellants, crew, and passengers; countdown, and launch.
- Perform flight support operations as required, or provide abort support if necessary.
- Refurbish pad and launcher and prepare for next use.

An expanded operations flow of the phase 6.0 example is shown in Figure 2.17-3. Tasks 6.3 and 6.4 are highlighted and will be used in the next subsection to illustrate the expanded timeline and job analysis typically performed during the study.

This figure shows a second-level functional flow diagram for the Mate Droptanks to Vehicle and Checkout phase. The actual methods used to perform the tasks in these operational phases are:

Spacecraft Erection and Installation Onto Launcher

- Deliver spacecraft from the Maintenance Annex through the north transfer aisle door of the VAB on the spacecraft's landing gear using a tug.
- Erect spacecraft in the VAB transfer aisle using the transfer aisle crane, a high-bay crane, and a tilt sling erection fixture.
- Install spacecraft onto launcher in a VAB high-bay cell using the high-bay crane, and spacecraft handling sling.
- Hookup all required ground support services to the spacecraft and verify satisfactory operation.

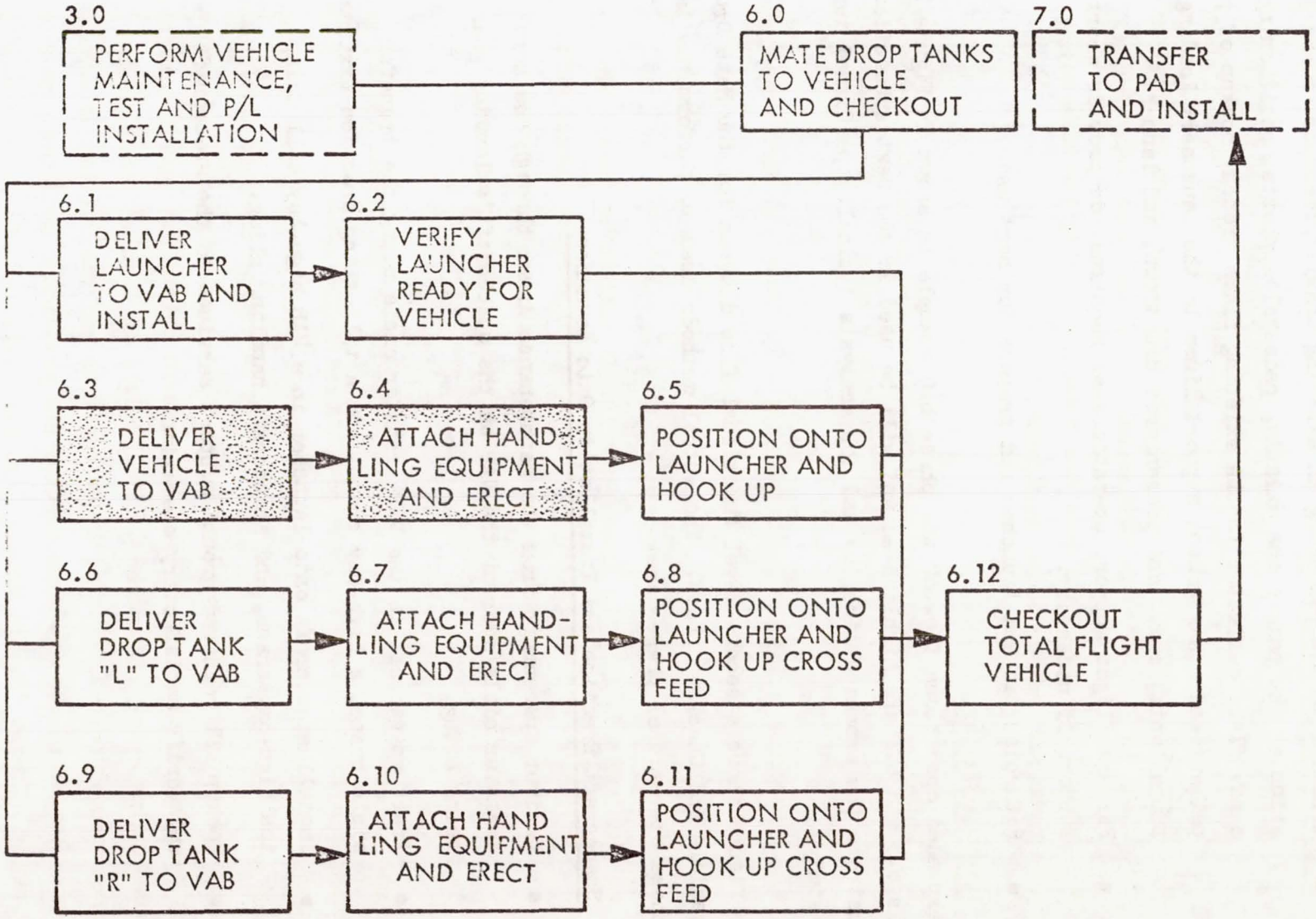


Fig. 2.17-3 Expanded Operations Flow

Droptank Erection and Mate

- Deliver tank half from the Tank Manufacturing Facility through the north transfer aisle door of the VAB on a tank transporter.
- Erect the tank half in the VAB transfer aisle using the high-bay crane and the pivot mechanism built into the tank transporter.
- Install the tank half onto the spacecraft using the high-bay crane and tank-handling equipment.

Droptank Hookup

- Install propellant plumbing between the droptanks, the spacecraft, and the launcher (ground fill hydrants) using the plumbing installation equipment.
- Leak-check all propellant feed and fill lines and verify propellant system ready for launch operations.

Vehicle Complete Systems Test (CST)

- Perform a CST on the launch-configured vehicle to verify the continuity and integrity of the major components - spacecraft, payload, and droptanks - using the onboard checkout system supplemented as necessary by ground-support equipment and operations.

One stage-and-one-half peculiar area associated with this phase is the method of mating the droptanks to the orbiter. Coordination between Ground Operations and Design personnel resulted in a mate-install concept utilizing a common "launch mount adapter" to provide the attach interfaces between the droptanks, vehicle, and launcher. These structures are initially attached to each side of the vehicle and used to provide lifting hard points for vehicle erection and positioning onto the launcher. The vehicle is then attached to the launcher support pedestals using these adapters and the lifting attachments removed. The droptanks are then erected and positioned onto these mounts and attached. These "launch mounts" are basically the aft structure of the

droptanks which initially separate from the launch pedestals with the vehicle on liftoff and then separate from the vehicle with the droptanks and are expended.

2.17.1.2 Timeline Analysis. The total time required for a nominal ground turnaround following a noneventful mission is 19 eight-hour shifts. Figure 2.17-4 gives a breakdown of this time by operational phases. The Post Landing Operations are accomplished in 8 hours and consist of personnel unloading, and vehicle safing and purging. The Vehicle Maintenance Operations are accomplished in 96 hours and consist of inspections, maintenance, acceptance tests and payload loading. The Mating Operations are accomplished in 32 hours and consist of erection and installation of the orbiter and droptanks, hookup of services and crossfeed, and total system checkout. The Transfer-to-Pad Operations are accomplished in 8 hours and consist of the actual transport on the crawler and the positioning and securing at the pad. The Prelaunch Operations are accomplished in 8 hours and consist of installation and hookup of the vehicle, loading and purging, vehicle checkout, and final countdown.

Figure 2.17-5 shows an example of the next level of timeline analysis, breaking the tasks into actual jobs on a per-hour basis. This further breakdown enables a first cut at manpower assignments to be made.

2.17.1.3 Manpower Assignments. Figure 2.17-6 shows an example of an online manpower assignment to a level-three operational task and time breakdown. Over two hundred and fifty (250) jobs were analyzed to arrive at the man-loading assignments required for crew sizing.

Using these man/elapsed time studies and the physical layout of the conceptual main base working facilities, an actual crew assignment was made to define a typical on-line ground operations crew size. Figure 2.17-7 presents these data as individual on-line crews by working area. This assignment is considered typical for the average nominal turnaround during the first year the launch density reaches 75 flights per year. (For the present bogie mission model, this is the 335th launch.)

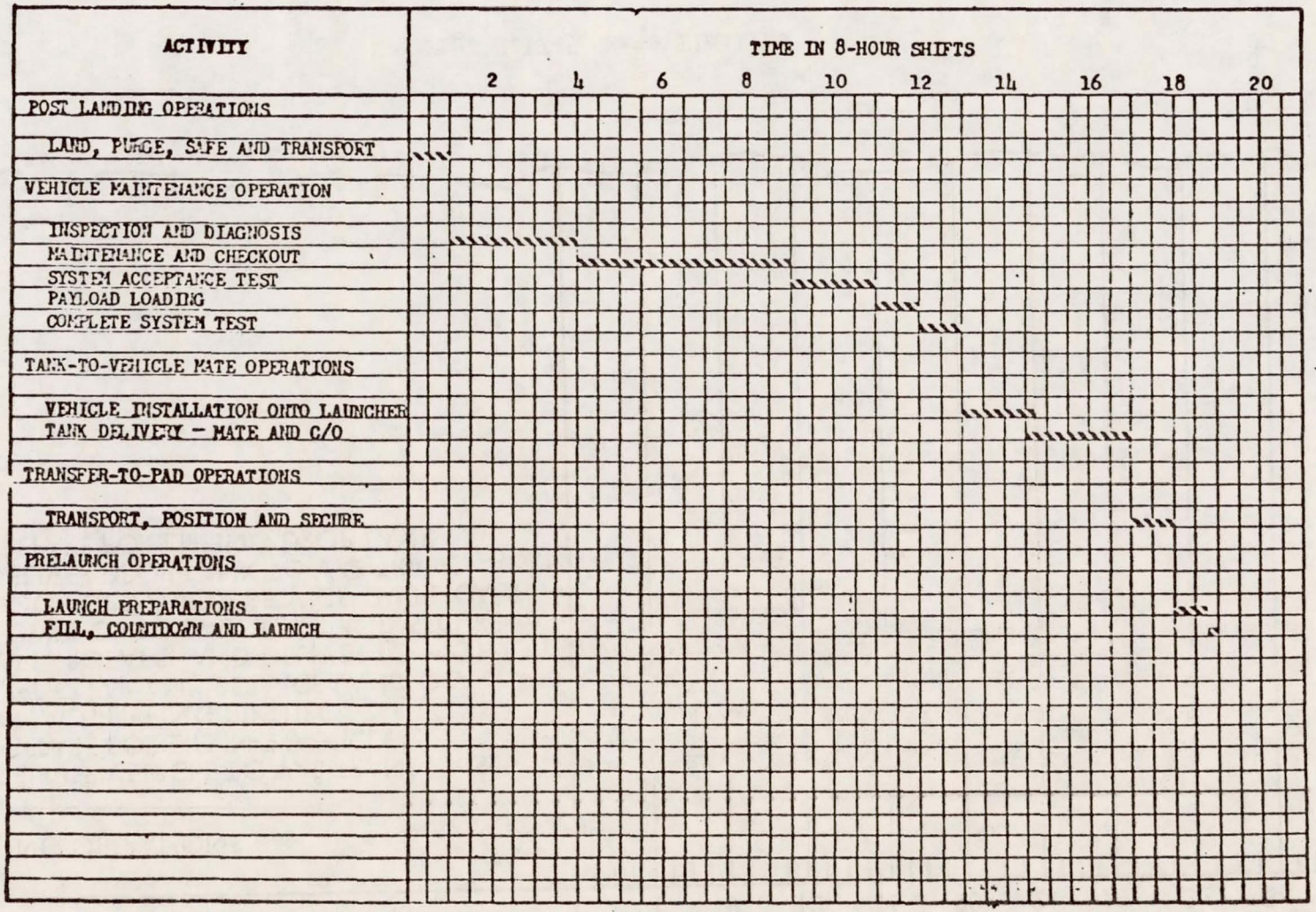


Fig. 2.17-4 Turnaround Estimates for Vehicle

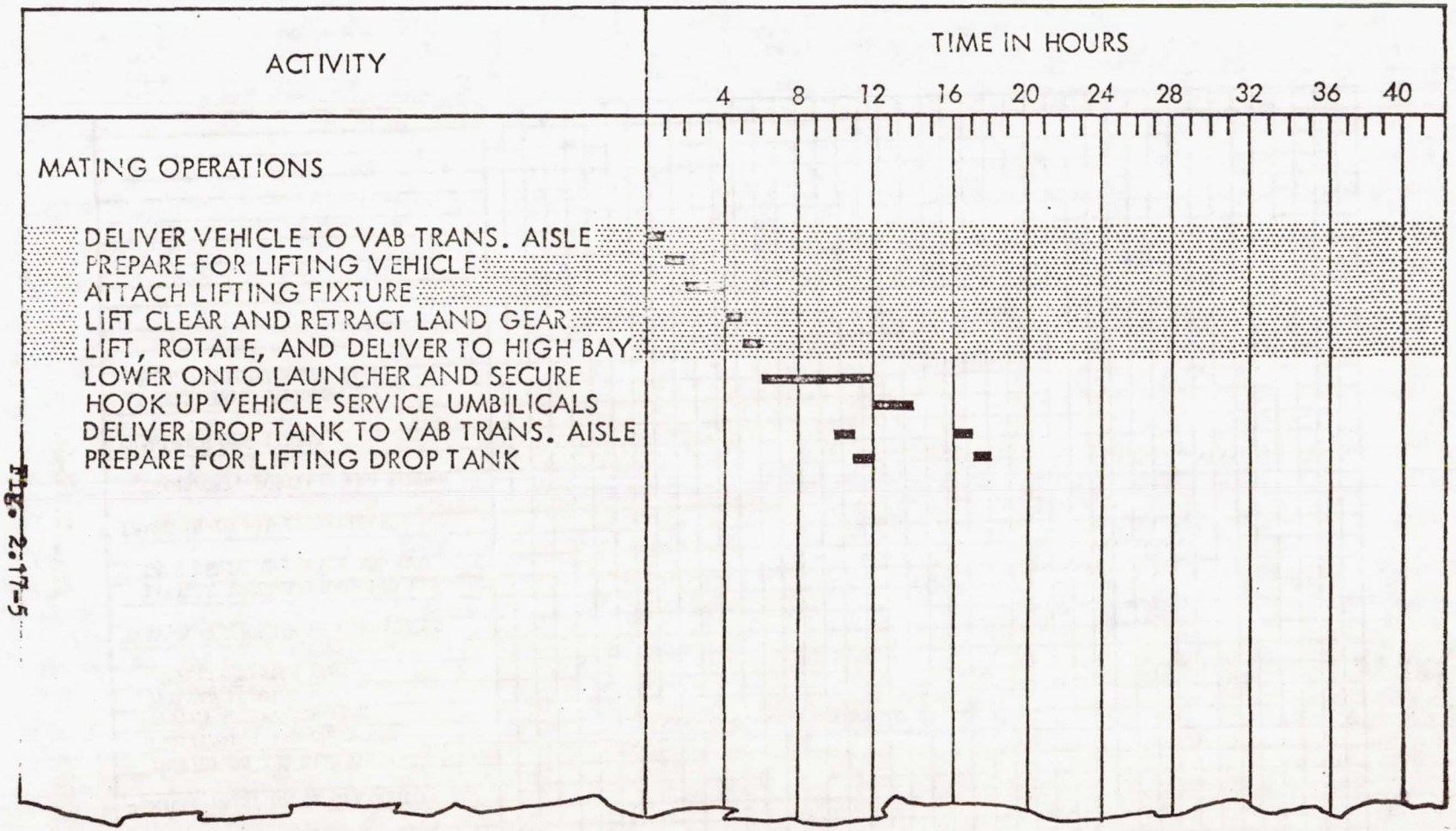
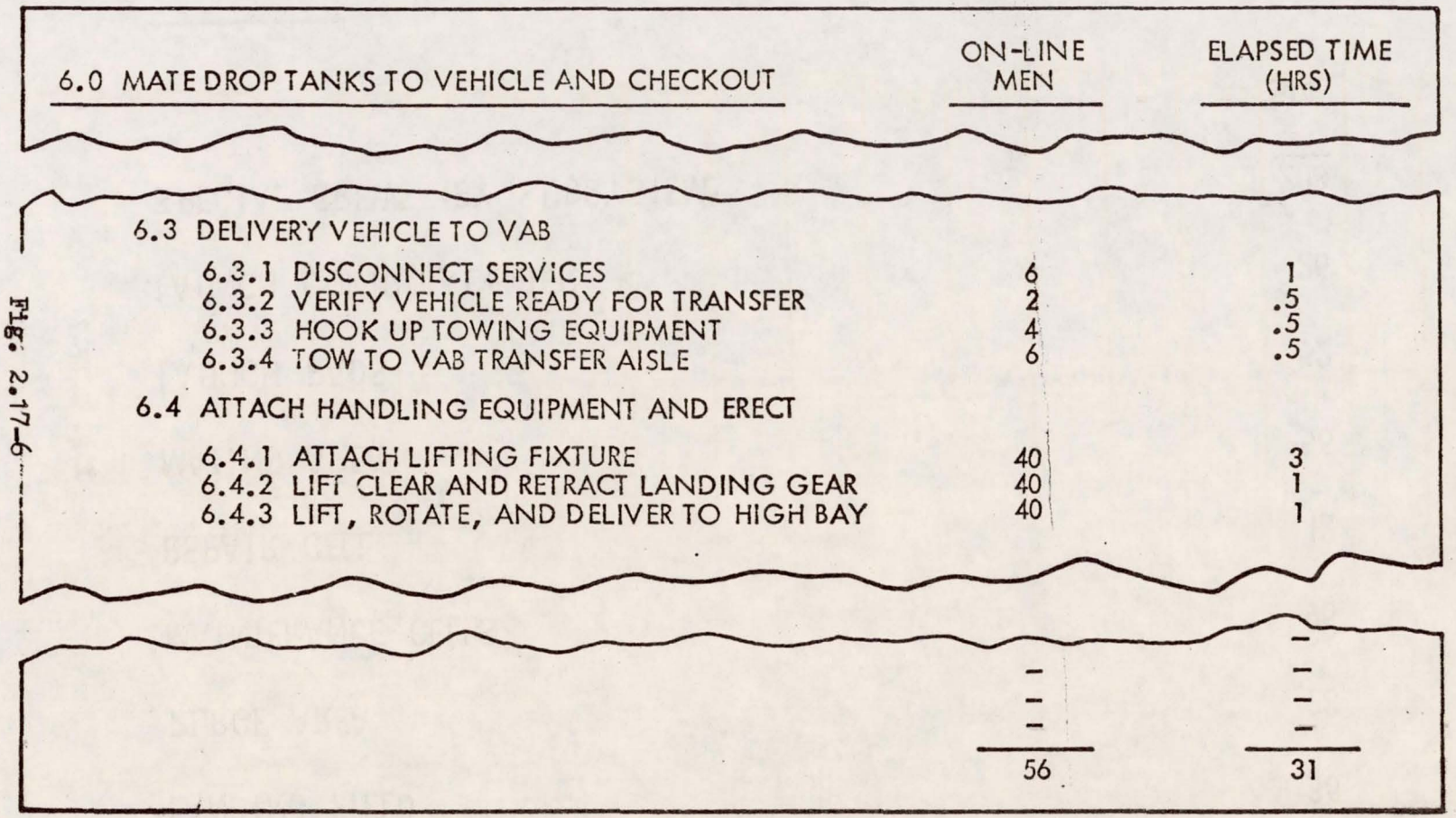


Fig. 2.17-5 Mate Timeline



6.0 MATE DROP TANKS TO VEHICLE AND CHECKOUT	ON-LINE MEN	ELAPSED TIME (HRS)
6.3 DELIVERY VEHICLE TO VAB		
6.3.1 DISCONNECT SERVICES	6	1
6.3.2 VERIFY VEHICLE READY FOR TRANSFER	2	.5
6.3.3 HOOK UP TOWING EQUIPMENT	4	.5
6.3.4 TOW TO VAB TRANSFER AISLE	6	.5
6.4 ATTACH HANDLING EQUIPMENT AND ERECT		
6.4.1 ATTACH LIFTING FIXTURE	40	3
6.4.2 LIFT CLEAR AND RETRACT LANDING GEAR	40	1
6.4.3 LIFT, ROTATE, AND DELIVER TO HIGH BAY	40	1
	–	–
	–	–
	–	–
	56	31

Fig. 2.17-6

Fig. 2.17-6 On-Line Mate Operations (Example)

	MEN
LANDING FIELD	36
PURGE AREA	28
MAINTENANCE CELLS	96
REPAIR CELL	12
MATING AREA	56
LAUNCH PADS	52
LAUNCH CONTROL CENTER	26
SPECIAL CREWS (BY SUBSYSTEM)	208
	514

KSC MAIN BASE, 75 FLIGHTS/YEAR, 5-DAY WEEK, SINGLE SHIFT

Fig. 2.17-7 On-Line Facility Manloading

After determining the on-line working effort in manhours and defining the on-line crews, the next job was to determine the amount of support required to perform the on-line tasks; and, as the shuttle operations should lie somewhere between existing aircraft and space vehicle operations, both were studied to determine existing support to on-line manpower levels. Analysis of both commercial (United Airlines) and military (MAC C-141) crew ratios produced a 6 to 1 support to on-line requirement. An attempt to determine this ratio for space vehicles proved impossible as no distinct on-line and support definition exists, and the actual vehicles are expended and not recycled. So, for the baseline fully operational system, a 6-to-1 ratio was adopted as a starting point. The resulting support crew sizes are shown in Figure 2.17-8.

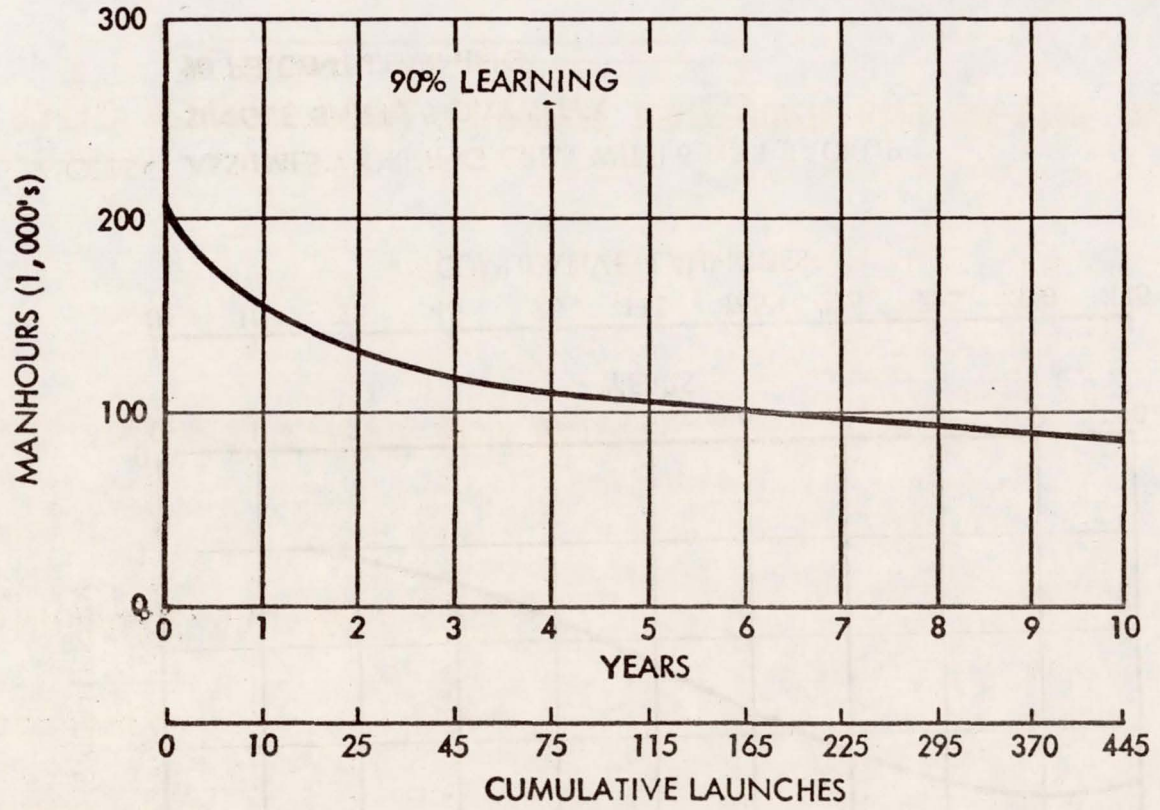
A plot of the main base turnaround manpower per flight was developed using the on-line and support crew sizes shown in Figures 2.17-7 and 2.17-8 (which are typical for the 335th launch in the middle of the 9th year) and projecting backward and forward using a 90 percent learning curve. This plot is shown in Figure 2.17-9. Using the turnaround manpower per flight and the actual flights per year in the 445-launch bogie model, a main base crew complement was determined on a per-year basis. A plot of this main base crew size is presented in Figure 2.17-10 and shows the typical manpower buildup required to support the bogie mission model. Crew size requirements for the first twenty (20) launches are not included in this analysis, as these first launches are part of the development test effort and cannot be rationally projected back from an analysis of the fully operational system.

2.17.1.4 Facilities and Equipment. The existing KSC MILA area including Complex 39, the adjacent KSC Industrial Area, and the undeveloped area north of the VAB are assumed to be the baseline location for the main Shuttle Launch Base. Figure 2.17-11 shows a conceptual layout for this concept. This layout of the Shuttle Main Base shows the actual facilities required to support shuttle operations. The new facilities required are: Purge and Safing Facility for vehicle cooling, safing, and purging; a taxiway connecting

	MEN
LANDING FIELD	144
PURGE AREA	82
MAINTENANCE AND MATE AREA	1,650
SUPPORT SHOPS	655
LAUNCH CONTROL CENTER AND PADS	550
	3,081

6:1 RATIO BETWEEN TOTAL SUPPORT MANPOWER AND ON-LINE MANPOWER

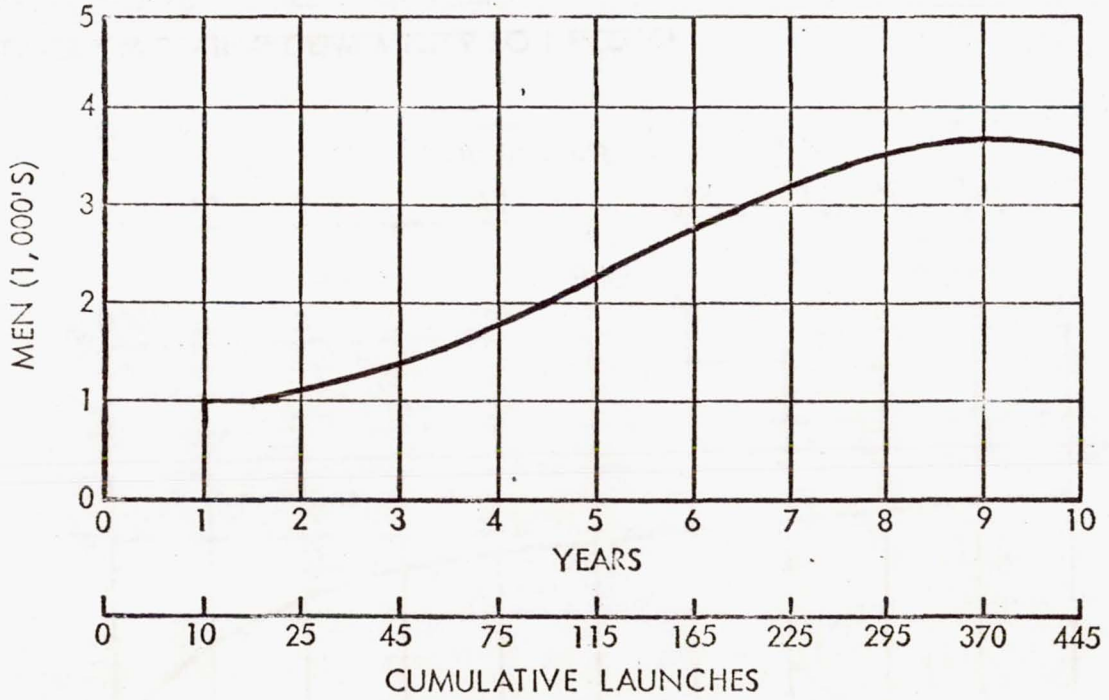
Fig. 2.17-8 Support Manpower



ASSUMES – WORKING CREW WITH 6 TO 1 BACKUP

ASSUMES – SINGLE SHIFT, 5 DAY WEEK

Fig. 2.17-9 Turnaround Manpower Per Flight



NOTES: **ASSUMES** WORKING CREW WITH 6 TO 1 BACKUP
SINGLE SHIFT, 5-DAY WEEK
90 PERCENT LEARNING

Figure 2.17-10

Fig. 2.17-10 Operations Crew Size

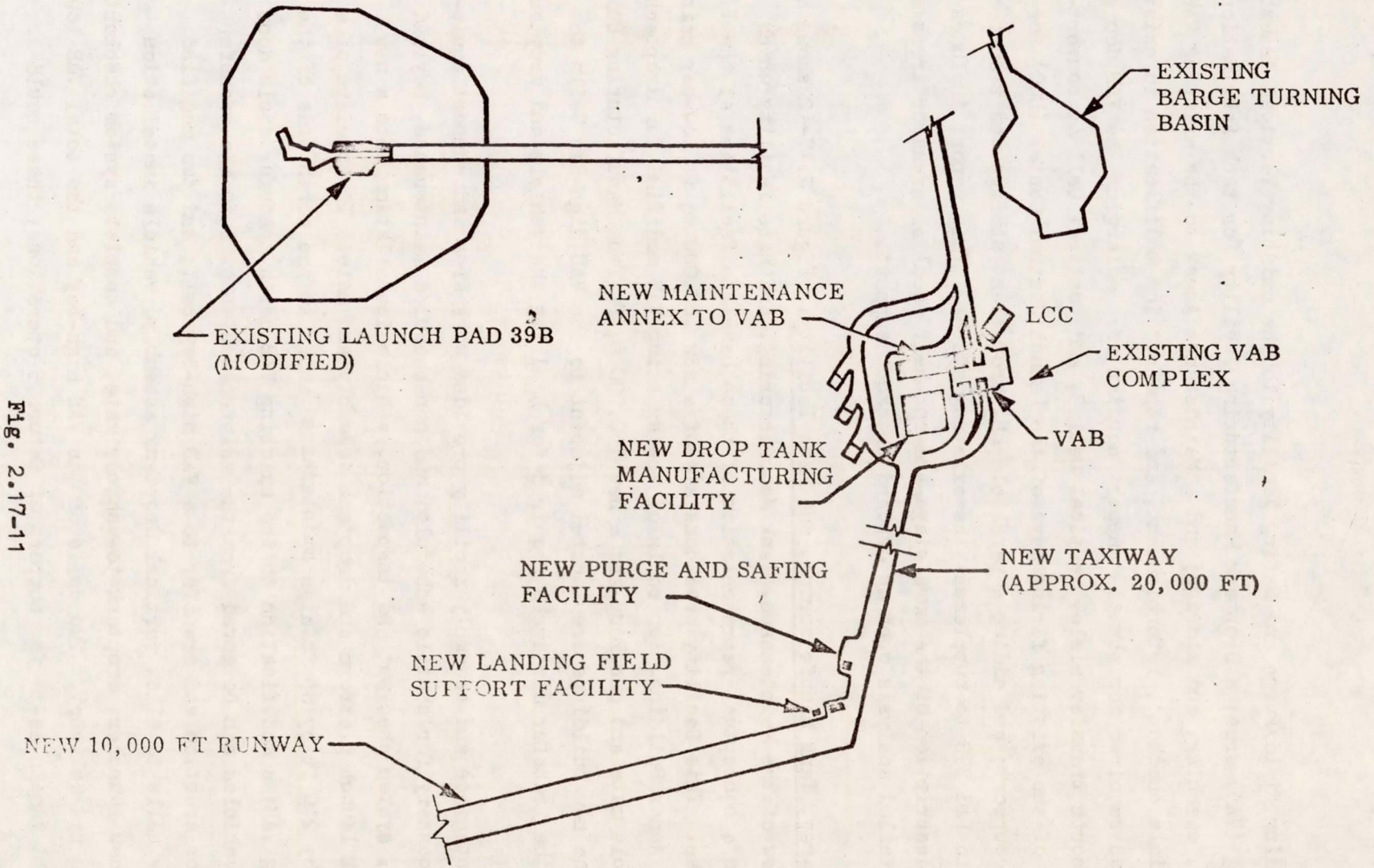


Fig. 2.17-11 Shuttle Facility Conceptual Layout

the Landing Field/Purge and Safing Facility to the existing Vertical Assembly Building (VAB) area; a Droptank Manufacturing Facility for tank fabrication, testing, cleaning, and storage; and a Maintenance Annex to the existing VAB for vehicle checkout, refurbishment, and repair. The modification to existing areas required for shuttle support consist of: enlarging the VAB doors at the north transfer aisle, vertical Cell 3, and vertical Cell 4; converting one of the existing firing bays of the Launch Control Center (LCC) for shuttle support; and adding a parallel LH_2 propellant storage and transfer system to Pad 39B to supplement the existing system. This report includes only a description of the more stage-and-one-half peculiar areas or areas of more detailed analysis such as onboard checkout support.

Maintenance, Tank Manufacturing, and Mate Facility. Figure 2.17-12 shows a plan view of the Maintenance, Tank Manufacturing, and Mate Facility which provides a conceptual representation of these baseline facilities at the KSC Main Base. This facility area consists of a new 160,000 sq ft low-bay Maintenance Annex Building, two cells of the existing VAB modified for droptank-to-vehicle mate and checkout and a new 750,000 sq ft Tank Manufacturing Building. The new buildings are located adjacent to the VAB High-Bay Cells to reduce the transfer distances required for handling the vehicle and droptanks.

The maintenance and assembly facility provides work area and support equipment necessary for shuttle scheduled and nonscheduled maintenance, payload loading, system checkout and inspection, vehicle installation onto a new vertical launch platform and droptank assembly and mate. It consists of a 775 ft by 210 ft by 85 ft high maintenance annex to the north side of the existing VAB; a modification to the existing VAB north transfer aisle door so the vehicles can be moved from the maintenance annex into the transfer aisle for erection and transfer to a VAB high-bay cell; and two modified high-bay cells to allow vertical droptank assembly, vehicle installation onto a new launcher, droptank-to-vehicle mate, and complete system checkout prior to pad delivery. Two cells of the VAB high-bay and the total VAB low-bay would remain as-is for support of Saturn 5 operations; these could

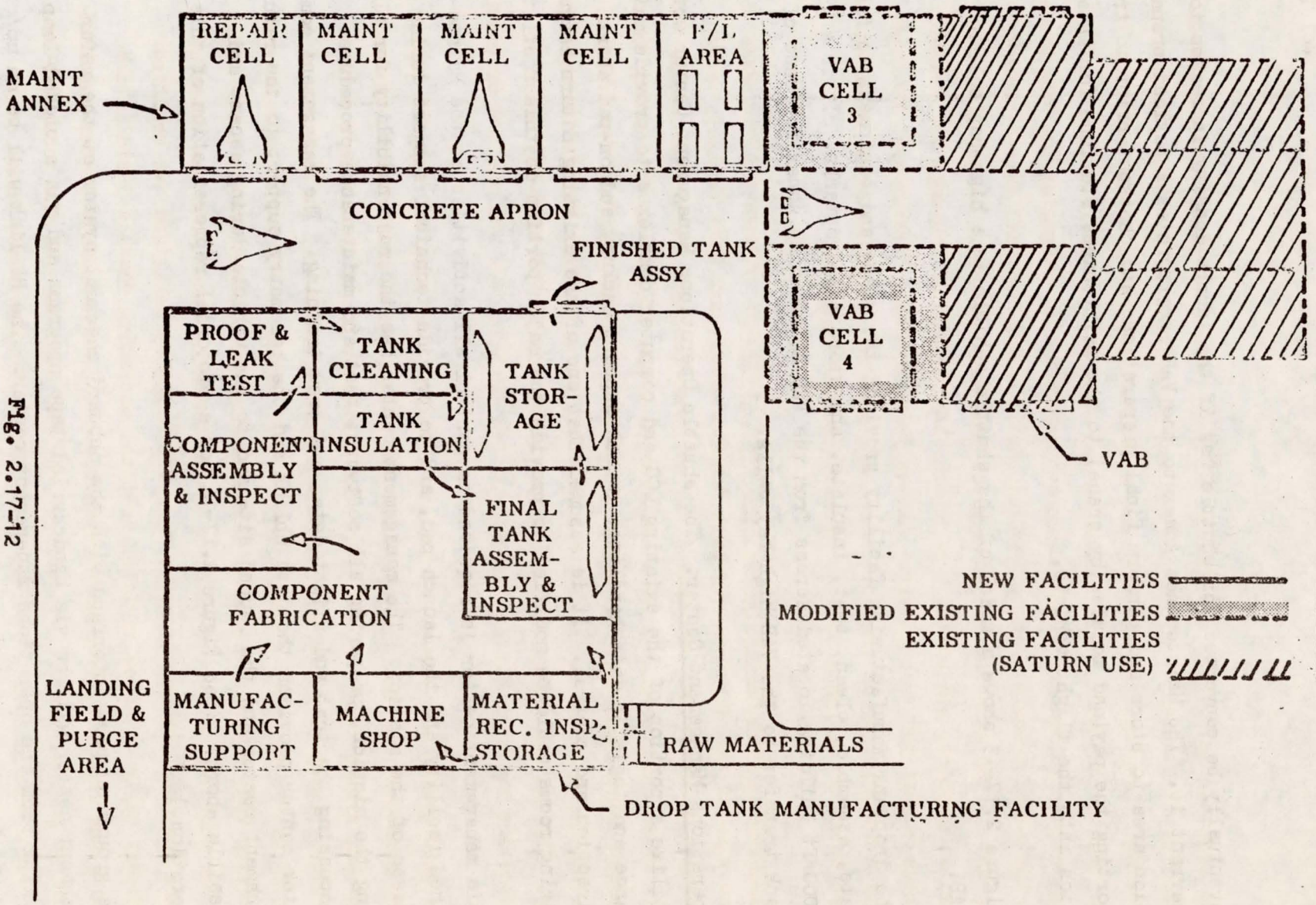


Fig. 2.17-12 Maintenance, Tank Manufacturing, and Mate Facility

eventually be converted for shuttle use if the launch density increases to warrant it. The Maintenance Annex to the VAB incorporates a payload preparation area to store and perform final preparations and checkout prior to transporting the payload module (by crane) to the maintenance stations for integration into the flight vehicle.

Figure 2.17-13 shows a model of the shuttle vehicle in a high-bay cell of the VAB.

The droptank manufacturing facility provides the space and equipment to fabricate, assemble, clean, test, insulate, and store the droptanks. It is a 750 x 1000 ft building located across from the VAB maintenance annex for ease of tank transfer to the VAB high-bay cells.

Operations Management Center. The shuttle Operations Management Center would utilize a portion of the existing LCC and computer complexes to provide maintenance and logistic data handling, launch support backup, and on-pad abort direction and control. It is assumed that one of the existing Saturn backup firing rooms could be modified to provide the major portion of this facility.

This management center is required to control all activities at the Maintenance facility and the launch pad, and to provide technical support during all phases of the mission. The management center has the responsibility for planning the mission, preparing all software, test and maintenance procedures, processing the test and flight data, and crew training. The management computer center provides the data uplink and the telemetry support to the refurbishment center and the launch site and to the vehicle during ascent and possible abort. (See Figure 2.17-14 for a pictorial representation of this operation.)

The ground support interface with the onboard checkout system at the refurbishment center will be via handcarried tape programs and via a closed-loop RF link from the operations management center. The RF link will be an up/down link for computer loading, telemetry, and voice communications. The

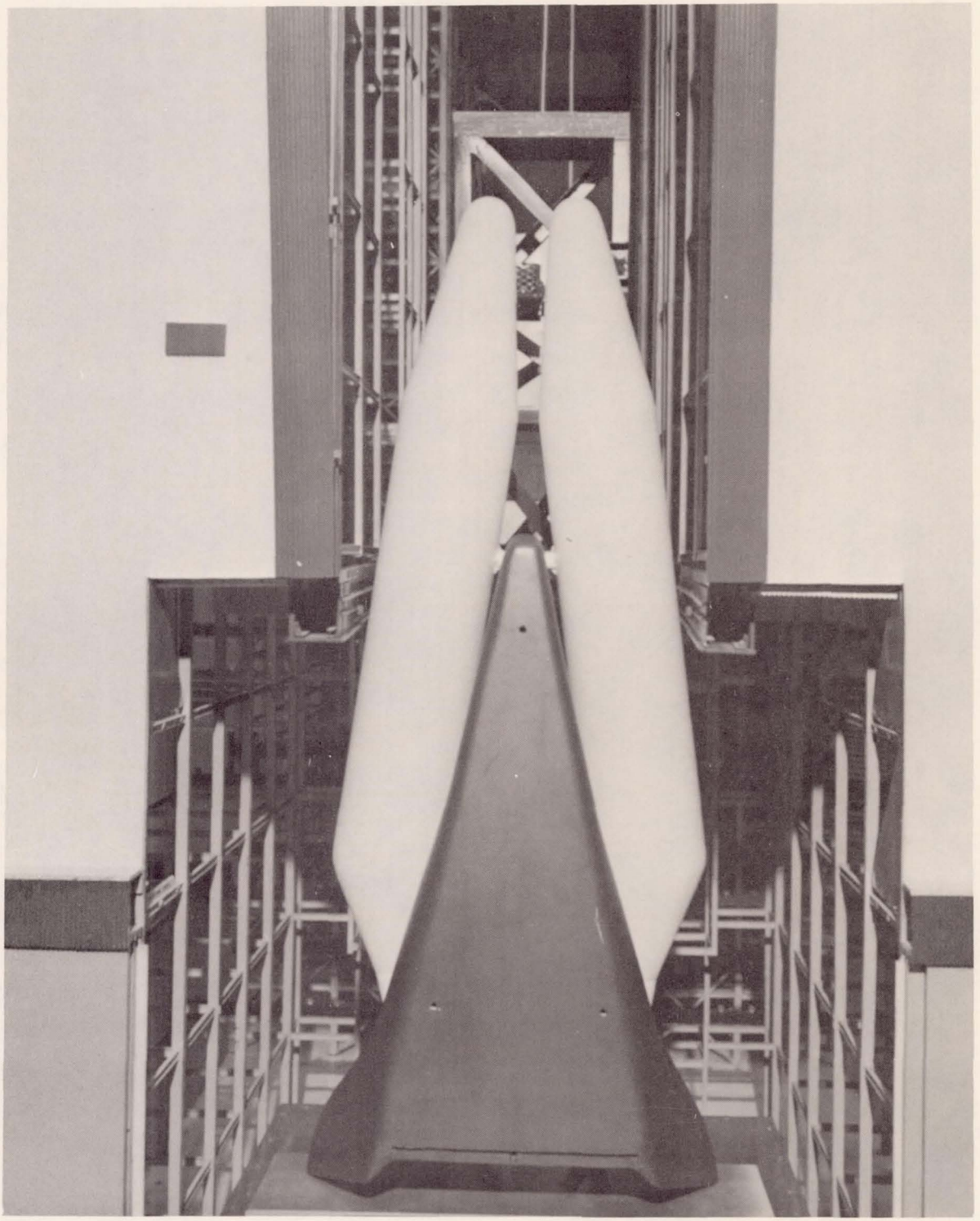


Fig. 2.17-13 Stage-and-One-Half in VAB Doorway

Fig. 2.17-13

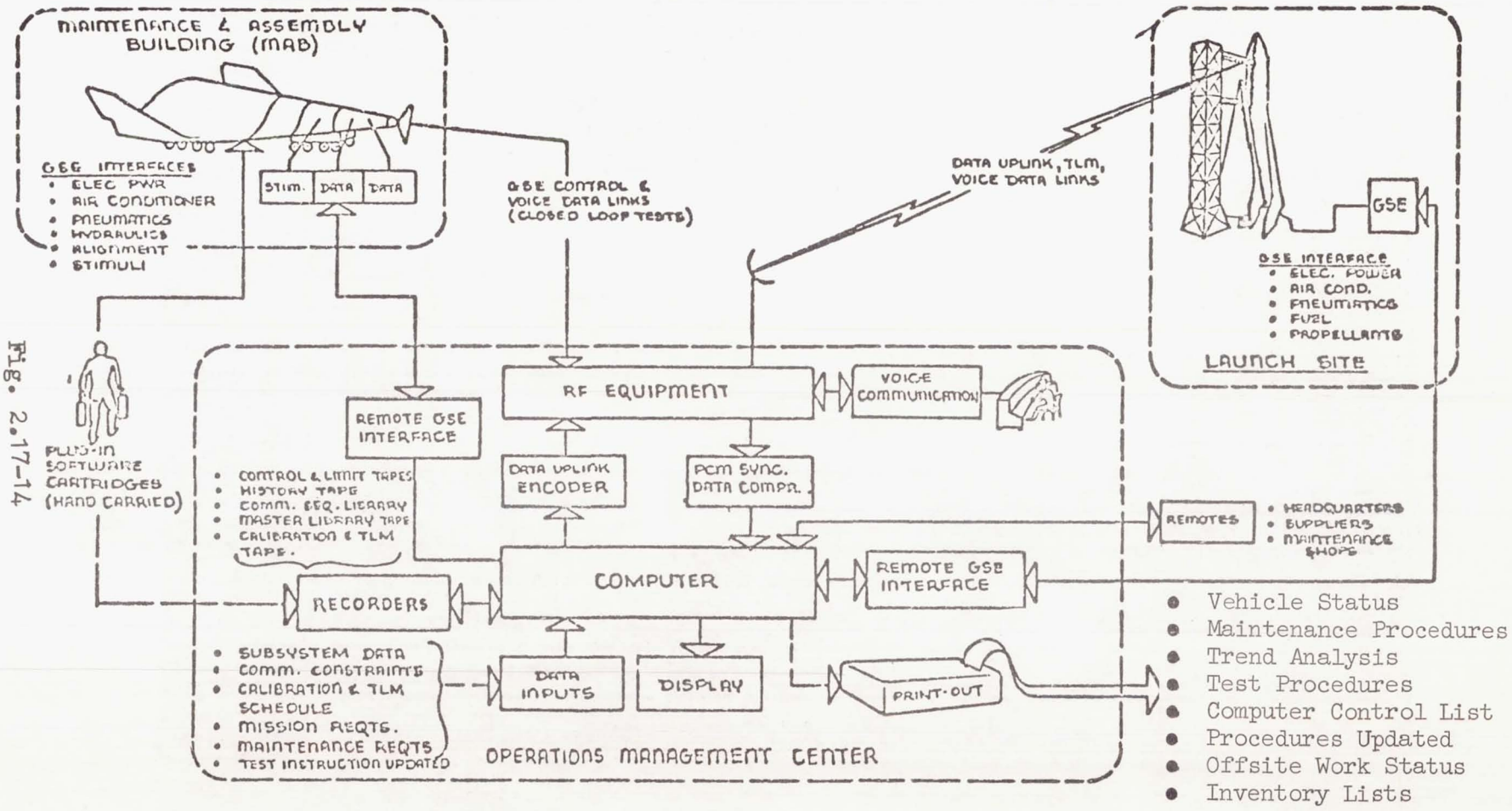


Fig. 2.17-14

Fig. 2.17-14 Ground Support/Vehicle Checkout Interface Flow Diagram

data uplink will be used to update the software programs and for linking the onboard data management and ground support equipment.

The determination of launch readiness will be made, basically, on board the vehicle by the same means as will be used during the mission. All communications and verifications of the onboard systems are through the RF link to the operationa management center. No hardlines are planned except for ground electrical power. The onboard checkout systems will exercise supervisory control over the loading of propellants, pneumatics,and air conditioning at the pad. Land lines will be required to the management computer center for control and checkout of the launch support facilities. Onboard computer programs can be changed on the pad, either via the RF data uplink, plug-in tape cartridges, or constants and coordinate inputs by crew keyset command.

A mobile equivalent of the latest production model flight vehicle avionics, including pilot displays and controls is required as a test tool. The tool should include programmable simulation of all vehicle data sources and load/ monitors for all command and control signal destination and would be used for checkout of launch pads, test stations, and onboard computer programs.

The simulator should be mobile to permit it to interface with the refurbishment stations for calibration and verification of engineering changes throughout the program. A single assembly should be adequate if a single fleet configuration is to be maintained. Experience has shown that a single configuration is to be maintained. Experience has shown that a single configuration is extremely difficult to implement, even in small test fleets. For this reasons, two simulators are anticipated.

One simulator should be to the predominant fleet configuration and be the primary vehicle for assuring uniformity of GSE interface, software, and support calibration. A second simulator is desirable as a lifetime development tool to reflect the configuration of the minority of the fleet which should be in a perpetual development cycle.

Configuration of the simulators must be mechanically and electrically identical at external interfaces (GSE, software input/output) but functionally identical in layout. Much simulator maintenance time can be saved by making electrical design identical to the flight vehicle (coupling, bonding, shielding, wiring) but simulating the installation configuration only to a limited degree. This technique has been successfully demonstrated on both the C-141 and C-5A airplanes.

Consideration was given to combining the simulator use with maintenance personnel training. It is concluded that the role of the simulator is sufficiently critical that training in its normal sense should be excluded. Use of trained personnel on a rotating basis is desirable.

The Dynamic Simulator is shown in the interface flow diagram of Figure 2.17-15.

Test Hardware. Operations Management Center - a central computer facility will support (1) mission flights (prelaunch through reentry and ferrying return to base), (2) maintenance testing and checkout of flight vehicles, (3) checkout and validation of launch pads, test stations, and mobile dynamic simulator, (4) development and validation of software required for mission flights and maintenance, (5) history file updating and report preparation for periodic flight test reports, maintenance reports, reliability reports, and safety reports, and (6) mission simulation for flight dynamics studies, abort techniques studies, and flight crew operational studies.

Scheduling priorities will be (1) mission flights (full time only during prelaunch, ascent, reentry, and other brief critical mission phases), (2) maintenance testing (full time only during brief occasional test data processing runs, (3) development and validation of mission flight software, and (4) all other computer center activities.

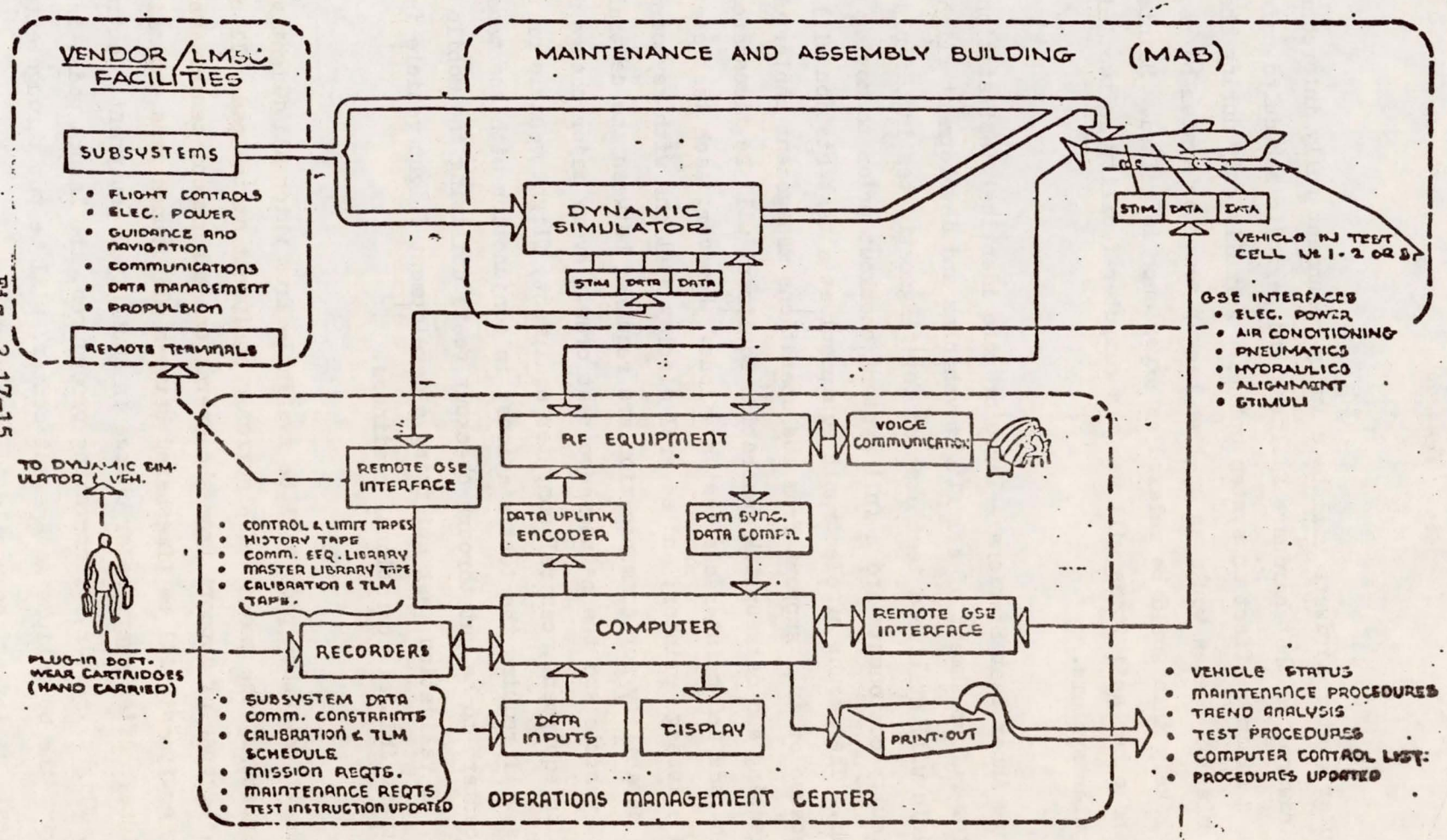


Fig. 2.17-15 Ground Support/Vehicle Subsystem Checkout Interface Flow Diagram

The computer center hardware will be a fully-redundant fully interconnected set of hardware, so that hardware failures and periodic scheduled maintenance will have a minimum effect on system uptime. Normally only during the most critical mission phases would an on-line backup computer system be required, and most likely this would be relaxed as experience is gained. Maintenance testing and other activities will use the computer facility as two independent computer systems.

Maintenance Annex Test Stations - Three or more identical test stations will be used for refurbishment of all flight vehicles and development flight vehicles. Each vehicle will utilize a test station for diagnostic testing, replacement of faulty components and a full systems checkout prior to moving onto the launch pad. The amount of electronic equipment at a test station will be minimized, because of full support from the operations management center where data processing and data uplink generation equipment will be timeshared between the launch pads and maintenance test stations. However, each test station will have a message printer; and several display stations with keyboard input and voice capability for communication are required between the central computer test vehicle and the maintenance test crew. Every major change in the test station equipment, central computer equipment, flight vehicle or test procedures will require the test stations, in conjunction with the management computer center, to be put through checkout tests utilizing the mobile dynamic simulator. A validation test utilizing a development flight vehicle is required to demonstrate initial operational readiness.

Launch Pads - Two launch pads will be available from which all flight vehicles will be launched. The amount of electronic equipment at the pad will be minimized, because of support from the operations management center, where data processing equipment will be timeshared between all launch pads and maintenance test stations. After every major change in launch pad equipment, central computer equipment, flight vehicle, or operating procedures, the launch pads, in conjunction with the operations management center, will be put through extensive checkout tests utilizing the mobile dynamic simulator.

Remote Terminals - The application of remote terminals for controlling vendor and factory equipment for testing subsystems or modules should be considered as an application of the onboard Data Management Subsystem, permitting units at a vendor bench to be tested per vehicle software present at the operations management center.

Facility Costs. The following cost breakdown is based upon KSC information and LMSC Facilities Group estimates. These amounts are required to provide a facility conforming to the baseline operational concepts presented in this subsection. The costs presented do not include design, contingencies, supervision, administration, activation, logistics, or escalation (inflation). These additional costs could increase the values shown by about 66% even without including escalation.

I. AIRPORT

A.	Runway	$10.00M
B.	Taxiway	2.00
C.	Control Tower	2.50
D.	Landing Aids and Control	2.50
E.	Jet Fueling System	.24
F.	A/P Maintenance Hangar	2.06
G.	Misc. Airport Equipment - Tugs, Fire Trucks, etc.	.48
	SUBTOTAL	$19.78M

A. RUNWAY

10,000 ft x 300 ft w. 4 in A.C. surf. 6 in base, 20 in subbase
1000 ft stabilized overrun D.B.T. surface, 6 in base, 15 in subbase
200 ft w stabilized shoulder 2 in A.C., 6 in base, 9 in subbase
700 ft safety zones cleared and graded

B. TAXIWAY

75 ft w taxiway, 3 ft A.C., 6 in base, 15 in subbase
50 ft w shoulders, 2 in A.C., 6 in base, 9 in subbase
400 ft x 800 ft apron - based on space for 2-707 and 1 super-guppy A/C. Construction similar to taxiway with 25 ft shoulders.

C. CONTROL TOWER (A.F. STD. DWG. AD-86-06-05R1)

42 ft 2 in to control room floor incl sound proofing and A/C

D. LANDING AIDS

Runway Hi-Intensity Edge Lighting (500w)
Taxiway Edge Lighting
Obstruction and approach lights (hi-int. strobe)
Nav. aids incl rotating beam, GCA, ILS, remote receiver & transm.

E. A/P FUELING SYSTEM

150,000 Gal. below ground storage w/truck load & unload normal rate pumping

F. A/P MAINTENANCE HANGAR (AD-39-01082)

300 ft w x 260 ft long 60° clear at ℄ of arch, incl. supporting shops

G. MISCELLANEOUS AIRPORT EQUIPMENT

3 - Major airport type fire trucks (incl foam generators)
20 - Misc. Apron Type Vehicles

II. PURGE AND COOLING FACILITY

Facility	$ .50M
Equipment	1.50
Subtotal	$2.00M

III. SHUTTLE MAINTENANCE ANNEX

A. Maintenance/Facility	$17.50M	
B. Handling and Accessory Equipment	3.00	
C. Test and Checkout Equipment	7.00	
Subtotal		$27.50M

A. MAINTENANCE/FACILITY

775 ft x 210 ft x 85 ft hook height building
4 Bays - 160 ft wide, payload area 135 ft x 210 ft, 5 sliding doors
3 - 25 ton cranes w/200 ft span
1000 ft x 300 ft concrete apron

IV. TANK MATE FACILITY (VAB)

A. VAB high bays (2 each) - $6 M/cell - door mods, new work platforms		$12.00M
B. Tank mate equipment		3.00
Subtotal		$15.00

V. LAUNCH PADS (COMPLEX 39A and 39B)

A. Propellant Facility Additions Modifications	$6.00M	
B. Gas Facility Additions Modifications	1.00	
C. Mobile Launcher (LUT Type) 2 each	15.50	
D. Mobile Launcher Equipment	2.00	
Subtotal		$24.50M

VI. LAUNCH CONTROL CENTER (Firing Room Mods.)

A. Checkout Equipment	$2.00M	
B. Transmission Equipment	2.00	
Subtotal		$ 4.00M

VII. SUPPORT AREA

A.	Second Line Maintenance Facility - Engine, Etc.	$1.30M	
B.	Second Line Maintenance Facility - Equipment	4.00	
C.	Passenger Facility	1.00	
D.	Passenger Facility - Equipment	.50	
E.	Nuclear Facility	3.50	
F.	Nuclear Facility - Equipment	1.00	
G.	Flight Training Facility - Modifications	1.00	
H.	Communications - Technical and Base Support	1.50	
	Subtotal		$13.80M
	TOTAL		$106.58

REFERENCES

2.17-1 EM L2-05-04-M1-1 P. 2.1.4-2, Fifth Letter Progress and Status Report, LMSC/A980397.

2.17-2 Subsection 2.17-2 MSFC-DRL No. 216, Line Item No. 4, LMSC/A980397.

2.17.2 Flight Operations

The six operational missions identified by NASA as part of space shuttle Level 1 program requirements are:

- Space Station/Base Logistics Support
- Satellite placement and retrieval
- Delivery of propulsive stages and payloads
- Delivery of propellants
- Satellite service and maintenance
- Short duration orbital mission.

This section describes general flight operations associated with these mission categories, with particular emphasis on the reference mission for space station/base logistics support. For this reference mission, the detailed sequence of events is based on a stage-and-one-half system.

2.17.2.1 Operational Summary. The primary mission of Space Shuttle is to transport cargo and personnel to and from Space Station/Base. In addition to cargo and personnel, Space Shuttle will be required to place at the station (both docked and free-flying) earth orbital experiment modules which would operate in conjunction with a Space Station/Base. The Space Station will normally be inserted into a 270 nm, 55 deg inclined orbit and eventually grow into the space base. Support of Space Station/Base in this orbit has been selected as the Space Shuttle design reference mission.

Objectives of the placement and retrieval mission are to place a number of self-contained satellites into a variety of orbits up to a maximum altitude of 800 nm. Return to earth operations will include retrieval of high priority satellites and wherever practical, space debris.

The delivery of propulsive stages and payloads mission is concerned with the delivery of payloads and propulsion stages to low earth orbit. The payloads would be eventually launched into high-altitude earth orbits or as unmanned interplanetary probes.

In the delivery of propellants mission category, the Space Shuttle is required to deliver LH_2 and LO_2 propellants to an orbital propellant storage facility (OPS) in a low earth orbit.

Space Shuttle requirements in the satellite service and maintenance mission are to periodically revisit modules and satellites to perform routine service and maintenance. In the case where extensive repairs are required, Space Shuttle would return the satellite to the ground.

Two mission modes are considered for the short duration mission. In Mode I, Space Shuttle performs as a dedicated mission vehicle conducting earth-sensing surveys for up to a 30-day period. Mode II operation is performed in response to a need for a "quick" evaluation and detailed observations of a given area. The duration of this mission is generally considered not to exceed three orbit revolutions. Return to launch site within one orbit revolution is desirable.

The mission and system requirements for each of the missions discussed above are summarized in Table 2.17-1. Inspection of the table indicates that mission groupings by requirements commonality exist. These groupings or classes are defined as (1) low-altitude delivery, (2) low-altitude data accumulation, and (3) high-altitude missions.

Within the low-altitude delivery class are the logistics supply, delivery of payloads with propulsive stages, and delivery of propellants missions. The mission profiles for these missions are relatively simple, requiring only about 2000 fps on orbit ΔV for 200-300 nm operation.

The low-altitude data accumulation missions consist of the Modes I and II short duration missions. These missions are characterized by mission sensor operational requirements which impose demanding requirements on guidance, navigation, and control and the data handling and communications subsystems. Further, this mission requires the highest crossrange maneuvering capability.

The high-altitude mission class encompasses the placement and retrieval of satellites and the satellite service and maintenance missions. The relatively high operating altitudes of these missions impose severe demands on the propulsion subsystem. A requirement for about 5000 fps on orbit ΔV with 14 discrete engine firings has been postulated for these missions. However, the requirement to operate at altitudes near 800 nm only to support a few low-cost, low-priority satellites such as Nimbus and Tiros, should be examined. For single payload missions limited to 500 nm orbits, the ΔV requirement would be reduced to about 3000 fps.

Although the quantitative values discussed above and presented in the requirements summary table are subject to revision, it is not expected that the qualitative mission influence on Space Shuttle design will change.

2.17.2.2 Space Station/Base Logistics. The primary mission of Space Shuttle is to transport cargo and personnel to and from Space Station/Base (see Fig. 2.17-16). In addition to cargo and personnel, Space Shuttle will be required to place at the station (both docked and free flying) earth orbital experiment modules which would operate in conjunction with a Space Station/Base.

The Space Station will normally be inserted into a 270 nm, 55 deg inclined orbit and eventually grow into the space base. Support of Space Station/Base in this orbit has been selected as the Space Shuttle design reference mission. Alternate orbits being considered for Space Station include geosynchronous and polar orbits. For support of Space Station in a geosynchronous orbit, Space Shuttle will be required to rendezvous in a low earth orbit with a space tug

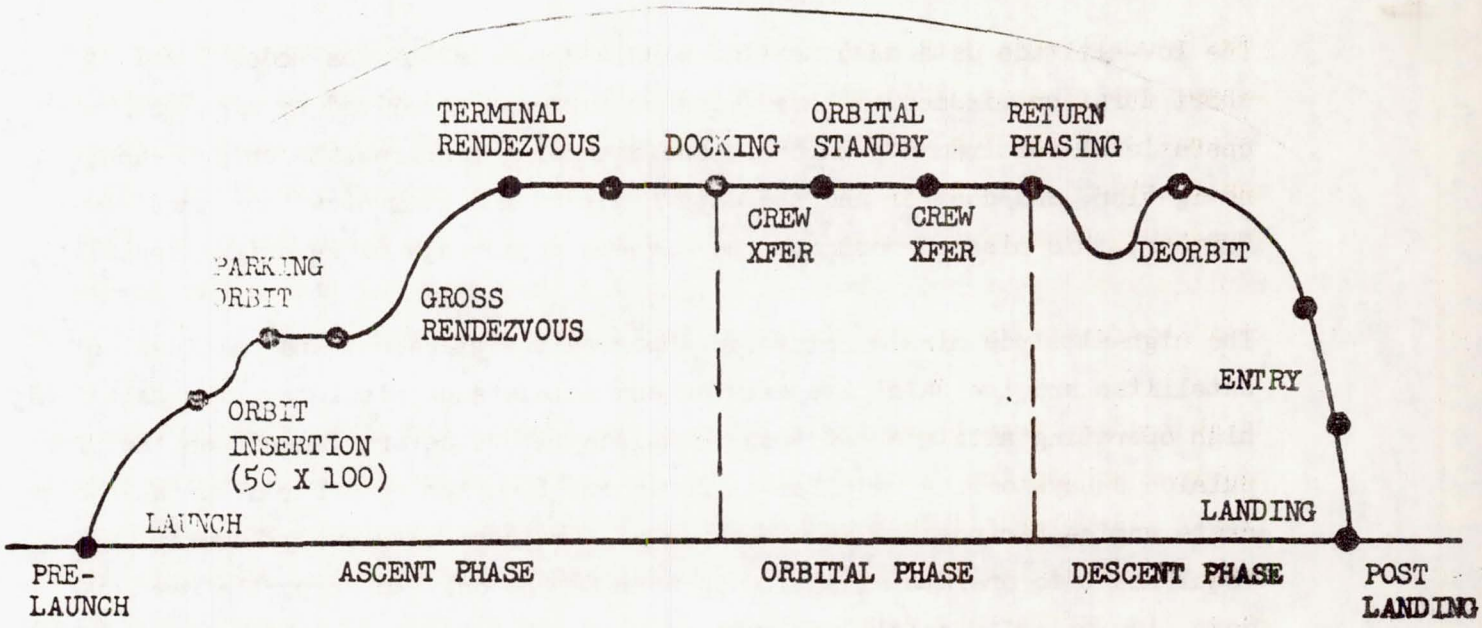


Fig. 2.17-16 Space Station/Base Logistics Mission

for passenger/cargo transfer and eventual delivery to the Space Station. For polar orbits, Saturn V payload launch capability limits the Space Station operations to about 200 nm.

Mission Flight Operation. During the prelaunch phase, launch vehicle activation, propellant loading, and systems checkout are performed. For this mission the Shuttle must be capable of launch within 2 hours from standby status. During this phase, the Space Station will be ground tracked and the ephemeris information processed to the Space Shuttle onboard computer. The tracking data are continually used in the orbit determination until the ephemeris is known to an acceptable degree of confidence to allow a launch decision. A flight plan to rendezvous with the space station is then generated in the onboard computer.

The mission assumes an ETR launch to a 55 deg inclination orbit with maximum acceleration limit of 3 g. Zero time is taken as the instant of liftoff. The rendezvous mission sequence of events begins with powered flight from the launch site at an initial launch azimuth of 39 deg for a Northerly launch

(or 141 deg for a Southerly launch). Orbit injection occurs at 369 sec later at perigee of an elliptical orbit having a perigee of 50 nm and an apogee of 100 nm. The Shuttle coasts to the apogee altitude of 100 nm where an incremental velocity is added to place the Shuttle in a parking orbit that will eventually time synchronize the Space Shuttle with the Space Station. The Shuttle remains in this orbit until phasing is proper for a transfer to the orbit which will complete the gross rendezvous maneuver.

At injection, some phase angle between the Shuttle and Space Station/Base can be expected and the Shuttle will have to perform a catchup maneuver by remaining in a lower altitude parking orbit. A rendezvous sequence which occurs over 17 orbits was selected for the design reference mission. For this mission, the central angle between the Shuttle and Station at insertion is about 353 deg. The rendezvous sequence of events of this mission was taken directly from MSC internal note 70-FM-104 and modified for the stage-and-one-half. Highlights of this mission include a 100/123 nm phasing orbit. A weight adjustment maneuver then occurs to raise the 100 nm perigee to 260 nm, and circularization to 260 nm. Each of these maneuvers is performed using orbital maneuvering system burns and requires a total ΔV of 1130 fps, including deorbit.

The final approach phase of the rendezvous sequence begins with the Space Shuttle 10 nm below and 120 nm behind the Space Station. A terminal phase initiation maneuver (TPI) and a theoretical braking maneuver (TPF) are accomplished to complete the rendezvous sequence. The total ΔV to perform these maneuvers, which are accomplished using RCS thrusting, is 77 fps. The total time to rendezvous and dock with the Space Station is 25 hours, 7 minutes, and 34 seconds.

The Shuttle on-orbit stay times can vary between less than 1 day to a maximum of 7 days depending on the cargo/crew transfer requirements, mission objectives, and return window passing. During this period, the Shuttle provides its own power. Subsystems are placed in a standby dormant condition with their status

monitored from within the Space Station. A typical on-orbit event allowance for the crew and cargo transfer to and from the Space Station and Checkout for preparation to return events is slightly greater than 8 hours.

The return phase begins with separation of the Space Shuttle from the Space Station/Space Base and an orbital loitering period of more than 1/2 hour for deorbit system checkout. During this period the guidance and navigation system performs all functions associated with deorbit, entry, and landing. These functions will include determination of the entry footprint, maximum entry g's, maximum heating, and range and the deorbit time and deorbit velocity vector required to permit landing at the selected site.

The orbiter must be rotated 180 deg for positioning so that the deorbit velocity vector is generated in a direction opposite the orbital velocity vector (θ_r = 0 deg). For this mission, a 469 fps retrograde deorbit velocity was assumed. At the completion of engine burn, the Space Shuttle is reoriented to its normal flight attitude for coast during the exoatmospheric phase. The duration of this phase, which is assumed to terminate at 400,000 ft, is 33 minutes. At this point the flight path angle is 1.8 deg and the orbiter is at an angle-of-attack of 30 deg with wings level (to reduce nose cap heating).

The RF blackout region is assumed to begin at 300,000 ft and extend down to 150,000 ft. At about 270,000 ft, the transition to aerodynamic control begins and pullup is initiated at about 250,000 ft followed by constant-altitude flight. During this phase, temperature and deceleration control are maintained by varying the angle-of-attack and by banking. Engine deployment occurs at 45,000 ft with engine start by 35,000 ft. The final approach, landing flare, and touchdown follow standard large aircraft practice.

A detailed sequence of events is shown in Table 2.17-1.

Table 2.17-1

BASELINE SEQUENCE OF EVENTS

00:01:21	Peak dynamic pressure
00:02:39	-3g acceleration
00:04:54	Reduce power 50 percent
00:04:59	Release droptanks
00:05:04	Full Power
00:05:23	-3g acceleration
00:06:09	Orbit insertion (h_e/h_a = 50/100); maintain cutoff attitude; verify engine cutoff.
00:06:35	Maneuver to local horizontal; orbital rate to maintain zero angle-of-attack; determine orbit and perform pre-thrust targeting for phasing burn.
00:38:15	Maneuver to desired burn attitude; hold inertial attitude.
00:48:15	Phasing burn (ΔV = horizontal, in-plane, posigrade; hold attitude during burn, then random drift).
21:03:21	Relative tracking of space station; perform preburn targeting for height adjustment maneuver; maneuver to and maintain local horizontal.
22:03:21	Maneuver to desired burn attitude; then hold inertial attitude.
22:13:21	Height adjustment burn (ΔV = 282 fps); horizontal, inplane, posigrade; maneuver to and maintain local horizontal; perform preburn targeting for coelliptic transfer maneuver.
22:49:09	Maneuver to desired burn attitude; hold inertial attitude.
27:59:09	Coelliptic burn (ΔV = 239 fps); horizontal inplane, posigrade.
23:00:00	Perform relative tracking of Space Station and preburn targeting for TPI burn; maneuver to and maintain LOS attitude to Space Station.

24:23:23	Maneuver to desired burn attitude for TPI burn.
24:33:24	TPI burn (ΔV = 22 fps) posigrade, pitched up 27 deg.
24:34:00	Maneuver to and maintain LOS attitude; perform preburn targeting for braking maneuver.
25:07:34	Begin braking maneuver using RCS ($\Delta V \cong 45$ fps); maintain LOS attitude during burn; stationkeep within 200 ft while preparations are completed for hard docking.
25:37:34	Hard docked (ΔV = 10 fps).
25:38:00	Begin crew and cargo transfer; dormant storage condition for all subsystems.
66:00:00	Activate subsystems; crew ingress; compute return trajectory; perform systems checks.
67:20:03	Separation; maneuver to position behind Space Station; apply 10 fps retrograde RCS separation burn maneuver; maintain local horizontal.
70:10:03	Maneuver to deorbit burn attitude: Maintain inertial orientation.
70:20:03	Deorbit maneuver (ΔV = 469 fps). Maintain attitude during burn.
70:21:00	Verify deorbit maneuver; reentry guidance update; compute entry footprint; orient to entry attitude retract sensors, antennas, etc.
70:49:00	Entry (400,000 ft); preblackout communications.
70:51:00	Aero control begins; modulate roll angle to fly within acceptable temperature profile.
70:53:00	Full aero control.
71:06:00	Ignite turbojets.
71:18:00	Adjust power setting; perform final landing check; lower landing gear.
71:22:00	Touchdown.

2.17.2.3 Placement and Retrieval of Unmanned Satellites. Mission objectives are to place a number of self-contained satellites into a variety of orbits up to a maximum altitude of 800 nm for independent operation. Return to earth operations will include retrieval of high-cost, high-priority satellites and wherever practical, space debris caused by U.S. and foreign "dead" satellites, expended upper stages, transtages, etc. For the mission being considered, payload weights will range between 200 and 33,000 lb allowing, in most cases, multiple payload delivery. Because orbital plane changes of more than a few degrees result in excessive propellant usage, multiple payload delivery missions will require satellite groupings by orbital inclination commonalities. Two orbit inclinations of major interest are a due east ETR launch (orbit inclination = 28 deg) and polar or near polar orbits.

To retrieve satellites, the shuttle must be capable of performing rendezvous and to dock with passive satellites. Also the target satellite will require a retrieval mechanism which is compatible with that of the Space Shuttle. This requirement, it is expected, will be incorporated into future satellite design.

Normal operations will be to deliver and retrieve satellites by remote controlled mechanical devices with EVA operations performed only as required. An example of EVA would be removal of protuberances, such as extendable booms or space-erected panels prior to satellite retrieval.

Flight Operations. For this mission, the mission profile is very dependent on the orbital and physical characteristics of the candidate satellites. A sampling of current and proposed candidate NASA and DoD payloads are listed below in Table 2.17-2.

Table 2.17-2

CANDIDATE PAYLOADS

Satellite NASA	Weight	Altitude	Inclination
Nimbus	1,463	600	Sun Sync
Tiros - M	651	770	Sun Sync
Biosatellite	1,485	200	28
Imp - J	730	100-140	
OAO	4,609	400	35
OGO	1,400	216-594	82
OSO	675	300	
Physics and Chem Lab	5,000	500	28
Gravity Waves	15,000	500	28
Gyroscope Precession	1,000	500	90
Hi-Energy Cosmic Ray	33,000	300	28
Sea State	200	500-600	80-90
Dod "A"	25,000	300	90-Sun Sync
Dod "B"	30,000	100-300	90-Sun Sync

An interagency satellite placement and retrieval mission has been postulated for placing six independent satellites into six different polar orbits. Mission objectives are to place a radiation and detection (IMP-J) satellite into a 100 x 140 orbit; a DoD satellite into a 100 x 300 orbit; two low-altitude space sensor platforms positioned 180 deg apart into 400 nm orbits; and both a gyroscope precession experiment and a sea surface height measurement satellite into a 500 nm orbit. Both the DoD and gyroscope precession satellites are high-cost, high-priority satellites, and will be retrieved.

The first burn injects the orbiter into a 50 x 100 elliptical orbit at an inclination of 90 deg. A second burn at apogee of the initial orbit places the Shuttle into a 100 x 140 nm orbit where the radiation and detection satellite is released. When phasing is proper, the Shuttle is transferred to a

100 x 300 nm orbit where the "dead" DoD satellite is retrieved and exchanged with an operational satellite. Since the two space sensor platforms are to replace "dead" satellites, which were a part of a low-altitude multisatellite net, the orbiter remains in the 100 x 300 nm orbit until phasing is correct for transfer to the 400 nm orbit. Transfer to the 400 nm orbit is made for placement of the first of the two satellites followed by a phasing period for transfer to the 500 nm orbit where a rendezvous and docking maneuver is performed for the retrieval and replacement of the gyroscope experiment satellite. Placement of the sea surface height measurement satellite at 500 nm completes the ascent orbital maneuvering phase.

The Shuttle remains in the 500 nm orbit until phasing is proper for a transfer to the 400 nm orbit and positioning of the second space sensor platform within the satellite net. Deorbit is then made directly from this altitude.

Assuming unfavorable phasing for all orbit transfers, the maximum duration for the preceding mission is about 7 days, requiring a total velocity of 2600 fps. The 3100 fps velocity includes all requirements for orbital transfer, phasing, rendezvous, docking, and deorbit. The performance requirement assumes a satellite is carried to orbit and exchanged with an equivalent satellite. For placement only (no exchange) phasing, rendezvous, and docking are not required, and the velocity requirements will be lower.

The preceding multiple orbit/payload placement and retrieval mission was presented only as one example of many possible operational modes. A satellite placement and retrieval mission launched due east from ETR would allow placement and retrieval of a Biosatellite, an automated physics and chemistry lab, an automated hi-energy cosmic ray physics lab, and a gravity waves experiment satellite.

2.17.2.4 Delivery of Propulsive Stages and Payloads. This mission category is concerned with the delivery of payloads and propulsion stages to low orbit.

These components would be launched into high-altitude earth orbits or as unmanned interplanetary probes. The following principal operational modes will be required to deliver payloads and stages:

Mode 1. The payload and stage are delivered to orbit in separate launches with orbital assembly, fueling, and launch. Mode 1 operation implies the use of an orbital facility to assemble, checkout, and fuel the stage and payload. The Space Shuttle would only deliver stages to the orbital assembly facility.

Mode 2. The fully assembled dry stage and payload are delivered to the OPS facility for fueling. Mode 2 will require the Space Shuttle to dock at the OPS facility for fueling the propulsion stage. This would be followed by checkout and deployment of the payload and stage by the Space Shuttle. Once deployed, the stage and payload would revert to the control of an orbital facility (Space Station) or ground control for final countdown and launch.

Mode 3. The fully loaded stage with payload attached is delivered to orbit, with subsequent checkout and deployment. Final checkout and launch are the same as Mode 2.

Flight Operations. This mission description is based on Mode 3 operation (see Fig. 2.17-17). Prelaunch activities associated with this mission would be as described in the Design Reference Mission. The Shuttle payload (stage and attached payload) would be attached to a standardized payload interface within the Space Shuttle payload compartment. This interface will provide structural support for flight and orbital deployment, as well as a crew function for checkout and monitoring of payload function.

A large number of payloads to be handled in this mission category will be unmanned planetary probes and those items designated for the lunar shuttle. The majority of launches will be made to low inclination orbits (near 28.5 deg) to an altitude of approximately 100 nm. The ascent sequence of events to this orbit will be the same as for the Design Reference Mission: injection into a

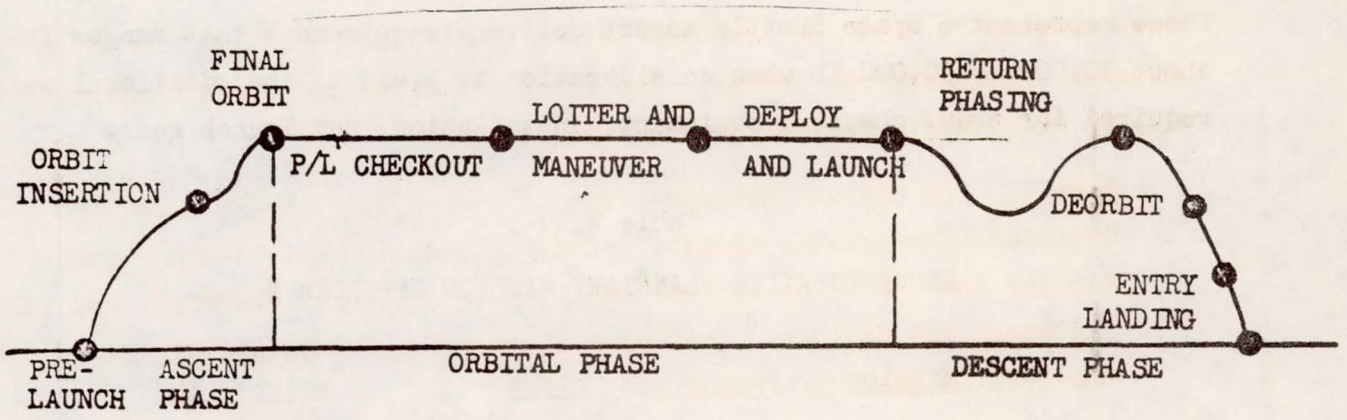


Fig. 2.17-17 Delivery of Propulsive Stages and Payloads

50 x 100 nm orbit and final circularization at 100 nm. For Mode 3 operation, the final orbit is the scene for checkout, deployment and launch of the payload. Orbital checkout of the payload is completed using equipment onboard the Space Shuttle. Following checkout, the vehicle maneuvers to the proper position and velocity coordinates and loiters to await the orbital launch window. Continuous checkout and monitoring are required during this period. As the launch time approaches, vehicle mechanical systems deploy the payload to a point external to the vehicle. While still attached, final system checks are completed. The payload is then mechanically separated from the vehicle with the correct attitude and position and the vehicle maneuvers away from the payload. Active control of the payload for final countdown and launch is then transferred to ground control. The vehicle maintains orbital standby and provides a backup monitor function for the ground station. At $t = 0$, the launch signal is given from the ground station. Following launch, the normal return phasing, deorbit, entry, and landing functions as described in the Design Reference Mission would be performed.

Table 2.17-3 lists a representative sample of spacecraft proposed for unmanned planetary missions. The table shows that the payload weight (satellite plus propulsive stage) will vary between 7500 lb for the Venus or Mars explorer orbiter to 47,500 lb for the 1984 Halleys comet rendezvous mission.

These represent a Space Shuttle ascent delivery requirement that ranges from about 10,000 to 50,000 lb when consideration is given to the additional weight required for crew, checkout equipment, installation, and launch gear.

Table 2.17-3

REPRESENTATIVE PLANETARY MISSION SPACECRAFT

Mission	Stage	Total P/L Weight (lb)
Mars/Venus Explorer	AJ-10-138	7,500
Saturn Orbiter	Centaur/ Burner II	39,500
Viking Orbiter	Centaur	43,700
Halley Rendezvous	Centaur/ N.E.	47,500

2.17.2.5 Delivery of Propellants. In this mission category, the Space Shuttle is required to deliver LH_2 and LO_2 propellants to an Orbital Propellant Storage facility (OPS) in low earth orbit. The OPS facility has the function of providing propellant for unmanned planetary missions, the spacebased nuclear lunar shuttle, and for the space tug operations required for lunar surface and geosynchronous station support. The Space Shuttle will be required to operate in three distinct tanker configurations to support this mission (1) as an LH_2 tanker, (2) as a combined LH_2/LO_2 tanker, (3) as a partial tanker to be used in conjunction with the normal delivery of supplies to the space station. During the high traffic periods of the program and for initial filling of the OPS, a dedicated tanker will be used for this purpose. For the dedicated vehicle, the tankage and propellant transfer system would be an integral part of the orbiter stage. Desirable operational orbits for the OPS would be 28.5 deg and 55 deg with an altitude sufficient to provide good drag life characteristics and to facilitate delivery to the space station at the higher inclination orbit. The mission profile is shown in Fig. 2.17-18.

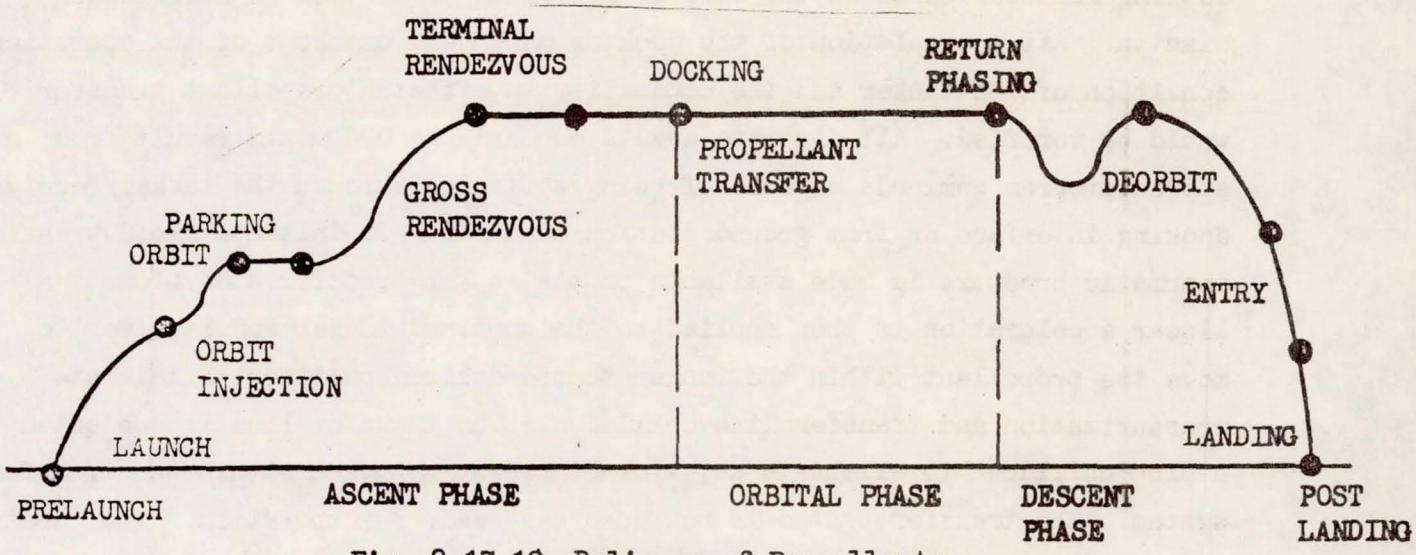


Fig. 2.17-18 Delivery of Propellants

The OPS facility itself would be comprised of structurally connected cylindrical tanks capable of long duration orbital storage of LH_2 and LO_2 with a tank mass approaching 1.2×10^6 lb. The OPS receiver would be a passive system maintaining a referenced stability and providing a docking capability with the Space Shuttle tanker. During orbital storage, operation and checkout of the OPS would be remotely controlled from the ground through the MSFN.

Flight Operations. Prelaunch activities associated with the mission would be described in the Design Reference Mission with the exception that both LO_2 and LH_2 tankers would require preparation of the propellant delivery tanks. Such basic functions as propellant delivery tank purge and fill, pressurization system purge and fill, and pneumatic system purge and fill would be required to insure safe ground operational procedures. With these sequential operations accomplished, liquid level sensing would function to maintain the required liquid level within specific limits. The tanker system is now filled with the required fluids and ready for launch.

For the purpose of description, it is assumed that the OPS facility is at a 55 deg inclination and 270 nm altitude. After liftoff, mission operations up to the attainment of the rendezvous orbit ($h \cong 260$ nm) and through the

docking maneuver would be the same as described in the Design Reference Mission. After completion of the docking maneuver, checkout of the operating condition of the tanker and the capability to effect a propellant transfer would be verified. All discrete events onboard the OPS would result from stored program commands capable of being activated through the tanker/receiving docking interface or from ground station commands. To initiate fluid transfer, pneumatic pressure is made available to the tanker transfer subsystems. A linear acceleration is then applied to the combined tanker and receiver to move the propellant within the tanker to the desired position to initiate pressurization and transfer line chilldown. The transfer line is subjected to a slow chilldown to eliminate any violent pressure surges within the transfer system. The transfer system is now open and ready for operation. The receiver shutoff valve is then actuated and propellant transfer begins. During transfer, the flow rate from the tanker and total flow to the receiver are constantly checked. When the tanker liquid level indicates the desired total flow, the tanker flow control valve is closed, completing the transfer operation. The Space Shuttle tanker is now ready to separate from the OPS receiver. After unlocking, the normal deorbit entry and landing functions as described in the Design Reference Mission would be performed.

2.17.2.6 Experiment Module/Satellite Service and Maintenance Mission. The purpose of this mission is to provide service and maintenance to large experiment modules and satellites operating in orbits at altitudes of up to 800 nm and inclinations ranging from 28.5 deg to 90 deg. (There is the possibility of orbits at inclinations lower than 28.5 deg as well.) While many of these modules or satellites may be operating in conjunction with a Space Station or Base, others may be in orbits that would be more readily accessible from the ground. These modules or satellites are logical candidates to be serviced and maintained by the Space Shuttle. The Shuttle would have the capability to revisit modules and satellites and bring them into an onboard facility (shuttle payload) where a service and maintenance crew could conduct these operations in a shirtsleeve environment.

Operating modes being considered include delivery of a satellite service module along with a logistics payload to the Space Station, where a tug would transfer the module to the satellite to be serviced. For maintenance payloads of only a few thousand pounds, this move is probably more efficient than direct delivery by Space Shuttle.

Flight Operations. The shuttle service and maintenance facility will contain equipment, instruments, and supplies trained personnel and will provide the capability to conduct servicing, maintenance, and repair operations. The servicing functions would be conducted on a periodic basis and would include such items as film changing and replenishment of attitude-control propellants.

Although highly automated satellites are designed for long-term operations, the capability to visit such satellites in case of malfunctions is highly desirable. The shuttle could provide the capability for on-orbit replacement of instruments and components. In cases where extensive repair might be required, the shuttle could either return the satellite or experiment module to the ground or transport it to a station or base (depending on the satellite orbit inclination). Satellites that operate for long durations would be designed to accept updated instruments and sensors to enhance their operational capability. This replacement function would be accomplished by the shuttle.

Although indications are that orbit operations of up to 15 days might be required to conduct on-orbit service and maintenance operations, the time would be highly dependent on the servicing requirement, number of personnel, equipment available, etc. Consumables for shuttle operations which exceed 7 days must be charged against the payload.

The mission profile for this mission is illustrated in Fig. 2.17-19.

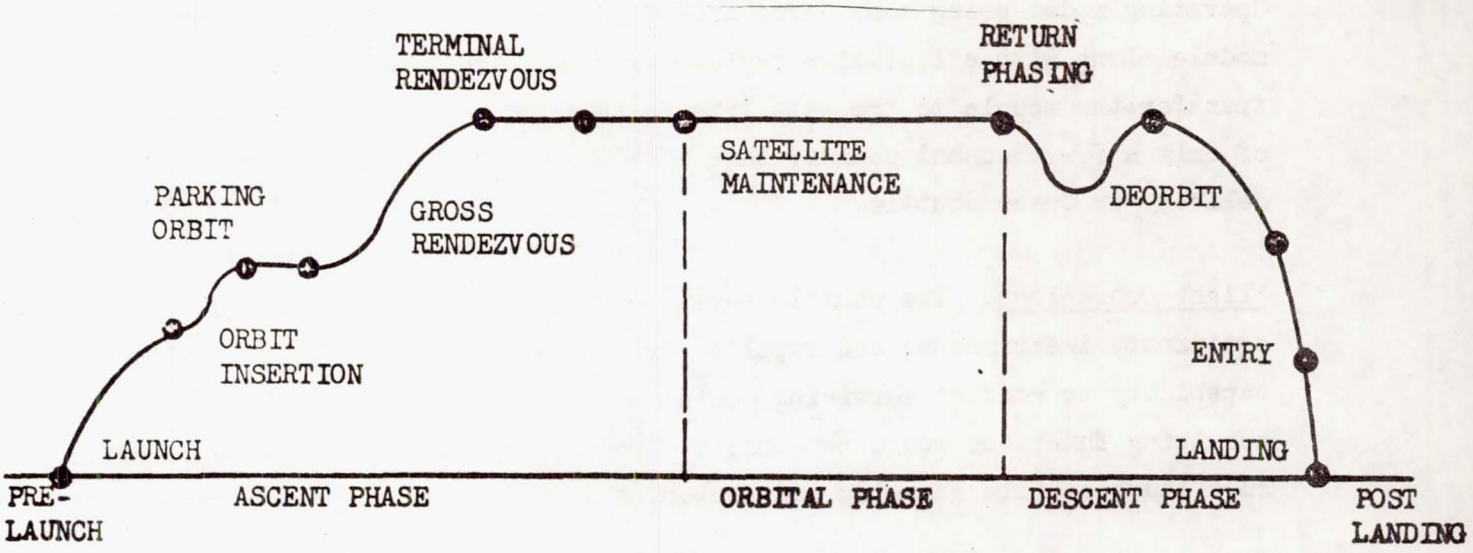


Fig. 2.17-19 Satellite Service and Maintenance

2.17.2.7 Short Duration Orbital Mission. As a spacecraft, Space Shuttle will have the capability of conducting earth-sensing surveys for up to 30 days stay time. Although many of the surveys will be conducted by the space base and unmanned satellites, Space Shuttle will complement their activities by providing in-depth coverage of selected areas. Surveys proposed to be performed with the short duration mission mode include investigations in the areas of cultural resources, natural resources, and earth sciences. Two mission modes are considered for this mission.

Mode 1. In this mode, Space Shuttle performs as a dedicated vehicle conducting earth-sensing surveys. The orbital characteristics and mission requirements for this mission are generally the same as for the baseline mission, therefore the mission profile will be similar. Normally, prelaunch activity will not be urgent and a launch response of about 5 hours will be sufficient. Launch will be in a southerly direction to an orbit having a perigee of about 100 nm and an apogee of 200 to 300 nm. Perigee will be located at the latitude which is of primary interest from the viewpoint of earth resources evaluations. The altitude and inclination will be selected to provide a ground track with a constant local sun time (sun synchronous orbit).

Remote sensing of the earth's surface involves use of high- and low-resolution imaging sensors over a wide range of the electromagnetic spectrum from the ultraviolet region and into the microwave bands. On-orbit operations will consist of activating these sensors over the areas of interest and storing the data for transmission to ground stations via electronic readout. Mission durations will range between 7 and 30 days, depending on the coverage requirements over the areas of interest.

Under normal operating conditions, there will be no urgency for the return to Earth-phase. At least one return opportunity per 24-hour period to a prime CONUS landing site will be available.

Mode 2. In response to a need for a "quick" evaluation and detailed observations in a given area (such as natural disaster), an on-request surveillance capability will be required. To accommodate quick evaluation, it is desirable that the orbiter return to the launch site within one orbit revolution. Since the operational requirements for the single-pass mode are, in many ways, more critical than the long stay time mode, it was selected for a detailed mission profile examination.

Flight Operations. Because this mission will be performed in response to an urgent situation, the capability of being launched within 2 hours from stand-by status is required. The prelaunch activities then, will be similar to those of the baseline mission.

The operational mode assumes a WTR launch with no launch azimuth constraints. The orbiter is launched into a low-altitude orbit at a launch azimuth to overfly the prime target on the first pass. Sensors are activated over the target area and data transmitted via relay satellites for "quick" ground assessment. Upon completion of the survey, the decision is made whether to return immediately to CONUS (or Hawaii) for detailed examination of the data, or to continue the mission for up to two additional passes followed by a return to Hawaii.

Single pass return to the launch site operation imposes stringent lateral range and downrange requirements which are directly related to the target location and direction of launch. For a given location, two launch azimuths (northerly and southerly) will result in first pass target overfly. Generally, for WTR launches, northerly launches will be used for first pass overfly of targets located north of about 35 deg south latitude. Target locations south of 35 deg south latitude will require southerly launches. These launch constraints result because the deorbit downrange (between the target and return sites) is insufficient to allow first pass return to the launch site without exceeding the allowable entry angle.

The crossrange maneuvering requirements for first pass return to Edwards and third pass return to Hawaii are listed below.

	Edwards AFB	Hawaii
First Pass	1100 nm	-
Third Pass	-	1700 nm

The mission profile of both the Mode 2 missions is illustrated in Fig. 2.17-20.

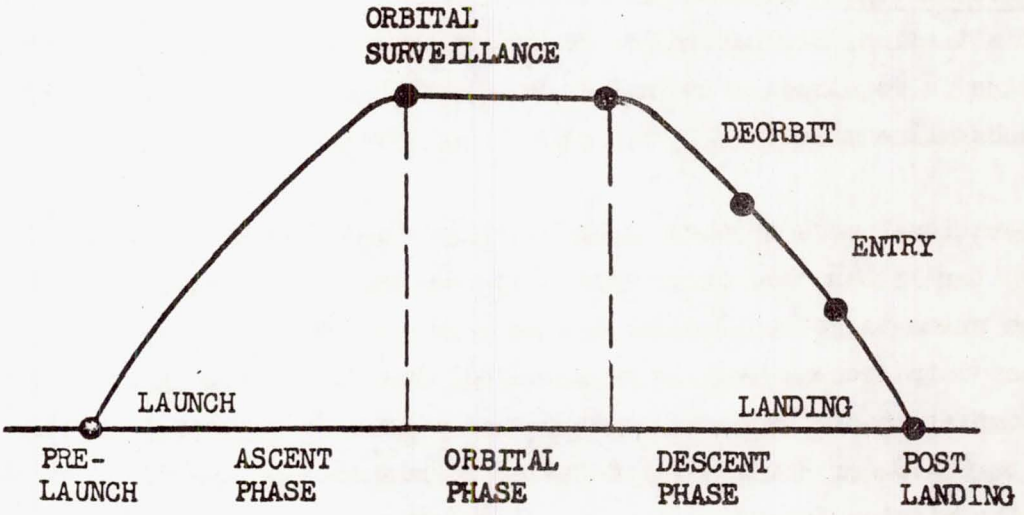


Fig. 2.17-20 Mission Profile

2.17.3 Ferry Operations

Landings at secondary or emergency fields impose two basic requirements upon shuttle program operations: (1) the need to supply certain support equipment and capability to that field, and (2) the necessity of returning the shuttle to a main operational base for turnaround and reuse. Consequently, this section describes the baseline ferry mode configuration and the methods of supporting and ferrying a stage-and-one-half shuttle from an alternate landing site. This baseline was derived, using the results of the analysis from section 2.4.8, which defines the ferry capability of the LS 200-10 vehicle and presents parametric ferry mode performance data for the ferry regime.

2.17.3.1 Ferry Configuration. The ferry-mode-configured stage-and-one-half includes the basic shuttle vehicle with payload removed, all main cryogenic propellant tanks purged and sealed, and a ferry "kit" installed. The ferry kit consists of a 65,000 lb capacity JP-4 fuel tank with an aerial refueling standpipe system installed in the payload bay. No additional engines are required and the vehicle is capable of rotation and takeoff without addition of auxilliary control surfaces or landing gear.

The baseline ferry vehicle has a standard-day, sea-level takeoff weight of 360,000 lb and a landing weight of 302,000 lb; it has a 270 nautical mile range with one go-around and a 15 percent fuel reserve. Tropical-Day (79°F) and high-altitude field (4,000 ft) takeoff capability limits range to 150 nm.

The ferry mode vehicle has a subsonic L/D of 5.85 with very little center-of-gravity (cg) shift throughout its flight as the JP-4 propellant tank is located far aft in the payload bay and quite close to the dry vehicle cg. Parametric ferry performance studies show that increasing L/D (base firings, etc.) and number of engines does not increase ferry performance or range very much; for example, increasing L/D to 7.0 increases the range only 75 nm, while adding an engine increases the range only 90 nm.

2.17.3.2 Ferry Methods. Since the ferry flight range for any shuttle vehicle is quite limited, as shown in the preceding discussion, the requirement for long-range ferry from secondary landing sites suggests the use of ancilliary or support transportation equipment to supplement the basic shuttle. Two prime candidates for this role are ocean-going barges to cover major distances or inflight refueling to increase the shuttle vehicle's range. In both of these cases the inherent vehicle ferry capability would still be relied upon to provide transportation to or from seaports and to provide safe return to earth in case of missed or aborted inflight refueling attempts.

Figure 2.17-16 is a pictorial example of the ferry hop or inflight refueling capability of the stage-and-one-half vehicle. Single hop distances are limited to about 270 nm, while inflight refueling of the 65,000 lb JP-4 tank provides an additional 290 nm before another refueling or landing is required. The inherent capability of the vehicle provides approximately 140 nm or 35 minutes to accomplish refueling after cruise altitude has been reached. Cursory analysis showed the KC-135 Tanker aircraft refueling altitude, velocity, and capacity to be compatible with the shuttle ferry requirements.

Figure 2.17-17 shows a typical single-ferry route for unassisted flight between Edwards Air Force Base and the proposed Kennedy Main Shuttle Base. The basic criteria being preferred use of military fields, distance between fields under 270 nm, and field runway lengths in excess of 10,000 ft. This figure is included to show that short hop ferry mode operation is possible. Also, an airfield matrix similar to this figure would be required in support of an inflight refueled trip, as backup fields would have to be defined and available for abort or missed hookups. The coast-to-coast ferry trip shown would require at least 7 inflight fuel transfers or 11 takeoffs and landings. Table 2.17.3-1 gives the distances for the routes shown in Fig. 2.17-17. The western portion of this route includes the Rocky Mountains and requires 9500 ft altitude minimum for IFR and 5000 ft for VFR. These altitudes do not impose any problems to a stage-and-one-half shuttle ferry flight as the nominal flight altitude for standard-day operation is 13,000 ft, and hot-day capability still allows climb to 6,000 ft.

STAGE-AND-ONE-HALF FERRY FLIGHT PROFILE

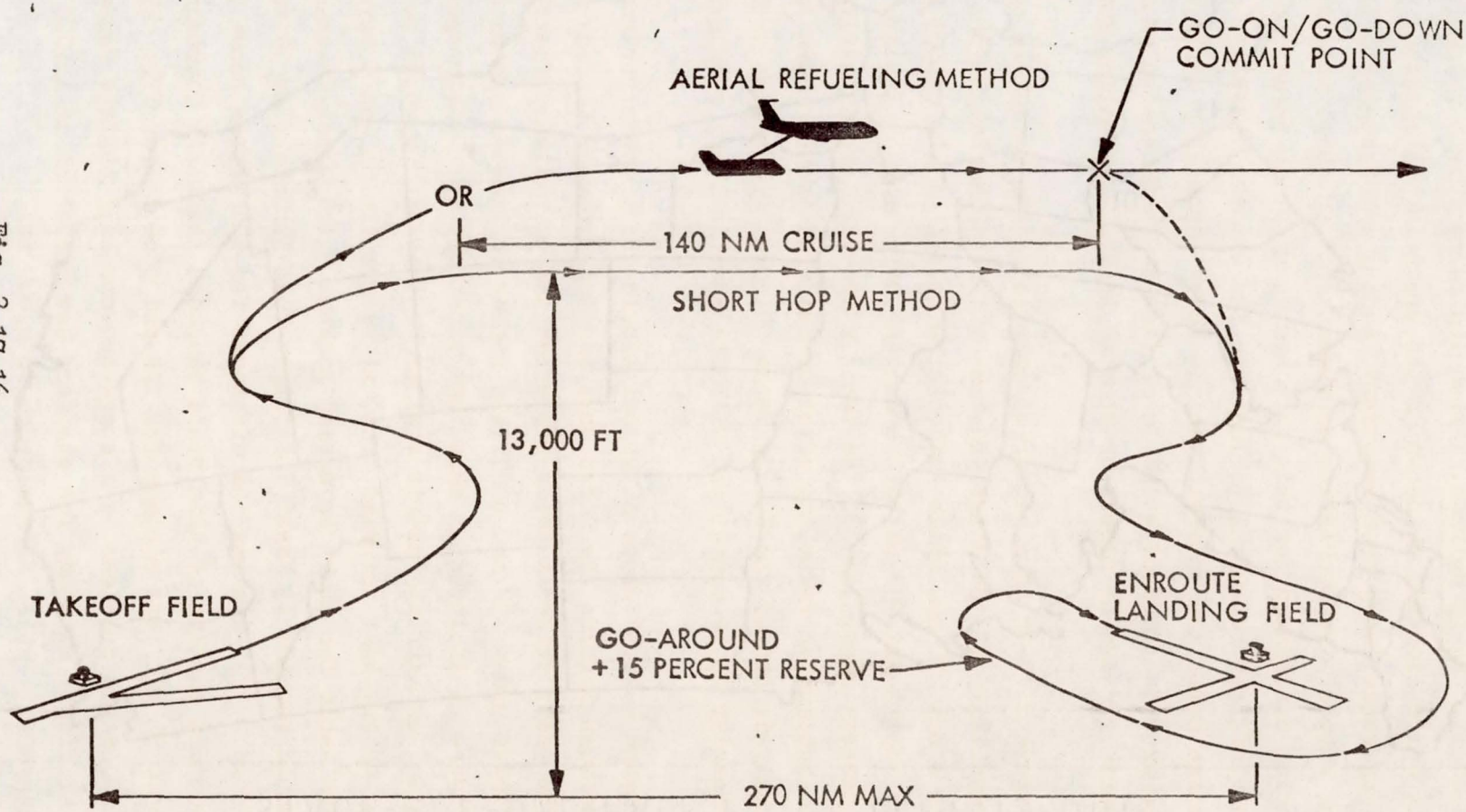


Fig. 2.17-16
2.17-53

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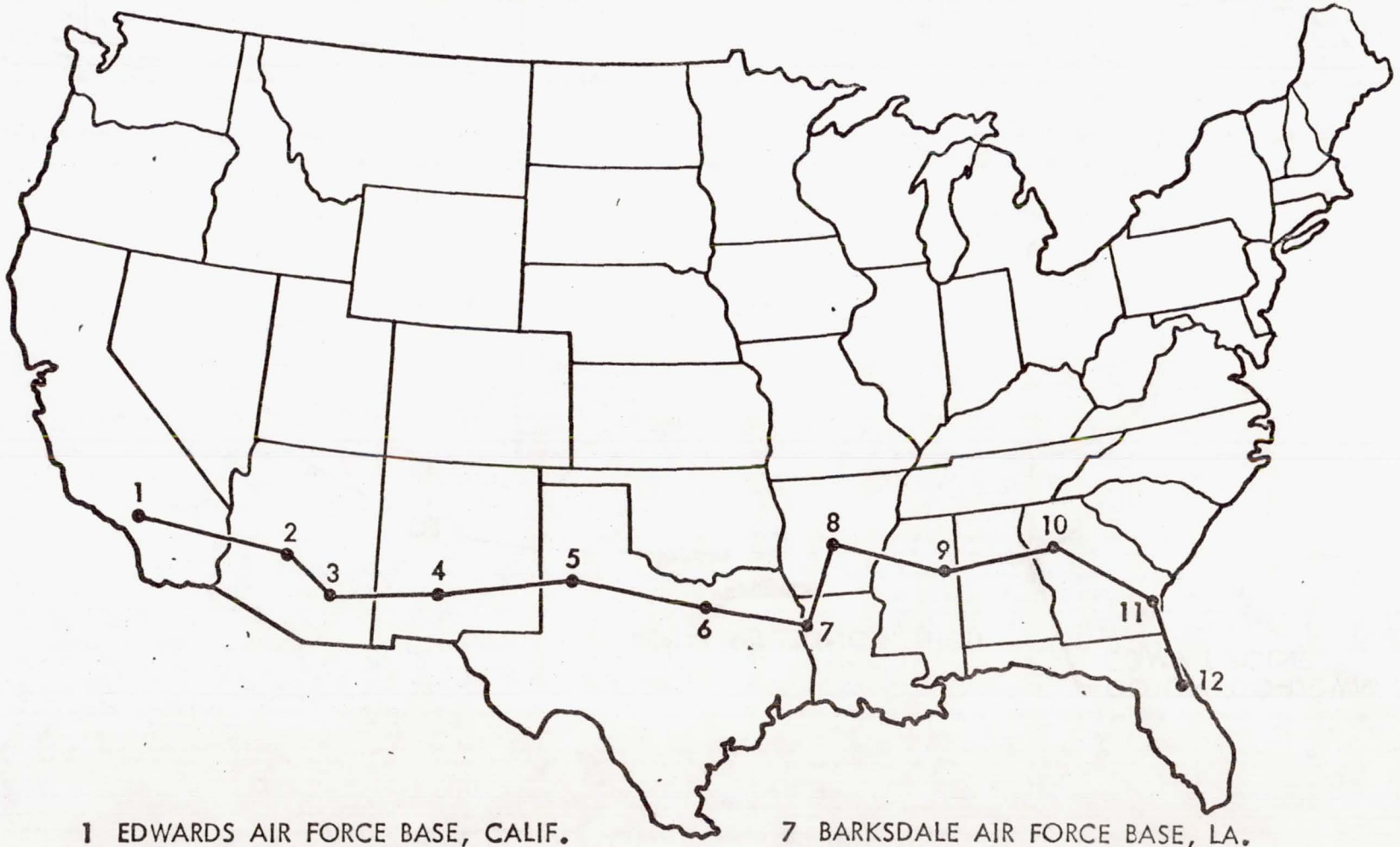
STAGE-AND-ONE-HALF FERRY FLIGHT ROUTE*



1 EDWARDS AIR FORCE BASE, CALIF.
2 LUKE AIR FORCE BASE, ARIZ.
3 DAVIS-MONTHAN AIR FORCE BASE, ARIZ.
4 HOLLOMAN AIR FORCE BASE, N.M.
5 REESE AIR FORCE BASE, TEX.
6 CARSWELL AIR FORCE BASE, TEX.
7 BARKSDALE AIR FORCE BASE, LA.
8 LITTLE ROCK AIR FORCE BASE, ARK.
9 COLUMBUS AIR FORCE BASE, MISS.
10 DOBBINS AIR FORCE BASE, GA.
11 HUNTER ARMY AIR FIELD, GA.
12 KENNEDY SHUTTLE BASE, FLA.

*POSSIBLE ROUTE BASED UPON 270 NM MAX FERRY RANGE AND MILITARY FIELDS WITH RUNWAYS >10,000 FT

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Fig. 2.17-17
2.17-54

TASK 3.17-1

STAGE-AND-ONE-HALF FERRY ROUTE DISTANCES

(Max. Range - 270 NM)

SEGMENT	FIELD	DISTANCE (N.M.)
1	Edwards AFB, Calif.	250
2	Luke AFB, Ariz.	100
3	Davis-Monthan AFB, Ariz.	225
4	Holloman AFB, N. M.	220
5	Reese AFB, Tex.	240
6	Carswell AFB, Tex.	210
7	Barksdale AFB, La	170
8	Little Rock AFB, Ark.	220
9	Columbus AFB, Miss.	205
10	Dobbins AFB, Ga.	220
11	Hunter AFB, Ga.	220
12	Kennedy Shuttle Base, Fla.	

Recovery of the shuttle from oversea bases or islands such as Hawaii or Guam would be by special converted ship. This method of ship transportation is also a consideration for trips from western U. S. landings to the east coast main shuttle base.

2.17.3.3 Secondary Field Support. The requirement to supply certain post-landing and preferry support at alternate post mission landing sites is met by making use of whatever site-located equipment can be used and by supplementing this with the necessary shuttle-peculiar support equipment flown in specifically for this operation.

The major support tasks from landing rollout to vehicle departure consist of:

Passenger and Crew Removal
Post-Landing Safing and Cooling
Payload Removal and Shipment
Ferry Kit Installation
Preflight Inspection, Checkout, and Maintenance
Preflight Fueling and Replenishing

The basic method of accomplishing the above tasks is to make use of the on-board vehicle systems and an air-transportable (fast response) support equipment package which provides ferry kit, payload removal and shipment, hydrogen tank inerting and whatever supplemental checkout and repair equipment is necessary.

Post-landing crew and passenger removal and transport, vehicle cooling, and tow-bar transporting of the shuttle will be provided with existing standard field ground support equipment. Safing and self checkout will be accomplished with on-board systems. Main hydrogen tank inerting, payload removal, and ferry kit installation will be accomplished with special equipment flow in as part of the support package. As the ferry kit only consists of a palletized JP-4 fuel tank with an aerial refueling stand-pipe system, this installation

is minimal. Inspection, checkout, and repair will be accomplished as required to meet a ferry flight minimum equipment list. Fueling will be accomplished using standard JP-4 fueling systems and cabin pressurization systems will be recharged using standard aircraft support equipment. The Auxiliary Power Units (APUs) require charging with hydrogen and oxygen which would be supplied from special refilling equipment.

As these APUs are the only vehicle subsystem that cannot be refilled or supported by enroute field ground support equipment, other methods of supplying power and hydraulics for control surface and landing gear activation are possible. Alternators and hydraulic pumps can be added to three of the air-breather engines which provide power pads for this purpose. However, this would make ferry flight preparations more complicated and imposes a tare weight penalty to the basic vehicle. Another alternative would be to add a JP-4 APU to the ferry kit pallet installed in the payload bay. But this would reduce the amount of ferry fuel carried and increase the complexity of the vehicle controls and ferry kit installation and checkout.

2.18 DEVELOPMENT TEST

Definition of space shuttle development test concepts considers planning and implementation of component, module, qualification, reliability, design verification, acceptance, and flight test aspects at all stages of the program leading to evaluation of the ultimate system development objectives and performance capabilities. Test requirements at each stage originate principally in the Engineering, Manufacturing, and Operations areas of the program, and the test activities are conducted in two major categories, engineering development test, and engineering flight test.

The objective of this section is to outline the technical approach and phase relationships for engineering development and flight test concepts incorporated in Volume III – PROGRAM PLANNING DATA, particularly the flight test program that paces the spans and phasing of the program master schedule for both stage-and-one-half and two-stage orbiter concepts. Alternative system development approaches are considered, leading to selection of an incremental approach utilizing single element orbiter suborbital vertical test flight modes with the primary rocket engines to penetrate transonic, supersonic, and hypersonic flight conditions that are otherwise unattainable with horizontal takeoff modes. This approach minimizes development risk and the degree of technical commitment at significant management milestones in the program by verifying design approaches in key areas of aerodynamics, thermodynamics, and structures at full scale and in the combined environments.

The potential capability of the orbiter vehicle to conduct test flights above Mach 5.0 at altitudes above 100,000 ft and return intact for a normal recovery is the most significant advantage offered by the reusable space shuttle concept in expediting low-cost development.

Although space shuttle development follows precedents set in aircraft experience, essential differences are apparent. An FAA certification program, such as that for the L-1011, involves 6 flight test vehicles concurrently over a

12-month span with 1700 test flights representative of normal operations. In terms of the number of events, this flight test activity is more than the total utilization of the space shuttle fleet during development and 10 years of operation. Further, the duty cycles expected in space shuttle are characteristically different and of two kinds. With a minimum fleet to support 75 launches per year, the active orbital systems cycle an average of 6 to 12 times per year with 200-hr continuous duty cycle requirements, whereas the rocket engine systems, propellant management systems, and airbreathing engine systems encounter the same frequency but with a duty cycle of 400 to 600 sec. Single element vertical flight test is costly and requires some special provisions, but is weighed against potential costs and development risk associated with reliance upon engineering analysis and horizontal flight tes[t] limited to the subsonic regime appropriate to most large aircraft develo[p]ments.

Comparison of a simplified space shuttle development schedule wit[h] the record of Saturn V system development in Fig. 2.18-1 tends to confirm feasibility of a 72-month baseline schedule to FMOF, although significant differences are noted. The most significant difference is the extensiv[e] ground test, static test firing, and developmental launch program in Satu[rn] V, characteristic of rocket design verification prior to manned flight[s]. In space shuttle, this is supplanted in part by longer design span as well as longer incremental flight testing before the vehicle system is committed to an "all-up" launch.

The following sections treat elements of the development risk assessment, and the engineering development and flight test approaches for the study.

2.18.1 Development Risk

A convincing quantitative measure of development risk is not apparent, in part because assessment of development risk is involved with assessment of unknowns. In this context, a program with low development risk is thought of as one with a low content of recognized unknowns and having a low probability of unidentified unknowns. However, since Explorer I stumbled into the Van Allen Belt, each new space flight development has exposed unknowns that affect

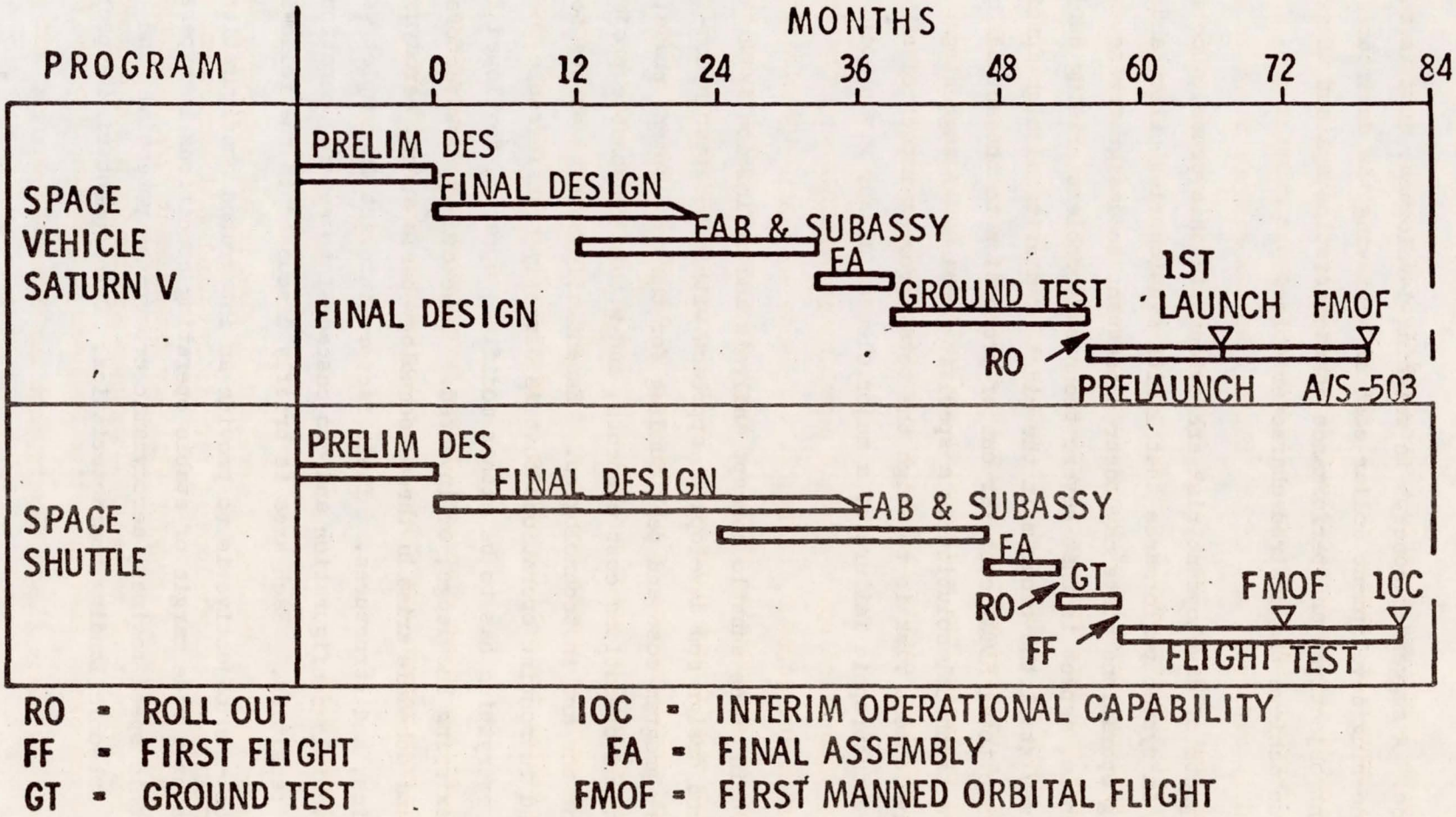

Fig. 2.18-1 Comparative Development Schedules

system performance. A suggested concept in weighing development risk is to balance the engineering development dollar cost of achieving the desired system development objectives and performance characteristics against the cost of failing to achieve the desired characteristics.

Two important aspects of development risk are evident in this sense. One is that degradation in system performance that limits mission flexibility and impacts recurring operations costs may occur subsequent to design freeze. Another, more subtle, aspect is that severe technical problems arising during the RDT&E phase may stall the program in the midst of a critical high funding period and increase total RDT&E cost far out of proportion to the actual direct cost of finding a technical solution to a specific problem. A suggested criterion for development risk is to weigh the total program impact of a vehicle loss or catastrophic failure of a major element at any given point in time.

A basic objective in space shuttle concept analysis and definition is to achieve a balanced design and development approach within acceptable constraints on total program cost and peak funding for the development phase, fleet investment and operations cost elements, and within reasonable projections of state of the art in technologies. Essentially nothing exists today at full scale and in routine operations that is directly transferrable to space shuttle; everything has to be either modified, uprated, developed from scratch within existing knowledge, or invented. Development risk factors in performance, cost, and schedule arise in three technology basic areas, aerodynamics, aerothermodynamics, and structures. These factors interact in complex ways throughout the vehicle configuration and can cascade into severe operational and performance penalties. Each area is briefly discussed in the following:

- Aerodynamics – An objective is to provide an integrated configuration that maintains a wide margin of stable operating conditions in hypersonic and supersonic speed regimes, aerodynamic crossrange potential, and acceptable subsonic landing characteristics. Development risk aspects

concern capability to predict aerodynamic characteristics across the entire flight regime in hypersonic, supersonic, transonic, and subsonic conditions, which affects attainable reentry crossrange performance, static margins of stability, and control functions in all flight regimes.

Typical design interactions include a sufficiently wide range of trimmed and stable vehicle pitch attitudes to maintain an altitude corridor above heating limits of the thermal protection system during hypersonic entry conditions; sufficient neutral or positive directional stability margins and aerodynamic control authority to maintain "fail-safe" attitude control from onset of "q" through critical reentry heating without dependence upon RCS, since more than 2 deg yaw may expose the "soft" portions of the heat shield to the full aerodynamic heating environment; and adequate subsonic handling qualities and aerodynamic performance to accommodate "fail-safe" power-off landing characteristics.

Aerothermodynamics – In thermal protection system design, a development risk aspect is capability to predict heat shield peak temperatures in the hypersonic regime. Prediction accuracy is affected by uncertainties in the thermal environment analysis models and equilibrium heat transfer models, as well as dispersions in entry trajectory guidance and control and atmospheric variation.

A further design interaction with the configuration is shockwave interaction with boundary layers and the extreme heating rate effects of shock impingement. Many of these effects are difficult to resolve short of full-scale flight experience.

Structures – Development risk aspects to be resolved in combined ground and flight test include primarily the dynamic environments; fluid dynamic effects in the propulsion system, tankage, and feedlines; and flutter, buffet, and aeroelastic instabilities in the external airframe and flight

control surfaces. Static structural design verification for inertially loaded members can largely be accomplished in ground test, whereas the induced dynamic environments must be experienced in flight throughout the transonic and supersonic regimes where significant interaction is expected.

Specific technology requirements in these three vital areas that bear investigation during engineering development and flight test phases are elaborated in Volume —Section 3; and the selected baseline test program spans, hardware requirements and phasing are covered in Volume III —Section 2.

In addition to technology aspects of development risk, there are some recognized tradeoffs in the two-stage and stage-and-one-half concepts addressed in this study. The two-stage concept is characterized by concurrent development of two airframes in a highly interactive approach, with ultimate demonstration of system objectives and performance dependent upon both elements. Dual primary engine configuration, facilities and GSE developments required are offset in some degree with commonality of subsystems and equipment. The stage-and-one-half concept integrates all functions into a single airframe with a single engine development, essentially 100 percent commonality except for the booster propellant containment function handled in the droptanks that either stage off in ascent or are carried to orbit, depending upon mission performance requirements. A low development risk aspect of the stage-and-one-half concept is that final droptank design is delayed until after the orbiter configuration design freeze at CDR, about 18 months into the program as discussed in Volume II - Section 2.2. This provides flexibility in sizing of the droptank propellant load and staging velocity until high confidence is established in orbiter configuration and weight, a programmatic development risk advantage.

Another more technical distinction concerns staging characteristics. Neither concept has a viable atmospheric abort mode under present considerations. Stage-and-one-half is committed to launch only with all engines running,

and has multiple engine-out capabilities for abort-to-orbit early in the ascent trajectory, i.e., fail-operational, fail-safe to abort orbit prior to droptank staging and fail-operational, fail-safe after staging. The two-stage concept has booster engine-out capability, but with two orbiter engines required to air start at staging, has single engine-out capability only for abort-to-orbit, i.e., fail-safe to abort orbit, fail dead. These development risk aspects reflect the different degree of technical commitment at significant milestones, for example at FMOF. Further, the stage-and-one-half concept is not required to demonstrate droptank staging at FMOF under the planned mode discussed in Volume III - Section 2.5 in which droptanks are carried to the injection orbit for a benign zero "g" separation. If committed to launch, the stage-and-one-half orbiter is assured an opportunity to exercise the entire mission profile. Thus significant configuration and concept aspects are involved in assessment of development risk in context with a low program content of recognized unknowns, and low probability of unrecognized unknowns.

2.18.2 Flight Test Philosophy

Alternative approaches to the engineering development and flight test objectives and phasing have been considered in examining the impact on master schedule to FMOF and the degree of system maturity at key management decision milestones.

Alternative Approaches - One approach to design validation test considered is the "all-up" first orbiter vertical flight to orbit, backed up with the horizontal flight test program and the engineering development test effort. This approach involves an unprecidented degree of technical commitment even compared to Saturn/Apollo 503; including entry with an airframe that has not demonstrated flight above approximately Mach 0.6 or flown vertically as a rocket. Present indications are that the orbiter cannot penetrate transonic and supersonic flight regimes from a horizontal takeoff; specifically, cannot take off on

turbojets with enough rocket fuel to accelerate for a supersonic "dash" due to weight limitations by landing gear load factors, take-off speed and lift coefficient. Also, regardless of weight limits the orbiter cannot accelerate on turbojets alone, because of inadequate thrust and the evident mismatch of engine air inlet and exhaust nozzle configuration, which requires the engines to be retracted at supersonic speeds to avoid extensive damage. Means to overcome these constraints are not evident.

The "all-up" approach makes no use of the reusable flight test capability offered by the orbiter, and places reliance entirely upon engineering analysis, simulation, ground test, and subscale model test.

A number of alternative approaches with varying degrees of incremental subscale precursor flight test vehicles have been proposed, ranging from B-52 drop tests similar to the X-15 and X-24A, to rocket launch such as Asset and Prime test vehicles. Cumulative flight time above Mach 5 in these programs is limited to less than 10 hours. Precursor test vehicle concepts for developing and demonstrating technologies required for advanced hypersonic and orbital entry systems are listed in Table 2.18-1 These options are based on the design and manufacture of a test vehicle that can be incrementally uprated in capability and performance in its flight envelope. Each contains its own technological developments which may be dead-ended and carry no direct contribution to full scale flight vehicle hardware other than proof of feasibility in a certain technology base. The schedule span indicated may overlap the design phase and is not simply a delay. Free flight subscale models fall generally into these categories:

- Subscale aerodynamic models of the full scale vehicle launched by B-52 to provide aerodynamic data at larger scale and higher Reynolds numbers than can be achieved in wind tunnels. Mach 2.0 capability is expected.
- Thermodynamic test sections, notably the nose and first several feet of the full scale vehicle could be flown by rocket launch in the hypersonic regime for testing heating prediction accuracy.

TABLE 2.18-1

DEVELOPMENT OPTIONS FOR PRECURSOR FLIGHT TEST VEHICLES

PROGRAM	ORDER OF PREFERENCE	SCHEDULE SPAN-MOS	OVERALL RATING
40—FT MANNED VEHICLE (LOW SPEED, B-52 LAUNCH)	1	18	FAIR
40—FT MANNED VEHICLE (HIGH SPEED, B-52 LAUNCH)	2	24	GOOD
40—FT UNMANNED VEHICLE (T-IIIC LAUNCH)	3	30	EXCELLENT
40—FT MANNED VEHICLE (T-IIIC LAUNCH)	4	40	EXCELLENT

Table 2.18-1

It is not certain that transition can be achieved on this type of vehicle, in which case only laminar heating would be realized if actual flight trajectory is flown. There is some hope that higher atmospheric density trajectories could be flown to promote transition as a basis for extrapolation to total vehicle heating.

- Construction of subscale models of the complete configuration to be flown on existing boosters offers some prospect for solutions, but due to the cost and time involved it is not believed that these alternatives should be adopted.

Since the orbiter is primarily an aluminum airframe, another alternative considered is a subsonic all-aluminum full scale prototype introduced about 18 months into the development program; that would accomplish the typical taxi tests, horizontal takeoff, turbojet engine installation shakedown, subsonic aerodynamic stability and control, handling qualities, development of flaps and other aerodynamic control surfaces, flutter and dynamics investigation, and verify the dead stick and powered landing characteristics of the configuration. The objective in this concept is to force engineering development by getting into the air as soon as possible in the program with a reasonable full scale prototype that takes over a large portion of the turbojet powered subsonic flight testing, thereby cutting perhaps 12 months out of the schedule to first vertical rocket powered orbiter flight test.

The extent to which precursor flight test vehicles may be used in space shuttle development is a matter of further definition study, and no recommendation is made at this point. A key decision element in assessment of development risk implications requiring application of precursor test vehicles is whether an "all-up" or vertical suborbital flight test mode is adopted.

Incremental Flight Test Approach - Previous NASA programs have of necessity been "all-up" to a considerable extent because of the use of expendable launch vehicles and spacecraft which could not take advantage of unique autonomous operation and intact recovery features available within the space shuttle concept.

The incremental flight test approach to design maturity is indicated qualitatively in Figure 2.18-2, which implies a minimum level of confidence to initiate FTV-1 horizontal flight test based on engineering development test, essentially at 80 percent demonstration of expected flight loads in the STV The interrelationship of engineering development and flight test activity is discussed in Volume III - Section 2.2 and 2.5. There is a maximum level of maturity achievable by ground test alone that is short of that required to initiate full environmental stress. At some design maturity level, sufficient confidence exists to initiate vertical rocket powered flight into the supersonic flight regime, proceding incrementally to build up stress levels in aerodynamic, thermodynamic and structural development risk areas, and with abort back to safe conditions previously experienced if incipient critical conditions are encountered. During this phase of testing reliance is placed on available extensive range support from mission control, deployed tracking and recovery forces, and crew escape capsules. The approach to implementing this test phase is discussed in the test plans outline in Volume III – Section 2.5.

Specific technical objectives of single element vertical flight are indicated in Fig. 2.18-3, which qualitatively presents some of the constraints that have to be dealt with. It is desirable to penetrate both heat shield temperature boundary and acceleration boundary conditions. In order to be able to get on the temperature boundary the orbiter has to have the capability to inject into the corridor defined between the equilibrium glide limit and the temperature and acceleration limits. These limits move to higher altitudes with increasing angle-of-attack, and the capability to achieve necessary velocity depends upon mass ratio available in the system concept. Capability of a two-stage orbiter is substantially higher than a stage-and-one-half orbiter, which can probably approach temperature boundary only by exceeding normal acceleration limits.

Flight Test Capability - Results of typical vertical flight test capability analysis are given in Fig. 2.18-4 for both stage-and-one-half and two-stage orbiter configurations, and at 15 deg angle-of-attack. Ranges of test conditions exist at other angles of attack, limited by mass ratio and stability

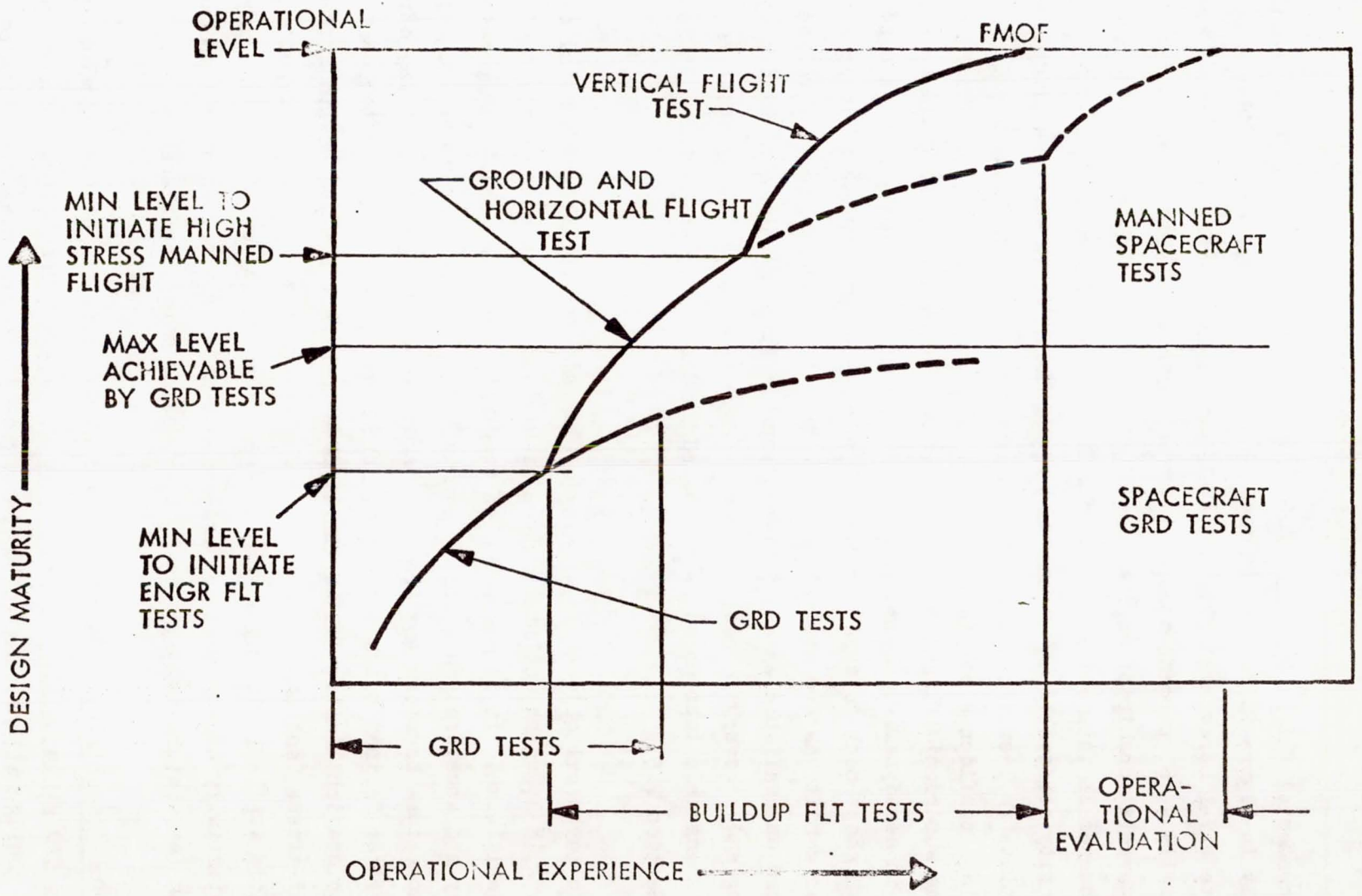

Fig. 2.18-2 Design Maturity

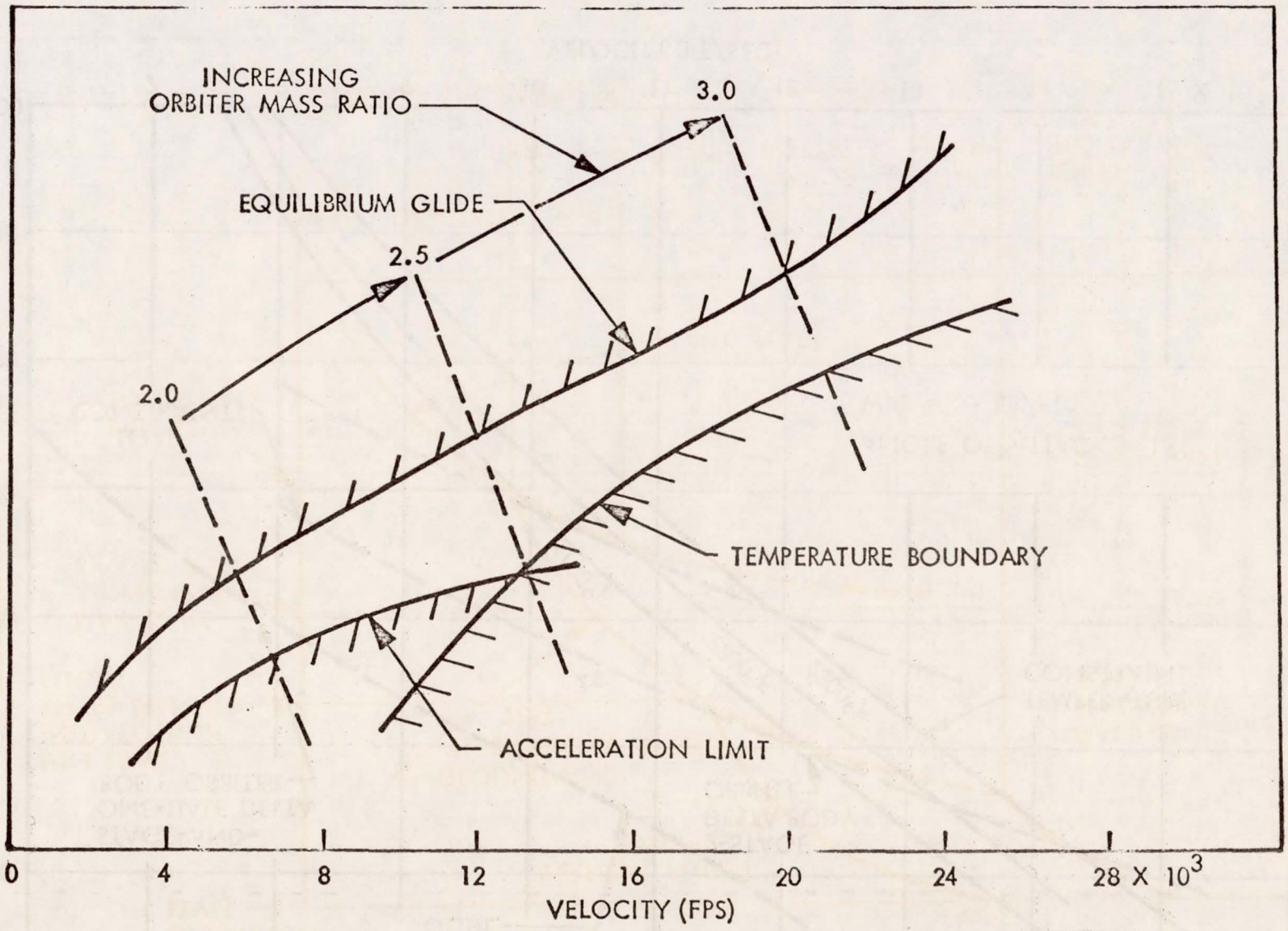

Fig. 2.18-3 Orbiter Flight Test Considerations

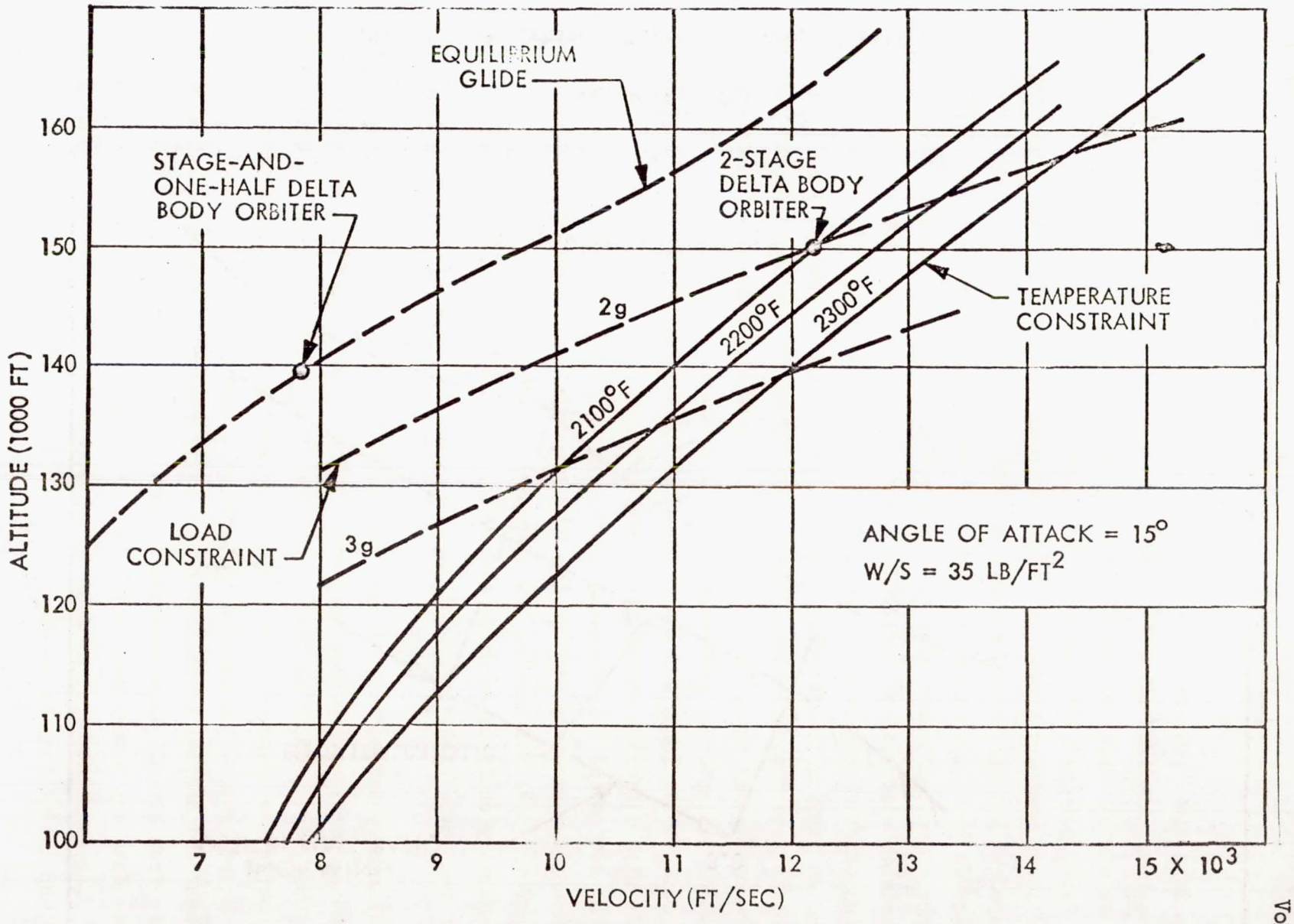

Fig. 2.18-4 Single Element Flight Test Capability

and trim considerations. For illustration, the two-stage orbiter case taken from prior studies with liftoff weight at 778,000 lb uses two booster engines in an ascent trajectory constrained by maximum "q" of 500 psf and alpha-"q" of 2,500 psf-deg, with injection at 150,000 ft since this altitude represents the lower boundary within the ascent dynamic constraints. The trajectory is shaped with insertion at a negative flight path angle. The injection velocity achieved is 12,200 ft per sec, and the test point achieved is the 2100°F temperature constraint with 2g acceleration. About 100 sec of flight time are available riding along the 2100°F boundary up to 3g acceleration limit, at which other maneuvers would be initiated to follow a typical reentry trajectory to landing and recovery. At any point, a pullup maneuver lofts the vehicle toward equilibrium glide and away from the acceleration and temperature boundaries using just the energy management techniques. Numerous injection trajectories have been run to achieve different test altitude and velocity conditions, and a wide range of descending flight paths are attainable with different energy management programs. A typical ascent range is about 140 nm and the unpowered glide range is approximately 1000 nm indicating a downrange recovery site on the launch azimuth for 55-deg inclination at Pease AFB near Portsmouth, New Hampshire, if this azimuth is chosen. This trajectory would be entirely over water and within range of deployed tracking and recovery forces.

Preliminary estimates of flight test performance of the LS 200-5 stage-and-one-half orbiter reported in Section 2.2 of Volume II provide an injection velocity of 7,700 ft per sec at 138,000 ft altitude in a typical case, with conservative assumptions of losses and inert weights to accommodate 170,000 lb of propellant in the payload bay, and utilizing the entire capacity of the ascent and orbital propellant tanks. Performance is evaluated for a gravity turn from liftoff with constant thrust, and for rotating earth model. Pertinent data are summarized in Tables 2.18-2 and 2.18-3. A burnout acceleration of 3.4g is obtained with coast to apogee and initiation of test conditions on the descending leg at about 140,000 ft altitude, and 343 nm downrange.

TABLE 2.18-2

LS 200-5M Flight Test Vehicle Weight Summary

Orbiter dry weight	269,427
Personnel	725
Residual propellant	2,821
Payload bay tankage	10,000
Additional Payload bay structure	10,000
Inert weight	292,973
Propellant reserve	8,609
Inflight losses	3,965
Propellant - cruise	3,477
Propellant - ascent*	361,700
Propellant - maneuver/ACS**	36,175
Total gross weight	706,899
Impulse propellant	393,260
Burnout Weight	313,639

* Include 170,000 lb in payload bay tank

**31,560 lb orbital propellant used for descent

Table 2.18-2

TABLE 2.18-3

Flight Test

LS 200-5M Test Vehicle Trajectory Characteristics
(Estimated)

1.	Liftoff thrust-to-weight	1.50
2.	Burnout velocity - fps (actual)	7,772
3.	Burnout altitude - ft	138,800
4.	Range - nm	343*
5.	Velocity losses - fps	3,848
	Gravity - 3340 fps	
	Drag - 123 fps	
	Thrust - 385 fps	
6.	Maximum ascent dynamic pressure - psf	780
7.	Burnout flight path angle - deg	30

*To 138,800 ft altitude on the descending leg

Table 2.18-3

At this point "q" is 190 psf and Mach number 7.3. The stage-and-one-half condition is shown also in Fig. 2.18-4, indicating injection very close to the equilibrium glide condition at the reference angle-of-attack at 15 deg. The temperature boundary can be approached at higher angles of attack, and energy management techniques applied to fly descending trajectories similar in principle to the two-stage orbiter. The full range of test conditions has not been explored for LS 200-5. The proposed downrange recovery site for launch from KSC is Myrtle Beach AFB in South Carolina, with an overwater trajectory. This site is adjacent to the Inland Waterway for return of the vehicle if this mode is considered. The available runway length of 9,500 ft is considered sufficient, but bearing strength of 165,000 lb would have to be uprated. Radar coverage and navigation aids, PAR, ASR, TACAN are adequate; ILS and VORTAC need to be provided.

The range of flight test conditions is also dependent upon wing loading and angle-of-attack for the attainable injection altitude and velocity range. At the higher wing loading indicated in Fig. 2.18-5, for 15-deg angle-of-attack, the 2200°F temperature boundary can be reached within 2g acceleration. A typical range of test capabilities for both stage-and-one-half and two-stage orbiters is indicated.

Another flight test performance analysis of interest is the benign staging mode for FMOF with stage-and-one-half. The due east launch capability of the LS 200-7 configuration reported in Section 2.3 of Volume II, with FPR and 650-ft per sec on-orbit reserve, is 16,000-lb to 100 nm with ABES installed, or 46,000 lb with ABES removed, in addition to injection of the droptanks at the 50 nm injection condition. The droptank weight quoted in this instance is 112,245 lb dry or 127,600 lb with residuals.

2.18.3 Development Test Approach

The schedule spans, interrelationship, and technical approach for implementing engineering development test and engineering flight test based on the incremental single element vertical flight test concept are described in Volume III - PROGRAM PLANNING DATA, Sections 2.2 and 2.5.

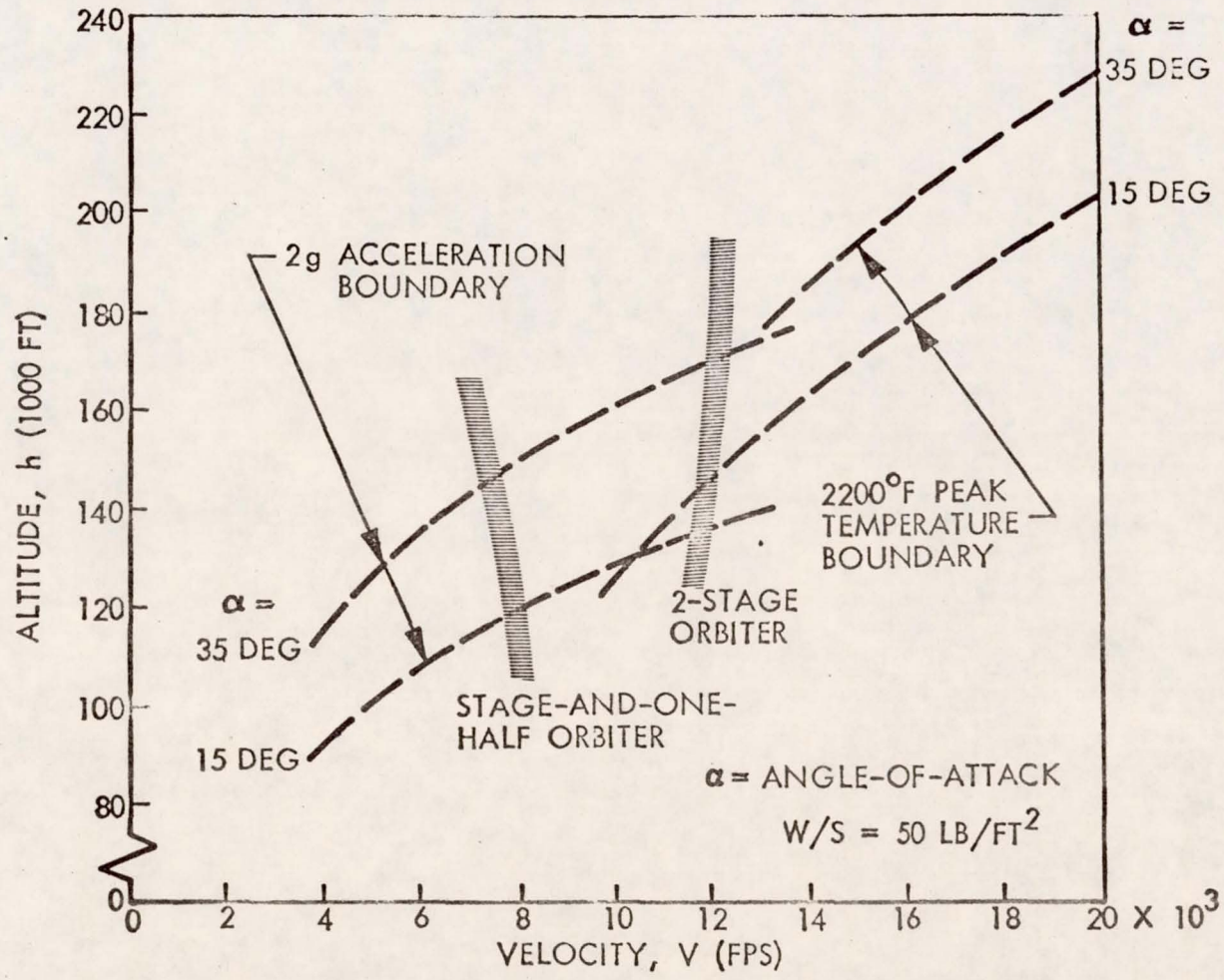

Fig. 2-18-5 Orbiter Hypersonic Test Conditions

2.19 SYSTEM COSTS

2.19.1 Cost Estimates

System costs were estimated for the LS 200-7 configuration of the stage-and-one-half system. These costs are covered in detail in Volume IV and summarized as follows:

DDT&E	$ 4,378 M
Recurring Production	487
Recurring Operations (44 flts)	3,158
Total Program	$ 8,023 M
Orbiter First Unit (including engines)	$ 143.4 M
Droptank First Set (2 tanks at 90% learning)	8.63 M
Average DDT&E Cost/Flight	$ 9.84 M
Average Recurring Production Cost/Flt	1.09
Average Recurring Operations Cost/Flt	7.10
Total Average Cost/Flt	$ 18.03 M

Cumulative cost versus time for this program is shown in Fig. 2.19-1. Annual funding was found to have a peak of $1.16 billion with a profile as shown in Fig. 2.19-2. Net present value at 10 percent discount rate is $3.96 billion as shown in Fig. 2.19-3.

When compared to the other systems costed, the stage-and-one-half proved to be the most economical in terms of total program cost, peak annual funding, and net present value.

2.19.1 Cost Sensitivities

For a situation where system performance must be maintained constant and all weight changes are accommodated by changes in GLOW, the following weight partials for the stage-and-one-half system have been derived:

$$\frac{\Delta \text{GLOW}}{\Delta \text{Orbiter Inert}} = 22.5$$

$$\frac{\Delta \text{Droptank Dry}}{\Delta \text{Orbiter Inert}} = 2.07$$

$$\frac{\Delta \text{ Propellant}}{\Delta \text{Orbiter Inert}} = 19.43$$

Total DDT&E costs for droptanks are:

$$C_{DDT\&E} = (C_D + C_{TH})\ 1.113$$

where C_D is the development cost for the droptanks, C_{TH} is the cost of the test hardware and 1.113 is the factor used to account for systems integration and program management.

Therefore:

$$\Delta C_{DDT\&E} = (\Delta C_D + \Delta C_{TH})\ 1.113$$

In the current CER's,

$$C_D = .225 \times 10^6\ (W)^{.578}$$

$$C_{TH} = 6 \times 5.95 \times 10^3\ (W/2)^{.607}$$

where W is the dry weight of one set of tanks (2 tanks).

Differentiating these equations:

$$\frac{\Delta C_D}{\Delta W} = (.225 \times 10^6)\ (.578)\ (W)^{-.422}$$

$$\Delta C_D = \frac{.130 \times 10^6}{W^{.422}}\ \Delta W$$

and

$$\frac{\Delta C_{TH}}{\Delta W} = \frac{35.7 \times 10^3}{(2)^{.607}}\ (.607)\ W^{-.393}$$

$$\Delta C_{TH} = \frac{14.26 \times 10^3}{W^{.393}}\ \Delta W$$

For the current droptank weight of 112,162 lb, these become:

$$\Delta C_D = 962\ \Delta W$$

$$\Delta C_{TH} = 148\ \Delta W$$

Then

$$\Delta C_{DDT\&E} = (962\ \Delta W + 148\ \Delta W)\ 1.113$$

$$\Delta C_{DDT\&E} = 1235\ \Delta W$$

In other words, an increase in droptank dry weight of 1 lb produces an increase in DDT&E costs of $1235.

The other element of droptank costs is in the area of recurring operations. These are given by:

$$C_{OPS} = (C_{PROD})\ 1.113 + (C_{SE} + C_{PROP})\ 1.28$$

and

$$\Delta C_{OPS} = (\Delta C_{PROD})\ 1.113 + (\Delta C_{SE} + \Delta C_{PROP})\ 1.28$$

where

C_{PROD} = the production costs for the droptanks
C_{SE} = the sustaining engineering costs during production
C_{PROP} = the cost of propellants for the program

C_{PROD} is the first unit cost of the tank projected on a 90 percent learning slope for 890 units (2 tanks/flight x 445 flights) or:

$$C_{PROD} = C_{TFU} \times 373.35$$

where

$$C_{TFU} = (5.95 \times 10^3)\ (W/2)^{.607} = 3.914 \times 10^3\ (W)^{.607} = \text{first unit cost}$$

Then:

$$C_{PROD} = 3.91 \times 10^3\ (W)^{.607} \times 373.35 = 1.460 \times 10^6\ (W)^{.607}$$

$$\frac{\Delta C_{PROD}}{\Delta W} = \frac{1.46 \times 10^6\ (.607)}{W^{.393}} = \frac{.886 \times 10^6}{W^{.393}}$$

$$\Delta C_{PROD} = 9.18 \times 10^3\ \Delta W$$

C_{SE} is calculated as 3 percent of C_{PROD}. Therefore

$$\Delta C_{SE} = .03 \times \Delta C_{PROD} = .03 \times 9.18 \times 10^3 \Delta W$$

$$\Delta C_{SE} = 275 \Delta W$$

C_{PROP} is the cost of propellant for 445 flights at \$.10/lb. Therefore:

$$C_{PROP} = 445 \times W_{PROP} \times .10 = 4.45 W_{PROP}$$

$$\Delta C_{PROP} = 44.5 \Delta W_{PROP}$$

Total change in recurring operations cost are then:

$$\Delta C_{OPS} = (9180 \Delta W)\ 1.113 + (275 \Delta W + 44.5 \Delta W_{PROP})\ 1.28$$

$$= 10{,}217 \Delta W + 352 \Delta W + 57 \Delta W_{PROP}$$

From the weight sensitivities,

$$\frac{\Delta W_{PROP}}{\Delta W} = \frac{19.43}{2.07} = 9.39$$

Therefore:

$$\Delta C_{OPS} = 10{,}217 \Delta W + 352 \Delta W + 535 \Delta W$$

$$\Delta C_{OPS} + 11{,}104 \Delta W$$

In other words, an increase in droptank dry weight of 1 lb produces an increase in recurring operations costs of \$11,104.

The total impact on program costs for a 1 lb increase in droptank dry weight is the \$1235 DDT&E increase plus the \$11,104 recurring operations increase or \$12,339. This relationship obtains for any case where a change is made in droptank design which increases its weight. As long as the droptank design is considered "rubberized" and, as such, is allowed to absorb all other weight changes in the system, these changes will be reflected in the form of contractions and expansions of droptank size and propellant capacity. Therefore, increases or decreases in droptank size may arise not only from a design change in the droptank itself but also from any changes which effect the weight of the orbiter.

The relationship between orbiter inert weight (W_o) and droptank dry weight (W) therefore becomes important. From the weight sensitivities,

$$\Delta W = 2.07\, \Delta W_o$$

Therefore, the cost sensitivities with respect to orbiter inert weight become:

$$\Delta C_{DDT\&E} = (2.07)\,(1235)\,\Delta W_o = 2556\ \Delta W_o$$

$$\Delta C_{OPS} = (2.07)\,(11{,}104)\ \Delta W_o = 22{,}985\ \Delta W_o$$

$$\Delta C_{TOTAL} = 25{,}541\ \Delta W_o$$

This says that an increase of 1 lb in orbiter inert weight will produce an increase of $25,541 in total program cost because of the additional droptank and propellant weights which it produces. Conversely, it says that any design change which produces an increase of 1 lb in the orbiter inert weight must yield a savings of $25,541 in other program costs in order to break even. In this case, it is assumed that the original 1 lb increase in orbiter inert weight is a change intended to reduce costs and does not cause any cost increase in the orbiter itself.

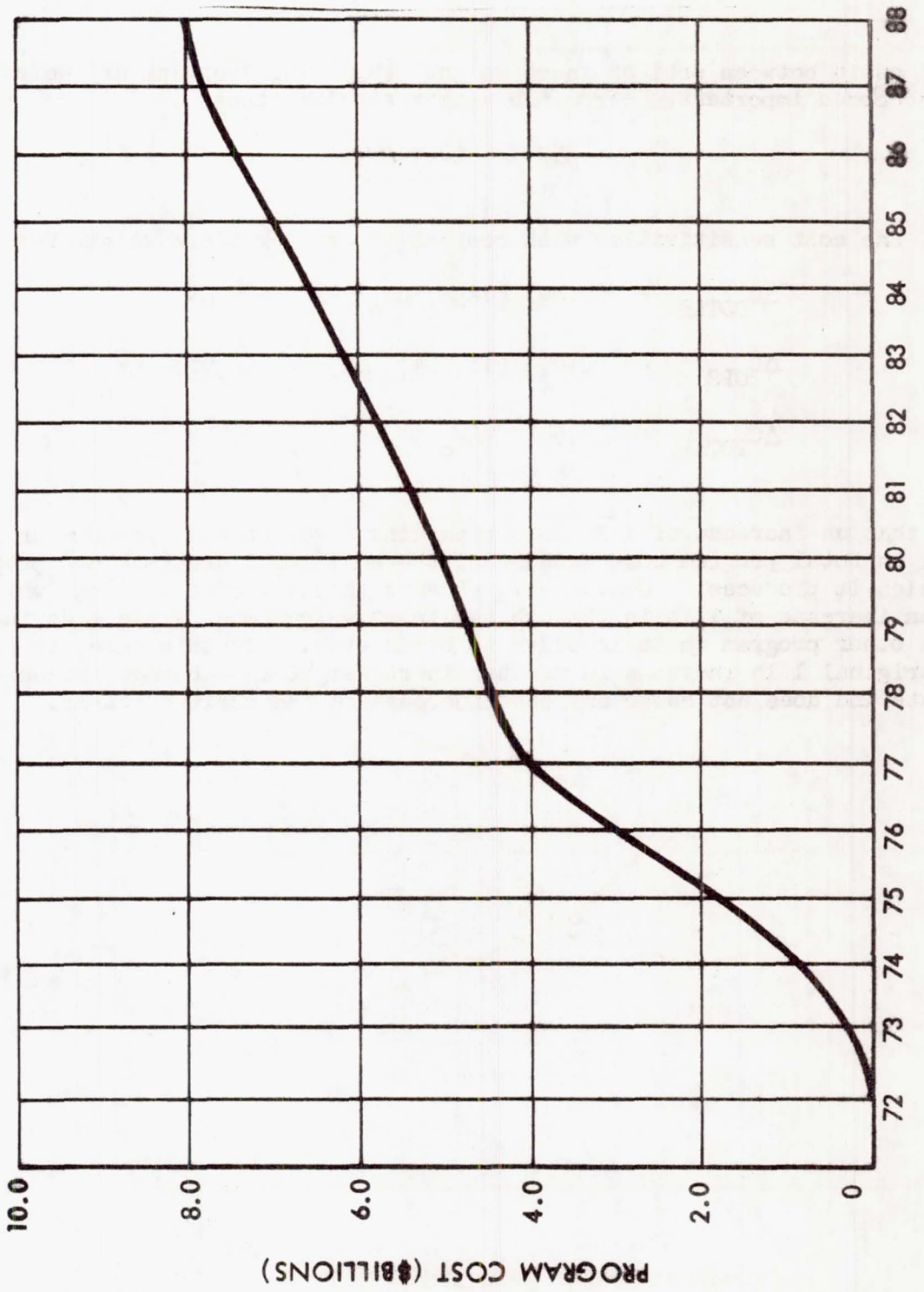

Fig. 2.19-1 Stage-and-One-Half Cumulative Cost

Fig. 2.19-1

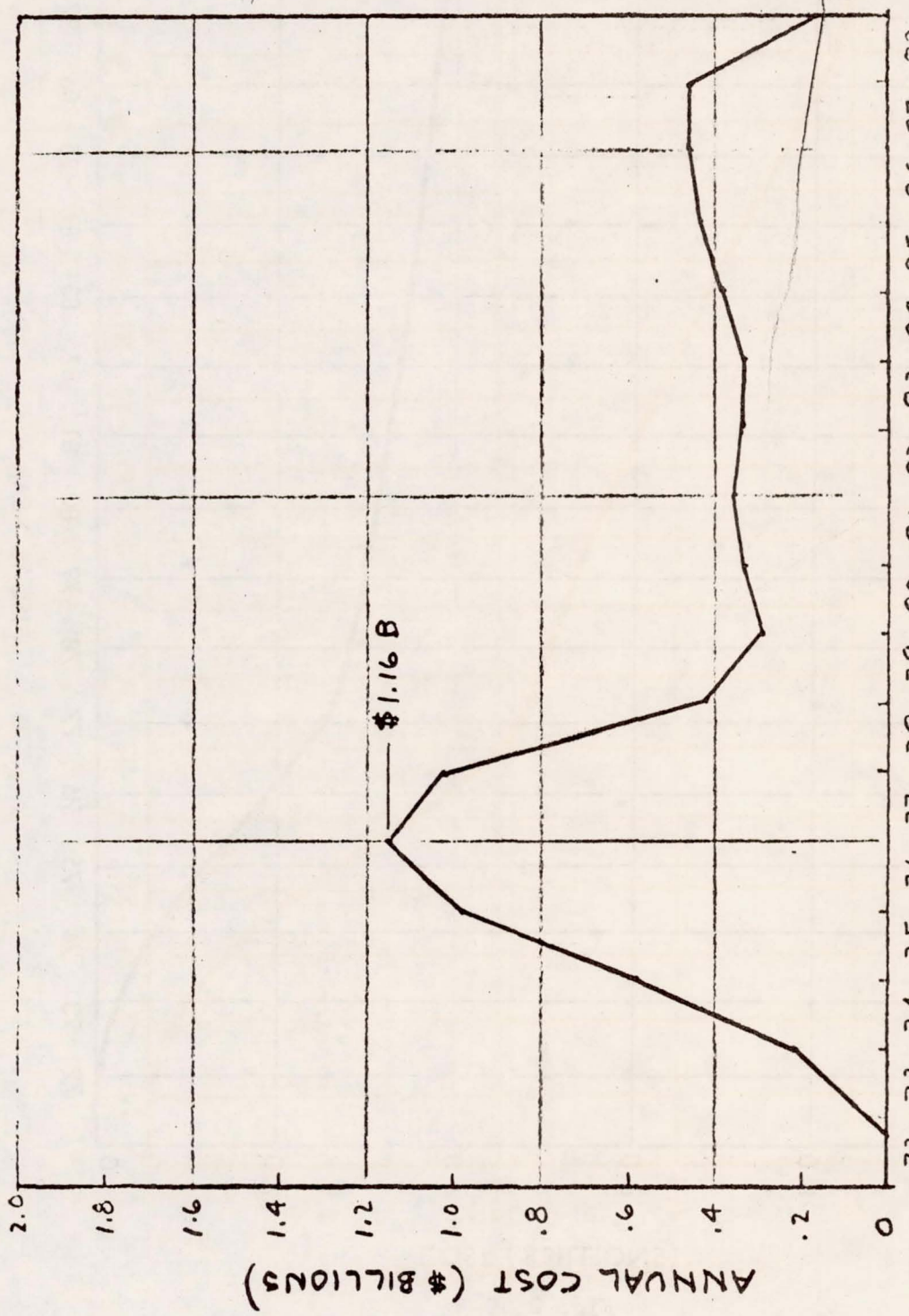

Fig. 2.19-2 Stage-and-One-Half Annual Cost

Fig. 2.19-2

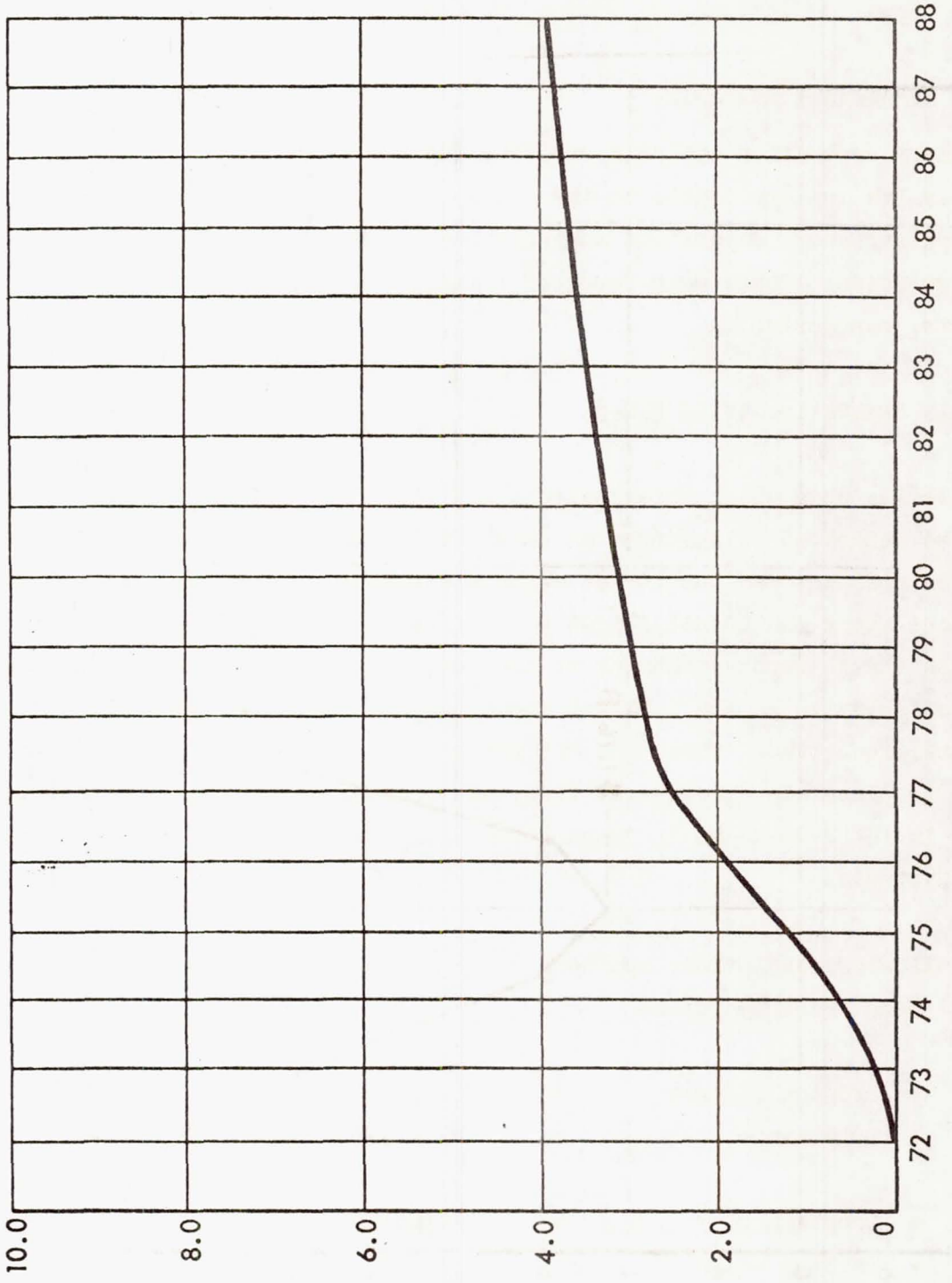

Fig. 2.19-3 Stage-and-One-Half NPV at 10%

Fig. 2.19-3

2.20 SEPARATION ANALYSIS

A primary separation analysis was done for the LS 200-3 configuration, but the results are applicable to the LS 200-10 as well. Subsequently, an atmospheric abort separation analysis was conducted for the LS 200-5 configuration. These were reported formally in the fifth and sixth monthly reports, respectively.

2.20.1 Separation of LS 200-3

The premise upon which this analysis was based is that the composite vehicle will be separated while under an acceleration of approximately 1g. The thrust will be constant and the rocket engines gimballed to provide that the orbiter continue along the normal flight path. The separation sequence will be to release the forward attachment at the initial signal, allow the droptanks to rotate (relative to the orbiter) about the aft pinned attachment until some predetermined time. Then, the aft pin will be pulled which will allow the tanks to translate upward away from the orbiter. This analysis is reported fully in EM L2-06-01-M4-2, "Separation Analysis for LS 200-3 Stage-and-One-Half System".

The following assumptions applied:

1. Mass Property Data -

• Droptank Weight	=	127,000 lb
• Droptank M.O.I.	=	12×10^6 slug-ft^2 (about cg)
	=	30×10^6 slug-ft^2 (about pivot)
• Droptank CG	=	73 ft from pivot
• Orbiter Weight	=	530,000 lb
• Orbiter M.O.I.	=	12.5×10^6 slug-ft^2 (about cg)

2. Flight Path Angle = 2 deg
3. TVC Gimbal Angle = 7 deg + 1/2 deg (6 deg used to allow 1-1/2-deg margin)
4. Thrust at 660K Total
5. Separation Sequence -
   - Forward attachment release at separation signal
   - Tank Rotation about aft pivot
   - Aft pin pull releasing droptanks
6. Thrust vector lines up with the pivot point and the composite cg.
7. Negligible effects from aerodynamics (q = 4 psf, M = 21.0, H = 272K ft)

With the rocket engines thrusting and providing approximately 1g acceleration on the vehicle, stage separation is possible without any augmentation. The initial angle of the droptanks should be about 5 deg to accomplish separation within the nominal 6 sec used for performance calculations. Constrained or unconstrained droptank trajectories are possible, depending on the judgement between the acceptability of sliding contact between the two bodies (a ramp) versus an additional 3 ft opening in the orbiter upper surface and a corresponding reduction of thrust to 20 percent.

The following tentative conclusions may be derived from this analysis:

1. No separation augmentation is necessary with engines thrusting.
2. The initial droptank angle should be at least 5 deg above the flight path angle to achieve separation in approximately 6 sec.
3. A method should be provided to assure that the droptanks have the proper angle at the time of separation signal.
4 Separation with zero ramp is possible if physical contract between the droptanks and the orbiter is acceptable for approximately 0.9 sec after release.
5. An unconstrained separation after release is possible with about a 3-ft opening aft of the pivot station if the engines could be throttled to 20 percent from 50 percent.

6. These conclusions should apply to the LS 200-5 and LS 200-10, as well, because the separation concepts are the same.

2.20.2 Atmospheric Abort Separation

To determine the interference aerodynamic characteristics, a selected abort condition, i.e., engine cutoff at 100 sec after launch and coast to a dynamic pressure 100 psf and a Mach number of 1.6 is utilized for the nominal initial abort conditions. The analysis is restricted to three-degrees-of-freedom in the pitch plane. Nominal controlled and uncontrolled staging events were analyzed for comparison with the abort results. The needed aerodynamic characteristic estimates for atmospheric staging were established. The LS 200-5 stage-and-one-half launch vehicle (Drawing SKS 100022) aerodynamic abort staging concept was be analyzed to allow better definition of the required aerodynamic test conditions and program.

The staging problem for stage-and-one-half vehicles at the initiation of tank release (front attachment released) consists of two, freely accelerating systems connected by an aft pinned joint. For the purpose of this study, the tanks were allowed to rotate until a predetermined, relative release-angle of 55 deg was attained. When this occured, the aft joint was released and total separation was reached. The aft joint incorporated a ramp which forced the tank attachment fitting to slide out of the spacecraft. The detailed design of this aft joint was not considered for this study, and the two bodies were taken to be mutually independent when the aft pin was released. Nominal staging was taken to occur at an altitude (H) of 272.3K ft, freestream velocity (V_{∞}) of 18.6K fps, flight-path angle (γ) of 2 deg, and thrust of 695Klb. The nominal abort considered was for total engine shutdown at T = 100 sec. The entire launch vehicle was allowed to coast after engine shutdown until q_{∞} = 100 psf, M_{∞} = 1.6, H = 80.5Kft and γ = 25.4 deg. (An active control system was assumed to be operating during the coast period.)

Nominal controlled and uncontrolled exoatmospheric staging analyses were conducted for the LS 200-5 one-and-one-half stage vehicle. The results indicated the successful attainment of the required relative tank-spacecraft angle at 6.8 and 8.4 sec after staging-initiation for controlled and uncontrolled staging, respectively. Spacecraft angular deviations with respect to the horizon were held to less than 2 deg. The results of these analyses are reported in EM L2-01-M1-6, "Atmospheric Abort Staging on the LS 200-5 Stage-and-One-Half Launch Vehicle".

Abort staging for total-engine shutdown at T = 100 sec is feasible with or without dumping the droptank LOX. The required separation angles can be obtained for a reasonable range of initial launch vehicle angles-of-attack. Staging times on the order of 7 and 4 sec for LOX-aboard and LOX-dumped can be obtained. Special care must be taken to minimize the large spacecraft negative angle-of-attack values attained for a wide range of initial conditions for the LOX-on cases. The feasibility of LOX dumping, resulting in an aerodynamically unstable launch configuration, must be further examined in relation to the level of stability augmentation attainable by the reaction control system.

3.1 REQUIREMENTS, GROUNDRULES, AND ASSUMPTIONS

The requirements and groundrules for the two-stage system are the same as those for the stage-and-one-half system as defined in Section 2.1. Since these are the same requirements against which the Phase B studies are working, the data here are directly comparable.

In order to derive total system performance, weight, and cost, a booster vehicle had to be defined and costed. Although design and weight data on the McDonnell Douglas booster as of March 1971 were available, this booster was larger than necessary for the Lockheed LS 400-7A two-stage orbiter. Consequently, scaling laws were derived by which the McDonnell Douglas booster could be resized (but not redesigned) to be compatible with the Lockheed orbiter. The resulting system, including the booster, was costed, using Lockheed CERs and in accordance with the costing assumptions and groundrules shown in Section 2.1.

Section 3
THE TWO-STAGE SYSTEM

Under Task 2, Growth to Two-Stage, which was added to the study at the end of the sixth month, it was necessary for Lockheed to define, size, and cost a two-stage space shuttle system to use as a reference against which the performance and cost of a system which would be converted from a stage-and-one-half to a two-stage system could be compared. A reference two-stage orbiter was designed based on the Lockheed delta body configuration and designed to be mated with a McDonnell Douglas Phase B booster. The system was refined, analyzed, and costed on the basis of a scaled down booster. Designated the LS 400-7A, this system, with its performance and cost, is reported here.

3.2 SUMMARY OF LS 400-7A SYSTEM

3.2.1 General Design Considerations, Operation and Performance

3.2.1.1 Sizing Considerations. The primary differences between the two-stage and stage-and-one-half space shuttle systems affecting vehicle sizing, design, and operation were discussed in Section 2.2.1. Summarizing, it was found that the determining design factors for the two-stage system differ from those for the stage-and-one-half system because of:

(1) A substantial reduction in staging velocity, resulting in
- A very large increase in the amount of propellant loaded in the orbiter

(2) The use of two separate and fully independent propulsion systems in the orbiter and the booster, resulting in
- A very great reduction in number of main rocket engines installed in the orbiter (from 9 to 2 when current Space Shuttle Engine ICD engines are used)
- The use of different engine expansion ratios in booster and orbiter
- The requirement to start the orbiter engines during flight under a high-altitude environment

As a result, the two-stage system orbiter is sized primarily by the requirements to provide space for the payload bay and the tankage for the ascent propellant. In addition, the effects of the reduced number of main rocket engines under the one-engine-out condition on performance and design requirements must be given special attention.

On the other hand, removal of the booster engines reduces the orbiter inert weight and also considerably alleviates the problem of balancing the cg with the aerodynamic forces. Consequently, the two-stage orbiter design and system performance are much less sensitive to main rocket engine weight characteristics than with the stage-and-one-half system.

Since to perform the design of the booster was not within the scope of the study, the required booster characteristic for establishing orbiter interface and system performance were determined by scaling the characteristics of a Baseline booster. For this baseline, the High-Crossrange, Canard Booster configuration documented by McDonnell-Douglas Mass Properties Status Report 8, dated 3 March 1971, was selected. Basic characteristics of this booster and the major assumptions made regarding the booster-orbiter attach points, other interface characteristics, and for the scaling of the booster are presented in Section 3.2.2.

For arriving at the two-stage vehicle configuration, in a first step by preliminary analysis, orbiter sizing requirements were established which were commensurate with baseline booster geometry and load-carrying capability and which also could be expected to come close to providing minimum GLOW with a vehicle system satisfying all mission requirements. Using these sizing requirements, a Baseline Orbiter configuration was designed and defined in detail. Combining the baseline orbiter with the baseline booster, a vehicle system was obtained to serve as the basis for supporting analyses performed for the orbiter design. This intermediate configuration will be referred to in the following as the Two-Stage Unadjusted Vehicle System. Since its performance capability was found to exceed the Level I requirements for the design mission and the two reference missions, a final Two-Stage Baseline Vehicle System was obtained by scaling the baseline booster characteristics to bring the system performance into accord with the specified values.

On the basis of earlier intensive two-stage performance and design analyses, which were updated to reflect the new NASA requirements, it was determined that the required propellant load for an optimal system could be approximately accommodated in the delta-body orbiter without changing the dimensions of the stage-and-one-half baseline configuration discussed in Section 2.2 of this report. Consequently, the Two-Stage Baseline Orbiter design was obtained by arranging the entire vehicle system around the largest propellant tanks of simple geometric design which could be packaged inside the LS-200-10 orbiter envelope. Lack of time prevented closing the loop by a final iteration in which orbiter size would also have been scaled for final systems optimization.

It is realized that the resulting performance and design characteristics of the two-stage system, reported in this document, do not represent a rigorously optimized system. However, this approach is justified, because

(1) In comparing the obtained design and performance characteristics with data quoted for current Phase B designs, the conclusion could be drawn that the assumed orbiter size is close to the optimum and that therefore final adjustments to the results of more rigorous analyses would have an insignificant effect on the results and conclusions of this study.

(2) The common aerodynamic shape and size of the stage-and-one-half and two-stage designs facilitated accomplishment of study objectives and vehicle comparisons and provided maximal validation for the obtained results, since the basis for extensive supporting studies, particularly with regard to weights, was maintained.

(3) The common aerodynamic configuration was a major factor for the validation of the favorable aerodynamic characteristics of the two-stage delta-body configuration since wind tunnel tests were performed for the stage-and-one-half configuration only.

(4) Uncertainty in the booster characteristics obtained by scaling rather than independent design are believed to have greater effect on system comparisons than small differences in vehicle size.

Constraining orbiter design by the characteristics of the main rocket orbiter engine specified by ICD 13M15000B for the Phase B program leads to the installation of two main rocket engines. With only one engine operating in the one-engine-out mode, the drop in thrust-to-weight with the two-stage configuration is sufficient to produce a noticeable performance loss for attaining the orbital capability specified for the abort maneuver. This is due, in comparison with the stage-and-one-half concept to the considerably reduced staging velocity. This loss must be accounted for in determining the propellant reserve in the orbiter. In addition, under this condition roll control capability must be provided by the ACP system, creating an additional design requirement on that system and further increasing the abort propellant reserve carried in the orbiter.

As will be shown later, this reserve, which equates with payload, can reach a considerable value. Based on these considerations, it appears that payload capability could be improved by reducing the thrust level of the orbiter engine sufficiently to permit installation of three engines. With three engines available for the post-separation injection phase, as with the stage-and-one-half system only two would be used under normal conditions and the third would be a standby providing undegraded performance capability in the one-engine-out condition. No abort propellant reserve would be required and the resulting saving in propellant weight can be expected to exceed the increase in installed engine weight.

The basic considerations leading to the determination of the propellant load in the orbiter for satisfying the payload weights specified for the three NASA missions were presented in Section 2.2.1-1 and apply equally to the two-stage system. However, for the two-stage configuration satisfaction of abort capability becomes the determining design requirement. For the 55 deg mission, which was critical with the stage-and-one-half system, the propellant requirement for abort is still covered by the propellant designated for the post-

insertion orbit maneuvers, since these are not considered required under the abort condition. For the south polar mission, however, and to a lesser degree for the due east mission also, additional propellant must be carried in the orbiter as a reserve. Consequently, the abort requirement for the south polar reference mission establishes the critical design condition for the two-stage orbiter.

On the basis of these considerations, and with the orbiter ascent tank volumes established by the assumed orbiter design, the orbit maneuver propellant tanks were sized to provide the required capacity to satisfy the critical design condition, and cross feeding capability between the orbit maneuver and main propellant systems was provided to permit use of additional propellant in the main engine system for the abort condition. For the due east and south polar missions, the orbiter tanks are filled to capacity, while for the 55 deg reference mission, orbit maneuver propellant was off-loaded corresponding to a 1500 ft/sec velocity capability. Consequently, the design capability is exceeded for the due east and 55 deg missions.

3.2.1.2 Configuration. The Two-Stage Baseline, fully reusable vehicle system selected by Lockheed, Model LS 400-7A, is shown in Fig. 3.2-1 in the launch configuration. It consists of an orbiter of modified delta planform configuration, combined with a booster whose characteristics are scaled from a baseline design defined by the 8th Mass Properties Status Report prepared by McDonnell-Douglas under the Space Shuttle Phase B program. The booster configuration shown in the figure displays the dimensions of the unaltered baseline design. The assumptions made in adjusting the baseline booster design to the baseline orbiter are presented in Section 3.2.2.

The orbiter vehicle has the same configuration and basic dimensions as the stage-and-one-half baseline orbiter discussed in Section 2.2. Except for characteristics determined by specific two-stage concept requirements, the

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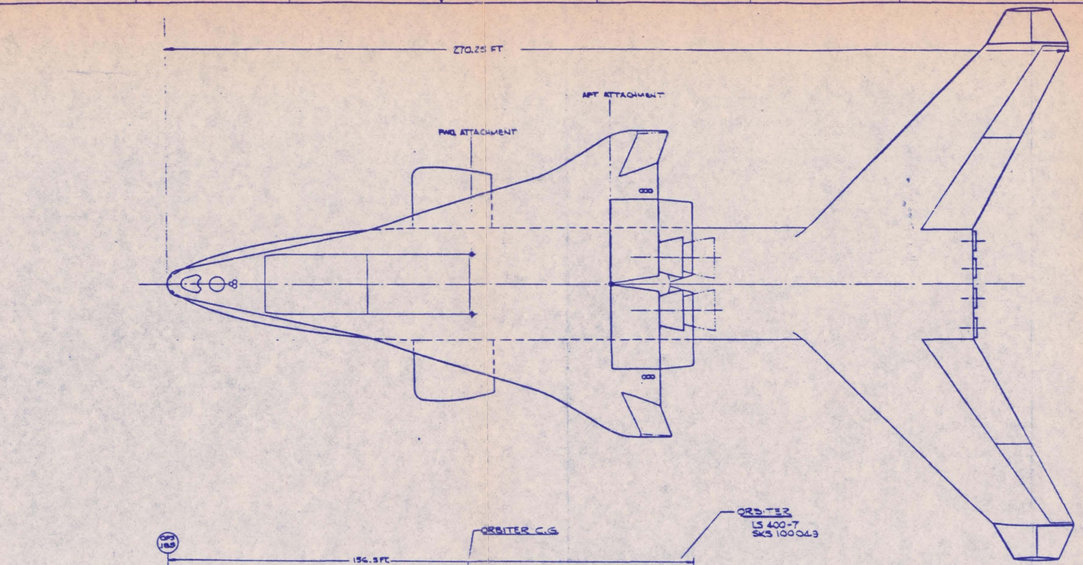

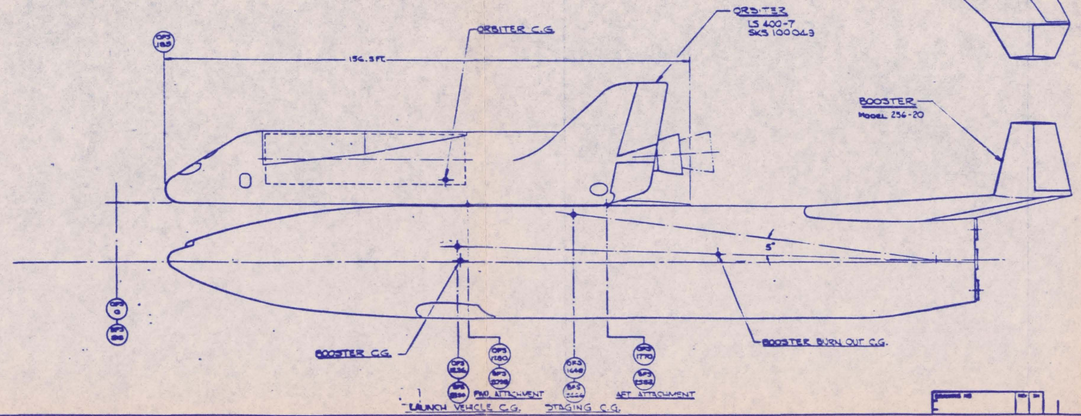

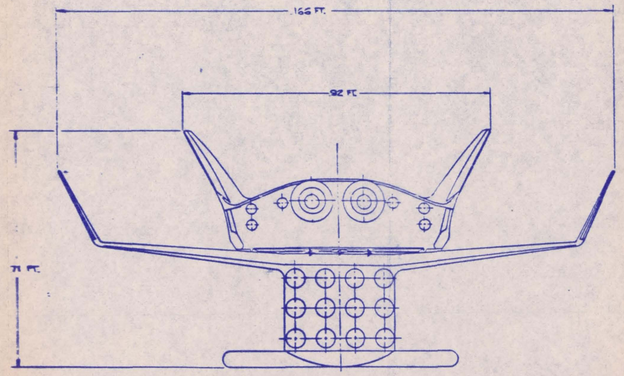

NOTE:
C.G. LOCATIONS ARE PRELIMINARY ESTIMATES.

Fig. 3.2-1

REVISIONS			
LIMITED CALENDAR LIFE	LIMITED OPERATING LIFE	SYMBOL	DESCRIPTION / DATE / APVD
		A	...LATED BOOSTER CONFIG. 3-2-77

REV HIST	CODE IDENT	PART OR IDENTIFYING NO.	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO.
PARTS LIST							

UNLESS OTHERWISE SPECIFIED DIMS ARE IN INCHES TOLERANCES ON FRACTIONS ± 1/16 DECIMALS .X ± 0.1 .XX ± .03 .XXX ± .010 ANGLES ± 1° 30'	DATE 3-10-77	LOCKHEED MISSILES & SPACE COMPANY GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION SUNNYVALE, CALIFORNIA	
	ENGR APVD	LAUNCH VEHICLE DELTA BODY ORBITER MODEL LS 400-7	
	MFG APVD		
	ENGRG APVD		
	CHKR APVD		
	MFG APVD		
NEXT ASSY USED ON APPLICATION	CONTR CCA 01-10	SIZE J	CODE IDENT DRAWING NO. SKH 100051 REV A
		SCALE NONE	SHEET 22

3-2-7

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internal arrangement and operation, as well as the functions assigned to and the designs of the subsystems are similar to those of the stage-and-one-half orbiter

In the following paragraphs, only those elements and characteristics of the two-stage system which are significantly different from the stage-and-one-half system, or which are peculiar to the delta-body configuration will be discussed.

3.2.1.3 Entry. The entry mode selected for providing the 1100 nm hypersonic crossrange capability is identical with that discussed for the stage-and-one-half configuration in Section 2.2.1.5. However, detail operational characteristics and resulting design requirements are changed, resulting from the reduced planform loading of the two-stage configuration.

3.2.1.4 Subsonic Flight. An airbreathing system can be installed for controlling the glide slope to a value compatible with instrument landing and for go-around capability. Since the stringent constraints of limited base area availability and cg balance capability affecting the stage-and-one-half vehicle design do not equally apply to the two-stage configuration, the jet engines are installed at the vehicle base. This arrangement provides distinct advantages with regard to operational characteristics, deletion of the complex engine deployment mechanism with its associated weight of approximately 1500 lb, availability of the space occupied by the stowed engines for propellant tankage, and reduction of a large heat shield penetration at the critical lower vehicle surface. However, with the installation at the base, the number of jet engines can be changed only in increments of two. For this reason, it was deemed acceptable to design the system with four installed jet engines, precluding the capability to satisfy the FAA engine-out climbing gradient requirement. Compliance with this specification would require the addition of two jet engines which was considered an undue penalty on the system.

With the assumed system, climb in the approach, engine-out condition can be made to 5000 ft altitude, but with less than 2.7 percent gradient. Climb in the landing configuration can be accomplished with a 3.2 gradient to 8000 ft altitude.

Landing speeds range from 122 knots up to 180 knots, depending on payload, cg location, and pilot technique. These values do not include ground effects which are expected to reduce touchdown speed by approximately 14 knots.

3.2.1.5 Ferry Operation. Ferry capability can be provided in the same manner as for the stage-and-one-half configuration by a ferry kit. However, for the two-stage orbiter inclusion of an additional jet engine will be required for all operational conditions. With this arrangement, and 65,000 lb fuel load, a range of approximately 300 nm is attainable which can be increased by in-flight refueling. Cruise altitude is over 12,000 ft or 5,500 ft with one engine out.

3.2.1.6 Abort. Intact abort through orbit is the primary mode for the critical engine-out condition at booster separation. The propellant reserves for performing this sequence with the reduced thrust capability of the remaining main engine are reflected in the quoted performance capabilities.

3.2.1.7 Aerodynamics and Stability. Diagrams showing the stability and performance characteristics for the two extreme aft and forward center-of-gravity locations and for appropriate trim conditions over the entire speed regime are presented in Figs. 3.2-2, 3.2-3, and 3.2-4. They demonstrate that the orbiter is aerodynamically stable and controllable under all conditions of operationally possible combinations of speed, center-of-gravity location, and flight attitude. The data shown are based on the results of extensive wind tunnel tests performed at Lockheed, Langley, and Ames on the LS 200-5 configuration.

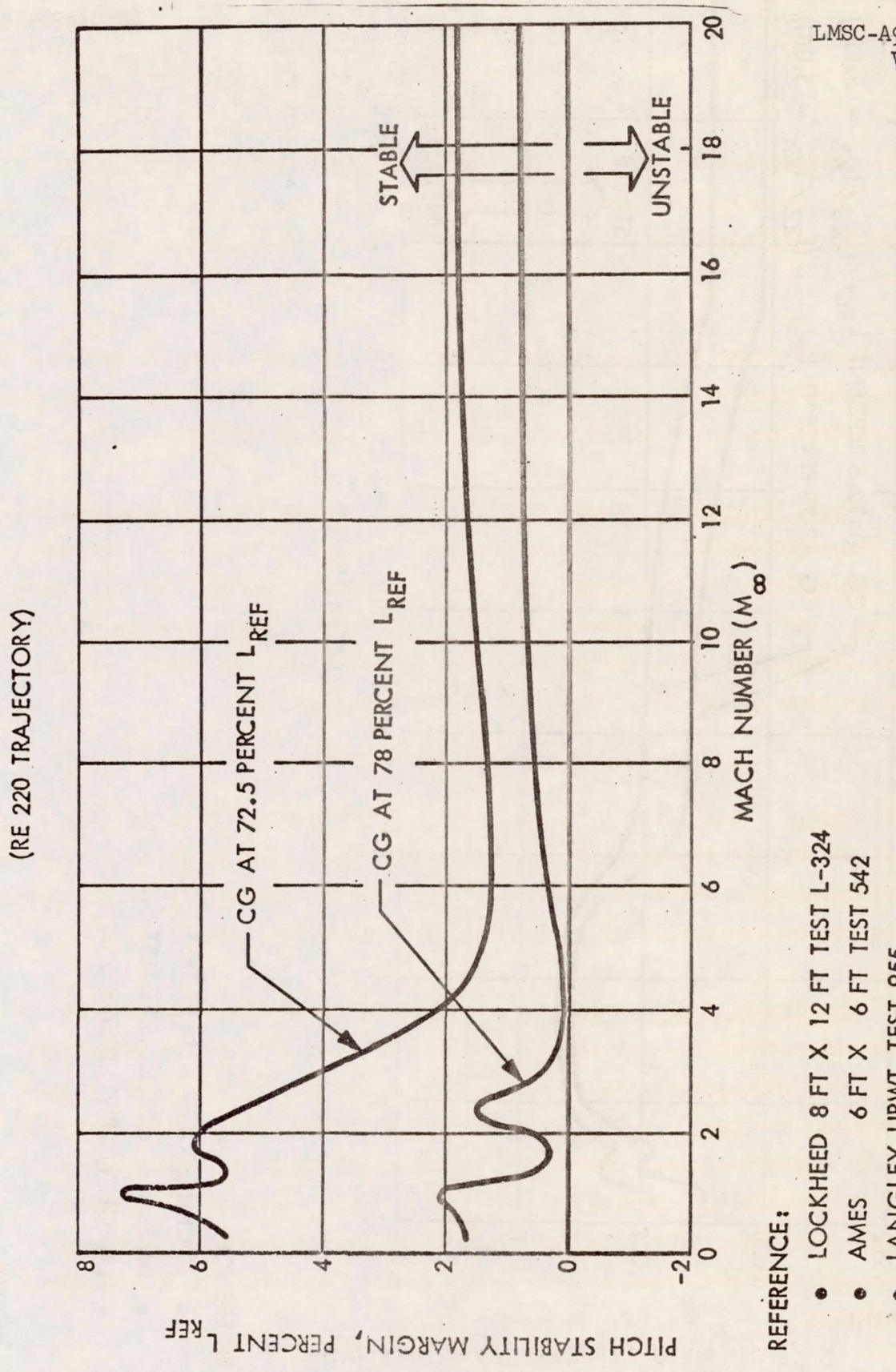

Fig. 3.2-2 2-Stage Delta-Body Longitudinal Stability Summary

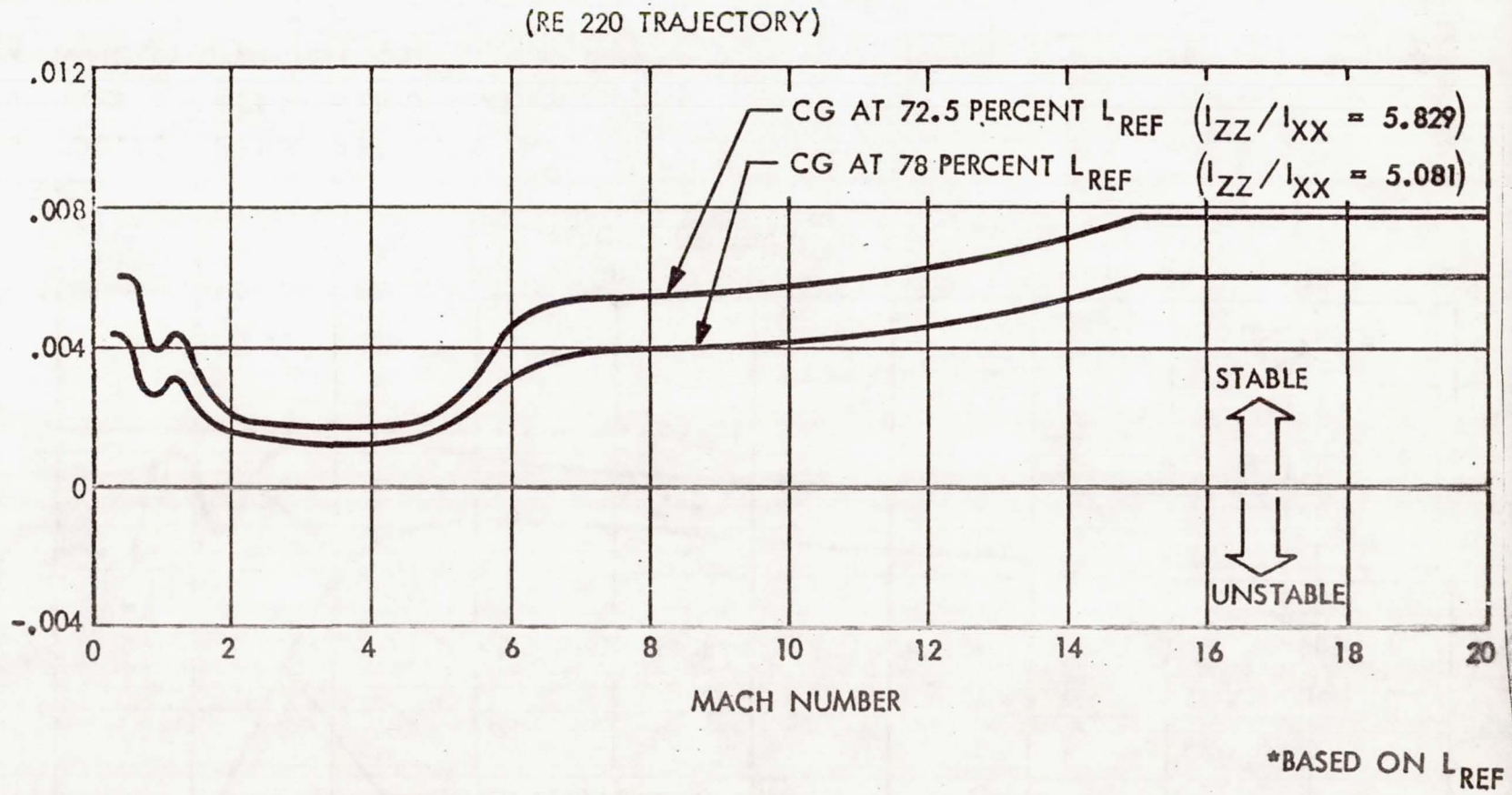

Fig. 3.2-3 2-Stage Delta-Body Directional Stability Summary

REFERENCE:

- LOCKHEED 8 FT X 12 FT TEST L-324
- AMES 6 FT X 6 FT TEST 542
- LANGLEY UPWT TEST 955

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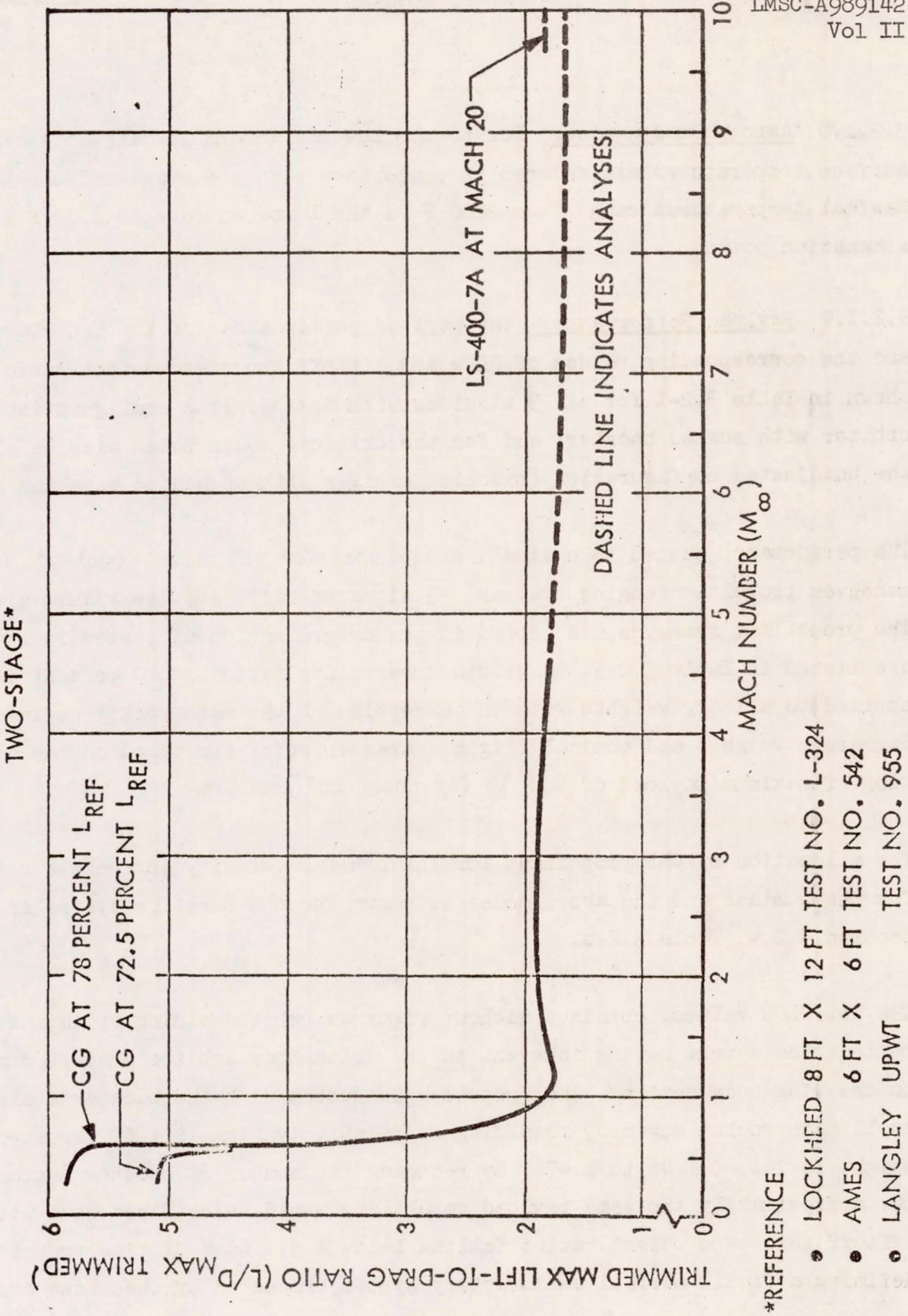

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Fig. 3.2-4 2-Stage Delta Body L/D Summary

3.2.1.8 Aerothermodynamics. Due to the lower planform loading, the orbiter surface temperatures are reduced in comparison with the stage-and-one-half. Maximal temperatures range from 2200°F on the lower surface to 2730°F at the stagnation point.

3.2.1.9 Payload Performance. The payload performance for the two-stage system, and the corresponding values of GLOW and liftoff thrust-to-weight ratio are shown in Table 3.2-1 for all 3 missions with the baseline configuration (baseline orbiter with scaled booster) and for the critical south polar mission also with the unadjusted configuration (baseline orbiter with unchanged baseline booster).

The performance quoted is nominal, except that for the main propulsion and orbit maneuver propulsion engine systems -3 sigma specific impulse values are used. The propellant reserves considered in the determination of system inert weight are listed in Table 3.2-5. A growth uncertainty factor of 10 percent is applied to all dry weights with the exception of the main rocket engines. Structure weights and nominal flight characteristics are based on the assumption of maximum payload of 40 K lb for entry and landing.

The allocation of OMP propellant for the orbital velocity increments required for the nominal and the abort modes is shown for the baseline system in Section 3.2.4, Table 3.2-8.

The low GLOW values, obtained without rigorous orbiter sizing optimization, reflect the weight saving inherent in the delta-body orbiter concept compared to the wing-body design. They represent a system with the booster scaled down to 11 main rocket engines, resulting in relatively high liftoff thrust-to-weight ratios of 1.46 to 1.47. By reducing the number of booster engines to 10, approximately the same payload capability would be retained, but with the liftoff thrust-to-weight ratios falling below 1.3. Lack of time prevented defining a final overall satisfactory system, bracketed by these two designs

Table 3.2-1

Payload Performance Characteristics

Mission	Design Due East	Reference South Polar		Reference 270 nm 55 Deg.Incl.
Payload Specified, Klb	65	40		25
Airbreather Engine Syst.	OUT	OUT		IN
Vehicle System	Baseline	Baseline	Unadjusted	Baseline
Booster	Scaled	Scaled	Baseline	Scaled
Payload, Klb	79.5	40.0	58.6	35.9
Liftoff,T/W	1.455	1.467	1.42	1.465
GLOW, Mlb	4,157	4,123	4,630	4,130

points. This would be obtained by adjusting, and also optimizing, orbiter size, and, if necessary, throttling of main booster engines, without significantly changing the performance potential indicated by the data of Table 3.2-1.

The vehicles for which these data are shown contain the required large propellant reserves for abort-through-orbit and are designed for the critical requirement established by the south polar mission. Consequently, excess payload capability is obtained for the two other missions. System weights for these two missions providing the design payloads by offloading the booster tanks were not determined because of the high thrust-to-weight values shown with fully loaded tanks. For the finally sized system with reduced liftoff thrust-to-weight ratio the adjustment to design payload capability by offloading the booster would be feasible for all missions.

3.2.1.10 System Characteristics Summary. The system characteristics are discussed briefly in Section 3.2.2 and 3.2.3 for the booster and for the orbiter. The most important quantitative system, design, dimensional, performance and mass property characteristics are summarized in tabular form in Section 3.2.4.

All established space shuttle performance and design requirements and constraints have been satisfied with the presented vehicle system. With this system, which uses the Lockheed-derived orbiter design in combination with a booster whose characteristics are derived directly from a recent configuration defined under the current Phase B program, GLOW values are obtained substantially below those currently quoted for systems composed entirely of vehicles developed under that program. This is caused by the use of an orbiter which, while smaller in every dimension, contains approximately 20,000 lb more propellant than comparable wing-body designs.

The results of this study confirm for the delta body the unique combination of high volumetric efficiency with a geometry particularly enhancing the design of lightweight structures, and which supplies the basis for the development of a configuration providing adequate performance, flight control and stability characteristics and temperature control capability over the entire flight regime.

3.2.2 Booster and Booster-Orbiter Interface Characteristics

Designing the booster for the two-stage system was not to be accomplished under this study. Consequently, the booster characteristics required for sizing and designing the orbiter and for the determination and evaluation of entire vehicle system performance were obtained by scaling pertinent booster characteristics of a baseline booster design to become representative of a correctly sized booster matching the baseline orbiter.

3.2.2.1 Baseline Booster. For the baseline booster, the McDonnell High Cross-range Canard Booster Model 256-20, designed under the Phase B Space Shuttle program and defined by the 8th Mass Properties Status Report was selected. The principal characteristics of this booster, as far as they affect orbiter and overall vehicle system design and performance, are summarized in Table 3.2-2.

In adjusting the baseline booster to the Lockheed two-stage vehicle system, the basic booster arrangement, structural design concept, and the general location of the orbiter attach points, determining the load path between the booster and orbiter, were maintained unchanged. Consequently, the major longitudinal and transverse interface loads are introduced at the forward attach point into the booster intertank structure; and only relatively low loads, and only in pitch direction through the aft attach point into the booster hydrogen tank. In detail, however, the design and function of the attach points were changed to suit the Lockheed orbiter design. However, no effort was spent on establishing a commensurate separation mode and resulting detail design requirements for the attach-separation subsystem. These assumptions are summarized in Table 3.2-3.

Table 3.2-2

MP-8 BOOSTER BASELINE CHARACTERISTICS

Weight, lb

• Booster Dry Weight**	520,409
Personnel	400
Cargo	0
Residuals	9,760
• Booster Inert Weight	530,569
Reserves	50,441
Inflight Losses	22,928
Ascent Propellant	3,064,000
Cruise Propellant	110,000
Man/ACS Propellant	1,222
• Booster Gross Weight	3,779,160

Area, ft^2

• Wing Area (Theo.)	= 6015.6
• Tail Area (Exp.)	= 876.0
• Canard Area (Theo.)	= 1660.0
• Body Area	= 24,104.0
• Tank Areas	= 16,766.0*
• TPS Area	= 37,231.0

Main Rocket Engines

Type:	Booster Engine ICD 13M15000B
No. of Engines	12

*Not including domes or intertank.
**Noseload affected structure = 4631 lb

Table 3.2-3

BOOSTER-ORBITER ATTACH POINT CHARACTERISTICS

Location	Forward		Aft	
Booster	MACDAC #20	LMSC LS 400-7A	MACDAC #20	LMSC LS 400-7A
Booster Structure Interface	Intertank Structure	Same	Intertank Structure	Same
Attach Points				
Number	1	2	3	1
Arrangement	Central	180 in. laterally spaced	One central Two 238 in. laterally spaced	Central
Loads Reacted	Omni-directional	Longitudinal, Lateral, and yaw couple by longitudinal force difference	Central: shear only Lateral: pitch only	Pitch only

In addition, the following constraints were assumed for the booster-orbiter interface:

$\alpha q_{max} = 2{,}800$ deg-lb/ft^2

relative separation velocity $\leq$ 11,000 ft/sec

The resulting combination of the baseline booster with the baseline orbiter is shown in Fig. 3.2-1. Included are the significant locations of the center-of-gravity of the composite vehicle system during the booster flight phase, and the resulting envelope for the booster main engine thrust vector. They indicate the compatibility of the selected combination with the gimbal capability of the booster engines.

3.2.2.2 Booster Scaling. The Baseline Two-Stage Configuration, Model LS 400-7A presented in this report consists, as discussed in the preceding section, of the combination of the Lockheed Two-Stage Baseline Orbiter Model LS 400-7A with a booster derived from the Baseline Booster by scaling its characteristics affecting vehicle system performance to values resulting in a vehicle system providing the specified mission performance characteristics.

Table 3.2-4 summarizes the basic rationale followed in scaling the baseline booster. In setting up the scaling laws, care was taken to insure that basic driving parameters most influential in determination of the weights were kept as close to the original booster design points as possible. These basic drivers were estimated to be.

(1) Initial thrust-to-weight ratio (T/W)

(2) Start-of-cruise wing-loading (W/S)

(3) Cruiseback engine performance

Item (1) was controlled by an option to delete main engines. This served to maintain a permissible band or tolerance effects on the maximum αq values for the booster, thereby permitting the use of constant fuselage unit weight values.

Item (2) was kept constant by ratioing the wing and tail areas to the cruise weight. This permitted the use of constant unit weight values for the TPS, and also permitted a simplified scaling law for cruiseback fuel requirements

by maintaining 10 constant cruiseback engines (the same as MP-8), so that total thrust and total S.F.C. were maintained in conjunction with item (2). Consequently cruiseback lift-to-drag ratios were kept reasonably constant, and the cruiseback fuel requirements could be expressed as a natural log function of the required range.

The remaining assumptions that were employed in the weights scaling rationale are defined in Table 3.2-4.

Table 3.2-4

Booster Weight Scaling Rationale

SYSTEM	SCALING FACTORS
WING AND TAIL	FIXED LB/FT^2. AREA SCALED TO KEEP WING LOADING CONSTANT.
BODY	THRUST STRUCTURE VARIES WITH THRUST, TANKAGE IS FIXED LB/LB ASCENT PROPELLANT. INTERSTAGE SCALED WITH ORBITER WEIGHT AND MISCELLANEOUS STRUCTURE IS CONSTANT VALUE.
THERMAL PROTECTION	FIXED LB/FT^2 FOR BODY, WING AND TAIL AND BASE AREAS. BODY AREA ADJUSTED WITH ASCENT PROPELLANT REQUIREMENT.
LANDING GEAR	FIXED PERCENTAGE OF LANDING WEIGHT.
ASCENT PROPULSION	CONSTANT PLUS FUNCTION OF NUMBER OF MAIN ENGINES.
CRUISE PROPELLANT	FIXED LB/JET ENGINE INSTL. TANKAGE SCALED WITH CRUISE PROPELLANT REQUIRED.
OTHER SUBSYSTEMS	FIXED WEIGHTS.
CONTINGENCY	CURRENT MAC/DAC PERCENT OF DRY WEIGHT (LESS ICD ENGINES)
PERSONNEL	FIXED WEIGHT.
RESIDUALS, RESERVES, LOSSES	CONSTANTS PLUS FIXED PERCENTAGES OF ASCENT AND CRUISE PROPELLANT WEIGHTS.
ASCENT AND CRUISE PRPL	COMPUTED, PER V_{STAGE} AND CRUISE RANGE.
MANEUVER/ACS PRPL	FIXED WEIGHT.

3.2.3 Orbiter Design Characteristics

3.2.3.1 General Arrangement and Inboard Profile. The general arrangement of the Baseline Orbiter, Model LS 400-7A, is shown in Fig. 3.2-5. The detailed internal arrangement of the orbiter is very similar to the inboard profile shown for the LS 400-7 configuration in Section 3.4 (Fig. 3.4-6), the only difference being that the payload deployment mechanism was moved from the rear to the front of the payload bay.

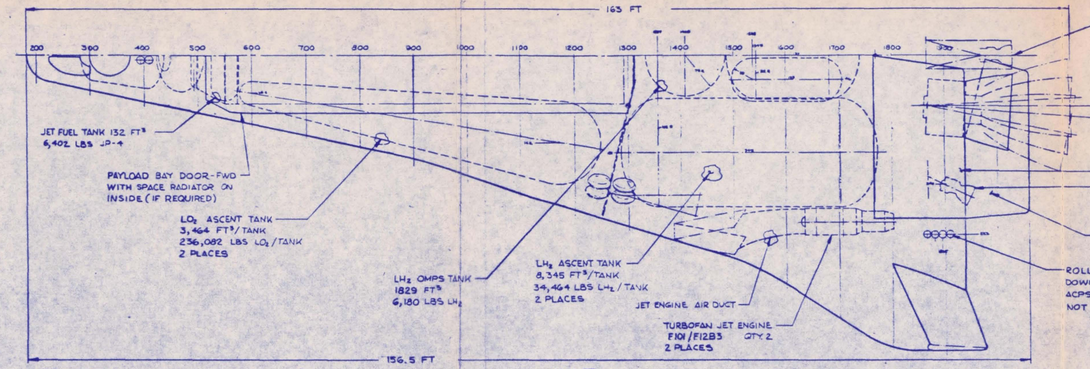
3.2.3.2 Configuration and Aeroshape. Size and aerodynamic configuration, with the exception of small changes at the aft body and vehicle base required for the accommodation of the reduced number of main rocket engines, are identical to the stage-and-one-half LS 200-10 configuration presented in Section 2.2. Consequently, the delta body two-stage orbiter provides the same satisfactory aerodynamic and flight performance characteristics over the entire flight regime, in combination with high volumetric and structural design efficiency as they were discussed for the stage-and-one-half configuration.

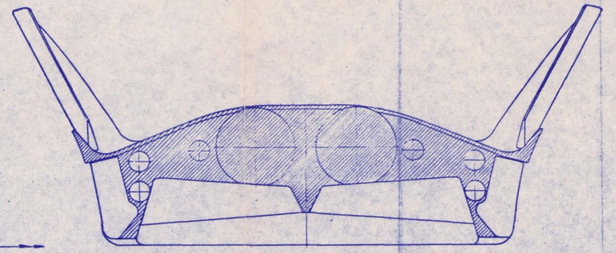
The resulting vehicle configuration is considered sized nearly optimal for a vehicle system with minimum GLOW satisfying all specified design and reference mission requirements. The packaging of the substantially (130 percent) larger propellant load in the envelope of the stage-and-one-half vehicle was achieved by moving the remaining two main rocket engines approximately 20 ft aft, and by moving the jet engines from their central location to the vehicle base. These changes were made possible without impairing the vehicle cg balance by the reduction in engine number from 9 with the stage-and-one-half configuration to 2 with the two-stage configuration. The resulting gain in length inside the vehicle in a region of maximal cross-section provided the required space for the installation of the additional tank volume. Accommodation of a total impulse propellant load of 584 klb in an orbiter vehicle of 156.5 ft total

length, without the use of complex tank configurations, clearly indicates the high packaging efficiency of the selected delta-body configuration. As with the stage-and-one-half orbiter, ample space is left for an expansion of crew and/or passenger accommodations and for an increase in equipment installation volume.

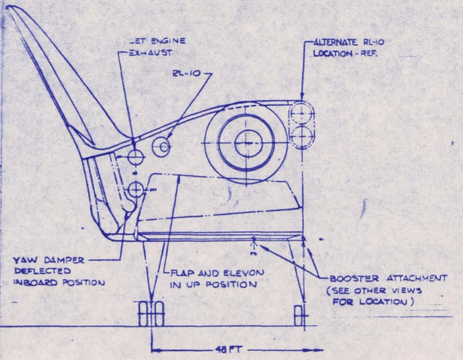
3.2.3.3 Structural Arrangement. Prior to staging, the fuselage forward of Station 1270 is designed by the maximum αq condition. For the section aft of this station, the maximum acceleration load becomes the design load condition. This load generates a large bending moment inside the orbiter, resulting from misalignment between the longitudinal accelerating force introduced at the forward booster attach point and the resisting inertia of the highly concentrated mass of the LO_2 carried in the ascent tanks. After separation, the maximum internal load is generated by the longitudinal acceleration of this oxygen mass by the orbiter main rocket engines.

For these load conditions, a highly weight and cost efficient structural design was developed. It combines the use of the thrust structure as a primary structural element with a design concept, in which the ascent tanks are used for carrying a large portion of the vehicle loads without making them an integrated part of the primary structure. The twin tank systems are assembled as two continuous beams, which are aft rigidly attached to the thrust structure, and forward at two support points to the fuselage by attachments designed to transfer only lateral loads and not to restrict longitudinal displacements for the accommodation of thermal contractions and expansions. With this arrangement, a direct longitudinal load path from the thrust of the orbiter rocket engines to the resisting mass of the LO_2 is created via the thrust structure, the hydrogen tank, and the intertank structure. The reaction to the bending moment during boosted acceleration is shared by tanks and fuselage structure without requiring integration with the primary structure. The longitudinal load resulting from the acceleration of the orbiter mass by the booster is transferred from the forward booster attach point to the thrust structure in tension and shear along the lower fuselage surface.





BASE = CRUISE CONDITION



AREA:
REF. PLANFORM : 4,878 FT²
TOTAL PLANFORM : 6,846 FT²
BASE : 1220 FT²

REVISIONS			
LTR	DESCRIPTION	DATE	APVD
A	CHANGED L3-400-7 TO L3-400-7A BY MOVING DEPLOYMENT MECH TO FWD. END OF P/L BAY	4-15-71	
B	JET ENGINE WAS JTF-22A FWD. BOOSTER ATTACHMENT WAS F.S. 970, - RL10 WAS ES. 1085 AND BL. 181 ORBIT PROPELLANT WAS LO₂ 472 FT³, 25,088 LBS LH₂ 1806 FT³, 6,077 LBS ADDED ALT. RL-10 (NOT L ADDED MAIN ENG. POS-CANT-4	4-29-71	MK

LIMITED CALENDAR LIFE | LIMITED OPERATING LIFE

Fig. 3.2-5

QTY REQD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO

PARTS LIST

INTERPRET DWG PER	UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES TOLERANCES ON FRACTIONS ± 1/16 DECIMALS X ± 0.1 XX ± 0.02 XXX ± .010 ANGLES ± 2 DEG
NEXT ASSY	USED ON APPLICATION
CONTR	CCA-01-10

DATE: MAR 13 '71
ENGR: W. Simon
APVD:
APVD:
ENGNG:
CHKD:
APVD:
APVD:

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

GENERAL ARRANGEMENT
DELTA BODY ORBITER
MODEL L3-400-7A

SIZE: J | CODE IDENT: 06397 | DRAWING NO: SKS 100049 | REV: B
SCALE: | SHEET:

DRAWING NO | REV | SH

3.2-25

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3.2.3.3 Thermal Protection System. The thermal protection system selected for the two-stage orbiter is basically identical with the one which was described in Section 2.2.3.7 for the stage-and-one-half configuration since similar thermal environments are experienced. Because entry-heating duration is reduced in comparison with the heavier stage-and-one-half orbiter, the insulation thicknesses can be reduced by about 8 percent. On the other hand, since with the two-stage configuration the lower trim flap is not retracted during ascent, its upper surface requires thermal protection from plume heating. This is accomplished by bonding 0.5 in. of LI-1500 to the titanium structure.

In establishing the orbiter vehicle system, the same attention as with the stage-and-one-half configuration was given to reducing to a minimum the number of penetrations in the heat shield, particularly in the lower surfaces exposed to high temperatures. On the lower surfaces, besides the inevitable disruptions caused by the hinges of aerodynamic surfaces, nine major penetrations are required: three for the landing gear, two for the air inlets for the air-breathing engines, and four for ACS thruster clusters. The first five penetrations, however, are considered not critical, since their protecting doors need not to be opened prior to reentry. Consequently, they can be sealed, and reliably checked out prior to launch by ground operations.

3.2.3.4 Tankage and Propellant Systems. The propellants for ascent and orbital maneuvers are stored in different tankage systems with only the orbital system designed for long-time storage. All tanks are non-integral with the primary structure, of simple geometric configuration and use external insulation, providing for low-cost development, manufacture and maintenance, and for efficient quality control and checkout capability. The ascent system uses two long, conical LO_2 tanks arranged along, and partially under the cargo bay, and three cylindrical LH_2 tanks installed in the large vehicle section between the cargo bay and the main engine thrust structure. The propellants for the orbit

maneuver and for the attitude control propulsion systems are contained in two cylindrical tanks which are also installed in the aft vehicle section.

The ascent and orbital propellant tank systems are interconnected, permitting propellant transfer for system flexibility for varying mission conditions and requirements. With the ascent propellant feed system, the oxygen propellant is fed directly to the individual rocket engines, and a cross-over capability close to the engines promotes maintenance of equal propellant drain from both tanks and enables each engine to use all propellant stored. The two outer LH_2 tanks are fed through the center tank into the feed lines. Gaseous hydrogen is prevented from being ingested in the engine by use of standpipe discharge into the center tank.

The main propulsion pressurization system uses gaseous oxygen and hydrogen bled from the main engines for pressurization of the oxygen and hydrogen tanks, respectively. Pressurant supplied from the attitude control propulsion system accumulators is used for engine start. For the orbit maneuver propulsion system, helium is used as pressurant on the basis of an appreciable weight saving in comparison to use of propellant gases bled from the RL 10 engine.

3.2.3.5 Main Propulsion and Orbit Maneuvers Rocket Engines. The main propulsion system uses two rocket engines each providing 632 lb thrust at vacuum with a 150:1 expansion ratio obtained with a nozzle equipped with a retractable extension skirt. They are identical with the orbiter engines specified by ICD 13M15000B. The engines are installed side-by-side at the vehicle base. Each engine is gimballed ± 7°deg in yaw and ± 5½ deg in pitch, giving full attitude control capability in the normal operating mode and also providing adequate pitch and yaw control capability when aligned with the vehicle cg in the one-engine-out mode.

The orbit maneuver propulsion system employs two RL10A-3-3A rocket engines, providing in excess of 90 percent of the required ΔV capability. The engines are installed on the thrust structure at a sufficient distance from the main engines to avoid plume impingement. The other characteristics and the function of this system are the same as with the stage-and-one-half discussed in Section 2.2.3.9.

3.2.3.6 Attitude Control Propulsion System. The attitude control system, using 40 identical thrusters, is similar to that used with the stage-and-one-half orbiter, except that it must also provide for roll control capability when only one main engine is operating in the main propulsion system one-engine-out and subsequent orbiter abort modes.

3.2.3.7 Airbreathing Propulsion System. Four GE F101/F12B3 engines are installed at the vehicle base. They use JP-4 fuel, stored in a forward-located tank to counteract excessive aft cg location during hypersonic flight. With this arrangement, the vehicle cg is not changed noticeably when the airbreathing system is removed. Two air intakes for this system are located one each at the outer surfaces of the vehicle fins, resulting in minimal flow disturbance around the vehicle surface and maximal intake and duct efficiencies.

3.2.3.8 Landing Gear and Lower Flap Assembly. The tricycle landing gear is of a design similar to that used with the stage-and-one-half orbiter. The lower flap is installed without provision for retraction, since the farther aft location and longer nozzle of the main engines removes the plume impingement interference encountered with the stage-and-one-half design.

3.2.3.9 Other Subsystems. The other subsystems use the same general and detail design concepts as with the stage-and-one-half configuration with modifications as required for the different design condition. A detailed description of the differences in the Avionics Systems requirements and design approaches between the two-stage and stage-and-one-half systems is presented in Section 4.8 of this report.

3.2.4 System Design and Performance Characteristics Summary Tables

Tables summarizing the design and performance characteristics of the Lockheed two-stage Baseline Model LS 400-7A vehicle system are presented in this section.

A summary of system design, performance, aerodynamic, and subsystem characteristics is given in Table 3.2-5.

The data demonstrating and defining the capability of the system to satisfy the performance requirements of the three Level I missions are presented in Table 3.2-6 for the baseline system and in Table 3.2-7 for the unmodified system using the baseline booster system.

The breakdown of propellant contained in the orbit maneuver tank system in relation to the incremental velocity requirements for maneuvers under nominal and abort flight conditions are presented for the baseline system and for the three missions in Table 3.2-8.

The major dimensional data of the vehicle system are summarized in Table 3.2-9.

A weight summary is presented in Table 3.2-10.

Table 3.2-5

PRINCIPAL VEHICLE SYSTEM CHARACTERISTICS OF TWO-STAGE
BASELINE SYSTEM LS 400-7A

SYSTEM SUMMARY

Concept	Two-Stage Reusable Booster Reusable Orbiter
Configuration	Orbiter Modified Delta Planform, Lifting Body Booster Scaled from Baseline Booster McDonnell-Douglas Canard High Crossrange 256-20 Ref. Eighth Mass Properties Status Report, dated 3 March 1971
Crossrange	1100 nm hypersonic crossrange capability
Performance	Satisfies with a Gross Liftoff Weight of $4.12 \leq GLOW \leq 4.16$ Mlb all Level I Mission Requirements Go-around capability with airbreathing propulsion system installed Orbiter trimmable and stable under all flight conditions Ferry capability with addition of one jet engine and fuel tanks, and possibly aerodynamic fairings in kit form.
Materials	Orbiter Primary Structures: Fuselage aluminum; aerodynamic surfaces, thrust structure and cargo bay door, titanium

Table 3.2-5 (Cont'd)

DIMENSIONS	Orbiter	Booster	Composite
Total Length, ft	163	270.25(1)	270.25(1)
Span, ft	92	166(1)	166(1)
Height, ft			
– Gear Up	36.75		71(1)
– Gear Down	50.00		
Planform Area, ft^2	6,846		
WEIGHTS, LB			
South Polar Reference Mission Baseline Vehicle Configuration (with scaled 11 eng. Booster)			
Dry	187,929	476,766	664,695
Ascent Propellant			
Impulse	546,439	2,643,026	3,189,465
Loaded	554,890	2,665,328	3,320,218
Orbit Maneuver Propellant(2)			
Impulse	36,617	NA	36,617
Loaded	37,847	NA	37,847
Attitude Control Propellant(3)			
Impulse	1,231	1,222	2,453
Loaded	1,757	1,450	3,207
Cruise Propellant			
Impulse	0	90,562	90,562
Loaded(4)	0	145,707	145,707
Cargo	40,000	NA	40,000

(1) Baseline Booster not scaled
(2) Including ACPS propellant for generation of 142 ft/sec-orbit ΔV
(3) Excluding ACPS propellant for generation of 142 ft/sec-orbit ΔV
(4) Including 39,000 lb of cruise fuel dumped before landing

Table 3.2-5 (Cont'd)

WEIGHTS, LB (Cont'd)	Orbiter	Booster	Composite
Landing	232,485	498,411	
Cruise		629,098	
Entry	233,565		
Staging	831,011		
Liftoff	831,093	3,292,248	4,123,341
Propellant Fraction λ' Dry/Wet(4)	.712/.701	.810/.803	
Thrust/Weight Ratio			
Liftoff	NA	NA	1.467
Maximum Ascent	3.0	3.0	3.00
At Staging	0	0	0
After Staging			
Initial	1.52	0	NA
Final	3.00	0	NA

(4) Propellant fractions are defined:

For the Orbiter

$$\lambda' \text{ Dry} = \frac{\text{Total loaded(ascent + OMP) propellant weight}}{\text{Liftoff less payload weight}}$$

$$\lambda' \text{ Wet} = \frac{\text{Total impulse(ascent + OMP) propellant weight}}{\text{Post separation less payload weight}}$$

For the Booster

$$\lambda' \text{ Dry} = \frac{\text{Total loaded (incl. holddown) propellant weight}}{\text{Liftoff + holddown propellant weight}}$$

$$\lambda' \text{ Wet} = \frac{\text{Total impulse propellant (without holddown) weight}}{\text{Liftoff weight}}$$

Table 3.2-5 (Cont'd)

	Orbiter	Booster	Composite
CG LOCATION			
Orbiter Fuselage Sta., in.			
Ascent 65 Klb Payload IN/OUT, at			
Liftoff		1248	1236
Prestaging		2170	1648
Orbit Injection			
Reentry			
40 Klb Payload IN, ABES OUT	1,271		
Payload OUT, ABES Eng. and Fuel IN	1,337		
Payload OUT, ABES OUT	1,346		
Landing			
40 Klb Payload IN, ABES OUT	1,271		
Payload OUT, ABES Eng. IN ABES Fuel OUT	1,365		
PLANFORM LOADING			
Maximum Loading Condition, lb/ft^2			
Reentry	40.0		NA
Landing	40.0		NA

RESERVES

Main Propulsion	FPR for 1 percent of ideal velocity capability to injection into reference insertion orbit
Orbit Maneuver Propulsion	Included in 1500 ft/sec ΔV requirement
Attitude Control Propulsion	10 percent
Airbreathing Propulsion	15 percent
Electric Power Reactants	20 percent

Table 3.2-5 (Cont'd)

RESERVES (Cont'd)	
Environmental Control Atmosphere	One day supply
APU Reactants	10 percent
Growth/Uncertainty	
Factor	Ten (10) percent of dry weight except main rocket engines

FLIGHT PERFORMANCE CHARACTERISTICS

270 nm 55 deg inclination Reference Mission

Ascent

Liftoff Thrust/Weight Ratio	
With 12 Main Booster (Unadjusted Baseline) Engines	
q_{max}, lb/ft^2/at time, sec	602/66
αq_{max}, deg-lb/ft^2/at time, sec	2800 (assumed for load analysis)
βq_{max}, deg-lb/ft^2/at time, sec	

Separation

Time, sec	201
Velocity ideal/relative, ft/sec	11,000/
Thrust/Weight Ratio	0
q, lb/ft^2	5

Injection

Time, sec	405

Entry

Crossrange, nm	Nominal 1100 from hypersonic maneuver
Footprint, nm	Probably exceeding 2200 wide, 2000 long
Angle-of-Attack, deg	32
Duration from 400 kft altitude to Mach = 1, min	28.4

Table 3.2-5 (Cont'd)

FLIGHT PERFORMANCE CHARACTERISTICS (Cont'd)

Maximum Load Factor, g	1.65
Max q, lb/ft^2	385
Landing	
Capability	With or without ABES installed
Speed, kts	
At $\alpha_{L/D_{max}}$/22 deg (tail scrape)	
40 klb cargo IN, ABE OUT	180/152
Cargo OUT, ABE IN	154/122
Ground Effect	No included, will reduce by about 14 kts
Powered Subsonic Flight	
Climb Gradient	Less than 2.7 percent to 5000 ft in approach configuration (gear-up, engine-out)
	3.2 percent to 8000 ft in landing configuration (gear down)
Cruise Altitude, ft	
All engines	Over 12,000
Engine Out	5,500
Go-Around Capability	Standard day from 200 ft altitude to 2000 ft altitude for return cruise to outer marker
Ferry Capability	Addition of kits providing one additional cruise engine, JP-4 tank for approximately 58,000 lb fuel in cargo bay and possibly some aerodynamic fairings for main rocket engines provides approximately 300 nm range. Larger ranges by in-flight refueling.
Abort Capability	Intact abort to orbit from liftoff to normal injection. Capability included in quoted payload performance.

Table 3.2-5 (Cont'd)

AERODYNAMICS	
$\alpha_{L/D_{max}}$, Nominal, deg	
Entry/Landing	20/15
L/D_{max}, Hypersonic/Subsonic	1.87(1)/5.10 to 5.85
Static Stability	
Longitudinal, Directional, Lateral	Neutral or better at all operating conditions
Trim and Control Authority	Adequate for all operation conditions
CG Location Capability	Adequate for all combinations Cargo IN/OUT and ABE IN/OUT
Transition Maneuver	Not required for high crossrange Stable, controllable transition at low hypersonic speeds if required for low crossranges
Handling Qualities	MIL SPEC 8795 B, Class 3, Level I from M 2.5 to landing. All conditions acceptable unaugmented except lateral/directional dutch roll mode (augmentation required)

NOTE:

(1) Based on hypersonic arbitrary body computer program, validity of which confirmed by wind tunnel data on LS 200-5 orbiter model.

Table 3.2-5 (Cont'd)

THERMAL PROTECTION SYSTEM

Orbiter

Type — Passive, fully reusable except for flame curtains used at vehicle base. Insulation designed to limit temperature to 300 deg on aluminum and to 600 deg on titanium substructures.

Location	Temperature Range (^{o}F)	Concept and Material
Body	≤1000	Titanium Heat Shield Fibrous insulation
Body	1000-2200	LI-1500, Titanium subpanel
Fins, Control Surfaces	1000-2200	LI-1500, Titanium structure
Nose Cap	≤ 2780	Coated Tantalum Heat Shield Silicon Carbide Foam
Base	2200	LI-1500 on Titanium Structure or Support – Flexible flame curtains on gimballed engines

PROPULSION

Location	Orbiter	Booster(5)
Main Propulsion System		
Propellant - Type	LO_2-LH_2	Same
O/F Ratio	6:1	Same

(5) Booster scaled from baseline for meeting critical South Polar Mission requirement.

Table 3.2-5

PROPULSION (Cont'd)

	Orbiter	Booster(5)
Engine		
Type	ICD 13M15000B	Same
No. - Installed	2	11
- Operating Lifeoff	0	11
- Operating at Staging	0	0
- Operating After Staging	2	0
Expansion Ratio	150:1	35:1
Thrust, Klb, (Sea Vac, sec)	632	550/604
Throttling Capability	50 percent	Same
I_{sp} Minimum Guaranteed, SL/Vac, sec	NA/456	397/436
Gimbal Capability, deg	± 7 pitch ±5 1/2 yaw	± 10
Attitude Control Propulsion System		
Propellant - Type	High pressure GO_2 and GH_2	
O/F Ratio (System)	3.3:1	
I_{sp}, sec		
- System	352 to 362	
- Thruster	420	
Minimum Impulse Bit, lb-sec		
- Thruster	50	
- Igniter	5	
Liquid-Gas Conversion Units		
- Number	3	
- Gas Pressure, psia	500 to 2000	
Thrusters		
Number	40	
Chamber pressure, psia	250	
Thrust, lb	1500	

Table 3.2-5

PROPULSION (Cont'd)

	Orbiter	Booster(5)
Thrust Modulation		
- Orbit Maneuvers	Thruster pulsing	
- Deadband Control	Igniter pulsing	
Sustained Operation Capability With Two Conversion Units Operating	4 Thrusters at 100 percent thrust	
Total Impulse From Stored Gas, lb-sec	87,500	
Orbit Maneuvering Propulsion System		
Propellant – Type	LO_2 and LH_2	
O/F Ratio	5:1	
Engine		
Type	RL10A-3-3A	
Number		
- Installed	2	
- Operating	1	
- Reserve	1	
Thrust (VAC), lb	15,000	
Gimbal Capability, deg	± 4	
I_{sp}, Minimum Guaranteed, (VAC), sec	439	
Flight reserve	allowance included in Δ V capability requirement	
Airbreathing Propulsion System		
Fuel - Type Orbiter and Booster	JP-4	
Engine - Type Orbiter	Low bypass ratio turbofan GE F101/FRB3/JTF 22 A-4	
- Number, Orbiter/Booster	4/11	
- Installation, Orbiter	Vehicle base	
- Thrust Rating (Sea Level, lb)	Classified	

Table 3.2-5 (Cont'd)

Ferry Propulsion System, Orbiter	
Fuel - Type	JP-4
- Weight, lb	Approximately 58,000
Type	Additional engine and tank and possibly aerodynamic fairings on main rocket engines installed in kit-form.
Engine - Type	TBD
- Number	1
- Installation	On pylon mounted in cargo bay (tentative)
Tankage - Installation	In cargo bay
- Capacity, lb	Approximately 58,000 lb
Orbiter Propellant Feed System	Orbiter ascent propellant system separated from orbital propellant system.
	ACPS, OMPS, and APU propellants contained in orbital tankage
PRIMARY STRUCTURAL MATERIALS	
Orbiter Fuselage and Tanks	Aluminum
Orbiter Thrust Structure, Aerodynamic Surfaces and Cargo Bay Door	Titanium

Table 3.2-5 (Cont'd)

SYSTEM SENSITIVITIES

Gross Liftoff Weight

Booster Sizing (= Scaling) only
South Polar Reference Mission, Abort Capability

$$\frac{\partial (GLOW)}{\partial (Payload)} = 41.7 \text{ (payload delivered and returned)}$$

$$\frac{\partial (GLOW)}{\partial (\Delta V_{orbital})} = 640 \text{ lb/ft-sec}$$

$$\frac{\partial (GLOW)}{\partial (Inert_{orbiter})} = 41.7 \text{ (with secondary effects)}$$

$$\frac{\partial (GLOW)}{\partial (Inert_{booster})} = 9.39 \text{ (with secondary effects).}$$

$$\frac{\partial (GLOW)}{\partial (Propellant_{orbiter})} = -2.79 \text{ (including tank penalty)}$$

$$\frac{\partial (GLOW)}{\partial (Isp_{booster})} = -19{,}200 \text{ lbm}^2/\text{lbf-sec}$$

$$\frac{\partial (GLOW)}{\partial (Isp_{orbiter})} = -24{,}500 \text{ lbm}^2/\text{lbf-sec}$$

$$\frac{\partial (GLOW)}{\partial (Thrust_{booster})} = -0.260 \text{ lbm/lbf including engine penalty}$$
$$-0.401 \text{ without engine penalty}$$

Table 3.2-5 (Cont'd)

Payload

	Due East[1] Design Mission ABES OUT	S. Polar[1] Ref. Mission ABES OUT	270 nm 55 deg Ref. Mission ABES IN
$\frac{\partial (\text{Payload})}{\partial (\Delta V_{orbital})}$, lb/ft/sec	-19.7	-23.2	-22.1
$\frac{\partial (\text{Payload})}{\partial (\text{Inert}_{orbiter})}$	-1	-1	-1
$\frac{\partial (\text{Payload})}{\partial (\text{Inert}_{booster})}$	-0.170	-0.196	-0.173
$\frac{\partial (\text{Payload})}{\partial (\text{Propellant}_{orbiter})}$	0.185	0.204	0.230
$\frac{\partial (\text{Payload})}{\partial (I_{sp_{orbiter}})}$, $lbm^2/lbf\text{-sec}$	765	829	748
$\frac{\partial (\text{Payload})}{\partial (I_{sp_{booster}})}$, lbm^2/lbf	632	741	664
$\frac{\partial (\text{Payload})^{(2)}}{\partial (\text{Thrust}_{booster})}$, lbm/lbf	0.0130	0.0143	0.0104
$\frac{\partial (\text{Payload})^{(2)}}{\partial (\text{Thrust}_{orbiter})}$, lbm/lbf	0.0578	0.0677	0.0103

(1) Abort capability.
(2) Without engine weight penalty.

Table 3.2-6

TWO-STAGE BASELINE MISSION PERFORMANCE CHARACTERISTICS
CONFIGURATION: TWO-STAGE BASELINE MODEL LS 400-7A
BOOSTER: SCALED BASELINE BOOSTER

Mission	Design Mission Due East	Reference Mission South Polar	Reference Mission 270 nm, 55 deg Inc.
Airbreather Engine System	Out	Out	In
ORBITER			
Impulse Propellant, K lb			
Ascent	546.5	547.0	546.4
Orbit Maneuver (1)	36.7	36.7	29.4
Cruise	0	0	5.4
Flight Reserve Prop't Ascent, K lb	5.9	5.4	6.0
Dry Weight, K lb	187.9	187.9	206.1
Ignition Weight, K lb	871.2	831.2	843.0
BOOSTER			
Impulse Propellant, K lb	2,643	2,643	2,643
Liftoff Weight, K lb	3,287.	3,292	3,289
Number Main Engines	11	11	11
LAUNCH VEHICLE			
Staging Velocity, ideal, ft/sec	14,300	14,430	14,400
Liftoff Thrust/Wt	1,455	1,467	1,465
GLOW, K lb	4,157	4,123	4,130
PAYLOAD, K LB			
Design	65.0	40	25
Calculated	79.5	40	35.9
ON-ORBIT SYSTEM ΔV, FT/SEC	1,359	1,605	1,500
LANDING WEIGHT W/O PAYLOAD, K LB	192.5	192.5	211.7

Note: (1) Detail Breakdown in Table 3.2-8.

Table 3.2-7

TWO-STAGE UNADJUSTED SYSTEM MISSION PERFORMANCE CHARACTERISTICS

CONFIGURATION ORBITER: TWO-STAGE BASELINE MODEL LS 400-7A
BOOSTER: UNCHANGED BASELINE BOOSTER

Mission	Design Mission Due East	Reference Mission South Polar	Reference Mission 270 nm, 55 deg Inc.
Airbreather Engine System	Out	Out	In
ORBITER			
Impulse Propellant, K lb			
Ascent		546.6	
Orbit Maneuver(1)		36.7	
Cruise		0	
Flight Reserve Prop't Ascent, K lb		5.8	
Dry Weight, K lb		187.9	
Ignition Weight, K lb		831.2	
BOOSTER			
Impulse Propellant, K lb	3,064.0	3,064.0	3,064.0
Liftoff Weight, K lb	3,779.2	3,779.2	3,779.2
Number Main Engines	12	12	12
LAUNCH VEHICLE			
Staging Velocity, ft/sec		32,080	
Liftoff Thrust/Wt	1.42	1.42	1.42
GLOW, K lb	4,630	4,630	4,630
PAYLOAD, K LB			
Design	65	40	25
Calculated		58.6	
ON-ORBIT ΔV, FT/SEC(1)		1,605	1,500
LANDING WEIGHT W/O PAYLOAD, K LB		192.5	

Note: (1) Related to propellant contained in OMP tank system.

Table 3.2-8 Δ V CAPABILITY AND PROPELLANT ALLOCATION FROM OMP TANK SYSTEM

2-Stage Baseline Configuration Model LS 400-7A

Mission	Design Due East							
Flight Condition	Normal				Abort			
Propellant and ΔV Increment, lb and ft/sec	Δ W_p LH_2	LO_2	$LH_2 + LO_2$	Δ V	Δ W_p LH_2	LO_2	$LH_2 + LO_2$	Δ V
Transfer for Nominal Ascent + FPR	1,180	7,082	8,262	(M)	1,180	7,082	8,262	(M)
Transfer for Abort Ascent	-	-	-	(M)	2,909	16,314	19,223	(M)
Available for Abort Roll Control	-	-	-	- (A)	1,344	4,301	5,645	214 (A)
Orbit Maneuvers	330	1,653	1,984	100 (O)	-	-	-	0
Plane Changes, Dispersions, Orbital FPR	387	1,934	2,321	108 (O)	-	-	-	0
Transfer, Rendezvous Orientation	897	2,871	3,768	142 (A)	117	376	493	20 (A)
Reserve for Abort	2,415	12,076	14,491	709 (O)				
Retro	978	4,890	5,868	300 (O)	486	2,433	2,919	150 (O)
Excess LH_2 or LO_2	22	-	22	-	173	-	173	
TOTAL	6,209	30,506	36,715	1,359	6,209	30,506	36,715	284

Table 3.2-8 Δ V CAPABILITY AND PROPELLANT ALLOCATION FROM OMP TANK SYSTEM (Cont'd)

2-Stage Baseline Configuration Model LS 400-7A

Mission	Reference South Polar							
Flight Condition	Normal				Abort			
Propellant and ΔV Increment, lb and ft/sec	Δ W_p LH_2	Δ W_p LO_2	Δ W_p $LH_2 + LO_2$	ΔV	Δ W_p LH_2	Δ W_p LO_2	Δ W_p $LH_2 + LO_2$	ΔV
Transfer for Nominal Ascent + FPR	1,117	6,703	7,820	(M)	1,117	6,703	7,820	(M)
Transfer for Abort Ascent	-	-	-	(M)	2,910	17,459	20,369	(M)
Available for Abort Roll Control	-	-	-	-	1,232	3,942	5,174	220 (A)
Orbit Maneuvers	337	1,683	2,020	100 (O)	-	-	-	-
Plane Changes, Dispersions Orbital FPR	283	1,414	1,697	108 (O)	-	-	-	-
Transfer, Rendezvous, Orientation	780	2,498	3,278	142 (A)	100	322	422	20 (A)
Reserve for Abort	2,805	14,025	16,830	955 (O)	-	-	-	-
Retro	837	4,183	5,020	300 (O)	416	2,080	2,496	150 (O)
Excess LH_2 or LO_2	50	-	50	-	434	-	434	-
TOTAL	6,209	30,506	36,715	1,605	6,209	30,506	36,715	390

LEGEND: (M) Main engine even
(O) RL-10 engine(s) burn
(A) ACS thrusters

TABLE 3.2-8 Δ V CAPABILITY AND PROPELLANT ALLOCATION FROM OMP TANK SYSTEM (Cont'd)

2-Stage Baseline Configuration Model LS-400-7A

Mission	Reference 270 NM 55 Deg Inclination							
Flight Condition	Normal				Abort			
Propellant and ΔV Increment, lb and ft/sec	ΔW_p LH_2	LO_2	$LH_2 + LO_2$	ΔV	LH_2	ΔW_p LO_2	$LH_2 + LO_2$	ΔV
Transfer for Nominal Ascent + FPR	-	-	-	-	-	-	-	-
Transfer for Abort Ascent	-	-	-	-	2,650	15,898	18,548	(M)
Available for Abort Roll Control	-	-	-	-	1,306	4,181	5,487	230 (A)
Orbit Maneuvers	2,169	10,846	13,015	659 (0)	-	-	-	-
Plane Changes, Dispersions Orbital FPR	644	3,218	3,862	199 (0)	-	-	-	-
Transfer, Rendezvous, Orientation	800	2,558	3,358	142 (A)	110	351	461	20 (A)
Reserve for Abort	-	-	-	-	313	1,567	1,880	104 (0)
Retro	1,528	7,637	9,165	500 (0)	453	2,262	2,715	150 (0)
Excess LH_2 or LO_2	-	-	-	-	309	-	309	-
TOTAL	5,141	24,259	29,400	1,500	5,141	24,259	29,400	504

Table 3.2-9

Summary of Dimensional Characteristics of Baseline
2-Stage Orbiter LS 400-7A

Orbiter	2-Stage LS 400-7A
VEHICLE GEOMETRY	
Linear Dimensions, ft	
Length -	
- actual	163
Span	92
Height - gear up	36.75
- gear down	50
Areas, ft^2	
Planform -	
- actual	6,846
Base	1,220
Wetted Area, Including Base	19,070
Volume, ft^3	
Mold Line Volume	99,950
Angular Dimensions, Deg	
Sweep Angle	78
FIN GEOMETRY	
Number of Fins	2
Linear Dimensions, ft	
Height	24.4
True Span	92
Root Chord	33.3
Tip Chord	16.75

Table 3.2-9 (Cont'd)

Orbiter	2-Stage LS 400-7A
Areas, ft^2	
Projected (True) Area Per Fin (including rudder)	624
Wetted Area Per Fin	998
Angular Dimensions, Deg	
Leading Edge Sweep	45
Roll Out, φ	30
Toe-In	4
Tip Wash Out	0
Volumes, ft^3	
Mold Line Volume, Per Fin, Including rudder and auxiliary surface	2,165
Aspect Ratio	2
Airfoil Sections (Root)	Modified Clark Y
CONTROL SURFACES GEOMETRY	
Upper Elevon	
MAC, ft	14.6
Span, ft/side	24
Area, ft^2	693
Hinge Line Sweep, Deg	0
Deflection Range, Deg	
- trailing edge down	0
- trailing edge up	40
- rate, deg/sec	15
Trim Flap	
MAC, ft	23
Span, ft	48
Area (including elevon), ft^2	1,086
Hinge Line Sweep, Deg	0
Deflection Range, Deg	
- trailing edge down	+10
- trailing edge up	-25
- rate, deg/sec	1

Table 3.2-9 (Cont'd0

Orbiter	2-Stage LS 400-7A
Lower Elevon	
MAC, ft	10
Span, ft/side	24
Area, ft^2	543
Hinge Line Sweep, Deg	15
Deflection Range, Deg	
- trailing edge down	+20
- trailing edge up	-20
Rudder	
Number	2
MAC, ft	10
Planform Area, Per Rudder, ft^2	237
Hinge Line Sweep, Deg	15
Auxiliary Control Surfaces	
Number	2
MAC, ft	11.25
Planform Area, Per Surface, ft^2	112.5
Hinge Line Sweep, Deg	10
Leading/Edges	
Radius, ft	
Fin	1
Body-Front	4
Body-Aft	3
Nose Cap (hemispherical)	4

Table 3.2-10

WEIGHT SUMMARY
TWO-STAGE BASELINE VEHICLE SYSTEM LS 400-7A

South Polar Reference Mission

Subsystem	Orbiter Weight (lb)	Booster (1) (lb)	Total (lb)
Aerosurface Structure	16,264	59,639	
Body Structure	42,598	136,059	
Thermal Protection	34,931	59,917	
Landing Gear	8,806	21,432	
Propulsion - Main Ascent	46,060	107,475	
Propulsion - Airbreathing	350	36,154	
Propulsion - Maneuver/ACPS	6,970	4,573	
Prime Power Sources	1,541	1,257	
Electrical System	3,747	1,363	
Hydraulics	2,073	5,290	
Surface Controls	4,085	4,910	
Avionics	3,678	3,682	
Environmental Control	1,274	4,137	
Personnel Provisions	210	290	
Contingency (10 Percent Less ICD Eng)	15,392	30,588	
DRY WEIGHT	187,929	476,766	
Personnel and Effects	725	400	
PAYLOAD	40,000	NA	
Reserves, Residuals and Losses	18,152	80,272 (2)	
Propellant - Ascent	546,439	2,643,026	
Propellant - Airbreathing		90,562	
Propellant - Maneuvers/ACPS	37,848	1,222	
LAUNCH WEIGHT	831,093	3,292,248	4,123,341

(1) Baseline booster scaled (11 engines) for system meeting mission payload requirement.

(2) Including 39,000 lb cruise fuel dumped before landing.

3.3 PERFORMANCE AND SIZING

3.3.1 Two-Stage System Sizing

The concept of total reusability has created the need for recognition of the strong interface between the disciplines involved in the launch system definition. The sizing logic employed recognizes this interface, particularly in the weights/design/performance aspects of the problem. Before a launch system is established, it has undergone a nested weights/performance iterative process. A computerized process is employed that has been developed in part by performance disciplines particularly in the area of design velocity requirements and by weights disciplines having established fixed, mission-dependent or weight-dependent weight laws.

As an example of the performance considerations in sizing, Fig. 3.3-1 indicates, for a given staging velocity, the significance of orbiter and launch system ignition thrust-to-weight ratio on design injection velocity. As indicated, the velocities (ideal) represented reflect the summation of the relative injection velocity and losses that occur during ascent. As thrust-to-weight ratio increases, the ideal injection velocity decreases, primarily because of reduced gravity loss. While not displayed, the influence of increased staging velocity has been found to reduce both the injection velocity and the sensitivity of injection velocity to thrust-to-weight ratio.

Such influences as developed from early two-stage reusable studies are maintained in data banks which encompass a range of ideal staging velocities from 12,000 to 24,000 fps, launch thrust-to-weight ratios of 1.25 to 1.6 and orbiter ignition thrust-to-weight ratios of 1.0 to 3.0. The data stored represent nominal ascent trajectory results. An updating process was achieved to account for the influence of engine-out at staging, a critical factor in two-stage sizing.

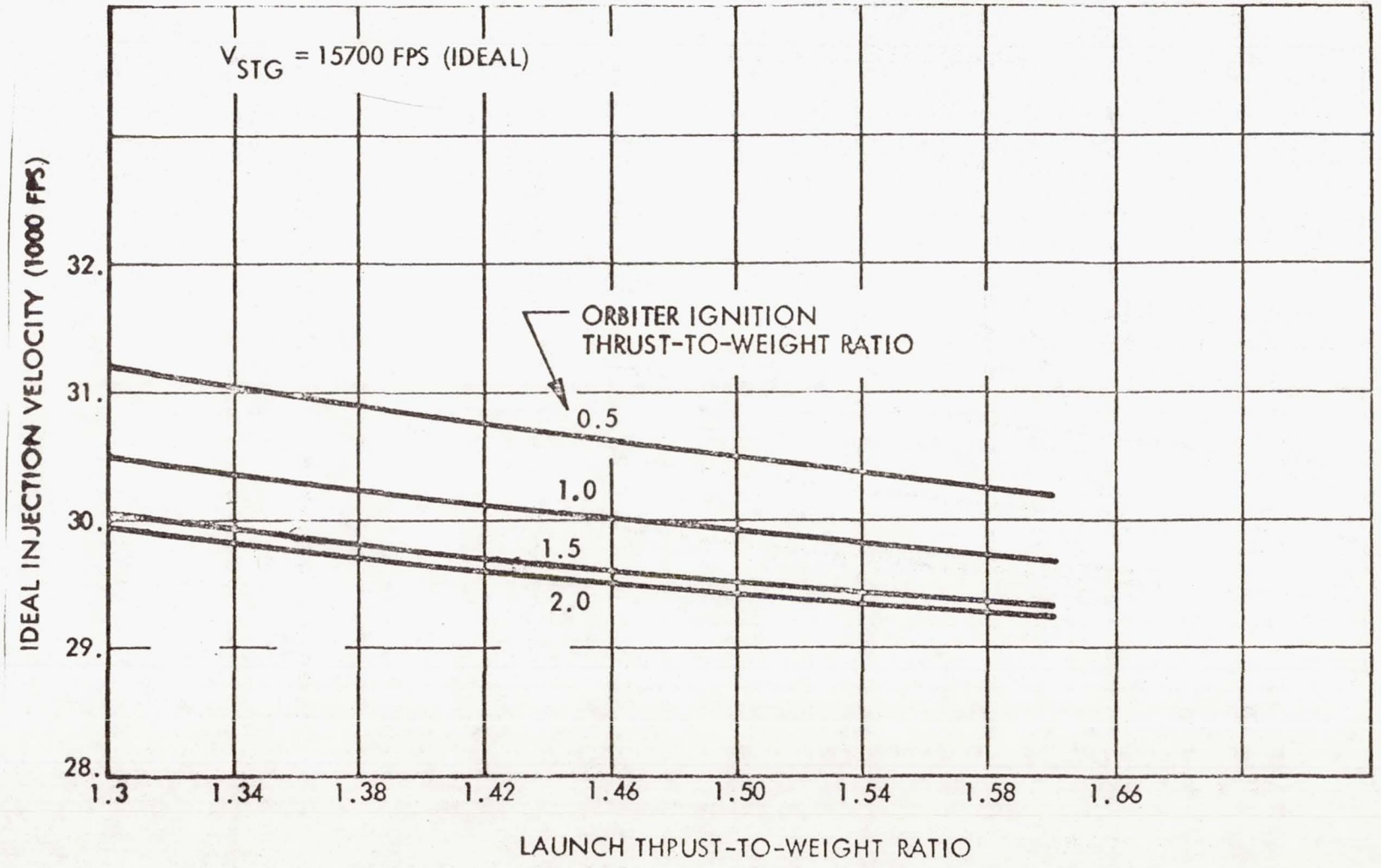
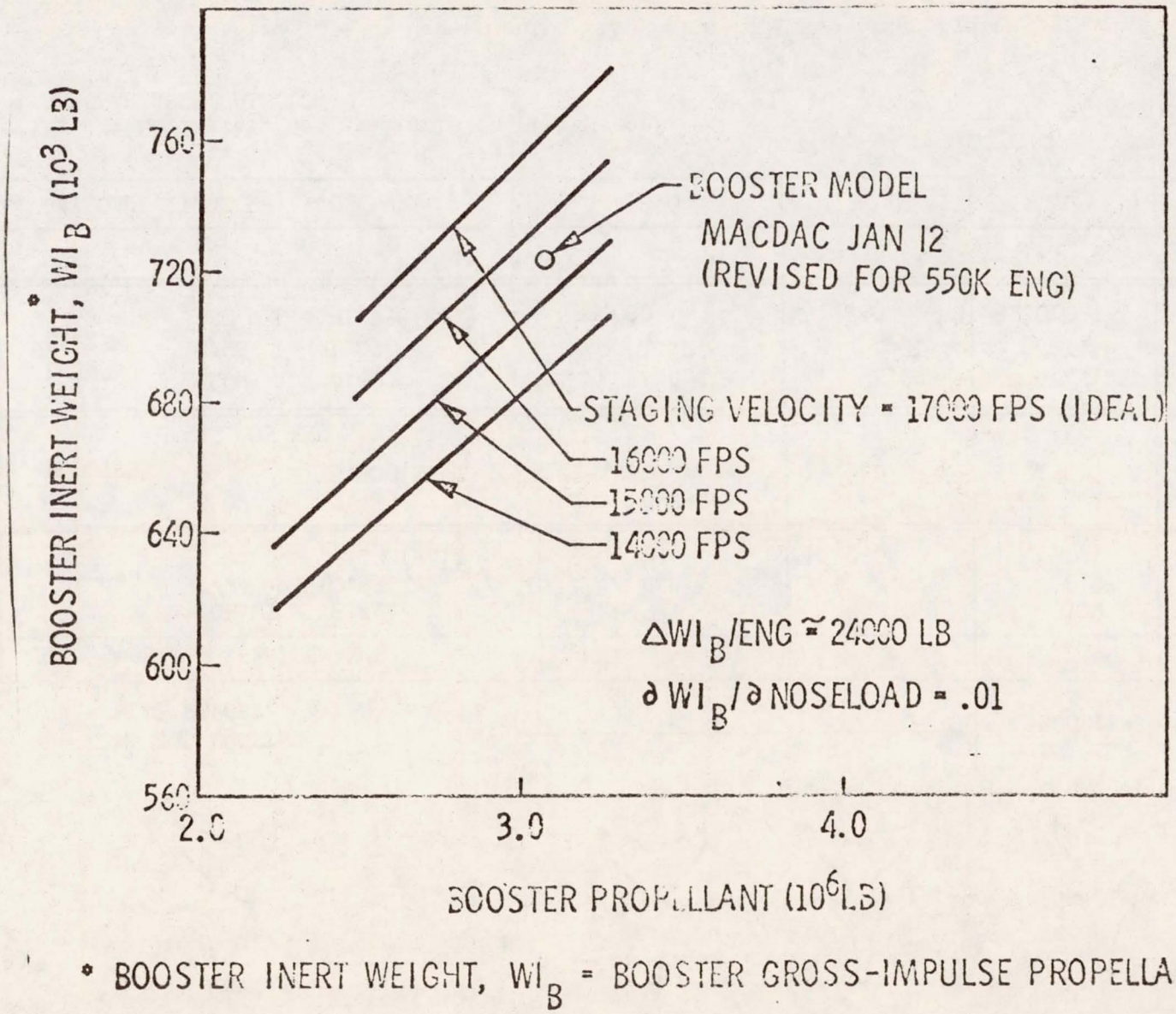


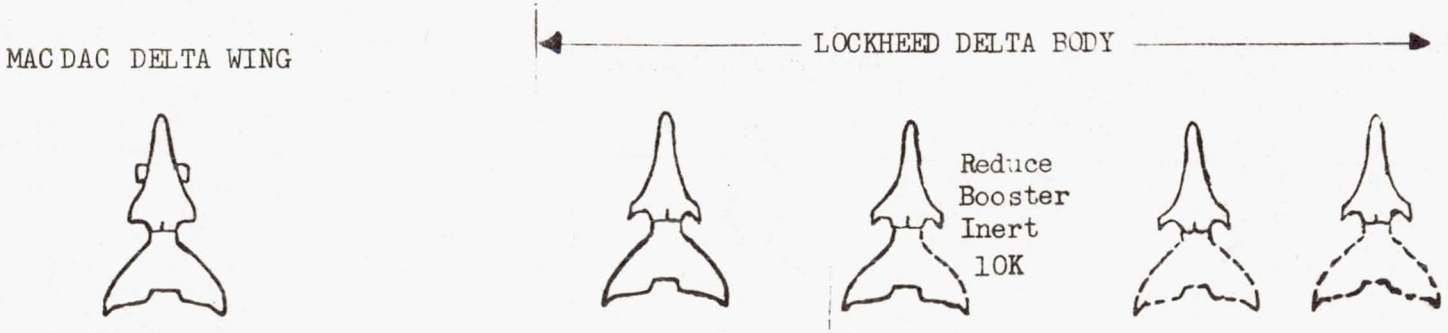
Fig. 3.3-1

Fig. 3.3-1 Velocity/Thrust-To-Weight Relationships



* BOOSTER INERT WEIGHT, WI_B = BOOSTER GROSS-IMPULSE PROPELLANT

Fig. 3.3-2 Booster Weight Summary (Example)



MACDAC BOOSTER
MP-8 REPORT

SCALED
BOOSTERS

ENG 550K (3σ PERFORMANCE)					
ORBITER	IGN WT = 818.4	849.7	843.0	831.2	831.2
	T/W = 0.931	0.811	0.82	0.83	0.83
	V_2 = 16,430(Est)**	16,830	17,090	17,580	17,580
BOOSTER	BLOW = 3.785	3.785	3.774	3.292	3.484
	V_{STG} = 15,300(Est)**	15,750	15,350	14,430	15,030
	NO. OF ENG = 12	12	11	11	10
LAUNCH SYSTEM	GLOW = 4.603	4.63	4.62	4.123	4.32
	T/WO = 1.43	1.42	1.31	1.467	1.275
	ASC VEL = 31,730(Est)**	32,080	32,440	32,010	32,610
PAYLOAD* (1000 LB) 40		58.6	51.9	40	40

**(Est) = Estimated, Not available in MP-8 Report
* - POLAR ABORT MISSION

Fig. 3.3-3 Deveopment of the Delta-Body Two-Stage System

Booster weight laws are developed through detailed studies which provide data output in accordance with NASA weight format requirements. Techniques used are developed in Section 3.2.2. For illustration purposes, a representative booster weight variation developed in the iterative weight cycle of the sizing program is shown in Fig. 3.3-2. Scaling laws for the data shown were developed from the M/DAC 12 Jan booster revised for the 550K main engine. The inert weight represented is the booster weight at post-separation and is therefore sensitive to staging velocity primarily because of the cruiseback propellant requirements. Linearity of booster weight with propellant load is evident. Booster weight, as it is influenced by number of main engines, is shown to be 24,000 lb per engine. This somewhat high sensitivity is due to the weight-cascading effects of such variables as the engine, thrust structure, landing gear, wing (required to maintain a constant wing loading) and cruiseback propellant weights. The final item projects a booster inert-to-orbiter ignition weight of .01.

Sizing normally implies the scoping of overall issues to launch system definition and determining through review the system which best satisfies mission objectives. Since scaling of both the orbiter and booster is undertaken, the sizing process is lengthy and time-consuming; but of particular concern, it loses reliability when large deviations from a previous launch system occur. The method employed in this section was to evolve a two-stage reusable system by maintaining a fixed size orbiter and resizing the booster. In selecting this technique, an orbiter which has had the full benefit of analysis reflected in its weight is maintained. The orbiter taken for study is the 150 ft long, LS 400-7A.

Figure 3.3-3 illustrates the determining process. The polar abort mission is reviewed, since it is the critical design mission of the two-stage reusable system (reviewed in Section 3.3.3). Flow is from left to right starting with the M/DAC launch system. Maintaining the MP-8 booster, the Lockheed delta-body orbiter was substituted for the delta-wing. This substitution, which involved an orbiter replacement smaller in every dimension than the delta-

wing but containing approximately 20,000 lb more propellant, resulted in a significant increase in payload capability. For the next step, the booster size was held fixed and one main engine was removed. Payload remained in excess of the 40K requirement. No further reduction in booster engines was not attempted, since the launch thrust-to-weight ratio would drop below reasonable values. The booster was then resized for the 40K payload requirement, resulting in a choice of two booster systems. If an 11-engine configuration is maintained, a GLOW of 4.123×10^6 lb and a launch thrust-to-weight ratio of 1.467 results. Reduction in booster main engines to 10 results in a GLOW of 4.32×10^6 lb and a launch thrust-to-weight ratio of 1.275.

The 11-engine booster is presently selected for the nominal two-stage system by virtue of the lower GLOW and higher launch thrust-to-weight ratio which promises growth capability. It is felt, however, that through further detailed analysis and sizing studies the 10-engine booster system could be improved in GLOW and thrust-to-weight ratio to the point at which it would be a more suitable candidate.

3.3.2 Two-Stage Ascent Trajectory

Two-stage ascent trajectories were developed for two booster-orbiter combinations of differing launch weight. These systems comprised a M/DAC type booster and a Lockheed delta-body orbiter. The selection was made to establish bounds within which a final system would lie.

The heavier of the two systems required 12 booster engines to maintain the launch thrust-to-weight ratio (1.41) characteristic of present two-stage systems. The lighter system was simulated with 10 booster engines, resulting in an initial T/W of 1.36.

Drag characteristics initially used for the simulation were those representative of the LMSC shuttle. Later, M/DAC drag data were used in the simulation, and the different effects of these two groups of data were assessed. Use of the M/DAC drag instead of the LMSC drag resulted in an increase of the mission ideal velocity to orbit requirement of 156 fps. In addition, the M/DAC drag produced lower dynamic pressure so that maximum q was 60 psf less than maximum q using LMSC drag. Because of the higher velocity requirement associated with the M/DAC drag, these data were selected to represent the two-stage system aerodynamic characteristics. Figure 3.3-4 presents the zero-lift drag coefficient as a function of Mach number.

Represented in Figs. 3.3-5 and 3.3-6 are the trajectory parameters associated with the 12-engine booster, two-stage launch system.

The groundrules and assumptions used to generate the two-stage system ascent trajectories were:

- Design reference orbit: A circular orbit of 270 nm altitude and inclined at 55 deg to the equator.
- Launch from ETR with an azimuth heading east of north
- Reference injection orbit: 50 x 100 nm

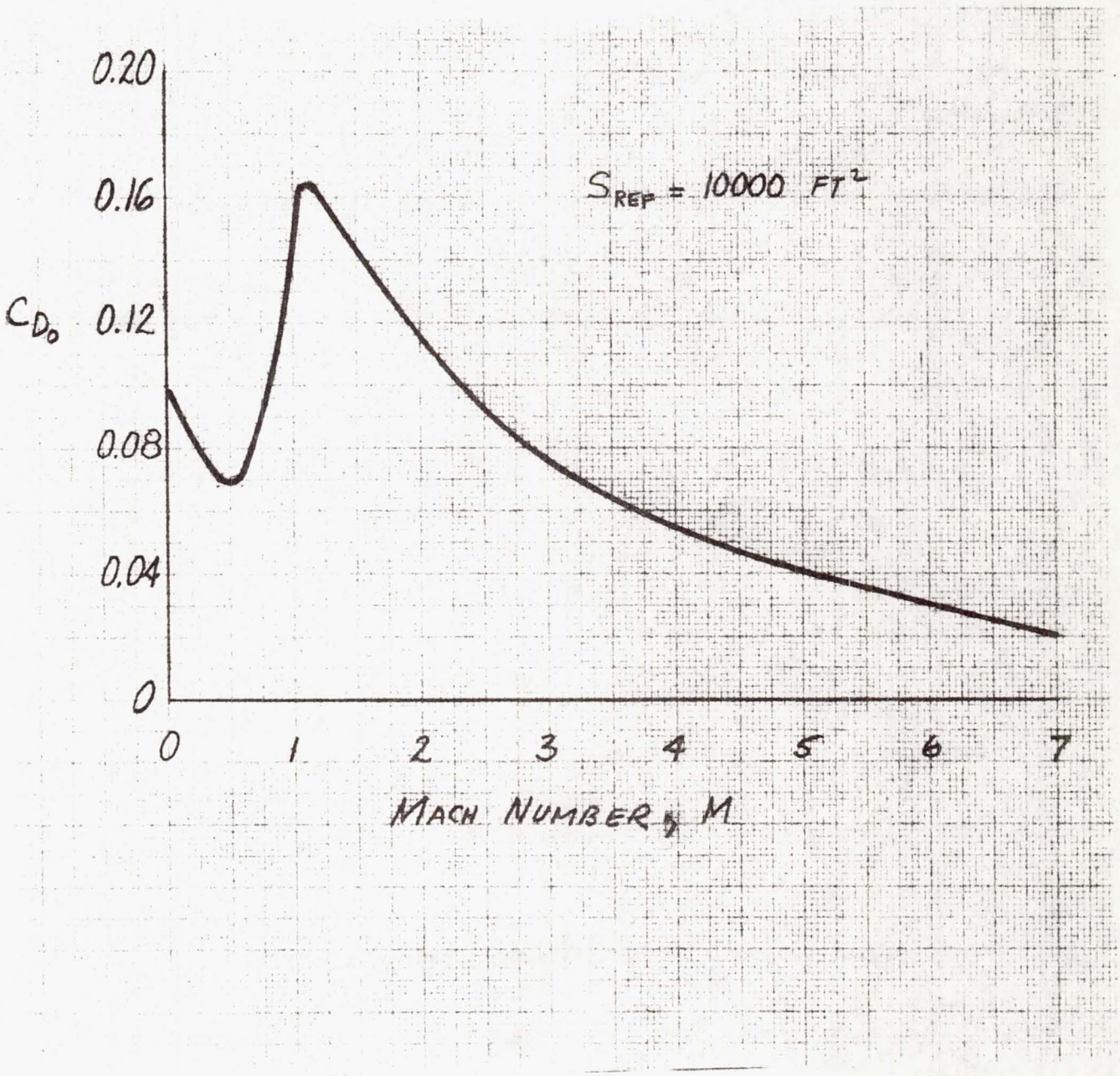


Fig. 3.3-4 Zero-Lift Drag Coefficient As a Function of Mach Number

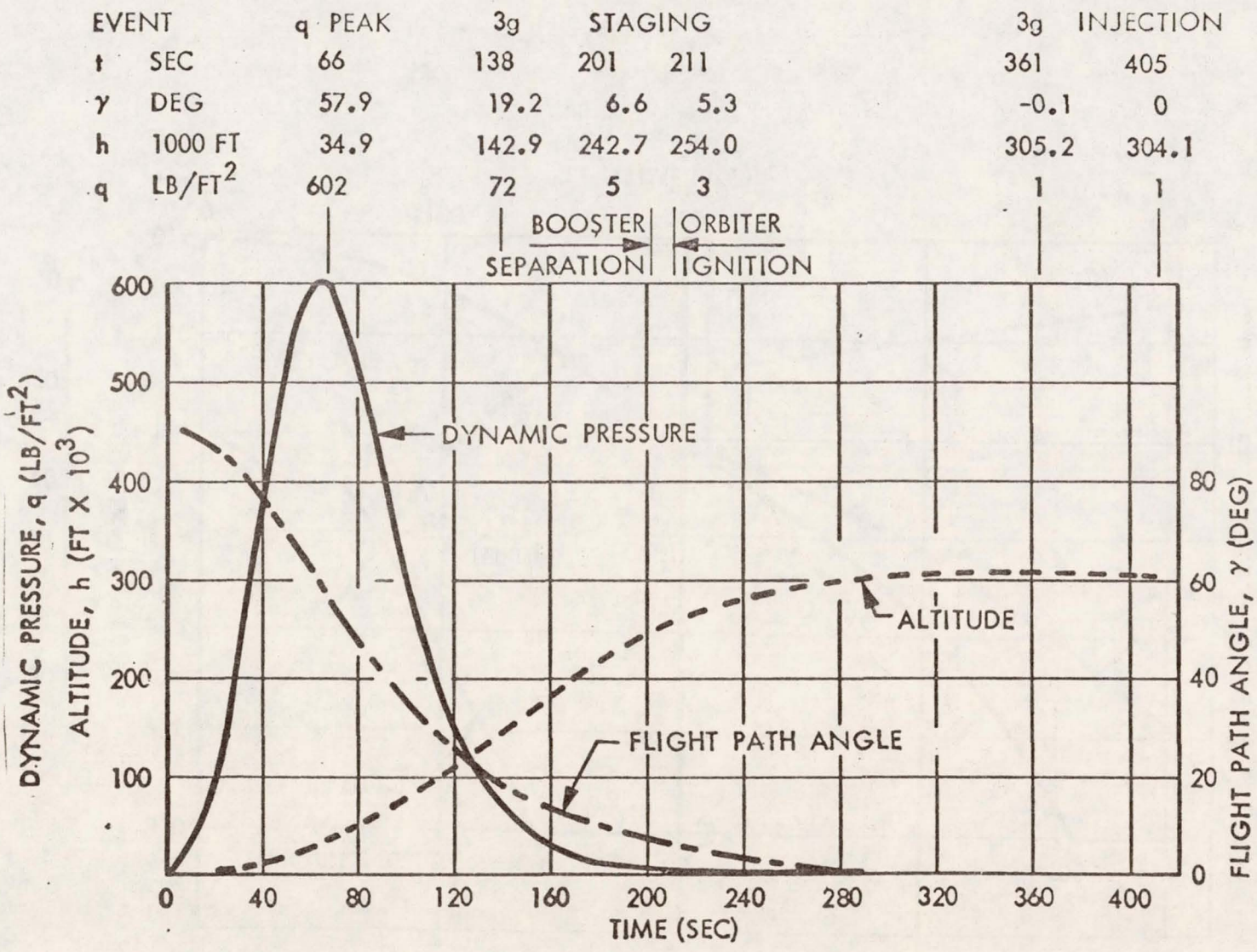


Fig. 3.3-5 Two-Stage Ascent Trajectory

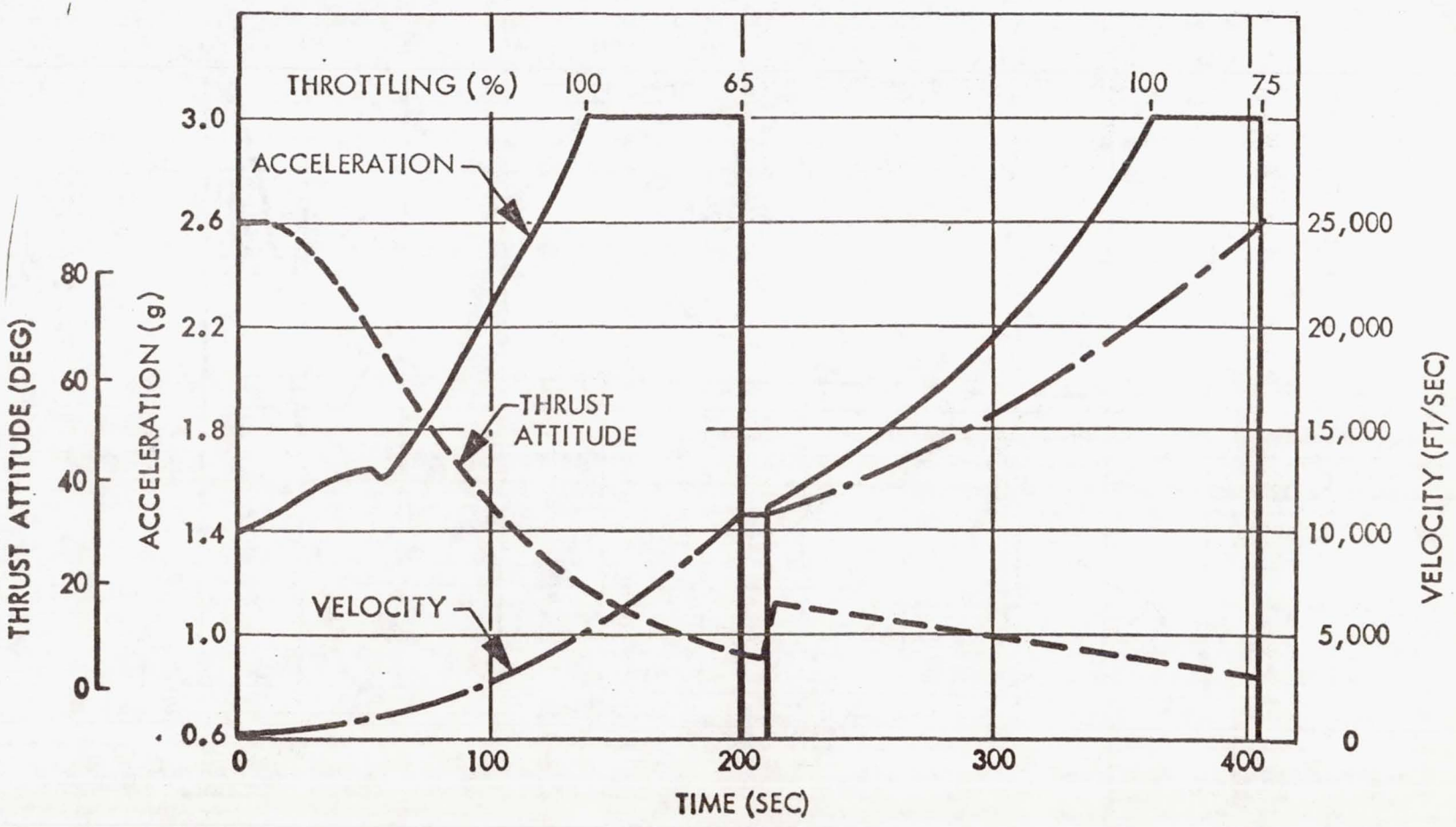


Fig. 3.3-6 Two-Stage Ascent Trajectory

- o Booster Propulsion
  - 12 engines (10 engines)
  - Sea-level thrust 550,000 lb/engine
  - Vacuum thrust 604,000 lb/engine
  - Nominal vacuum I_{sp} 439 sec
  - Nominal sea level I_{sp} 400 sec
- o Orbiter Propulsion
  - 2 engines
  - Vacuum thrust 632,000 lb/engine
  - Nominal vacuum I_{sp} 459 sec
- o 1962 Standard atmosphere
- o Operational Modes
  - Vertical flight for 14 sec
  - Gravity turn during booster phase
  - Ten-second coast between booster separation and orbiter ignition
- o Optimal thrust attitude programming to injection
  - 3-g acceleration limit during booster and orbiter phases
  - Booster throttling by maintaining all engines in operation with I_{sp} degradation of 3 sec between 100 and 50 percent

Three-sigma engine performance was used for both the booster and orbiter. Because of this, booster I_{sp} for the 10-engine configuration was slightly less than the 12-engine vehicle.

In addition to generating the two-stage ascent trajectory to 55 deg inclination, two other inclinations were included, 28.5 and 90 deg. These cases were used to determine the critical mission involving abort and are discussed in the following section. For the additional inclinations, the optimum launch azimuth was used for each inclination, due east and almost due south.

3.3.3 Abort-to-Orbit (Two-Stage Orbiter)

Providing capability to perform a once-around trajectory with a one-engine-out condition from staging introduces a complex interplay between the nominal and abort mission requirements and the sizing of a compatible orbiter/booster combination. To provide an understanding of the problem scope, a study has been completed which reviews the considerations attendant to ascent and return modes and develops the energy requirements associated with critical mission requirements whether it is determined from the nominal or orbit mode.

3.3.3.1 Launch System Models. Abort and nominal mission performance considerations are reviewed in Fig. 3.3-7. The three reference missions are studies. All performance studies are based on 3 σ engine performance.

Concern as to the influence of staging velocity on engine-out performance required that a bounding investigation be accomplished. For this reason, two launch system sizes are considered. System I represents a present two-stage system size and is made up of a 12-engine booster and the Lockheed delta-body orbiter. System II represents the same Lockheed delta-body orbiter; however, in this case, the booster is reduced in size and number of main propulsion engines. The major comparative performance parameters between the two systems are ignition thrust-to-weight (both stages) and staging velocity. The breakdown in ascent and orbital tank volumes in the orbiter is developed from the 55-deg mission which requires 1500 fps on-orbit velocity capability but has on-orbit tanks sized for 2000 fps.

3.3.3.2 Ascent Considerations. Representative altitude-time histories of ascent are shown in Fig. 3.3-8 for a nominal ascent mode (a) as well as two abort modes (b) available for consideration. The solid line of Fig. 3.3-8 (b) indicates the velocity penalty of 960 fps associated with abort from the nominal staging point. The second mode directs the launch system to fly an optimal profile so that, if an engine-out occurs at staging, the summation of velocity losses from launch to injection is minimized. As can be seen,

ANALYSIS MODEL

MISSIONS
ETR LAUNCH – DUE EAST
WTR LAUNCH – POLAR
ETR LAUNCH – 55 DEG INCLINED

PROPULSION
550K RATED SL POWERHEAD $I_{sp_{del}} = I_{sp_{nom}} - 3/\sqrt{NE}$
BOOSTER $I_{sp_{nom}} = 439$
ORBITER $I_{sp_{nom}} = 459$
NE = NO. OF ENGINES
EPL = 109% RATED THRUST
(2) RL - 10 I_{sp} = 439 SEC
ACS I_{sp} = 352 SEC

LAUNCH SYSTEMS		
SYSTEM	I	II
LAUNCH GLOW	4.69×10^6	4.04×10^6
NO. OF BOOSTER ENGINES	12	10
BOOSTER PROPELLANT	3.13×10^6	2.59×10^6
ORBITER GLOW	838.5×10^3	834.5×10^3
NO. OF ORBITER ENGINES	2	2
ORBITER PROPELLANT	DETERMINED	DETERMINED
LAUNCH T/W	1.41	1.36
ORBITER T/W	1.51	1.52
V_{STAGE} (IDEAL)	15,435	14,450
V_{STAGE} (RELATIVE)	11,100	9,950

TRAJECTORY ANALYSIS

D03017

Fig. 3.3.3-7

Fig. 3.3.3-7

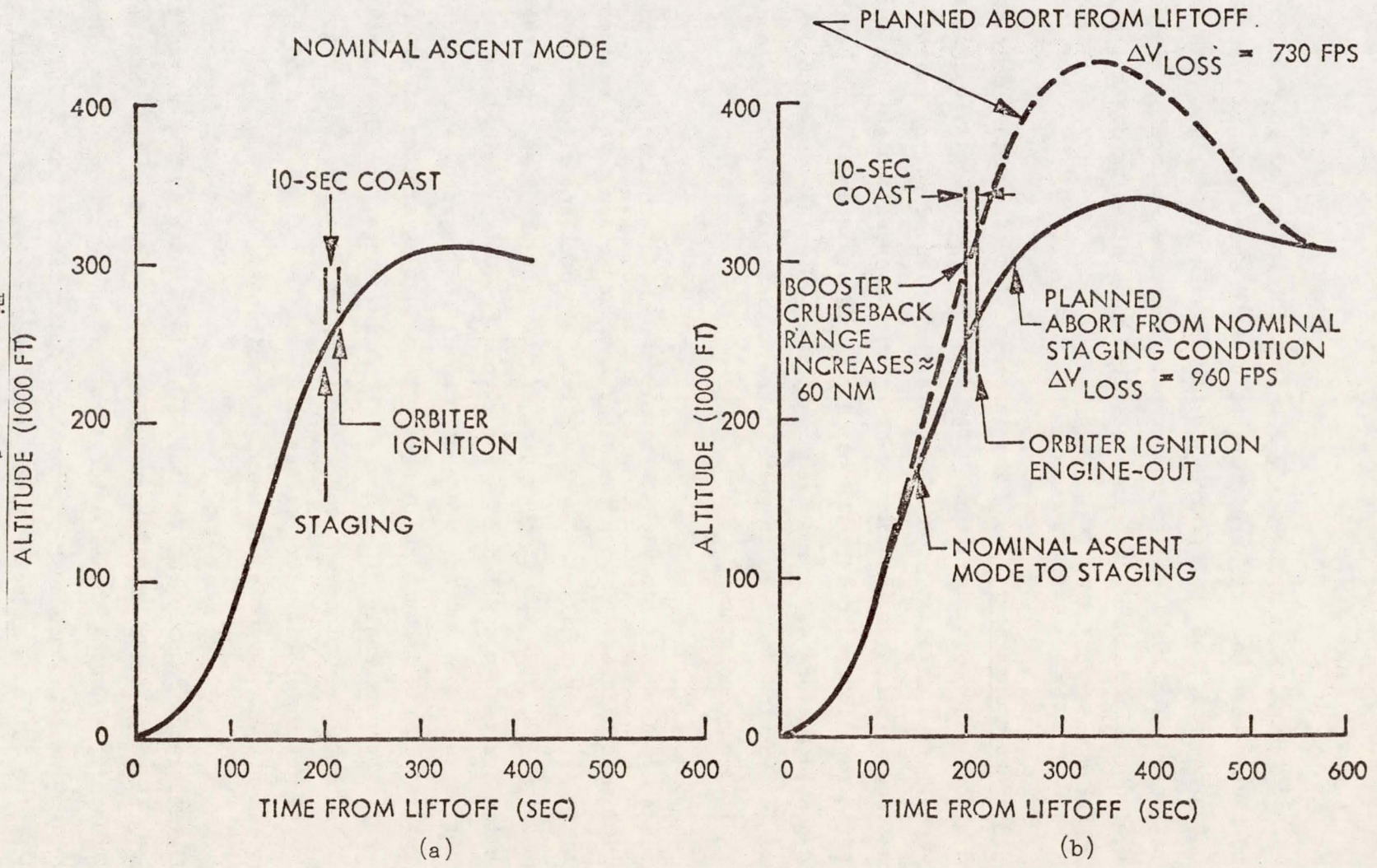


Fig. 3.3.3-8 Abort Through Orbit - Two-Stage

the velocity penalty is reduced 130 fps from the unplanned abort mode. A second factor must be considered for the planned abort mode is that the nominal staging condition is perturbed to higher values of flight path angle and altitude. The result is an approximate increase in booster cruiseback range of 60 nm, a factor which is considered in evaluating overall payload effects at a later point in this study.

3.3.3.3 Return Considerations. The determination of a return mode is influenced by two major considerations. The first is the velocity budget that must be allotted in addition to the losses experienced during ascent. The second consideration is the time involved in the operation and the corresponding earth rotation and crossrange buildup during this abort time period. These considerations are summarized in Fig. 3.3-9. Investigation was directed toward determining the case in which crossrange required is less than the design 1100 nm condition.

It is assumed for this study that the orbiter is injected into the nominal 50 x 100 nm condition. During the staging to injection mode, roll control authority is relegated to the ACPS system. A propellant allotment derived by assuming full roll thrust operation during half the time from staging to injection has been converted to a corresponding 300 fps velocity. Orientation velocities for retro and entry are assumed at 10 fps each. The selection of the return ellipse is such as to place the returning orbiter at 400,000 ft with a flight path angle of -1 deg, 4000 nm downrange from the launch base. For this particular placement condition, the retro velocity derived was 107 fps. The retro velocity derived was increased to 150 fps to allow for other entry flight path angles and downrange conditions.

Total crossrange requirement for the WTR polar launch is 880 nm and is based on a 33 nm range drift that occurs during ascent added to the rotation of the launch site during the 4100 sec transfer and return process. The crossrange requirement increases to 940 nm for polar launch from KSC due to the launch site latitude effect. Since these crossranges are less than the 1100

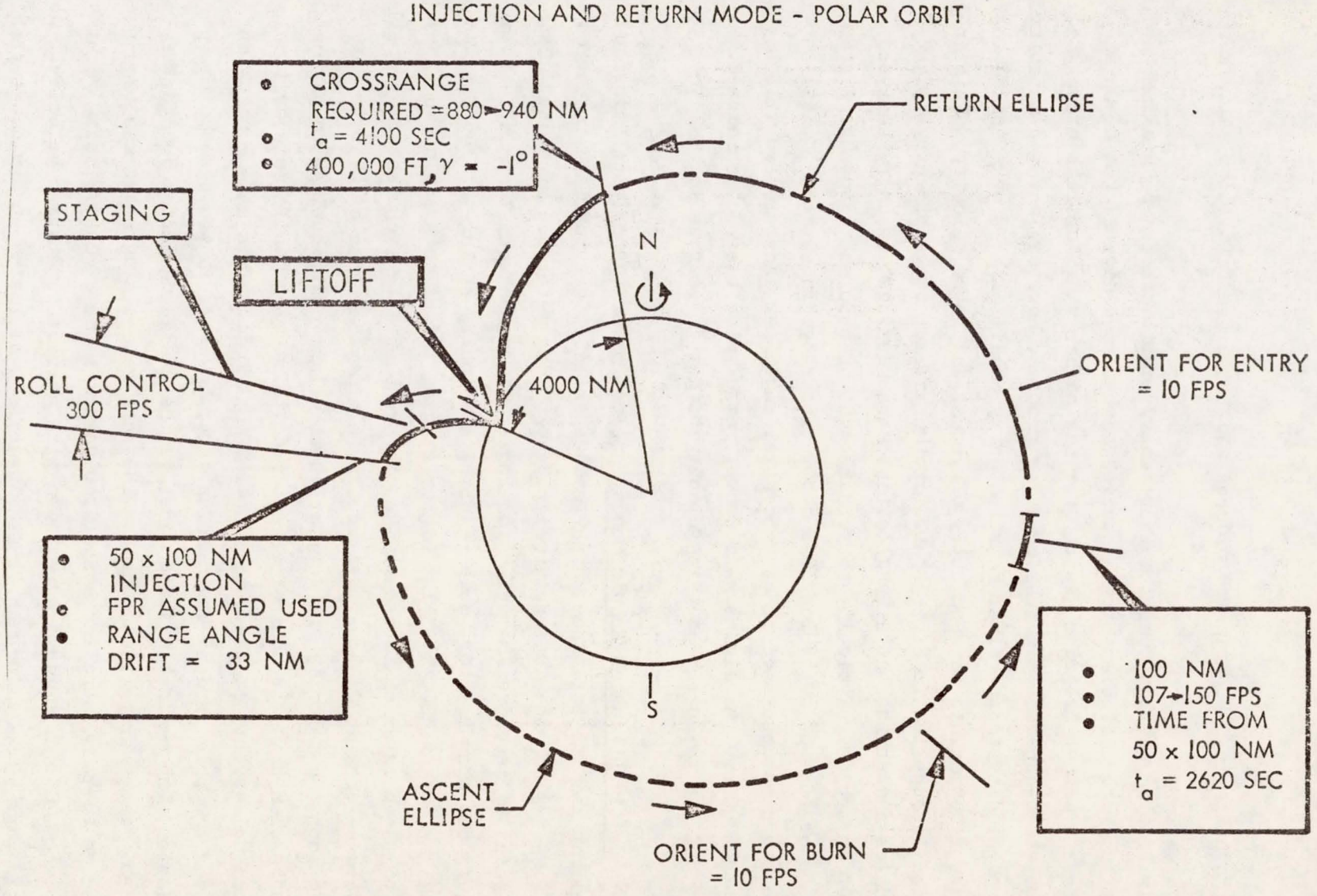


Fig. 3.3.3-9 Abort Through Orbit

nm design criteria, the transfer process and logic are felt to be satisfactory.

There is little question that the abort entry conditions can be accommodated, especially since sufficient trade studies have been conducted (EM L2-06-01-M2-7, Fourth Letter Progress Report) for the higher wing loadings associated with the stage-and-one-half orbiter. The problem, however, was the matching of retro velocity to the final selected entry condition. It was decided to scope the relationship between entry placement and retro velocity. This information is supplied in Fig. 3.3-10 for the maximum range of entry flight path angles and earth central angles that have been used in entry analyses from 100 nm orbits. Typically for crossranges of 1500 nm, downrange distances in the order of 5000 nm. Since the abort crossrange is between 880 and 940 nm, the expected downrange or earth central angle for abort return will be less than 4000 nm. As Fig. 3.3-10 shows, retro velocities do not exceed 150 fps and the selection of this velocity budget is reasonable.

3.3.3.4 Critical Mission Requirements. Selection of the critical mission requirement and its influence on payload capability is summarized in Tables 3.3-1 to 3.3-3 for the due east, polar, and 55 deg reference missions. Using Table 3.3-3 to describe the selection logic, it can be seen that in terms of velocity, the planned abort mode results in velocity savings of 148 fps, benefitting payload by 3000 lb. The higher staging flight path angle, which adds 60 nm to the booster cruiseback range, is translated to a cruiseback propellant increase of 21,600 lb. This results in a payload loss of 4750 lb. By summing both the velocity and booster inert weight contributions to payload, it can be seen that the nominal ascent mode to staging, while more critical than nominal ascent, is the better operational mode. This same conclusion holds for the polar and 55 deg mission data; however, in the case of the 55 deg mission, the nominal mission model is the designing factor due to the high orbit maneuvering requirement. The same determining logic has been applied to System II and for comparison purposes the excess velocity requirements (nominal and abort) are reviewed with respect to staging velocity. As shown in Figs. 3.3-11 through 3.3-13 the effect of losing an engine at staging

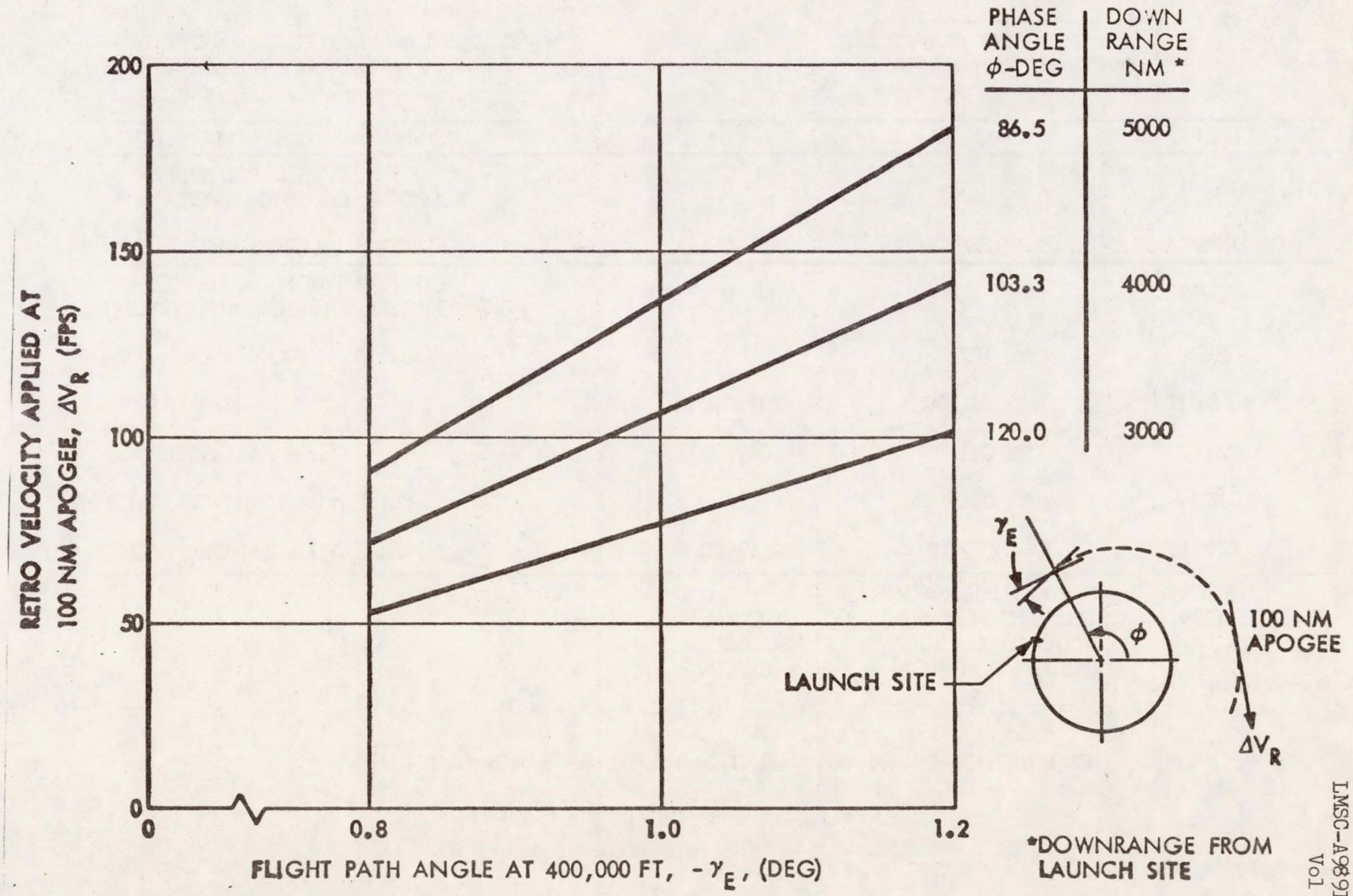


Fig. 3.3.3-10 Abort-to-Orbit Analysis — Retro Velocity Requirement

Table 3.3-1

Ascent Mode Selection

System I - VSTG (REL) = 11,100 fps

WTR LAUNCH - 90-DEG INCLINED, 100-NM CIRCULAR ORBIT

MODE	1 NOMINAL ASCENT MODE	2 NOMINAL ASCENT MODE TO STAGING, ENGINE-OUT TO INJECTION	3 PLANNED ABORT MODE
IDEAL ASCENT VELOCITY (FPS)	30,691	31,652	31,423
ROLL CONTROL ΔV (FPS)	0	300	300
ON-ORBIT ΔV (FPS)	650	170	170
Σ ΔV (FPS)	31,341	32,122	31,893
ΔV FROM NOMINAL (FPS)	0	781	+552
INCREASE IN BOOSTER CRUISEBACK FUEL (FROM NOMINAL) (LB)	0	0	21,600
ΔPAYLOAD DUE TO VELOCITY	0	-15,620	-11,040
ΔPAYLOAD DUE TO BOOSTER WEIGHT INCREASE	0	0	-4,750
PAYLOAD POTENTIAL CHANGE	0	-15,620	-15,790

∂ PAYLOAD/ ∂ V = 20 Lb/fps
∂ PAYLOAD/ ∂ BOOSTER WT = 0.216 LB/LB

DETERMINE PAYLOAD CAPABILITY

Table 3.3-2

Ascent Mode Selection

System I - VSTG (REL) = 11,100 fps

ETR LAUNCH – 55 DEG INCLINED + 1500 FPS

MODE	1 NOMINAL ASCENT MODE	2 NOMINAL ASCENT MODE TO STAGING, ENGINE OUT TO INJECTION	3 PLANNED ABORT MODE
IDEAL ASCENT VELOCITY (FPS)	29,752	30,567	30,393
ROLL CONTROL ΔV (FPS)	0	300	300
ON-ORBIT ΔV (FPS)	1,500	170	170
Σ ΔV (FPS)	31,252	31,037	30,863
ΔV FROM NOMINAL (FPS)	0	-215	-389
INCREASE IN BOOSTER CRUISEBACK FUEL (FROM NOMINAL) (LB)	0	0	+21,600
ΔPAYLOAD DUE TO VELOCITY	0	+4,300	+7,780
ΔPAYLOAD DUE TO BOOSTER WEIGHT INCREASE	0	0	-4,750
PAYLOAD POTENTIAL CHANGE	0	+4,300	+3,030

∂ PAYLOAD/ ∂ V = 20 Lb/FPS
∂ PAYLOAD/ ∂ BOOSTER WT = 0.216 LB/LB

DETERMINE PAYLOAD CAPABILITY
SIZE ASCENT TANKS
SIZE ORBIT TANKS FOR 2000 FT/SEC

Table 3.3- 3

Ascent Mode Selection

System I - VTSG (REL) = 11,100 fps

ETR LAUNCH - 28.5 DEG INCLINED, 100 NM CIRCULAR ORBIT

MODE	1 NOMINAL ASCENT MODE	2 NOMINAL ASCENT MODE TO STAGING, ENGINE OUT TO INJECTION	3 PLANNED ABORT MODE
IDEAL ASCENT VELOCITY (FPS)	29,221	29,953	29,805
ROLL CONTROL ΔV (FPS)	0	300	300
ON-ORBIT ΔV (FPS)	650	170	170
Σ ΔV (FPS)	29,871	30,423	30,275
ΔV FROM NOMINAL (FPS)	0	+ 552	+404
INCREASE IN BOOSTER CROSSRANGE FUEL (FROM NOMINAL) (LB)	0	0	+21,600
ΔPAYLOAD DUE TO VELOCITY	0	- 11,040	-8,080
ΔPAYLOAD DUE TO BOOSTER WEIGHT INCREASE	0	0	-4,750
PAYLOAD POTENTIAL CHANGE	0	- 11,040	-12,830

DETERMINE PAYLOAD CAPABILITY

∂PAYLOAD/∂V = 20 LB/FPS
∂PAYLOAD/∂BOOSTER WT = 0.216 LB/LB

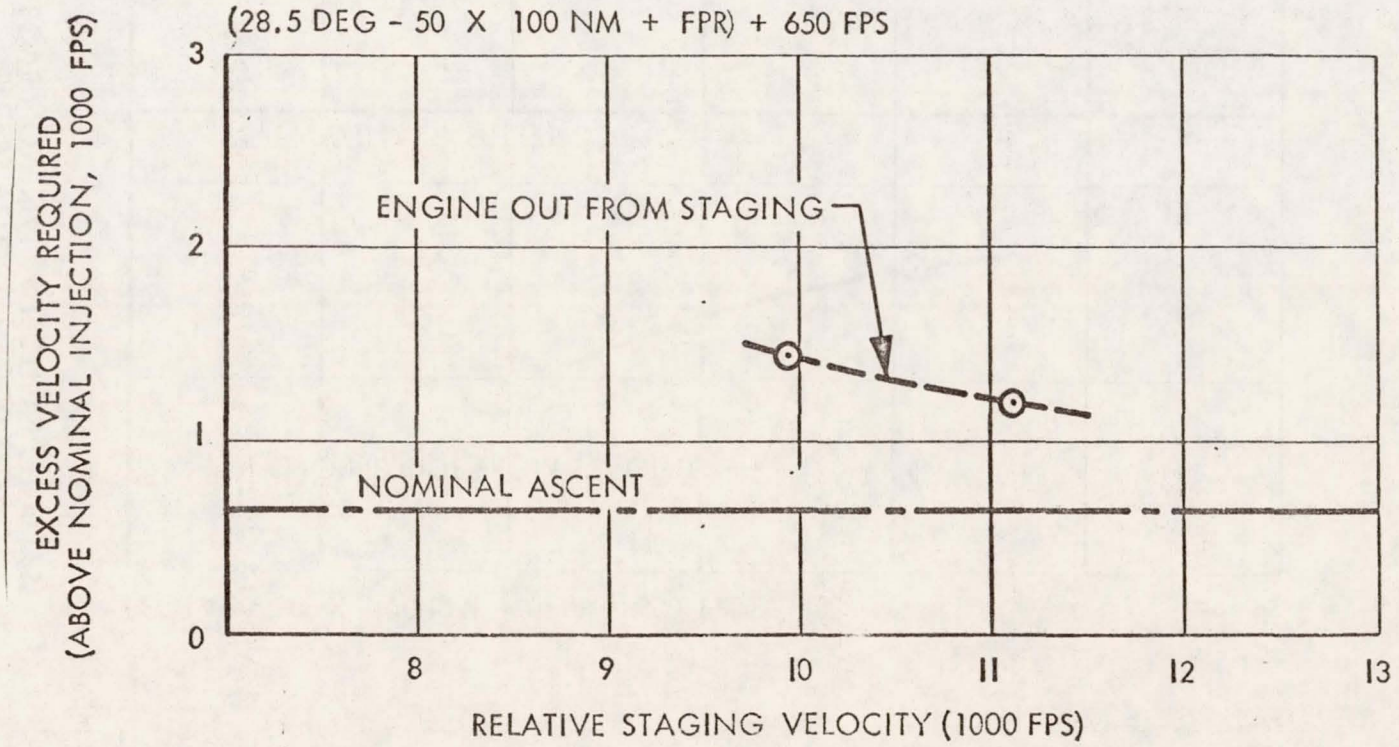


Fig. 3.3.3-11 2-Stage Velocity Criteria

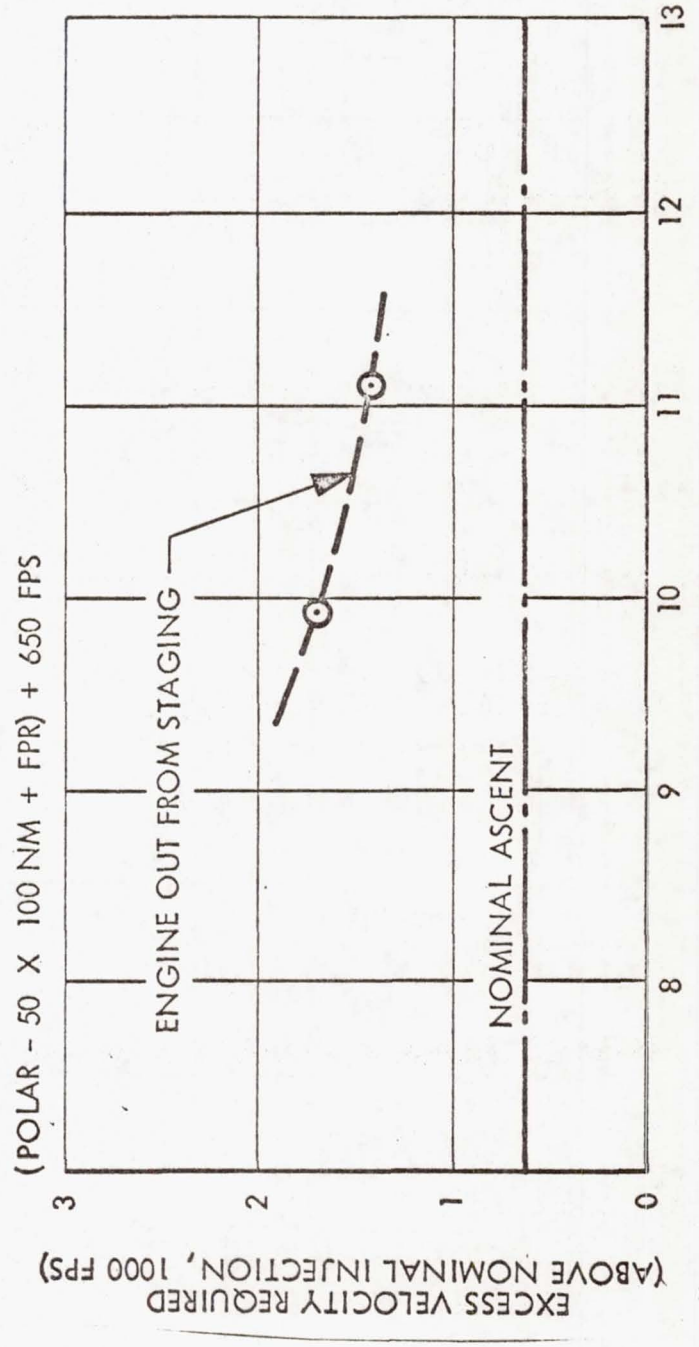


Fig. 3.3.3-12 2-Stage Velocity Criteria

Fig. 3.3.3-12

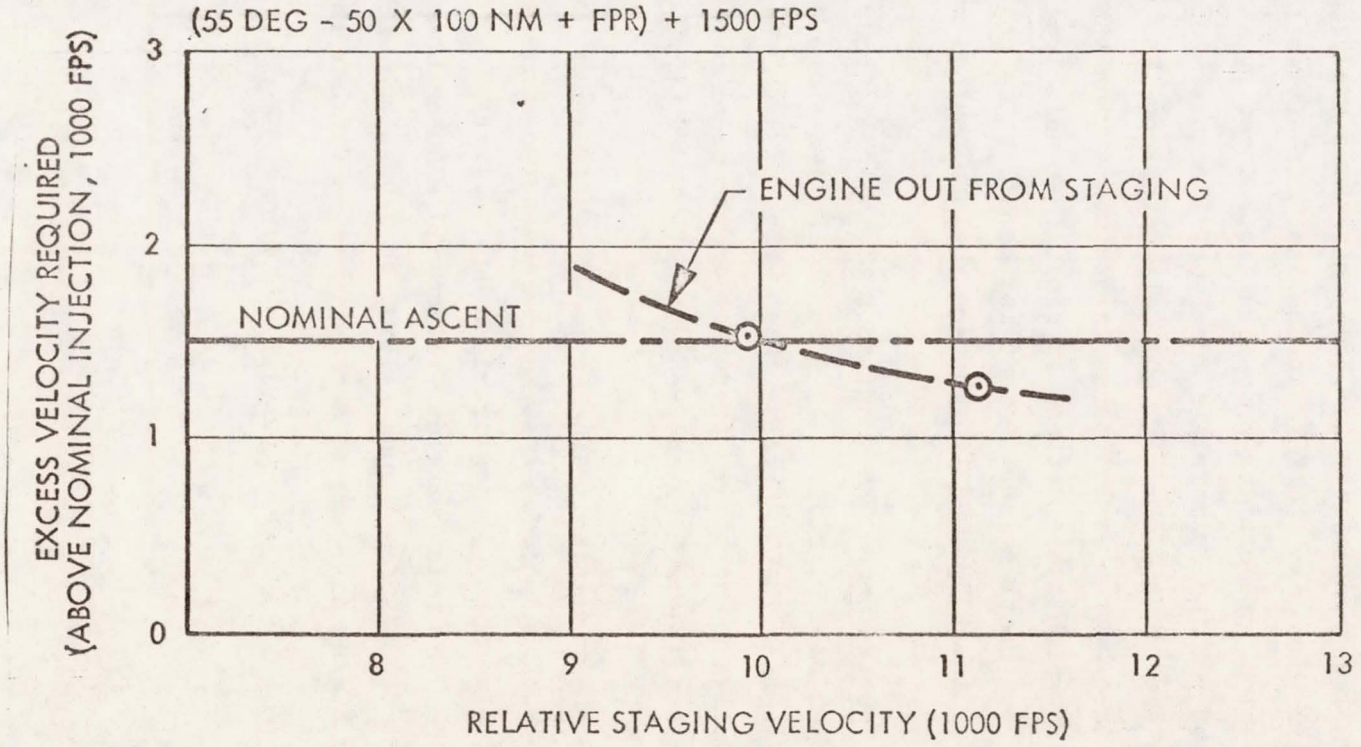


Fig 3.3.3-13 2-Stage Velocity Criteria

for System II is the dominating term for all three missions.

Payload delivery capability for the two systems analyzed is shown in Table 3.3-4 and is based on the fixed propellant capacities listed. For the 55-deg 270 nm mission, the orbiter propellant is reduced by an equivalent on-orbit velocity of 500 fps to maintain the Level 1 requirements. Full propellant loads are used in the due east and polar missions, and transfer of propellant from the on-orbit to ascent tanks is assumed during the abort modes. The stage inert weights are defined as gross less main impulse propellant and payload weight. Booster inert weight is therefore the post-staging weight while the orbiter inert becomes its post-retro less payload weight.

3.3.3.4 Conclusions. The adaptation of the delta body orbiter into the two-stage system will significantly enhance the two-stage performance potential. It is clear that when booster size is maintained at Phase B levels (System I), payload delivery capability increases in the order of 50 to 120 percent for the polar and 55-deg missions respectively.

The polar abort mission is the critical requirement. Note that System II with a GLOW of 13 percent less than present two-stage systems is sufficient for the due east and 55-deg missions but delivers 7,400 lb less payload than required for the polar mission. It should also be considered that there was no attempt to optimize through sizing the two-stage GLOW. It can be expected that if a sizing analysis is accomplished, a further improved system will result.

Table 3.3-4

Payload Capabilities

	SYSTEM					
	I			II		
	Due East	Polar	55°	Due East	Polar	55°
Booster Propellant (10^6 lb)	3.125	3.125	3.125	2.594	2.594	2.594
Booster Inert (10^3 lb)	.729	.736	.731	.602	.619	.615
Orbiter Propellant (10^3 lb)	589,296	589,768	580,768	589,296	589,296	580,768
Orbiter Inert (10^3 lb)	195,809	195,809	219,887	195,809	195,809	219,887
Orbiter ABES	Out	Out	In	Out	Out	In

Design Requirement	Abort	Abort	Nominal	Abort	Abort	Abort
PAYLOAD (1,000 lb)	101.3	60.3	55.3	70.7	32.6	29.4
GLOW (10^6 lb)	4.749	4.715	4.720	4.060	4.039	4.048

3.3.4 Payload Performance

The inherent differences between the stage-and-one-half and two-stage concepts require that the design effects of the mission requirements be reviewed in a separate manner. It has been established (Section 3.3.3) that the combined effects of payload and engine-out from staging for the polar mission are the critical factors to the two-stage launch system size. An added complexity to the two-stage launch system is the requirement for on-orbit tanks sized for 2000 ft/sec in the 55 deg 270 nm mission. These effects were accounted for in the system size by (1) determining the ascent and on-orbit propellant breakdown for the 55 deg 270 nm mission, and (2) with the established orbiter, sizing the booster to accommodate the polar abort mission.

With the propellant and system weights fixed, payload delivery capability was then determined. Total propellant in the orbiter varies between missions. For the due east and polar missions, the orbiter tanks were filled to usable capacity, while for the 55 deg mission on-orbit propellant was offloaded by an equivalent 500 fps. In all cases, 3σ engine performance is used in the payload development.

The three mission payload potentials are reviewed in Table 3.3-5 for the worst of nominal or orbit requirements. Since the polar abort requirement sized the booster, the fixed vehicle capability is 40 lb and is in agreement with the polar mission requirement. For the due east and 55 deg missions, excess payload capabilities of approximately 14K and 10K respectively are achieved. Propellant offloading of the booster and/or orbiter is projected, if significant operational costs can be accrued.

The accommodation of the 550K ICD engine in the two-stage orbiter leads to a two-engine configuration due to engine/orbiter design integration and improved payload capabilities for nominal (nonabort) operations. Since the polar abort requirement sizes the launch system, it is felt that some design steps should be undertaken. Two are available. Resizing of both booster and orbiter to

accomplish higher nominal staging velocities will reduce the significance of engine-out at staging; however, this is not felt to be the best answer from the overall system standpoint. A recommended approach is a three-engine orbiter design. This second approach would considerably alleviate the abort performance aspects of the design but a lower thrust engine than the present ICD is required.

Table 3.3-5

TWO STAGE SYSTEM

MAXIMUM PAYLOAD POTENTIAL - FIXED LAUNCH VEHICLE

MISSION	DUE EAST	POLAR	55 DEG
Orbit Altitude (nm-Circ)	100	100	270
Design for:	Abort	Abort	Nominal Mission
GLOW (10^6 lb)	4.15	4.12	4.13
Booster Propellant (10^6 lb)	2.64	2.64	2.64
Orbiter Ignition Wt (1000 lb)	871.2	831.2	843
Ascent Propellant (1000 lb)	580.0*	581*	546.3
FPR Propellant (1000 lb)	5.9	5.4	6.0
On Orbit Propellant (1000 lb)	3.4	2.9	29.4
Launch Thrust to Weight Ratio	1.46	1.47	1.46
Orbiter Ignition Thrust to Weight Ratio	.80	.83	1.50
Ideal Velocity - Ascent (fps)	30,370	32,010	29,650
Staging (fps)	14,300	14,430	14,400
Payload (1000 lb)	79.5	40.0	35.9

* Accomplished by fuel transfer from orbital maneuver tanks.

3.3.5 Airbreather Performance

The logic of Section 2.4.4, which was employed for the stage-and-one-half orbiter, has been maintained in the airbreather performance analysis of the two-stage orbiter. As before, the objective of the study was the determination of fuel requirements and flight capability through review of the combined effects of orbiter weight, aerodynamics, and jet engine candidates.

The representative return-orbiter weight from the 55-deg 270 nm reference mission is 244,000 lb. With its 25,000-lb payload, the vehicle cg is located at approximately 74-percent of reference length, resulting in a trimmed lift-to-drag ratio and lift coefficient of 5.4 and 0.52, respectively.

Fuel requirements to accomplish the terminal airbreather profile from 40,000 ft altitude through go-around and landing are shown in Fig. 3.3-14. As design startup weights decrease and number of engines increase, a crossover in fuel requirement is evidenced. This is due to the startup and flight idle fuel that increases with number of engines. Although the go-around fuel requirements reduce with increased engine number, the savings is not sufficient to compensate for the fuel expended during the early high altitude flight mode. Variations of $\pm$ 30,000 lb in vehicle design weight indicate fuel requirement variations from +500 lb to a worst-case of +1,300 lb.

Estimated airbreather system weights are summarized in Fig. 3.3-15. For the same number of engines, the P&W and GE systems are within 1,800 lb; to the advantage of the P&W system. The exception is in the four-engine arrangement where the GE system is approximately 300 lb lighter. From the standpoint of payload effects, the selection of the P&W or the GE system is of little significance. For any given system, as the engine number is reduced, however, weight savings up to 4,000 lb are evidenced. This latter effect is felt to be of sufficient importance to attempt to maintain a minimum engine configuration.

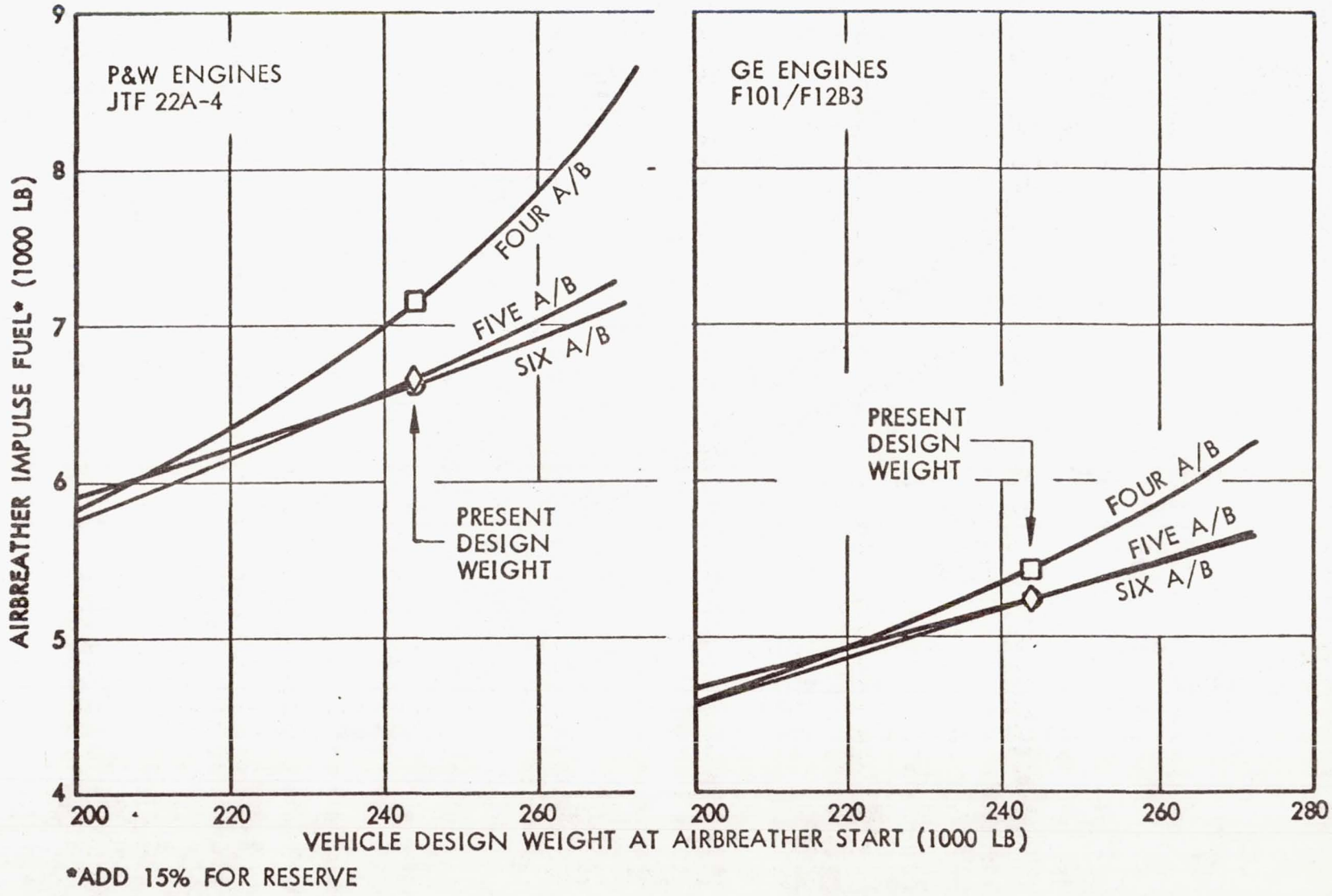


Fig. 3.3-14 FUEL REQUIRED – STANDARD DAY

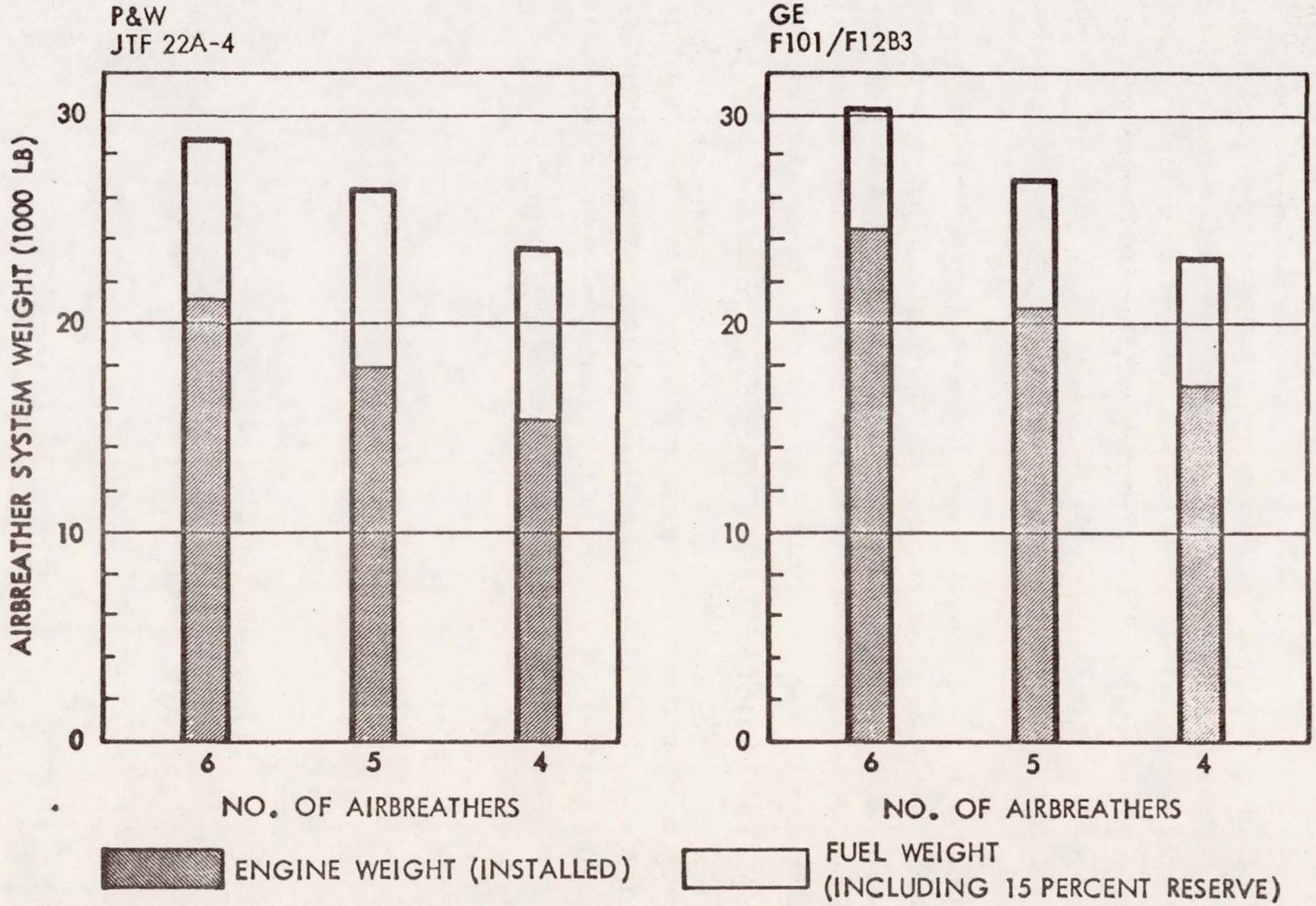


FIG. 3.3-15 VEHICLE DESIGN WEIGHT AT A/B START = 244,000 LB

FAA climb gradient and level-flight capability are summarized in Figs. 3.3-16 through 3.3-18. As in the case of the stage-and-one-half orbiter, the engine-out condition is the critical parameter. The GE system, by virtue of its superior performance, appears as the better solution since, even in the engine-out mode, level-flight capability of 1,000 ft in excess of the worst reference airport condition is possible for the four-engine configuration.

The combination of a significant weight savings and even number of engines required for base integration promotes the four-engine GE arrangement as the better two-stage orbiter airbreather system. However, acceptance of this precludes the capability of the FAA engine out climb gradient requirement. Discussions with the FAA to negotiate for engine out level-flight capability with 1,000 ft airport-altitude clearance is recommended.

The representative airbreather terminal profile, utilizing four GE F101/F2B3 engines is shown in Fig. 3.3-19 (a & b). Total airbreather time from 40,000 ft through go-around and touchdown in 1160 sec; a factor accounted for in the thermal protection system weight.

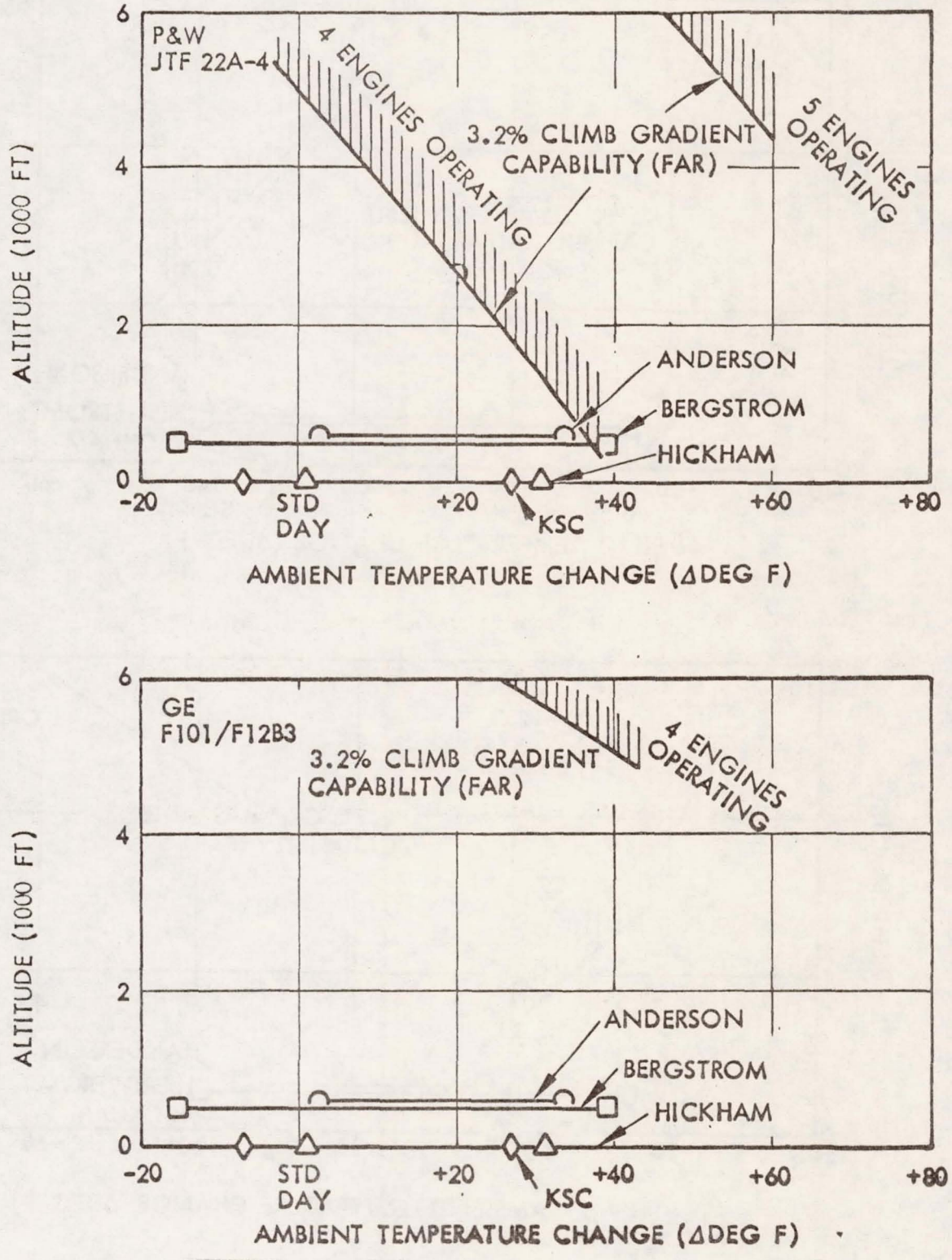
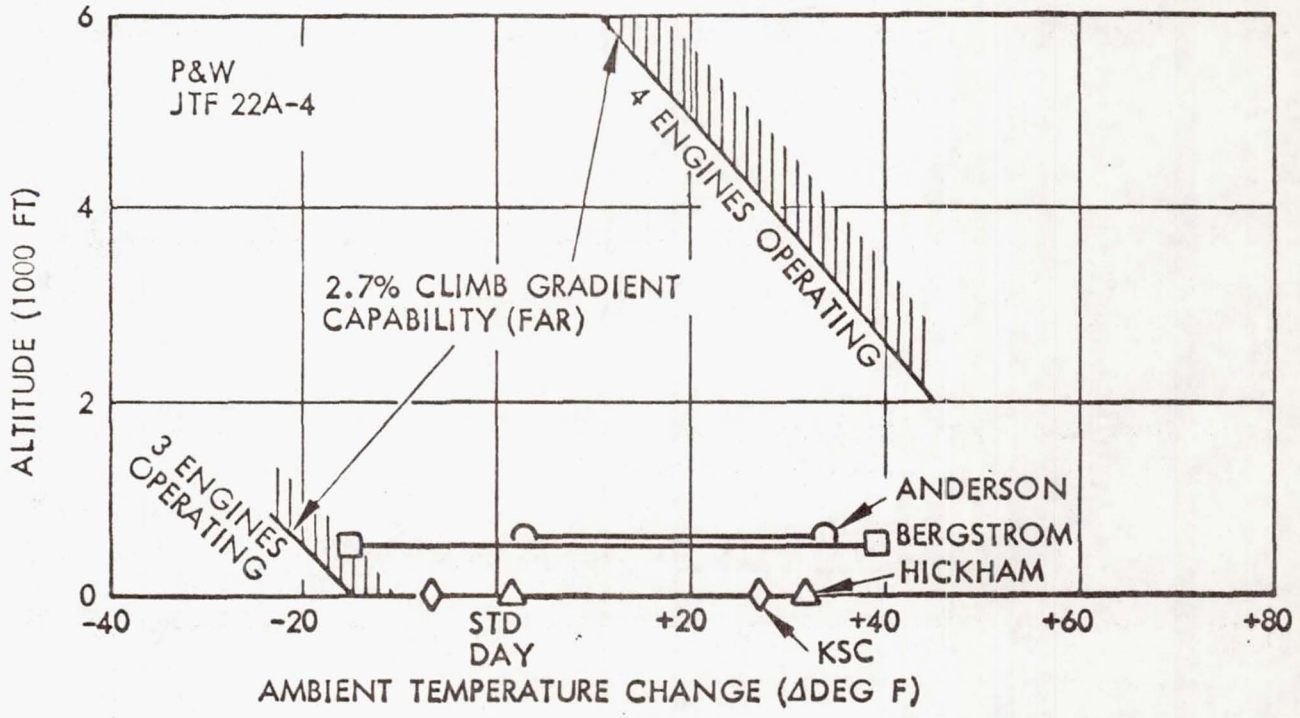


Fig. 3.3-16 2-Stage Orbiter Climb Summary - Landing Configuration



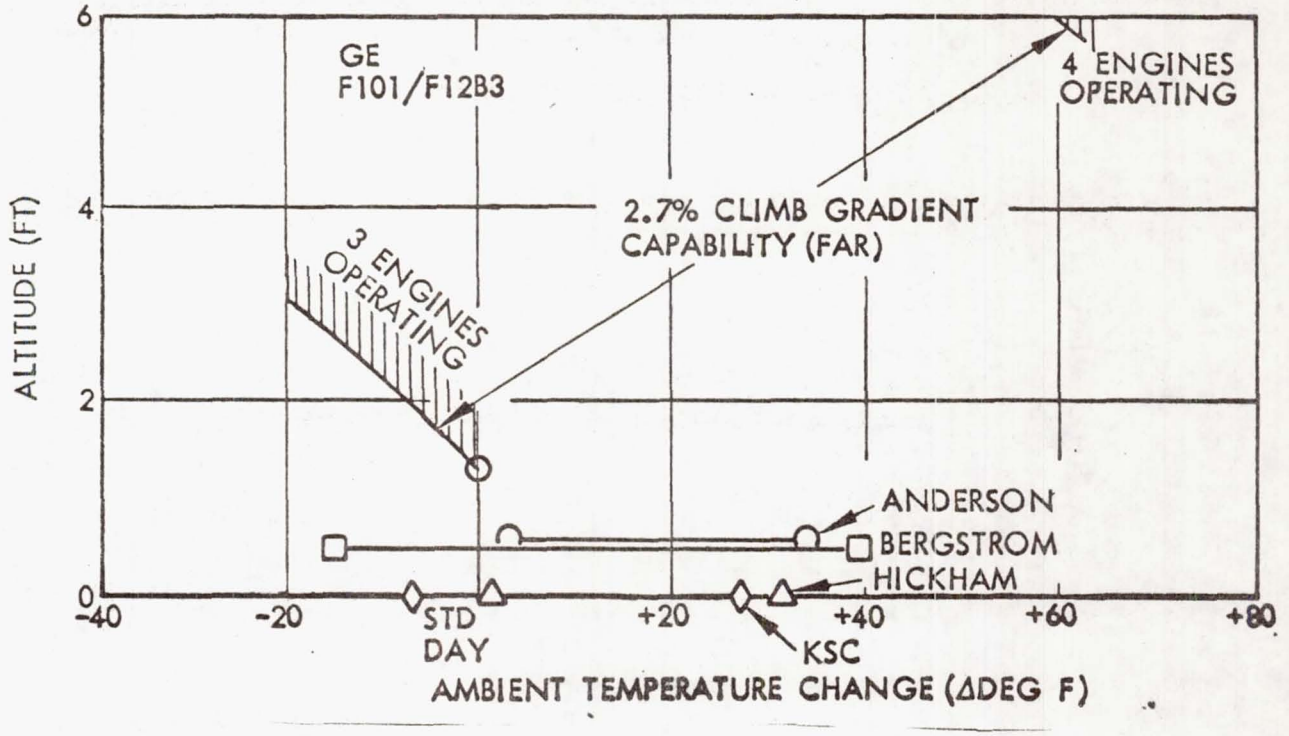


Fig. 3.3-17 2-Stage Orbiter Climb Summary - Approach Configuration

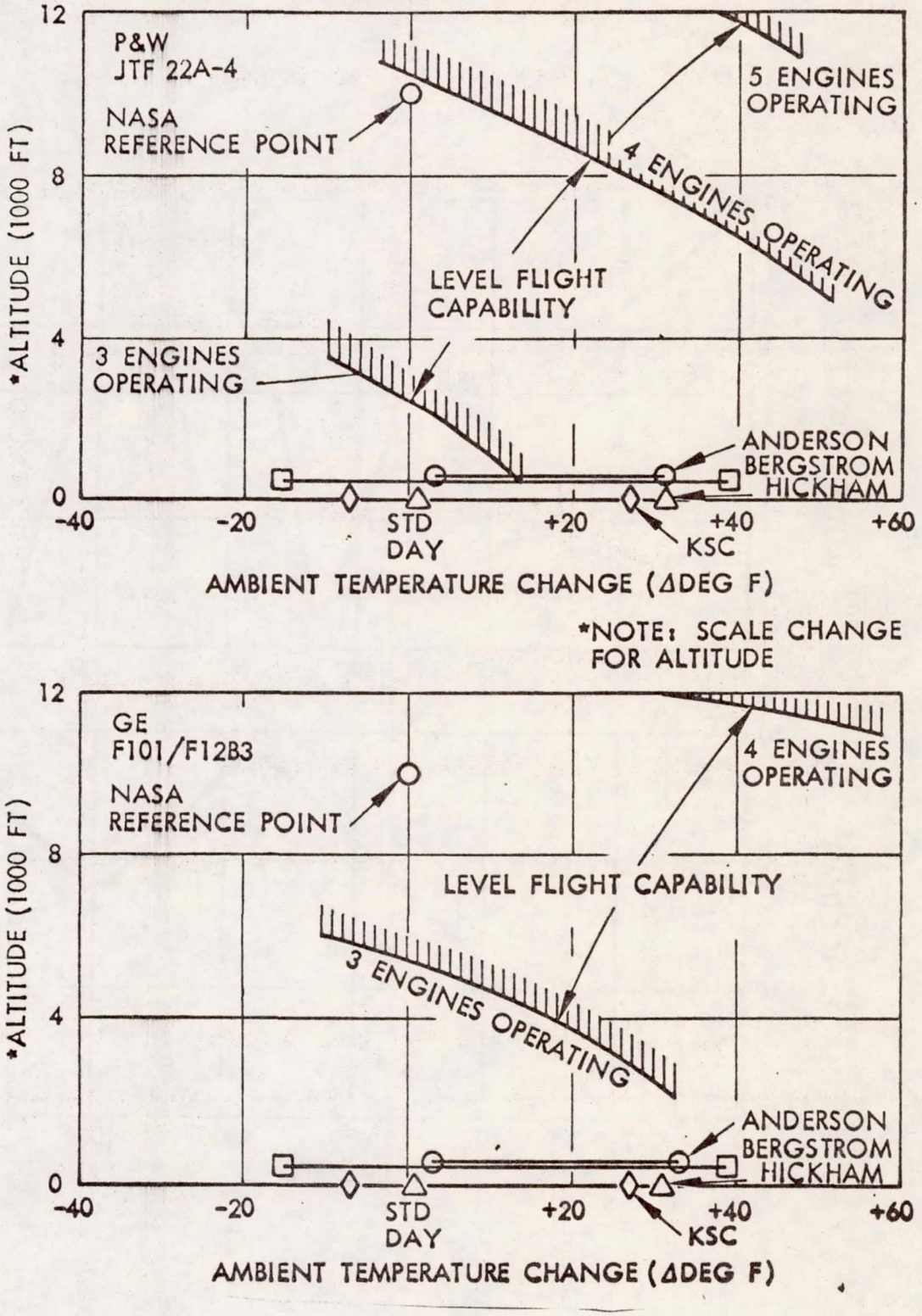


Fig. 3.3-18 2-Stage Orbiter Level Flight Ceiling Cruise Configuration

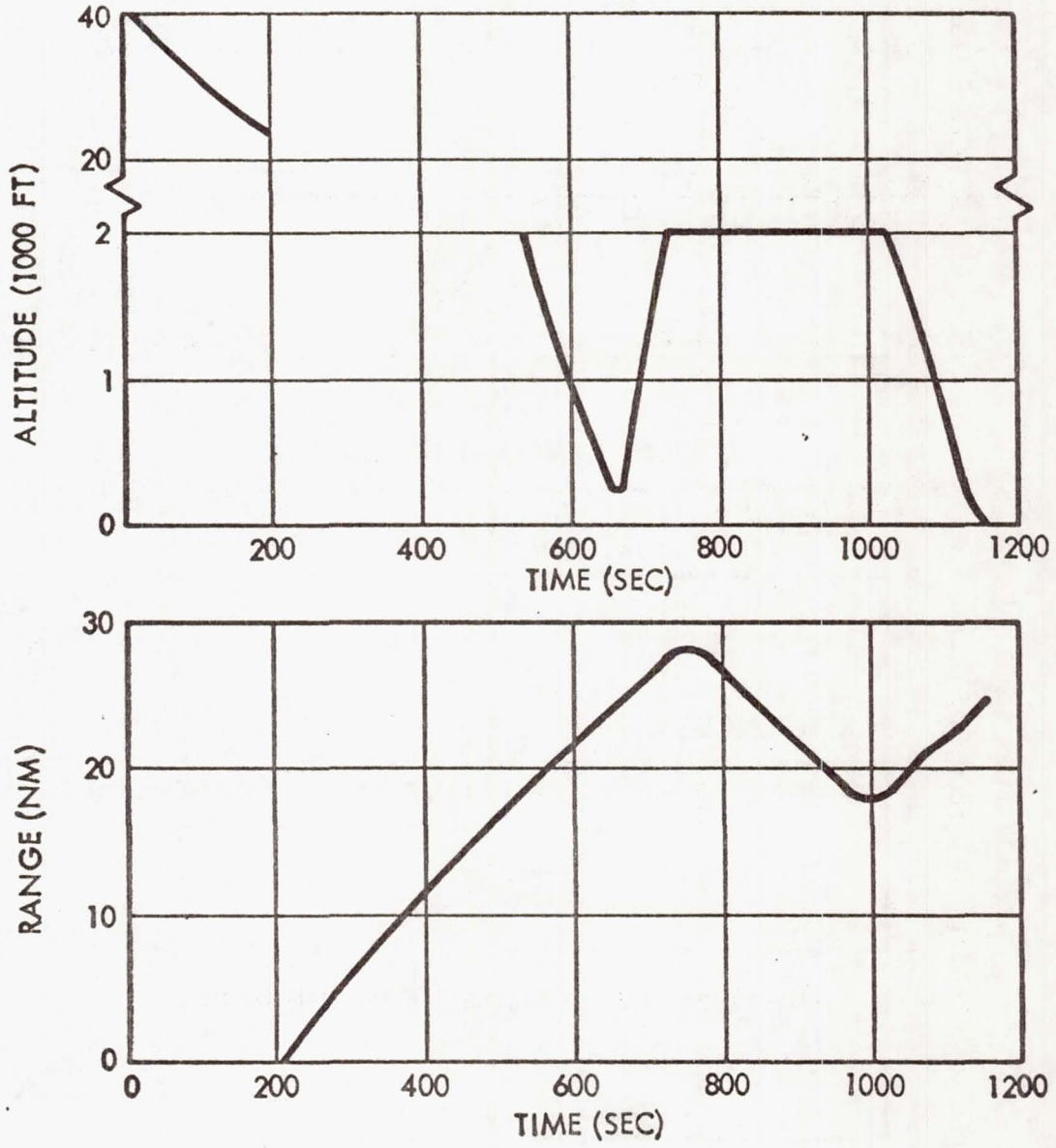


Fig. 3.3-19(a) Two-Stage Orbiter Terminal Airbreather Profile

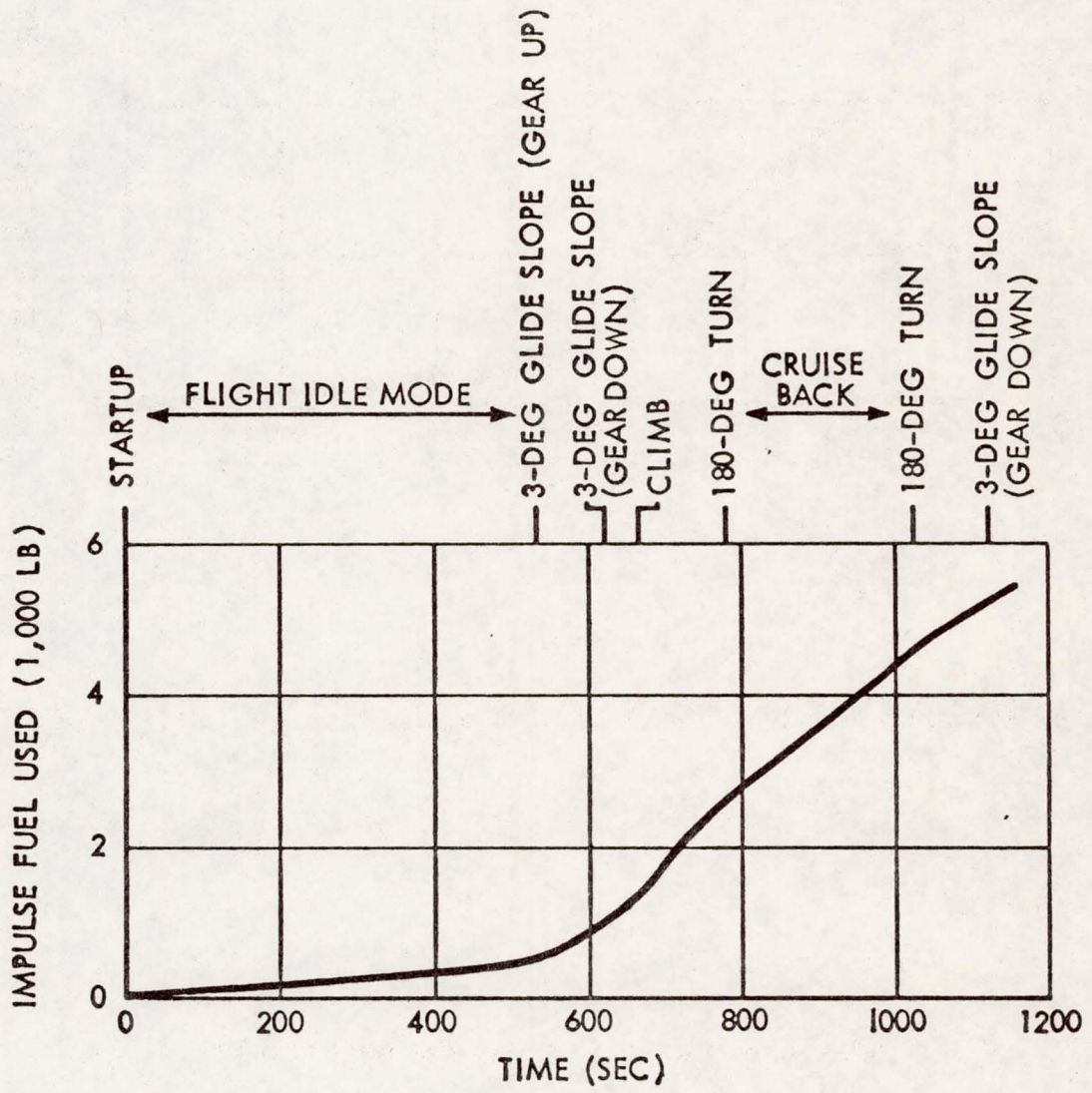


Fig. 3.3-19(b) Two-Stage Orbiter Terminal Airbreather Profile

3.3.6 Reentry

The selection of the delta-body orbiter for use in the two-stage reusable system analysis results primarily in the study of the effects of lower wing loadings on the entry profile. The aerodynamic characteristics and the temperature boundary data established for the stage-and-one-half orbiter are retained for study. In this section, the development of the two-stage orbiter nominal entry trajectory is reviewed. Also discussed is an additional trade-off study promoted by the lower wing loading characteristics of the two-stage orbiter on the effects of lower temperature trajectories on crossrange.

3.3.6.1 Nominal Reentry Trajectory. A nominal reentry trajectory for the two-stage vehicle was obtained for sizing the TPS. Essentially the same criteria was followed in developing the trajectory as were described in Section 2.4.5 for the stage-and-one-half system. The control history that was determined for the two-stage descent was also similar to that of the stage-and-one-half. Differences between the vehicles in the control histories and the resulting trajectories are attributed primarily to the different wing loadings and their effect on the bank angle crossrange sensitivity. Figs. 3.3-20 through 3.3-25 present the trajectory data for the two-stage vehicle. The trajectory parameters shown are identical to those described previously for the stage-and-one-half.

3.3.6.2 Lower Temperature Trajectories. An additional study was made of the two-stage orbiter to determine the effects of requiring a cooler descent on crossrange capability. To accomplish the objective, nominal temperature boundaries described by altitude and velocity were biased by increasing increments of altitude. Since the study was intended as a trade analysis, the constraints on angle-of-attack were relaxed during the final control phase. The control history logic for this modified descent is shown compared with nominal values in Table 3.3-6.

Table 3.3-6

CONTROL HISTORIES FOR TWO-STAGE ORBITER

TYPE	PHASE	ANGLE OF ATTACK, (DEG)	BANK ANGLE, (VALUE/CONDITION)	END CONDITION
Nominal	1	32	0	T.B. pullup, = 0.07 deg
	2	32	On T.B.	V = 16,000 fps
	3	32 → 25	Constant	H ≃ 100 deg
	4	25 → 15	Constant	= 0
	5	15 → 10	0	V = 0
Cooler	1	32	0	
Higher Temperature	2	32	On T.B.	V = 16,000 fps
Boundary	3	32 → 25	Constant	H ≃ 100 deg
	4	25	Constant	= 0
	5	25	0	V = 0

Definitions:

α = entry angle-of-attack

γ = flight path angle

V = Velocity

H = Heading change relative to orbit plane

T.B. = Temperature boundary

→ Modulated to

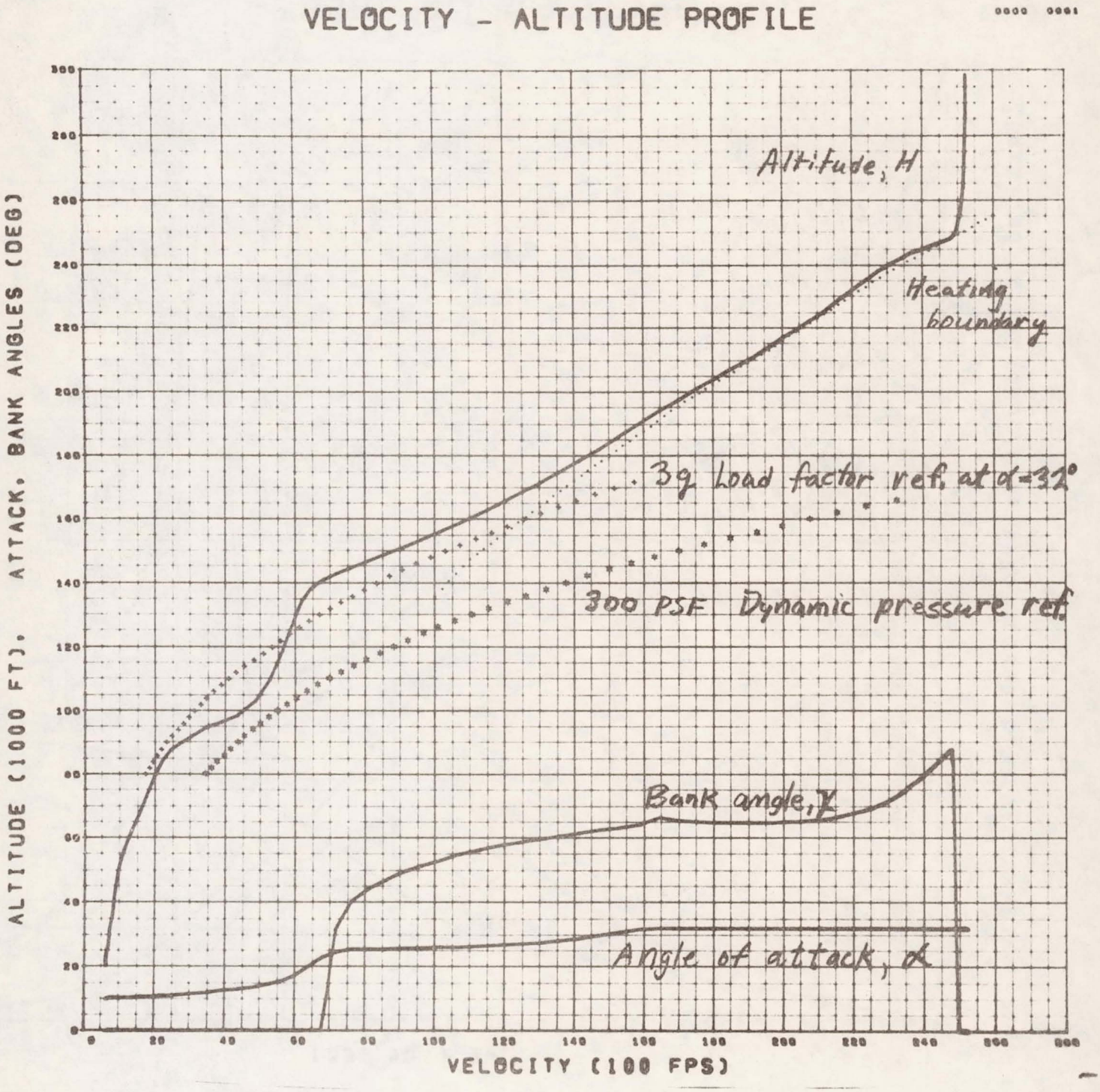


Fig. 3.3-20 Two-Stage Orbiter H-V Profile

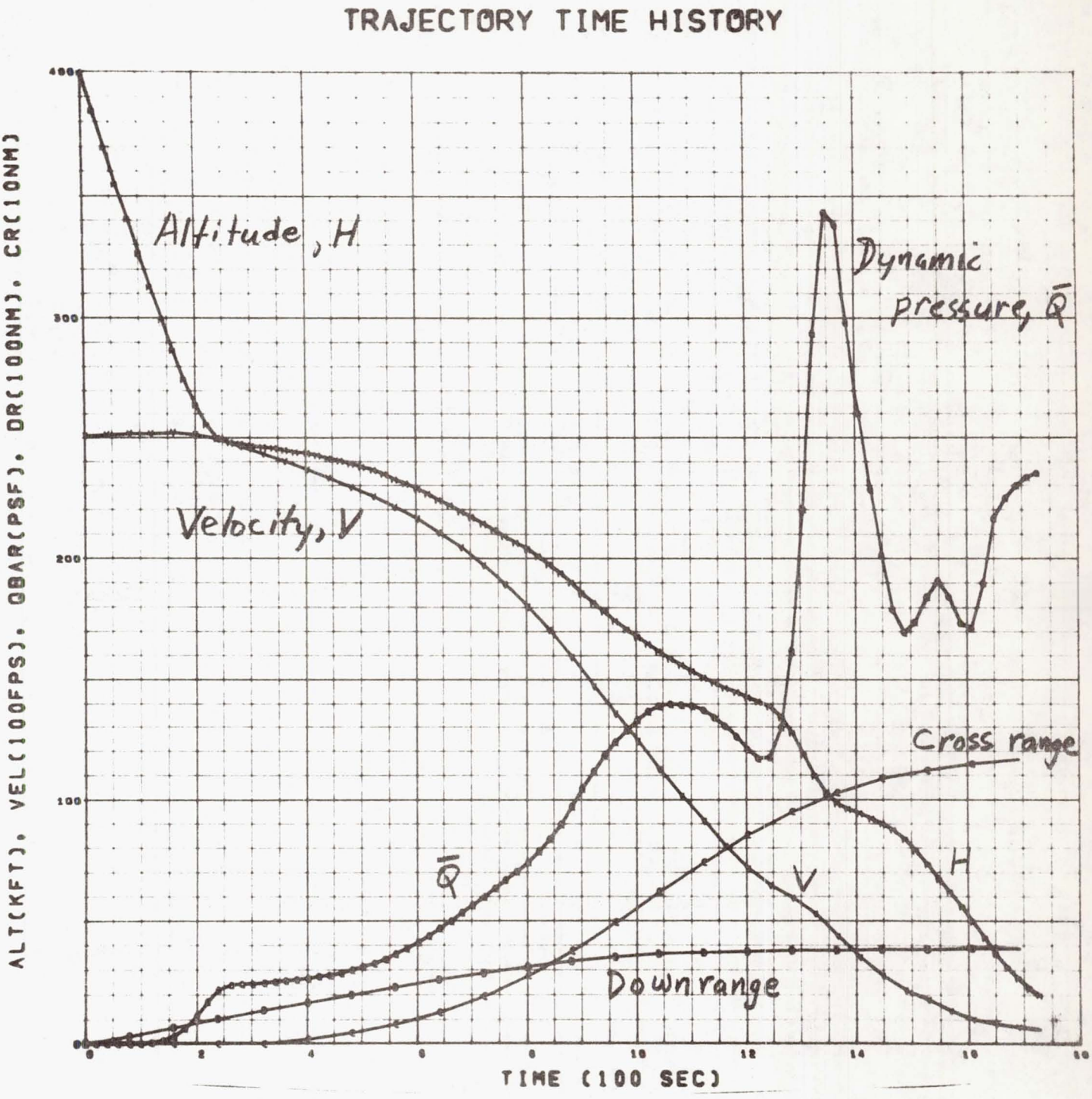


Fig. 3.3-21 Two-Stage Orbiter, H, V, $\bar{Q}$ History

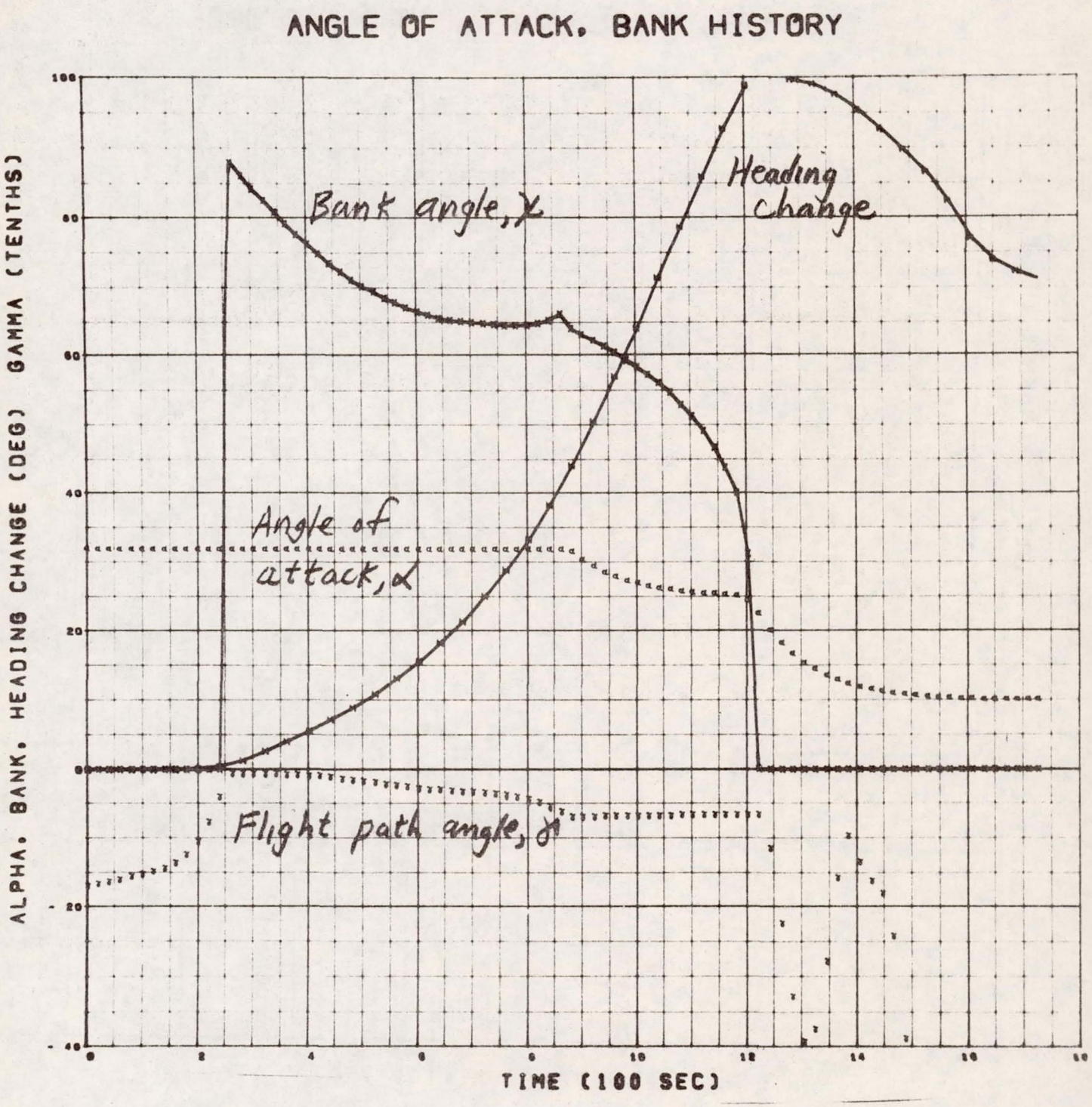


Fig. 3.3-22 Two-Stage Orbiter α , χ , γ History

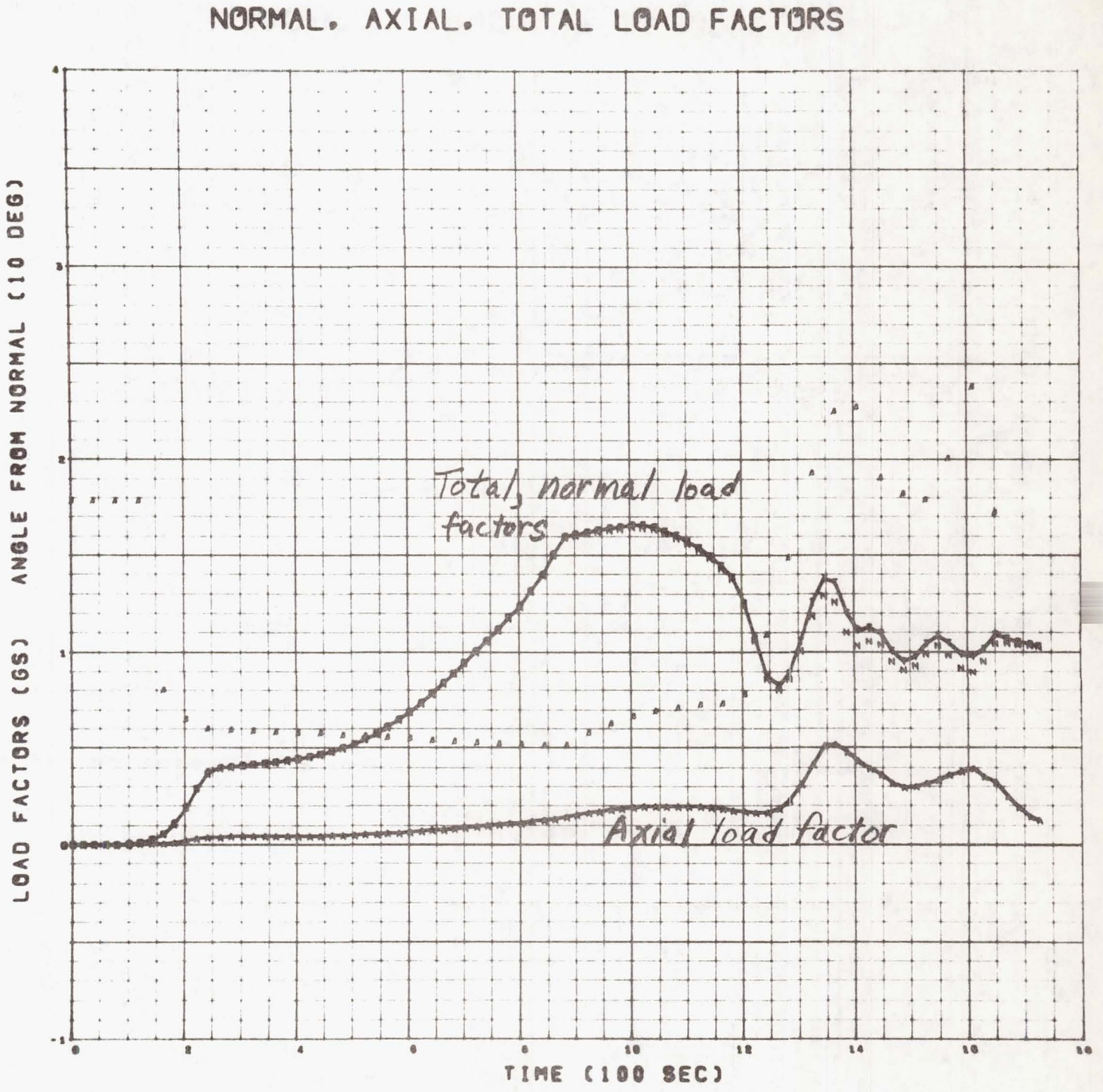


Fig. 3.3-24 Two-Stage Orbiter Load Factor

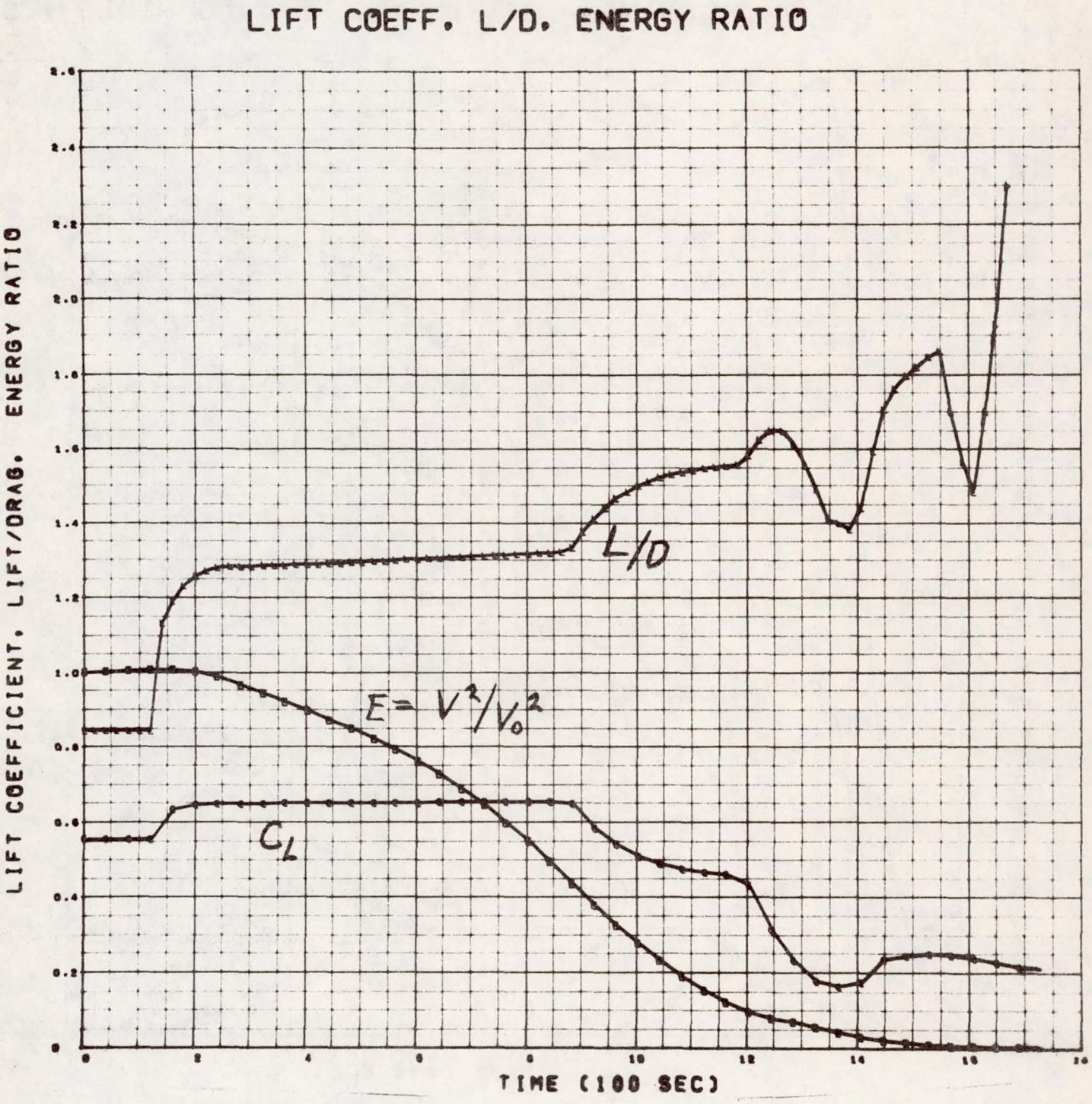


Fig. 2.2-24 Two-Stage Orbiter Lift Coefficient

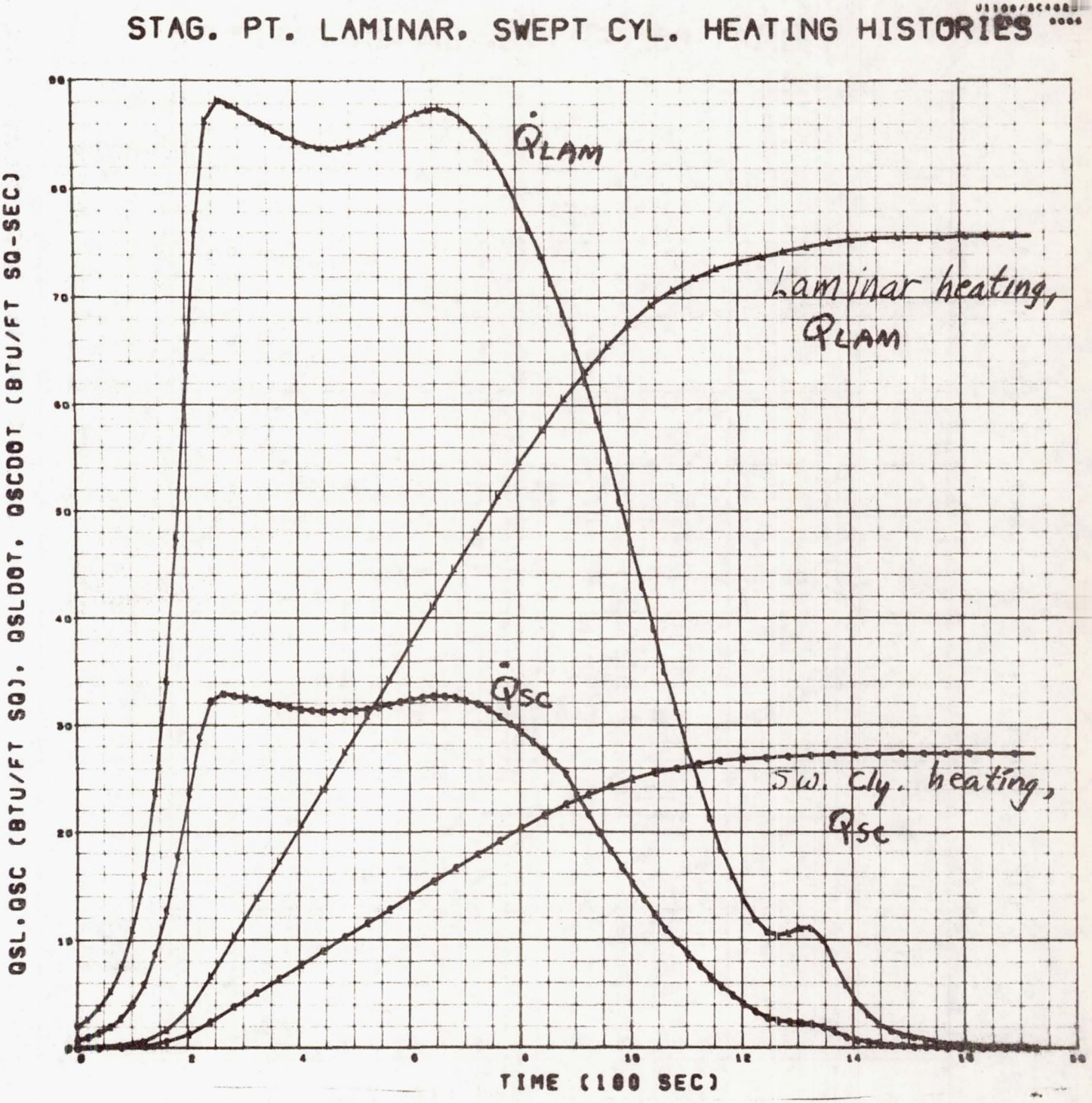


Fig. 2.2-25 Two-Stage Orbiter Heating History

Figure 3.3-26 summarizes the results of the above study. Crossrange and time from entry to landing are shown as functions of an altitude increment Δ H. The Δ H is the altitude increase imposed on the temperature boundary cruve for a given trajectory. In effect, it raises the altitude of the pullup point and that portion of the trajectory along the temperature boundary by the constant amount Δ H. Note that increasing the pullup altitude also results in a shallower entry flight path angle.

Figure 3.3-26 shows first that the modified control history provided a small (30 nm) increase in crossrange at the nominal temperature altitude (Δ H = 0). The crossrange then increases to a maximum of about 120 nm over the nominal at a Δ H of 8,000 ft. Increasing Δ H further decreases the crossrange until it drops below the design requirement of 1100 nm at a Δ H of about 16,400 ft. When this data was reviewed by the thermodynamics group, a temperature of 1800 deg was developed. Thus, the two-stage nominal design can accommodate a significantly cooler trajectory while still providing adequate crossrange capability.

The entry times obtained for these trajectories indicated that flying a 16,000 ft higher temperature boundary curve will require approximately 50 percent longer entry time. However, these are not minimal times, and it should be possible to shorten them by further modifications in the control history if the lower temperature is to be a TPS design criterion.

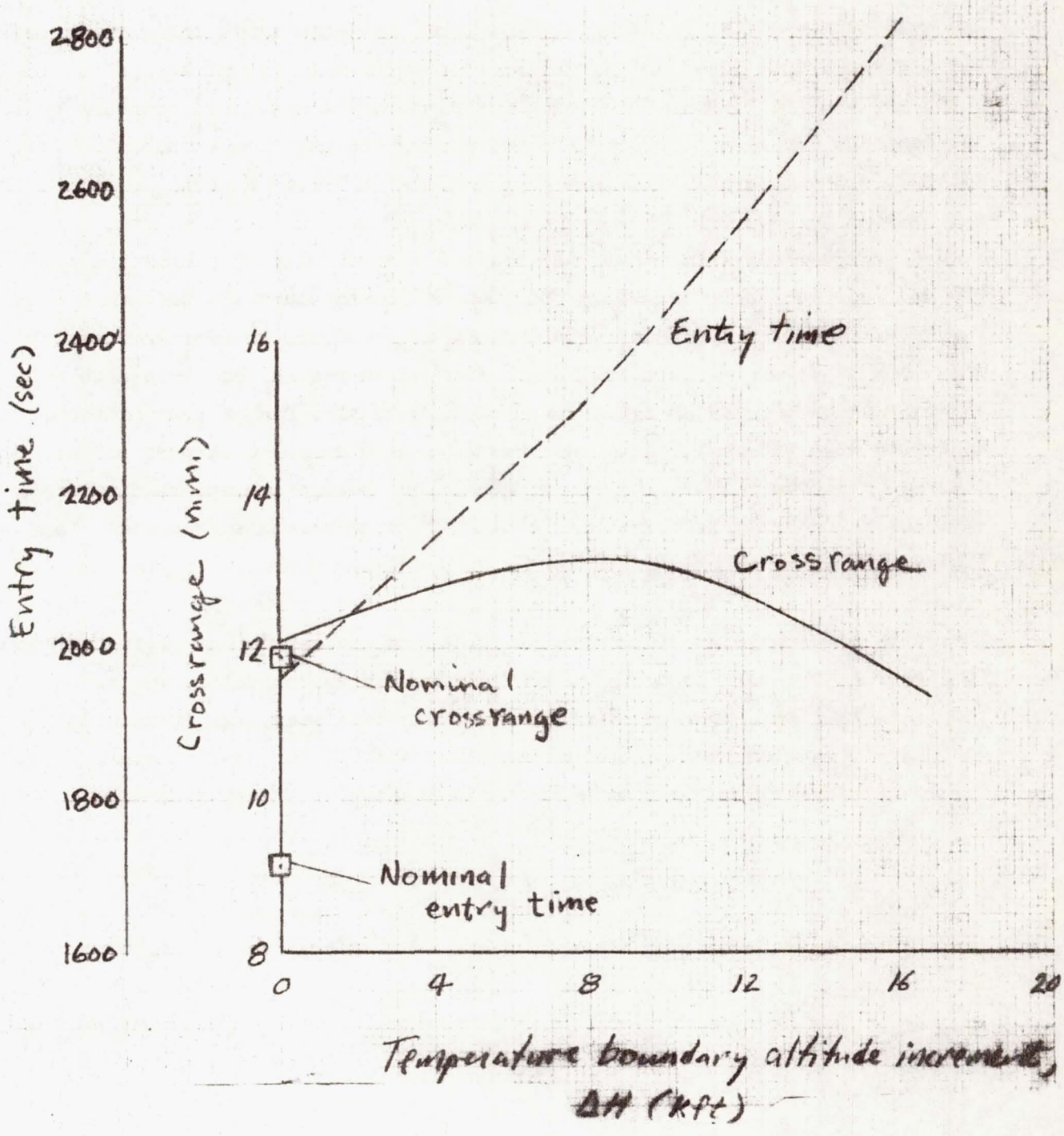


Fig. 3.3-26 Sensitivity of Two-Stage Orbiter Entry Trajectory to Temperature Boundary Altitude Increase – Effects on Crossrange and Entry Time

3.3.7 System Sensitivities

Performance and design sensitivities for the two-stage system are useful for design/cost tradeoffs, for evaluation of risk in program development due to design and engine performance uncertainties, and for calculation of flight performance reserves. Two types of sensitivities are presented in this section: variable gross liftoff weight (GLOW) and variable payload. The variable liftoff weight sensitivities assume a variable booster size and fixed orbiter size and payload. This type of sensitivity is done for the critical mission, 40,000 lb payload to a polar orbit with an orbiter engine out. It is useful for tradeoffs early in the design cycle when the principle features of the orbiter design and booster are already established and only relatively minor modifications are made. The booster propellant load and airframe size are allowed to change, however. For the second type of sensitivity, variable payload, the entire vehicle design is assumed to be fixed, and any performance or weight change affects the payload capability of the vehicle. This type of sensitivity may be used to compute flight performance reserves or changes in payload capability because of different mission requirements. A note of caution is advised on the use of these partials. They are very non-linear, and should be used for relatively small changes in the parameters of interest. It is also important that the assumptions relating to their use be read and observed.

Variable Liftoff Weight Sensitivities. The variable liftoff weight sensitivity for the critical mission assumes that only the booster is rubberized, and that the payload, the orbiter airframe size and propellant loading, and the main engine number and thrust level on both booster and orbiter are all fixed. Rubberizing the booster changes the launch weight, and thus the liftoff thrust-to-weight ratio $(T/W)_o$, since the sea level thrust is fixed at 550,000 lb per engine.

As the liftoff thrust-to-weight ratio is reduced, the velocity losses to achieve orbit injection increase, so that the ideal velocity required from

the booster must increase. The velocity losses also increase if the staging velocity or orbiter ignition thrust-to-weight ratio decreases, so that the total ideal velocity required increases.

The basic booster weight scaling equations used for the variable liftoff weight sensitivities are derived by a combination of analytical and empirical techniques, and are based on the McDonnell-Douglas MP-8 booster. These weight scaling equations include the effect of propellant load on volume and surface area, the effect of staging velocity on booster thermal protection system weight and booster flyback range and fuel, and the effect of orbiter gross weight on booster-orbiter attach structure. The increase in booster volume because of increases in propellant loading is reflected in many weight items, both directly through surface area and indirectly as a function of other weights. It contributes primarily to body structure, thermal protection system, aerodynamic surfaces and landing gear. The weights of the main engines are fixed. For further details on the rubberized booster weight scaling laws, see Section 3.2.2.

Table 3.3.7-1 shows the effect of weight and performance changes in gross liftoff weight (GLOW), booster inert weight (separation weight) and dry weight, and orbiter inert weight (post-retro less payload) and dry weight for the polar mission with ABES out and engine-out abort to orbit capability. Both ascent and return payload are fixed at 40,000 lb. This mission is the critical mission for the two-stage system.

One sensitivity that requires additional clarification is that of weight of cargo landed. No airbreathing engines are required for the polar mission, so there are no changes in ABES engine or fuel weight. It is assumed that the orbiter thermal protection system weight, landing gear and associated frames, and entry ACPS propellant are changed as the weight of cargo returned changes. However, the return cargo is not carried through any impulsive ΔV burn. Note that the sensitivity of orbiter inert weight to return cargo is used in the

calculation of several other partials, since any direct inert weight added to the orbiter requires a secondary increase in orbiter inert weight, so that the direct inert weight can be returned. For example, if one additional pound of weight is required for the crew cabin, an additional 0.138 lb of weight is required in landing gear, thermal protection system, and other miscellaneous systems. Contingency is not applied to the direct pound of weight added, but it is applied to the secondary weight increases.

The sensitivity of the two-stage system to booster inert weight is treated similarly. If 1 pound is added to the booster inert weight, and no additional flyback fuel, other residuals, or dry weight is required, then the sensitivity to booster inert weight with no secondary effect should be used. This might be the case, for example, if the weight was jettisoned at staging and did not have to be carried through booster reentry, cruiseback and landing. This sensitivity may also be used if the total weight added is known through a separate study. The second type of sensitivity to booster inert weight, with the secondary effects accounted for, represents adding 1 lb to the booster and increasing the weight of other systems on the booster to carry back this pound. This requires more flyback fuel, larger reserves, a larger wing (wing loading is held constant), heavier landing gear, and various other indirect effects. Notice that the sensitivity accounting for secondary effects is nearly twice as large as the sensitivity not accounting for them. It is considered the more realistic sensitivity, and the one that should be used for design/cost tradeoffs on the booster.

The sensitivity of the system to orbiter propellant loading is given for three cases: (1) No orbiter tankage weight change reflected by changing the propellant, (2) Internal non-load carrying tank weight changes only, assuming volume changes occur within a fixed airframe, and (3) Airframe changes plus internal tank changes to accommodate propellant volume changes. As the airframe size changes, the structural thermal protection system and aerodynamic surface weights change, as well as other miscellaneous systems. The general method of computing the propellant sensitivities including a weight effect is illustrated below. The sensitivity of GLOW to propellant weight, assuming the airframe changes, is given as an example:

$$\frac{d\ GLOW}{d\ W_{P_2}} = \left(\frac{\partial\ GLOW}{\partial W_{I_2}}\right) + \frac{\partial GLOW}{\partial W_{I_2}} \cdot \frac{dW_{I_2}}{dW_{P_2}} = -4.98 + 36.6 \times .130$$

$$= -4.98 + 4.75 = -.23$$

Notice that the sensitivity of GLOW to orbiter inert weight does not include the secondary effect, since $\frac{dW_{I_2}}{dW_{P_2}}$ already accounts for this. The meaning of this sensitivity is that if more propellant is added to the orbiter, GLOW decreases, so that the minimum GLOW staging velocity must be less than the staging for the nominal system. However, the actual decrease in GLOW would be very small, and because it represents a small difference between large numbers, is considered negligible.

Total thrust is handled in a similar way for the orbiter and booster. Sensitivities both with and without weight changes are considered. When thrust is increased without a resulting weight change, the thrust-to-weight ratio increases and velocity losses decrease, resulting in a smaller system. If increased engine weight is added as the thrust increases, there is a tradeoff between engine weight and thrust. This sensitivity is determined by a combination of the sensitivity to thrust without weight, and to weight without thrust, as shown below, for the sensitivity of GLOW to orbiter booster thrust:

$$\frac{d\ GLOW}{dT_1} = \left(\frac{\partial GLOW}{\partial T_1}\right)_{W_{I_1}} + \frac{\partial GLOW}{\partial W_{I_1}} \frac{dW_{I_1}}{dT_i} = -.401 + 9.39 \times (.015) = -.260$$

Note that the sensitivity of booster inert weight to sea-level thrust (fixed number of engines) is .015 lb/lb, representing engine weight, thrust structure and increased feed lines. The sensitivity of GLOW to booster inert accounts for secondary effects, since the .015 factor does not. For the orbiter, the corresponding direct ratio of orbiter inert weight to thrust is .021 lb/lb. However, since the critical mission is for one orbiter engine out during abort,

one pound of thrust added to the orbiter requires a weight increase in both engines, for a total of 2 x .021 = .042 lb. Since this is a direct ratio, and does not include secondary effects, the sensitivity of GLOW to inert weight including secondary effects is used.

$$\frac{d\ GLOW}{dT} = \left(\frac{\partial GLOW}{\partial T}\right)_{W_{I_2}} + \frac{\partial GLOW}{\partial W_{I_2}} \frac{dW_{I_2}}{dT_2} = -1.85 + 41.7 \times .042 = -.10\ lb/lb$$

The liftoff weight decreases as orbiter thrust increases, even with an engine out.

Other interesting results using these sensitivities may be obtained. For example, if the engine powerhead is increased in size, then both the orbiter and booster thrust increase, resulting in a liftoff weight decrease but a net dry weight increase. This is shown in the calculations below, where T_1 is the total sea-level thrust of the booster, T_2 is the total vacuum thrust of the orbiter, and w is the propellant flow rate of a single engine powerhead.

$$T_1 = \dot{w} \times I_{s_p} \times \text{No. booster engines on} \times \text{throttle ratio}$$

$$\frac{dT_1}{\dot{dw}} = 439 \times 11 \times 1 = 4830$$

$$T_2 = \dot{w} \times I_{sp_V} \times \text{No. orbiter engines on} \times \text{throttle ratio}$$

$$\frac{dT_2}{\dot{dw}} = 459 \times 1 \times 1.09 = 500$$

$$\frac{dGLOW}{\dot{dw}} = \frac{dGLOW}{dT_1} \frac{dT_1}{\dot{dw}} + \frac{dGLOW}{dT_2} \frac{dT_2}{\dot{dw}} = -.260 \times 4830 - .10 \times 500 = -1310\ lb/lb/sec$$

$$\frac{dW_{DRY_1}}{\dot{dw}} = \frac{dW_{DRY_1}}{dT_1}\frac{dT_1}{\dot{dw}} + \frac{dW_{DRY_1}}{dT_2}\frac{dT_2}{\dot{dw}} = .005 \times 4830 - .008 \times 500 = 20.2 \text{ lb/lb-sec}$$

$$\frac{dW_{DRY_2}}{\dot{dw}} = \frac{dW_{DRY_2}}{dT_1}\frac{dT_1}{\dot{dw}} + \frac{dW_{DRY_2}}{dT_2}\frac{dT_2}{\dot{dw}} = 0 \times 4830 + .048 \times 500$$

$$= 24.0 \text{ lb/lb-sec}$$

$$\frac{dW_{Total\ Dry}}{\dot{dw}} = \frac{dW_{DRY_1}}{\dot{dw}} + \frac{dW_{DRY_2}}{\dot{dw}} = 44.2 \text{ lb/lb-sec}$$

Thus if the main engine propellant flow rate is increased in size, GLOW will decrease but total dry weight will increase mostly because of the extra engine weight. The net effect will be an increase in cost.

Variable Payload Sensitivities. The variable payload sensitivities for a fixed two-stage system (LS 400-7A orbiter with the scaled MP-8 booster described in Section 3.10) are presented for the due east and polar abort missions (engine out) and nominal resupply mission in Table 3.3.7.2. Sensitivities for the abort case, rather than the nominal (both engines operating) are used for the polar and due east missions since it is the abort requirements that determine the payload that can be carried for the nominal mission. These sensitivities assume that propellant tank capacities and dry weights are fixed, as well as the thrust of the engines. The only major weights which vary are the ascent payload, and the flyback cruise fuel, which varies with staging velocity and booster flyback weight. It is assumed that changing the return payload has no effect on the orbiter dry weight. The return payload does affect the ACPS reentry propellant and airbreathing engine fuel. The variable payload sensitivities apply to a fixed vehicle, where no design changes in the vehicle are permitted, but propellant loadings for some items may vary from mission to mission.

In deriving the sensitivity of payload to ascent ΔV, it is assumed that the orbiter and booster ascent tanks are fully loaded and the propellant load is not changed as the ΔV required varies. For the polar and due east abort missions, the OMPS tanks are assumed fully loaded in deriving the sensitivity of payload to on-orbit ΔV. The on-orbit ΔV is defined as that required during an engine-out abort, for retro and phasing purposes, rather than the nominal mission, where excess on-orbit ΔV is available. For the resupply mission, as the on-orbit ΔV requirement changes, the propellant in the OMPS tank also changes, since excess volume is available. This accounts for the decreased sensitivity of payload to on-orbit ΔV for the resupply mission.

The sensitivity of payload to booster inert weight may be expressed in either of two ways: (1) With no change in booster flyback fuel as the booster inert weight changes (no secondary effect) and (2) flyback fuel changes with booster inert weight (with secondary effect). The first would be applied in a situation where the total gross change in booster inert weight is known (including the effects on booster flyback fuel). The second should be used for all other cases.

The other sensitivities are rather straightforward. Sensitivities of propellant, specific impulse, and thrust are derived assuming no dry weights change. The only inert weight that changes, other than the payload, is the booster flyback fuel, which changes as the ascent trajectory is altered. Note that the payload in the polar and due east abort missions is much more sensitive to orbiter thrust, since only one engine is operating, than for the resupply mission.

Table 3.3.7-1

TWO-STAGE VARIABLE LIFTOFF WEIGHT SENSITIVITIES
CRITICAL MISSION – ENGINE OUT POLAR ABORT (ABES OUT)

Parameter	$\frac{\partial GLOW}{\partial Parameter}$	$\frac{\partial Booster\ Inert}{\partial Parameter}$	$\frac{\partial Booster\ Dry}{\partial Parameter}$	$\frac{\partial Orbiter\ Inert}{\partial Parameter}$	$\frac{\partial Orbiter\ Dry}{\partial Parameter}$
Payload, lb					
(1) Delivered and returned	41.7	4.65	2.33	.139	.138
(2) Returned only (not carried through ΔV burn)	5.09	.569	.285	.139	.138
On-Orbit ΔV (fps)	640	75.4	36.9	0	0
Ascent ΔV (fps)	633	74.7	36.6	0	0
Orbiter Inert Weight, lb					
(1) No secondary effect	36.6	4.09	2.05	1.00	1.00
(2) With secondary effect	41.7	4.65	2.33	1.139	1.138
Booster Inert Weight, lb					
(1) No secondary effect	5.10	1.38	1.22	0	0
(2) With secondary effect	9.39	2.53	1.84	0	0
Orbiter Propellant, lb					
(1) No orbiter tankage penalty	-4.98	-.792	-.352	+.001	0
(2) Internal tank penalty only	-2.79	-.546	-.229	.061	.060
(3) Airframe and tank enlarged	-.23	-.260	-.085	.132	.130
Orbiter Specific Impulse, sec (Thrust Constant)	-24,500	-3,140	-1,370	0	0
Booster Specific Impulse, sec	-19,200	-1,720	-1,030	0	0
Total Orbiter Vacuum Thrust, lb (I_{sp} Constant)					
(1) No engine weight penalty	-1.85	-.218	-.107	0	0
(2) With engine weight penalty	-.10	-.022	-.008	.049	.048
Total Booster Sea Level Thrust, lb (I_{sp} Constant)					
(1) No engine weight penalty	-.401	-.047	-.023	0	0
(2) With engine weight penalty	-.260	-.009	+.005	0	0

Table 3.3.7-2

TWO-STAGE VARIABLE PAYLOAD SENSITIVITIES

Parameter	Polar (Abort) ABES OUT	Due East (Abort) ABES OUT	Resupply (Nominal) ABES IN
On-Orbit ΔV (fps)	-19.7	-23.2	-17.0
Ascent ΔV (fps)	-19.6	-23.0	-21.9
Orbiter Inert Weight (lb)	-1.	-1.	-1.
Booster Inert Weight (lb)			
(1) No secondary effect	-.145	-.167	-.148
(2) With secondary effect	-.170	-.196	-.173
Orbiter Propellant (lb)	.185	.204	.230
Booster Propellant (lb)	.0472	.0558	.0528
Orbiter Specific Impulse (sec) (Thrust Constant)	769	829	748
Booster Specific Impulse (sec) (Thrust Constant)	632	741	664
Orbiter Thrust (lb) (Specific Impulse Constant)	.0578	.0677	.0103
Booster Thrust (lb) (Specific Impulse Constant)	.0130	.0143	.0104
Orbiter Thrust-to-Weight Ratio (.1 Unit T/W)	4540	5530	880
Booster Thrust-to-Weight Ratio (.1 Unit T/W)	4990	6050	4410

3.4 SYSTEM DESIGN

A conventional two-stage system was derived and finally designated as the Design Model LS 400-7A. The initial design work leading to the final configuration was initiated during the growth to two-stage task where baseline two-stage (LS 400-6) and stage-and-one-half (LS 200-7) orbiters were established for comparison to a converted two-stage orbiter (conversion from a convertible stage-and-one-half orbiter). The following drawings associated with this task and related to the generation of a conventional two-stage system describe the Model LS 400-6 arrangement:

Fig. 3.4-1 General Arrangement (2-B and 2-C) Delta Body - Two-Stage Model LS 400-5 and -6, SKS-100040

Fig. 3.4-2 Launch Vehicle (2-B) Delta Body - Two-Stage Model LS 400-6, SKT-100044

This baseline two-stage (2-B) orbiter system is shown in Fig. 3.4.1.* The arrangement differs from the baseline stage-and-one-half ($1\frac{1}{2}$-B) system as follows:

a. Aerodynamic Shape and/or Geometry

- Basic shape is the same
- Nose section lower surface incorporates a 3-deg up ramp
- Upper aft surface extends aft to cover and protect the rocket-engine power heads

b. Main Rocket Engine System

- Two engines
- Uses a two-position nozzle system employing a 150:1 area ratio
- Installation is well aft which permits a fixed flap/elevon system

*The drawings show 6 cruise engines installed, while finally their number was reduced to 4.

c. Main Propellant Tankage System

- Arrangement is similar
- Propellant requirements are much greater
- Number of tanks is 7 instead of 6
- Three hydrogen ascent tanks required
- Oxygen ascent tanks are located forward
- Orbital propellant requirements are less
- Load path - adopted for baseline approach similar to stage-and-one-half

d. Jet Engine System

Generally the same except that fuel (JP-4) requirements are less and, therefore, the fuel tank is installed further forward to balance-out the fixed jet-engine installation. If only four jet engines are required, the lower one on each side would be removed and the intake duct lines improved.

e. Landing Gear Installation

- Arrangement - maintained tricycle concept with main gear installed inboard of ascent tankage

f. Control Surface Arrangement

- Same, except upper-surface flaps are installed further aft.

g. Nose Cap System - Same

h. Booster Interface

A three-point booster-to-orbiter attachment system was adopted and located on the lower aft-fuselage surface.

A representative launch vehicle associated with the fully recoverable two-stage (2-B) system is shown in Fig. 3.4.2. A MACDAC high-wing booster arrangement was used with the orbiter located so that its fully loaded center-of-gravity was positioned over the booster's forward attachment station. This composite arrangement resulted in the booster nominal thrust

line operating through an approximate 9-deg angle-of-travel from the initial launch vehicle center-of-gravity condition to the staging center-of-gravity condition. Location of the three-point attachment system is shown with the after center-line attachment designed to resist loads only in the pitch direction.

The final approach to the two-stage vehicle arrangement is illustrated in the following design drawings:

Fig. 3.4-3 Launch Vehicle Delta-Body - Two-Stage Model LS 400-7, SKH 100051(A)

Fig. 3.4-4 Layout - General Arrangement - Delta-Body - Two-Stage Model LS 400-7A, SKS-100049(B)

Fig. 3.4-5 Contour Lines Delta-Body - Two-Stage Model LS 400-7, SKS-100047

Fig. 3.4-6 Inboard Profile Delta-Body Orbiter Model LS 400-7, SKS-100050 (3 sheets)

Fig. 3.4-7 Structural Concept Delta-Body Orbiter - Two-Stage Model LS 400-7, SKG-100120 (2 sheets)

Fig. 3.4-8 Plumbing System Delta-Body Orbiter - Main Propulsion Model LS 400-7, SKT-100224(A)

Fig. 3.4-9 Volume and Wetted Area Curve Delta-Body Orbiter - Model LS 400-7, SKS-100054

Fig. 3.4-10 Alternate Inboard Profile Delta-Body Orbiter - Model LS 400-7, SKS-100060 (2 sheets)

The two-stage launch vehicle arrangement used for the final approach to the fully reusable concept is shown in Fig. 3.4-3. An updated MACDAC booster version with a high-wing, 12-rocket engine system is employed. The LMSC orbiter has been placed so that the forward booster/orbiter attachment is located over the desired booster forward attachment station. The resulting estimated center-of-gravity travel during the boost phase indicates a nominal booster rocket-engine thrust-line excursion of approximately 5 deg.

The structural attachment between the orbiter and booster is accomplished at three points. The forward two attachments resist loads in all directions, including yawing moments as a fore-and-aft force couple, while the aft tie is a two-force member resisting loads only in the pitch plane.

The two-stage orbiter arrangement is illustrated in Fig. 3.4.4. This current arrangement differs from the previous two-stage orbiter (Fig. 3.4-1) as follows:

a. Aerodynamic Shape and/or Geometry

- Lower forward surface ramp eliminated
- Upper surface boattailing revised
- Aft body (base) contour changed
- Overall length reduced by 2-$\frac{1}{2}$ ft to 156.5 ft

b. Jet Engine System

- Jet engine inlet ducts moved to lower fin surface
- Four jet engines installed instead of six

c. Landing Gear Installation

- Main landing gear installed outboard of ascent fuel tankage

d. Orbital Maneuver Propellant System

- LH_2 orbit-maneuver propellant tank installed vertically

e. Payload/Orbiter Interface System

- Payload mounting and deployment mechanism installed forward of payload bay

Figure 3.4-5 indicates the contour lines defining the Model LS 400-7 delta-body orbiter, while Fig. 3.4-6 shows the locations of the main internal systems associated with this concept.

The current structural concept for the after portion of the two-stage, delta-body orbiter is shown by Fig. 3.4-7. This concept is generally the same as

in Section 4.5 reported for the two-stage (2-B) orbiter (Fig. 4.5-8), except that additional details of the intertank ring, jet-engine installation, main landing-gear bays, booster-attach fittings, and main rocket-engine thrust structure are shown with respect to the Model LS 400-7 contours.

Figure 3.4.8 illustrates the Main Propulsion Plumbing System Arrangement. Individual LO_2 ascent tank feed lines (14-in. diameter) feed individual rocket engines with a cross manifolding which includes a circulation loop valve; the single-point oxidizer-fill line is connected to this manifold. The center ascent fuel tank acts as a sump tank into which the two main ascent fuel tanks drain via individual standpipes. Individual feed lines connect from the sump tank to the fuel intakes of each rocket engine. The single-point fuel-fill line is connected directly to the sump tank. Also, preliminary fill and feed, and pressurization and venting system schematics are shown.

The volume and wetted area associated with the Model LS 400-7 orbiter is indicated on Fig. 3.4.9.

An alternate approach to the general arrangement of the major internal systems is shown in the inboard profile drawings indicated in Fig. 3.4-10. The primary change in arrangement over the basic system is reflected in the arrangement of the (1) flight station and the passenger/crew cabin system, (2) payload deployment mechanism, (3) LH_2 OMPS tank installation, and (4) the relocation of the orbital maneuvering rocket engines. The rearrangement of the flight station and passenger/crew cabin system was done to principally permit the space accommodation of ten additional passengers within the orbiter cabin system.

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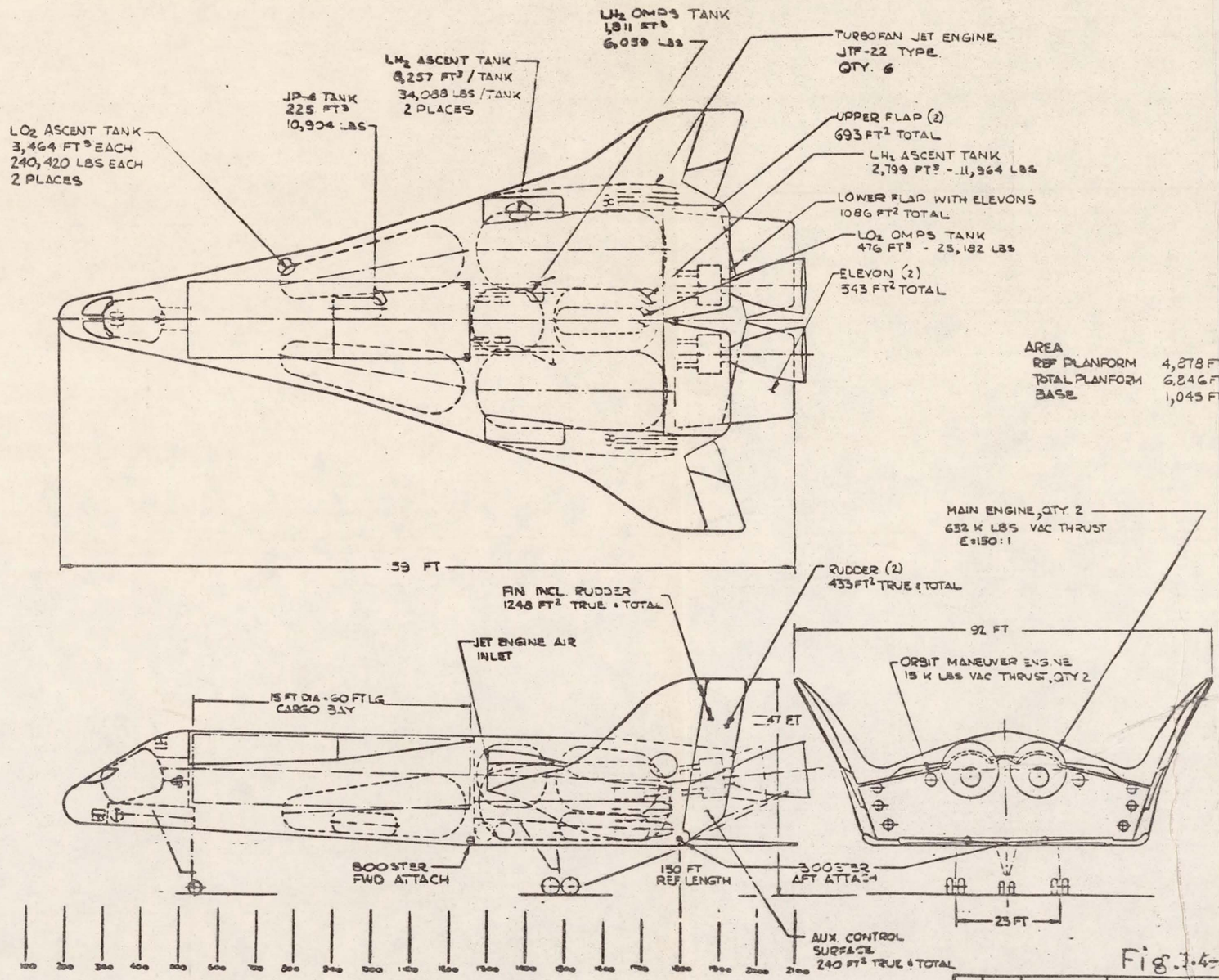

Fig. 3.4-1

2C
2B

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				PARTS LIST			

UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES TOLERANCES ON: FRACTIONS ± 1/16, DECIMALS: .X ± .1, .XX ± .02, .XXX ± .010, ANGLES ± 2 DEG	DATE 2-17-71	LOCKHEED MISSILES & SPACE COMPANY A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION SUNNYVALE, CALIFORNIA	
	DWG APVD	GENERAL ARRANGEMENT (2B&2C) DELTA BODY - 2.5 STAGE VOOR LS 400-5 8-6	
	APVD		
	ENGRG		
	CHKD		
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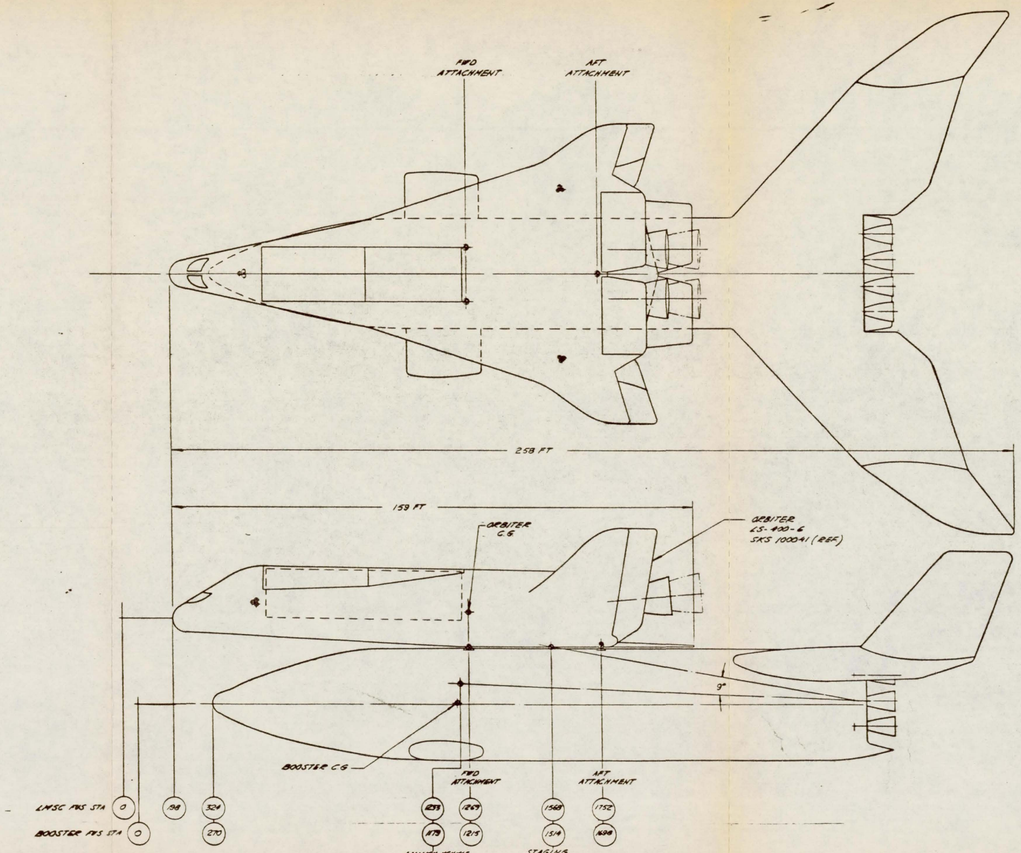

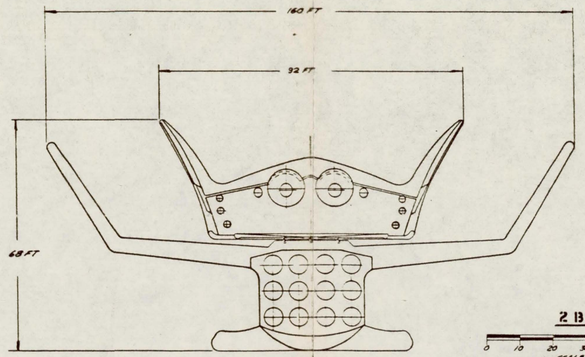

NOTE
L.C.G. LOCATIONS ARE PRELIMINARY ESTIMATES.

2 B

0 10 20 30 40 50 FT
SCALE

Fig. 3.4-2

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			PARTS LIST				

UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES TOLERANCES ON FRACTIONS ± 1/16 DECIMALS .X = ± 1 .XX = ± .03 .XXX = ± .015 ANGLES ± 2 DEG

NEXT ASSY | USED ON
APPLICATION | CONTR NAS 9-150

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

SATURN IB/V STUDY
DELTA 3031-2 STAGE
MODEL LS-400-6

SIZE J | CODE IDENT | DRAWING NO. SAT 700044 | REV

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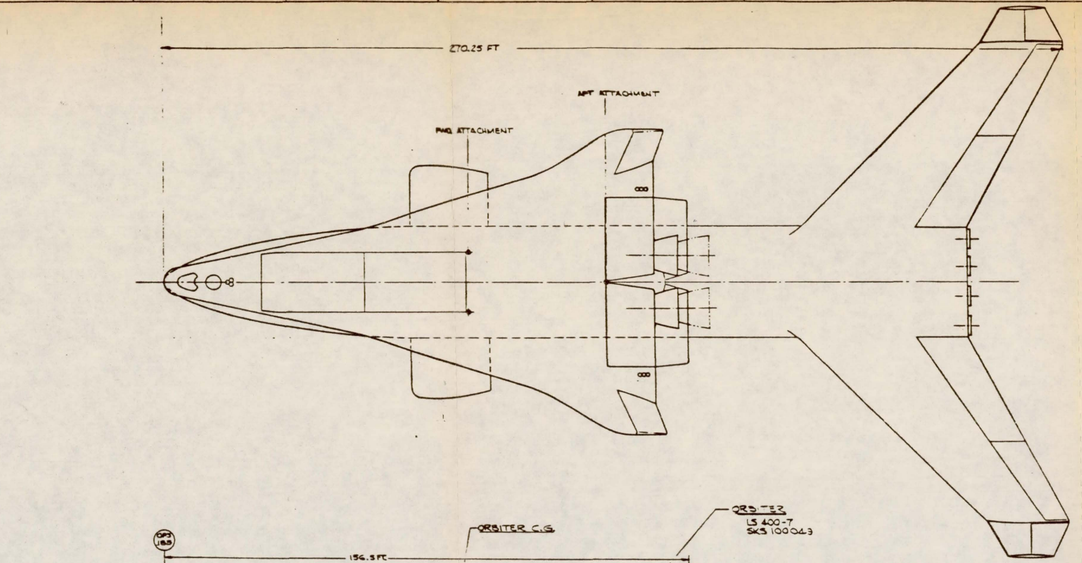

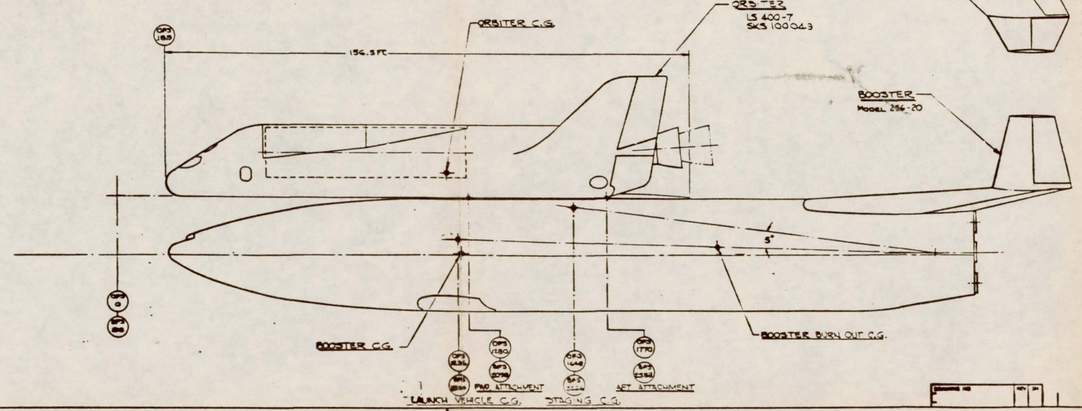

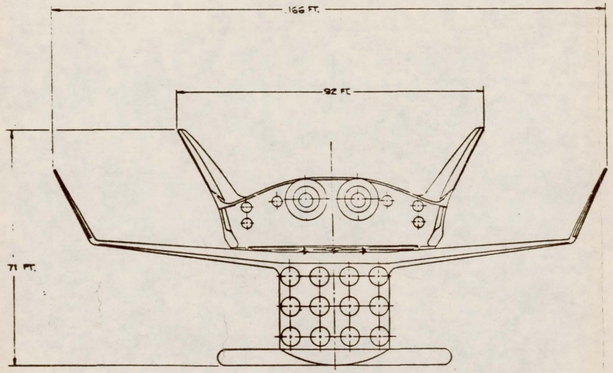

NOTE:
E.C.G. LOCATIONS ARE PRELIMINARY ESTIMATES.

Fig 3.4-3

LOCKHEED MISSILES & SPACE COMPANY
LAUNCH VEHICLES
DELTA BODY ORBITER
MODEL LS400-7
SKH 100051 A

3.4-11

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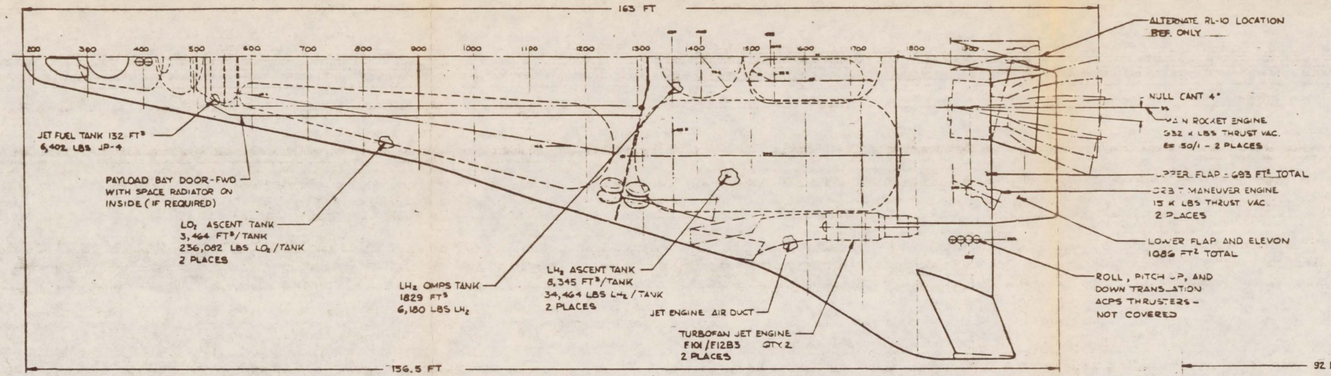
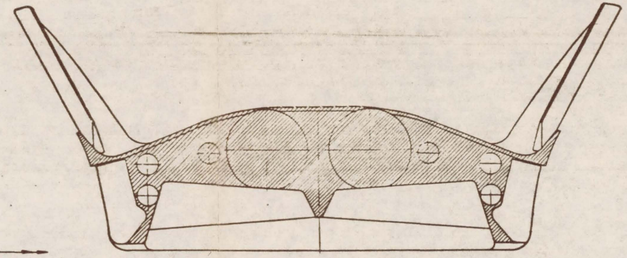
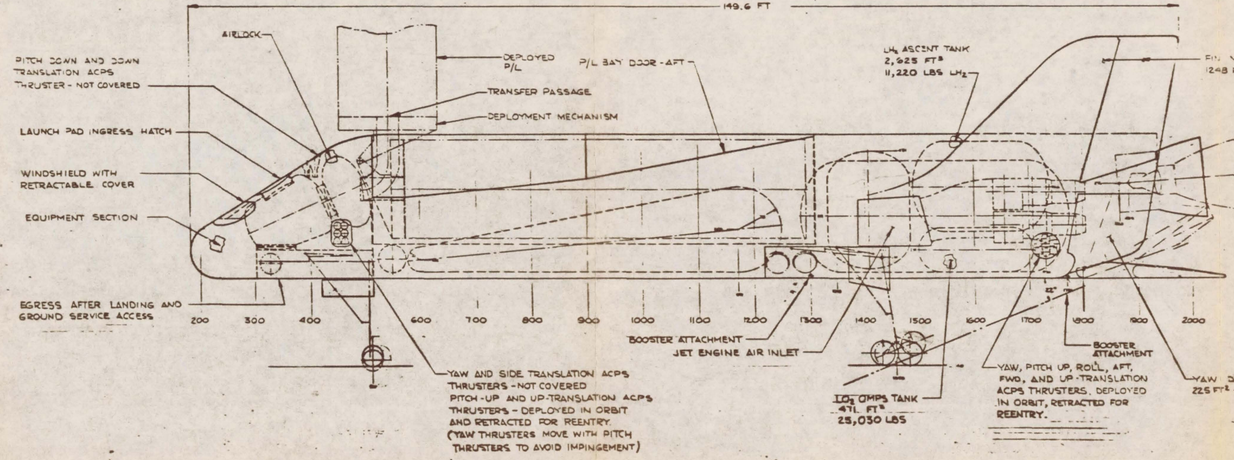


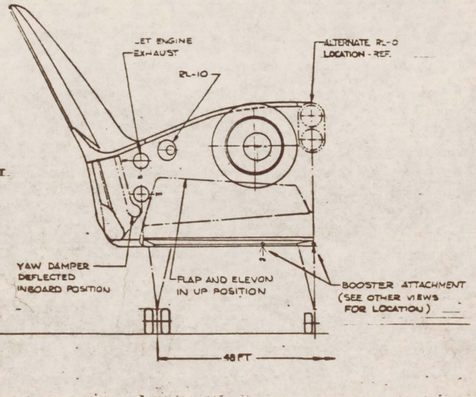


BASE - CRUISE CONDITION

CONFIG	LTR	DESCRIPTION	DATE	APVD
	A	CHANGED LS-400-T TO LS-400-T A BY MOVING DEPLOYMENT MECH. TO FWD BHD OF P/L BAY	10/8/71	
	B	JET ENGINE MDL J79-21B FWD. BOOSTER ATTACHMENT MRS P.S. 1370 - RL10 MDL PS. 1585 AND B.L. 181 OMPS PROPELLANT WAS LO₂ 472 FT³, 33,388 LBS LH₂ 1806 FT³, 6,077 LBS ADDED ALT RL-10 W.S.T.L. +25 IN. W ADDED MAIN BAG PER CANT-6		

LIMITED CALENDAR LIFE | LIMITED OPERATING LIFE





AREA:
REF PLANFORM : 4,878 FT²
TOTAL PLANFORM : 6,846 FT²
BASE : 1220 FT²

Fig. 3.4-4

QTY REQD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO
			PARTS LIST				

COPYRIGHT 1970 PER

UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES. TOLERANCES ON FRACTIONS ± 1/16 DECIMALS 1 PL ± .1 2 PL ± .03 3 PL ± .010 ANGLES ± 2 DEG	DATE	9-12-70
	DWN	[illegible]
	APVD	[illegible]
	ENGRG	
	OTHER	
	APVD	
CONTR	APVD	
CUST APPL		

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

GENERAL ARRANGEMENT
DELTA BODY ORBITER
MODEL LS-400-T A

SIZE A | CODE IDENT 1 20267 | DRAWING NO SAS 10049 | REV B

SCALE | SHEET

3.4-13

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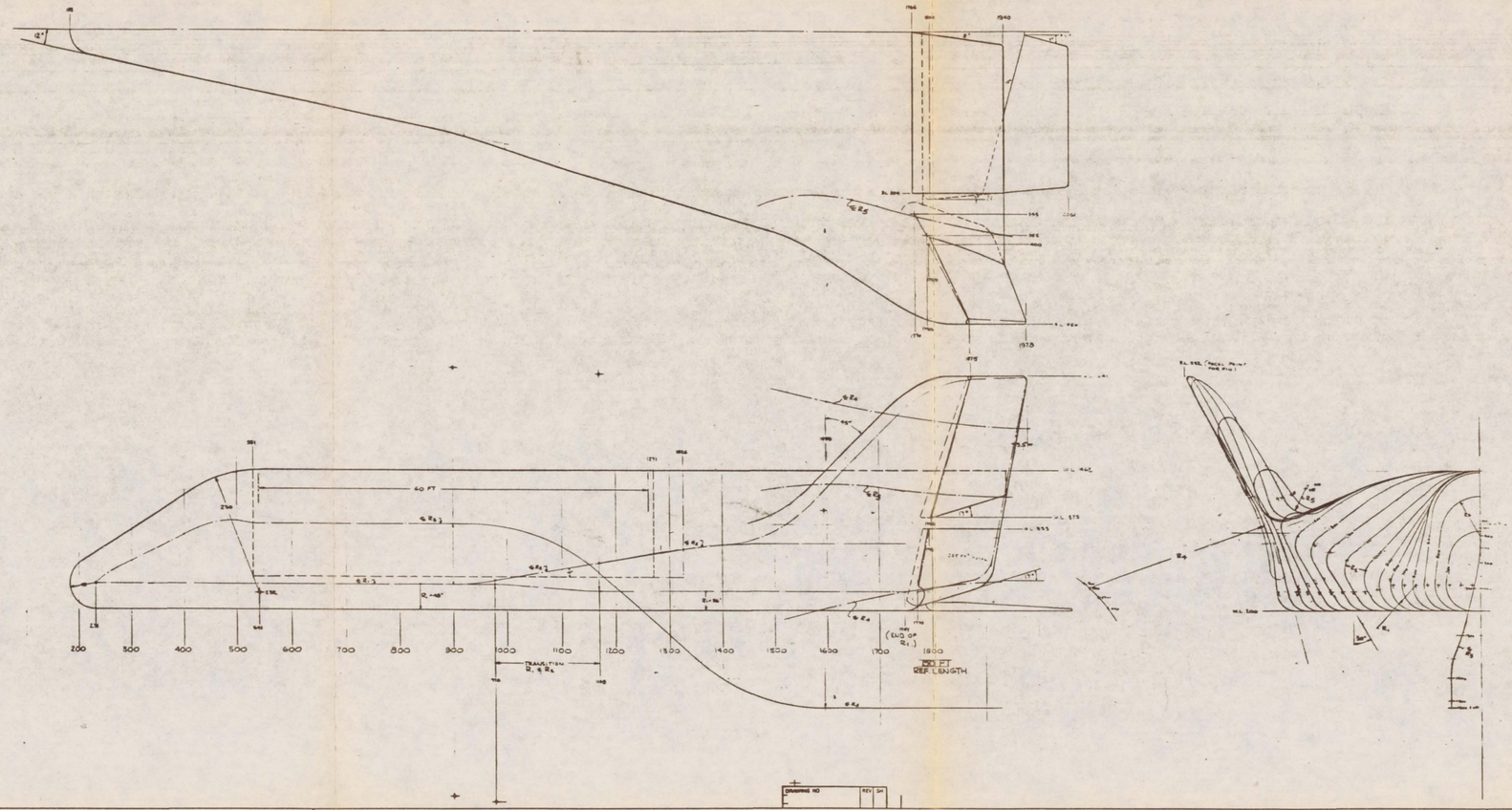

Fig. 3.4-5

QTY REQD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO
			PARTS LIST				

INTERPRET DWG PER	UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES TOLERANCES ON	DATE 3-1-71	LOCKHEED MISSILES & SPACE COMPANY A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION SUNNYVALE, CALIFORNIA
		DWN [illegible]	
		APVD	
	FRACTIONS ± 1/16	APVD	CONTOUR LINES
	DECIMALS X = ± .1	ENG'RG	DELTA BODY DAB/TER
	XX = ± .03	CHKR	MODEL LS-400-7
	XXX = ± .015	APVD	
	ANGLES ± 2 DEG		SIZE CODE IDENT DRAWING NO REV
BODY ASSY USED ON APPLICATION	CONTR	APVD	J 51847 LOCKHEED 100247
	CCA-01-70		SCALE 1/2 SHEET 1 OF 1

3.4-15

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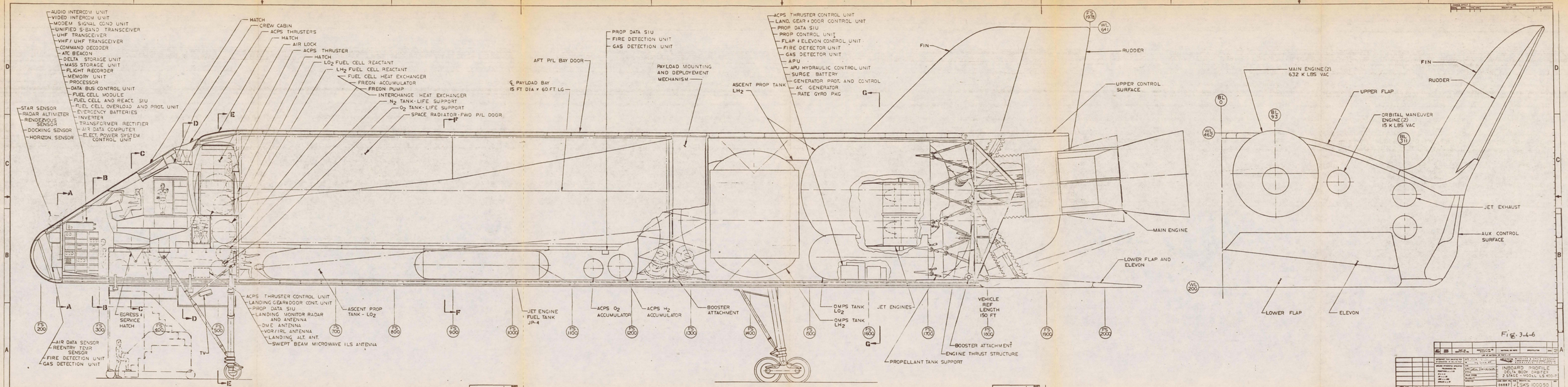

Fig. 3.4-6

INBOARD PROFILE
DELTA BOOSTER
2 STAGE - MODEL LS 400-7

06887 2 SKS 100050

3.4-17

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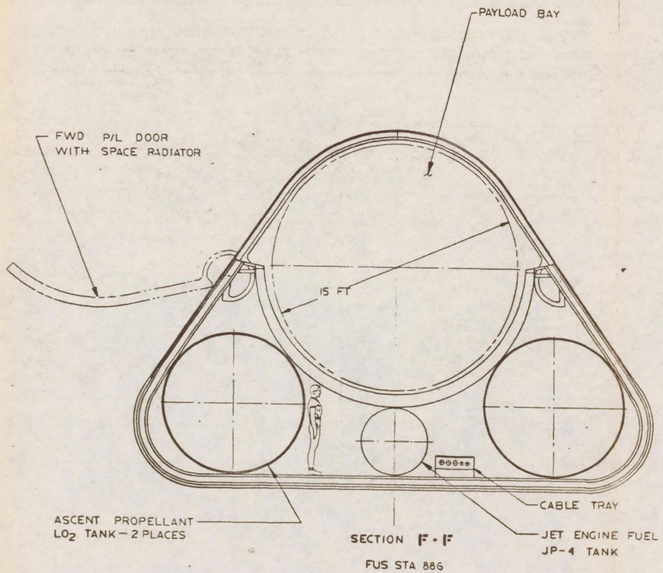
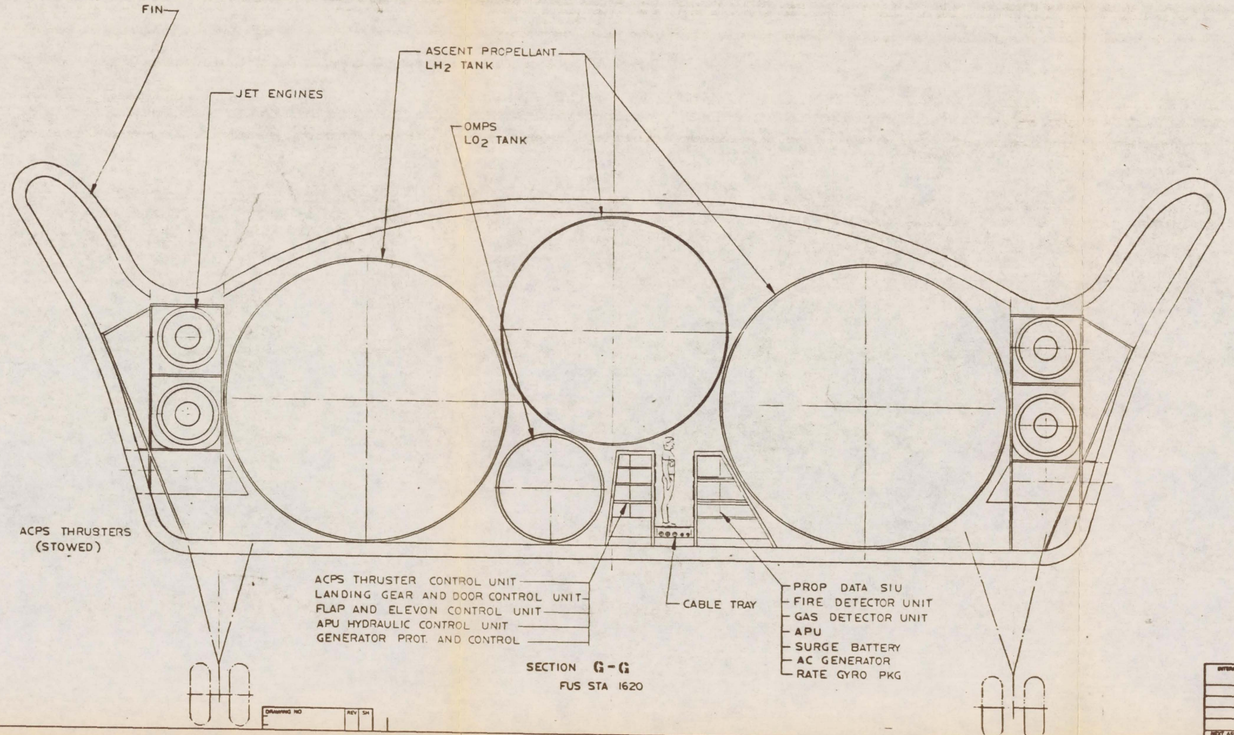


SECTION F-F
FUS STA 886



SECTION G-G
FUS STA 1620

Fig. 3.4-6

LIMITED CALENDAR LIFE	LIMITED OPERATING LIFE

REVISIONS			
ZONE	LTR	DESCRIPTION	DATE	APVD

QTY REQD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO

PARTS LIST

INTERFACE DWG PER	UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES TOLERANCES ON FRACTIONS = ± 1/16 DECIMALS .X = ± .1 .XX = ± .03 .XXX = ± .010 ANGLES = ± 2 DEG	DATE 4-2-73	LOCKHEED MISSILES & SPACE COMPANY A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION SUNNYVALE, CALIFORNIA
		DWN [illegible]	
		APVD [illegible]	
		APVD	INBOARD PROFILE DELTA BODY ORBITER 2 STAGE - MODEL LS #OO1-7
		ENGNG	
		OTHER	
		APVD	SIZE J CODE IDENT 07687 DRAWING NO SKS 1600501 REV
NEXT ASSY USED ON APPLICATION	CONTR	APVD	SCALE NONE SHEET 1 OF 1
	CCA-01-P/D		

3.4.10

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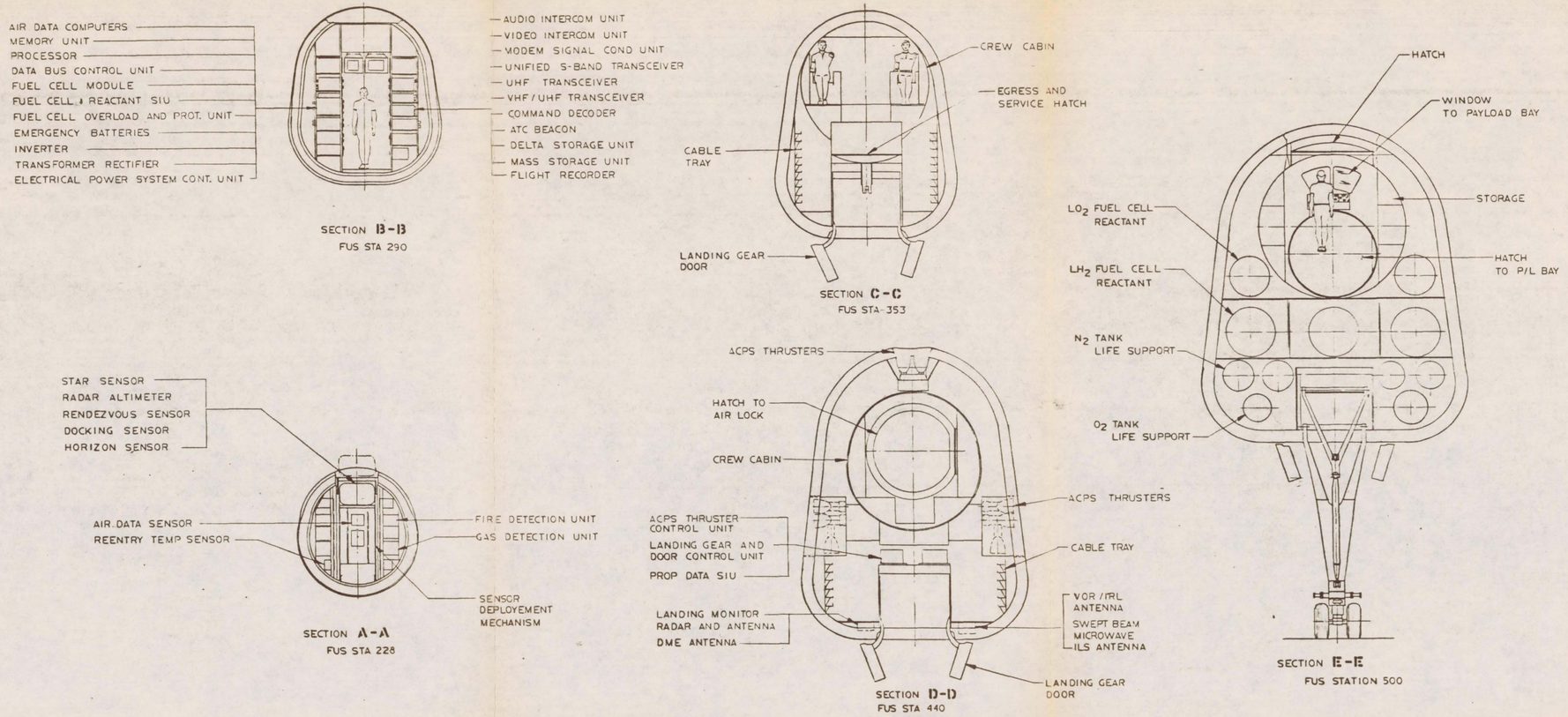

Fig. 3.4-6

QTY REQD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO

PARTS LIST

REVISIONS			
ZONE LTR	DESCRIPTION	DATE	APVD

LIMITED CALENDAR LIFE | LIMITED OPERATING LIFE

UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES TOLERANCES ON FRACTIONS = ± 1/16 DECIMALS X.X = ± .1 X.XX = ± .03 X.XXX = ± .010 ANGLES = ± 2 DEG

NEXT ASSY | USED ON | APPLICATION | CONTR | CAGE CODE

DATE | DWN | APVD | CHKD | APVD | APVD

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

INBOARD PROFILE
DELTA BODY ORBITER
2 STAGE MODEL LS 400-7

SIZE J | CODE IDENT 94797 | DRAWING NO SKS 100050 | REV

SCALE | SHEET 1 OF 1

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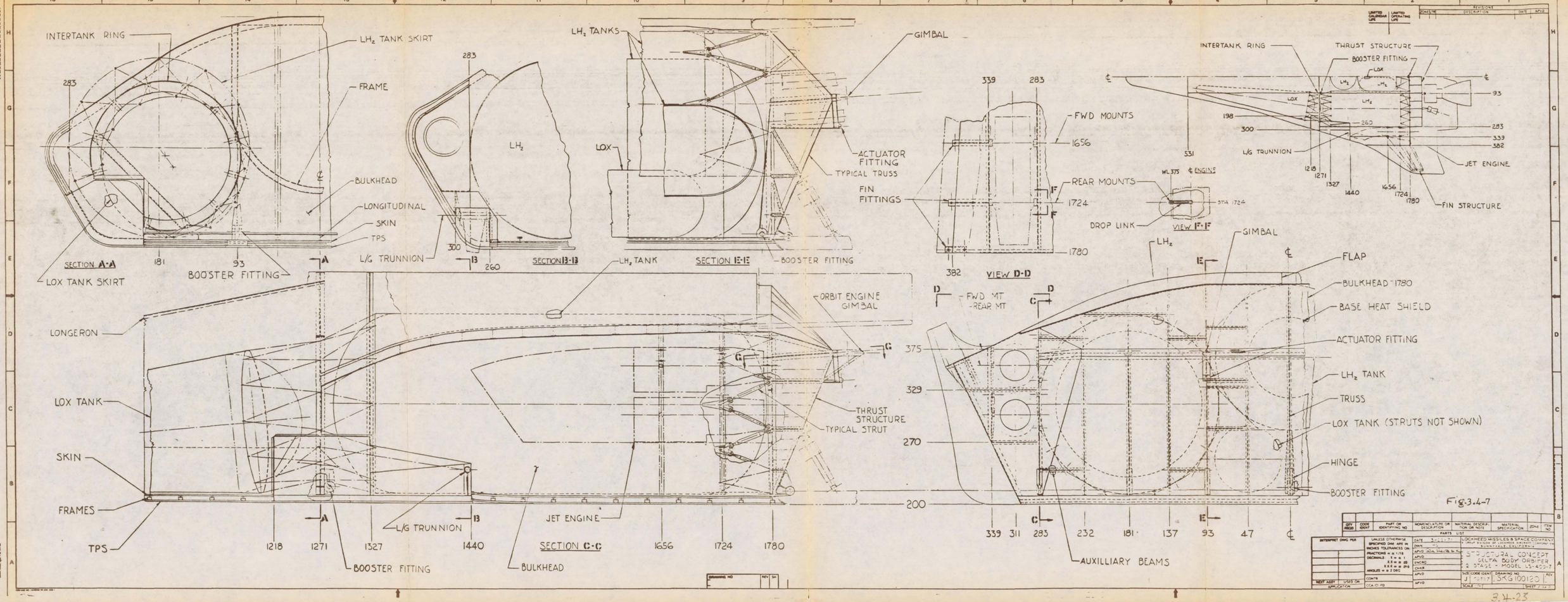

Fig. 3.4-7

QTY PER	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO
PARTS LIST							

DATE	3-27-7
DWN	
APVD	
APVD	
ENGRG	
CHKR	
APVD	
APVD	

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

STRUCTURAL CONCEPT
DELTA BODY ORBITER
2 STAGE - MODEL 43-4597

SIZE J | CODE IDENT | DRAWING NO SKG100120 | REV
SCALE | SHEET

3.4-23

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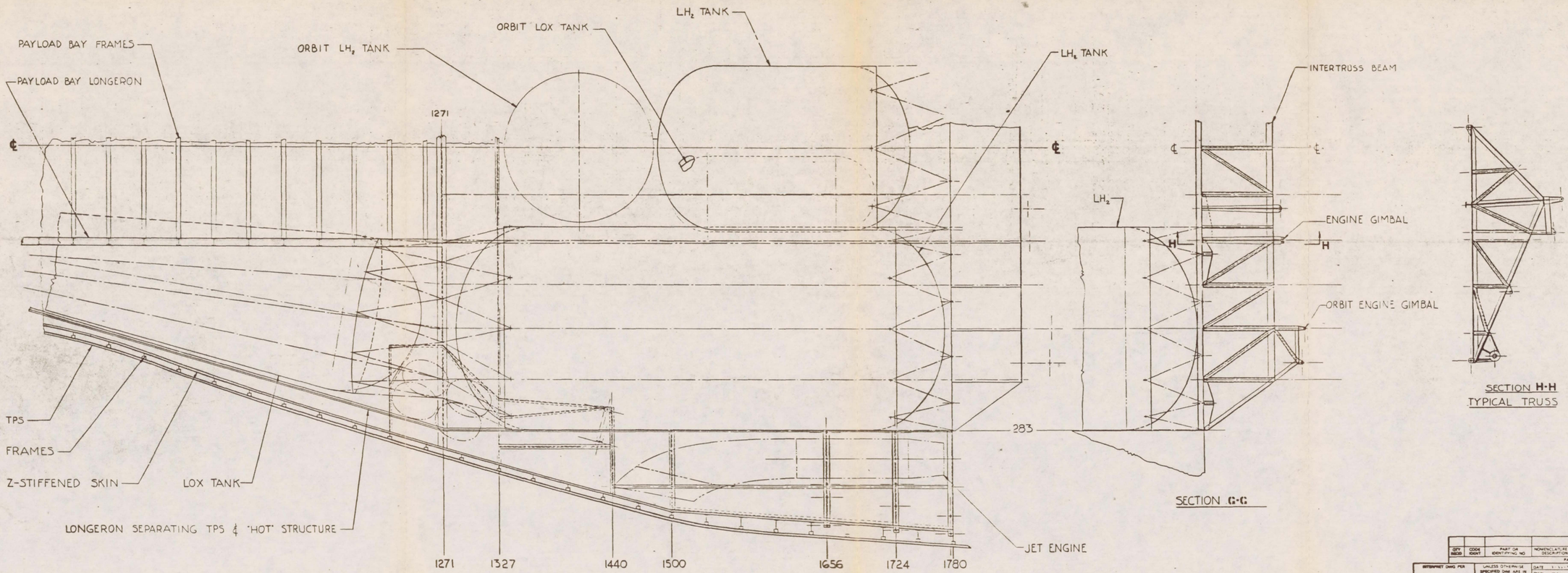

SECTION G-G

SECTION H-H
TYPICAL TRUSS

Fig. 3.4-7

REVISIONS			
ZONE/LTR	DESCRIPTION	DATE	APVD

LIMITED CALENDAR LIFE | LIMITED OPERATING LIFE

QTY REQD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO
			PARTS LIST				

INTERFACE DWG PER

UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES TOLERANCES ON FRACTIONS ± .x 1/16 DECIMALS .X ± .1 .XX ± .03 .XXX ± .010 ANGLES ± 2 DEG

NEXT ASSY | USED ON | APPLICATION

CONTR | USA CI 113

DATE | DWN | APVD | APVD | APVD | ENGRG | CHKR | APVD | APVD

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

STRUCTURAL CONCEPT
SEGMENTED BODY ORBITER
2 STAGE - PARALLEL LOX-400-7

SIZE | CODE IDENT | DRAWING NO | REV
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SCALE 1" = 1 | SHEET 2 OF 2

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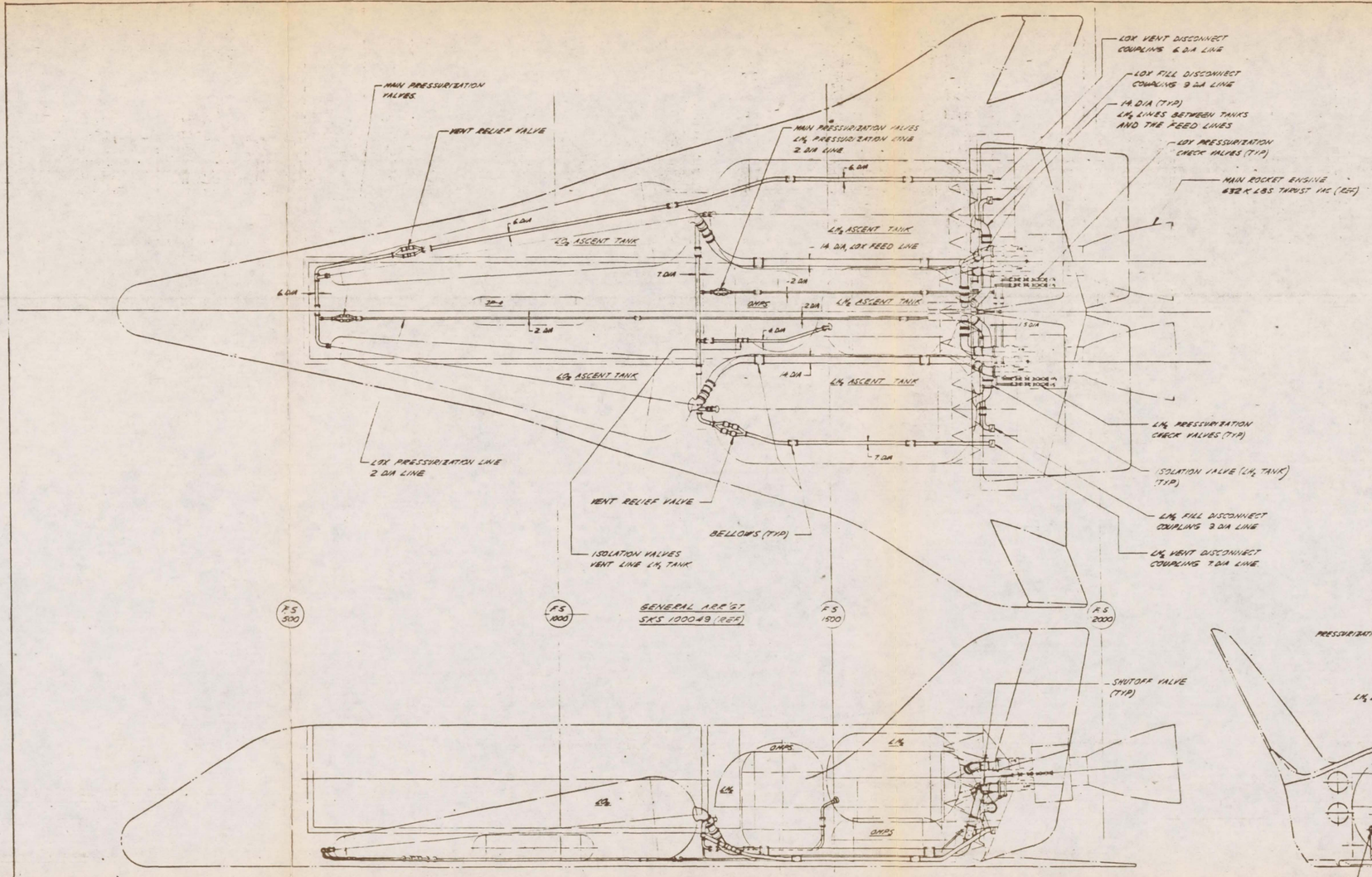
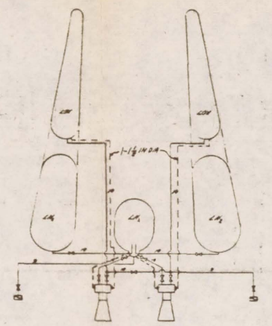
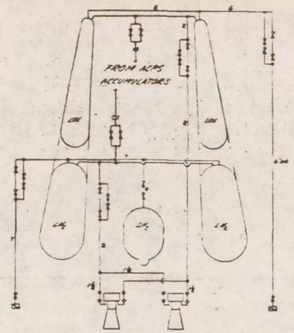




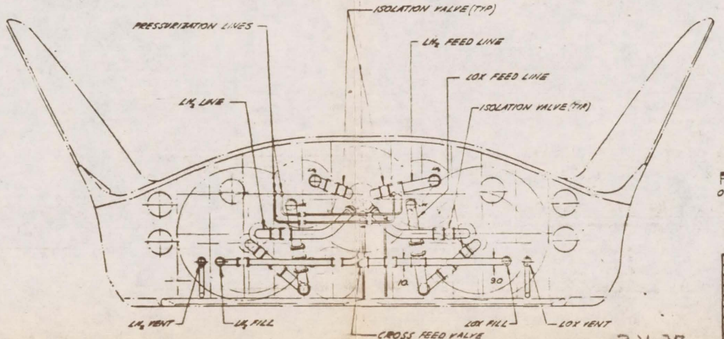
SCHEMATIC
FILL & FEED SYSTEM



SCHEMATIC
PRESSURIZATION & VENTING SYSTEM

SYMBOLS

- DISCONNECT
- ISOLATION VALVE
- CHECK VALVE
- LO_2 LIQUID OXYGEN
- LH_2 LIQUID HYDROGEN



PROPULSION SYSTEM SCHEMATIC
REF SKS 03197

0 5 10 15 20 25 FT
SCALE

Fig. 3.4-8

DATE 3-21-71	LOCKHEED MISSILES & SPACE COMPANY		
DESIGN ENGR	A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION SUNNYVALE, CALIFORNIA		
APPROVAL	PLUMBING SYSTEM		
APPD	MAN MOBILITY VEHICLE SYSTEM		
ENGRG	MSFC - 5-40337		
CHK			
APPD	SIZE J	CODE IDENT 06897	DRAWING NO SAT 100
APPD	SCALE NONE		SHEET 6 REV A

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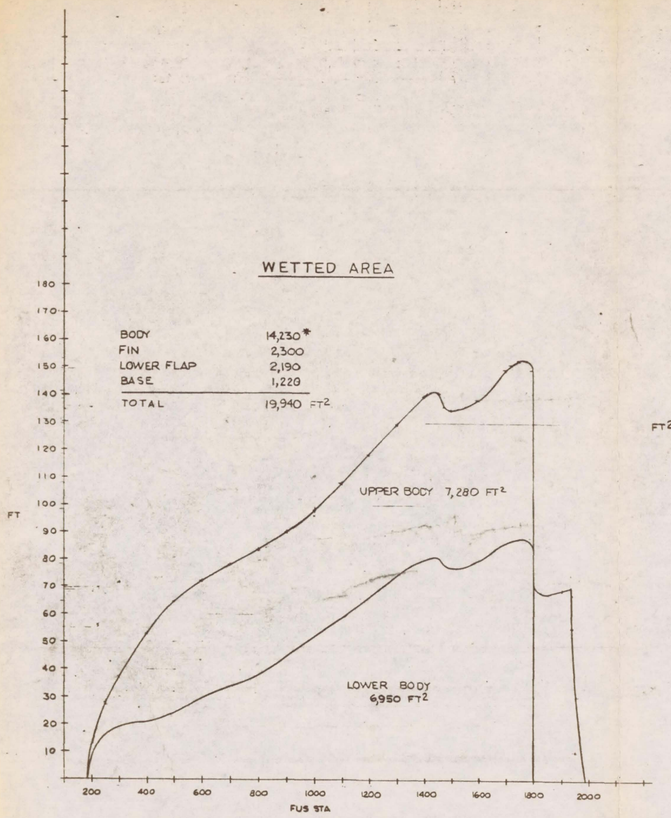
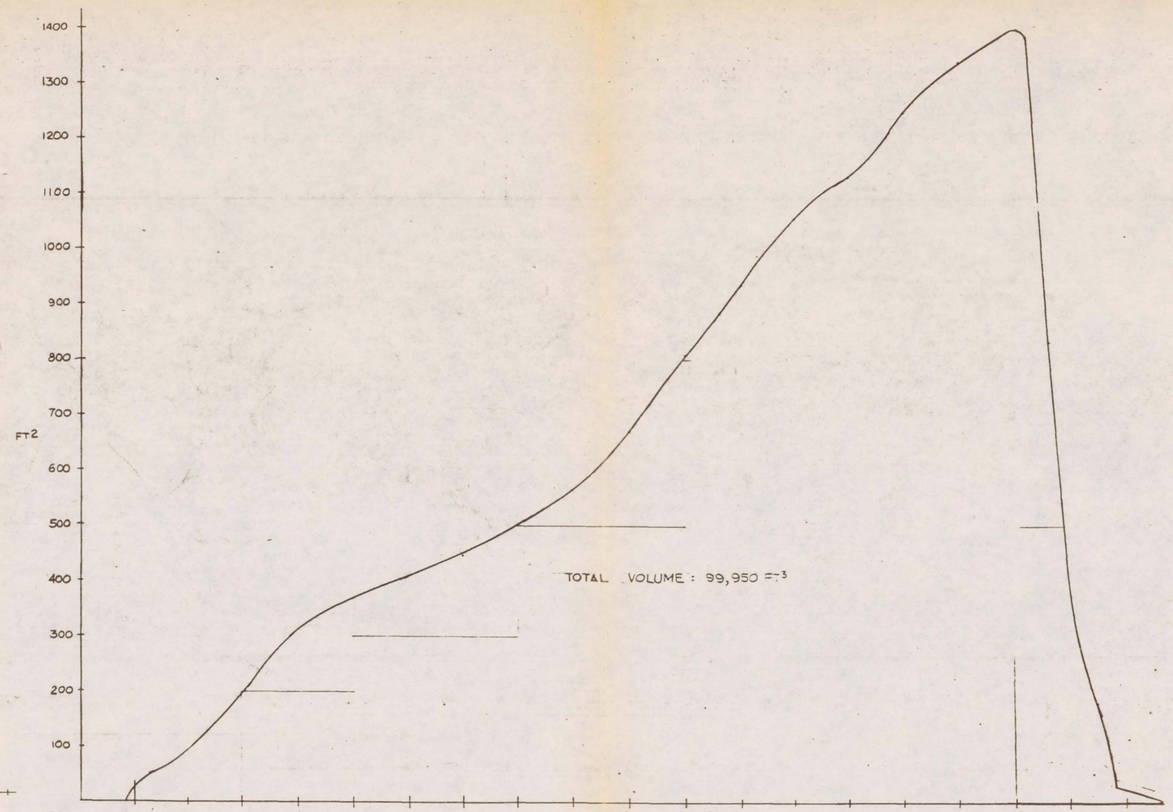
WETTED AREA

BODY	14,230*
FIN	2,300
LOWER FLAP	2,190
BASE	1,220
TOTAL	19,940 FT²



*INCLUDING UPPER FLAP



REF. DWG. SKS 100046

Fig. 3.4-9

QTY PER	CODE IDENT	PART OR IDENTIFYING NO.	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO.

PARTS LIST

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

VOLUME + WETTED AREA CURVE
DELTA BODY ORBITER
MODELS LS-400-T

SIZE | CODE IDENT | DRAWING NO | REV
J | 04597 | SKS 100054 |
SCALE | SHEET

3.4-29

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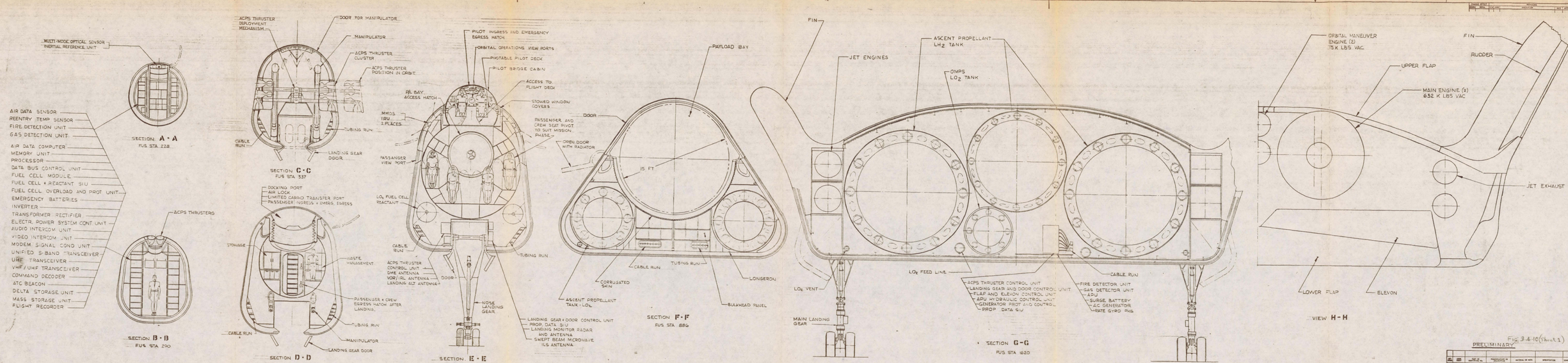

Fig 3.4-10(Sheet 1)

PRELIMINARY

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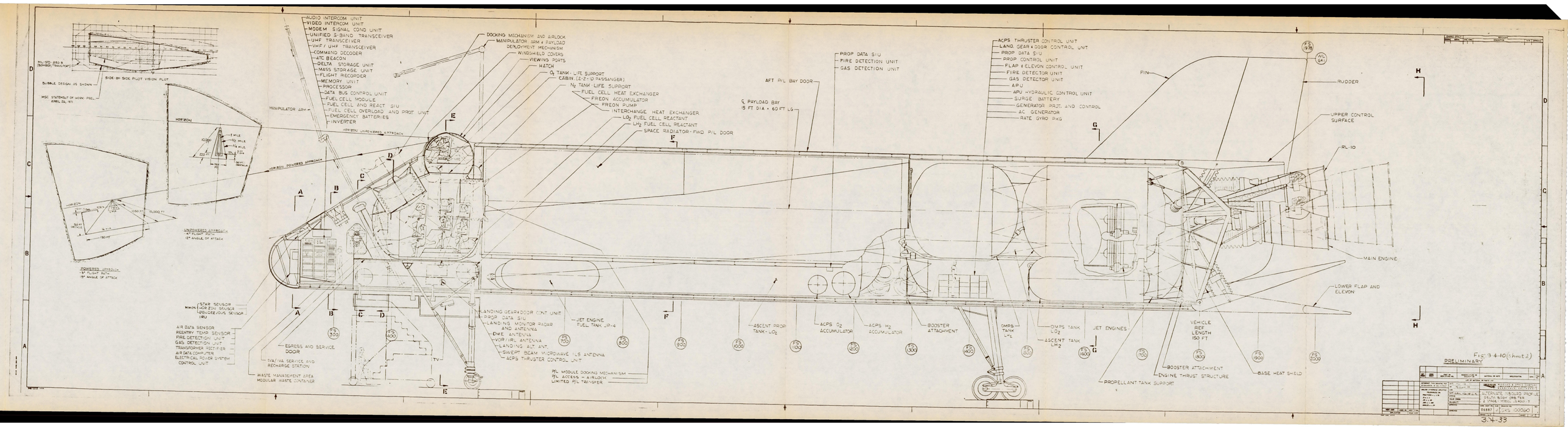

PRELIMINARY

Fig. 3.4-10 (Sheet 2)

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3.5 AERODYNAMICS

3.5.1 Introduction

Aerodynamic characteristics are presented for the Lockheed delta lifting-body orbiter to be used in conjunction with the Space Shuttle Two-Stage system.

3.5.2 Aerodynamic Requirements (Orbiter)

Several basic aerodynamic requirements have been imposed on the orbiter configuration – either from the NASA or from Lockheed. Since the majority of entry flight time is in the subsonic and hypersonic-speed regimes, the aerodynamic design emphasis has been largely influenced by the requirements associated with these particular regimes. The basic aerodynamic performance philosophy is to attain the majority of the required 1100-nm crossrange through the use of the hypersonic lift-to-drag ratio (L/D) and to achieve adequate subsonic L/D for an acceptable landing approach; also, subsonic L/D significantly impacts the orbiter ferry-range capability. Therefore, the transonic/supersonic performance has been purposely desensitized; the prime requirements in this regime being stability and control. Neutral or better longitudinal lateral, and directional static stability have been emphasized throughout the operating attitudes and Mach numbers. Sufficient trim authority must be available to accommodate the center-of-gravity extremes afforded by the various payload combinations both with and without airbreather engines. When aerodynamic heating is severe, an additional requirement has been imposed to limit the windward trim and control surface deflections to 5 deg or less, thus assuring TPS material compatibility with anticipated thermal environment.

3.5.3 Configuration Description (Orbiter)

The orbiter is a delta planform lifting body with appropriate aerodynamic surfaces, configured for adequate performance, stability, and control. Leading edge sweep and radii have been parametrically studied to assure aerodynamic and

aerothermodynamic compatibility. A triangular, flat-bottom body surface is required to assure a successful compromise between subsonic trim and hypersonic longitudinal stability and trim. The aft body is shaped to eliminate flow separation and to minimize the large base area associated with the delta-body orbiter. Aft-body compression-sharing surfaces have been utilized in an attempt to optimize the basic body shape for hypersonic yaw-stability considerations. Ideally, the body sides are rolled in to prevent excessive heating, drag, and loss of directional stability at the hypersonic design angle-of-attack. To facilitate improved packaging capability – specifically where it enhances a forward cg location – the forebody side roll-in requirement has been relaxed to some extent.

To accommodate the aft cg locations inherent in the delta-body orbiter, the addition of aerodynamic surfaces is necessary to provide acceptable longitudinal and directional static stability margins. Two aft-mounted side fins have been sized and positioned (toe-in and rollout) – based on directional stability requirements throughout the operating speed regime. The fins also contribute significantly to the low-speed lift-to-drag performance and longitudinal stability.

An aft trim flap hinged at the body-bottom base has been sized to allow acceptable hypersonic stability and subsonic trim authority. Pitch control is accomplished by deflection of two elevon surfaces which comprise the aft portion of the trim flap. These same surfaces can be differentially deflected for roll control and roll damping.

Roll coordination, dutch roll mode damping, crosswind landing and sideslip control capability, pitch trim authority, and hypersonic yaw stability are provided by two rudders (one on each fin). Also, a speed brake-control surface is located beneath each rudder to provide glide path control and can likewise be used to increase hypersonic yaw stability.

Two elevons are mounted on the orbiter upper surface between the vertical fins. The functions of these surfaces are roll control and damping, takeoff rotation, landing trim and derotation, transonic/supersonic pitch stability, and trim authorit

3.5.4 Flight Characteristics (Orbiter)

An experimental wind tunnel program was funded under the Alternate Concept Study to investigate the aerodynamic characteristics of the Lockheed (LS 200-5) delta lifting-body orbiter and the stage-and-one-half launch vehicle in the NASA Ames 6 ft x 6 ft and the Langley Unitary Plan Wind Tunnels. Concurrent with this contract effort, a 0.03 scale model of the LS 200-5 orbiter with parametric variations was designed, fabricated and tested by LMSC in the Lockheed 8 ft x 12 ft Low Speed Wind Tunnel. The objectives of both programs respectively were: (1) to provide an experimental data verification of the aerodynamic estimates, and (2) to define a credible baseline of data for future engineering studies.

An experimental wind tunnel program was funded under the Alternate Concept Study to investigate the aerodynamic characteristics of the Lockheed (LS 200-5) delta lifting-body orbiter and the stage-and-one-half launch vehicle. In addition, a low-speed test on the orbiter was conducted at contractor expense.

The current orbiter configuration identified as LS 400-7A utilized the findings of the LS 200-5 wind tunnel tests and, consequently, reflects minor external geometry differences. Unless noted otherwise, the LS 200-5 aerodynamic characteristics presented in the following discussion are considered to be representative of the current LS 400-7A design.

The two-stage orbiter, depending upon mission requirements, can have cg locations between 72.5 and 78 percent of the reference length (146 ft) i.e., F.S. 1270 and 1367, respectively. The following charts will present pertinent aerodynamic characteristics throughout the entry speed regime emphasizing these two cg's and the flexibility of the delta-body orbiter to accommodate these extremes. Worthy of mention is the fact that the 78-percent cg condition occurs only at landing; the farthest aft cg above Mach 0.6 is 76.8 percent.

These data will be discussed by speed regime-subsonic/transonic/supersonic and hypersonic - followed by summary charts of general aerodynamic characteristic and specific cases relating the reference entry trajectory. Figure 3.5-1 identifies the axis system and nomenclature used throughout this section.

3.5.4.1 Subsonic. A low-speed wind tunnel test (Fig. 3.5-2) was conducted at the Lockheed 8 ft x 12 ft Low-Speed Wing Tunnel on a 0.03 scale model of the delta lifting-body orbiter. A wide variety of configuration combinations were run in an effort to optimize the low-speed aerodynamic configuration. Figure 3.5-3 indicates acceptable longitudinal stability and trim characteristics for the anticipated cg extremes. The trim lift coefficients and lift-to-drag ratios are presented in Fig. 3.5-4. Maximum trimmed lift-to-drag ratios of 5.10 and 5.85 are predicted for the fore and aft cg, respectively. All L/D data include the total drag of the vehicle (including realistic base drag). Landing speeds have been computed at various angles-of-attack using the aforementioned trimmed lift coefficients. Again, the mission extremes have been presented to illustrate the range of anticipated landing speeds and attitudes (see Fig. 3.5-5).

Condition	Landing Speed at $\alpha_{L/D_{Max}}$	Landing Speed at $\alpha^*_{Tailscrape}$
40K lb Cargo In/Airbreather Engines Out (Most Forward CG)	180 Kt	152 Kt
Cargo Out/Airbreather Engines In (Most Aft CG)	154 Kt	122 Kt

* $\alpha_{Tailscrape}$ = 22 deg

It should be emphasized that ground effects have not been included but are available and indicate reduced landing speeds and/or attitudes from those shown in the above table.

3.5.4.2 Transonic/Supersonic. A 0.01 scale steel model of the orbiter configuration was tested in NASA/Ames 6 ft x 6 ft tunnel and the NASA/Langley Unitary Plan Wind Tunnel (Fig. 3.5-6). The majority of effort was concentrated on obtaining configuration combinations which would allow acceptable static stability and trim authority for the cg range. Considerable pitch and sideslip data are available between Mach number 0.6 and 4.6, and these are presented in Figs. 3.5-7 through 3.5-15. Neutral or better longitudinal stability and adequate trim characteristics are typical of this speed regime.

3.5.4.3 Hypersonic. Machine drawings from the Hypersonic Arbitrary Body Computer Program, HABCP are shown in Fig. 3.5-16. Hypersonic characteristics are presented in Figs. 3.5-17 and 3.5-18. These are based on the HABCP modified to reflect previously obtained increments between experimental and analytical results on a similar configuration. The vehicle is longitudinally stable and trimmable over a wide range of angles-of-attack. Consistent with the anticipated thermal environment and the thermal protection system material, no windward deflections in excess of 2 deg are required for hypersonic trim of the 78 percent c.g. at L/D_{max}. However, the actual cg of 76.8 percent L_{REF} requires no windward deflection ($\delta_E < 0$ deg). Maximum trimmed L/Ds of 1.60 to 1.65 are attained for the fore and aft cg location. The LS 400-7A configuration has a hemispherical nose cap rather than the 2:1 ellipsoid of revolution employed by LS 200-5. Also, it has reduced fin leading edge radii. These differences should produce decreased hypersonic drag resulting in maximum trimmed L/Ds of 1.87. No significant change in stability would be predicted.

3.5.4.4 Summary. Directional stability has been expressed in terms of the dynamic yaw-stability derivative, $C_{n_{\beta Dynamic}}$, where:

$$C_{n_{\beta Dynamic}} = C_{n_{\beta Body\ Axis}} \cos\alpha - C_{l_{\beta Body\ Axis}} \sin\alpha \left(\frac{I_{zz}}{I_{xx}}\right)$$

Figures 3.5-19 through 3.5-21 are plots of $C_{n_{\beta_{Dynamic}}}$ versus Mach number and angle-of-attack at various rudder combinations; rudders deflected in 20 deg, 0 deg, and out 10 deg. The selected mode of rudder biasing is dependent on the Mach number. Emphasizing aerodynamic heating constraints, the rudders are nominally set at 0 deg above Mach 6.0. For the aft cg condition, in the speed regime of Mach numbers 2 to 6, the rudder are flared 10 deg out to increase directional stability thus effectivesly increaseing the angle-of-attack range where stability is maintained. The rudders remained undeflected for the forward cg case between Mach 2 and 6 because of the increased directional stability afforded by the forward cg location. The rudders are biased inward 20 deg between Mach numbers 0.6 to 2.0 to improve the trim authority as required for the far forward center-of-gravity locations. In the landing approach (Mach 0.6 to touchdown), the rudders have been nominally set at 25 deg and 15 deg in for the fore and aft cg location, thus taking advantage of the positive trim pitching moment increment, increased L/D, and slightly increased longitudinal stability offered by the rudder biasing.

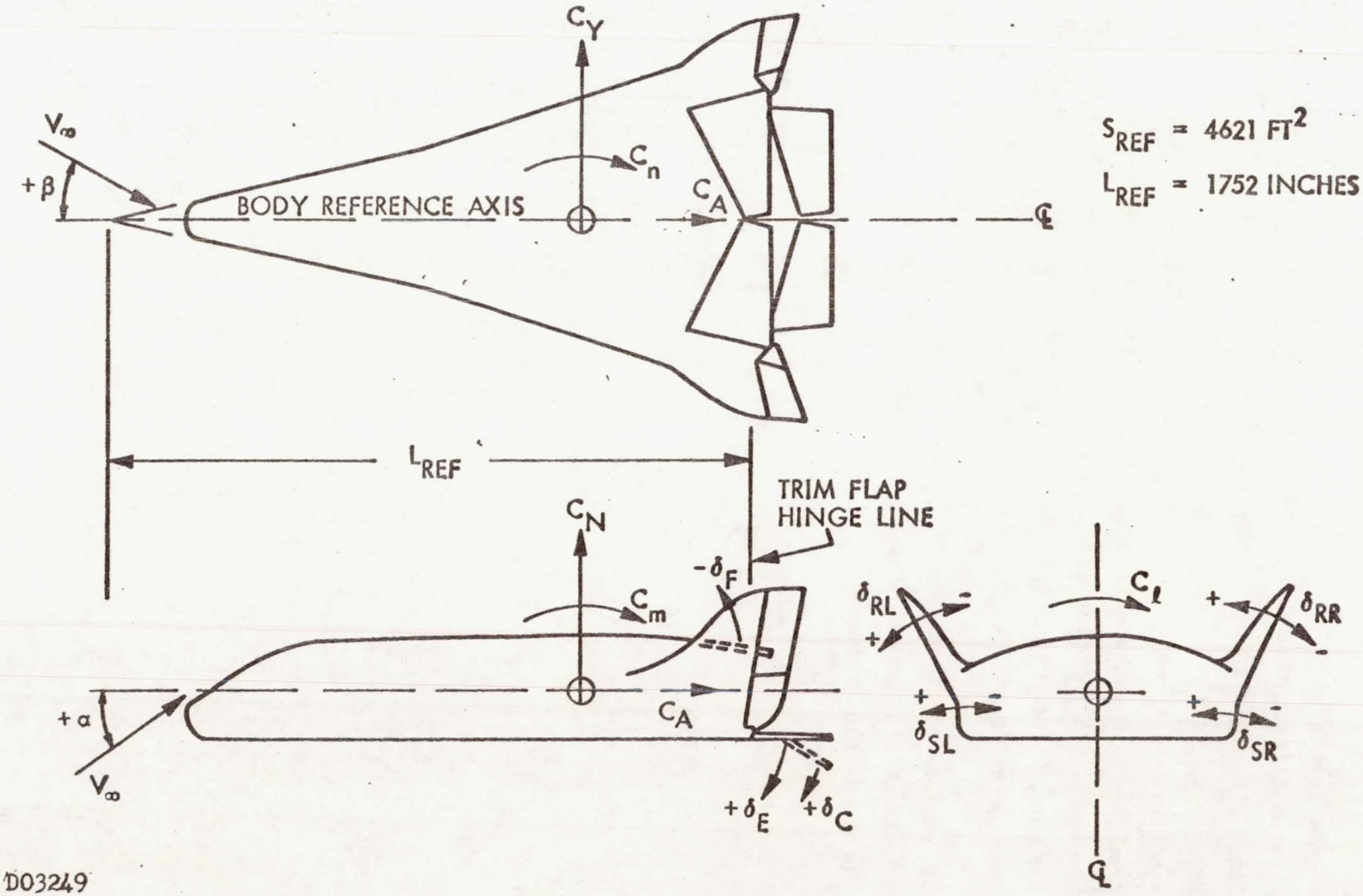
Maximum trimmed lift-to-drag ratio is presented versus Mach number for the cg extremes. Full-scale subsonic conditions indicate trimmed L/D_{Max} of 5.10 and 5.85 and 1.60 and 1.65 for the hypersonic conditions (see Fig. 3.5-22). The hypersonic L/Ds are representative of flight conditions at Mach 20 and 200,000 ft altitude. As previously discussed under hypersonic lift-and-drag characteristics (Fig. 3.5-18), the maximum trimmed L/D of the LS 400-7A orbiter is estimated at 1.87 rather than 1.65.

The most descriptive summary plots illustrating the delta-body orbiter flight capability are shown in Figs. 3.5-23 and 3.5-24. These plots clearly illustrate the trimmed angle-of-attack range where neutral or better static stability are achieved by indicating the boundaries where neutral stability occurs. Also, the figures indicate areas where the boundaries are determined by criteria other than stability – e.g., pitch trim authority. As can be observed from both figures, there is a sizable operating corridor throughout the entry Mach numbers where the vehicle is statically stable and trimmable. There are "NO OPERATING AREAS OF AERODYNAMIC INSTABILITY."

Figure 3.2-25 is a typical reference entry trajectory for the high-crossrange mission illustrating angle-of-attack, Mach number, altitude, and crossrange. Since the majority of crossrange is attained at hypersonic speeds, considerable freedom is available to tailor the trajectories below Mach 6.0 to conform with the aforementioned aerodynamic operating corridor. Figures 3.5-26 and 3.5-27 present the pitch stability margin and the directional stability summaries of the delta-body orbiter for this particular reference entry trajectory. As is evident, the configuration is longitudinally and directionally stable and attains at least 1100-nm crossrange for all cases.

In conclusion it can be stated that the Lockheed delta lifting-body orbiter has excellent aerodynamic characteristics and is acceptable as a candidate for the Space Shuttle Orbiter.

ORBITER BODY AXIS SYSTEM



D03249

Fig. 3.5-1

LOW-SPEED WIND TUNNEL TEST

- LOCKHEED 8 FT x 12 FT LOW-SPEED WIND TUNNEL
- TEST CONDITIONS
  - MACH NUMBER = 0.23
  - ANGLE-OF-ATTACK = $-4 \longrightarrow 26^{0}$
  - SIDESLIP ANGLE = $-4 \longrightarrow 10^{0}$
  - REYNOLDS NUMBER = 1.7×10^{6} PER FT
- TEST DURATION = 40 HR – 60 RUNS
- CONFIGURATION PARAMETERS
  - TRIM FLAP DEFLECTION
  - LOWER ELEVONS DEFLECTION
  - UPPER ELEVONS DEFLECTION
  - RUDDER DEFLECTION
  - DRAG BRAKE DEFLECTION
  - TRIM FLAP
  - BOAT-TAILING
  - CENTER FIN
  - UPPER ELEVON LOCATION
  - GROUND PLANE

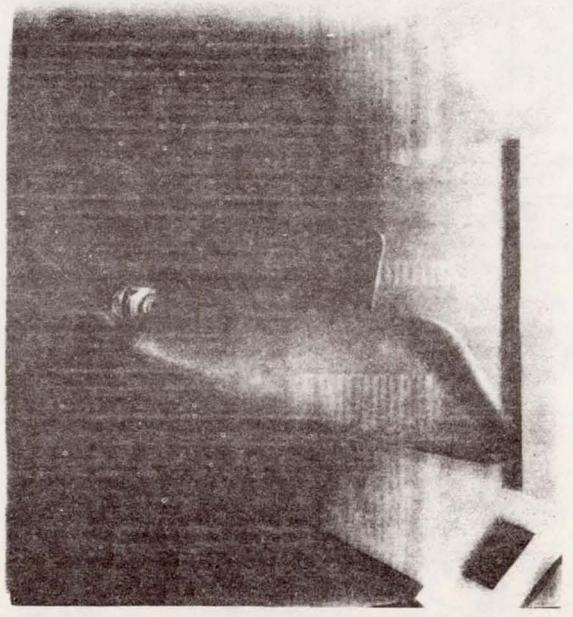
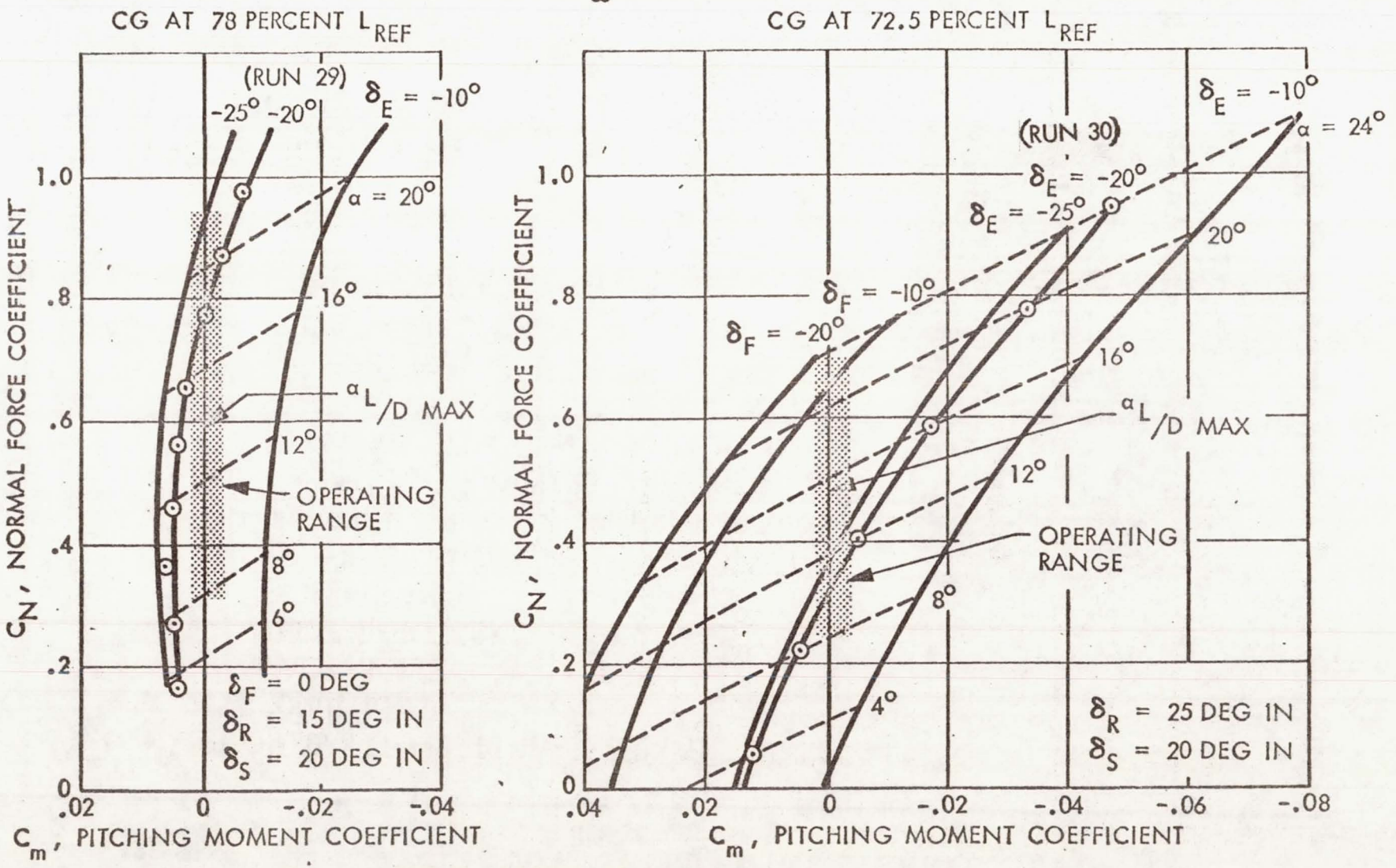


Fig. 3.5-2
3.5-9

fav
3-2-71
DQ2552 (1)

$M_\infty = 0.23$*



*LOCKHEED 8 FT X 12 FT WIND TUNNEL TEST L-324

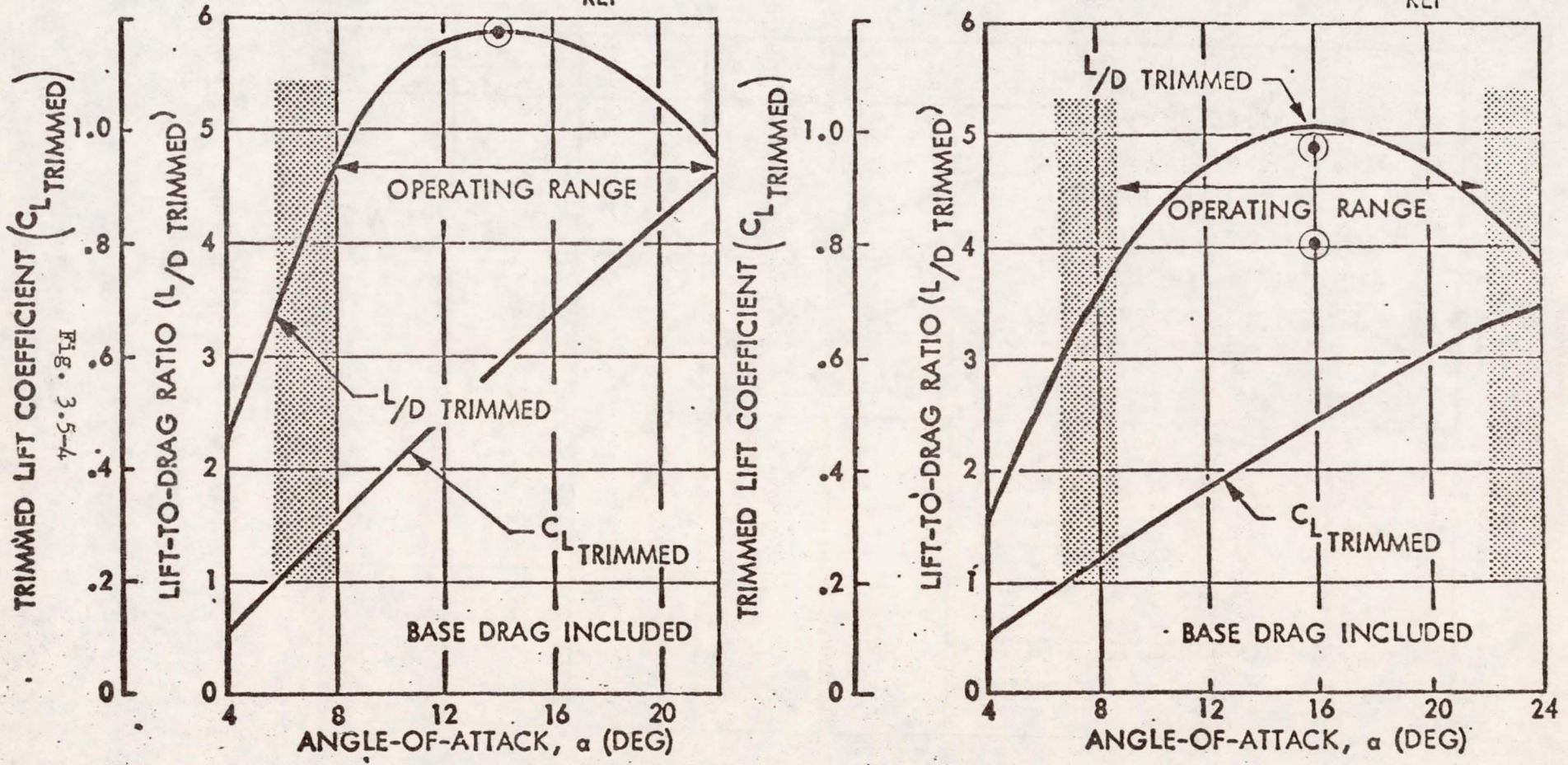
Fig. 3.5-3

3.5-10

D03266

LIFT AND DRAG CHARACTERISTICS

$M_\infty = 0.23$*



*LOCKHEED 8 FT X 12 FT WIND TUNNEL TEST NO. L-324

◉ LANGLEY DATA

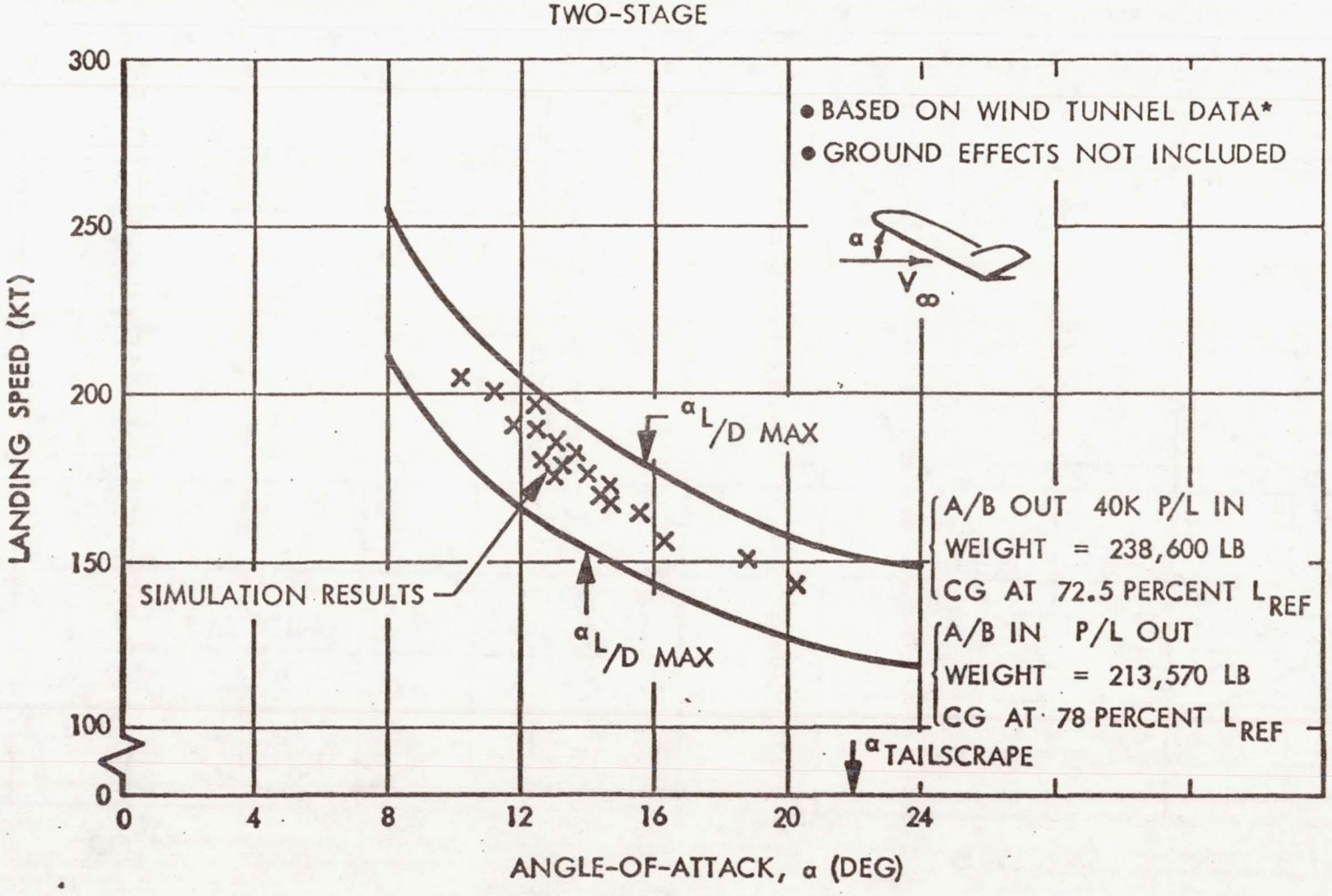
Fig. 3.5-4

3.5-11

D03267 (2)

LMSC-A959142 Vol II

LANDING SPEED AND ATTITUDE



*LOCKHEED 8 FT X 12 FT WIND TUNNEL TEST NO. L-324

Fig. 3.5-5
3.5-12

D03068

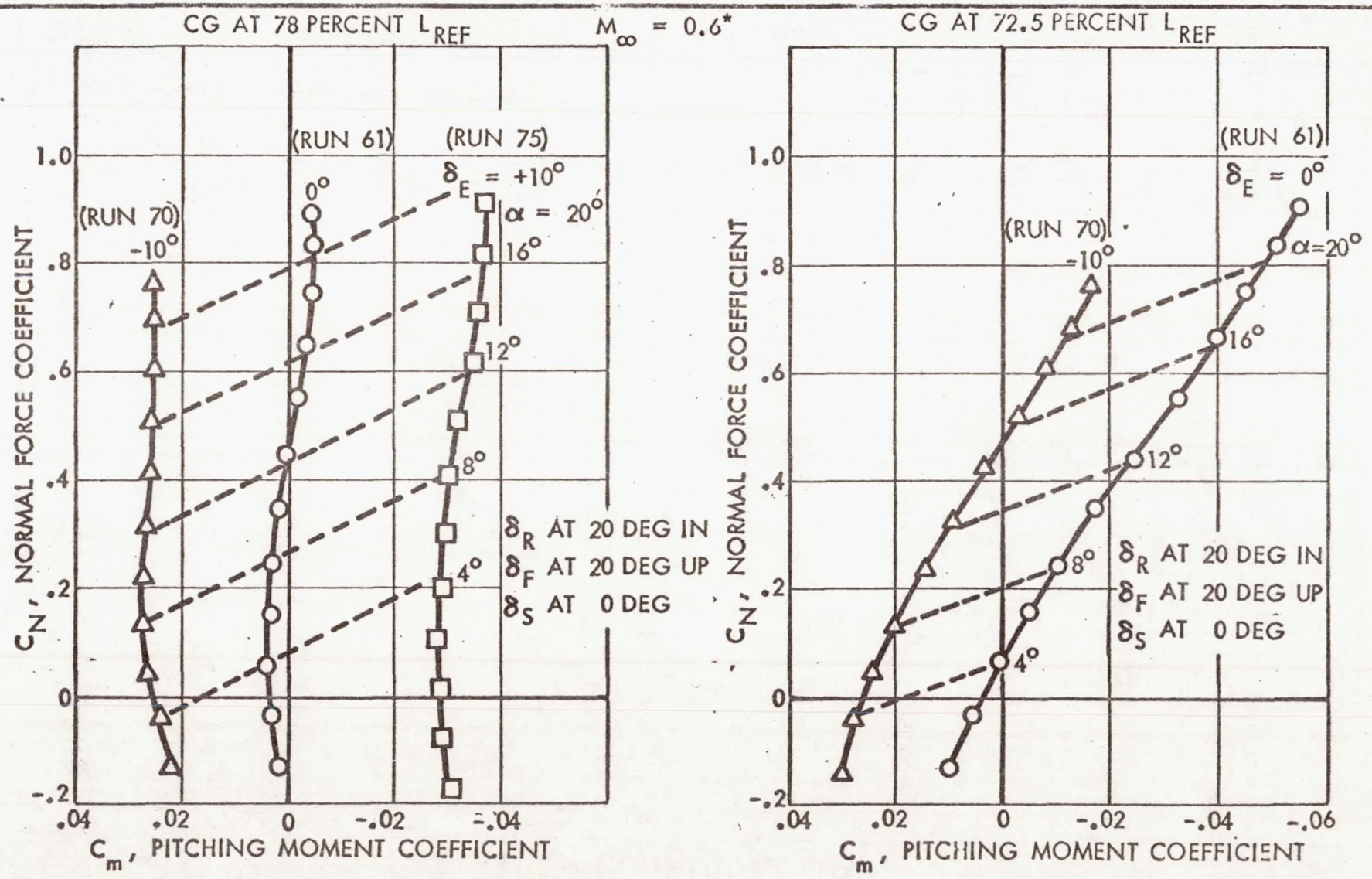
ORBITER CONFIGURATION

	ORBITER	
TEST FACILITY	AMES 6 x 6 FT	LaRC-UPWT
RUNS	67	84
MACH NUMBER	0.60 to 2.0	2.3 TO 4.6
ANGLE-OF-ATTACK (DEG)	-4 TO 22	0 TO 60
SIDESLIP ANGLE (DEG)	-4 TO 10	-4 TO 10
REYNOLDS NUMBER PER FOOT	2.5×10^6 TO 4.0×10^6	2.5×10^6 TO 4.5×10^6

Fig. 3.5-6

3.5-13

D03010

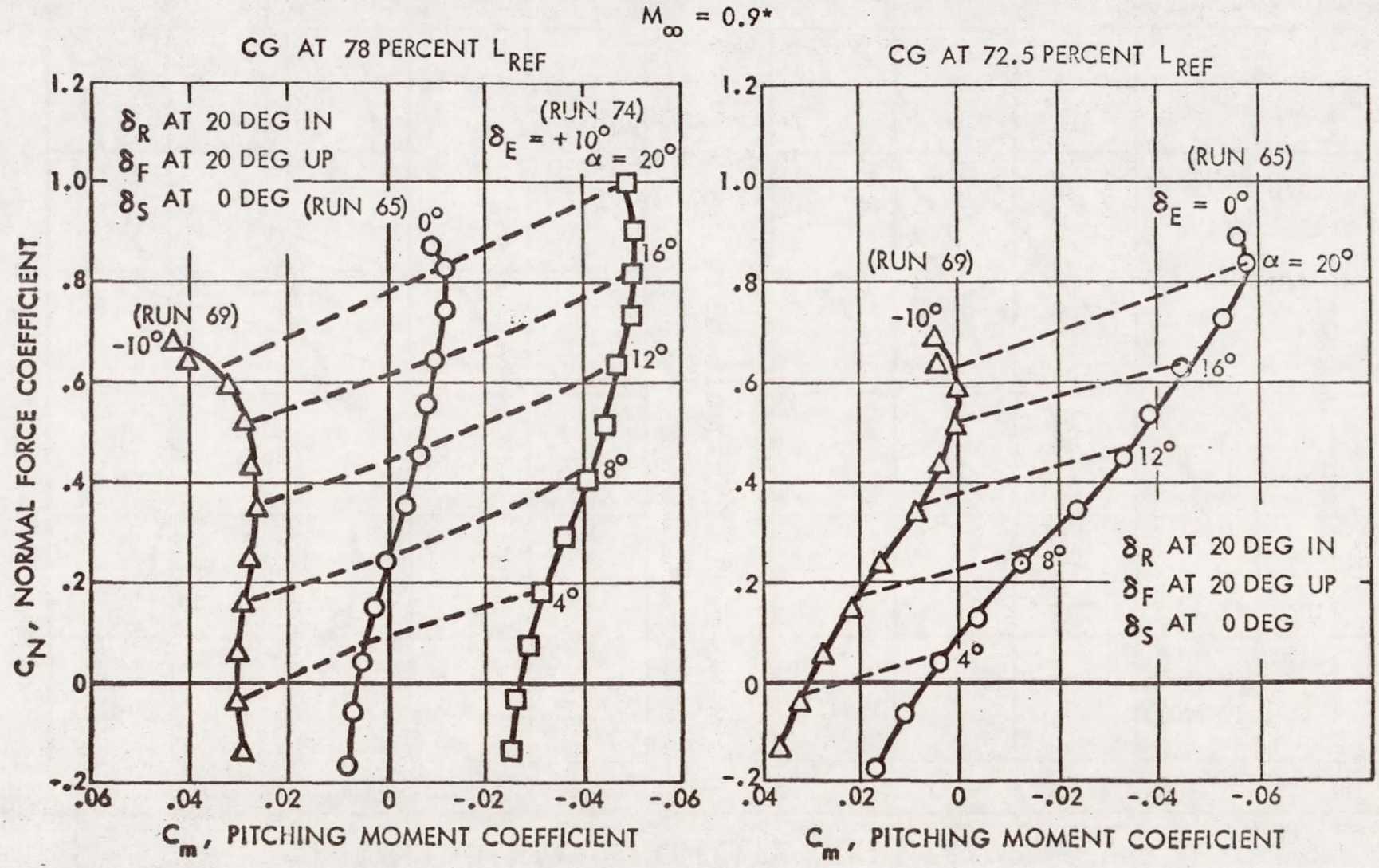


*AMES 6 FT X 6 FT WIND TUNNEL TEST NO. 542

D03268

Fig. 3.5-7

3.5-14

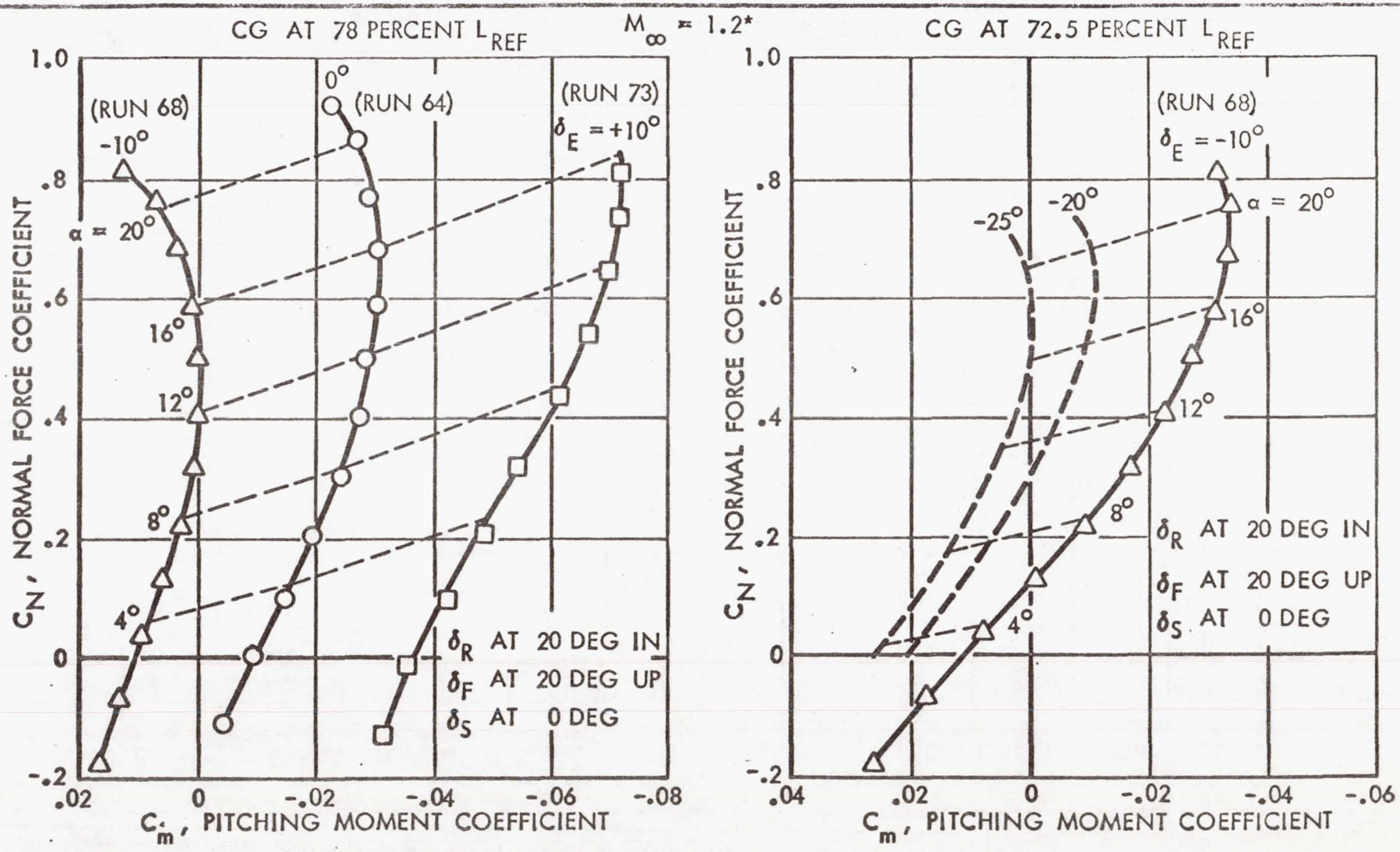


*AMES 6 FT X 6 FT WIND TUNNEL TEST NO. 542

Fig. 3.5-8

3.5-15

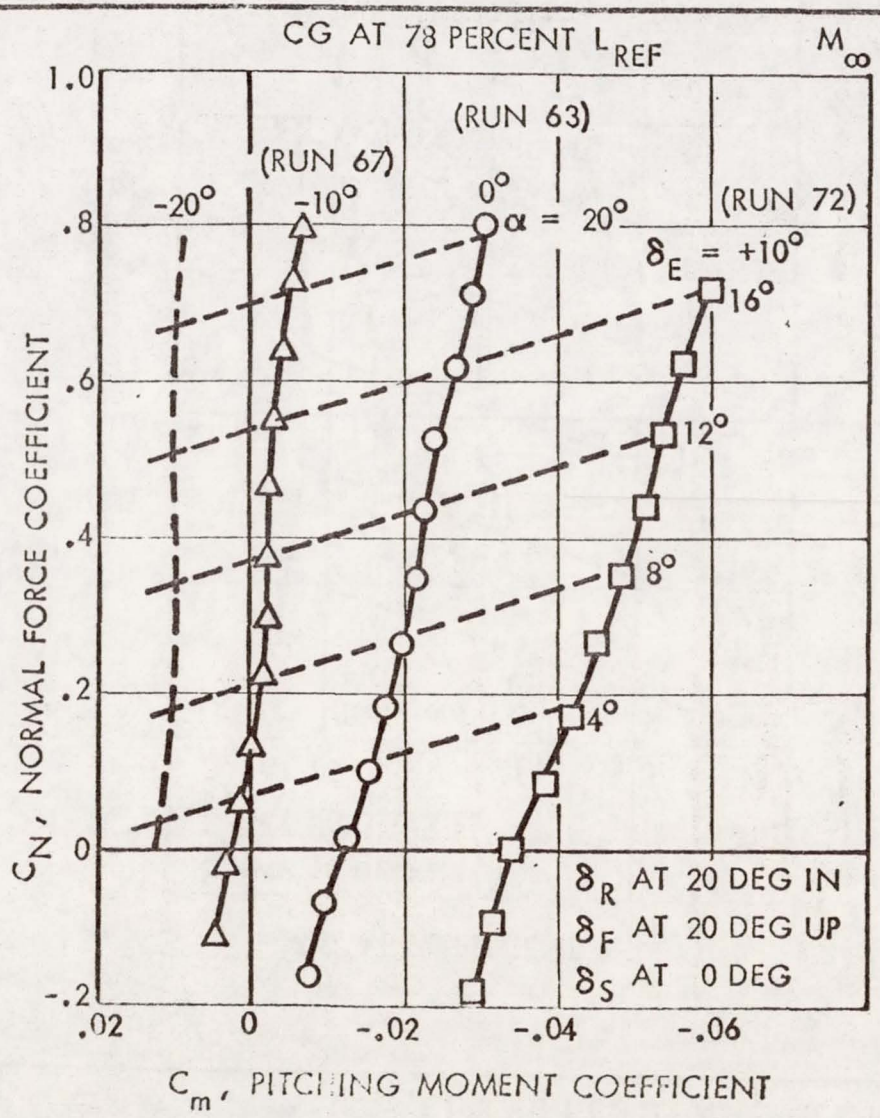
D03269

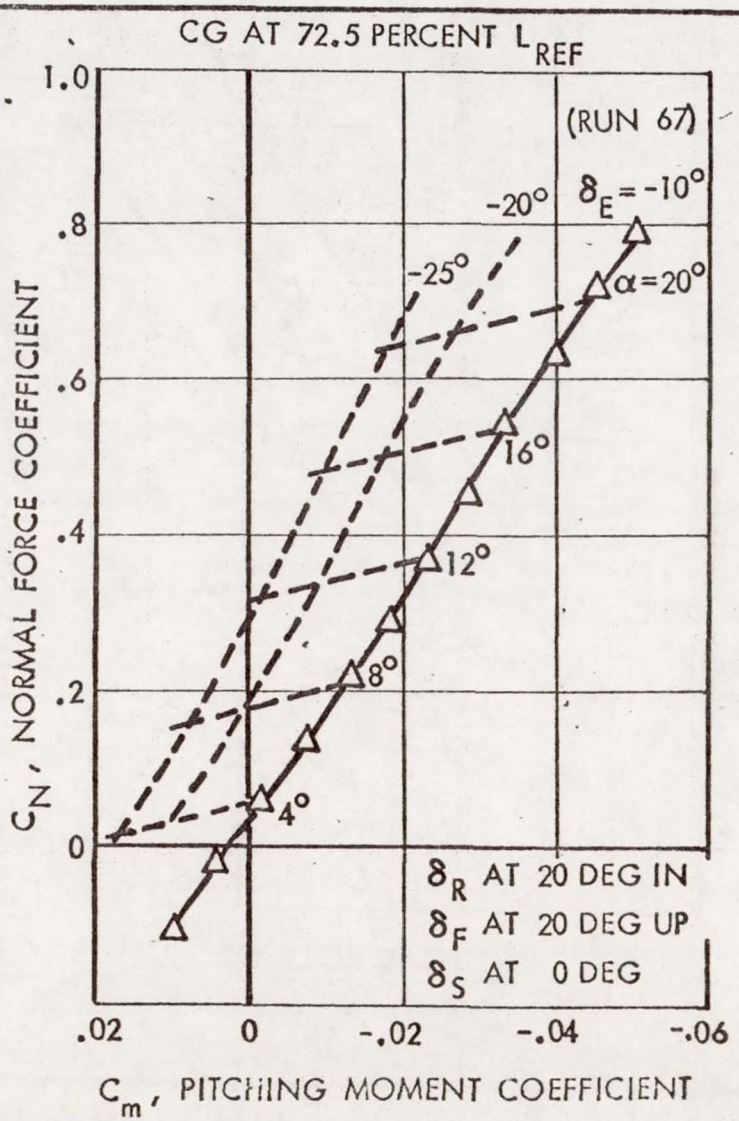


*AMES 6 FT X 6 FT WIND TUNNEL TEST NO. 542

D03270

Fig. 3.5-9





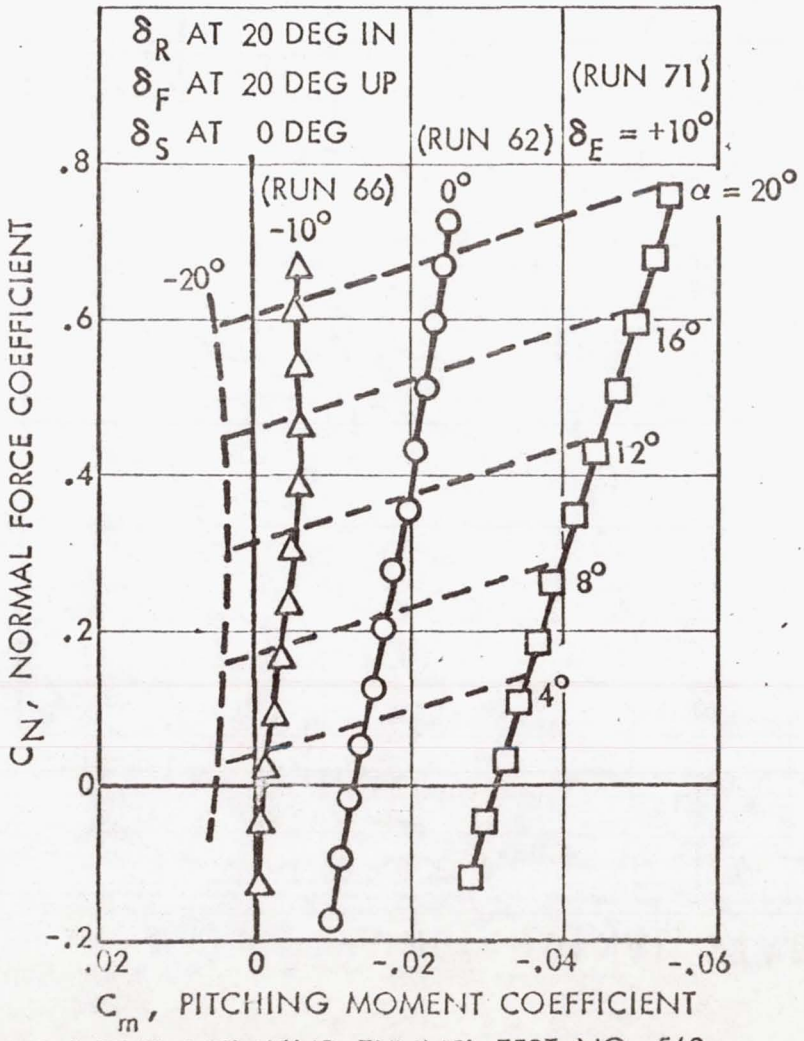
*AMES 6 FT X 6 FT WIND TUNNEL TEST NO. 542

D03271

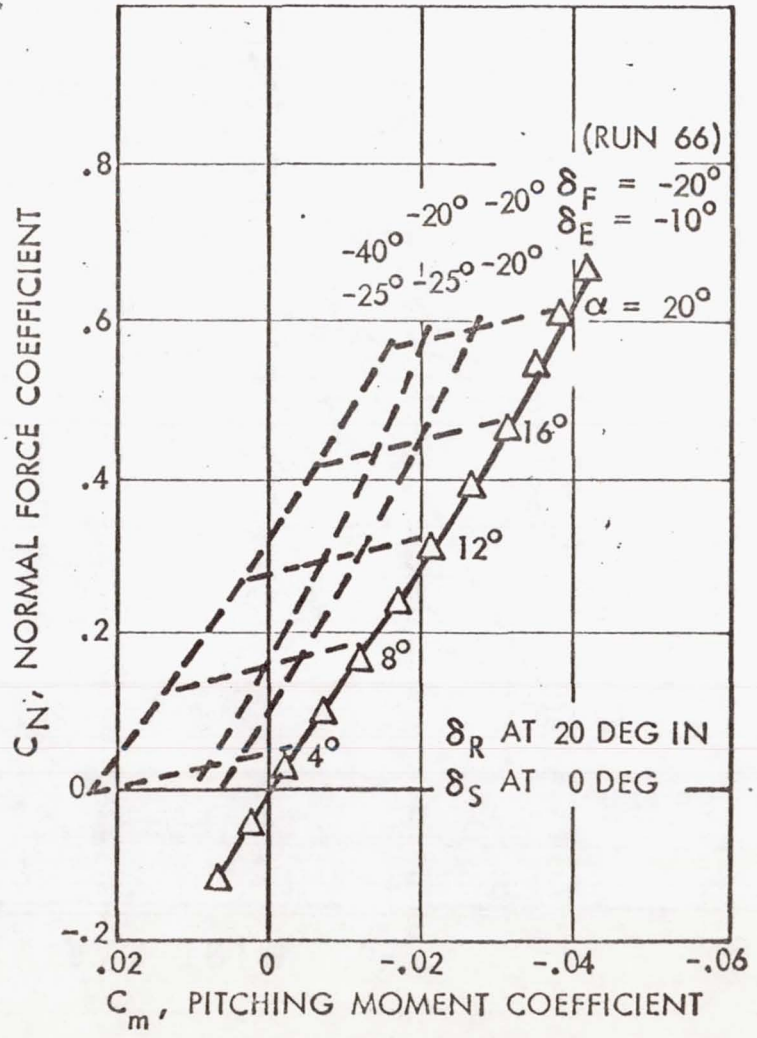
Fig. 3.5-10

$M_\infty = 2.0$*

CG AT 78 PERCENT L_{REF}



CG AT 72.5 PERCENT L_{REF}

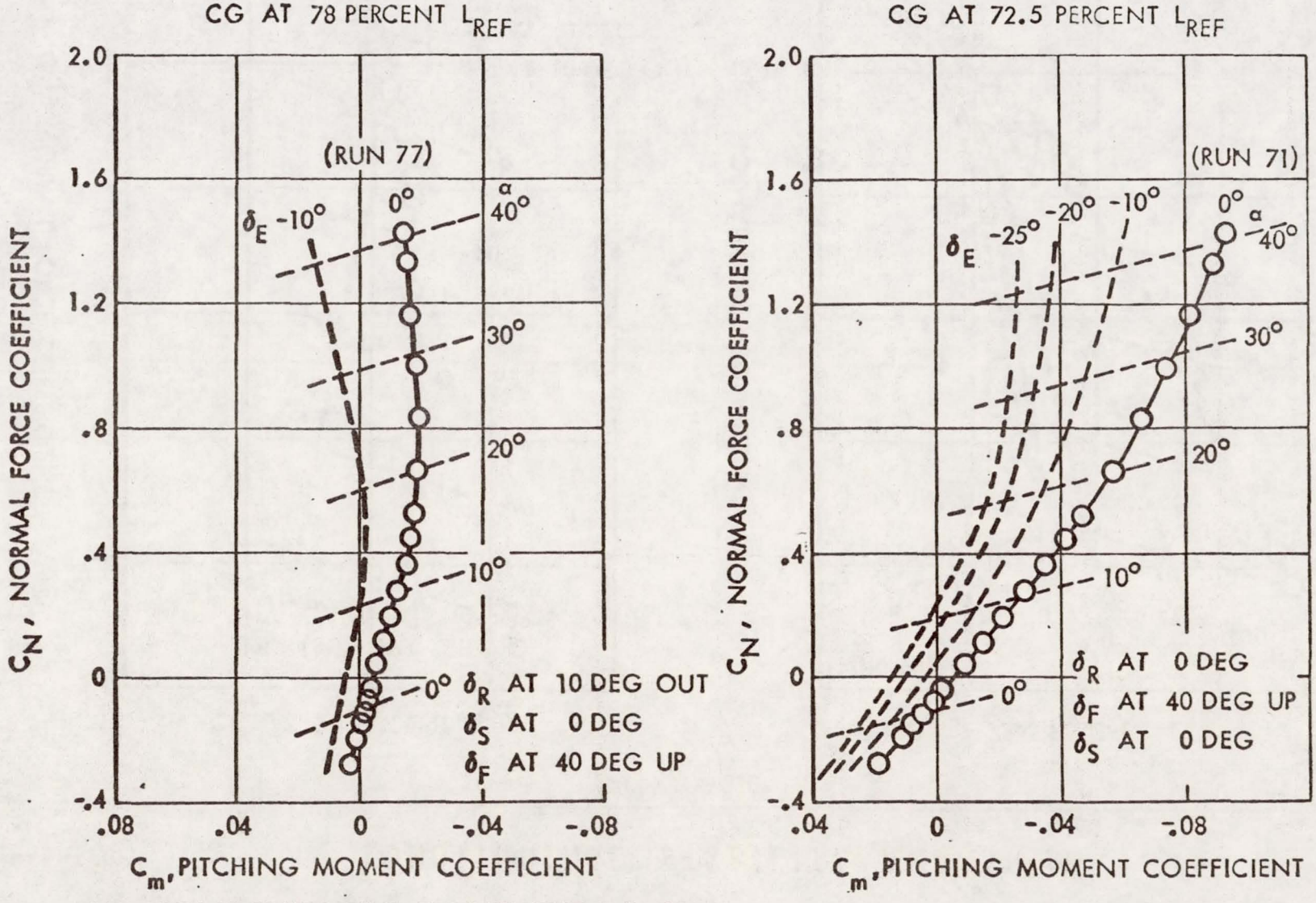


*AMES 6 FT X 6 FT WIND TUNNEL TEST NO. 542

D03272

Fig. 3.5-11

M_∞ = 2.3*

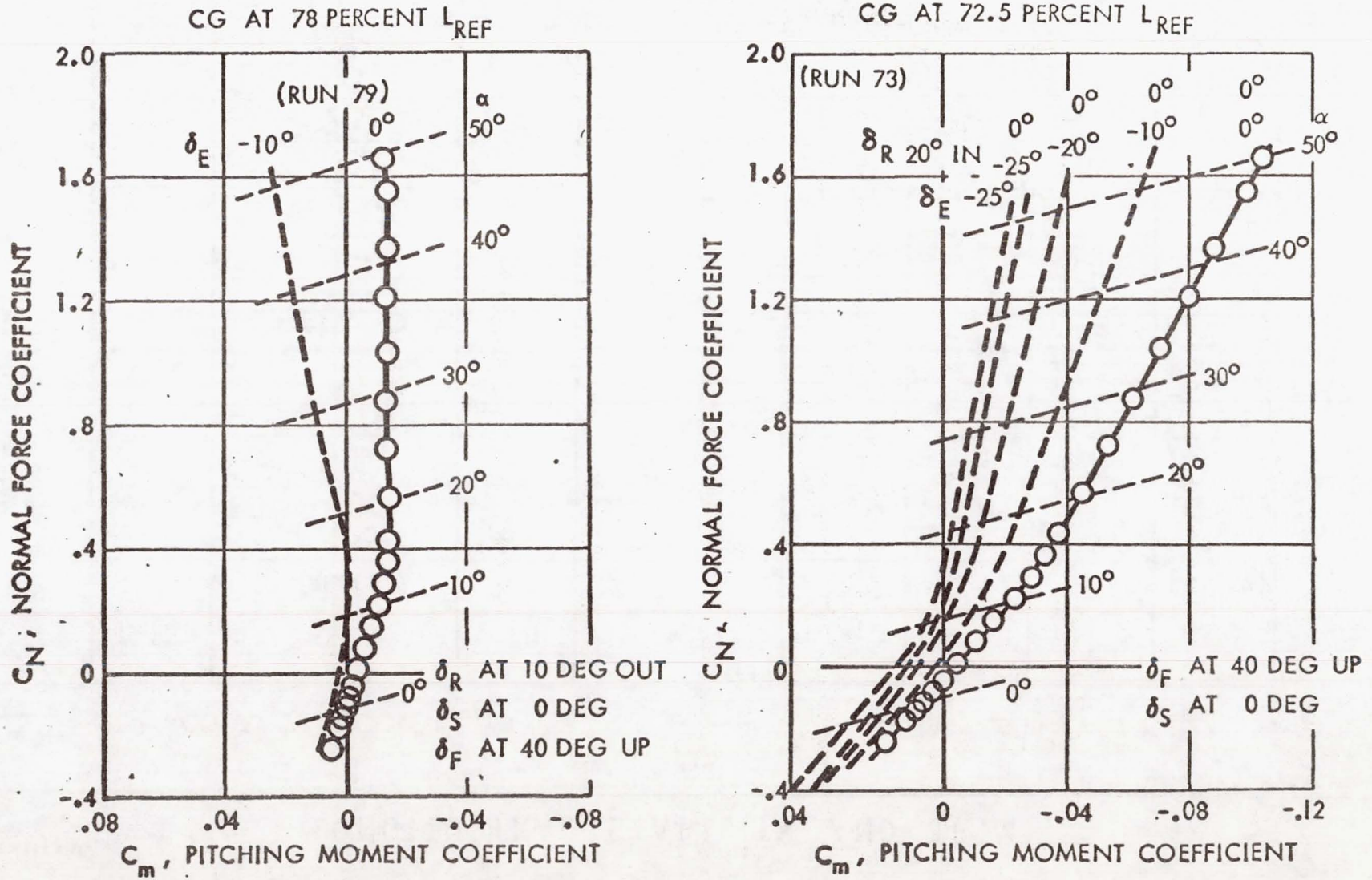


*LANGLEY UNITARY WIND TUNNEL TEST NO. 955

D03273

Fig. 3.5-12

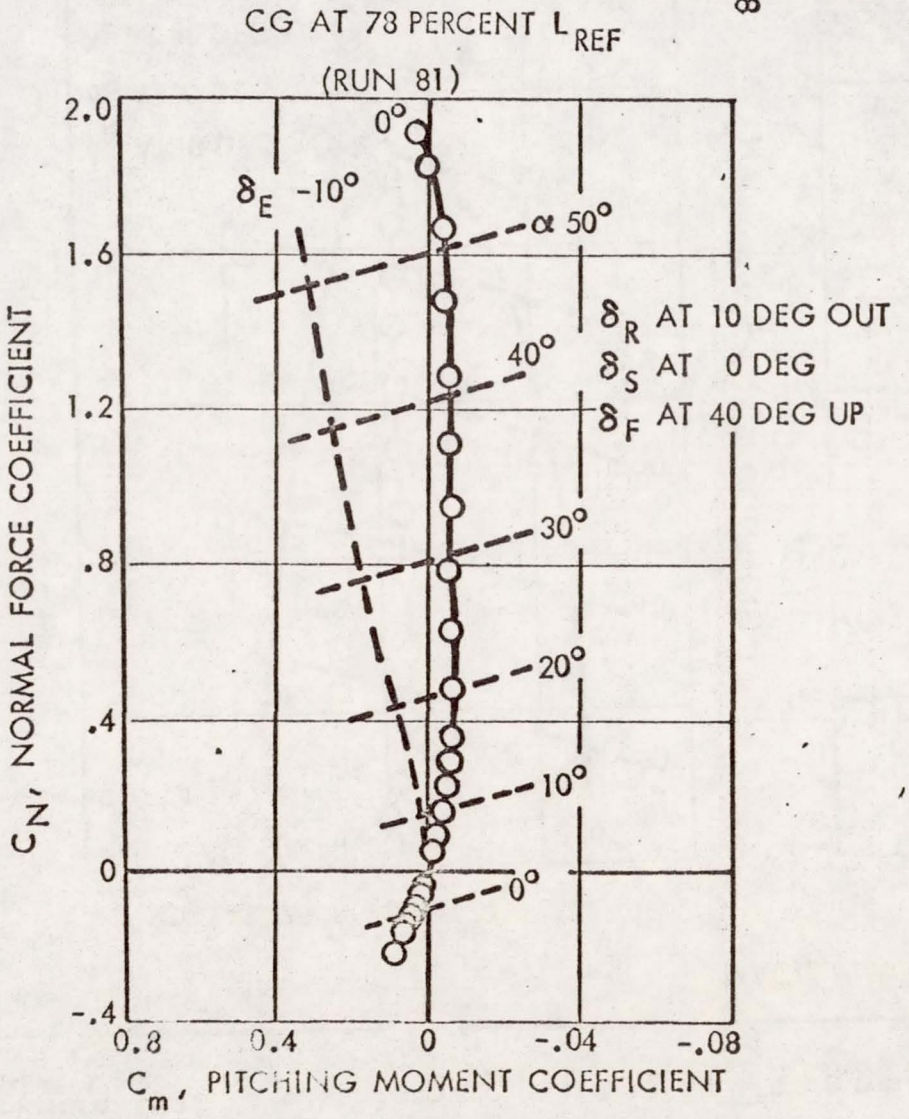
$M_\infty = 3.0$*

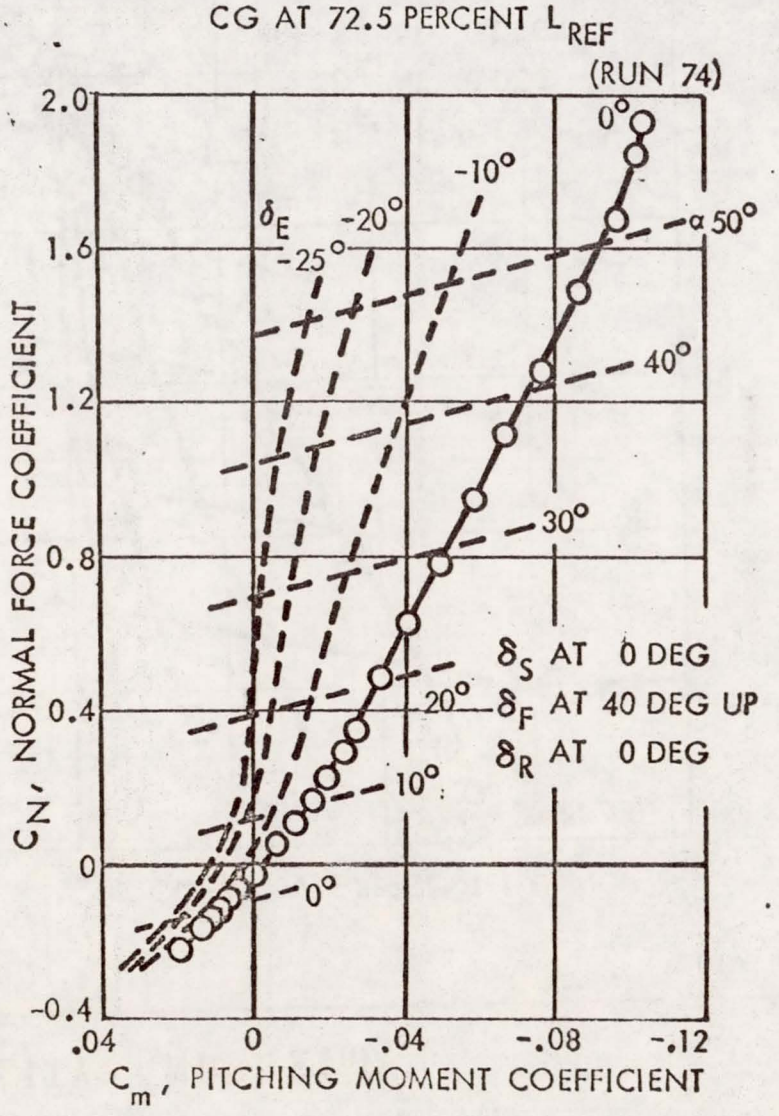


*LANGLEY UNITARY WIND TUNNEL TEST NO. 955

D03274

M_∞ = 4.0*





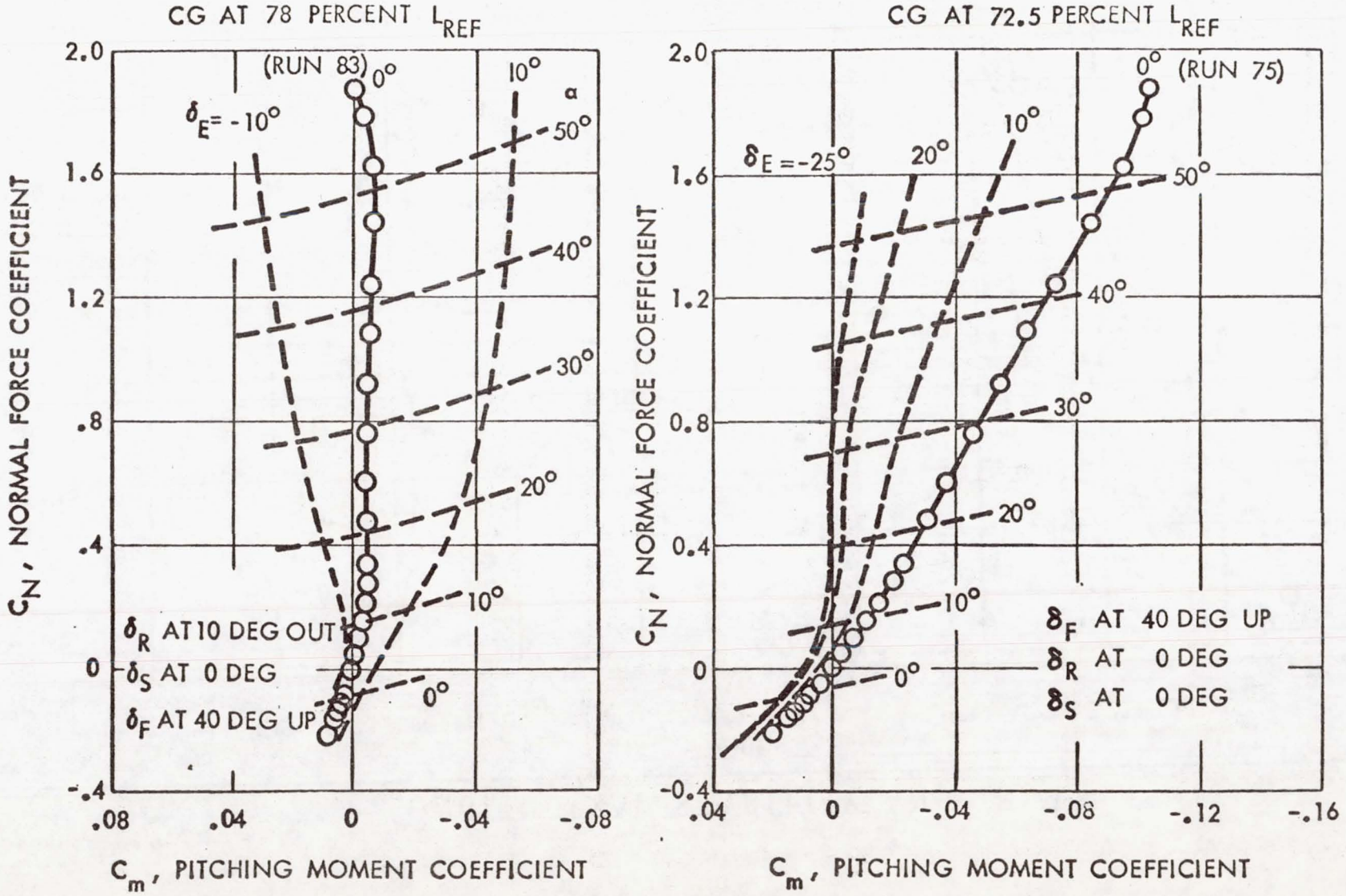
*LANGLEY UNITARY WIND TUNNEL TEST NO. 955

D03275

Fig. 3.5-14

3.5-21

MACH 4.6*



*LANGLEY UNITARY WIND TUNNEL TEST NO. 955

DELTA LIFTING BODY ORBITER

(HYPERSONIC ARBITRARY BODY COMPUTER PROGRAM)
(LS-400-7A)

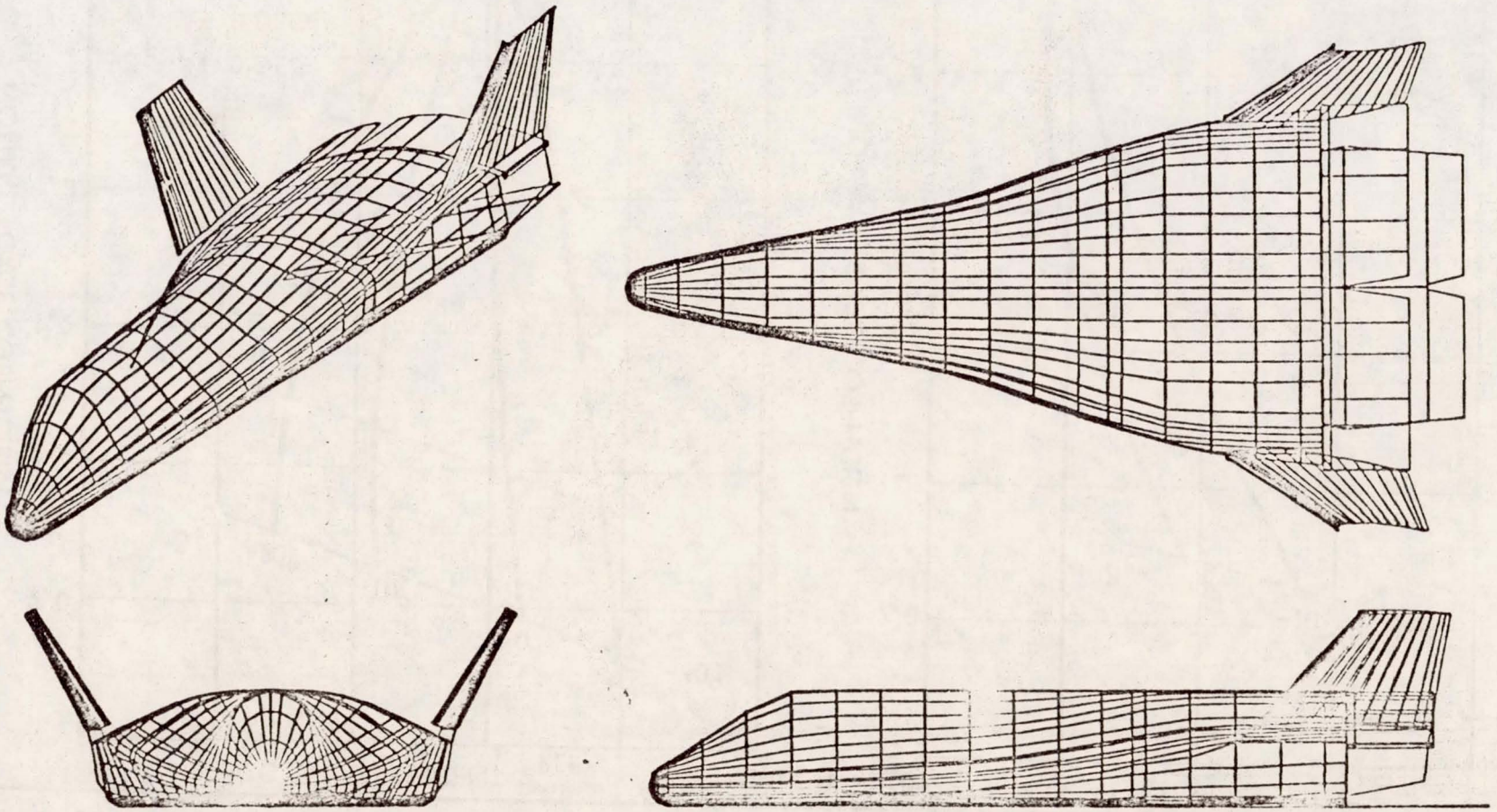
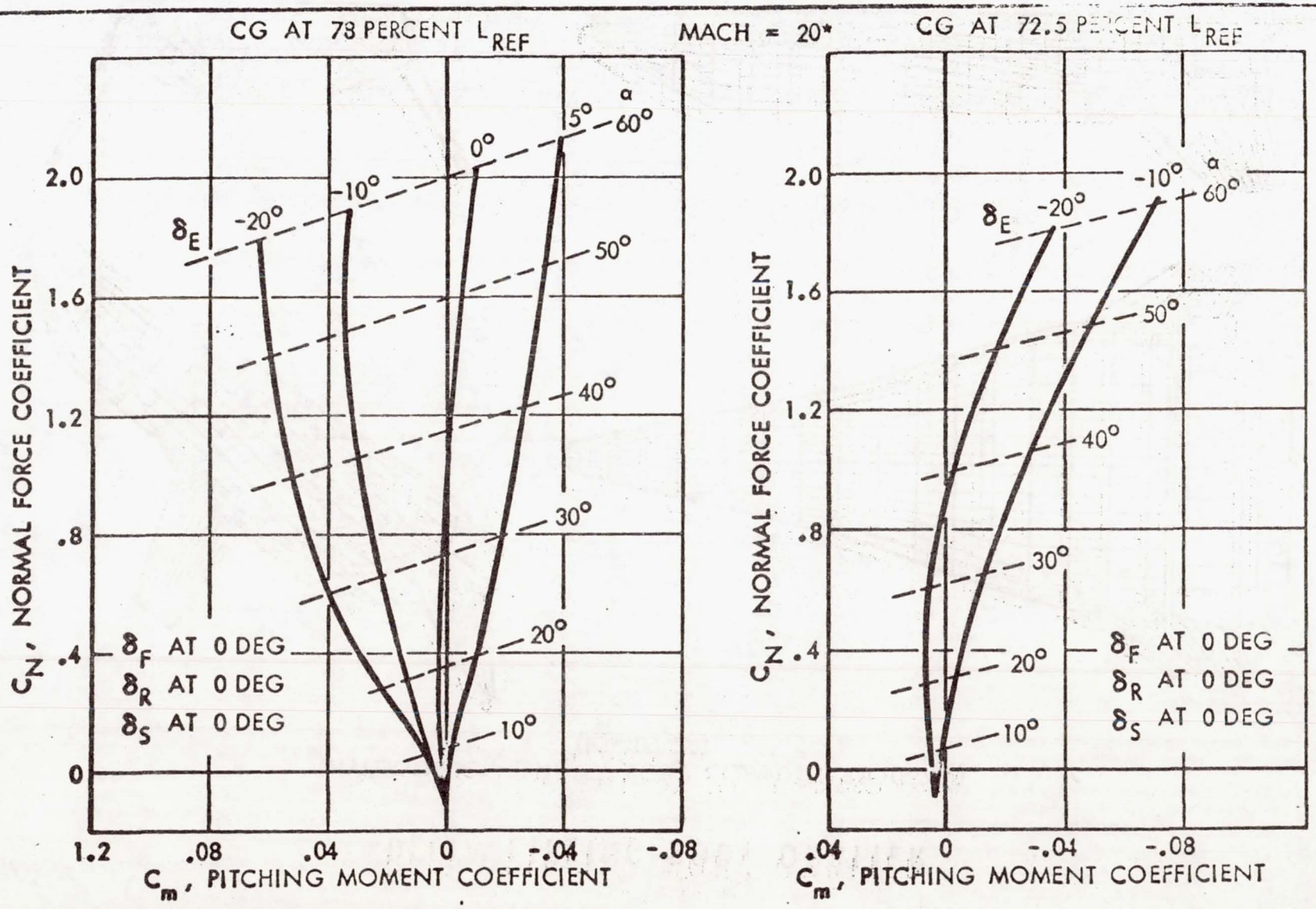


Fig. 3.5-16



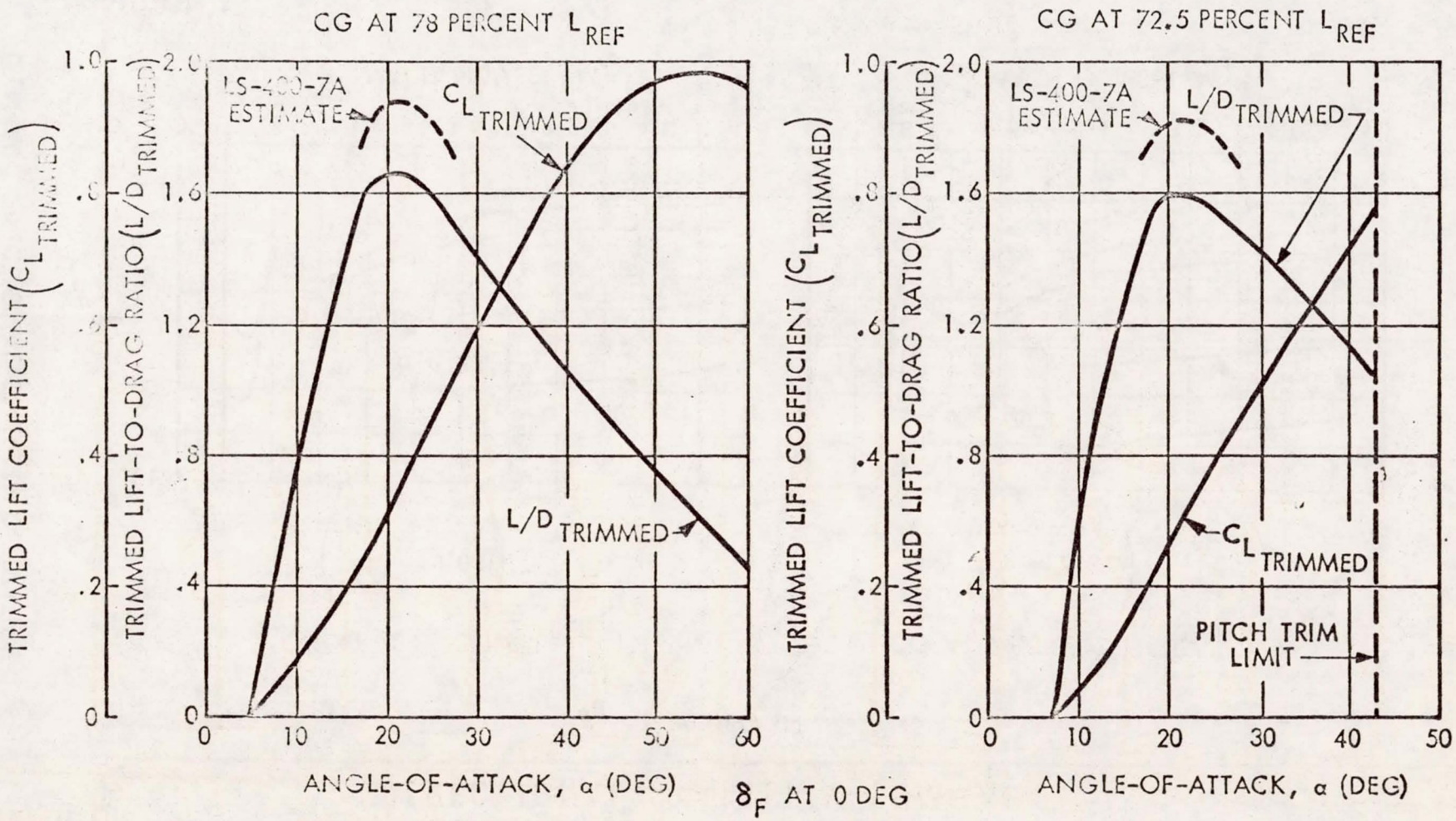
*MODIFIED HYPERSONIC ARBITRARY BODY PROGRAM

Fig. 3.5-17

3.5-24

LIFT AND DRAG CHARACTERISTICS

MACH = 20*



δ_F AT 0 DEG
δ_R AT 0 DEG
δ_S AT 0 DEG

*MODIFIED HYPERSONIC ARBITRARY BODY PROGRAM

Fig. 3.5-18

3.5-25

D03278

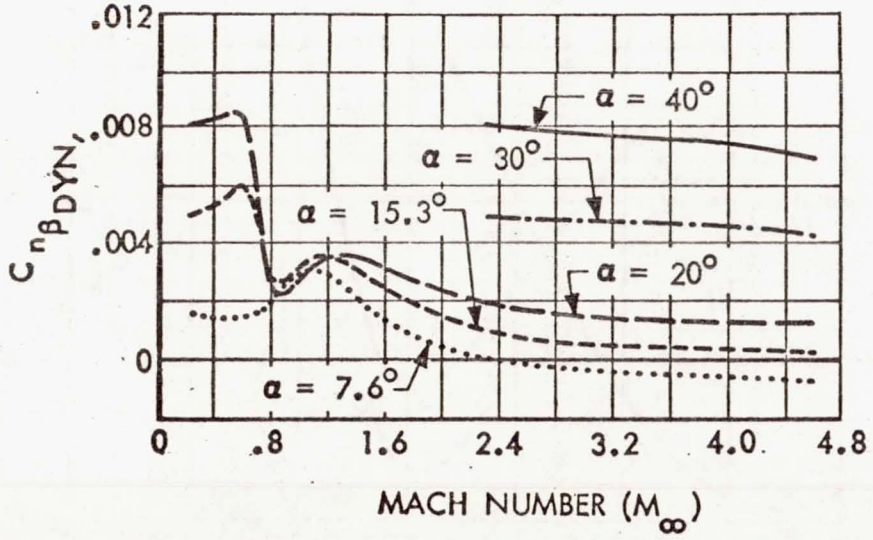
DIRECTIONAL STABILITY SUMMARY

δ_R AT 20 DEG IN

DYNAMIC DIRECTIONAL STABILITY DERIVATIVE*

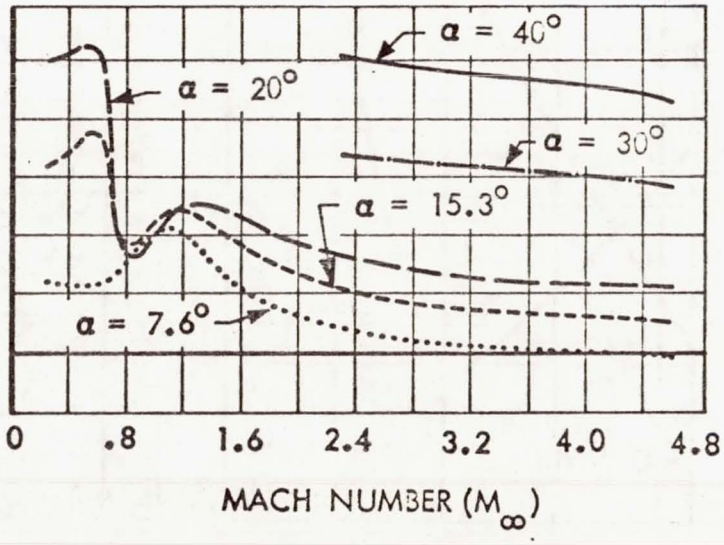
CG AT 78 PERCENT L_{REF}

$\left(\frac{I_{ZZ}}{I_{XX}} = 5.081\right)$



CG AT 72.5 PERCENT L_{REF}

$\left(\frac{I_{ZZ}}{I_{XX}} = 5.829\right)$



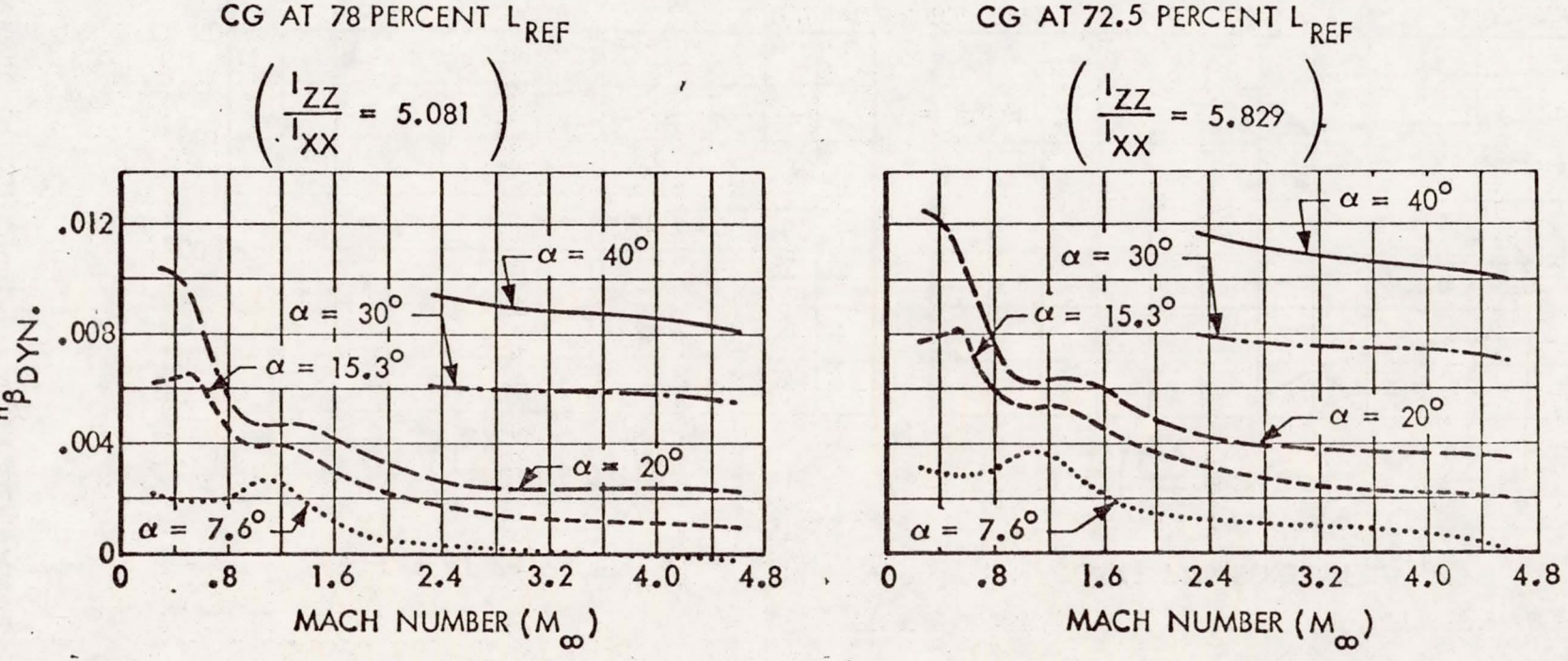
*BASED ON L_{REF}

- LOCKHEED 8 FT X 12 FT TEST L-324
- AMES 6 FT X 6 FT TEST 542
- LANGLEY UPWT TEST 955

Fig. 3.5-19
3.5-26

DIRECTIONAL STABILITY SUMMARY

δ_R AT 0 DEG



*BASED ON L_{REF}

- LOCKHEED 8 FT X 12 FT TEST L-324
- AMES 6 FT X 6 FT TEST 542
- LANGLEY UPWT TEST 955

D03280

Fig. 3.5-20

3.5-27

LMSC-A989472
Vol II

DIRECTIONAL STABILITY SUMMARY

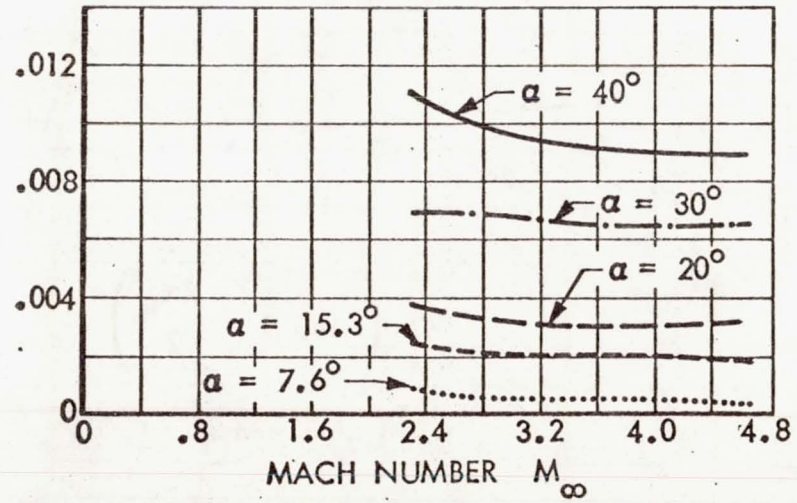
δ_R AT 10 DEG OUT

DYNAMIC DIRECTIONAL STABILITY DERIVATIVE *

$C_{n_{\beta_{DYN}}}$

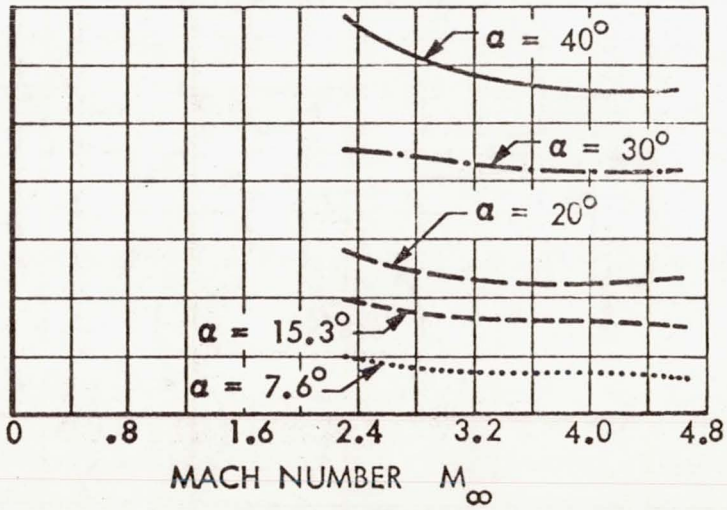
CG AT 78 PERCENT L'_{REF}

$\left(\frac{I_{ZZ}}{I_{XX}} = 5.081\right)$



CG AT 72.5 PERCENT L_{REF}

$\left(\frac{I_{ZZ}}{I_{XX}} = 5.829\right)$



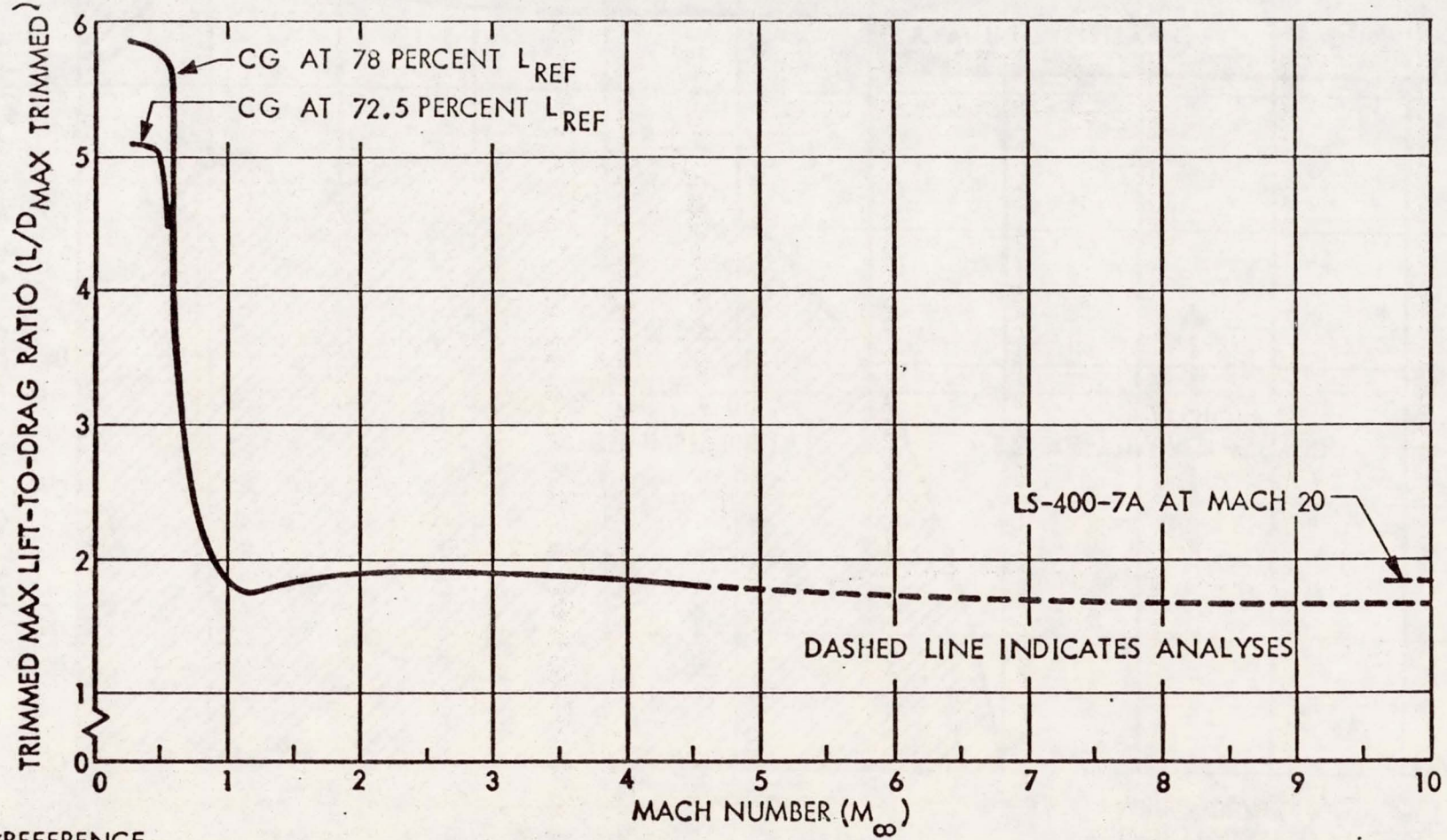
*BASED ON L_{REF}

- LOCKHEED 8 FT X 12 FT TEST L-324
- AMES 6 FT X 6 FT TEST 542
- LANGLEY UPWT TEST 955

D03281

AERODYNAMIC LIFT-TO-DRAG SUMMARY

TWO-STAGE*



*REFERENCE

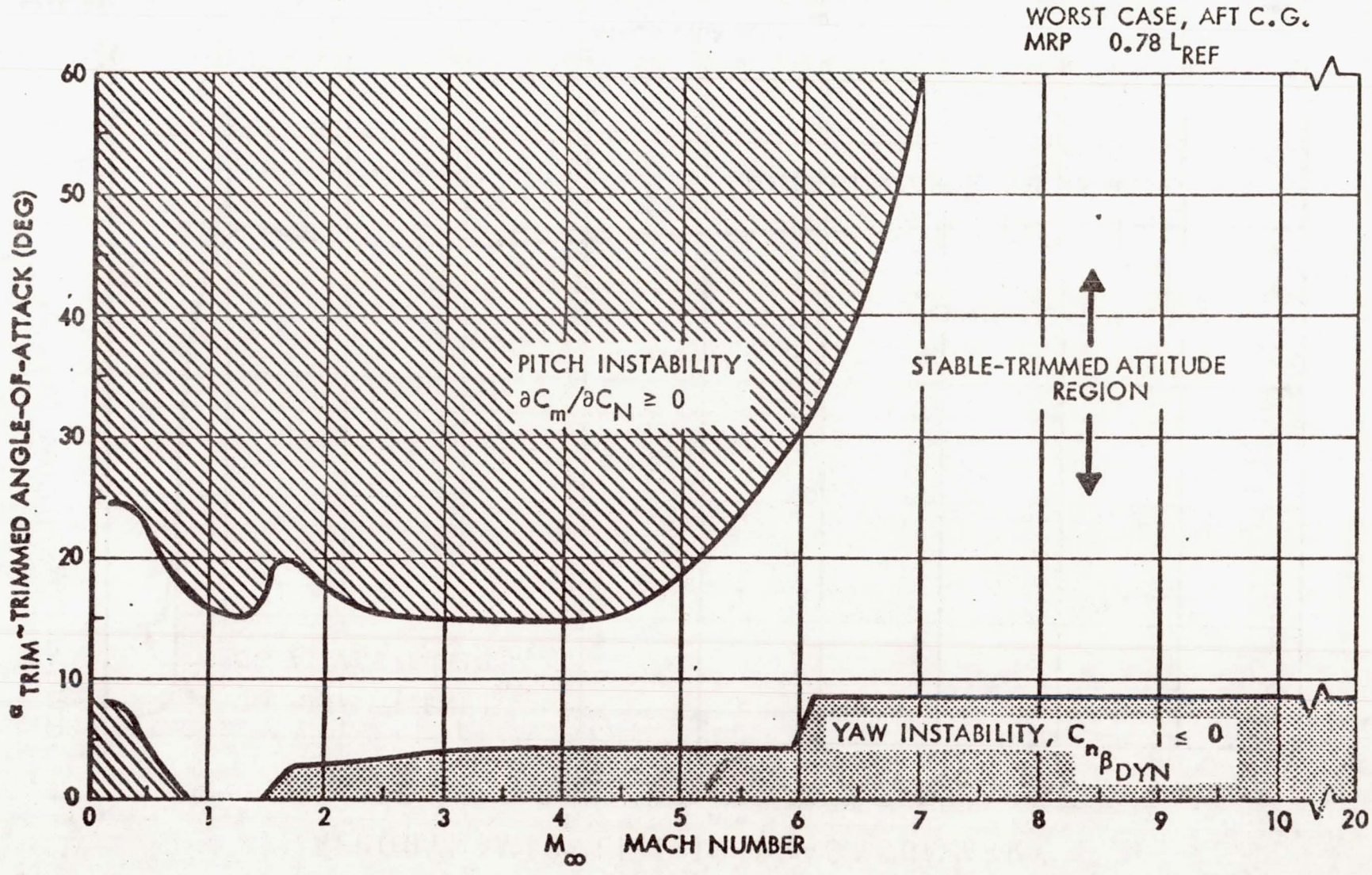
- LOCKHEED 8 FT X 12 FT TEST NO. L-324
- AMES 6 FT X 6 FT TEST NO. 542
- LANGLEY UPWT TEST NO. 955

Fig. 3.5-22

3.5-29

D03066

TRIM ANGLE-OF-ATTACK RANGE



D03015(1)

TRIM ANGLE-OF-ATTACK RANGE

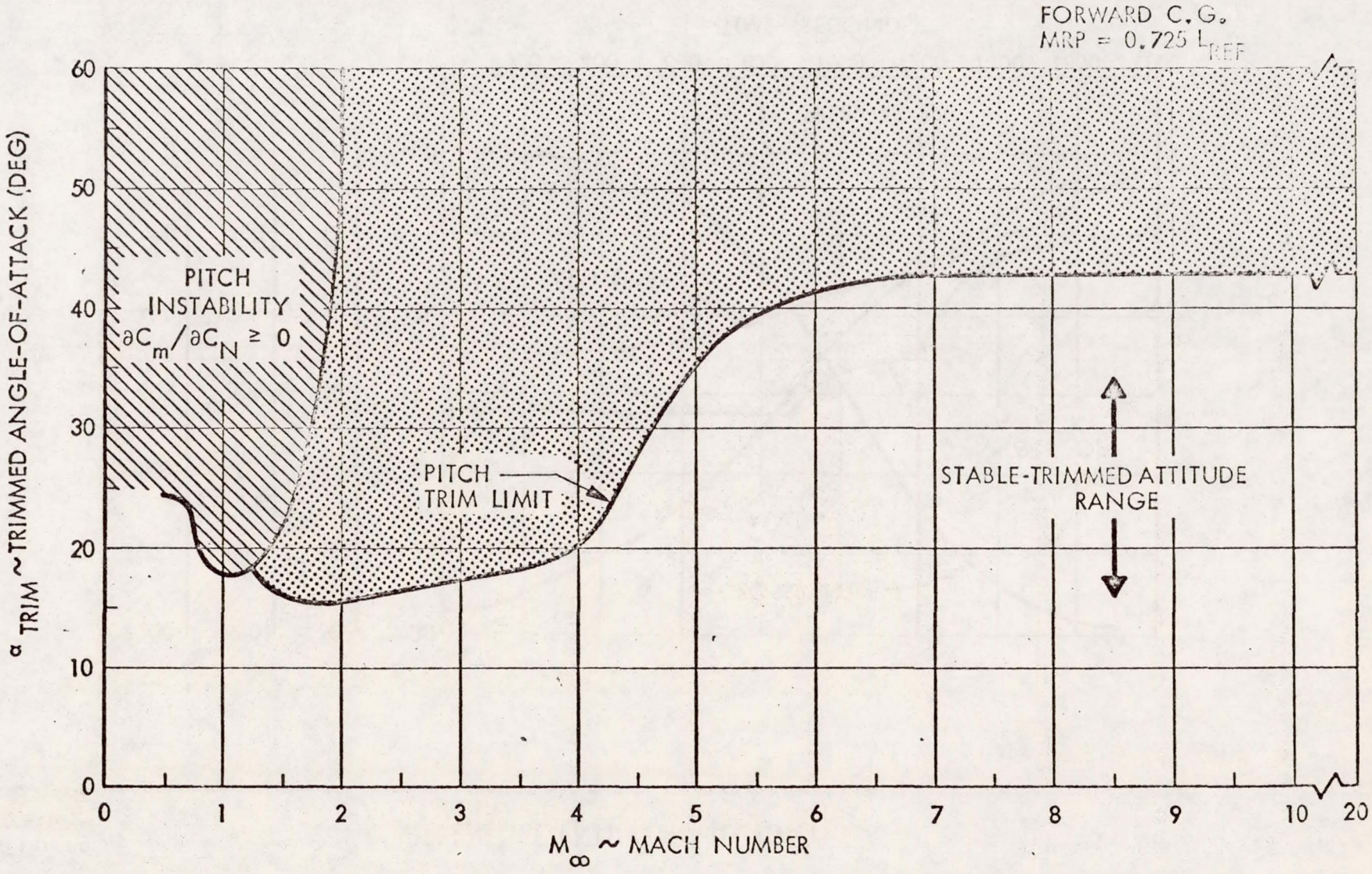


Fig. 3.5-24

D03013(1)

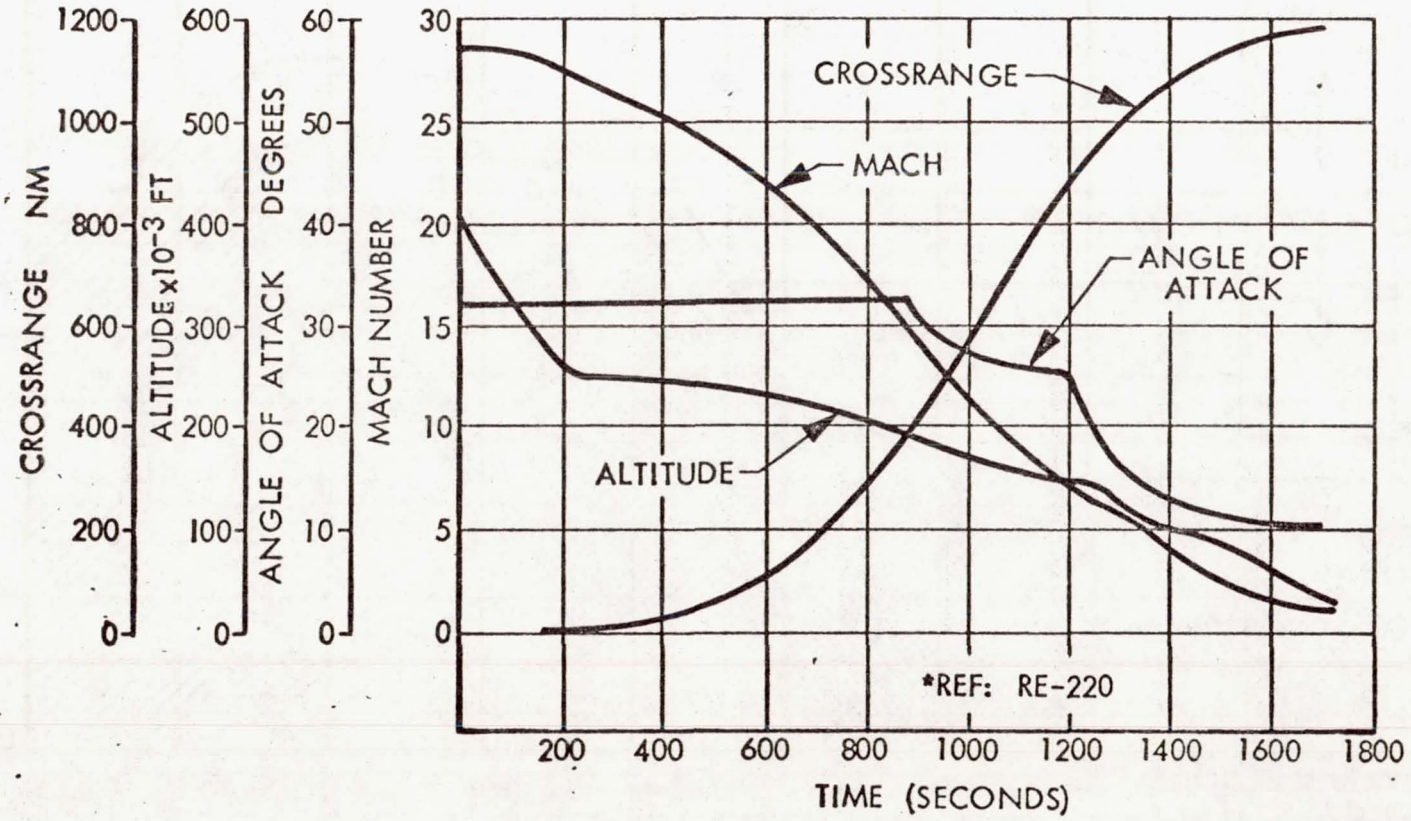
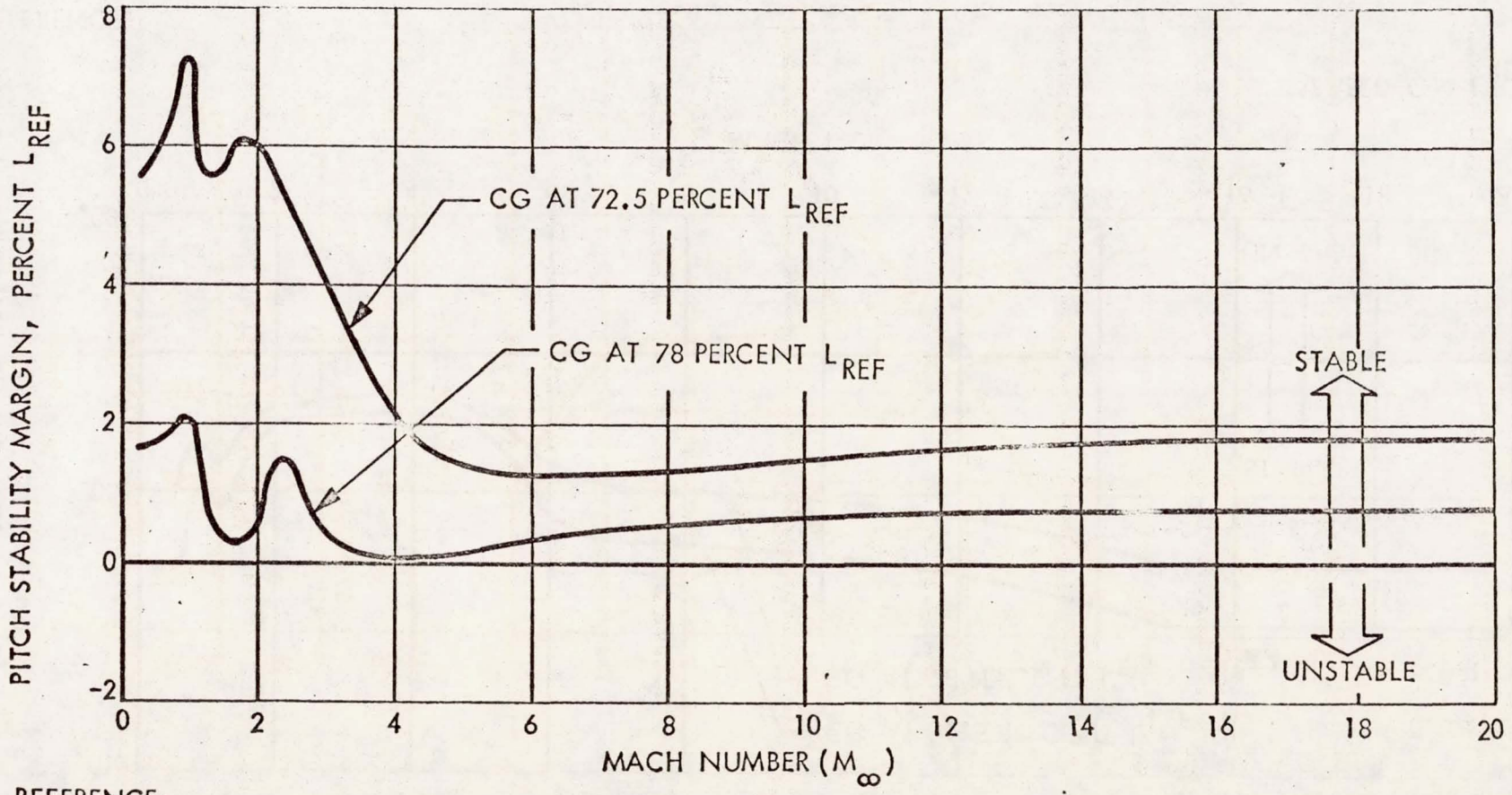

CROSSRANGE NM
1200
1000
800
600
400
200
0
ALTITUDE x10⁻³ FT
600
500
400
300
200
100
0
ANGLE OF ATTACK DEGREES
60
50
40
30
20
10
0
MACH NUMBER
30
25
20
15
10
5
0
CROSSRANGE
MACH
ANGLE OF ATTACK
ALTITUDE
*REF: RE-220
200
400
600
800
1000
1200
1400
1600
1800
TIME (SECONDS)

Fig. 3.5-25

LONGITUDINAL STABILITY SUMMARY

(RE 220 TRAJECTORY)



REFERENCE:

- LOCKHEED 8 FT X 12 FT TEST L-324
- AMES 6 FT X 6 FT TEST 542
- LANGLEY UPWT TEST 955

D03282

Fig. 3.5-26

DIRECTIONAL STABILITY SUMMARY

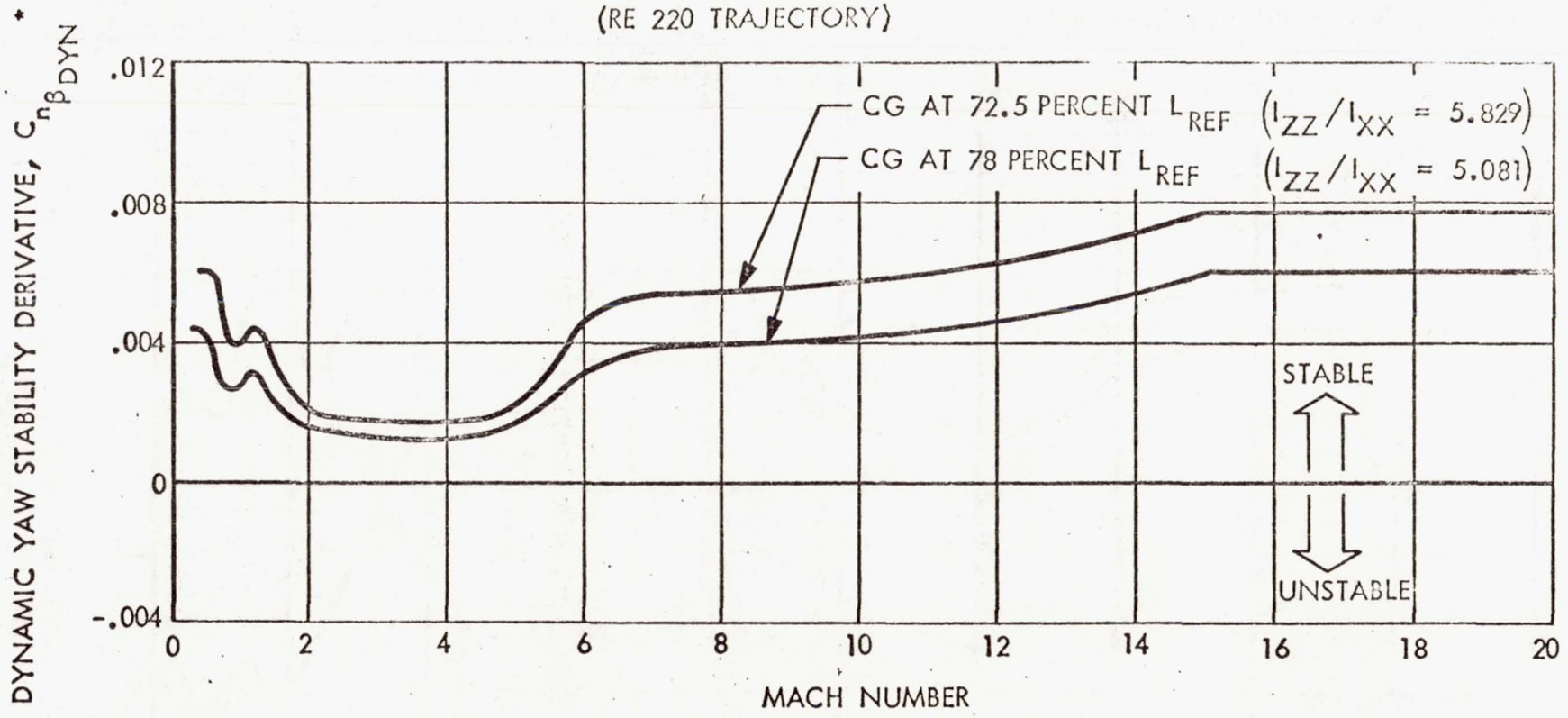


Fig. 3.5-27

REFERENCE:

- LOCKHEED 8 FT X 12 FT TEST L-324
- AMES 6 FT X 6 FT TEST 542
- LANGLEY UPWT TEST 955

D03283

3.6 AEROTHERMODYNAMICS AND TPS

Aerothermodynamic characteristics and thermal protection system (TPS) requirements for the two-stage orbiter are described in this section. Aerodynamic-heating prediction methods and entry-heating boundaries are presented in Section 2.7.

3.6.1 Entry Thermal Environment

Figure 3.6-1 shows the two-stage orbiter design entry trajectory. This trajectory, designated RE-220, generates 1,177-nm crossrange from a 270-nm, 55-deg inclination orbit. Entry time from 400,000 ft to touchdown is 2,840 sec, including approximately 500 sec for go-around. The heating duration is approximately 1,600 sec.

Figure 3.6-2 shows surface-temperature histories at representative stagnation region and lower centerline locations for trajectory RE-220, and Figure 3.6-3 shows peak-surface temperatures in the form of isotherms. These are radiation equilibium temperatures based on a surface emittance of 0.8. Peak temperatures are 2730°F on the nose cap, 1940°F on the body leading edge, 2300°F on the fin leading edge, 2210°F on the lower surface, and less than 800°F at most leeward surface locations. Turbulent heating determines peak temperatures on virtually the entire lower surface but is relatively insignificant for leading edge regions.

Figures 3.6-4 and 3.6-5 show representative surface-temperature histories and peak-temperature isotherms for an alternate, cooler 1100-nm crossrange trajectory designated RE-221 (see paragraph 2.4.5). This trajectory is appropriate for a metallic heat shield, since the moderate temperatures permit use of a nonrefractory heat shield on the lower surface and body leading edges, and a coated columbium heat shield on the nose cap. Peak temperatures are 2470°F on the nose cap, 1750°F on the body leading edge,

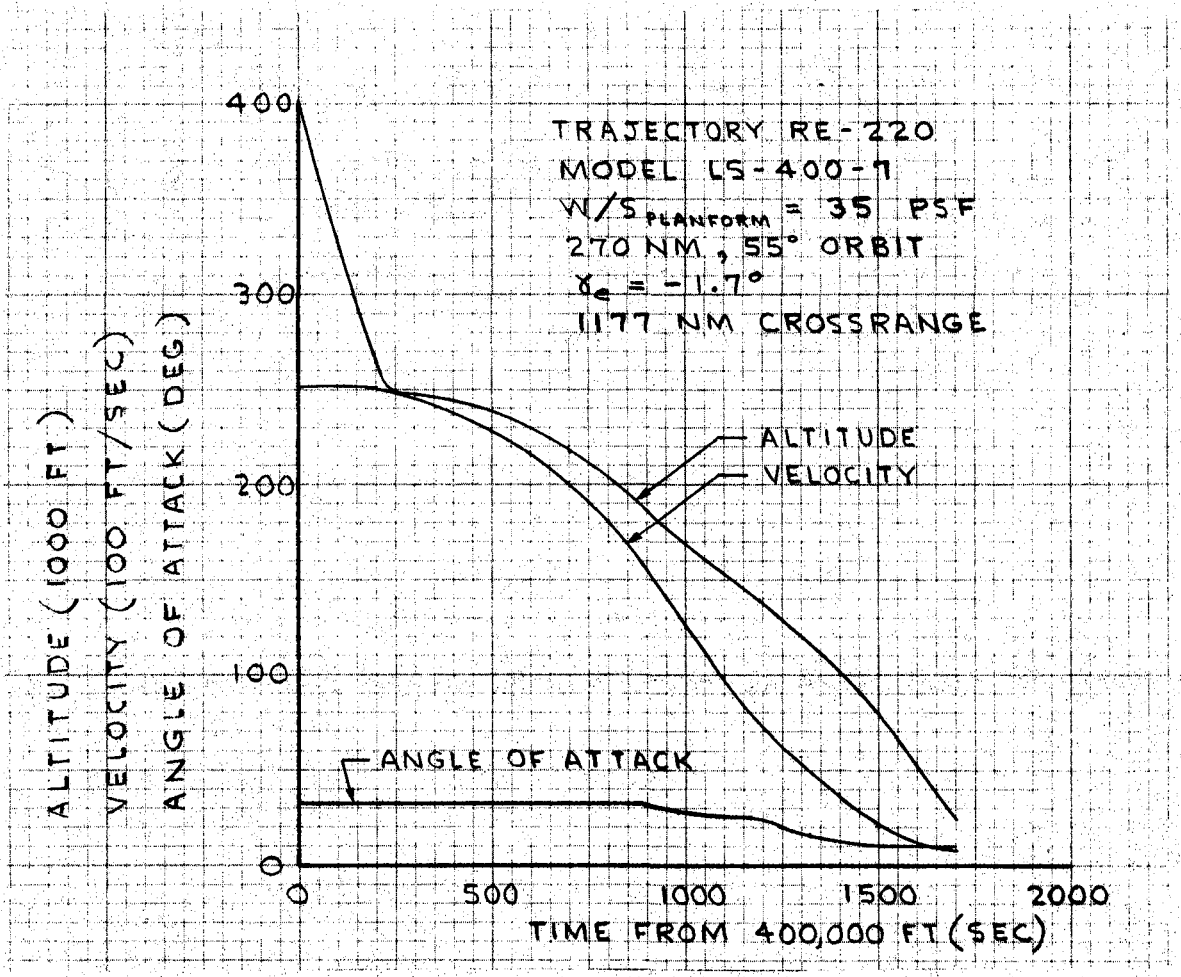


Fig. 3.6-1 Two-Stage Orbiter 1100-nm Crossrange Design Entry Trajectory (RE-220)

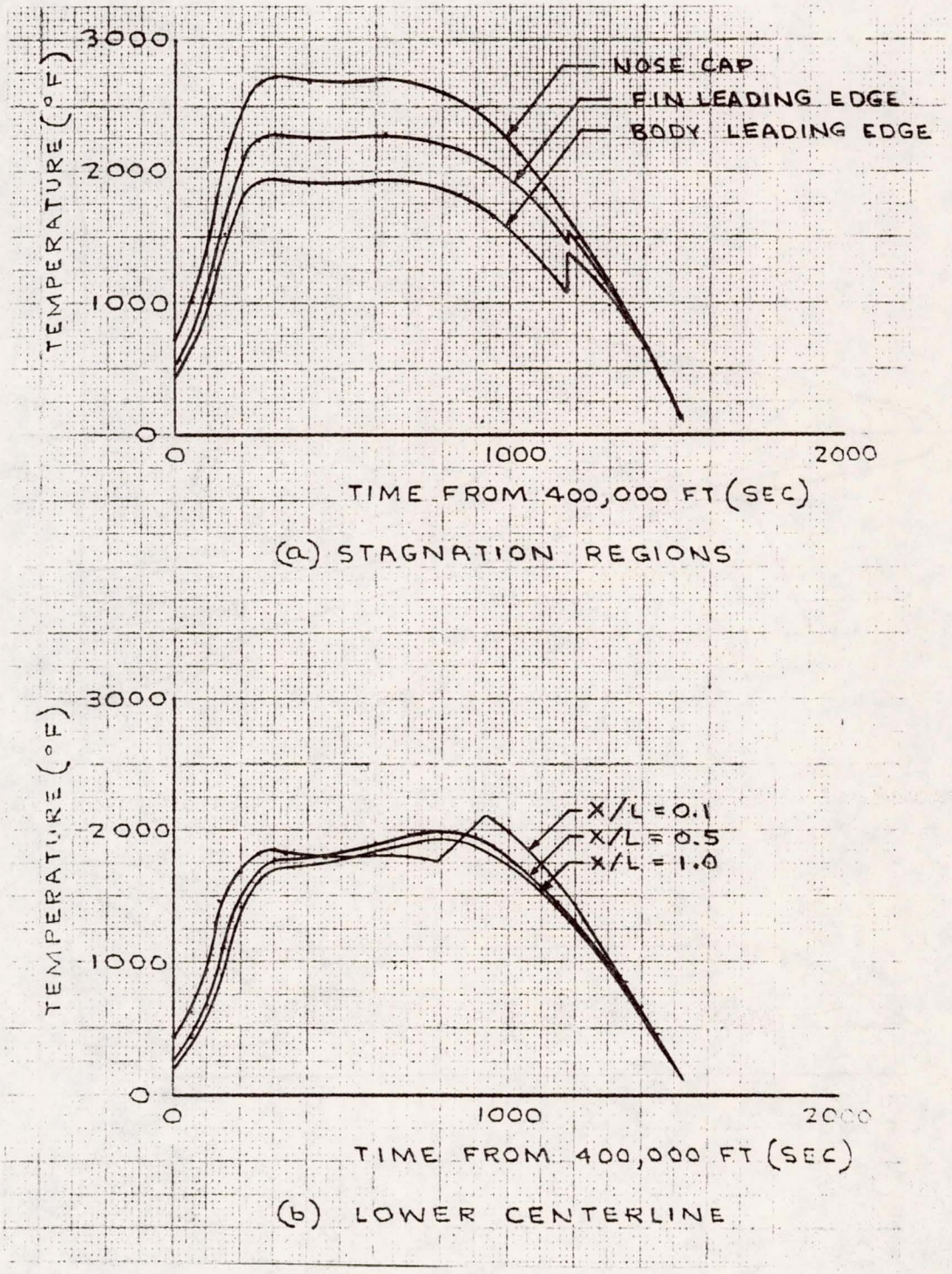


Fig. 3.6-2 Two-Stage Orbiter Surface Temperature Histories for Design Entry Trajectory (RE-220)

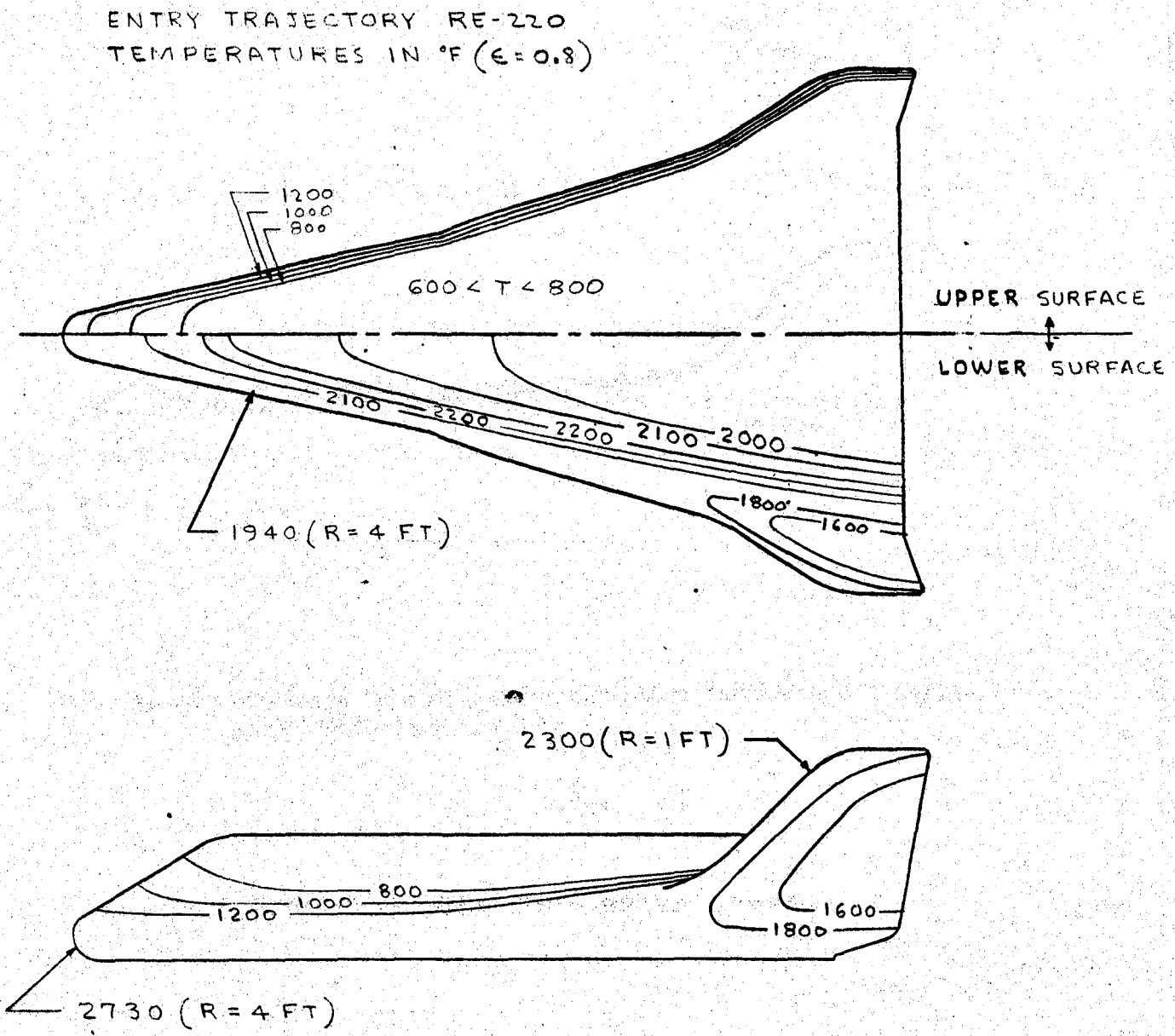


Fig. 3.6-3 Two-Stage Orbiter Peak Temperature Isotherms for Design Entry Trajectory (RE-220)

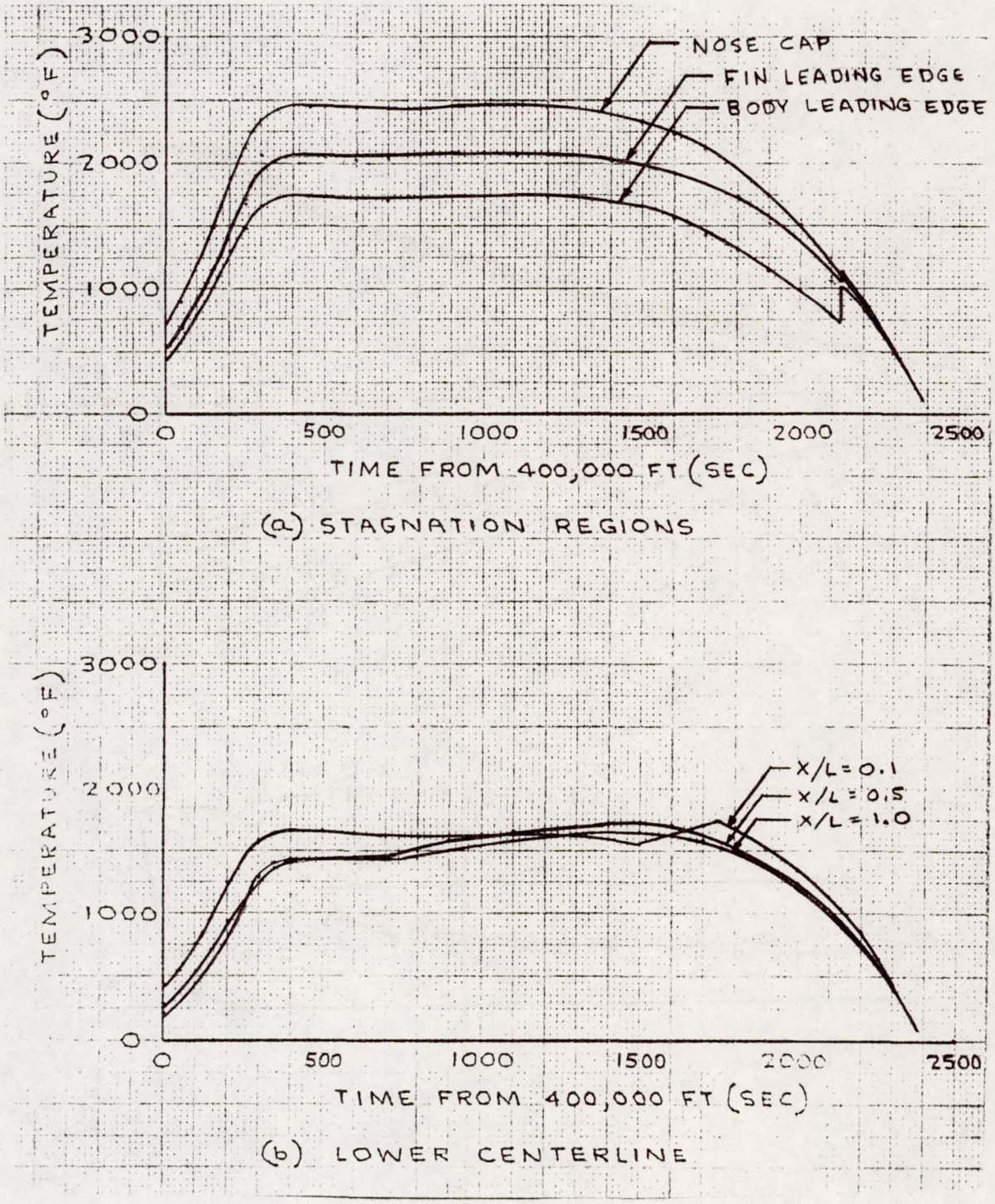


Fig. 3.6-4 Two-Stage Orbiter Surface Temperature Histories for Low Temperature Trajectory (RE-221)

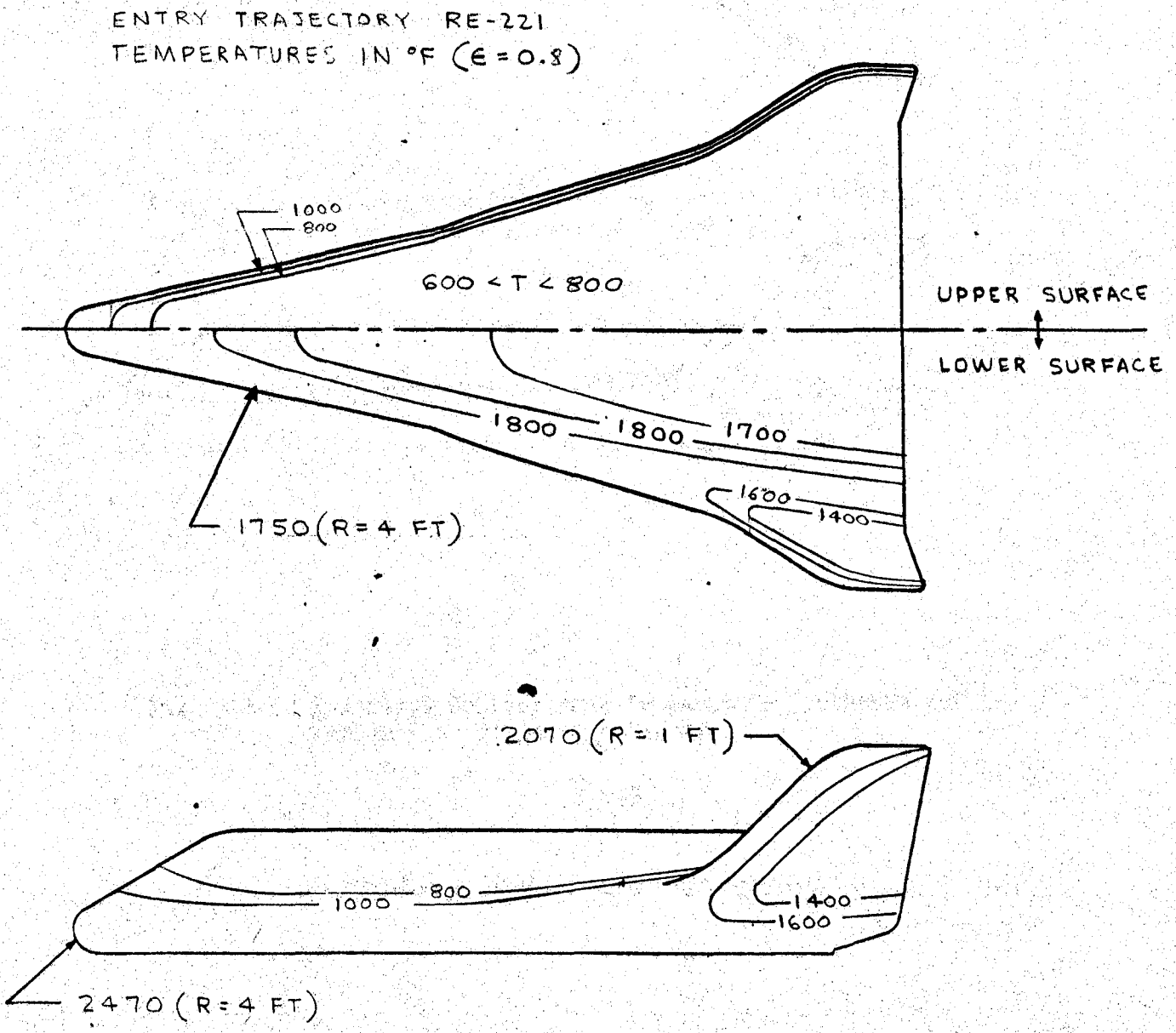


Fig. 3.6-5 Two-Stage Orbiter Peak Temperature Isotherms for Low Temperature Trajectory (RE-221)

2070°F on the fin leading edge, 1830°F on the lower surface, and generally less than 800°F on the upper surface.

3.6.2 Thermal Protection System

Because the stage-and-one-half and two-stage orbiters experience similar entry thermal environments, the thermal protection system selected for the two-stage orbiter is identical to that shown in Figure 2.7-8 for the stage-and-one-half. Heat shield materials are titanium for surface temperatures up to 1000°F, LI-1500 for $1000°F \leq T \leq 2300°F$, and coated tantalum on the nose cap where the peak temperature is 2730°F.

Table 3.6-1 summarizes insulation thickness and TPS weights for the two-stage orbiter, based on the 1,100 nm design entry trajectory with go-around. To minimize insulation weight, the aluminum structure is assumed to be cooled with air supplied by ground support equipment, starting five minutes after touchdown. Insulation thicknesses are based on temperature limits of 600°F for titanium structure and 300°F for aluminum structure. Because the entry-heating duration is less than that for the heavier stage-and-one-half orbiter, the two-stage insulation thicknesses are about 8 percent less than those shown in Table 2.7-2 for the stage-and-one-half. Also, since the two-stage lower-trim surface is not retracted during ascent, the upper surface requires thermal protection from plume heating. This is accomplished by bonding 0.5 in. of LI-1500 to the titanium structure. As shown in Table 3.6-1, total TPS weight for the two-stage orbit is 34931 lb, based on application of thermal protection to 16,828 ft^2 of surface area.

Table 3.6-1
TWO-STAGE ORBITER TPS WEIGHTS

Location	Area (ft^2)	t (in.)	W (lb)
Body Lower Surface	4,110	1.67	13,558
Body Leading Edge	2,753	1.30	7,743
Body Upper Surface ($T \geq 1000°F$)	937	1.10	2,389
Body Upper Surface ($T < 1000°F$)	3,667	0.50**	3,166
Fin Leading Edge	284	0.83	377
Fin Side	1,482	0.65	1,618
Lower Trim Surface (Bottom)	1,086	1.43	2,297
Lower Trim Surface (Top)	1,086	0.50	972
Nose Cap and Skirt	203	----	1,147
Base	1,220	0.75	1,492
Base Flame Curtain		----	172
TOTAL	16,828		34,931

* LS 400-7A Configuration (Drawing SKS 100049A)

** 6 PCF Dyna-Flex; Remainder is 15 PCF LI-1500

3.7 STRUCTURES

3.7.1 Introduction

The two-stage delta-body orbiter, designated LS 400-7A, is structurally similar to the stage-and-one-half delta-body orbiter. Major structural differences occur because of the change to booster-orbiter interstage attachment and changes in propellant loading and the thrust structure. Since only two rocket engines are employed on this orbiter, the thrust structure requirements become less complex and significantly lighter. Many of the structural aspects associated with the stage-and-one-half orbiter that are common to the two stage, such as the payload compartment door, aerodynamic surfaces, landing gear, and the crew cabin were discussed in Section 2.8, and do not require repeating. The sections on materials and structural criteria, also discussed in Section 2.8, apply equally well to the two-stage orbiter and are also not repeated here.

The structural definition of the two-stage orbiter is not as complete as the stage-and-one-half configuration. This design started late in the program and remains to be completed. The areas completed to date are the structural loads, initial orbiter airframe structural sizing, and the propellant tankage sizing. These areas are discussed subsequently. Areas not common to the stage-and-one-half configuration, such as the thrust structure, airbreathing engine support, and the landing gear (reduced landing weight) were scaled using mass-properties scaling equations based on prior work or available statistics.

3.7.2 Loads

3.7.2.1 Loads Criteria and Assumptions. The loads criteria prepared for Phase B proposal are contained in EM L-1-02-10-M1, Vol. 2, Appendix B. Excerpts and deviations to this E.M. are included in the following paragraphs.

3.7.2.1.1 Maximum αq Condition. The maximum product of angle-of-attack and dynamic pressure is assumed to be $\pm$2800-deg psf. This value is considered to be a conservative estimate which accounts for the 95 percentile steady-state wind plus gust and also uncertainties in the guidance system and vehicle response to wind shears during the ascent-flight phase. The sidewind case has been considered for the two-stage vehicle, since loads relief studies, similar to studies performed on the stage-and-one-half vehicle, have not been made for the two-stage vehicle.

A normal load factor of 0.4g and an axial load factor of 1.7g have been derived from previous trajectory analysis and the referenced criteria.

3.7.2.1.2 Maximum Axial Acceleration; Booster Burnout. The criteria axial load factor of 3g has been assumed and is considered to include both rigid-body and dynamic effects. The booster thrust vector is assumed to be directed through the composite vehicle cg and is misaligned 11 deg* from the orbiter reference axis. Resulting component load factors are 2.942 axial load factor and 0.572g sideload factor. The loads at this condition are assumed to be entirely inertial and geometry induced.

3.7.4.1.3 Orbiter Landing. Two landing conditions have been considered. The first condition includes main gear touchdown, which results in a criteria load factor of 2.7g. One g is attributed to aerodynamic lift and 1.7g reaction is through the main gear. The second landing condition occurs when the orbiter rotates onto the nose gear. A 1g load factor is assumed; no aerodynamic lift is considered.

3.7.2.1.4 Noncritical Conditions. Conditions investigated and found to be noncritical orbiter loading conditions are as follows:

a. Launch release where the rigid-body plus dynamic axial load factor is 1.7g.

*The misalignment angle has been reduced to seven degrees in the final configuration.

b. Post-release with one engine out. Rigid-body load factor is 1.25g and the flexible-body effects are assumed to be 0.1g.

3.7.2.2 Results of Analysis. The analysis consists of determining design loads for a two-stage orbiter vehicle designated LS400-7. Loads found to be critical to the major structural components are presented on the following table and figures:

a. Orbiter/Booster Attachment Reactions (Table 3.7-1)

b. Bending Moment; Maximum Neg. αq; ascent (Fig. 3.7-1)

c. Bending Moment; Maximum positive αq; Ascent (Fig. 3.7-2)

d. Pitch Plane Bending Moment: Maximum βq; Ascent (Fig. 3.7-3)

e. Yaw Plane Bending Moment; Maximum βq; Ascent (Fig. 3.7-4)

f. Axial Load; Maximum $\pm \alpha q$ and $\pm \beta q$; Ascent (Fig. 3.7-5)

g. Bending Moment; Maximum Axial Load Factor; Booster Burnout Condition (Fig. 3.7-6)

h. Axial Load; Booster Burnout (Fig. 3.7-7)

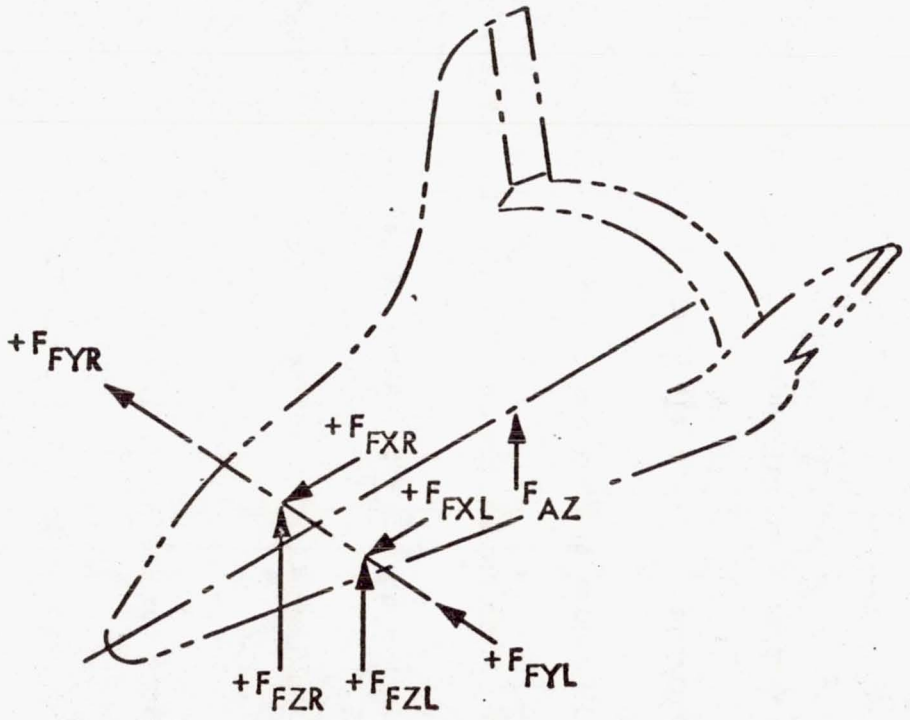
i. Bending Moment at Maximum Main Gear Load (Fig. 3.7-8)

j. Bending Moment at Maximum Nose Gear Load (Fig. 3.7-9)

3.7.2.2.1 Maximum αq, βq Condition. The αq criterion of 2800-deg psf and the normal load factor of 0.4g have been selected from previous trajectory studies of similar vehicles, as design conditions for the ascent phase for the two-stage orbiter. The criteria is conservative pending an indepth ascent-

Table 3.7-1

BOOSTER-ORBITER INTERFACE REACTIONS



ORBITER REACTIONS ~ K LBS

CONDITION	F_{FZL}	F_{FZR}	F_{FXL}	F_{FXR}	F_{FYL}	F_{FYR}	F_{AZ}
MAX + αq	+ 72.8	+ 72.8	+ 881.3	+ 881.3	0.0	0.0	- 9.69
MAX - αq	-350.9	-350.9	+ 881.3	+ 881.3	0.0	0.0	+581.3
MAX + βq	-201.9	- 21.9	+1,102.1	+ 660.5	+53.1	+53.1	+258.0
MAX - βq	- 21.9	-201.9	+ 660.5	+1,102.1	-53.1	-53.1	+258.0
BOOSTER BURNOUT	+ 57.58	+ 57.58	+1,193.0	+1,193.0	0.0	0.0	+348.7

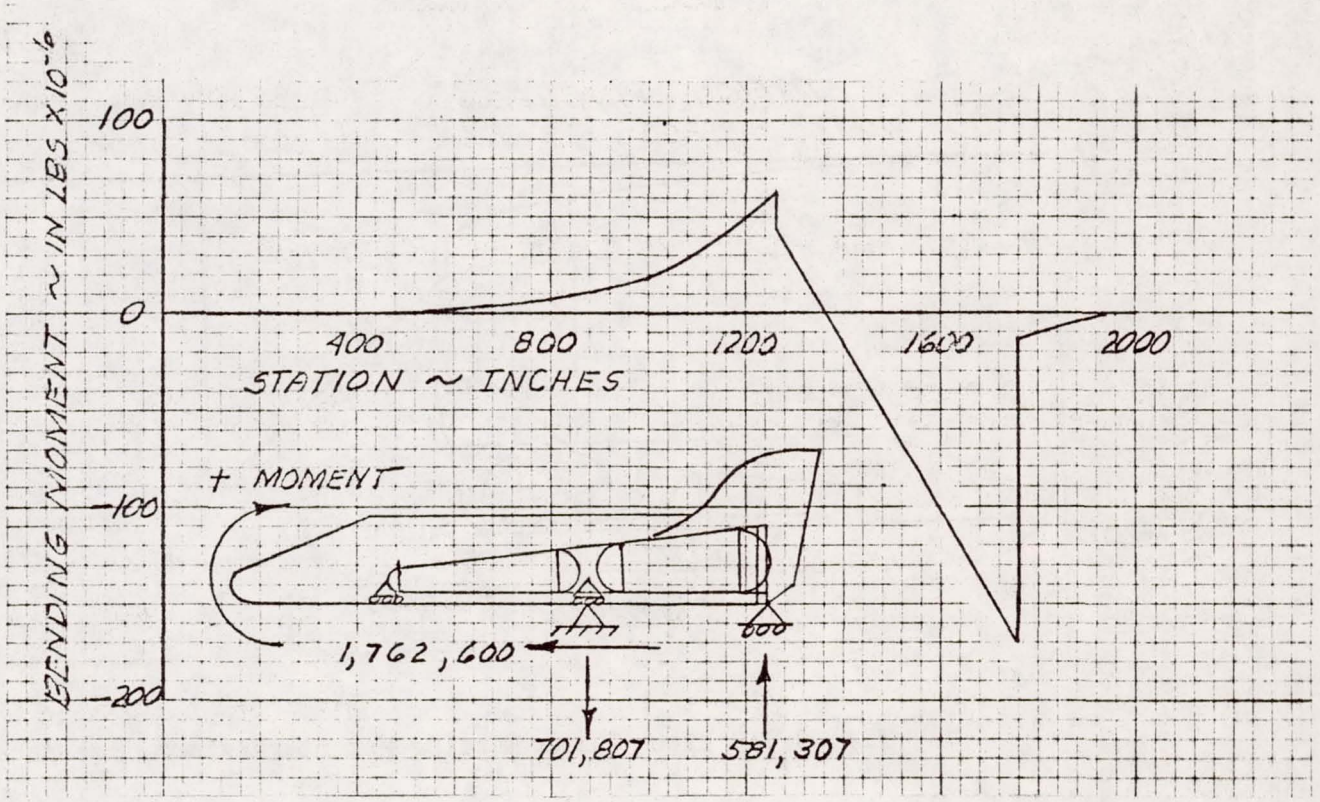


Fig. 3.7-1 Bending Moment; Maximum Neg. αq; Ascent

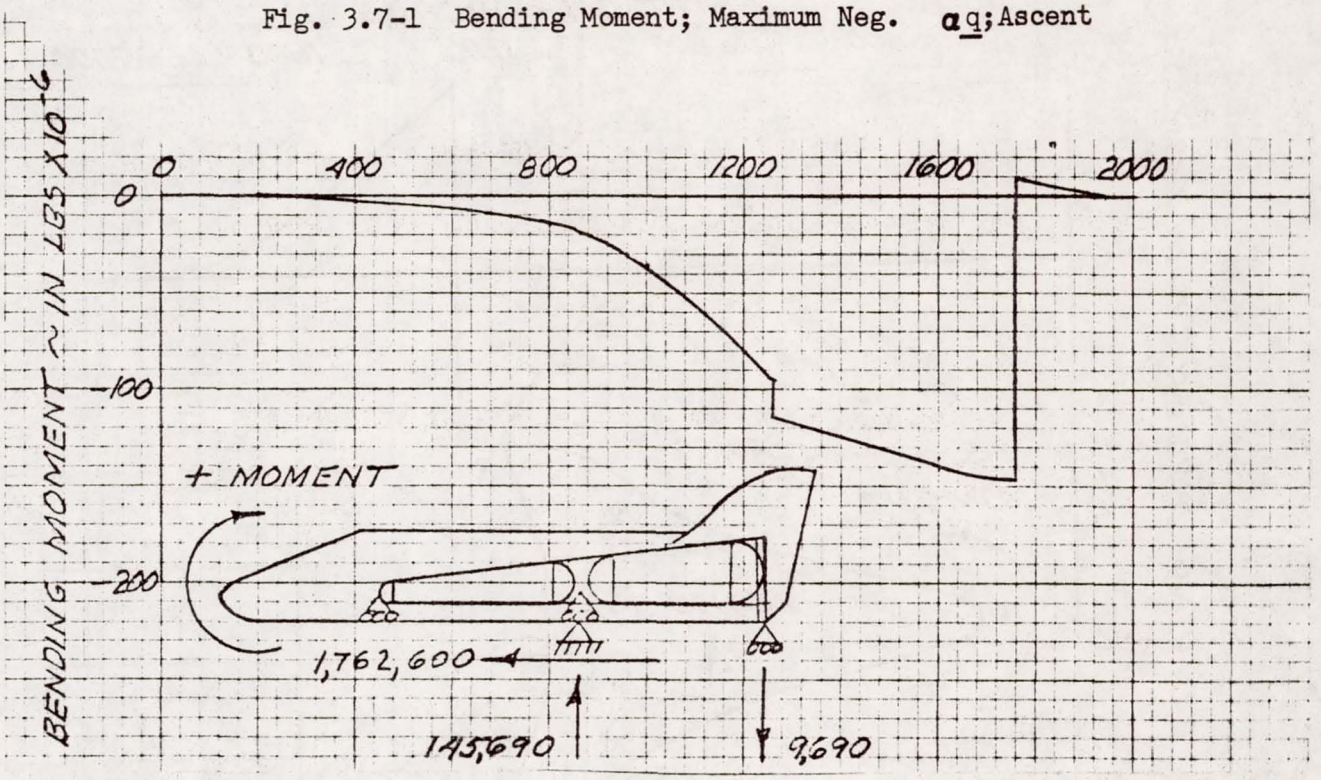


Fig. 3.7-2 Bending Moment; Maximum Positive αq; Ascent

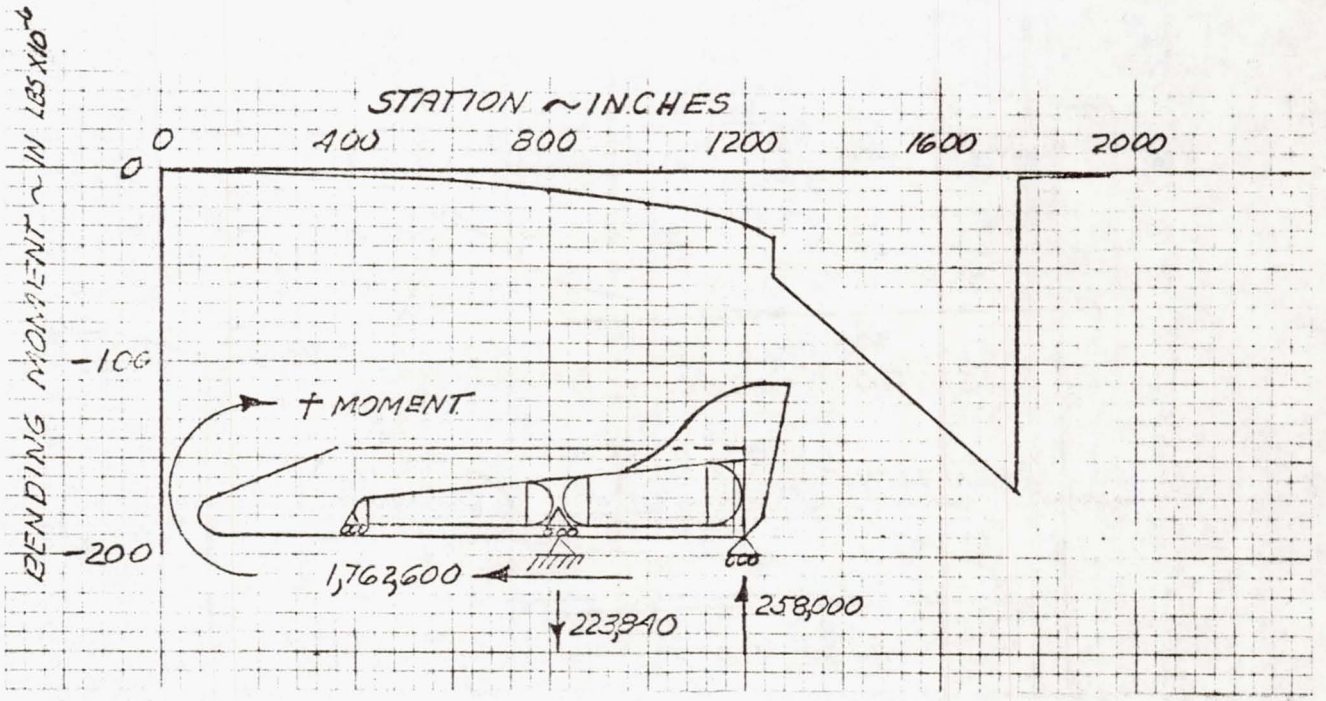


Fig. 3.7-3 Pitch Plane Bending Moment: Maximum βq; Ascent

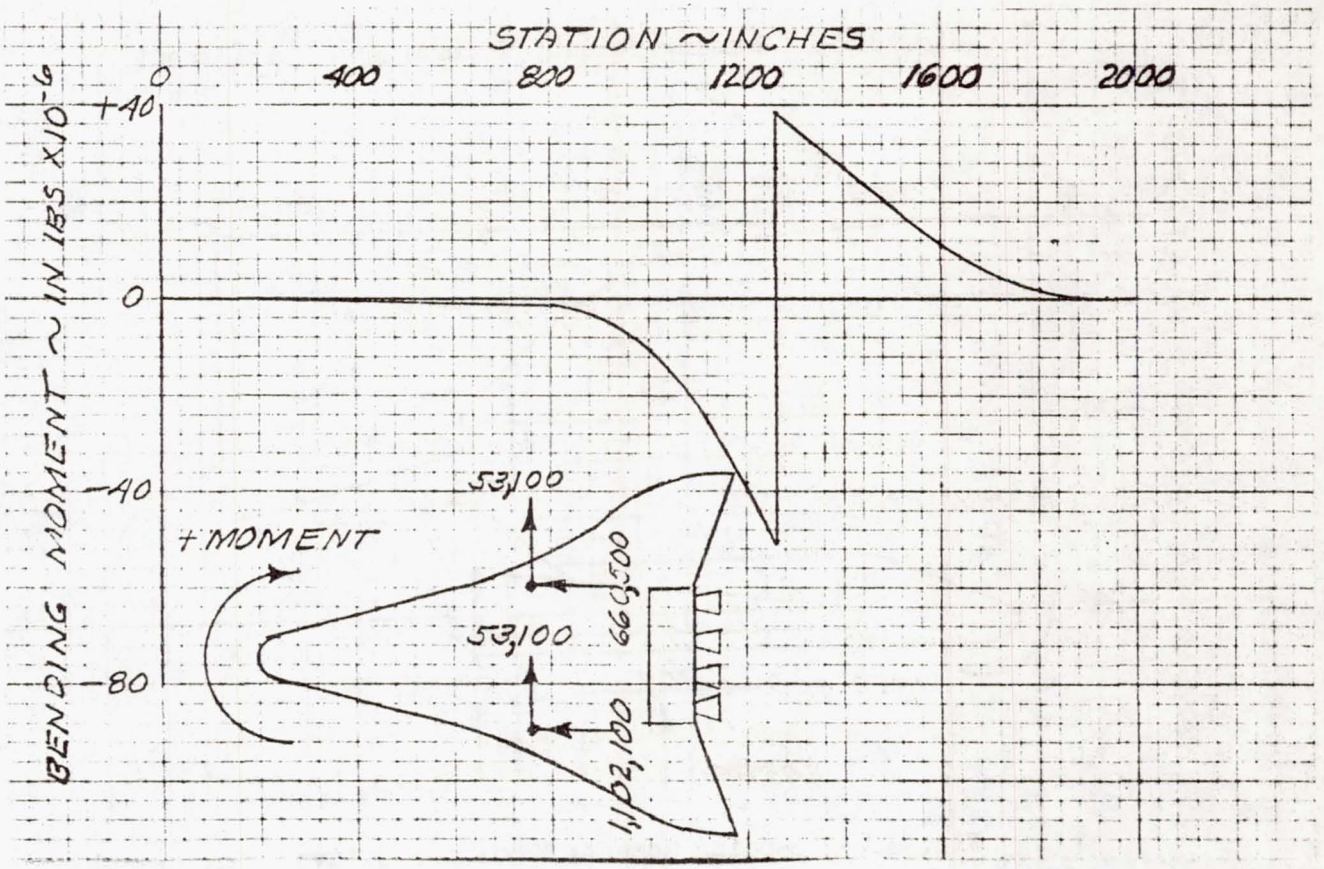


Fig. 3.7-4 Yaw Plane Bending Moment; Maximum βq; Ascent

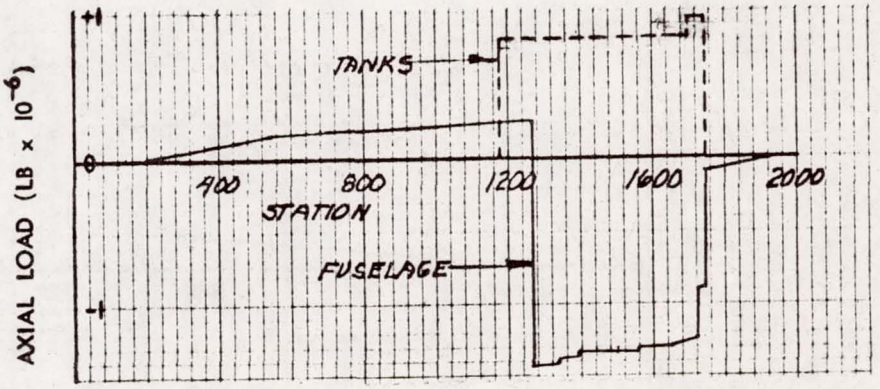


Fig. 3.7-5 Axial Load; Maximum $\pm \alpha \underline{q}$ and $\pm \beta \underline{q}$; Ascent

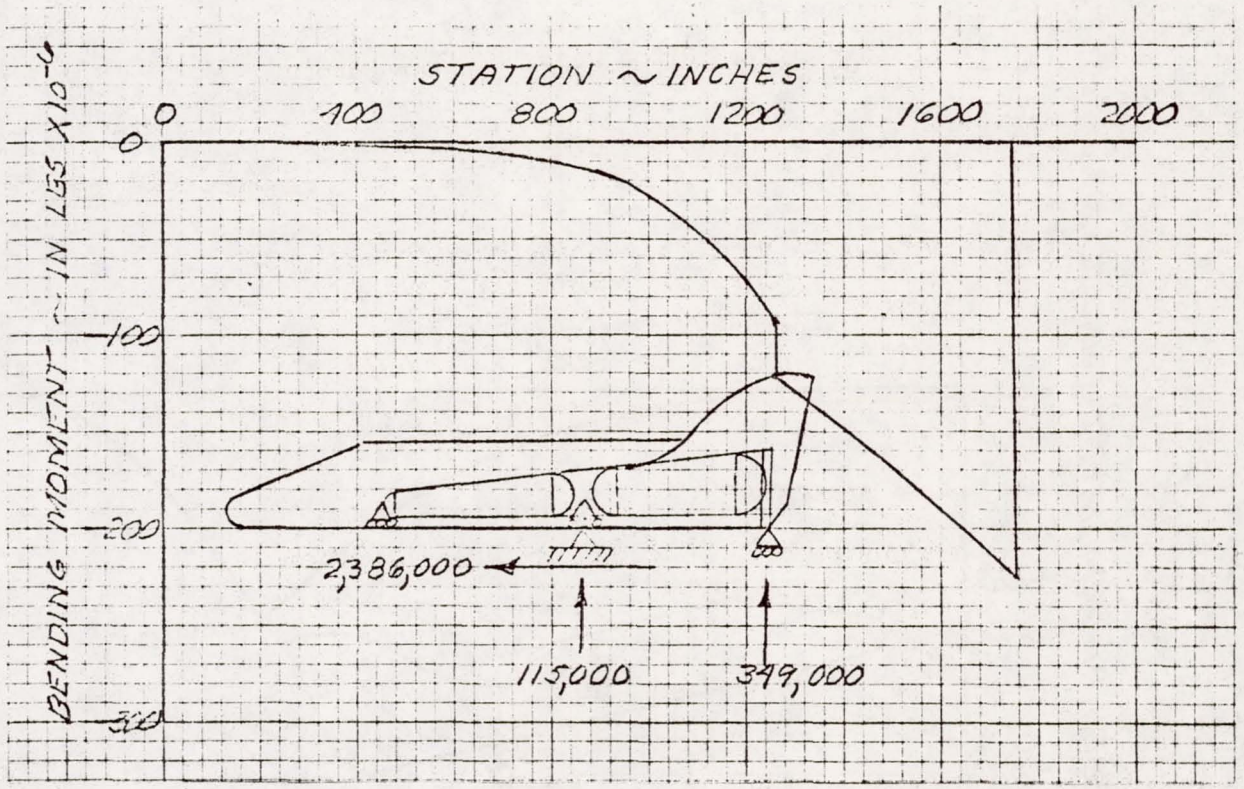


Fig. 3.7-6 Bending Moment; Maximum Axial Load Factor;
Booster Burnout Condition

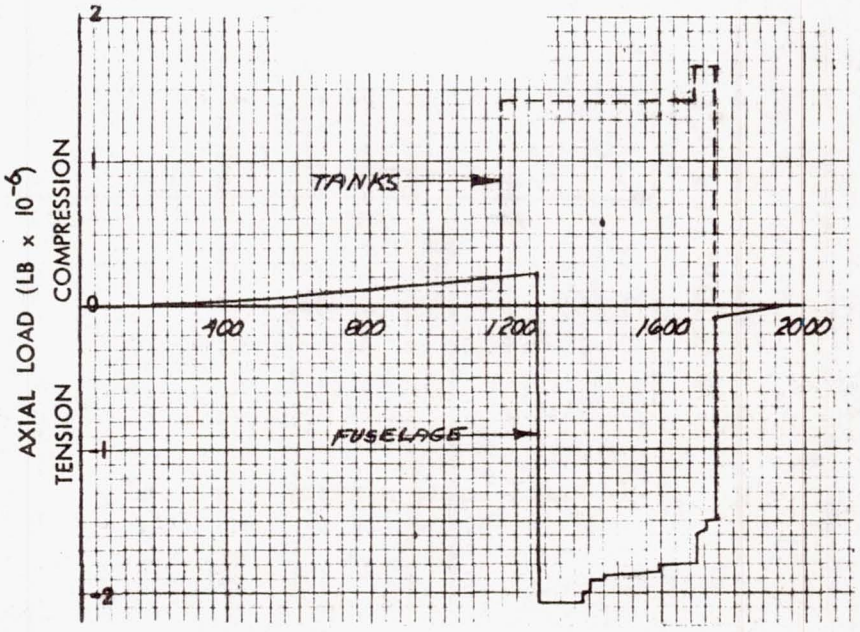


Fig. 3.7-7 Axial Load; Booster Burnout

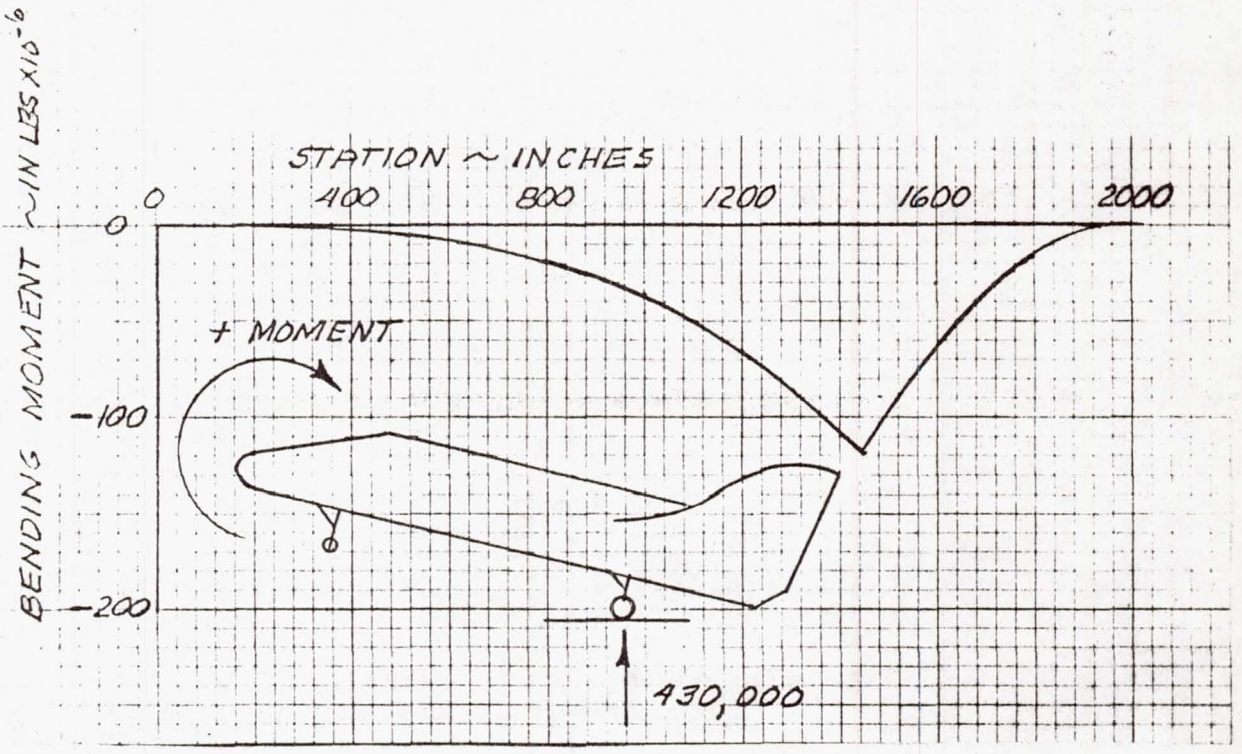


Fig. 3.7-8 Bending Moment at Maximum Main Gear Load

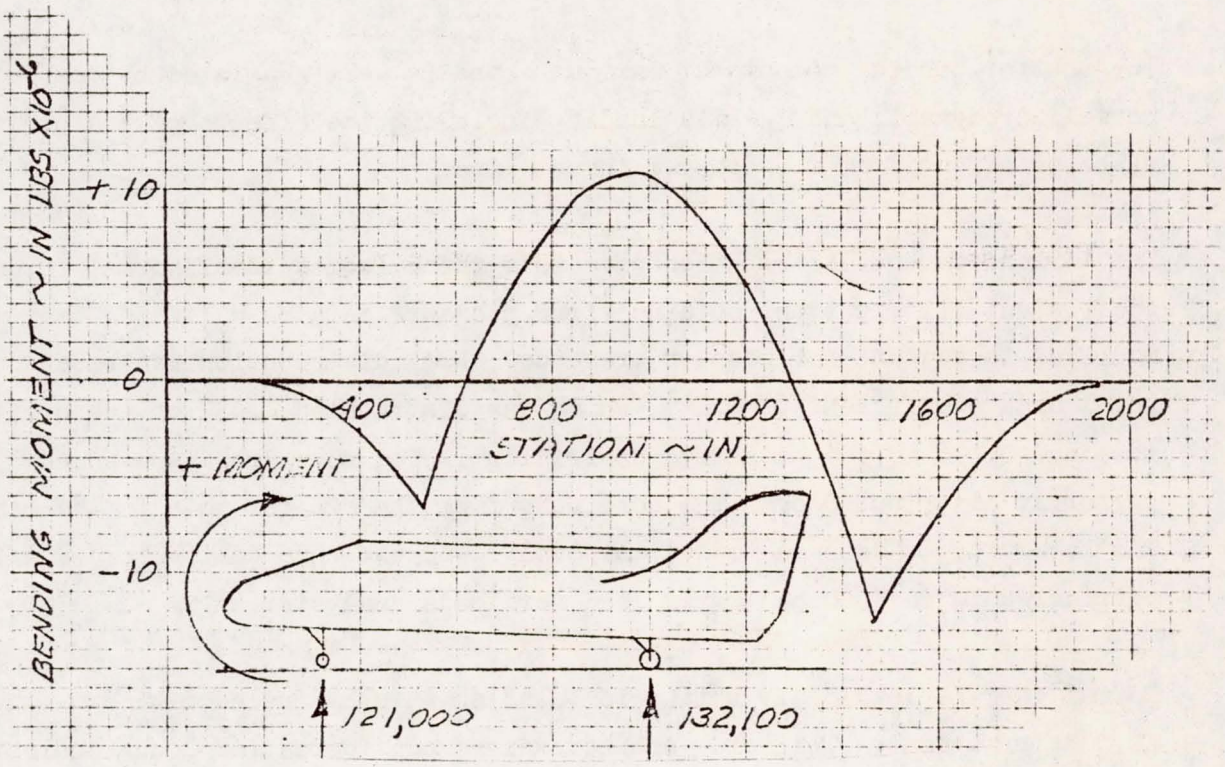


Fig. 3.7-9 Bending Moment at Maximum Nose Gear Load

trajectory analysis.

The orbiter bending moments at maximum α and βq were generated by combining normal aerodynamic and inertia loading including the effects of the orbiter cg being offset from the booster thrust line.

The airload moment distribution were integrated from a local normal force coefficient distribution, obtained from pressure data on a lifting body and modified to account for booster presence. Drag distribution was based on a total drag coefficient of 0.145, which was distributed 0.037 on the orbiter nose and 0.108 as the base drag coefficient. The nose drag is assumed uniformly distributed over the nose ramp from the orbiter nose to the Sta 500. The base drag is assumed to act uniformly over the base heat shield and is considered to be a concentrated load for loads analysis.

Normal inertia load distribution is based on a fully fueled orbiter with a 40,000 lb payload. The total orbiter weight considered for loads analysis is 811,000 lb.

The orbiter/booster attachment loads are found by balancing the total aerodynamic and inertial loading, considering two forward attachment points at orbiter Sta 1270 and a single attachment at Sta 1752. The forward attachment points are located at orbiter hard points on either side of the payload bay, and the aft attachment point is in the area of the orbiter thrust structure. The aft attachment is designed to take loads only in the Z direction; all orbiter yaw moments are taken by the forward attachments.

It has been assumed that the orbiter axial load is carried by the orbiter fuselage shell to the thrust structure, where the propulsion unit (including propellant tanks) and the fuselage are joined axially. Therefore,

the moment due to the eccentricity of the attach point to the orbiter center-of-gravity is shown at the orbiter thrust structure.

The booster attachment loads are included in Table 3.7-1. Bending moment distributions for maximum α, $\beta\underline{q}$ conditions are shown on Figs. 3.7-2 through 3.7-4. Axial loads for this condition are shown on Fig. 3.7-5.

3.7.2.2.2 Maximum Axial Load Factor; Booster Burnout. Normal loads are induced in the orbiter for this condition due to the misalignment of the booster thrust vector and the orbiter axis. It has been assumed that the booster thrust vector is through the cg of the booster orbiter combination which results in an 11 deg misalignment. The resulting component orbiter load factors are 2.942g axial and 0.572g normal.

Booster burnout occurs at altitudes great enough that air loads can be ignored in loads analysis, the loads being entirely inertia and geometry induced. The inertia properties and orbiter/booster attachment geometry are as described for the maximum α, $\beta\underline{q}$ case. Again, the concentrated moment due to orbiter eccentricity is transferred to the orbiter thrust structure.

Bending moment and axial loads for the maximum axial load factor case are shown on Figs. 3.7-6 and 3.7-7 respectively. Orbiter/booster interface loads are shown in Table 3.7-1.

3.7.2.2.3 Orbiter Landing Condition. Two cases have been considered for the landing condition. The first is maximum main gear reaction. It has been found from previous analysis that a 10-ft/sec MIL-spec-derived sink rate at landing can be resolved into a 1.7 g load factor acting through the main gear struts with 1g aerodynamic lift on the orbiter (Ref. 3.7-1) LMSC-A972148. The tire and strut stroke required for the 1.7 g reaction was determined to be 18 in. Rotational acceleration of the orbiter is ignored for this condition. The bending moment distribution for this condition is shown on Fig. 3.7-8.

The second landing condition considered occurs when the obiter pitches down on the nose gear. Analysis performed in Ref. 3.7-1 shows that with a 21-in. nose gear tire and strut stroke, the nose gear reaction will be held to 2.35 times the static 1g nose gear load. The total nose gear load in this case is 121,000 lb. A 1g vehicle acceleration has been assumed; air loads are assumed negligible. Nose-up pitch acceleration has been accounted for. The bending moment distribution for the nose gear slap-down is shown on Fig. 3.7-9.

3.7.3 Structural Arrangement

Basic structural arrangement of the two-stage delta-body orbiter is similar to the stage-and-one-half orbiter, in outward appearances, but internal arrangements and load paths differ. Internal propellant loading differs by almost a factor of two. Fortunately, rocket-engine requirements between the two vehicles compensate to the extent that no change in vehicle size is required. A minor change in the extension of the upper flap, the relocation of airbreathing engines, the removal of the lower-flap retraction-track assembly, and the change in thrust structure represent the major changes in addition to the interstage connection.

The two-stage orbiter is supported from the booster at three locations. The selection of three attach points is based on consideration of several factors: (1) to provide a statically determinate support system, (2) to facilitate a reliable separation system, and (3) to minimize interface design requirements. Two attach points, providing omnidirectional support, are located at the orbiter aft payload-compartment bulkhead (Sta 1270), at buttline (BL) $\pm$90. The third tiepoint, providing only vertically transverse orbiter support, is located at the lower-flap hingeline at BL 0.

There is no one critical load condition which designs the two-stage orbiter as will be discussed subsequently. All ascent conditions cause a severe orbiter-fuselage bending moment to be present because of the large eccentricity between the center-of-mass of the orbiter and the connection to the booster.

To minimize this problem, the delta-body orbiter ascent tankage is designed to share the orbiter fuselage bending moment loads. The twin propellant tank systems are assembled as two continuous beams, supported within the orbiter fuselage at three locations: the forward bulkhead of the payload compartment (Sta 530), the aft bulkhead of the payload compartment (Sta 1270), and the interface of the LH2 tanks with the thrust structure (Sta 1770). The forward and intermediate supports do not restrict longitudinal tank deformations; transverse (shear) loads only are reacted at these locations. The aft support is rigidly attached to the thrust structure which, in turn, is connected to the fuselage. Axial loads associated with the LO2 ascent tanks are directly supported by the LH2 ascent tanks. Such a statically indeterminate support system prevents the orbiter airframe from bending without inducing bending of the tankage. The bending moment distribution, therefore, becomes a function of the relative stiffness, EI, between the tankage and airframe, where E is the material modulus of elasticity and I is the cross-section area moment-of-inertia. Details of this analysis will be discussed subsequently. The finite element modeling and structural sizing of the fuselage is presented in EM L2-11-01-M1-1 and the structural sizing of the tankage is presented in EM L2-11-04-M5-1, Vol. 2, Appendix B.

3.7.3.1 Orbiter Airframe. Orbiter fuselage and ascent tankage bending moments were obtained using the SNAP/STATICS computer code. SNAP/STATICS is a general purpose finite element program based on the direct stiffness version of the displacement method. It is applicable to any structure which can be idealized as an assemblage of beam, membrane and bending elements. Elastic relations ranging from isotropic to general anisotropy are considered. The program solves static problems involving imposed displacements, mechanical loadings and thermal loadings.

The fuselage and ascent tankage were represented by a finite element model consisting of 35 discrete beam elements and 34 joints. Orbiter fuselage and ascent tankage bending moments were obtained for two structural design iterations.

Because of the eccentricity of the orbiter center of gravity relative to the lower surface booster attach points at F.S. 1270, a servere bending moment is

introduced during ascent. The eccentric booster attach moment forms a couple whose magnitude is equal to the longitudinal booster attach reaction times the transverse distance to the orbiter center of gravity. The two concentrated longitudinal booster attach loads are introduced into the lower surface of the fuselage shell at F.S. 1270. However, because of shear lag effects, the longitudinal booster attach loads do not become fully effective as a fuselage bending moment until F.S. 1752 at the thrust structure. Conceptually, the means of transferring the booster attach couple of the aft end of the orbiter may be idealized as a tierod and bell crank mechanism. The booster attach reaction pulls on the tierod and introduces a couple at the aft end of the fuselage. In turn, the couple is resisted by the fuselage and ascent tank structure in properties to their respective stiffness.

Design loading conditions considered were the maximum acceleration, maximum positive αq, maximum negative αq and landing impact. For each loading condition, load factors, unit inertia load distributions, and airload distributions were used to determine nodal point lateral loads for the finite element model.

For the initial design iteration, the orbiter was considered as a homogeneous body with no moments due to eccentric masses applied to the fuselage or ascent tankage elastic axes. However, for the second design iteration, a concentrated moment was applied between the ascent LO_2 and LH_2 tanks to account for a 15-in eccentricity between the longitudinal tank axes. Because of increased propellant volume requirements, it was not possible to eliminate the 15-in eccentricity.

The second design iteration fuselage and ascent tank bending moments are shown in Figures 3.7-10 to 3.7-13 for the ascent and landing impact loading conditions. For the ascent loading conditions, approximately 2/3 of the booster attach couple at F.S. 1752 is resisted by the fuselage and 1/3 resisted by the ascent tankage.

ORBITER FUSELAGE AND ASCENT TANKAGE BENDING MOMENTS

MAXIMUM ACCELERATION CONDITION

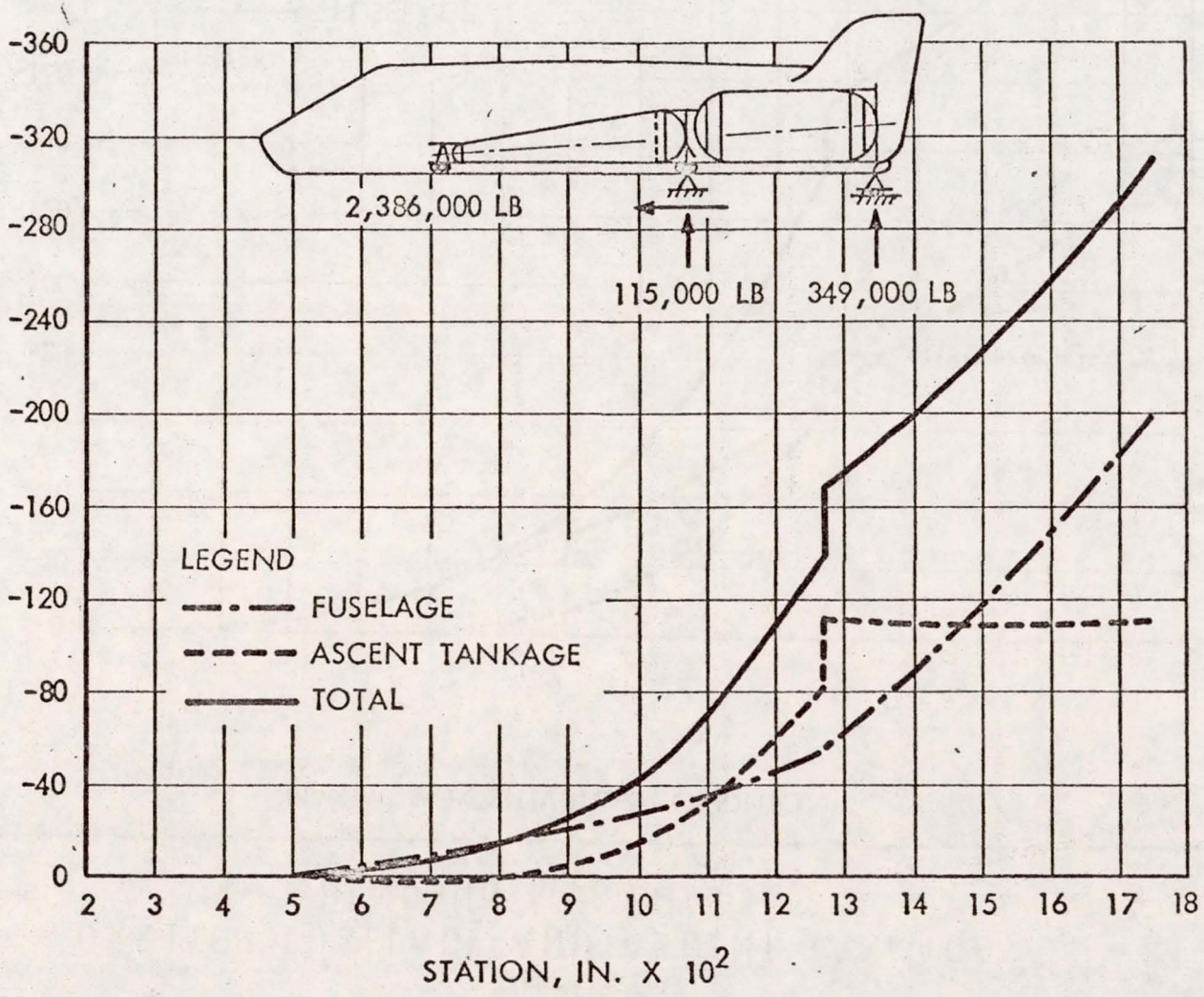


Figure 3.7-11

ORBITER FUSELAGE AND ASCENT TANKAGE BENDING MOMENTS

MAXIMUM POSITIVE aq CONDITION

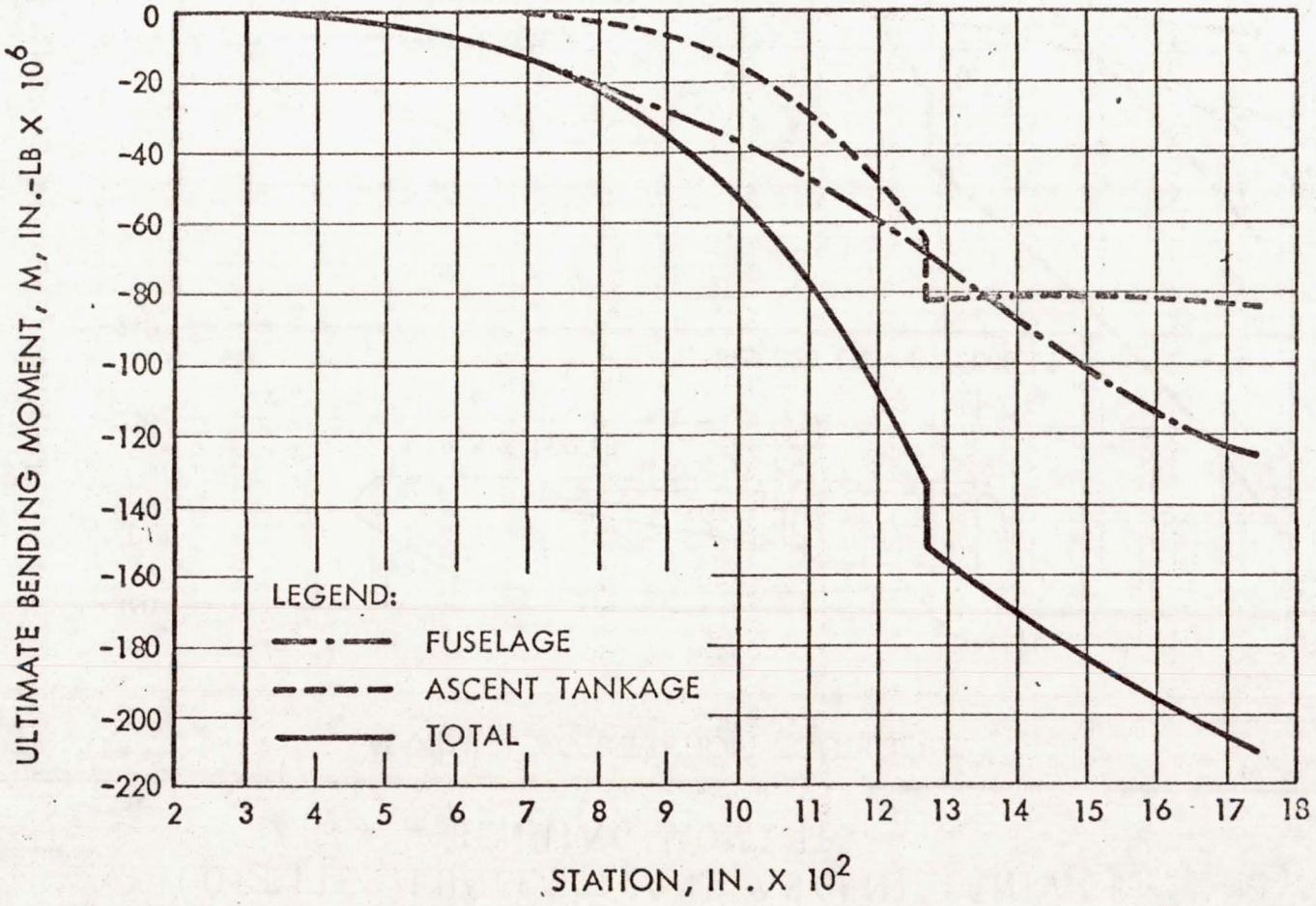


Fig. 3.7-11

Fig. 3.7-12

ORBITER FUSELAGE AND ASCENT TANKAGE BENDING MOMENTS

MAXIMUM NEGATIVE αq CONDITION

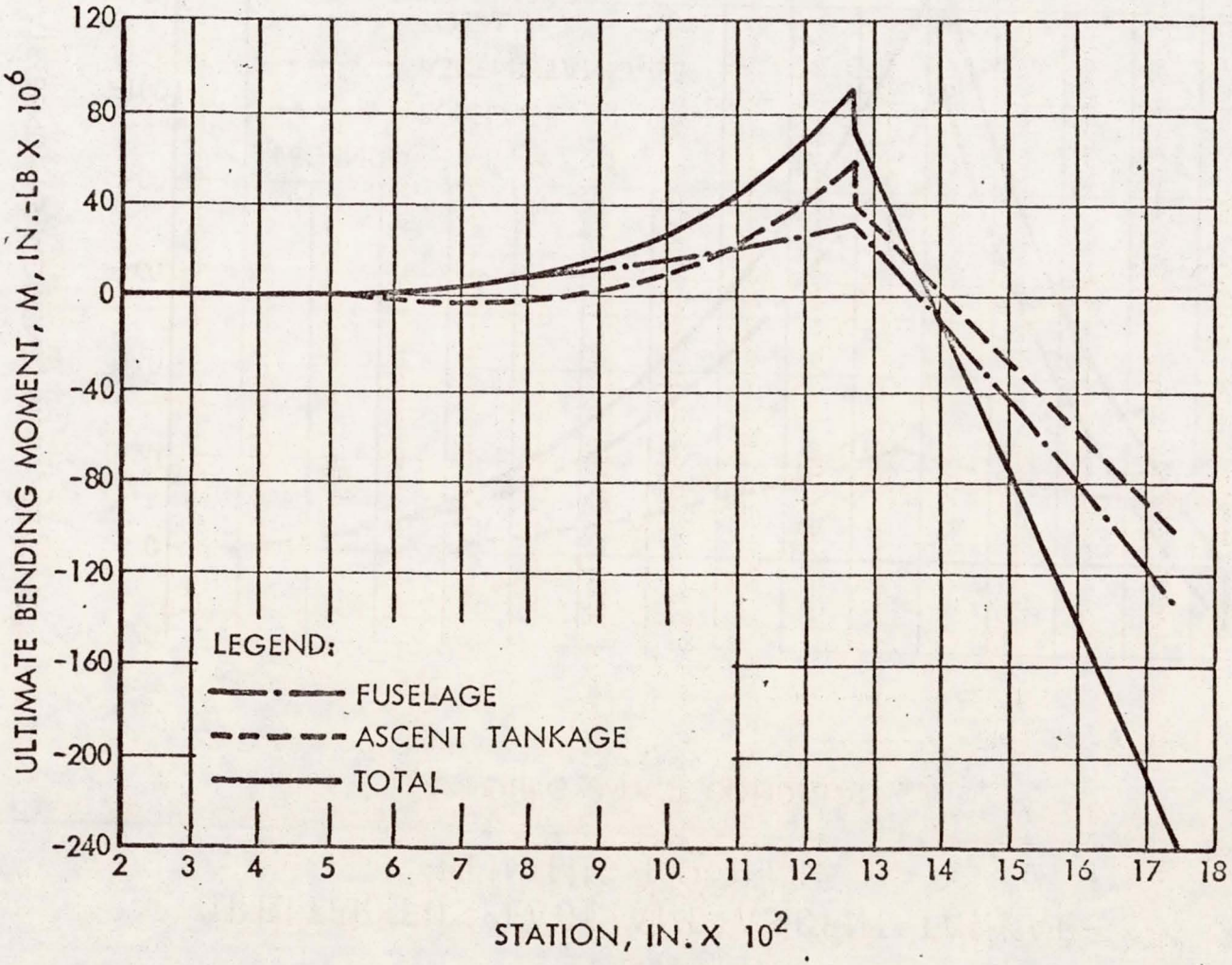
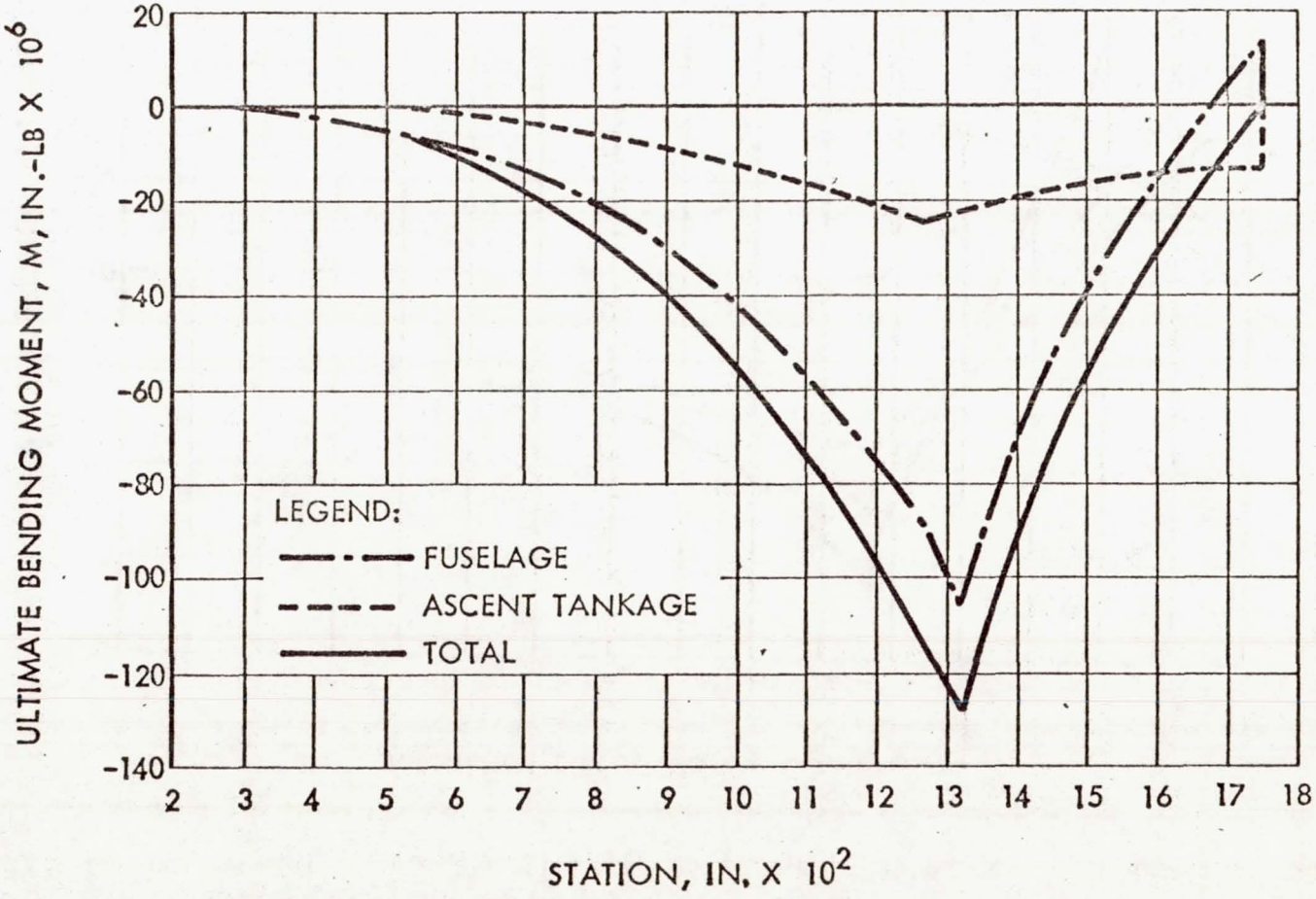


Fig. 3.7-12

Figure 3.7-13

ORBITER FUSELAGE AND ASCENT TANKAGE BENDING MOMENTS

LANDING IMPACT CONDITION



Deflections for the maximum acceleration condition are shown in Figure 3.7-14. Comparison of first and second design iteration bending moments for the landing impact condition show almost identical results. Hence, the change in bending stiffness between the first and second design iteration has little effect on fuselage and ascent tank bending moments. Comparison of first and second design iteration bending moments for the ascent loading conditions show the following results:

- Fuselage bending moments are reduced slightly
- Ascent LO_2 tank bending moments are increased slightly
- Ascent LH_2 tank bending moments are increased essentially by the magnitude of the moment due to the 15-in eccentricity between the LO_2 and LH_2 tanks

Since the 15-in tank eccentricity has little effect on the orbiter transverse center of gravity location, the total eccentric booster attach couple at F.S. 1752 is unchanged.

3.7.3.1.1 Forward Fuselage Line Loads. Orbiter forward fuselage line load intensities were obtained using the FAST computer code. The FAST computer code calculates section properties and load intensities of a fuselage cross-section having at least one axis of symmetry. Section properties are obtained from geometric data specified in terms of discrete areas and the coordinates of the centroid of the discrete areas. In addition, the effective width of each element is specified to permit calculation of line loads. Load intensity is expressed in terms of stress, line load and shear flow (for closed sections) at each discrete element. Multiple loading conditions and fuselage cross-sections are considered.

Using the FAST computer code, ultimate line load intensities, based on from 9 to 17 discrete elements per half fuselage cross-section, were obtained at

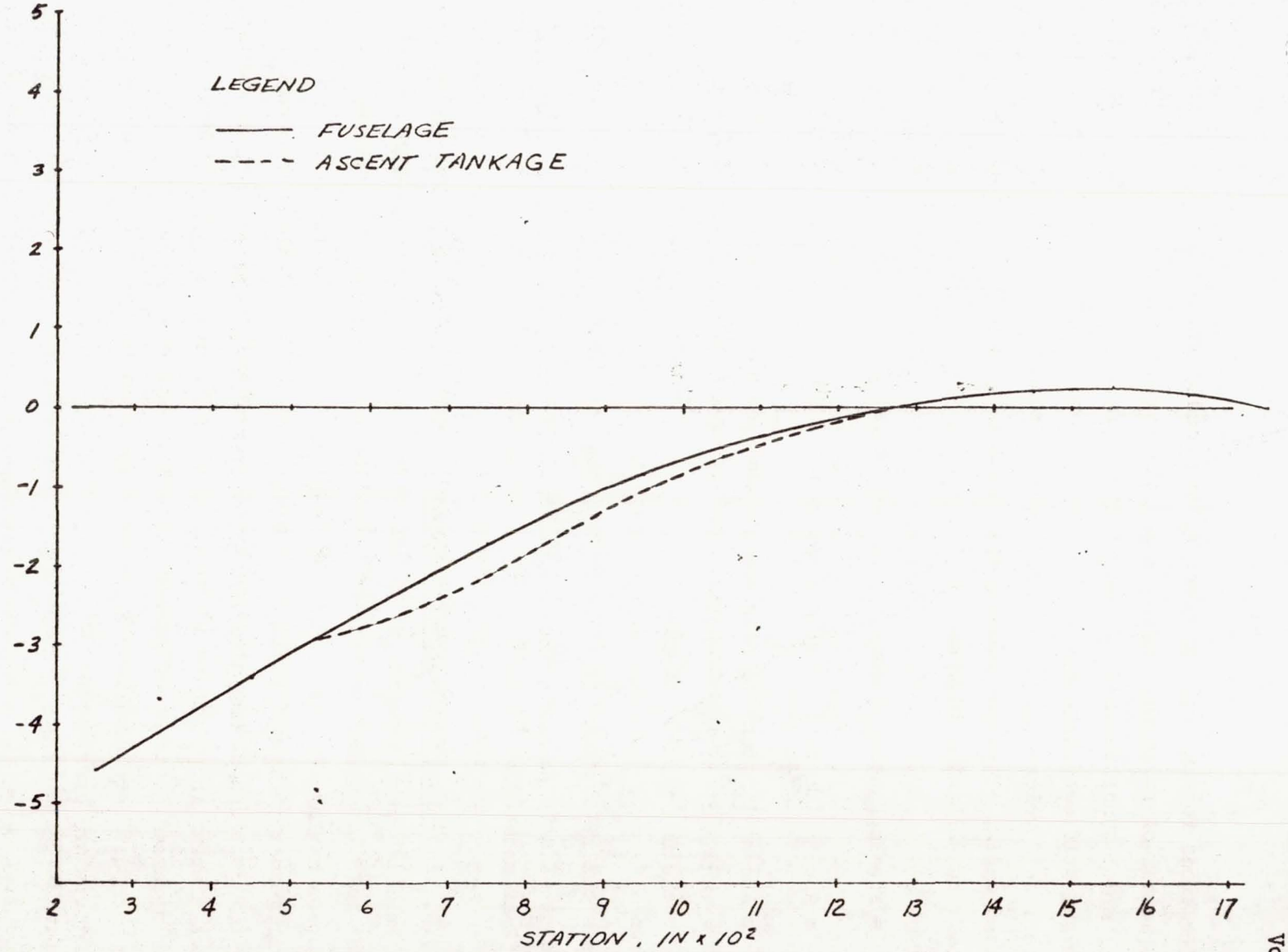


Fig. 3.7-14 Orbiter Fuselage and Ascent Tankage Transverse Deflection - Maximum Acceleration Condition

5 fuselage stations from F.S. 300 to the booster attach point at F.S. 1270. The maximum acceleration, maximum $\pm \alpha q$ and landing impact loading conditions were considered. Factors of safety for the ascent and landing conditions were 1.4 and 1.5, respectively.

Maximum upper and lower surface line loads are, respectively, shown in Figures 3.7-15 and 3.7-16, for the fuselage structure forward of the booster attach point at F.S. 1270. Critical loading conditions for the upper and lower surface are the maximum negative αq and maximum positive αq conditions, respectively. Critical loading condition for the fuselage structure aft of the booster attach point at F.S. 1270 is the maximum acceleration condition. Thus, the fuselage shell is sized entirely by ascent loading conditions. Although an active design condition for landing gear bulkheads, the landing impact condition does not size any of the fuselage shell structure. For the forward fuselage, maximum upper and lower surface compressive line loads are $(N_X)_{MAX}$ = -600 lb/in (ULT) and $(N_X)_{MAX}$ = -720 lb/in (ULT) respectively.

3.7.3.1.2 Forward Fuselage Structure Sizing. Based on results presented in Ref. 3.7-1 , aluminum primary structure with LI-1500 bonded to a titanium subpanel, where surface temperatures are between 1000°F and 2300°F, and a titanium heat shield with Dynaflex insulation, where temperatures are below 1000°F, was selected for the two-stage orbiter fuselage structure/TPS concept.

A trade study was performed to determine the effect of maximum aluminum fuselage shell temperature and exposure time on weights for the orbiter fuselage structure and TPS. By increasing the aluminum fuselage temperature from 200°F to 300°F, thermal analysis showed an incremental TPS weight reduction of 0.64 lb/ft^2 on the fuselage lower surface. Material properties for 7075-T6 show that the time-sensitive aluminum has a relatively stable structure up to about 200°F. As this temperature is violated, preciptiation proceeds, and overaging, with its associated loss in strength and corrosion resistance,

Figire 3.7-15

MAXIMUM UPPER SURFACE DESIGN LOADS
LS 400-7 ORBITER

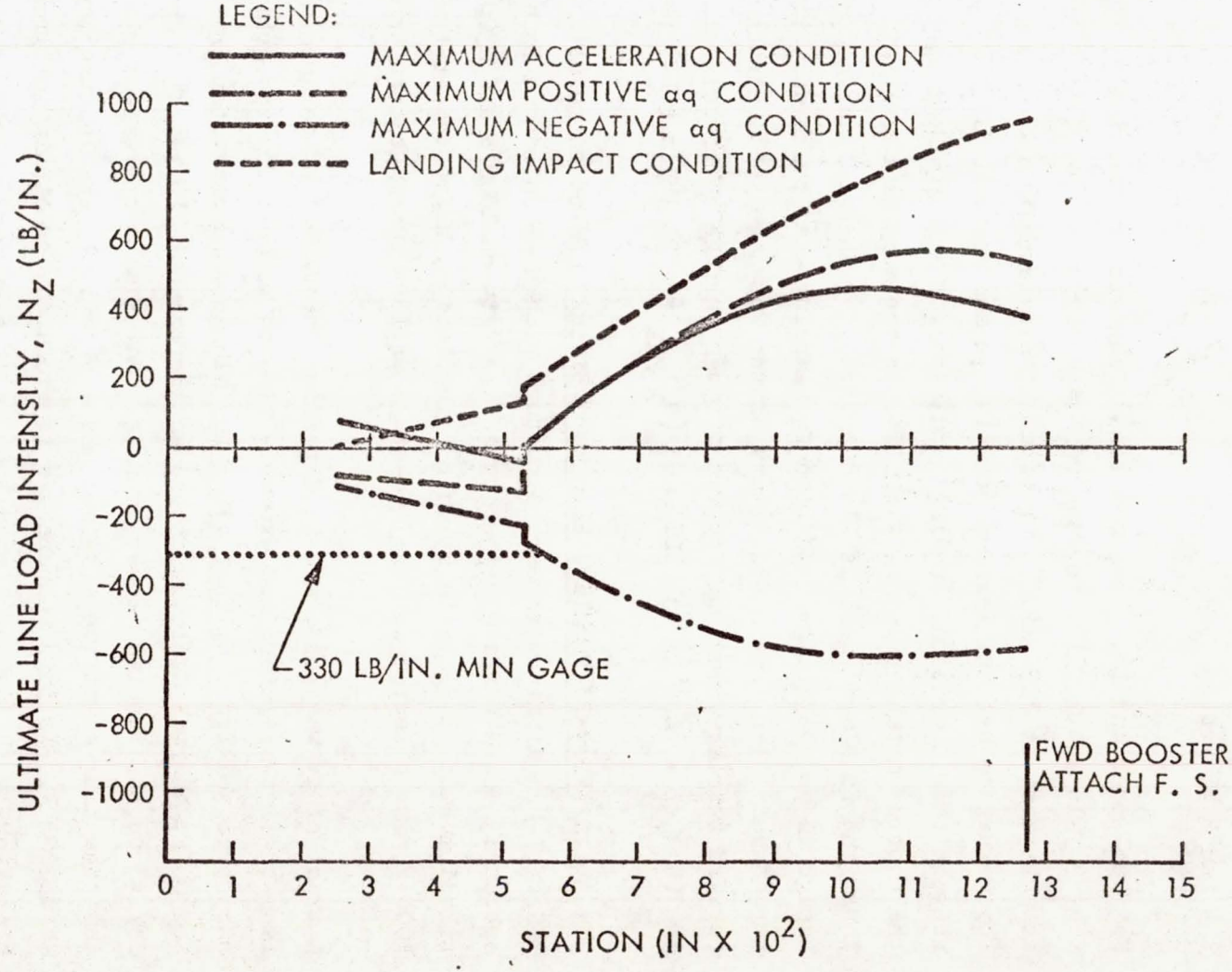


Fig. 3.7-15

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Figure 3.7-16

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MAXIMUM LOWER SURFACE DESIGN LOADS
LS 400-7 ORBITER

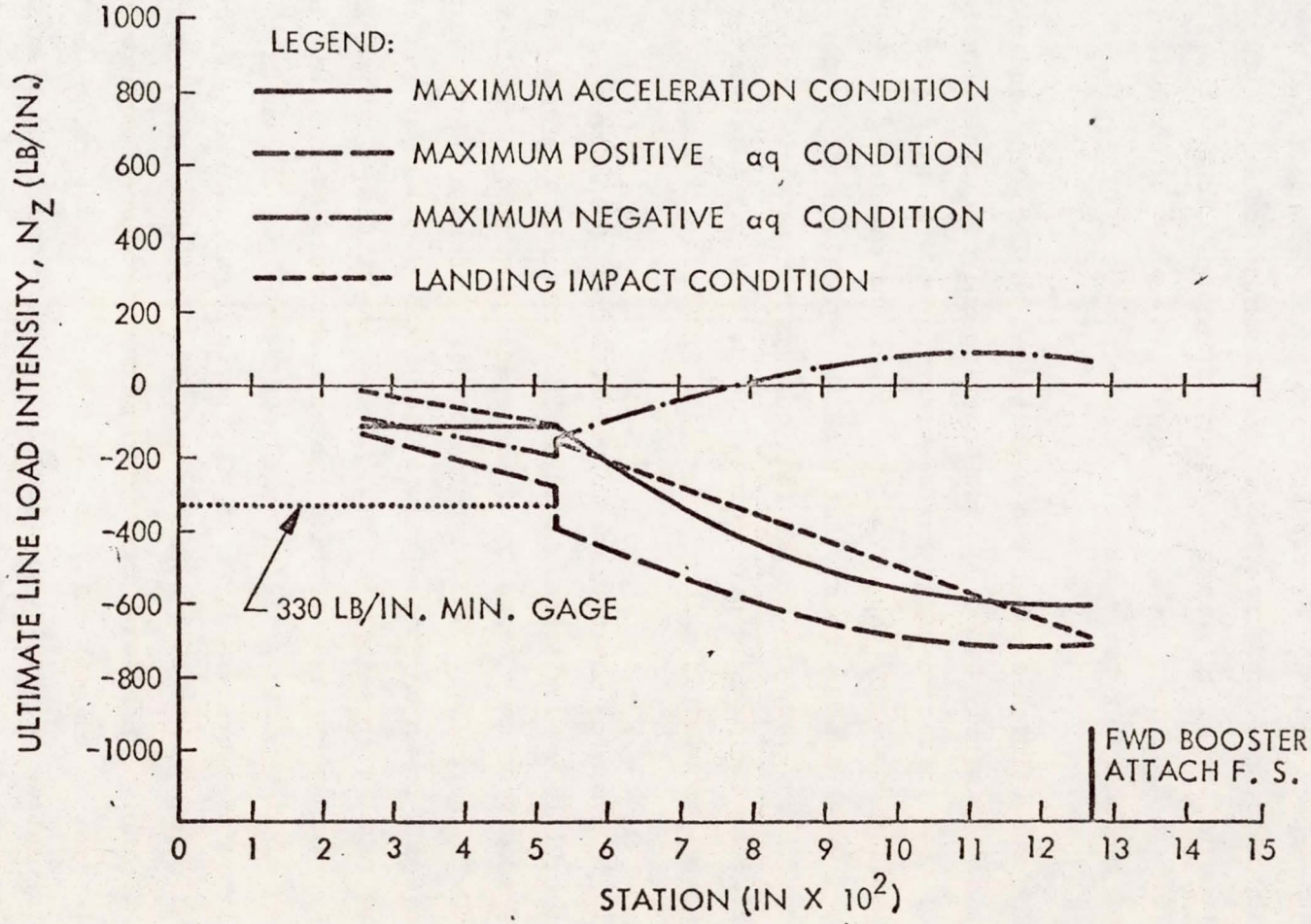

Fig. 3.7-16

3.7-23

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becomes a function of time. Based on a 6 percent DOD high crossrange and 94 percent NASA low crossrange mission distribution and a vehicle life of 100 missions, exposure time of the aluminum fuselage structure at elevated temperature is of relatively short duration. Thus, it was concluded that 7075-T6 aluminum, though time sensitive, could be exploited at 300°F because of the short exposure time.

The zee-stiffened panel was selected as the leading candidate fuselage cover panel primarily based on manufacturing and cost considerations. Since most of the orbiter fuselage surface is tapered with single or double curvature, tooling costs for the corrugated panel and machining costs for the integral zee panel are not consistent with the space shuttle design philosophy of low cost. However, the trapezoidal corrugation panel was selected for application to the payload bay. Since the payload bay is a simple cylinderical shape, manufacturing considerations are less critical.

In determining the optimum forward fuselage frame spacing, unit weights for the LI-1500 insulation, titanium TPS subpanel, aluminum fuselage panel and frames were considered for frame spacings varying from 20-in to 60-in in 10-in increments. To determine optimum fuselage panel and frame sizes, an average lower surface ultimate line load of 600 lb/in was selected. Based on results presented in Ref. 3.7-2 , zee-stiffened subpanels were selected. A preliminary lower surface subpanel pressure loading was established from an earlier study of a reusable reentry vehicle TPS where the ultimate collapse pressure was given as +2.5 psi and corresponding burst pressure a -1.6 psi for a 2.5 g subsonic maneuver with a temperature of 600°F at the subpanel. Based on a maximum aluminum structure temperature of 300°F, a parametric thermal analysis was performed to determine the required LI-1500 insulation thickness. Optimum titanium subpanel and aluminum fuselage panel equivalent thicknesses were used in the thermal analysis. Non-optimum factor of 1.05 was applied to the LI-1500 insulation to account for coating and

bondline. Non-optimum factor of 1.25 was applied to the TPS subpanel and fuselage structure to account for fasteners, edge closeouts and manufacturing considerations. As shown in Fig. 3.7-17 the minimum weight average fuselage frame spacing was found to be 36-in. However, the total unit weight of fuselage structure and TPS was nearly constant for frame spacings between 30-in and 40-in. Therefore, to minimize the number of fuselage frames, a frame spacing of 40-in was selected.

Zee-stiffened aluminum panel and frame average unit weight is shown in Figure 3.7-18 for the LS 400-7 orbiter forward fuselage. Loading intensities at 65 nodal points were used to determine fuselage panel and frame unit weights. A minimum gage face sheet thickness of 0.015-in was used to determine a lower bound of 0.538 lb/ft^2 on the aluminum panel unit weights. For a given fuselage cross section, the average unit weight is obtained from the ratio of total cross section weight per in. divided by cross section perimeter. The maximum unit fuselage structural weight of 1.27 lb/ft^2 includes the payload bay trough and non-optimum factor of 1.25.

3.7.3.1.3 Aft Fuselage Loading Intensities and Structural Sizing. Due to the concentrated longitudinal booster attach loads at F.S. 1270, severe tensile line loads and shear flows are introduced in the lower surface aft fuselage shell. Since elementary beam theory does not account for concentrated load effects, the SNAP/FSD computer code was employed to determine line loads, shear flows and structural weights for the aft fuselage. The SNAP/FSD computer code automatically generates fully stressed designs of large bar/shear panel structures. Given the initial design, the program computes all element stresses for each loading condition, then determines the extreme stresses occurring in each element over the entire range of loading conditions. Bar areas and panel thicknesses are then altered in proportion to the ratio of maximum stress to

Figure 3.7-17

OPTIMUM FORWARD FUSELAGE FRAME SPACING

TWO-STAGE ORBITER, MODEL LS-400-7

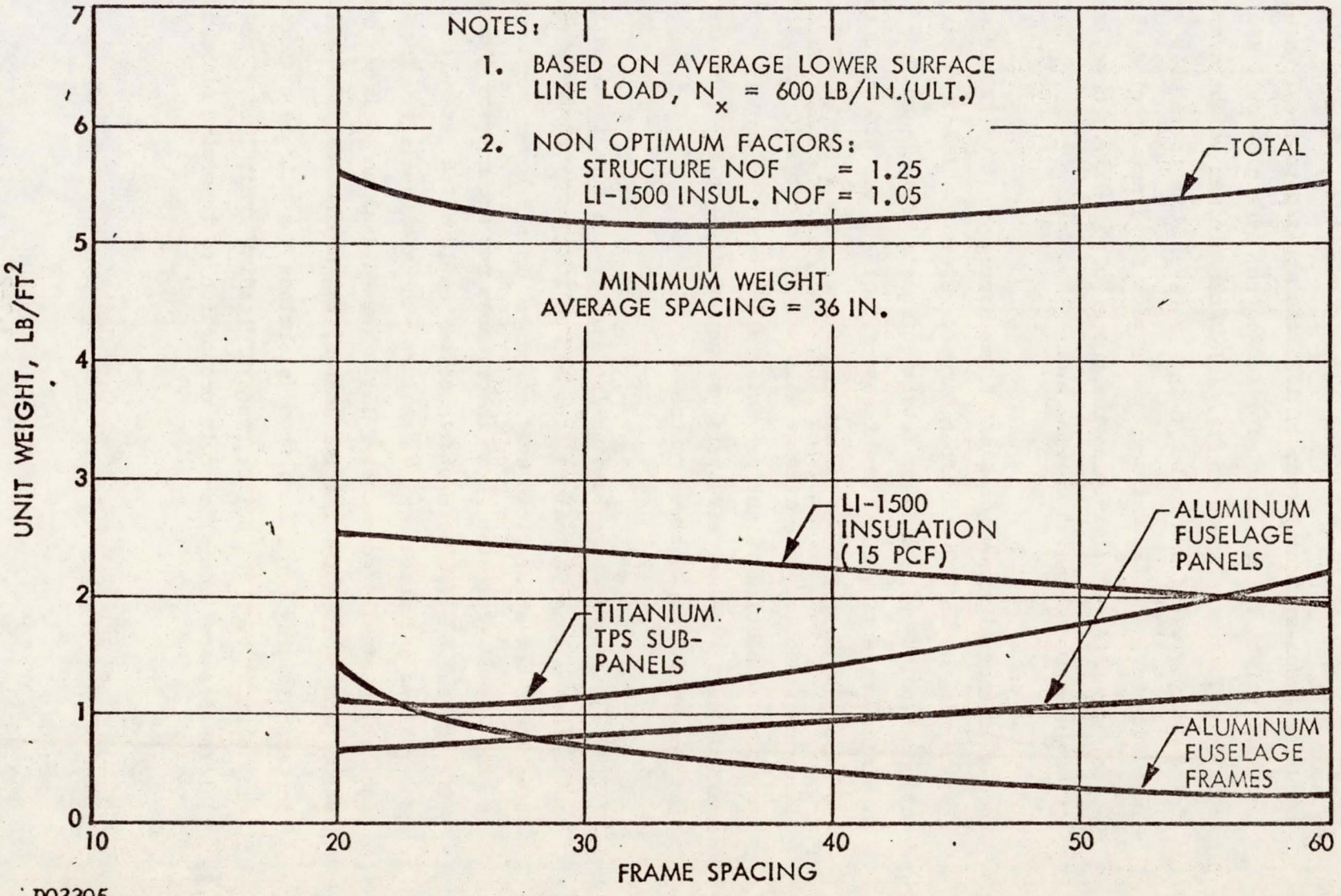

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Fig. 3.7-17

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ALUMINUM PANEL AND FRAME AVERAGE UNIT WEIGHT, FWD SECTION MODEL LS 400-7, 2-STAGE ORBITER

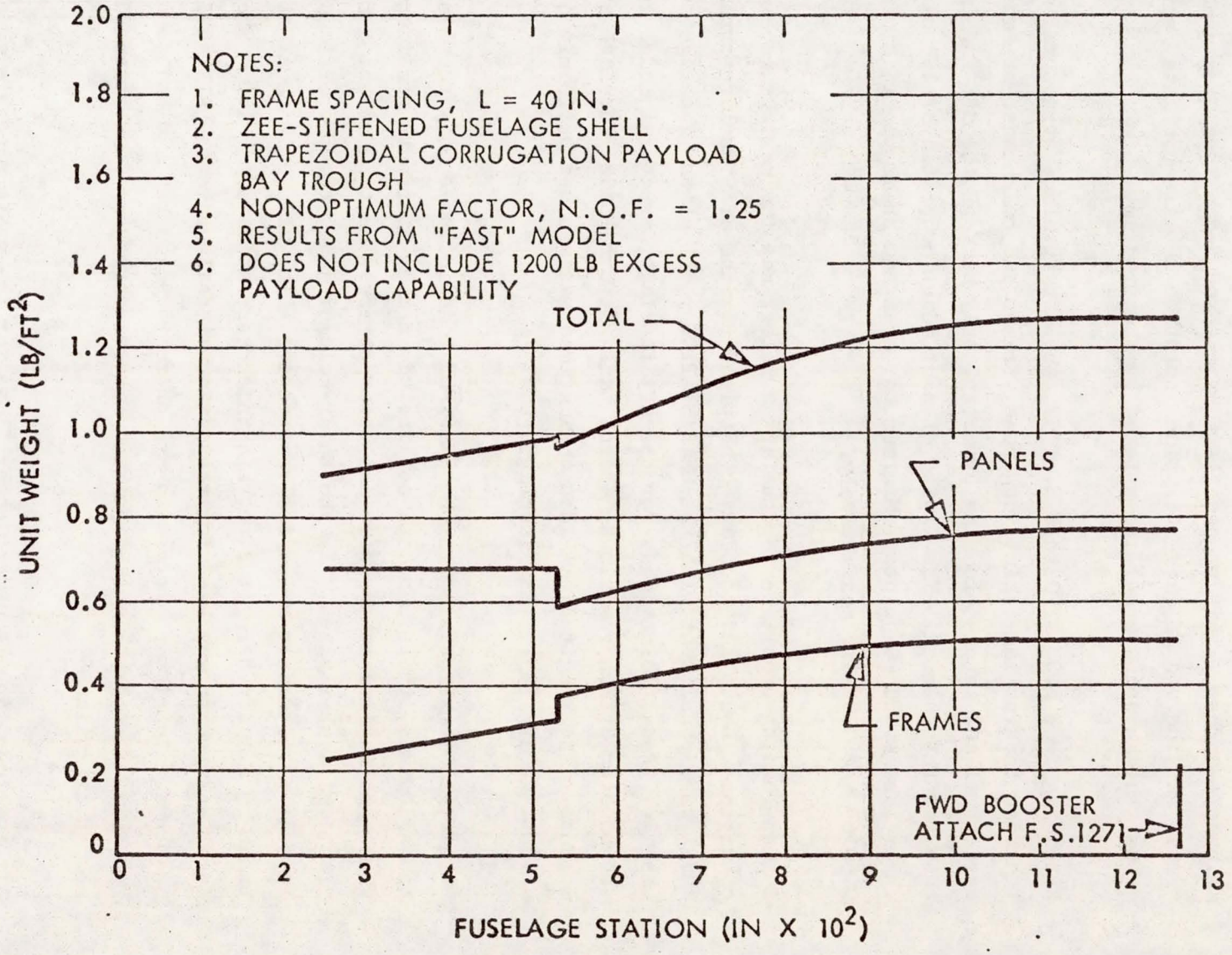


Fig. 3.7-18

D02927
apt

allowable stress. This process is then re-executed a specified number of times. At the end of each iteration, solution details (weights, stresses, areas, thicknesses, etc) are printed by the program.

Extending from F.S. 1270 to F.S. 1752, the aft fuselage structural model included the fuselage skin, longerons, frames and bulkheads. Because of structural and assumed loading symmetry, only half of the aft fuselage structure was represented. The finite element model consisted of 153 bars, 44 shear panels and 60 joints. Minimum bar area and shear panel thickness was 0.5-in^2 and 0.02-in, respectively.

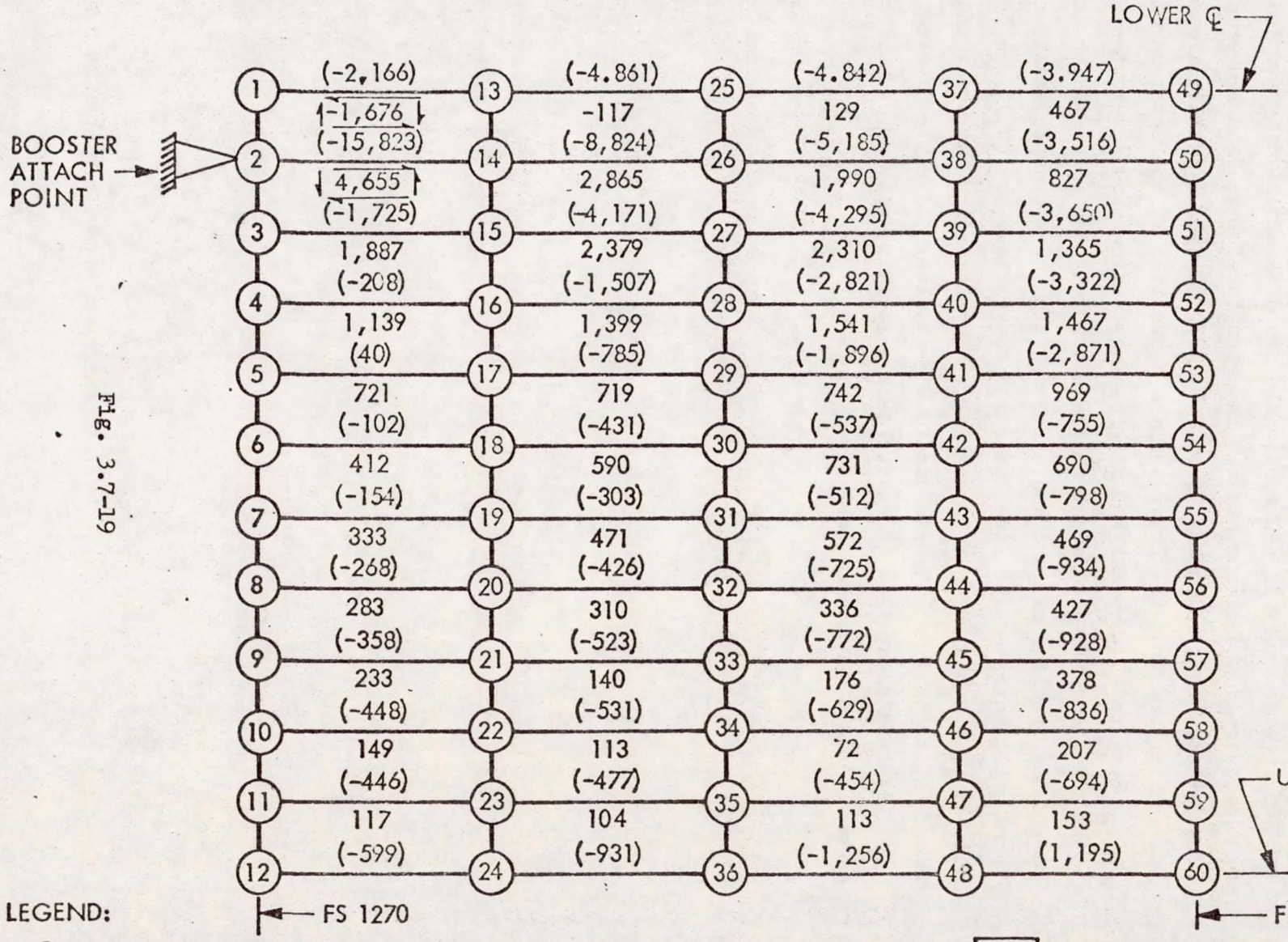
Design loading condition for the aft fuselage was the maximum acceleration condition. Loads and boundary conditions for the aft fuselage structural model were consistent with the SNAP/STATICS model of the fuselage and ascent tankage. Nodal point loads for the aft fuselage structural model included the booster attach reactions, ascent tank support loads, fuselage shears and moments. The longitudinal ascent tank load was assumed uniformly distributed across the aft end of the fuselage.

Structural member sizing in the SNAP/FSD code includes only the member force and allowable stress and does not include member stiffness considerations. Thus, the maximum compressive and shear stresses must be selected to provide practical member sizes. In the Euler column range, the proportional limit, F_{pl}, is a reasonable value for compression. The allowable shear stress was based on a stiffener pitch to skin thickness ratio, $b/t = 100$. The ultimate tensile strength was used for the maximum tensile stress. Allowable stress values for tension, compression and shear were, 80 ksi, 50 ksi, and 20 ksi, respectively. Excellent weight and stress convergence was obtained after five design iterations.

Aft fuselage line loads and shear flows for the maximum acceleration condition are shown in Figure 3.7-19. Corresponding discrete bar element cross-sectional

Figure 3.7-19

TWO-STAGE ORBITER AFT FUSELAGE ULTIMATE LINE LOADS AND SHEAR FLOWS MAXIMUM ACCELERATION CONDITION



LEGEND:

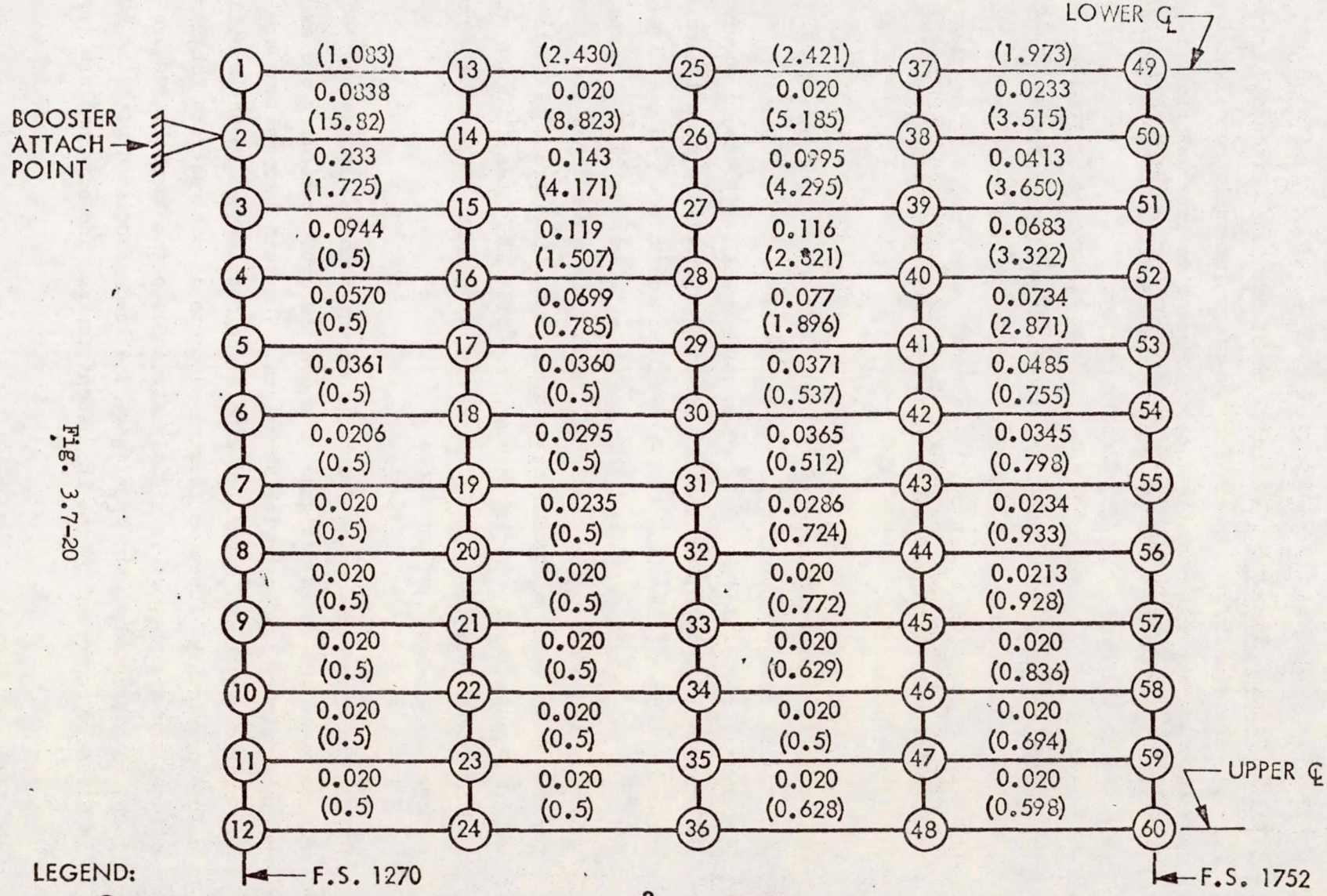
○ NODAL POINT (xxx) ULTIMATE LINE LOAD, LB (+ = COMPRESSION) [xxx] ULTIMATE SHEAR FLOW, LB/IN

D03311

areas and shear panel thicknesses are shown in Figure 3.7-20. Although the aft fuselage geometry is 3-dimensional, two dimensional diagrams are employed to depict loads and member sizes. The structural loads and member size diagrams extend, longitudinally, from F.S. 1270 to F.S. 1752 and, around the fuselage perimeter, from the lower surface centerline to the upper surface centerline. Results of the SNAP/FSD aft fuselage sizing show severe tensile line loads and shear flows adjacent to the booster attach points which decay rapidly toward the fuselage aft end and upper surface. For example, maximum lower surface ultimate tensile line loads and shear flows are $(N_x)_{MAX}$ = 15,800 lb/in and $(q)_{MAX}$ = 4,655 lb/in, respectively. Corresponding bar areas and shear panel thicknesses are 15.8-in^2 and 0.233-in, respectively. However, since the line load and shear flow intensity decays rapidly, many upper surface bars and shear panels are minimum size.

Figure 3.7-20

TWO-STAGE ORBITER AFT FUSELAGE DISCRETE BAR CROSS SECTION AREAS AND SHEAR PANEL THICKNESSES



D03310

3.7.3.2 Tankage Structure. Material considerations, discussed in section 2.8.5.5, are equally applicable for two-stage internal propellant tanks. The factors-of-safety for reusable tanks subjected to internal pressure are identical to those used for the stage-and-one-half. Also, the material selected is the same, 2219-T87 Aluminum alloy.

The design criteria used for sizing the tankage differs slightly from that presented for the stage-and-one-half system. Ullage pressure for the ascent LD2 tank is 20 psia in lieu of the cascaded value used for the stage-and-one-half LO2 ascent tank. All other data presented in Tables 2.8-5 through 2.8-11 are applicable.

All tanks are initially sized for internal pressure considerations. The LO2 ascent tank design condition is a combination of two conditions: the forward 25 percent of the tank is designed by maximum ullage pressure occurring in the warmed-up empty condition, and the remainder of the tank is designed by the maximum 3g acceleration full-propellant-condition. To minimize plumbing and residual weight penalties, the LH2 ascent tanks in line with the LO2 tanks are cascaded to the center LH2 tank. All three tanks are sized initially for the empty warm-up condition, when maximum ullage pressure is reached. No weight penalty results from cascading the side LH2 tanks into the middle LH2 tank. The on-orbit LO2 and LH2 tanks are also critical for the warmed-up maximum ullage-pressure condition.

Unlike the stage-and-one-half orbiter ascent tankage, the ascent tankage system for model LS 400-7A is designed to share the orbiter bending and axial loads, thus being a primary load-carrying structure. The twin tank systems are assembled as two continuous beams, supported within the fuselage at these locations, as described previously. The two forward supports do not restrict longitudinal deformations; the aft support is rigidly attached to the thrust structure which, in turn, is connected to the fuselage. The load associated with the LO_2 ascent tanks is directly supported by the LH_2 ascent tanks. The aft fuselage (primarily the

lower surface) provides the load path from the LO2/LH2 tank systems supported off the thrust structure to the booster attach fittings. Axial load distribution, which is shown in Fig. 3.7-7, represents the maximum condition-booster maximum acceleration. The severe bending moment, resulting primarily from the eccentricity of the LO2 center-of-mass to the booster attach fittings, is shared by both the fuselage and the ascent tankage.

A computer model of combined tankage and fuselage was analyzed, using an iterative analytical procedure of EI stiffness distributions calculated at discrete locations. A typical moment distribution resulting from the analysis is shown in Fig. 3.7-10. This figure shows how the total moment is shared between the tankage and the airframe. Fig. 3.7-21 shows the results from the computer analysis for the tank system. The moment diagrams shown in this figure include an ultimate factor-of-safety of 1.4 for ascent-loading conditions and 1.5 for the landing-load condition. The moments shown in Fig. 3.7-21 combined with axial loads and tank internal pressures are used for tank structural sizing.

For the LO2/LH2 tank system to resist the bending and axial loads associated with the ascent and landing conditions, structural reinforcement is required. Two types of reinforcement may be considered: monocoque shell with intermediate rings and the second to include the addition of longitudinal stiffeners. To satisfy the loads to which the tanks are subjected, the monocoque design requires the shell thickness to be increased beyond that which is required for maximum internal pressure. The alternate approach of adding longitudinal stiffeners in lieu of thickening the shell membrane, in general, will be the more structurally efficient (lighter design) way to resist the external loads.

Comparing designs using both approaches revealed that significant weight savings are achieved by consideration of longitudinal stiffeners for the LO_2 tank. The critical load condition occurs during landing impact when the tank is essentially unpressurized. Accordingly, it was decided to provide

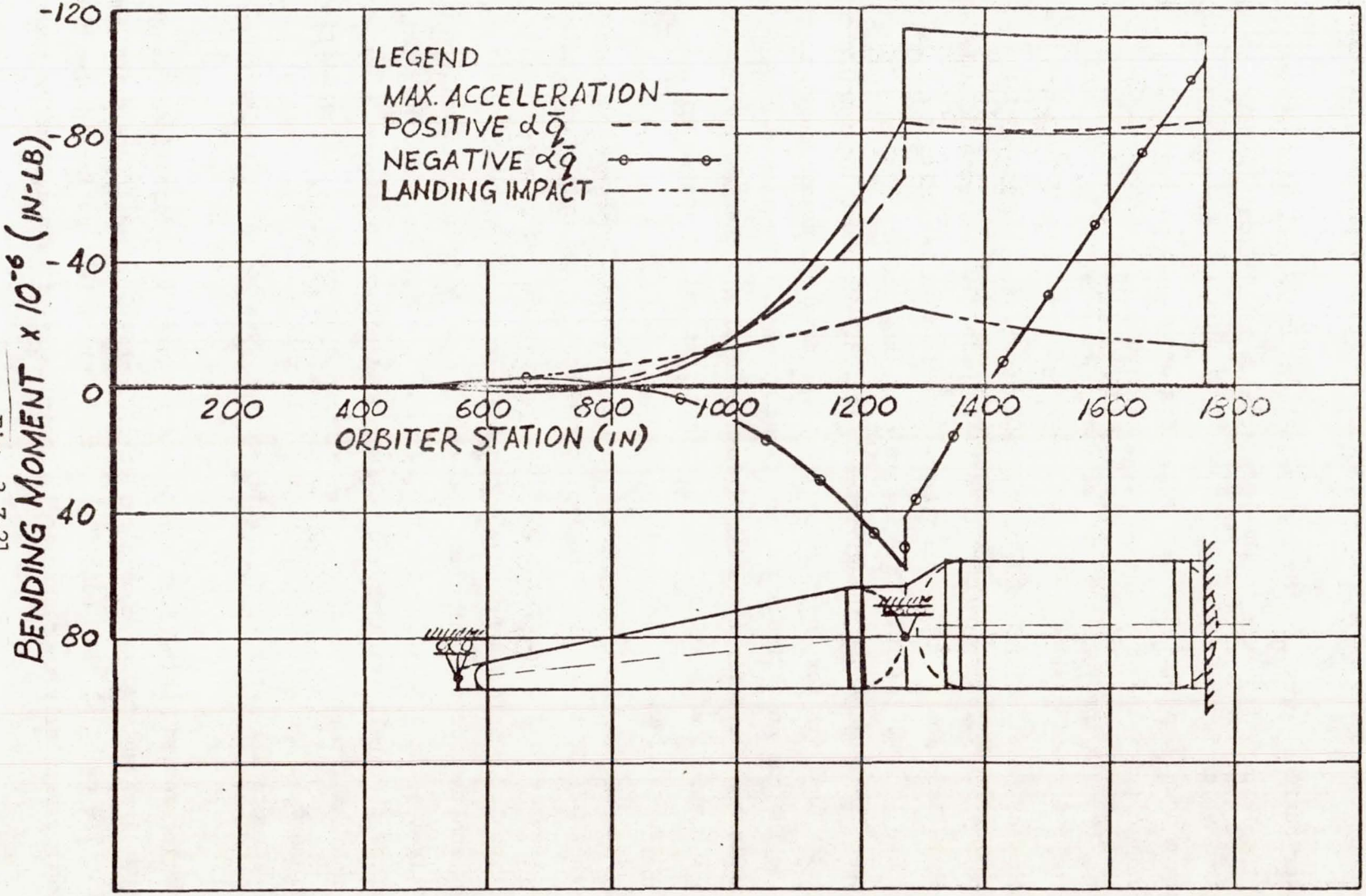


Fig. 3.7-21 2-Stage Orbiter Ascent Tanks Ultimate Bending Moment

rings and longitudinal stiffeners to the LO_2 tank membrane to resist external loading.

For the LH_2 tank, thickening the shell sufficiently to resist the external loads results in a severe weight penalty when compared to the latter design approach of adding longitudinal stiffeners. Therefore, it was decided to add internal rings (to additionally support slosh baffles) and external longitudinal stiffeners to the membrane shell thickness, which is sized for internal pressure, as was done for the LO_2 tank.

The analytical approach follows the same procedure described previously in section 2.8.5.1 for the sizing of the stage-and-one-half droptank shells.

The intertank structure, tying the LO2 tank to the LH2 tank is a twelve-pointed trussed-frame structure which attaches to the aft payload-compartment bulkhead at Sta 1270. This attachment is designed so that only transverse loads are reacted at that location, thus permitting the tanks to translate longitudinally. A similar attachment is designed for the forward LO2 tank connection to the forward payload-compartment bulkhead.

Support structure attaching the aft end of the LH2 tank to the thrust structure is also a twelve-pointed trussed-frame.

The non pressure vessel structures are made from 7075-T6 Aluminum tubular extrusions, mechanically fastened to form the cylindrically shaped sections. Similar types of supports are used for the other propellant tanks.

The detailed analysis of the propellant tankage is presented in EM L2-11-04-M5-1.

3.7.3.3 Thrust Structure. The thrust structure concept for the two-stage delta-body orbiter is similar to the stage-and-one-half concept but different in complexity and weight. It consists of five vertical trusses, interconnected

to a horizontal beam, with auxiliary supports for lateral stability and attachment to the fuselage, as illustrated in Fig. 3.7-22. Intermediate vertical trusses help distribute thrust loads to the aft fuselage, while providing lateral stability and intermediate continuity of the horizontal beam. The network of trusses acting with the beam provides a base for the tank supports to distribute tank propellant loads to the fuselage, hence to the booster.

Time did not permit detailed design of this structure to be pursued, but it is assumed that 6Al-4V Titanium alloy will result in being the lightest among the conventional materials, and is, therefore, selected as the baseline candidate. Weight-scaling laws were used to help determine the weight of this structure.

THRUST STRUCTURE CONCEPT FOR TWO-STAGE DELTA-BODY ORBITER

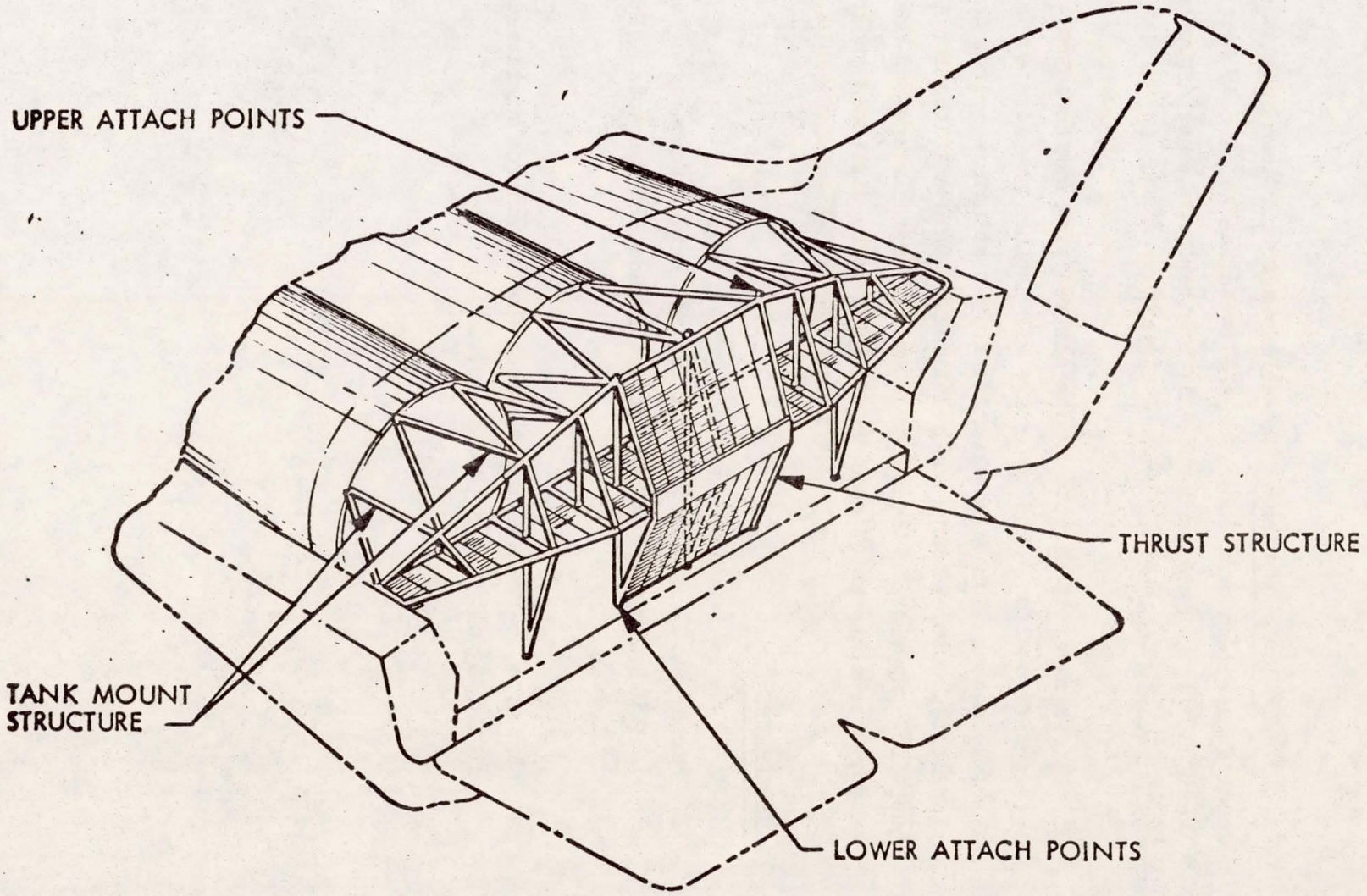


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Figure 3.7-22

3.7.3.4 Major Subsystems. The aerodynamic surfaces, payload compartment door, landing gear, and crew cabin for the two-stage delta body orbiter are structurally identical to the stage-and-one-half orbiter. Comments discussed in Section 2.8.5 apply equally well here.

There is a major difference in the airbreathing engine support structure, primarily because of location. The two-stage A/B engines are located in the base of the aft fuselage and supported from the thrust structure. From the standpoint of simplicity, cost, weight, and reliability, this aft location is, without question, superior to the belly location for the stage-and-one-half A/B engine support structure which must be designed to be distended and retracted.

REFERENCES

1. Richie, C.C., "Structural Sizing and Weight Comparison for LS 200-2 and LS 200-5 Orbiter Fuselage Structure and TPS Concepts," EM No. L2-01-01-M1-3, 17 December 1970.

2. Chinn, A., "Preliminary Subpanel Evaluation for Rigidized Insulation," EM No. L3-01-01-M3, 3 March 1971.

3.8 PROPULSION SYSTEMS

Propulsion requirements for orbit injection, orbit transfer, orbital maneuvers, reentry retro, vehicle attitude control; and subsonic landing assistance are satisfied by the operational capabilities of four propulsion systems. These four systems and their major functions are listed as follows:

1. Main propulsion system (MPS) for vehicle injection into orbit.
2. Orbit-maneuver propulsion system (OMPS) for orbit-to-orbit transfer, on-orbit maneuvers, and reentry retro.
3. Attitude-control propulsion system (ACPS) for vehicle orientation and minor translations on-orbit and vehicle orientation during the initial portion of reentry after retro.
4. Airbreathing propulsion system (ABPS) for landing field approach and landing go-around. Supplemental propulsion subsystems may be required for subsonic ferry flights between the landing field and the launch site. These supplemental subsystems are separate from the basic airbreathing propulsion system and are provided in kit form.

The propulsion systems for the 2-stage orbiter vehicle have been designed to overcome the vehicle-peculiar problems and to produce maximum performance with logical conservatism.

3.8.1 Main Propulsion System

The main propulsion system (MPS) provides impulse to inject the space shuttle orbiter vehicle into a 50 nm by 100 nm elliptical orbit. A layout drawing of the system is presented in Fig. 3.8-1.

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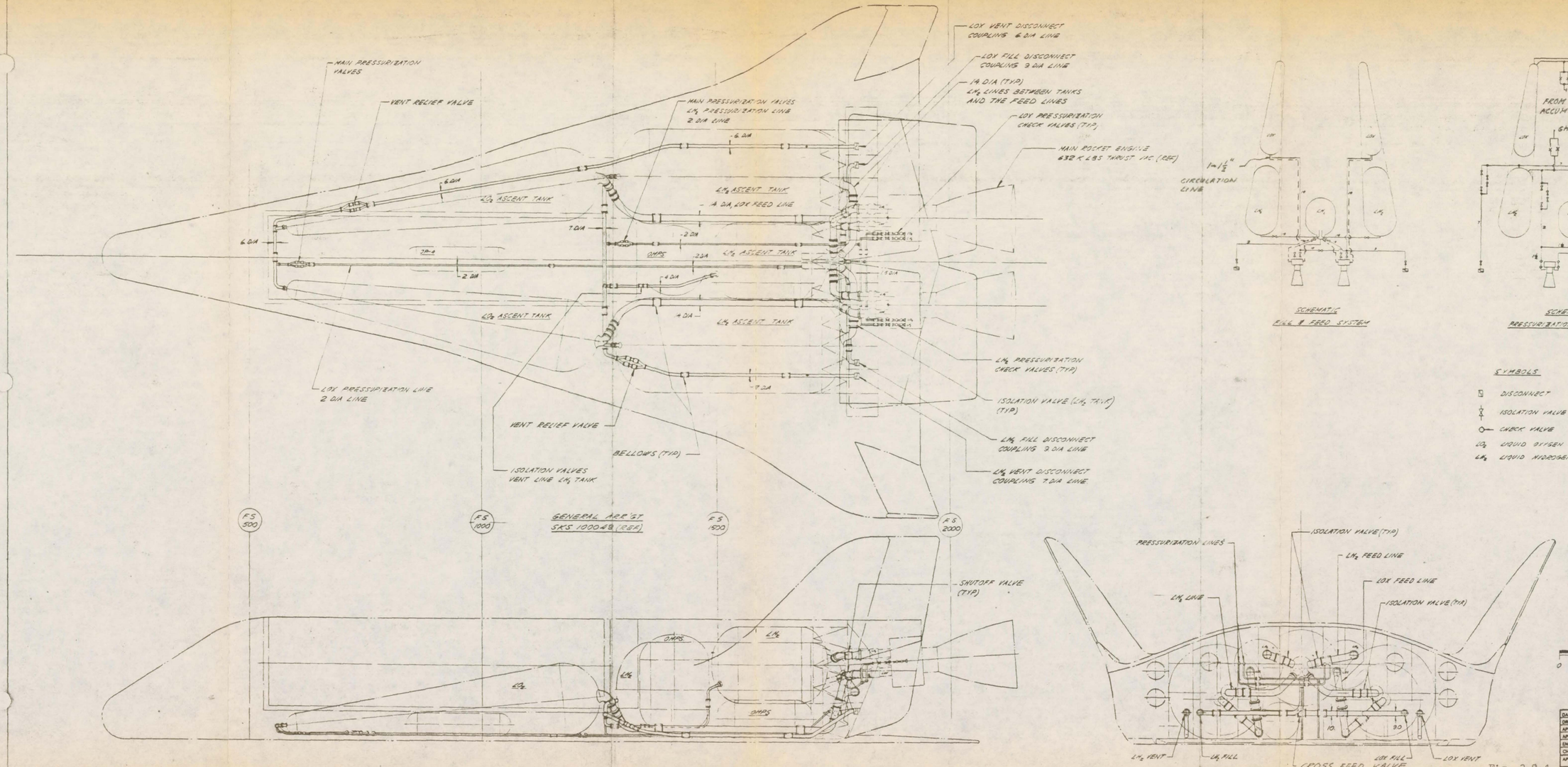
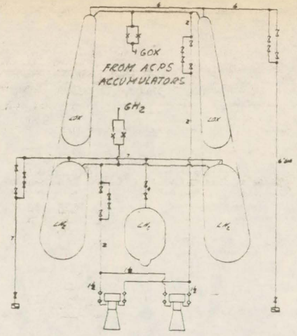


SCHEMATIC
FUEL & FEED SYSTEM



SCHEMATIC
PRESSURIZATION & VENTING SYSTEM

SYMBOLS

- DISCONNECT
- ISOLATION VALVE
- CHECK VALVE
- LO_2 LIQUID OXYGEN
- LH_2 LIQUID HYDROGEN

PROPULSION SYSTEM SCHEMATIC
REF. SKH 03197

Fig. 3.8-1

0 5 10 15 20 25 FT
SCALE

DATE 3-21-71	LOCKHEED MISSILES & SPACE COMPANY, A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION, SUNNYVALE, CALIFORNIA		
DR ENG TOLLO	PLUMBING SYSTEM		
APPD ENGRG	MAIN PROPULSION SYSTEM		
APPD CHK	MODEL LS-100-7		
APPD	SIZE 7	CODE IDENT 06987	DRAWING NO. SKT 100224
APPD	SCALE: FS	SHEET	REV A

Fig. 3.8-1

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This propulsion system consists of two high-pressure oxygen-hydrogen rocket engines supported by a propellant storage and feed system incorporating flow controls including propellant utilization control, pressurization, venting, and thermal protection subsystems. The two rocket engines in the base of the vehicle are defined by and installed in compliance with the NASA Interface Control Document 13M15000B dated 1 March 1971. Both rocket engines are gimballed ± 7 deg (square pattern) to provide thrust vector control (TVC) in roll, pitch, and yaw modes during the ascent-to-orbit injection. Ignition of the MPS rocket engines after stage separation is accomplished using GO_2 and GH_2 for engine start prepressurant, drawn from the pressurized gas accumulators which feed the ACPS thrusters. Both engines operate at the NPL thrust until the vehicle reaches the 3 g level and are throttled thereafter so that the vehicle acceleration does not exceed 3g. In the event that it becomes necessary to shutdown one engine and continue operating the remaining engine in an abort mode, provisions are included for transferring propellants from the OMPS tanks to the MPS tanks to offset increased gravity losses induced by the reduction in the vehicle acceleration-time profile. Vehicle roll control will be provided by the ACPS thrusters, which are estimated to use no more than 174 lbm of propellants for this purpose.

Propellant tankage for the MPS consists of two conical LO_2 tanks, arranged on either side of the payload bay, and three cylindrical LH_2 tanks, arranged to feed both engines from the central tank as depicted in Fig. 3.8-1. A brief summary of MPS system characteristics is presented in Table 3.8-1.

3.8.1.1 Propellant Feed System. The propellant feed system with plumbing for filling the tanks, as shown in Fig. 3.8-2, has been configured to minimize residuals and propellant losses within the limitations imposed by the arrangement of tankage in the orbiter vehicle. Propellant tank sizes are established from impulse requirements for the reference mission (south polar, 100 nm circular orbit) and from operational analyses which take into account the average density of propellants in the tanks at liftoff with volume allowances

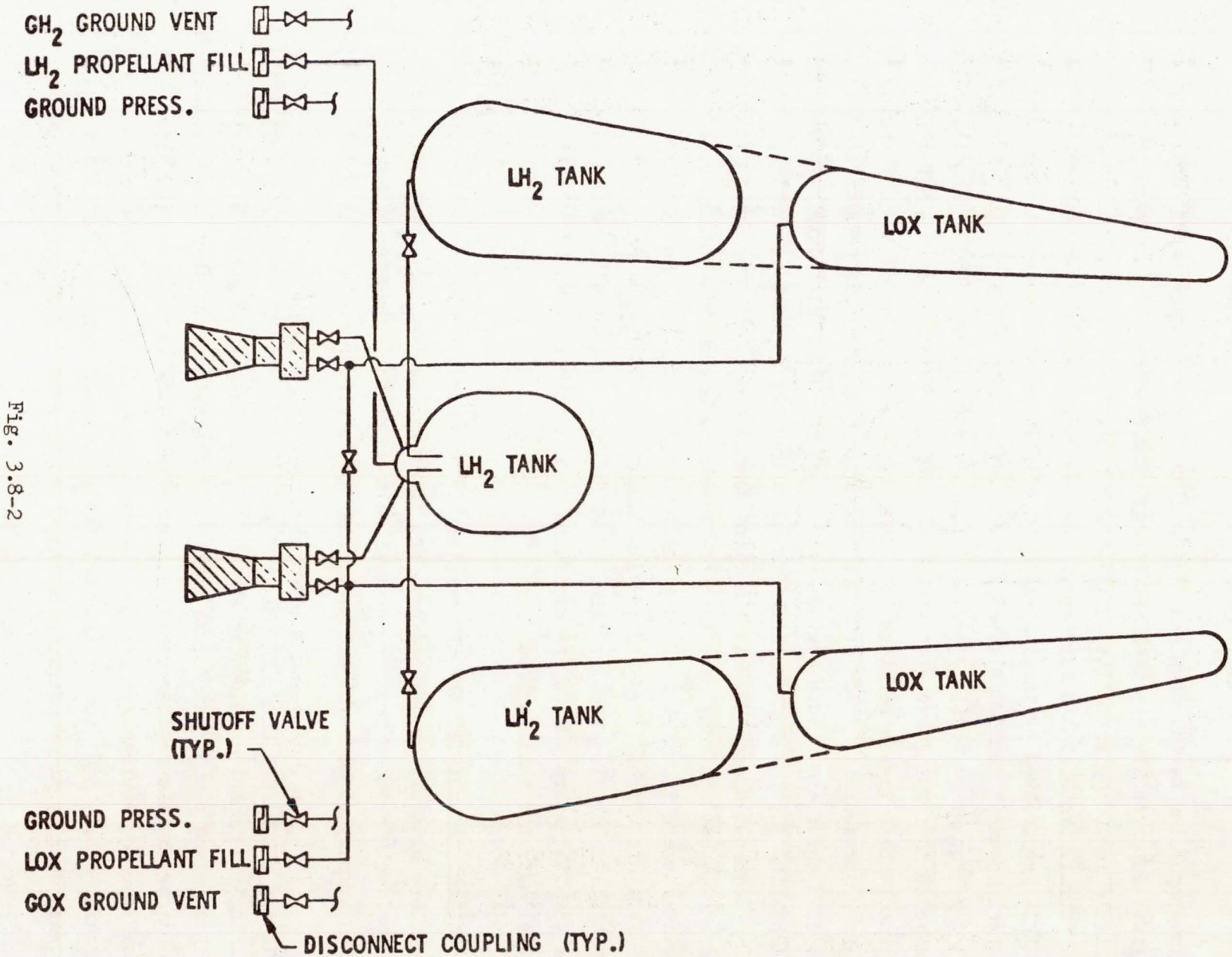

Fig. 3.8-2 2-Stage Orbiter Propellant Fill and Feed System

Table 3.8-1

Main Propulsion System Characteristics Summary

A. Engine System	
Engine Type	As defined by Space Shuttle Vehicle/Engine 550K (SL) Interface Control Document 13M15000B
Propellant	LO_2 and LH_2
Nozzle Expansion Ratio	150:1
Throttle Range, Percent NPL	50 to 109
Number of Engines	2
Thrust Vector Control:	
Engine Gimbal Capability, deg	±7 square pattern
Overshoot, deg	0.5
Performance, NPL:	
Thrust (2 engines), nominal, lb	1,264,000 (vacuum)
Specific Impulse, sec	459 ±3 (vacuum)
Mixture Ratio O/F (nominal)	6.0
B. Propellant System	
Propellant Weight, lb:(1)	
Loaded	559,669
Impulse	546,439
Reserve	6,000
In-flight losses	6,444
Residuals	786
Tank Ullage Pressures:	
O_2 Tank (max)	
Boost	TBD
Operation	TBD
H_2 Tank (max)	
Boost	TBD
Operation	TBD

Table 3.8-1

(1) For South Polar, 100 NM Circular Orbit Reference Mission

for the impulse propellant (based on minimum performance of the rocket engines), residuals, in-flight liquid/vapor losses, ullage space, flight performance reserve, loading tolerances; and tank volumetric changes induced by internal pressure/cryogenic liquid temperature.

The vehicle system operational requirement for propellant loading to be accomplished within 2 hours from a standby status, results in a procedure for chilling and filling oxygen and hydrogen propellant tanks simultaneously. Inert, dry nitrogen gas, contained in the tanks under a slight positive pressure to preclude induction of moisture and particulate contaiminants (and ambient oxygen in the hydrogen tanks), is purged from the tanks with gaseous oxygen and gaseous hydrogen prior to the initiation of the liquid propellant fill operation. Liquid hydrogen is pumped into the three tanks via the central tank and flows evenly into all three tanks after the liquid level in the center tank exceeds the heighth of the 14 in. diameter crossflow standpipes. When the center tank is full (3 percent ullage for this tank is provided for in the two side tanks), the vent valve for this tank (See Fig. 3.8-3) is closed, which forces all the remaining hydrogen to be loaded into the two side tanks. Liquid oxygen is pumped into the 9 in. diameter crossflow pipe near the rocket engines and flows evenly up the 14 in. diameter feedlines and into the tanks. Control of propellant fill operations in both oxygen and hydrogen tanks is provided through the use of in-tank point-level sensors with fill completion and topping controlled thru the use of redundant linear capacitance probes arrayed about the tank full-level point.

Formation of ice/liquid air on the propellant tanks is prevented by a dry nitrogen purge within the orbiter stage plus a layer of foam insulation approximately 1 in. thick applied to the hydrogen tanks to prevent the formation of liquid/solid nitrogen.

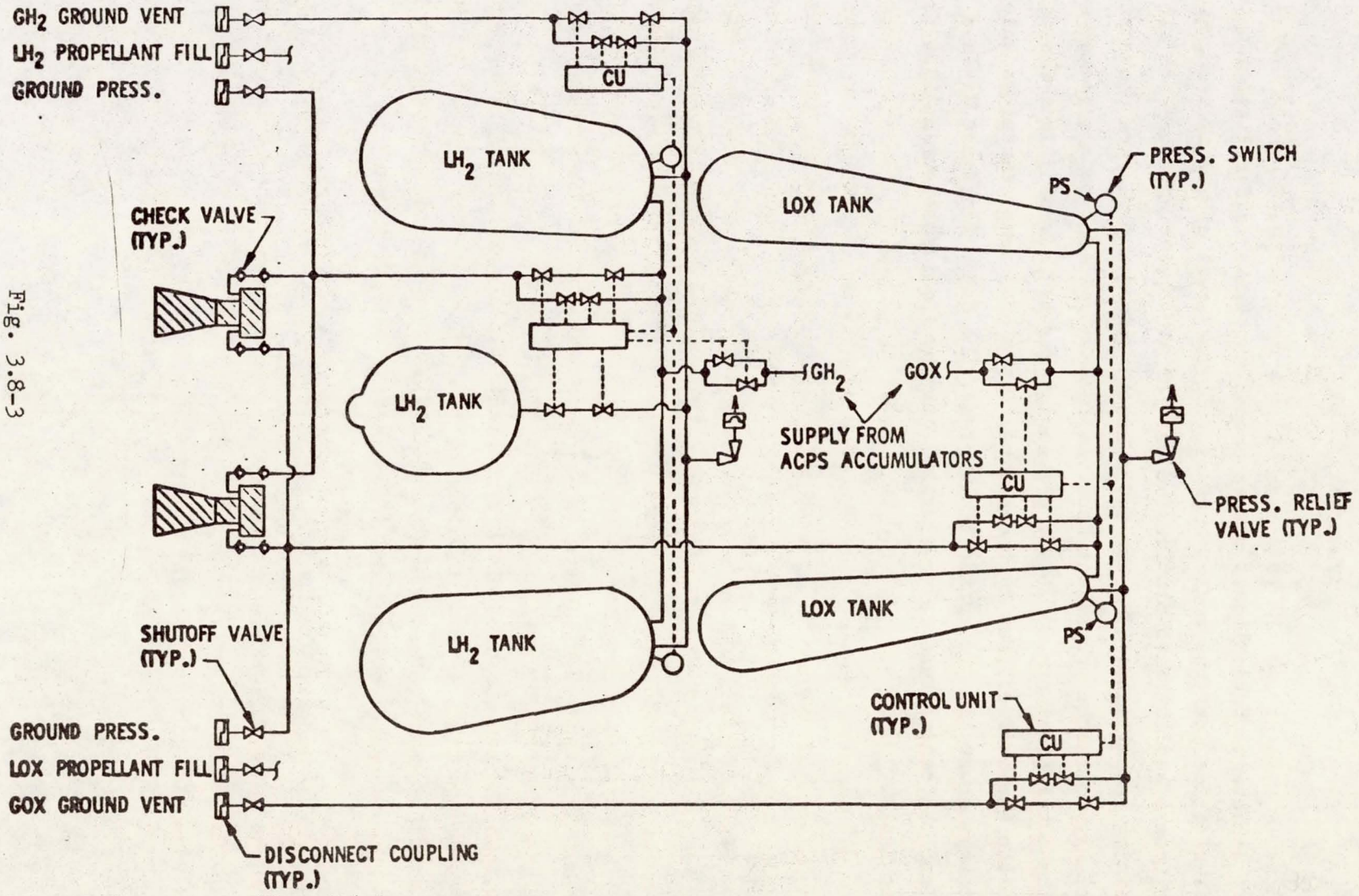

Fig. 3.8-3 2-Stage Orbiter Main Propulsion Pressurization and Venting System

Equal depletion of the liquid oxygen propellant tanks is facilitated by the cross-feedline near the rocket engines. In the event of a rocket engine failure, this cross-feedline also provides access for the operating engine to all the remaining propellants. The two outer-liquid hydrogen tanks are pressurized initially to induce propellant flow via the central tank to the rocket engines. As the outer tanks are depleted, pressurant gas is prevented from induction into the engine feedlines by the height of the crossfeed standpipes which discharge upward in the center tank. Thus, the outer tanks are drained essentially dry and the center tank is pressurized only by liquid/pressurant flow from the outer tanks.

The feedlines and engine pumps are kept chilled prior to launch and up to the time of engine ignition after stage separation by means of a pumped propellant-circulation system. Electrically powered centrifugal pumps (not shown in Fig. 3.8-2) submerged in the oxidizer and fuel tanks circulate propellant to the engines downstream of the pumps via series-redundant check valves. From the engine, the propellants flow counter-current into the respective feedlines and back to the propellant tank. This approach eliminates the type of geysering that would take place if the tanks were filled prior to chilling the engines/feedlines.

3.8.1.2 Pressurization/Venting System. The pressurization/venting system for the 2-stage orbiter vehicle incorporates separate plumbing and pressure control components for pressurization and venting. The preferred mode for pressurization system operation is based on a capability to start/stop rocket engine bleed gas flow so that provisions for venting excess pressurant overboard are not required. Preliminary evaluations of an alternate pressurization operational concept, based on a constant, uninterrupted flow of pressurizing gas from the rocket engine, indicates that a flow rate can be selected which will minimize requirements for overboard venting. Ullage pressure requirements for engine prestart are satisfied at minimum system weight when using gas drawn from the ACPS gas accumulators at approximately 380°R. This operational concept is reflected in Fig. 3.8-3. The baseline pressure control system - consisting of redundant pressure switches, voting logic circuits, series-redundant pressurization control valves, and series-parallel pressure relief valves - is backed up by direct-sensing mechanical valves for overpressure safety and tank rupture prevention. A second feature of this system arrangement, separate ground pressurant supply couplings, enables introduction of helium during ground-fill operations, if necessary to prevent an impending tank implosion during the chill-fill cycle.

3.8.2 Orbit Maneuvering Propulsion System

The orbit-maneuvering propulsion system (OMPS) provides the impulses for orbit maneuvers after injection into the reference injection orbit. In conjunction with impulses for small impulse increment maneuvers delivered by the ACPS thrusters, it is required to provide a total orbital velocity increment of 1,500 ft/sec. The tank volume is sized for an additional capacity of 500 ft/sec.

This system consists of two RL-10 engines and the separate propellant tank and feed system designed for the long storage-time requirement. Only one RL-10 engine is used for normal operation, the second being a standby providing engine-out capability. The two RL-10 engines are gimballed and appropriately canted for alignment with the vehicle cg, permitting single-engine operation as the normal mode. The system as designed provides a fail-operational capability in the event of an RL-10 engine failure. If the second RL-10 fails, the retro maneuver would have to be accomplished with the ACPS, at an additional weight penalty in propellants because of the lower system specific impulse of the ACPS. This penalty is not reflected in the propellant data given. Before incorporating it, an analysis should be performed to determine whether it is more effective to add the additional propellant capability or to provide a third OMPS engine. This latter course appears to be lighter but may impose additional installation problems.

The basic system design is capable of efficient storage of propellants over a 30-day period in orbit. For the baseline configuration, the insulation is sized for a 7-day mission. A system schematic is presented in Fig. 3.8-4. A summary of system characteristics is presented in Table 3.8-2.

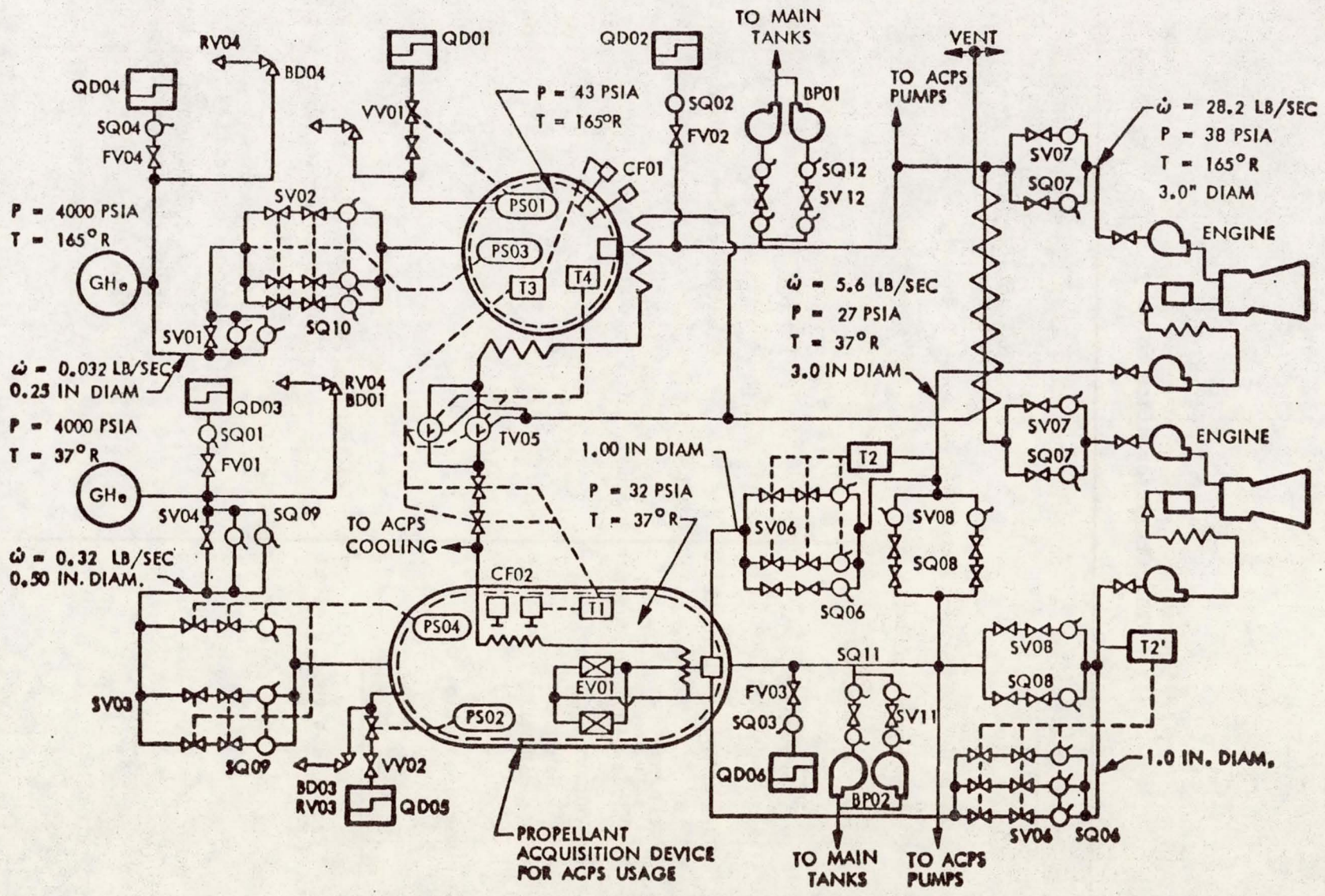

Fig. 3.8-4 Orbit Maneuvering Propulsion System

Table 3.8-2

OMPS Characteristics Summary

A. Engine System	
Engine Type	PWA RL10A-3-3A
Propellant Type	LO_2 and LH_2
Nozzle Expansion Ratio	57:1
Number of Engines:	
Total	2
Operating	1
Standby	1
Thrust Vector Control	Electric-motor gimbal actuation
Performance:	
Thrust (nom vacuum per engine), lb	15,000
Vacuum I_{sp}, sec	
Nominal	444
Minimum	439
Mixture Ratio, O/F	5
B. Propellant System	
Propellant Feed System:	Tankage separated from ascent tankage
	Designed for 30-day propellant storage capability
	Sized to 2,000-ft/sec performance capability for reference mission (1)
	H_2 tank shared with ACPS, APS
	O_2 tank shared with ACPS, APS

Table 3-8-2 (Cont'd)

B. Propellant System (Cont'd)

Propellant Weight, lb: (1)	
Loaded	39,674 (2)
Impulse	36,715 (3)
Flight Performance Reserve	0 (4)
In-flight Losses	363
Residuals	137
Tank Ullage Pressure, psi:	
H_2 Tank	32
O_2 Tank	43
Other Requirements:	
Insulation	Tanks-Multilayer Lines-Vacuum-jacketed
Line and Tank Temp Control	By TCU System
Propellant Gauging and Instrumentation	TBD
System Electric Power (max.), KWDC	0.25
System Helium, Total, lb	230
Performance	

(1) Reference Mission: 55 deg inclination, 270 nm circular orbit

(2) Includes ACPS and APU Propellant

(3) 2,000 ft/sec Orbital-Vehicle ΔV capability (reduced by 142 ft/sec provided by ACPS)

(4) Flight reserve allowance included in ΔV capability requirement.

Table 3.8-2 (Cont'd)

3.8.2.1 OMPS Rocket Engine. The rocket engine selected for OMPS is the PWA Model RL-10A-3-3A. This rocket engine is identical to the RL-10A-3-3 presently being used on the Centaur stage, except that provisions have been incorporated for operating the engine at reduced propellant feed NPSP. The RL-10 rocket engine uses a regeneratively cooled thrust chamber and a turbopump-fed propellant system operating on a closed expander cycle. Multiple-start capability is provided. Normal operation requires the vehicle to supply helium gas at 500 psi for valve actuation and electrical power for ignition and valve control at 24 to 32 vdc (9 amp maximum).

3.8.2.2 OMPS Propellant System. Helium is used for prepressurization and pressurization of both propellant tanks. Helium prepressurization is used to eliminate the problems associated with the condensation consideration of gaseous propellant vapors used in an autogenous system. For the multiple burn operation of the OMPS, the last prepressurization operation requires almost as much helium as is required for the burn itself. Thus, the simplest system is helium pressurization. The helium is stored in two tanks, one for each propellant. These tanks are located outside the propellant tank but under its insulation so that the helium is at the same temperature as the propellant to be pressurized. The eliminates collapse problems during operation, and does not increase the tank temperature since a hot gas is not introduced. Thus the tank design can fully utilize the higher material properties at cryogenic temperatures.

The propellant transfer system is designed to feed either of the two RL-10 engines. A different technique is used for each propellant. For the hydrogen, selector valves located near the tank are used to isolate the lines to each engine, so that only one need be chilled for each operation. The line is dumped at engine shutdown. Line chilling up to the engine is accomplished by a secondary flow line taking propellant from the propellant acquisition device in the tank. The propellant acquisition and retention syslem is configured to satisfy ACPS operational requirements for on-orbit, low g conditions. The oxygen feed system uses selector valves located near the engine and between engine firings the feed line is chilled up to that point, using hydrogen vapor from the Thermal control unit (TCU) in the hydrogen tank.

The TCU operates in an open expansion refrigeration cycle to maintain a constant LH_2 temperature ($\simeq$ 37°R) in orbit. Hydrogen is expanded through a Joule-Thompson valve and is passed through an in-tank heat exchanger to remove excess heat from the tank contents. Small circulation fans stir the hydrogen tank contents to ensure uniform conditions. The hydrogen vent gas is then used to chill the LO_2 tank and its acquisition and retention device and the LO_2 feedline. Circulation fans in the LO_2 tank are driven by a motor located outside the tank and magnetically coupled to the fans. Tank temperature sensors are used to open the hydrogen vent system and select whether the tank is chilled or not. A separate H_2 vent line, separately controlled, is used to provide hydrogen for cooling the ACPS pumps.

The integrated storage tanks hold the hydrogen and oxygen required for the OMPS, the ACPS, and the APUs. The tanks and the OMPS hydrogen feedlines are insulated with high-performance multilayer insulation, which is optimized for a 7-day period in orbit and purged during atmospheric operations. The OMPS LO_2 feedlines are filled during ground operations, and are insulated with vacuum-jacketed multilayers insulation. It is noted that this insulation system may prove unsatisfactory during the entry and landing phases when some APU reactants are required. If so, a small vacuum-jacketed tank, filled from the main tank, may be required.

Provisions have been made to transfer propellant from the OMPS tanks to the main tanks in the event of an abort due to a main engine failure. The system uses redundant boost pumps, to ensure that the abort can be successfully completed. Each pump is protected by two squib valves and a shutoff valve in series so that this emergency system can be isolated during normal orbital operation in the event of leakage.

3.8.3 Attitude Control Propulsion System

The attitude control propulsion system provides roll control during operation of the OMPS. During orbit and reentry operations, it provides attitude control and three-axis rotational and three-axis translational maneuver capability. It also provides roll control during an abort resulting from loss of a main engine.

The thrusters use high-pressure GO_2-GH_2 propellant stored in liquid form in the orbital propellant tanks. The liquids, pumped through heat exchangers, are converted to propellant vapors and stored in accumulators from which they flow at regulated pressure to the thrusters.

Forty identical 1,500-lb thrust thrusters are used, with 16 located in the forward section and 24 in the aft section. The number of thrusters results from the selection of single thrust-level thrusters and from the necessity to meet fail-operational/fail-safe capability under all conditions including reentry when the down-firing thrusters must be retracted. Pure rotational couples about all three axes is achieved during orbital flight. During the reentry flight phase, however, pure couples are not possible about all axes, because all down-firing thrusters are retracted into the vehicle. Forward and aft and up and down translation in orbit is accomplished without cross-coupling. Laterial translations in orbit, however, produce cross-coupling.

A summary of system characteristics is presented in Table 3.8-3. The design data assumptions, which have been the basis of sizing analyses, are listed in Table 3.8.4.

2.8.3.1 Thruster Characteristics. The minimum impulse value per thruster in its regular operational mode is 50 lb-sec. In order to minimize propellant consumption for attitude hold, when the use rate is determined chiefly by the minimum impulse of the thrusters in a limit cycle mode of operation, the desired low thrust can be achieved by pulsing the igniters only. The minimum impulse value per thruster obtained by this mode of operation is assumed to be 5 lb-sec.
Other performance characteristics are listed in Table 3.8-5.

Table 3.8-3

ACPS Design/Performance Characteristics

Propellant:	
Type	GO_2 and GH_2
Weight, lb	
Loaded	5,016
Impulse	4,475
Reserve (10% of impulse)	447
Thrusters:	
Type	TBD Single Design
Number	40
Thrust, per Thruster, lb	1,500
Min. Impulse Bit, lb-sec	
Thruster	50
Igniter Pulsing	5
Chamber Pressure, psia	250
Specific Impulse, Steady-State, sec	425
Propellant Supply System:	
Propellant Storage	In OMPS Tanks
Propellant Conditioning	Gas Generator - Heat Exchanger Assemblies
Number of Units, O_2/H_2	1/1
Outlet Temperature, ^oR	380
Outlet Pressure, Range psia	2,000/500
Gas Accumulators	
Number O_2/H_2	3/3
Operating Pressure Range, psia	2,000/500
Storage Capacity (usable total) O_2/H_2, lb	208.5
System Power Requirements:	
Maximum, KWDC	3.5
System Performance:	
Specific Impulse, sec	
Sustained Burn,	362
Pulsed Burn	352
O/F Ratio	3.3
System Orbital Control Capability	
Rotational Acceleration, deg/sec^2	1.0 (all axes)
Translation Acceleration, ft/sec^2	0.5 (all axes)
Total Vehicle Translation Capability, ft/sec	142
Sustained Operation Capability	4 Thrusters at 1.5K thrust
Total Impulse Capability Without Reloading of Gas Accumulators, lb-sec	88,600

Table 3.8-3

Table 3.8-4

ACPS Design Data Assumptions

Vehicle Weight On-Orbit, lb	373,000 to 335,000
Moments Inertia, slug/ft^2:	Reentry
Pitch	10.4×10^6
Yaw	13.6×10^6
Roll	2.7×10^6
Orbital Rotational Acceleration, deg/sec^2	1. (all axes)
Orbital Translational Acceleration, ft/sec^2	0.5 (all axes)
Reentry Rotational Acceleration, deg/sec^2	
Pitch	0.8
Yaw	1.0
Roll	0.67

Table 3.8-4

Table 3.8-5

ACPS THRUSTER CHARACTERISTICS

Thrust, lb	1,500
Min Impulse Bit, lb-sec:	
Thruster	50
Pulsing Igniter	5
Chamber Pressure, psia	250
O/F Ratio:	
Thruster	4.0
Pulsing Igniter	1.0
Specific Impulse, sec	
Full Thrust	425
Minimum Impulse Bit (Thruster)	TBD
Minimum Impulse Bit (Igniter Pulsing)	TBD

3.8.3.2 Thruster Installation Arrangement. Forty 1,500-lb thrust thrusters on the orbiter, with 16 located in the forward section and 24 in the aft section as indicated in Fig. 3.8-5. With this thruster configuration and thrust level limited to 1,500 lb per thruster, it is possible to achieve rotation and translation in all axes except lateral tranlation without cross-coupling during orbital flight. During reentry, however pure couples are not possible in all axes, because all down-firing thrusters are retracted in the vehicle. Under this condition, pure couples are not possible in pitch and roll, because only the up-firing thrusters are available for operation. Cross-coupling also occurs in yaw during reentry, because only the forward side-firing thrusters are available at this time. Forward and aft translation in orbit is accomplished without cross-coupling. Lateral translation, however, produces cross-coupling due to the use of single level thruster and available thruster locations, resulting in unequal lever arms about the cg.

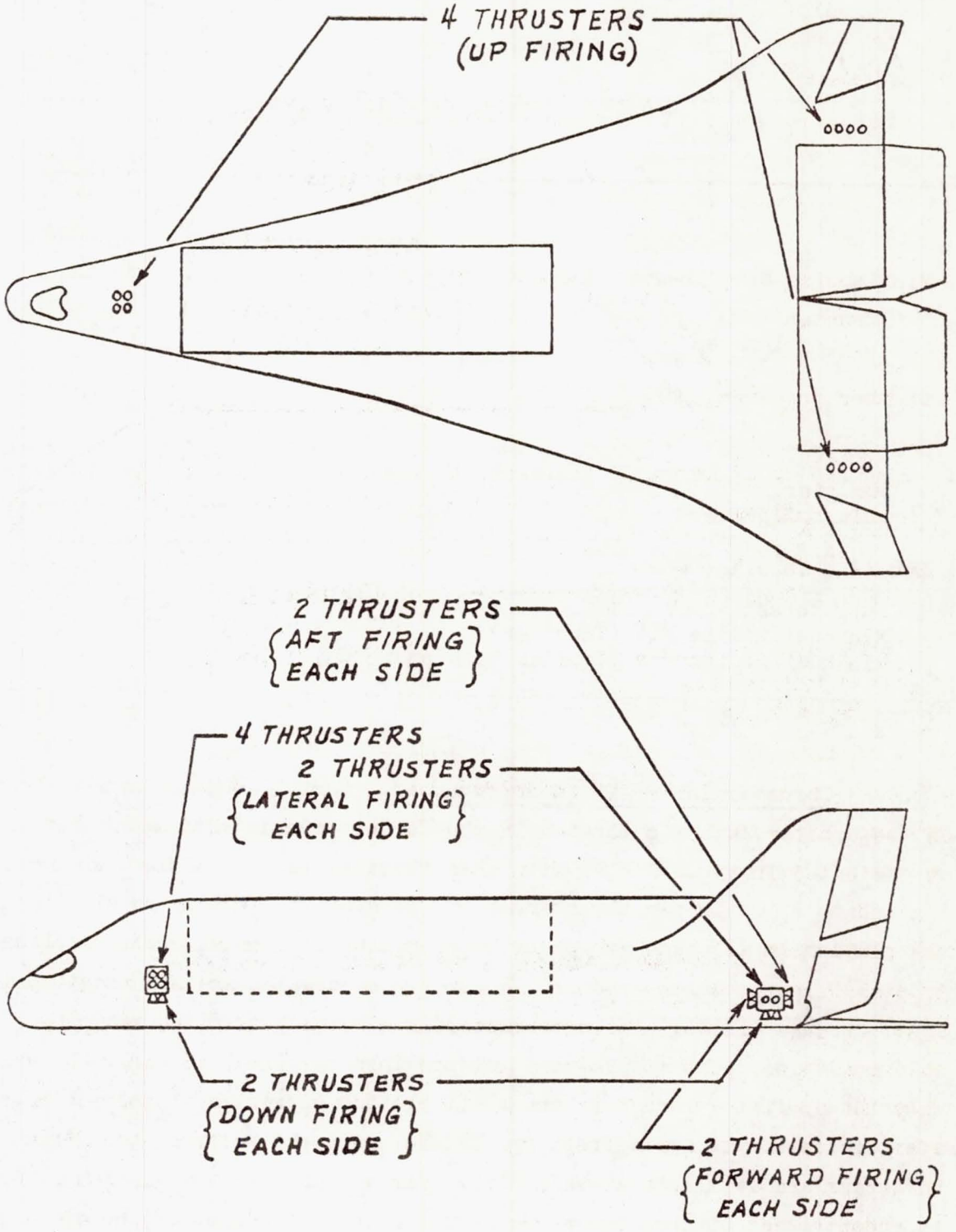

Fig. 3.8-5 ACPS Thruster Locations

3.8.3.3 Propellant System. Liquid propellants for the ACPS are integrated with the APU reactants in the OMPS storage tanks as noted in Fig. 3.8-4 and Table 3.8-2. A capillary feeding system in the OMPS tanks provides positive propellant feeding under zero and adverse low acceleration conditions. The required thermal gradient for capillary operation is maintained by a thermal conditioning unit (TCU) in the OMPS hydrogen tank which removes heat by the refrigeration action of hydrogen expansion and vaporization. This TCU also provides hydrogen for cooling the ACPS pumps and feedlines used in the three separate propellant conditioning units provided for each propellant. Each unit consists of a turbine driven pump and a heat exchanger with two gas generators, one to drive the turbine and one to supply hot gas to the heat exchanger, as shown in Fig. 3.8-6. A temperature sensor at the pump inlet controls the cooling flow by opening and closing the valve in the cooling line downstream of the pump. Redundant valves upstream of the pump provide a shutoff of these individual vent lines in the event of a failure of the downstream valve. Cooling gas from the hydrogen pump can be used to cool the oxygen pump or be vented overboard.

Each propellant conditioning unit is sized to provide 50 percent of the full-flow requirement. Therefore, full capability is available with the failure of one oxygen and/or one hydrogen conditioning unit. Two conditioning unit failures in either propellant system still provide 50 percent capability which will provide safe operating characteristics.

The pumps transfer the high pressure propellants to heat exchangers which raise its temperature to 380°R for storage in the accumulators at a maximum pressure of 2000 psia. Separate gas generators are used to provide gas to the heat exchangers, and to power the turbines which drives the pumps, since the turbine exhaust gases do not provide sufficient heat to condition the propellants. Each pair of generators are supplied from the accumulators through lines with separate shutoff valves at the generators, and common redundant normally-open valves to ensure that two failures in one line does not result in an open drain in the accumulators. Doubly redundant pressure

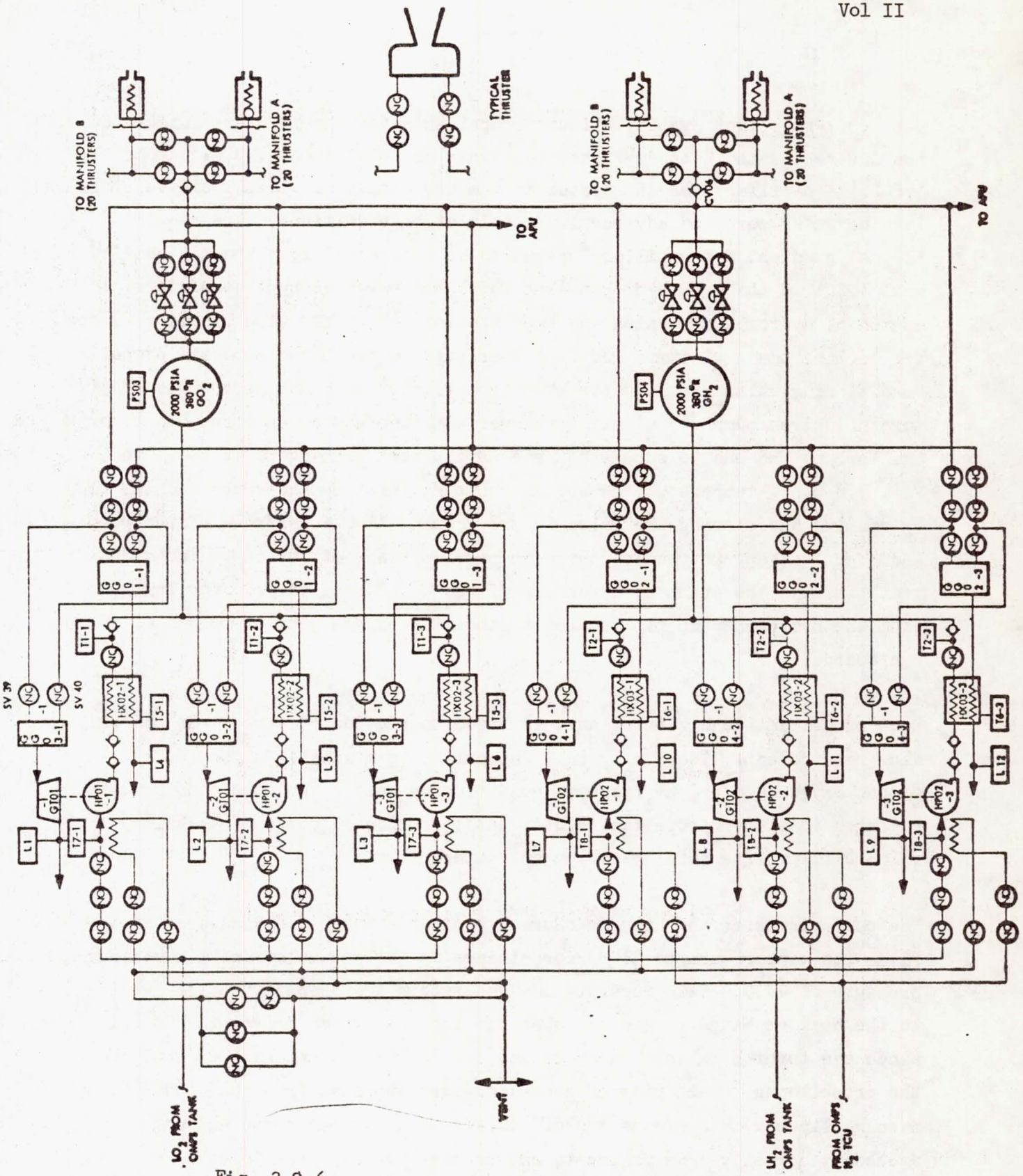

Fig. 3.8-6 ACPS Propellant Conditioning System

switches on the accumulators shutoff the conditioning units at 2000 psia, and start them at 900 psia, a broad range selected to minimize the number of pump cycles.

Flow from the accumulators passes through a triple redundant regulator set. Each leg is provided with a solenoid valve and a squib valve to insure that it can be closed in the event of a double failure. Flow from the regulators feeds the two ACPS manifolds, each serving 20 thrusters, as well as the APU system. Two normally-open valves are provided so that each propellant leg of each manifold can be isolated if excessive line leakage occurs which cannot be isolated elsewhere. The thruster units themselves are each equipped with series redundant valves. Analysis showed that the fail-operational, fail-safe criteria could be met with a lighter system by adding nine additional thrusters with series redundant valves rather than using quad redundant valves on the 31 thrusters which are minimally required.

3.8.4 Airbreathing Propulsion System

The airbreathing propulsion system (ABPS) provides thrust for powered approach to the landing field and for landing go-around in the event that the initial approach is aborted. Performance requirements and the sizing analysis presented in Section 3.3.5 constitute the basis for the system description presented here. Although the schematic shown as Fig. 3.8-7 includes tankage, etc. for ferry operations, the system weights presented in Table 3.8-6 are limited to fuel and components required for return from an orbital mission. ABPS characteristics are presented in Table 3.8-6.

This system is composed of 4 low-bypass-ratio turbofan engines supported by a fuel storage and feed system. Sizing analyses that have been conducted have established the landing go-around conditions as the limiting conditions. The design condition assumed for the baseline configuration includes a spacecraft weight of 240,000 lbm and L/D = 5.4 at a flight Mach number of 0.25 in the go-around configuration. Jet engine selection to meet the thrust requirements involved considerations of engine thrust-to-weight ratio, installation volume, engine specific fuel consumption, and system weight. The selection of the GE F101/F12B3 turbofan engine is based on the operational requirements, system design considerations, and the engine performance as presented in GE Report No. R71 AEG198 (Confidential document) "Space Shuttle ABE Study Data F101/F12B3 Turbofan Engine," (title unclassified), dated April 1971.

The engines are installed inside the vehicle between the outer hydrogen tank and the vehicle outer surface structure so that the exhaust gases discharge into the vehicle base area. (See Fig. 3.4-4.) The jet engine inlet is provided by swinging a portion of the vehicle outer skin structure inward which enables the freestream air to flow nearly directly into the engines. The selection of this installation arrangement was made after evaluating several alternates in

AIRBREATHING PROPULSION SYSTEM (JP-FUEL)

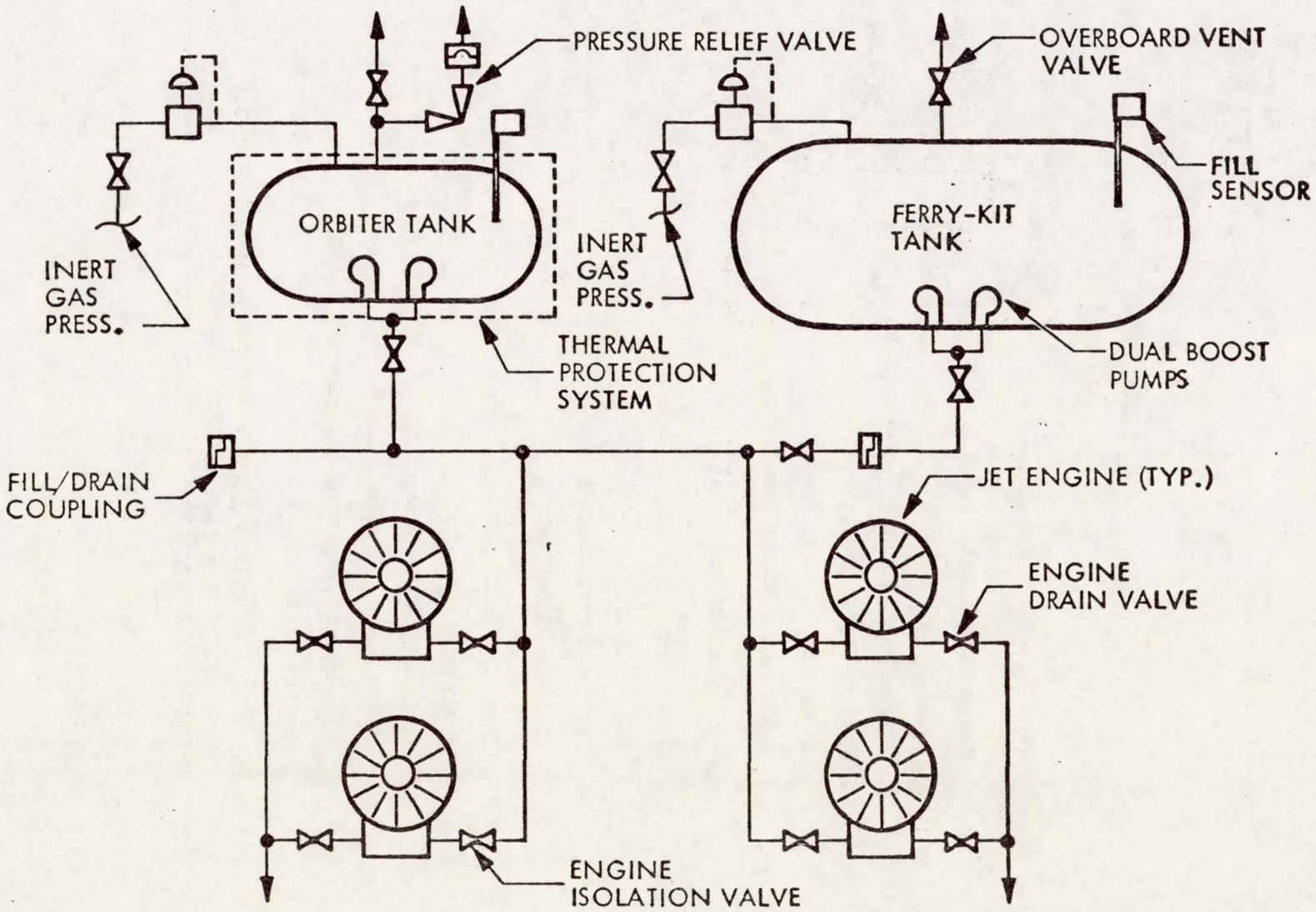

Fig. 3.8-7

D03236(1)

Table 3.8-6

ABPS

Requirements	
Gross Net Thrust, lbf	66,550
Altitude, ft	2,000
Flight Mach Number	0.25
Engine Start Altitude, ft	35,000
Engine Start Mach Number	0.55
Maximum Operating Time, min	19.3
Design Characteristics	
Number of Airbreathing Engines	4
Engine Type	GE-F101/F12B3
Engine Thrust Rating	Classified data – See GE Report R71AEG198
Engine SFC Rating	
Ducting Thrust Losses, %	6
Fuel Type	JP-4
Fuel Weight, Loaded, lbm	6,210
Fuel Weight, Impulse, lbm	5,400
Fuel Weight, Reserve (15 percent), lbm	810
Engine Installation	Internal – fixed
Engine Weight, each, lbm	2,665
Installation Weight (Engines, tanks, and plumb.), lbm	15,756

Table 3.8-6

conjunction with the internal location of propellant tankage and landing gear. Briefly, the structural and volumetric criteria for installation of the cryogenic propellant tanks were first satisfied to provide the lowest vehicle weight. The turbojet engines were then installed in the most usable remaining volume which is outboard of the outer hydrogen tanks near the vehicle base. Consideration was given to the use of retractable engine inlets extending into the airstream on top of the vehicle. Duct losses incurred by the top-mounted inlet would be greater than the selected side inlet and, more importantly, the duct extension above the vehicle surface would seriously reduce the vehicle lift component provided by the upper surface.

A propellant feed system schematic is shown in Fig. 3.8-7. The JP fuel storage tank is installed under the payload bay between the conical liquid oxygen tanks. A thermal protection system consisting of insulation and electric heating units is employed to prevent fuel freezing while in orbit. To satisfy fail-operational/fail-safe criteria, the inert gas pressurant supply is designed to satisfy pressure and flow requirements for feeding JP fuel to the jet engines in the event that both boost pumps fail. Engine starting is accomplished by windmilling the engine rotor after the inlet is opened with supplemental starting torque supplied by a hydraulic motor driving through the engine gear box. Hydraulic power for this motor is furnished by the vehicle APU system.

3.9 POWER AND AVIONICS

The two-stage orbiter and the stage-and-one-half vehicle electrical power and avionics systems are constrained by identical requirements, with relatively few but notable exceptions. The changing requirements between the two vehicle concepts do not reflect into a change in the fundamental system concepts developed in support of the stage-and-one-half vehicle. There are detailed design changes which have been identified, they are presented in Section 4.8 in the context of Conversion Requirements (Ref. EM L2-01-03-M1-5).

Stage-and-one-half Engineering Memorandums are being revised to reflect two-stage orbiter detail design configurations under in-house funding; revisions available during the contract span will be included in the final report.

No baseline avionics for the booster exists for this study, but a reasonable baseline can be extrapolated from the original stage-and-one-half avionics. The following discussion gives the details of this extrapolation. The referenced EMs were submitted in the Fifth Progress Report.

3.9.1 Electrical Power System (EM L2-01-06-M1-1A)

The major equipment difference between the stage-and-one-half and the two-stage booster is that there is no need for the fuel cells for long-time power requirements. The basic power system for the booster will be from AC generators, instead of the fuel cells. Emergency backup will be from batteries. Therefore, removing the fuel cells, dc controls and inverters reduces the weight by 850 lb, but the addition of the one AC generator rectifier increases the weight by 100 lb, giving a net reduction of approximately 750 lb for the electrical power system. Everything else will be the same, since the other avionics power requirements and peak demands are within the basic design capacity, including the main engines (11 or 12).

3.9.2 Guidance and Navigation (EM L2-01-03-M1-2A)

There is no need for the star sensor, horizon sensor, orbital radar altimeter, rendezvous radar, and docking sensor. All other G & N equipment will be the same. Removing these equipments reduces the weight by approximately 300 lb and the power by 120 watts.

3.9.3 Communication (EM L2-01-03-M1-3A)

The satellite communication subsystem will be removed and replaced with another UHF/VHF system. The booster will always be within range (208 miles) of a standard Tacan station and will never be more than 400 miles away from the launch base. An S-band link will be left onboard for use during launch/ascent and final approach on NASA bands. Therefore, removing the satellite equipment and adding UHF equipment reduces the weight by 50 lb and electrical power required is essentially the same.

3.9.4 Data Management System (EM LM-01-03-M1-1A)

The reduction of G & N sensors reduces the quantity of substation controllers by three. This is a weight reduction of approximately 15 lb and power reduction of approximately 10 watts. Other equipment will be considered essentially the same. Addition of 3 main engines over the 9 required for stage-and-one-half will require 9 SSCs for a net increase of 30 lb and 10 watts.

3.9.5 Control and Display (EM LM-01-03-M1-4)

Since the duration of the flight on the booster is approximately two hours, it is possible to reduce some of the operational functions needed for the orbiter. Also, most of the two hours is used in returning to the launch base in the atmosphere. Therefore, the following equipment can be reduced: One TV film reader, HUD, one HSD, and one film projector. This gives a weight reduction of approximately 100 lb and a power reduction of approximately 400 watts.

Considering that the booster is approximately 1-1/2 times the length and width of the orbiter, the wire and cable weight will increase. Taking this as approximately 700 to 900 lb additionally, there is a net weight and power savings in the booster of the two-stage over the stage-and-one-half of approximately 360 lb and 510 watts. These estimates are within 5 to 10 percent of the stage-and-one-half numbers, and the accuracy of estimating the original stage-and-one-half numbers for weight and power is no better than this percentage; therefore, we can say the booster avionics (weight and power) will be the same as the stage-and-one-half. This means that in the two-stage shuttle versions, the avionics equipment requirement is doubled in terms of weight and power.

SUMMARY

The operating principal of the booster avionics is essentially the same as the stage-and-one-half with minor differences. These differences give a net change in weight of -360 lb and a change in power of -520 watts.

3.10 MASS PROPERTIES

The two-stage system has essentially the same external aerodynamic shape as the stage-and-one-half system. Some differences exist in the aft end of the orbiters due to the differences in main engine philosophies. For the two-stage system, the requirement to package more propellant in the vehicle shifts the main engines aft slightly. To compensate for this, a fairing extention on the upper surface was required to shift the upper flap aft to provide for the effect on vehicle longitudinal c.g. Because the two vehicles are basically similar, particularly with regard to internal systems, this discussion is limited to those areas of dissimilarity. If a system is not treated in the following discussion, it may be assumed that it is the same as the stage-and-one-half system and may be found in section 2.14 of this report.

The design reference mission for the two-stage design becomes the South Polar Mission because of the abort mode. This will be the vehicle shown in detail. Since this is the designing case for the two-stage, the procedure was to scale a booster (Section 3.2.2) to satisfy this mission requirement and then use this fixed booster design to achieve maximum payload potentials for the Due East and 55 deg x 270 nm Missions. The two-stage orbiter and booster weights shown in this section are compatable with the NASA weight reporting format as well as current two-stage groundrules. Weight summaries for the due east and 55 deg/270 nm Reference Missions are also shown. A 4,123,341 lb GLOW was achieved using the 134.7 ft delta-body orbiter and a scaled MDAC booster as shown in MDAC 1 March 1971 MP8 Report. The weight of 831,092 lb reflects a 40K payload delivered to south polar orbit without an airbreathing propulsion system (ABPS). The orbiter gross weight of 857,230 lb reflects a 65K payload without ABPS for the due east mission. It is designed to land with 40K payload. The resupply mission (55 deg/270 nm) orbiter weight of 832,398 lb reflects a 25K payload delivery and return capability with ABPS in.

The orbiter dry weight for the south polar, due east, and 55 deg 270 nm missions are 187,929 lb, 187,929 lb and 206,130, respectively. These weights include a 10 percent growth and contingency factor applied to all dry weight less 19,164 lb of main engine weight.

3.10.1 Body Group

The two-stage fuselage shell and frames were modeled on two separate computer programs rather than one as the stage-and-one-half system was (Section 2.14). This was required because of the orbiter-to-booster interface loads in the aft body section. The loads here led to a redundant structural analysis, since the aft body shell and main ascent LH_2 tanks share the loads by the ratio's of their relative stiffness parameters. (For a more rigorous discussion of this, see Section 3.7 of this report.)

The section forward of Station 1272 was modeled by the "Fast" computer program described in Section 2.14, while the body shell, frames, and two primary bulkheads aft of fuselage station 1270 were modeled by a finite-element program SNAP/FSD described in Section 3.7. The forebody section, like the section described in Section 2.14, carries an additional N.O.F. of 25 percent over the theoretical weights. The aft body section, however, because of the loading uncertainies has a N.O.F. of 50 percent applied.

The remaining internal secondary structural item weights were determined by the same methods as those described in Section 2.14.

3.10.2 Induced Environmental Proection

Differences, although slight in terms of total weight, exist between the stage-and-one-half and the two-stage. The stage-and-one-half system, because of its higher wing loading, requires higher LI-1500 thicknesses along the body

but does not require insulation on the lower flap upper surface as the flap is retracted during ascent. For these reasons, the total differences in weight are only 785 lb with the two-stage being the lighter system.

3.10.3 Main Ascent Propulsion

The two-stage design weights are based upon two ICD 13M15000B engines, gimballed, with an expansion ratio of 150 to 1. Feed and drain system weights were calculated from preliminary layouts and schematics. Tank membrane weights were obtained by the same methods as described in Section 2.14 with stiffening elements added to the LH_2 tank. These elements were provided by structural analysis of the section from basic loads.

3.10.4 Propulsion-Cruiseback

The cruise propulsion system weights are based upon four fixed G. E. F101/F12B3 engines in the base of the vehicle. Each engine weighs 2835 lb with accessories. Provisions were made for a ducting system with movable doors or inlets in the sides of the vehicle. In addition, a nacelle package for each engine was estimated.

3.10.5 Systems

Slight differences exist in system weights between the two-stage and the stage-and-one-half for the following reasons.

Prime Power - Two-stage is lighter because main engine requirements are less than for stage-and-one-half.

Surface Controls - Two-stage is lighter than stage-and-one-half since no actuation of ABPS is required.

All other systems are the same as, or derived in the same manner as, the stage-and-one-half system.

A comparative weights summary for the three reference missions is shown in Table 3.10-1. The basic design mission (south polar) is detailed in Tables 3.10-2 through 3.10-11, respectively. The due east mission data will be found in Tables 3.10-12 and 3.10-13, and the final mission, the 55 deg x 270 nm is detailed in Tables 3.10-14 and 3.10-15.

Table 3.10-1

Two-Stage Weight Summary

Item	South Polar		Due East		55 deg/270 nm	
	Booster	Orbiter	Booster	Orbiter	Booster	Orbiter
GLOW	4,123,341		4,156,868		4,130,424	
Propellant-Ascent	2,643,026	546,439	2,643,026	545,852	2,643,026	546,411
Pre-Separation	649,222	831,011	642,102	871,740	644,102	843,296
Orbiter Injection		272,616				273,919
ABPS Fuel	90,562	0	82,562	0	85,562	5,400
Maneuvers/ACS	1,222	37,848	1,222	37,723	1,222	29,513
Other, (Reserve, Residual, Loss)		18,152		20,001		19,219
Personnel	400	725	400	725	400	725
Cargo		40,000		79,510		35,898
Dry Weight	476,766	187,929	476,766	187,929	476,766	206,130
Main Engine Weight		19,164		19,164		19,164
Growth and Contingency	30,588	15,342	30,588	15,432	30,588	16,997
Payload						
Required		40,000		65,000		25,000
Potential		40,000		79,510		35,898

LOCKHEED

Table 3.10-2

SOUTH POLAR	PERFORMANCE DATA NUMBER				
CONFIGURATION	TWO-STAGE SYSTEM	By	P.W.	Date	4 May 71

ITEM	UNITS	ORBITER	BOOSTER
WEIGHTS DATA:			
Burnout Weight	10^3 Lb	233.5 (1)	649.2 (2)
Nominal Propellant Load	10^6 Lb	.590	2.643
Payload	10^3 Lb	40	831.2
Gross Weight	10^3 Lb	.831	3.292
VELOCITY DATA:			
Nominal Ascent Velocity	Ft/sec	17584 (3)	14430 (3)
Flight Performance Reserve	Ft/sec	317	0
Total Ascent Velocity	Ft/sec	32331 (3)	--
Ascent Specific Impulse	Sec	456 (4)	438.1 (4)
On Orbit Maneuver Velocity	Ft/sec	170 (5)	--
Ascent Thrust/Weight (initial)	-	.829 (3)	1.467
On Orbit Maneuver I_{sp}	Sec	439 (5)	--
PAYLOAD SENSITIVITY DATA *			
Specific Impulse	Lb/sec	769	632
Inert Weight	Lb/lb	-1.0	-.170
Propellant Load	Lb/lb	.185	.0472
On Orbit Maneuver Velocity	Lb/ft/sec	-19.7	
Ascent Velocity	Lb/fps	-19.6	
Thrust/Weight - (initial)	Lb/(0.1 T/W)	4540 (6)	4990
Orbit Inclination	Lb/deg	-510	
Launch Site Altitude	Lb/ft	1.26	
Gross Weight	Lb/lb	.024	
INJECTION ORBIT CHARACTERISTICS			
Apogee	nm	50	
Perigee	nm	100	
Inclination	Deg	90	
Launch Site Latitude	Deg	34.6	
Launch Site Altitude	Ft above SL	0	

NOTE: Sensitivities are estimates for polar abort mission with 550K engines

(1) Post Retro
(2) Post Staging
(3) Abort
(4) 3σ Performance
(5) Retro ΔV = 150 fps (Isp = 430), ACPS ΔV = 20 fps (Isp = 352)
(6) One engine operating (abort)

Table 3.10-3

SOUTH POLAR GENERAL DATA SUMMARY					
CONFIGURATION	2-Stage Orbiter	BY		DATE	4 May 71
ITEM	LS 400-7A	ORBITER			TOTAL
GEOMETRIC DATA					
Length (Base to Nose) - Ft		134.7			
Wing Span - Ft					
Wing Area (Theoretical) - Sq Ft					
Wing Area (Exposed) - Sq Ft					
Vehicle Planform Area - Sq Ft					
Body Wetted Area - Sq Ft		13721			
Vehicle Wetted Area - Sq Ft		19070			
Body Outer Mold Line Volume - Cu Ft					
Vehicle Outer Mold Line Volume - Cu Ft		99950			
Ascent Propellant Tank Volume - Cu Ft		10923			
" " Line Volume-Cu Ft		1033			
AERODYANMIC DATA					
Entry Angle-of-Attack - Deg		32			
Hypersonic L/D Max. (Trimmed)		1.87			
Angle-of-Attack (Subsonic L/D Max) - Deg		15			
Subsonic L/D Max (Trimmed)	(1) 5.10 to (2) 5.85				
Cruise L/D (Average)	(1) 5.10 to (2) 5.85				
Cruise Range (No Wind) - nm					
cg Limits Fwd/Aft - % L_{REF}	(1) 72.5 to (1) 78				
Landing Speed - knots					
@ L/D Max = 15°	(2) 154 to (1) 180				
@ Tailscrape = 22°	(2) 122 to (1) 152				
PROPULSION DATA					
Ascent Engine Thrust - lb					
Sea level					
Vacuum		632K			
Ascent Engine Expansion Ratio		150:1			
Retracted					
Extended					
Fixed					
Number of Ascent Engines					
Cruise Engine S.L. Thrust - lb		N/A			
Number of Cruise Engines		4			
Cruise Fuel Type		JP-4			
REFERENCE DRAWING NUMBER					

(1) 40 Klb Cargo IN/Airbreather Engines OUT
(2) Cargo IN/Airbreather Engines OUT

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Table 3.10-4

SOUTH POLAR	DESIGN DATA SUMMARY		PAGE 1 of 8
CONFIGURATION	TWO-STAGE ORBITER	BY	DATE 4 May 1971

MAX DYNAMIC PRESSURE, LB/SQ.FT _____
MAX q α PSF DEGREE 2800
ENTRY VELOCITY FT/SEC _____
BALLISTIC COEFFICIENT $W/C_d A$, LB/SQ.FT _____
ENTRY ANGLE DEGREES _____

ULTIMATE LOAD FACTOR	Nx	Ny	Nz	@ WEIGHT, LB
Ascent	4.2	± 1.6	± 1.6	
Abort				
Separation	4.2			
Entry				
Cruise				
Landing			4.0	

1. WING GROUP N/A

	INSIDE FUSELAGE	EXPOSED	
Gross Area SQ FT			
Torque Box			
Leading Edge (Fixed)			
Trailing Edge (Fixed)			
Movable Surfaces			
Volume - CU.FT			

	MAC	ROOT THEORETICAL	BODY JUNCTION	PLANFORM BREAK	TIP
CHORD LENGTH (FT.)					
CHORD THICKNESS (FT.)					

SPAN BETWEEN CHORDS (FT)
DIHEDRAL ANGLE (DEG) _____ SPAN BETWEEN DIHEDRAL BASES FT _____
SWEEP BACK ANGLE (MEAN CHORD) _____ CHORD _____
AIRFOIL DESCRIPTION _____
TAPER RATIO _____
THICKNESS/CHORD: ROOT _____ TIP _____
DESIGN LOAD _____
CRITICAL LOAD CONDITION _____

CONTROL SURFACES	TYPE	AREA — SQ FT RETRACT	EXTEND
L. E. Flaps			
Spoilers			
Speed Brakes			

PRIMARY STRUCTURAL MATERIALS	MATERIAL	DESIGN TEMP °F
Torque Box		
Leading Edge		
Trailing Edge (Fixed)		
Movable Surfaces		

Table 3.10-4 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE 2 of 8
CONFIGURATION TWO-STAGE ORBITER		BY		DATE 4 May 1971

2. TAIL GROUP	Speed Brake	Fin Rudder	Flap Upr HORIZONTAL	Flap Lwr	
EXPOSED AREA FT^2 TOTAL	(225)	(1248)	(693)	(1086)	3252
Torque Box		874			( )
Leading Edge (Fixed)					( )
Trailing Edge (Fixed)					( )
Movable Surfaces	225	374	693	1086	(2378)
CARRY THROUGH AREA - FT^2					
EXPOSED SEMI-SPAN-FT^2					
CARRY THROUGH SPAN-FT					
NO. OF SURFACES/VEHICLE	2	2	2	1	
VOLUME – CU FT	*	4330		1330	
PRIMARY STRUCT.MATERIAL	Ti	Ti	Ti	Ti	
CHORD LENGTH					
Root (Theoretical) Ft.					–
Mac Ft					
Body, Junction					
Tip, Ft					
CHORD MAX. THICKNESS					
Root Ft					
Tip, Ft					
DIHEDRAL ANGLE, DEGREE					
SWEEPBACK, 25% CHORD					
CRITICAL LOAD CONDITION					

3. BODY GROUP	FWD BULKHEAD	BARREL	AFT BULKHEAD	COMMON BULKHEAD	
INTEGRAL TANK WETTED AREA - FT^2					
Oxidizer Tank					( )
Fuel Tank					( )
Inter-Tank Structure					( )
ULLAGE PRESSURE – PSI	OXIDIZER		FUEL		
BASIC STRUCTURE WETTED AREA – FT^2	FWD (1187)	CTR (5725)	AFT (5542)	SKIRT (815)	13269
Sidwalls					
Bulkheads					
Partitions					
Thrust Structure (Main Ascent Engine)					( )
PRIMARY STRUCTURAL MATERIAL	Al	Al/Ti	Al	Al	

* Incl in Fin/Rudder

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Table 3.10-4 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY		PAGE 3 of 8
CONFIGURATION	TWO-STAGE ORBITER LS 400-7A	BY	DATE 4 May 1971

3. BODY GROUP (Continued)

MISCELLANEOUS	WETTED AREA-SQ.FT.	VOLUME CU.FT.	ULT DES.PRESS. DIFF. – PSI
Crew Compartment	1026	645	
Equipment Compartment			
Radome Antennas			
Speed Brakes			
Doors Payload	1300	N/A	
Tanks - Oxidizer			
Tanks - Fuel			

4. INDUCED ENVIRONMENT PROTECTION

		LWR HOR. FLAP	VERT.TAIL	BODY	
TOTAL VEH.WETTED AREA–FT^2		(2190)	(1996)	(203)	19070
LEADING NOSE CAP AREA – SQ FT	( )			(203)	203
Material Ti/Ta/Dynaflex				203	
Material					
Material					
SURFACE PROT. AREA-SQ.FT.	( )	(1086)	(964)	(11468)	13518
Material LI-1500		1086	964	7801	
Material Ti/Dynaflex				3667	
Material					
Material					
Material					
Material					
Material					
Material					
UNPROTECTED AREA–SQ FT	( )	1104	1032	1993*	4129
BASE: MATERIAL LI-1500/RSF**				1220	1220

*Incl. Area under Upper Flap

TOTAL VEHICLE VOLUME – CU. FT. (OUTER MOLDLINE)					99,950
VOLUME INSIDE PRIMARY Structure – Cu Ft					
TPS VOLUME CU FT					
LEAD. EDGE/NOSE CAP RAD.–FT				4 ft Hemisphere	
LEAD. EDGE/NOSE CAP MAX TEMP ^oF					
LOWER SURFACE MAX TEMP ^oF					
SIDE SURFACE MAX TEMP ^oF					
UPPER SURFACE MAX TEMP ^oF					

**Reinforced silicone elastomer.

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Table 3.10-4 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE 4 of 8
CONFIGURATION	TWO-STAGE ORBITER LS 400-7A	BY		DATE 4 May 1971

5. LANDING, RECOVERY, DOCKING

ALIGHTING GEAR	NO./VEH	DESIGN LOAD - LB	EXTENDED STRUT LGTH - IN	STROKE IN	PRIMARY MATL	BRAKE MATL
Main Gear	2	363,800	178	18	Stl	Be
Nose Gear	1	40.800	178	20	Stl	Be

Max. Design Landing Wt - Lb = 238000
Landing Speed - Knots = See Table 3.10-3
Angle of Attack @ Landing - Deg = _____

Limit Landing Sink Speed
10 Ft/Sec @ 238000 LB. Landing Weight
10 Ft/Sec @ 238000 LB. Max Design Landing Weight

SEPARATION SYSTEM
Design "q" @ Separation - psf = 5.25
Max. Axial Acceleration - g's = 3 gs
Max. Design Separation - Wt. Lb. = _____

DECELERATION CHUTE
Diameter - Ft = _____
No./Vehicle = _____

6. PROPULSION - MAIN ASCENT

	THRUST-SL	THRUST-VAC	EXP RATION	I_{sp} VAC	CHAMB PRES PSI
ENGINE	N/A	632K	150:1	459 ± 3	3000

PROPELLANT SYSTEM	FUEL	OXIDIZER	(OPERATING LIMIT)
Ullage Pres. - psi Burnout	26	28	
Propel Type	LH_2	LO_2	
Ullage (%)	3	3	( )
Pressurant	GH_2	GO_2	
Total Tank Vol - Ft	19315	6928	
TOTAL PROP VOL - FT^3*	19577	7030	
Total Len of Feedlines VOL			

TANKAGE - NONINTEGRAL	NO.	SHAPE	WET AREA SQ FT	VOLUME CU FT
Oxidizer	2	Frust.	1553 ea	3490 ea
Oxidizer				
Oxidizer				
Fuel	2	Cyl	2272 ea	8345 ea
Fuel	1	Cyl	969	2626
Fuel				

*Incl. Lines Vol.

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Table 3.10-4 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE 5 of 8
CONFIGURATION	Two-Stage Orbiter LS 400-7A	BY	Broadhead	DATE 4 May 71

7. PROPULSION – CRUISEBACK

No. of Engines	4
Engine Thrust – S.L. Static – Lb	N/A
Specific Fuel Consumption Lb/Lb – Thrust Per Hr @ Nominal Cruise Altitude – Ft	
Nominal Cruise Altitude – Ft	
Nominal Cruise Speed – Knots	
Cruise Altitude Engine Out – Ft	
Cruise Speed, Engine Out – Knots	
Cruise Range (Actual Req) – NAMI	
Cruise Range (Max Avail – No Headwind, All Engines Up)	
Cruise Lift Drag Ratio	
Engine Thrust Sized By	
Lift Coefficient for Critical Thrust Condition	

	TYPE	TANK VOL CU FT	TANK MATL	TANK PRES-psi	BURST FACTOR	NO. OF TANKS
FUEL SYSTEM	JP-4	135	AL			1
PRESSURIZATION SYSTEM						

AIR INTAKE – LENGTH–Ft
– DIAMETER–Ft
NACELLE WETTED AREA – EACH– SqFt

8. PROPULSION – AUXILIARY

THRUSTERS

Thrust (Vac) – Lb	I_{sp} – Sec	ACS	QUANTITY REQ MANEUVER	COMBIN	( )
1500	352	40			( )
RL-10 (15000)	439		2		( )
					( )
					( )
					( )

PROPELLANT SYSTEM	TYPE	TANK** VOLUME	TANK MATL	TANK PRES–psi	BURST FACTOR	NO. OF TANKS
Fuel	LH_2	471	AL	27	2.0	1
Oxidizer	LO_2	1829	AL	50	1.78	1
Fuel Pressurant	GH_2					
Oxidizer Press.	GO_2					

**Net Usable Plus Ullage

9. PRIME POWER

	SPECIFIC POWER		TOTAL POWER		TYPE
Batteries	60	Watt-Hr/Lb	12000	Watt-Hrs	Ag-ZN
Engine/Turbine		HP-Hr/Lb Fuel	120K	HP – Hr	
	2.0	HP/Lb of Engine (	800	HP	
Fuel Cell	1235	Watt-Hrs/Lb Fuel	864	Watt-Hrs	H_2O_2
	30	Watts/Lb of Fuel Cell	16K	Watts	

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Table 3.10-4 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE	6 of 8
CONFIGURATION	Two-Stage Orbiter LS 400-7A	BY	Broadhead	DATE	4 May 71

10. ELECTRICAL POWER CONVERSION/DISTRIBUTION

System Voltage	= 28	VOLTS	
Peak Power	= 6900	WATTS	
Average Power	= 4800	WATTS	

11. HYDRAULIC POWER CONVERSION/DISTRIBUTION

System Nominal Oper Pressure	= 4000	PSI	
Peak Power	= 344	HORSEPOWER	
Average Power	= 115	HORSEPOWER	
Total Volume of Fluid	=	FT^3	

Fluid Type MIL-H-5606 Max Oper Temp - °F

12. SURFACE CONTROLS

SURFACE	AREA FT^2	MAX DEFL RATE DEG/SEC	MAX DEFL DEG	DESIGN HINGE MOM. FT/LB	NO/ VEHICLE
UPR FLAP	693	15	-40	628K	2
RUDDER	374	15	+10-30±15	101K	2
AUX SURF	225	2.5	-20 +40	334K	2
FLAP LWR	543	1	+10 -25	811K	1
ELEVON LWR	543	15	± 20	186K	2

13. AVIONICS

14. ENVIRONMENTAL CONTROL

	TOT. STOR VOL- FT^3	STORAGE PRES.	TANK MATL	NO. OF TANKS
Gas Supply System	10			
Primary Oxygen & Cooling H_2				
Second Oxygen (Super Critical)		900 PSI	STL.	1
Diluent N_2 (Super Critical)		900 PSI	Ti	1

Gas Requirement Average Rates

Metabolic	= 11.6	Lb Man-Day
Leakage	= 2.0	Lb Day
Repressurize	= H_2=29.4; O_2 = 8.9	Lb Repressurize (CABIN)
Repressurize	=	Lb/Repressurize (AIRLOCK)

Heat Transport System Capacity = 35,000 Btu Hr (PEAK) Radiator
= 10,000 Btu Hr (PEAK) Total System

Radiator Area = 640 Sq Ft

Water Management System Capacity

Drinking Water	= 6*	Lb Man-Day	14	Man Days
Washing	= 2*	Lb Man-Day	14	Man Days
Cooling	= 5*	Lb Btu		BTU's

* Supplied by Fuel Cell

15. PERSONNEL PROVISIONS

16. RANGE SAFETY AND ABORT

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Table 3.10-4 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE 7 of 8
CONFIGURATION	2-Stage Orbiter LS 400-7A	BY	Broadhead	DATE

17. BALLAST

Design C.G. FWD = 72.5 %L AFT = 78 %L
Nominal C.G. = %L ENTRY
Nominal C.G. with 25K Lb = 75.4 %L ENTRY
Max. Design Landing Wt - Lb @ % MAC. C.G.

* L = 1752 IN.

18. GROWTH/UNCERTAINTY

Current Allowance = 15,342 LB
Contractors Est of Allowance Needed to guarantee
Current Payload and Gross Wt = LB*
Remaining Growth Allowance for
Customer Changes = LB

19. OPEN

20. PERSONNEL

No. of Crew = 2 ; ave percentile man = 37*
No. of Personnel = 0 ; ave percentile man =
TOTAL

*Ref SAWE HDBK

21. CARGO

CARGO BAY VOLUME = 10,770 Cu Ft

Bay Dia = 15 FT
Bay Lgth = 60 FT

22. ORDNANCE

23. RESIDUAL FLUIDS - DEFINE WEIGHT ESTIMATING RATIONALE

TANKS/LINES (Asct/ Orbit) - Maintain Tank Pressure
ACPS - Accumulator Gasses
Service - Cabin Radiator Freon plus Coolants
Hydraulic Fluids and Fuel Cell Residual

25 RESERVE

Ascent - 1 percent ΔV ideal
Maneuver - Incl. in on-orbit ΔV Req
ACPS - 10 percent Impulse
Service
ECS - 1 day supply
EPS - 10 percent on APU and 20 percent on fuel cell Prop. Wt.

Table 3.10-4 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE 8 of 8
CONFIGURATION	Two-Stage Orbiter LS 400-7A	BY	Broadhead	DATE 4 May 71

27 - 29 PROPELLANTS EXPENDED

	ASCENT	CRUISE	MANEUVER	ATTITUDE
Oxidizer/Fuel Ratio (By Weight)	6	N/A	5; 3.52	3.52
Oxidizer Ullage Volume - Percent*	3	N/A	N/A	N/A
Fuel Ullage Volume - Percent*	3			
Fuel Density - pcf	4.274	49	4.274	4.274
Oxidizer Density - pcf	70.2		70.2	70.2
Fuel Bias - Percent				
Incremental Velocity -fps				
Inertial				
Maneuver Losses				
Gravity Losses				
Drag Losses				
Back Pressure Losses				
Engine Cant				
Earth Rotation				
FLT. Performance Reserve	6000			
		( )		

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Table 3.10-5

SOUTH POLAR	GROUP WEIGHT STATEMENT			Page 1 of 6
CONFIGURATION	Two-Stage Orbiter LS 400-7A	Broadhead	DATE	4 May 71

1. WING GROUP					N/A
		CARRY THROUGH	EXPOSED SURFACE		
Basic Structure					
Torque Box					
Leading Edge					
Trailing Edge					
Secondary Structures					
Variable Geometry (incl ____ lbs mechanism)					
Doors Insulation Fairings					
Control Surfaces		SURFACE	SUPT/MECH		
Elevon — (incl. bal wt. ____ lb)					
T.E. Flaps					
L.E. Flaps					
Spoilers					
Speed Brakes					
2. TAIL GROUP		VERTICAL	HORIZ		16264
Basic Structure		5242		5242	
Torque Box					
Carry Through					
Leading Edge					
Trailing Edge					
Secondary Structure					
Control Surfaces		SURFACE	SUPT	11022	
Rudder (incl. bal wt 0 lb)		2246			
Body Flap – UPR		2772			
ELEVON		5104			
AUX. CONT. SURFACE		900			
3. BODY GROUP					42598
Integral Tankage					
Fuel Tank					
Oxidizer Tank					
Between tanks (cmn blkhd)					
Insulation					
Basic Structure	FWD.	CTR.	AFT.	SKIRT	
	1746	16963	10390	8254	37353
Sidewalls	1146	12125	8880	1718	
Bulkheads	600	917	1510	2115	
Partitions		3921			
Thrust Structure (main ascent engine)				4421	
Secondary Structure					5245
Crew Compartment/AIRLOCK/ACCESS				2620	
Equipment Compartments				710	
Payload Attach & Deploy				1000	
Speed Brakes					
Engine Heat Protection				915	
Interstage (incl. mech. ____ lbs)					
Doors/fairings				INCL.	
Gear/wing provisions/ABES					
Contingency					

Table 3.10-5 (Cont'd)

SOUTH POLAR	GROUP WEIGHT STATEMENT			Page 2 of 6
CONFIGURATION	Two-Stage Orbiter LS 400-74	By	Broadhead	Date 4 May 71

	WING	HOR.TAIL	VER. TAIL	BODY		
4. INDUCED ENVIRONMENTAL PROTECTION						34931
Thermal Protection					6411	
Leading Edge/Nose Cap*		(3269)	(1995)	(1147)		
Ta/Ti/Dynaflex				1147		
Surface Protection*					26856	
Body Lwr (LI-1500)				13558		
Body L/E (LI-1500)				7743		
Body Upr T>1000° (LI-1500)				2389		
Body Upr T<1000° (Ti/Dynaflex)				3166		
FIN L/E (LI-1500)			377			
FIN Side (LI-1500)			1618			
Lwr Trim Surf (LI-1500)		3269				
Base Heat Protection*					1664	
LI-1500 (Incl. Flame Curtain)				1664		
Sound Protection						
Meteorite / Radiation Protection						
Contingency						

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Table 3.10-5 (Cont'd)

SOUTH POLAR GROUP WEIGHT STATEMENT						Page 3 of 6	
CONFIGURATION	Two-Stage Orbiter LS400-7A			By Broadhead		Date	4 May 71
5. LANDING, DOCKING							8806
Alighting Gear		* ROLLING	STRUC-TURE	CON-TROLLS		8806	
Main		3970	3178	630			
Nose		197	605	226			
Docking	* Incl. Brakes						
Auxiliary Systems							
Deceleration chutes				N/A			
Flotation gear							
Handling gear							
Contingency							
6. PROPULSION – MAIN ASCENT							46060
Engine & Accessories						20149	
Engine (as supplied)				17620			
Gimbal System				1544			
Ignition and Control System							
Propellant Utilization System				300			
Accessories/Misc Systems				685			
Installation, Ducts, Shrouds							
Propellant System			LO_2	LH_2		4349	
			(2697)	(1652)			
Pressurization							
Fill & Vent Lines			1143	247			
System			759	556			
Valves			520	699			
Feed Systems							
Vortex, Flow Control System							
Supports and Install			275	150			
Tankage – Nonintegral	Tank	Insul.	Supports			21562	
Fuel	7028	603	4253				
Oxidizer	4716	0	4962				
Contingency							
7. PROPULSION – CRUISE BACK							350
Engine & Accessories							
Engine							
Ignition and Control Sy							
Lubrication Sy (dry)							
Accessories							
Installation, Ducts, Shroud						350	
Air Induction							
Engine Mounting							
Nacelles Pylons (incl ____ lb mech)							
Exhaust System							
Propellant System							
Fill Drain							
Pressurization (dry)							
Vent System							
Pump							
Feed System							
Transfer System							
Dump System							
Supports/Installation							
Tankage – Non Integral							
Fuel							
Contingency							

Table 3.10-5 (Cont'd)

SOUTH POLAR GROUP WEIGHT STATEMENT						Page 4 of 6
CONFIGURATION	Two-Stage Orbiter LS 400-7A Broadhead					Date 4 May 71
	LS 400-7A		ATT. Cont	Maneu-ver		6970
			(4793)	(2177)		
8. PROPULSION – AUXILIARY			1160	700		
Thruster Installation						
Thruster	1160			700		
Accessory						
Propellant System			2978		485	
Fill/Drain/Vent Lines & Valves	80			147		
Pressurization						
Heat Exchanger	265					
Feed System/Accumulators	2458			275		
Conditioning						
Supports	175			63		
Tankage	LO_2		655	LH_2	992	
Tanks	296			660		
Insulation	104			164		
Supports	255			168		
						1541
9. PRIME POWER	POWER UNIT	MTG INSTAL	PROPEL TANK/SYS	TEMP CONTROL	CON-TROLS	TOTAL
Batteries	198	36				234
Engine Turbine	368	80			24	472
Fuel Cells	480	85	270	Incl.		835
Contingency						
						3747
10. ELECTRICAL		SUPPLY	CONVER-SION	CONTROL UNITS		
Equipment		120	192	600		912
Distribution and Control Circuitry						2270
Utility Systems						565
Supports/Installation						
Contingency						
						2073
11. HYDRAULIC		PRIMARY (2073)				
Power Supply		144				
Distribution Control Ctr		1725				
Temperature Control Sy		16				
Auxiliary Systems						
Supports/Installation		188				
Contingency						
						4085
12. SURFACE CONTROLS						
Cockpit Controls						120
Flight Control System						
System Actuation	CONTROLS	POWER XMISSION	ACTU-ATOR	FEEL SY	SUPTS INSTALL	
		(350)	(3240)		(375)	3965
ELEVON UPR			720			
ELEVON LWR		350	1600			
Rudder/AUX SURFACE			870			
DUCT DOOR			50			
Speed Brake						
Contingency						
						3678
13. AVIONICS	UNITS (2732)	CIRCUITRY (321)	COOLING	ANTENNAS (58)	INSTALL. (567)	
Guid/Nav	174	35			35	
Flight Control	909	91			180	
Data Management	1102	118			235	
Communicate	107	33		58	30	
Config. Seq.	440	44		INCL.	87	
Instrumentation	INCL.			INCL.		
Displays	INCL.					
Contingency						

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Table 3.10-5 (Cont'd)

SOUTH POLAR GROUP WEIGHT STATEMENT							Page 5 of 6	
CONFIGURATION	Two-Stage Orbiter LS400-7A				By	Broadhead	Date	4 May 71
14. ENVIRONMENTAL CONTROL								
Gas Supply System (dry)							62	
Gas Management, Processing (dry)							177	
Heat Transport System (dry)							996	
Water Management System (dry)							39	
Purge System							N/A	
Insulation								
Contingency								
15. PERSONNEL PROVISIONS								210
Seats / Restraint Sys (No =)							84	
Fixed Life Support Equipment							112	
Emergency Equipment								
Cargo Handling							14	
Furnishings							INCL.	
16. RANGE SAFETY AND ABORT								0
17. BALLAST								0
18. GROWTH / UNCERTAINTY								15342
19. OPEN								
SUBTOTAL (Dry Weight)								187929
20. PERSONNEL	NO	MAN	GARMENTS	HELMET	PRESS. SUIT	ACCESSORIES		725
Crew	(2)	330	6	14	70	22		
Passenger	( )							
Personal Gear/Accessories							126	
Life Support							157	
Food / MEDICAL						102		
Equipment - Portable						55		
21. CARGO								40000
22. ORDNANCE								
23. RESIDUAL & UNUSABLE FLUIDS								3191
Ascent							786	
Cruise							0	
Maneuver (Incl. All Aux. Tk. Residuals)							137	
Attitude Control							94	
ECS							451	
EPS							35	
Hydraulic							1688	
Misc (Shock struts, etc.)								
24. OPEN								
SUBTOTAL (Inert Weight)								231845

Table 3.10-5 (Cont'd)

SOUTH POLAR	GROUP WEIGHT STATEMENT			Page 6 of 6
CONFIGURATION	Two-Stage Orbiter LS400-7A	By	Broadhead	Date 4 May71

25. RESERVE FLUIDS			6640
Ascent	6000		
Cruise	0		
Maneuver	0		
Attitude Control	432		
ECS	10		
EPS	123		
APU	75		
26. INFLIGHT LOSSES			8321
Ascent	6444		
Cruise	0		
Maneuver (Incl. All Aux. Tk Boiloff)	1092		
Attitude Control	INCL.		
ECS	145		
EPS	640		
27. PROPELLANT - ASCENT			546,439
28. PROPELLANT - CRUISE			
	MANEUV.	ACS	
29. PROPELLANT - MANEUVER/ACS	33531	4317	37848
TOTAL (Gross Weight)			831093

SOUTH POLAR **MISSION WEIGHT SUMMARY**

CONFIGURATION: Two-Stage Orbiter LS 400-7A | BY: | DATE: 4 May 71

CODE	SYSTEM	A	B	C	D	E	F	G	H
1	WING GROUP	N/A							
2	TAIL GROUP	16264							
3	BODY GROUP	42598							
4	INDUCED ENVIR PROTECTION	34931							
5	LANDING, RECOVERY, DOCKING	8806							
6	PROPULSION - ASCENT	46060							
7	PROPULSION - CRUISE	350							
8	PROPULSION - AUXILIARY	6970							
9	PRIME POWER	1541							
10	ELECT CONVER & DISTR	3747							
11	HYDRA CONVER & DISTR	2073							
12	SURFACE CONTROLS	4085							
13	AVIONICS	3678							
14	ENVIRONMENTAL CONTROL	1274							
15	PERSONNEL PROVISIONS	210							
16	RANGE SAFETY & ABORT								
17	BALLAST								
18	GROWTH/UNCERTAINTY	15342							
19									
	SUBTOTAL (DRY WEIGHT)	187929							
20	PERSONNEL	725							
21	CARGO	40000							
22	ORDNANCE								
23	RESIDUAL FLUIDS	3191							
24									
	SUBTOTAL(INERT WEIGHT)	231845	231845	231845	231845	231845	231845	231845	231845
25	RESERVE FLUIDS	6640	6640	640	640	640	640	640	640
26	IN FLIGHT LOSSES	8321	8239	8239	2283	2283	615	615	
27	PROPELLANT - ASCENT	546439	546439	271936	0	0	0	0	0
28	PROPELLANT - CRUISE	0	0	0	0	0	0	0	0
29	PROPELLANT— MANEUV/ACS	37848	37848	37848	37848	9682	8916	465	0
30									
	TOTAL(GROSS-WEIGHT)LB.	831093	831011	550508	272616	244450	242016	233565	232485

DESIGNATORS:

EVENTS

A LAUNCH WEIGHT
B IGNITION (EFFECTIVE)
C 50% BURN
D INJECTION
E ON-ORBIT
F PRE-RETRO
G ENTRY
H LANDING

NOTES & SKETCHES:

SYSTEM MASS PROPERTIES Table 3.10-7

CONFIGURATION SOUTH POLAR LS 400-7A Two-Stage Orbiter

DATE: 4 May 1971

NO.	SYSTEM	WEIGHT (LB)	CENTER OF GRAVITY (IN)			MOMENT OF INERTIA MILLION SLUG FT^2/1000		
			X	Y	Z	ROLL	PITCH	YAW
A	LAUNCH WEIGHT	831,093	1,177	-2	286	4,794	19,101	22,640
B	IGNITION (EFFECTIVE)	831,011	1,177	-2	286	4,794	19,100	22,640
C	50% BURN	550,508	1,205	-3	293	3,353	14,282	16,626
D	INJECTION	272,616	1,320	-5	312	1,941	10,724	11,888
E	ON-ORBIT-STATION DOCKING	244,450	1,292	-2	318	1,885	10,261	11,450
F	PRE-RETRO	242,016	1,291	-1	318	1,882	10,238	11,427
G	ENTRY	233,565	1,272	0	320	1,864	10,018	11,215
H	LANDING	232,485	1,277	0	320	1,972	9,461	10,404

X = ROLL AXIS
Y = PITCH AXIS
Z = YAW AXIS

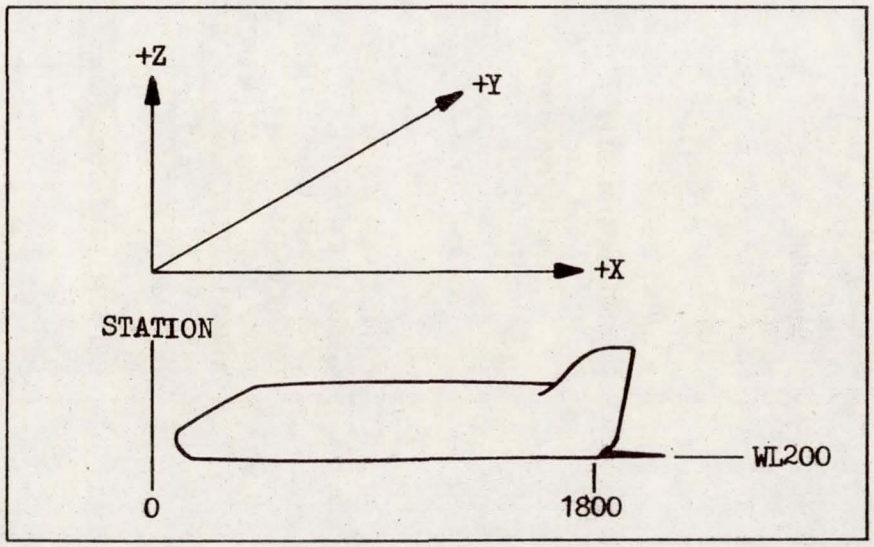

TABLE 3.10-7

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Table 3.10-8

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE 1 of 8
CONFIGURATION	BOOSTER	BY	MV	DATE 4 May 71

MAX DYNAMIC PRESSURE, LB/SQ.FT _____
MAX q α PSF DEGREE _____
ENTRY VELOCITY FT/SEC _____
BALLISTIC COEFFICIENT $W/C_d A$, LB/SQ.FT _____
ENTRY ANGLE DEGREES _____

ULTIMATE LOAD FACTOR	Nx	Ny	Nz	@ WEIGHT, LB
Ascent				
Abort				
Separation				
Entry				
Cruise				
Landing				

1. WING GROUP	INSIDE FUSELAGE	EXPOSED	
Gross Area SQ FT	(1746)	(3667.7)	5413.7*
Torque Box	569	1142.6	
Leading Edge (Fixed)	N/A	1357.7	
Trailing Edge (Fixed)	N/A	212.2	
Movable Surfaces		955.2	
Volume - CU.FT		10477	

	MAC	ROOT THEORETICAL	BODY JUNCTION	PLANFORM BREAK	TIP
CHORD LENGTH (FT.)					
CHORD THICKNESS (FT.)					

SPAN BETWEEN CHORDS (FT)
DIHEDRAL ANGLE (DEG) _____ SPAN BETWEEN DIHEDRAL BASES FT _____
SWEEP BACK ANGLE (MEAN CHORD) 37° 25% CHORD _____
AIRFOIL DESCRIPTION NACA 00107-64
TAPER RATIO .435
THICKNESS/CHORD: ROOT .107 TIP .107
DESIGN LOAD _____
CRITICAL LOAD CONDITION REENTRY

CONTROL SURFACES	TYPE	RETRACT	AREA — SQ FT EXTEND
	N/A	N/A	N/A
L. E. Flaps	N/A	N/A	N/A
Spoilers	N/A	N/A	N/A
Speed Brakes	N/A	N/A	N/A

PRIMARY STRUCTURAL MATERIALS	MATERIAL	DESIGN TEMP °F
Torque Box	Alum.	200
Leading Edge	Alum.	200
Trailing Edge (Fixed)	Alum.	200
Movable Surfaces	Alum.	200

NOTES: N/A = Not Applicable
* Theo. Area

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Table 3.10-8 (Cont'd)

SOUTH POLAR DESIGN DATA SUMMARY					PAGE 2 of 8
CONFIGURATION BOOSTER			BY MV		DATE 4 May 71
2. TAIL GROUP	INBOARD VERTICAL	OUTBOARD VERTICAL	FWD HORIZONTAL	AFT	
EXPOSED AREA FT^2 TOTAL	(N/A)	(791)	(1097)	(N/A)	1888
Torque Box		160	542		(702)
Leading Edge (Fixed)		195	239		(434)
Trailing Edge (Fixed)		69	108		(177)
Movable Surfaces		367	208		(575)
CARRY THROUGH AREA - FT^2		0	402		402
EXPOSED SEMI-SPAN-FT^2		21.8	24.4		
CARRY THROUGH SPAN-FT		0	18.5		
NO. OF SURFACES/VEHICLE		2	1		
VOLUME - CU FT		967	4843		5810
PRIMARY STRUCT.MATERIAL		ALUM	ALUM		
CHORD LENGTH					
Root (Theoretical) Ft.					-
Mac Ft					
Body, Junction					
Tip, Ft					
CHORD MAX. THICKNESS					
Root Ft					
Tip, Ft					
DIHEDRAL ANGLE, DEGREE					
SWEEPBACK, 25% CHORD					
CRITICAL LOAD CONDITION					
3. BODY GROUP	FWD BULKHEAD	BARREL	AFT BULKHEAD	COMMON BULKHEAD	
INTEGRAL TANK WETTED AREA - FT^2					20222
Oxidizer Tank	236	4096	1145	N/A	(5477)
Fuel Tank	1145	10367	1145	N/A	(12657)
Inter-Tank Structure		2088		N/A	(2088)
Primary Str. Mat'l	ALUM.	ALUM.	ALUM.		
ULLAGE PRESSURE - PSI	OXIDIZER 30*		FUEL 30*		
BASIC STRUCTURE WETTED AREA - FT^2	FWD ( )	CTR ( )	AFT ( )	SKIRT ( )	
Sidwalls					
Bulkheads					
Partitions					
Thrust Structure (Main Ascent Engine)					
Body Volume - Cu Ft (Total)					( )
PRIMARY STRUCTURAL MATERIAL					

* Max Relief Pressure

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Table 3.10-8 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE 3 of 8
CONFIGURATION	BOOSTER	BY	M.V.	DATE 4 May 71

3. BODY GROUP (Continued)

MISCELLANEOUS	WETTED AREA-SQ.FT.	VOLUME CU.FT.	ULT DES.PRESS. DIFF. – PSI
Crew Compartment			
Equipment Compartment			
Radome Antennas			
Speed Brakes			
Doors Fairings			
Tanks - Oxidizer			
Tanks - Fuel			

4. INDUCED ENVIRONMENT PROTECTION

	WING	HOR.TAIL	VERT.TAIL	BODY	
TOTAL VEH.WETTED AREA–FT^2	7214	3015	1596	21715	33540
LEADING EDGE NOSE CAP AREA – SQ FT	( )				
Material					
Material					
Material					
SURFACE PROT. AREA-SQ.FT.	( )				
Material					
Material					
Material					
Material					
Material					
Material					
Material					
Material					
Material					
UNPROTECTED AREA–SQ FT	( )				
BASE: MATERIAL					
TOTAL VEHICLE VOLUME – CU. FT. (OUTER MOLDLINE)					
VOLUME INSIDE PRIMARY Structure – Cu Ft					
TPS VOLUME CU FT					
LEAD. EDGE/NOSE CAP RAD–FT					
LEAD. EDGE/NOSE CAP MAX TEMP ^oF					
LOWER SURFACE MAX TEMP ^oF					
SIDE SURFACE MAX TEMP ^oF					
UPPER SURFACE MAX TEMP ^oF					

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Table 3.10-8 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE 4 of 8	
CONFIGURATION	BOOSTER	BY	M.V.	DATE	4 May 71

5. LANDING, RECOVERY, DOCKING

ALIGHTING GEAR	NO./VEH	DESIGN LOAD - LB	EXTENDED STRUT LGTH – FT	STROKE FT	PRIMARY MATL	BRAKE MATL
Main Gear	2				Steel	Carbon
Nose Gear	1				Steel	N/A

Max. Design Landing Wt - Lb =
Landing Speed - Knots =
Angle of Attack @ Landing - Deg =

Limit Landing Sink Speed
Ft/Sec @ LB. Landing Weight
10 Ft/Sec @ LB. Max Design Landing Weight

SEPARATION SYSTEM
Design "q" @ Separation – psf = TBD
Max. Axial Acceleration - g's = 3
Max. Design Separation - Wt. Lb. =

DECELERATION CHUTE
Diameter - Ft = TBD
No./Vehicle = TBD

6. PROPULSION – MAIN ASCENT

	THRUST–SL	THRUST– VAC	EXP RATION	I_{sp} VAC	CHAMB PRES PSI
ENGINE	550000	605000	35:1 *	(439-3/V NE)	3000

PROPELLANT SYSTEM	FUEL	OXIDIZER	
Ullage Pres. - psi	25-28.5	18-28.5	(OPERATING LIMIT)
Propel Type	LH_2	LO_2	
Ullage Vol - Ft^3	4109	1410	(5519)
Pressurant	GH_2	GO_2	
Total Tank Vol - Ft^3 *	93064	33649	126713
Usable Prop Vol - Ft^3 **	88790	32709	121499
Total Len of Feedlines - Ft			

TANKAGE – NONINTEGRAL	NO.	SHAPE	WET AREA SQ FT	VOLUME CU FT
Oxidizer				
Oxidizer				
Oxidizer				
Fuel				
Fuel				
Fuel				

* N.E. = No. of Engines

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Table 3.10-8 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE 5 of 8
CONFIGURATION	BOOSTER	BY	M.V.	DATE 4 May 71

7. PROPULSION – CRUISEBACK

No. of Engines	10
Engine Thrust – S.L. Static – Lb	18000
Specific Fuel Consumption Lb/Lb – Thrust Per Hr @ Nominal Cruise Altitude – Ft	
Nominal Cruise Altitude – Ft	18000
Nominal Cruise Speed – Knots	286
Cruise Altitude Engine Out – Ft	14000
Cruise Speed, Engine Out – Knots	281
Cruise Range (Actual Req) – NAMI	439
Cruise Range (Max Avail – No Headwind, All Engines Up)	692
Cruise Lift Drag Ratio	6.56
Engine Thrust Sized By	
Lift Coefficient for Critical Thrust Condition	

	TYPE	TANK VOL CU FT	TANK MATL	TANK PRES-psi	BURST FACTOR	NO. OF TANKS
FUEL SYSTEM	JP-4					4
PRESSURIZATION SYSTEM						

AIR INTAKE – LENGTH–Ft
– DIAMETER–Ft
NACELLE WETTED AREA – EACH– SqFt

8. PROPULSION – AUXILIARY

THRUSTERS

Thrust (Vac) – Lb	I_{sp} – Sec	ACS	QUANTITY REQ MANEUVER	COMBIN	
1750	381 Steady	16			(16)
	302 Pulsed				( )
					( )
					( )
					( )

PROPELLANT SYSTEM	TYPE	TANK** VOLUME	TANK MATL	TANK PRES–psi	BURST FACTOR	NO. OF TANKS
Fuel	LH_2	326*		30	2.0	1
Oxidizer	LO_2	31*		30	2.0	1
Fuel Pressurant	GH_2	INCL. in Above Tank				
Oxidizer Press.	GO_2	INCL. in Above Tank				

**Net Usable Plus Ullage

9. PRIME POWER

	SPECIFIC POWER		TOTAL POWER		TYPE
Batteries	N/A	Watt-Hr/Lb		Watt-Hrs	
Engine/Turbine	.422	HP-Hr/Lb Fuel	168	HP–Hr	H2/02
		HP/Lb of Engine (		HP	
Fuel Cell		Watt-Hrs/Lb Fuel		Watt-Hrs	
		Watts/Lb of Fuel Cell		Watts	

Table 3.10-8 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE 6 of 8	
CONFIGURATION	BOOSTER	BY	M.V.	DATE	4 May 71

10. ELECTRICAL POWER CONVERSION/DISTRIBUTION

System Voltage	=	28/115	VOLTS
Peak Power	=	40000	WATTS
Average Power	=	25000	WATTS

11. HYDRAULIC POWER CONVERSION/DISTRIBUTION

System Nominal Oper Pressure	=	3000	PSI
Peak Power	=	TBD	HORSEPOWER
Average Power	=	TBD	HORSEPOWER
Total Volume of Fluid	=	TBD	FT^3

Fluid Type Petroleum Base Max Oper Temp - ^{o}F 275

12. SURFACE CONTROLS

SURFACE	AREA FT^2	MAX DEFL RATE DEG/SEC	MAX DEFL DEG	DESIGN HINGE MOM. FT/LB	NO/ VEHICLE
Inboard Elevon	531.5	30	+30	TBD	2
Outboard Elevon	531.5	30	+30	TBD	2
Rudder	187	30	+30	TBD	2
Canard Flap	177	30	+30 -60	TBD	2

13. AVIONICS (LIQUID COOLED, 120 VDC POWER, QUAD REDUNDANT, FO/FO/FS)

14. ENVIRONMENTAL CONTROL

	TOT. STOR VOL - FT^3	STORAGE PRES.	TANK MATL	NO. OF TANKS
Gas Supply System				
Primary Air	1.7	3000	TBD	2
Second Oxygen (Emerg.)	TBD	TBD	TBD	2
Diluent				

Gas Requirement Average Rates

Metabolic	=	2.15	Lb Man-Day
Leakage	=	2.0	Lb Day
Repressurize	=	N/A	Lb Repressurize (CABIN)
Repressurize	=		Lb/Repressurize (AIRLOCK)

Heat Transport System Capacity = 46000 Btu Hr (PEAK) Radiator
= 38000 Btu Hr (PEAK) Total System

Radiator Area = None SqFt Material
Cryogenic Hydrogen = Lb/Btu Btu's

Water Management System Capacity

Drinking Water	=	2	Lb Man-Day	1	Man Days
Washing	=	0	Lb Man-Day	0	Man Days
Cooling	=	TBD	Lb Btu		BTU's

15. PERSONNEL PROVISIONS (2 Crew)

16. RANGE SAFETY AND ABORT

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Table 3.10-8 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE 7 of 8
CONFIGURATION	BOOSTER	BY	M.V.	DATE 4 May 71

17. BALLAST

Design C.G. Envelope FWD = _____ %L AFT = _____ %L
Nominal C.G. without Ballast = _____ %L ENTRY
Nominal C.G. with _____ Lb Ballast = _____ %L ENTRY
Max. Design Landing Wt-Lb _____ @ _____ % MAC. C.G.
(FWD and AFT JP Tanks and Fuel Transfer System Provided for c.g. Control)

18. GROWTH/UNCERTAINTY

Current Allowance = 30,588 LB
Contractors Est of Allowance Needed to guarantee
Current Payload and Gross Wt = _____ LB*
Remaining Growth Allowance for
Customer Changes = _____ LB

*For System Requirements as Defined by _____

19. OPEN

20. PERSONNEL

No. of Crew = 2 ; ave percentile man = 95
No. of Personnel = 0 ; ave percentile man = N/A
TOTAL

21. CARGO CARGO BAY VOLUME = N/A Cu Ft

Bay Dia = N/A FT
Bay Lgth = N/A FT

22. ORDNANCE

23. RESIDUAL FLUIDS - DEFINE WEIGHT ESTIMATING RATIONALE

24. OPEN

25. RESERVE FLUIDS

26. INFLIGHT LOSSES

Table 3.10-8 (Cont'd)

SOUTH POLAR	DESIGN DATA SUMMARY			PAGE 8 of 8
CONFIGURATION	BOOSTER	BY	M.V.	DATE 4 May 71

27 - 29 PROPELLANTS EXPENDED

	ASCENT	CRUISE	MANEUVER	ATTITUDE
Oxidizer/Fuel Ratio (By Weight)	6:1	N/A		4.5:1
Oxidizer Ullage Volume - Percent*	4.2			5
Fuel Ullage Volume - Percent*	4.4	35.3		5
Fuel Density - pcf	4.38	48.2		4.38
Oxidizer Density - pcf	70.59	N/A		70.59
Fuel Bias - Percent	.151			
Incremental Velocity -fps	(14430)			
Inertial				
Maneuver Losses				
Gravity Losses				
Drag Losses				
Back Pressure Losses				
Engine Cant				
Earth Rotation				
FLT. Performance Reserve	0			
Range - NA MI		(692)		
Ground Miles		439		
Head Wind Allow		TBD		
Contingency		TBD		
Equivalent Burn Time at Full Thrust Provided for Landing/Go-around-minutes				

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Table 3.10-9

SOUTH POLAR	GROUP WEIGHT STATEMENT			Page 1 of 6
CONFIGURATION	BOOSTER	BY	MV	DATE 4May71

1. WING GROUP				46,239
	CARRY THROUGH	EXPOSED SURFACE		
Basic Structure			39,738	
Torque Box	11,779	20,023		
Leading Edge		7,650		
Trailing Edge		286		
Secondary Structures			6,501	
Variable Geometry (incl ____ lbs mechanism)				
Doors Insulation Fairings				
Control Surfaces	SURFACE	SUPT/MECH		
Elevon — (incl. bal wt. ____ lb)				
T.E. Flaps (ELEVONS)	6,501			
L.E. Flaps				
Spoilers				
Speed Brakes				
2. TAIL GROUP	VERTICAL	HORIZ		13,400
Basic Structure	(1,943)	(7,771)	9,714	
Torque Box	1,085	2,919		
Carry Through		2,923		
Leading Edge		1,315		
Trailing Edge	858	614		
Secondary Structure				
Control Surfaces	SURFACE	SUPT/MECH	3,686	
Rudder (incl. bal wt 0 lb)	2,496			
Body Flap	964	226		
3. BODY GROUP				136,059
Integral Tankage			88,120	
Fuel Tank	51,750 + 2,502 = 54,252			
Oxidizer Tank	17,625 + 675 = 18,300			
Between tanks (cmn bkhd)	11,523 + 576 = 12,099			
Insulation		3,469		
Basic Structure	FWD. CTR. AFT.	SKIRT		
	(794)	(33,325)	34,119	
Sidewalls				
Bulkheads				
Partitions				
Thrust Structure (main ascent engine)				
Secondary Structure			13,820	
Crew Compartment		1,500		
Equipment Compartments				
Payload Attach & Deploy				
Speed Brakes				
Engine Heat Protection				
Interstage (incl. mech. lbs) 4,342+221		4,563		
Doors/fairings		1,190		
Gear/wing provisions/ABES		6,567		
Contingency				

Table 3.10-9 (Cont'd)

SOUTH POLAR	GROUP WEIGHT STATEMENT			Page	2 of 6
CONFIGURATION	BOOSTER	By	MV	Date	4May71

	WING	HOR.TAIL	VER. TAIL	BODY		
4. INDUCED ENVIRONMENTAL PROTECTION						59,917
Thermal Protection						
Leading Edge/Nose Cap						
Surface Protection						
Base Heat Protection						
Sound Protection						
Meteorite / Radiation Protection						
Contingency						

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Table 3.10-9 (Cont'd)

SOUTH POLAR	GROUP WEIGHT STATEMENT			Page 3 of 6
CONFIGURATION	BOOSTER	By	MV	Date 4May71

5. LANDING, DOCKING							21,432
Alighting Gear	ROLLING	STRUCTURE	CONTROLS				
Main					14,359		
Nose					7,073		
Docking							
Auxiliary Systems							
Deceleration chutes							
Flotation gear							
Handling gear							
Contingency							
6. PROPULSION – MAIN ASCENT							107,475
Engine & Accessories					85,480		
Engine (as supplied)				81,070			
Gimbal System				2,904			
Chilldown SP.				948			
Propellant Utilization System				100			
Accessories and Misc.				458			
Installation, Ducts, Shrouds							
Propellant System		FUEL	OXIDIZER		21,995		
Purge							
Pressurization and Vent		1,394	1,554				
Fill, Drain and Dump		851	740				
PCV System							
Pneumatic Sy.		2,486					
Feed Systems		5,022	9,948				
Vortex, Flow Control System							
Supports and Install							
Tankage – Nonintegral							
Fuel							
Oxidizer							
Contingency							
7. PROPULSION – CRUISE BACK							36,154
Engine & Accessories					25,040		
Engine				22,000			
Ignition and Control Sy							
Lubrication Sy (dry)							
Accessories				3,040			
Installation, Ducts, Shroud					2,580		
Air Induction							
Engine Mounting							
Nacelles Pylons (incl ____ lb mech)							
Exhaust System					1,384		
Propellant System							
Fill Drain and Dump				139			
Pressurization (dry)				246			
Vent System							
Pneumatic Control				157			
Feed System				842			
Transfer System							
Dump System							
Supports/Installation					3,757		
Tankage – Non Integral	2,799						
Sealant	958						
Contingency					3,393		

Table 3.10-9 (Cont'd)

SOUTH POLAR GROUP WEIGHT STATEMENT						Page 4 of 6
CONFIGURATION BOOSTER				By MV		Date 4May71
8. PROPULSION – AUXILIARY						4,573
Thruster Installation			549			
Thruster		549				
Accessory						
Propellant System			3,328			
Fill/Drain/Vent Lines & Valves						
Pressurization						
Feed System/Accumulators						
Conditioning						
Supports						
Tankage			696			
Tanks						
Insulation						
Supports						
Contingency						
9. PRIME POWER	POWER UNIT	MTG INSTAL	PROPEL TANK/SYS	TEMP CONTROL	CON-TROLS	TOTAL	1,257
Batteries						
Engine Turbine						1,257
Fuel Cells						
Contingency						
10. ELECTRICAL		SUPPLY	CONVER-SION	CONTROL UNITS			1,363
Equipment						448
Distribution and Control Circuitry and Protection		99	260	89		855
Utility Systems						
Supports/Installation						60
Contingency						
11. HYDRAULIC							5,290
Power Supply						
Distribution Control Ctr						
Temperature Control Sy						
Auxiliary Systems						
Supports/Installation						
Contingency						
12. SURFACE CONTROLS							4,910
Cockpit Controls						
Flight Control System						
System Actuation	CONTROLS	POWER XMISSION	ACTU-ATOR	FEEL SY	SUPTS INSTALL	(SEE AVIONICS)
Aileron						
Elevator						
Rudder						
T.E. Flap						
Speed Brake						
Contingency						
13. AVIONICS	UNITS	CIRCUITRY	COOLING	ANTENNAS	INSTALL.		3,682
Guid / Nav						2,282
Flight Control						
Data Management						
Communicate						155
Config. Seq.						
Instrumentation						216
Displays						1,029
Contingency						

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Table 3.10-9 (Cont'd)

SOUTH POLAR GROUP WEIGHT STATEMENT								Page 5 of 6	
CONFIGURATION	BOOSTER					By	MV	Date	4May71
14. ENVIRONMENTAL CONTROL									4,137
Gas Supply System (dry)									
Gas Management, Processing (dry)									
Heat Transport System (dry)									
Water Management System (dry)									
Purge System									
Insulation									
Contingency									
15. PERSONNEL PROVISIONS									290
Seats/Restraint Sys (No =)								120	
Fixed Life Support Equipment									
Emergency Equipment								29	
Cargo Handling									
Furnishings								141	
16. RANGE SAFETY AND ABORT									0
17. BALLAST									0
18. GROWTH/UNCERTAINTY									30,588
19. OPEN									0
SUBTOTAL (Dry Weight)									476,766
20. PERSONNEL	NO	MAN	GARMENTS	HELMET	PRESS. SUIT	ACCESSORIES			400
Crew (2)									
Passenger( )									
Personal Gear/Accessories									
Life Support									
Food									
Equipment - Portable									
21. CARGO									0
22. ORDNANCE									0
23. RESIDUAL & UNUSABLE FLUIDS									8,804
Ascent									
Cruise									
Maneuver (Incl. All Aux. Tk. Residuals)									
Attitude Control									
ECS									
EPS									
Hydraulic									
Misc (Shock struts, etc.)									
24. OPEN									0
SUBTOTAL(Inert Weight)									485,970

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Table 3.10-9 (Cont'd)

SOUTH POLAR	GROUP WEIGHT STATEMENT			Page 6 of 6
CONFIGURATION	BOOSTER	By	MV	Date 4May71
25. RESERVE FLUIDS				51,441
Ascent				
Cruise (Eng. Out, with Headwinds)			51,000	
Maneuver				
Attitude Control			228	
ECS / EPS / Hydraulic			213	
				-39,000
Cruise Fuel Dumped				20,027
26. INFLIGHT LOSSES				
Ascent				
Cruise				
Maneuver (Incl. All Aux. Tk Boiloff)				
Attitude Control				
ECS and Misc.				
APU			1,127	
Hydraulic				
27. PROPELLANT - ASCENT				2,643,026
28. PROPELLANT - CRUISE (NOMINAL)				90,562
(DUMPED)				39,000
		MANEUV.	ACS	
29. PROPELLANT - MANEUVER/ACS		0	1,222	1,222
TOTAL (Gross Weight)				3,292,248

Table 3.10-10

SOUTH POLAR — **MISSION WEIGHT SUMMARY**

CONFIGURATION: BOOSTER | BY: MV | DATE: 4May71

CODE	SYSTEM	A	B	C	D	E	F	G	H
1	WING GROUP	46,239							
2	TAIL GROUP	13,400							
3	BODY GROUP	136,059							
4	INDUCED ENVIR PROTECTION	59,917							
5	LANDING, RECOVERY, DOCKING	21,432							
6	PROPULSION - ASCENT	107,475							
7	PROPULSION - CRUISE	36,154							
8	PROPULSION - AUXILIARY	4,573							
9	PRIME POWER	1,257							
10	ELECT CONVER & DISTR	1,363							
11	HYDRA CONVER & DISTR	5,290							
12	SURFACE CONTROLS	4,910							
13	AVIONICS	3,682							
14	ENVIRONMENTAL CONTROL	4,137							
15	PERSONNEL PROVISIONS	290							
16	RANGE SAFETY & ABORT								
17	BALLAST								
18	GROWTH/UNCERTAINTY	30,588							
19									
	SUBTOTAL (DRY WEIGHT)	476,766							
20	PERSONNEL	400							
21	CARGO	0							
22	ORDNANCE	0							
23	RESIDUAL FLUIDS	8,804							
24									
	SUBTOTAL(INERT WEIGHT)	485,970							
25	RESERVE FLUIDS	51,441	51,441	51,441	51,441	51,441	51,441	12,441	
26	IN FLIGHT LOSSES	20,027	20,027	20,027	20,027	1,127			
27	PROPELLANT - ASCENT	2,643,026	1,982,270	1,321,513					
28	PROPELLANT - CRUISE	90,562	90,562	90,562	90,562	90,562			
29	PROPELLANT— MANEUV/ACS	1,222	1,222	1,222	1,222				
30									
	TOTAL(GROSS-WEIGHT)LB.	3,292,248	2,631,492	1,970,736	649,222	629,100	537,411	498,411	

DESIGNATORS:

EVENTS

A LAUNCH WEIGHT
B ~~IGNITION EFFECTIVE~~ 25% BURN
C 50% BURN
D ~~MECO~~ BURNOUT
E ~~CRUISE~~ CRUISE START
F ~~CRUISE~~ CRUISE END
G ~~TOUCH~~ LANDING
H ~~DOCKING~~

NOTES & SKETCHES:

11 MAIN ENGINE BOOSTER

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Table 3.10-11

SOUTH POLAR WEIGHT SUMMARY (LAUNCH CONDITION)

CONFIGURATION: Two-Stage LS 400-7A | BY: | DATE: 4 May 71

CODE	SYSTEM	ITEM OR MODULE: BOOSTER	EST	CAL	ACT	ORBITER	EST	CAL	ACT
1	Wing Group	46239							
2	Tail Group	13400				16264			
3	Body Group	136059				42598			
4	Induced Envir Protection	59917				34931			
5	Landing, Recovery, Docking	21432				8806			
6	Propulsion - Ascent	107475				46060			
7	Propulsion - Cruise	36154				350			
8	Propulsion - Auxiliary	4573				6970			
9	Prime Power	1257				1541			
10	Elect Conver & Distr	1363				3747			
11	Hydra Conver & Distr	5290				2073			
12	Surface Controls	4910				4085			
13	Avionics	3682				3678			
14	Environmental Control	4137				1274			
15	Personnel Provisions	290				210			
16	Range Safety & Abort								
17	Ballast								
18	Growth/Uncertainty	30588				15342			
19									
	SUBTOTAL (Dry Weight)	476766				187929			
20	Personnel	400				725			
21	Cargo	0				40000			
22	Ordnance								
23	Residual Fluids	8804				3191			
24									
	SUBTOTAL (Inert Weight)	485970				231845			
25	Reserve Fluids	51441				6640			
26	In Flight Losses	20027				8321			
27	Propellant - Ascent	2643026				546439			
28	Propellant - Cruise	90562				0			
29	Propellant - Maneuv/Acs	1222				37848			
30									
	TOTAL (Gross-Weight) LB	3292248				831093			

DESIGNATIONS:

Class of Weight
EST - Percent Estimated Weight
CAL - Percent Calculated Weight
ACT - Percent Actual Weight

NOTES & SKETCHES:

Gross Launch Wt = 4,123,341

Table 3.10-12

DUE EAST ABORT PERFORMANCE DATA NUMBER					
CONFIGURATION	TWO STAGE ORBITER LS400-7A	By	P.W.	Date	4 May 71

ITEM	UNITS	ORBITER	BOOSTER
WEIGHTS DATA:			
Burnout Weight	10^3 -Lb	273.2 (1)	642 (2)
Nominal Propellant Load	10^6 -Lb	.590	2.643
Payload	10^3 -Lb	79.5	871.1
Gross Weight	10^6 -Lb	.871	3.285
VELOCITY DATA:			
Nominal Ascent Velocity	Ft/sec	16072 (3)	1.4293 (3)
Flight Performance Reserve	Ft/sec	300	
Total Ascent Velocity	Ft/sec	30665 (3)	
Ascent Specific Impulse	Sec	456 (4)	438.1 (4)
On Orbit Maneuver Velocity	Ft/sec	170 (5)	
Ascent Thrust/Weight (initial)	-	.791 (3)	1.459
On Orbit Maneuver I_{sp}	Sec	439 (5)	
PAYLOAD SENSITIVITY DATA			
Specific Impulse	Lb/sec	829	741
Inert Weight	Lb/lb	-1.	-.167
Propellant Load	Lb/lb	.204	.0558
On Orbit Maneuver Velocity	Lb/ft/sec		
Ascent Velocity	Lb/fps	-23.2	
Thrust/Weight - (initial)	Lb/(0.1 T/W)	5530 (6)	23 6050
Orbit Inclination	Lb/deg	-	330
Launch Site Altitude	Lb/ft	1	.43
Gross Weight	Lb/lb		
INJECTION ORBIT CHARACTERISTICS			
Apogee	nm	50	
Perigee	nm	100	
Inclination	Deg	28.5	
Launch Site Latitude	Deg	28.5	
Launch Site Altitude	Ft above SL	0	

NOTE: Sensitivities are estimates for due east abort

(1) Post Retro
(2) Post Staging
(3) Abort
(4) 3σ Performance
(5) Retro ΔV = 150 fps (Isp = 439), ACPS ΔV = 20 fps (Isp = 352)
(6) One engine operating (Abort)

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Table 3.10-13

DUE EAST — WEIGHT SUMMARY (LAUNCH CONDITION)

CONFIGURATION: TWO-STAGE | LS 400-7A | BY: Broadhead | DATE: 4 May 71

CODE	SYSTEM	ITEM OR MODULE: BOOSTER	EST	CAL	ACT	ORBITER	EST	CAL	ACT
1	Wing Group	46,239				N/A			
2	Tail Group	13,400				16,264			
3	Body Group	136,059				42,598			
4	Induced Envir Protection	59,917				34,931			
5	Landing, Recovery, Docking	21,432				8,806			
6	Propulsion - Ascent	107,475				46,060			
7	Propulsion - Cruise	36,154				350			
8	Propulsion - Auxiliary	4,573				6,970			
9	Prime Power	1,257				1,541			
10	Elect Conver & Distr	1,363				3,747			
11	Hydra Conver & Distr	5,290				2,073			
12	Surface Controls	4,910				4,085			
13	Avionics	3,682				3,678			
14	Environmental Control	4,137				1,274			
15	Personnel Provisions	290				210			
16	Range Safety & Abort								
17	Ballast								
18	Growth/Uncertainty	30,588				15,342			
19									
	SUBTOTAL (Dry Weight)	476,766				187,929			
20	Personnel	400				725			
21	Cargo	0				79,510			
22	Ordnance								
23	Residual Fluids	8,684				3,441			
24									
	SUBTOTAL (Inert Weight)	485,850				271,605			
25	Reserve Fluids	51,441				8,169			
26	In Flight Losses	20,027				8,391			
27	Propellant - Ascent	2,643,026				545,852			
28	Propellant - Cruise	82,562				0			
29	Propellant - Maneuv/Acs	1,222				37,723			
30									
	TOTAL (Gross-Weight) LB	3,285,128				871,740			

DESIGNATIONS:

Class of Weight
EST - Percent Estimated Weight
CAL - Percent Calculated Weight
ACT - Percent Actual Weight

NOTES & SKETCHES:

Gross Launch Wt = 4,156,868

Table 3.10-14

55 DEG/270 NM PERFORMANCE DATA NUMBER					
CONFIGURATION	Two-Stage Orbiter LS 400-7A	By		Date	4 May 71

ITEM	UNITS	ORBITER	BOOSTER
WEIGHTS DATA:			
Burnout Weight	10^3-Lb	254.2(1)	644.1(2)
Nominal Propellant Load	10^6-Lb	.582	2.643
Payload	10^3-Lb	35.9	843.
Gross Weight	10^6-Lb	.843	3.287
VELOCITY DATA:			
Nominal Ascent Velocity	Ft/sec	15277(3)	14377(3)
Flight Performance Reserve	Ft/sec	301	--
Total Ascent Velocity	Ft/sec	29954	--
Ascent Specific Impulse	Sec	456.9(4)	438.1 (4)
On Orbit Maneuver Velocity	Ft/sec	1500	--
Ascent Thrust/Weight (initial)	-	1.5	1.464
On Orbit Maneuver I_{sp}	Sec		
PAYLOAD SENSITIVITY DATA			
Specific Impulse	Lb/sec	748	664
Inert Weight	Lb/lb	-1.	-.173
Propellant Load	Lb/lb	.230	.0528
On Orbit Maneuver Velocity	Lb/ft/sec	-17.0	
Ascent Velocity	Lb/fps	-21.9	
Thrust/Weight - (initial)	Lb/(0.1 T/W)	880(6)	4410
Orbit Inclination	Lb/deg	-310	
Launch Site Altitude	Lb/ft	1.	35
Gross Weight	Lb/lb		
INJECTION ORBIT CHARACTERISTICS			
Apogee	nm	50	
Perigee	nm	100	
Inclination	Deg	55	
Launch Site Latitude	Deg	28.5	
Launch Site Altitude	Ft above SL	0	

NOTE: Sensitivities are estimates for nominal 55 deg/270 NM mission with 550K engines

(1) Post Retro
(2) Post Staging
(3) Nominal Mission
(4) 3σ Performance
(5) On Orbit Δ V = 1358 fps (Isp = 439) ACPS Δ V = 142 fps (Isp = 352)
(6) two engines operating (nominal)

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Table 3.10-15

55DEG/270 NM WEIGHT SUMMARY (LAUNCH CONDITION)									
CONFIGURATION	TWO-STAGE	LS 400-7A	BY	Broadhead		DATE	5May71		
		ITEM OR MODULE							
CODE	SYSTEM	BOOSTER	EST	CAL	ACT	ORBITER	EST	CAL	ACT
1	Wing Group	46,239				N/A			
2	Tail Group	13,400				16,264			
3	Body Group	136,059				43,738			
4	Induced Envir Protection	59,917				34,931			
5	Landing, Recovery, Docking	21,432				8,806			
6	Propulsion - Ascent	107,475				46,060			
7	Propulsion - Cruise	36,154				15,756			
8	Propulsion - Auxiliary	4,573				6,970			
9	Prime Power	1,257				1,541			
10	Elect Conver & Distr	1,363				3,747			
11	Hydra Conver & Distr	5,290				2,073			
12	Surface Controls	4,910				4,085			
13	Avionics	3,682				3,678			
14	Environmental Control	4,137				1,274			
15	Personnel Provisions	290				210			
16	Range Safety & Abort								
17	Ballast								
18	Growth/Uncertainty	30,588				16,997			
19									
	SUBTOTAL (Dry Weight)	476,766				206,130			
20	Personnel	400				725			
21	Cargo					35,898			
22	Ordnance								
23	Residual Fluids	8,684				3,382			
24									
	SUBTOTAL (Inert Weight)	485,850				246,135			
25	Reserve Fluids	51,441				7,496			
26	In Flight Losses	20,027				8,341			
27	Propellant - Ascent	2,643,026				546,411			
28	Propellant - Cruise	85,562				5,400			
29	Propellant - Maneuv/Acs	1,222				29,513			
30									
	TOTAL (Gross-Weight) LB	3,287,128				843,296			

DESIGNATIONS:

Class of Weight
EST - Percent Estimated Weight
CAL - Percent Calculated Weight
ACT - Percent Actual Weight

NOTES & SKETCHES:

Gross Launch Wt = 4,130,424

3.11 OPERATIONS

There is no fundamental difference in Baseline Operational Concept between a two-stage shuttle and a stage-and-one-half shuttle. The only real difference is the replacement of the facilities for manufacture and assembly of droptanks by the equivalents for maintenance and repair of booster vehicles. The functional flow of this baseline operation is shown in Fig. 3.11-1.

This functional flow diagram presents all the major phases of the shuttle support operations and shows their relationship and relative time-to-accomplish. The mainline cycle consists of the Post-Landing phase, Maintenance phase, Mate and Checkout phase, Pad Transfer and Installation phase, Prelaunch phase, and Flight Support phase. It requires approximately 19 eight-hour shifts to accomplish the ground operations for a normal turnaround.

Supporting these mainline cycle operations is the Payload Support phase, Pad Refurbish phase, and Abort Support phase. These operations are provided in parallel with the mainline operations and do not affect the normal turnaround time.

The tasks required for a complete vehicle turnaround are:

- Landing at the new landing field
- Orbiter off-loading, purging, safing, and cooling at a new Purge and Safing Area
- Booster and Orbiter scheduled and unscheduled maintenance at a new Maintenance Annex to the Vertical Assembly Building (VAB)
- Payload preparation and support in one of the existing KSC Industrial Area facilities
- Payload installation into the orbiter in the Maintenance Annex
- Delivery of the Booster and Orbiter vehicles from the Maintenance Annex to the VAB on transfer dollies; erection of the vehicles in a

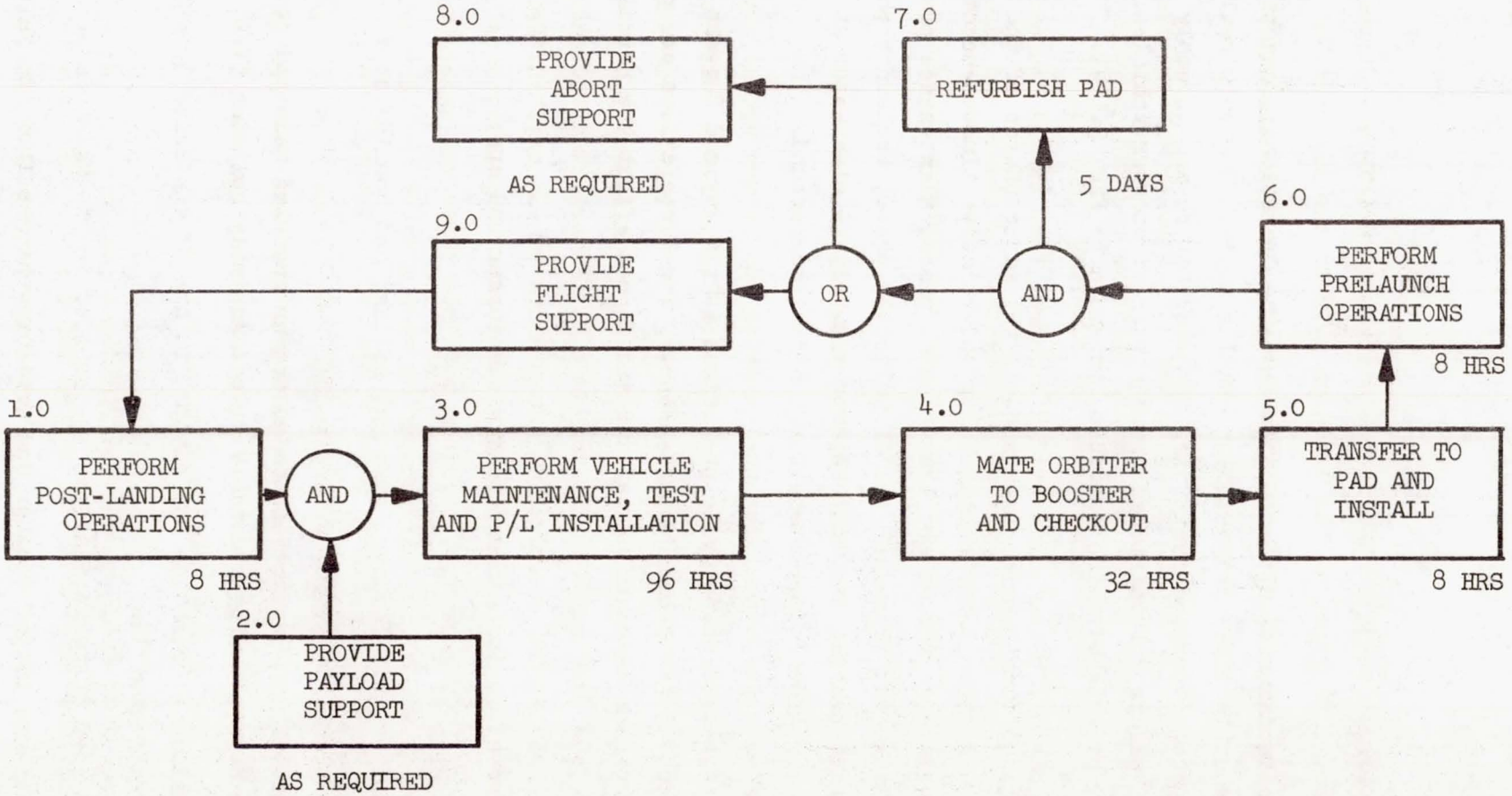

Fig. 3.11-1 Vehicle Ground Operations Flow

VAB high bay; lift and installation onto a LUT-type launcher (same base dimensions as the existing LUT) in the adjacent VAB high bay.

- Vertical mate of the orbiter to the booster in the VAB high-bay cell.
- Transfer to the pad using an existing Crawler/Transporter and installation at the pad on the existing pedestal supports; positioning of a new flame deflector in the existing flame trench; hookup of propellants and other ground support lines to the launcher; loading of propellants, crew, and passengers; countdown and launch.
- Perform flight support operations as required, or provide abort support if necessary.
- Refurbish pad and launcher and prepare for next use.

This baseline concept utilizes the existing KSC Saturn/Apollo equipment and facilities insofar as possible (see Section 2.17.1 for a complete facility description). The changes required in the KSC facilities to support this concept include: a new Landing Field; a new Safing Facility; a new taxiway connecting the Landing Field/Safing Area to the existing Vertical Assembly Building (VAB) Area; a new Maintenance Annex to the existing VAB for booster and orbiter checkout, refurbishment, and repair; modifications to the VAB consisting of rearrangement of fixed work platforms and enlargement of the big doors through which the LUT/Transporter must move the mated vehicle; conversion of one of the existing firing bays of the Launch Control Center (LCC) for shuttle support; addition of a parallel LH_2 propellant storage and transfer system to Pad 39 to supplement the existing systems; two new Launcher Umbilical Towers (LUTs); and new flame deflectors at the launch pads because of the different rocket motor pattern of the shuttle.

The greatest difference in the modifications required between a two-stage and a stage-and-one-half shuttle is in the LUT design. The launcher base is 160 ft long and 135 ft wide. The two-stage vehicle is installed crosswise

on the launcher, with the personnel access tower/stabilizer at one end. This two-stage configuration employing the MacDac booster is 166 ft wide at the tips of the swept wings, the wing tips extend well below the launch support points, the orbiter tail is 118 ft above the booster tail, and the overall vehicle length, mated, is over 270 ft; therefore, a much higher personnel access tower/stabilizer and very tall launch pedestals are required on the launcher base. Service umbilicals for the orbiter must now run up the tower and across swing arms to the vehicle. Personnel access to both vehicles is at higher levels. Because the booster is between the orbiter and the tower, crew and passenger access to the orbiter is more difficult. Cantilevered swing arms that are retracted before launch provide personnel access to both vehicles.

3.12 SYSTEM COSTS

System costs were estimated for the LS 400-6 configuration of the 2-stage system. These costs are covered in detail in Volume IV and summarized as follows:

DDT&E	$ 6,725 M
Recurring Production	560
Recurring Operations (445 Flts)	1,731
Total Program	$ 9,016 M
Orbiter First Unit (including engines)	$ 105.7 M
Booster First Unit (including engines)	165.9 M
Average DDT&E Cost/Flight	$ 15.11 M
Average Recurring Production Cost/Flight	1.26
Average Recurring Operations Cost/Flight	3.89
Total Average Cost/Flt	$ 20.26 M

Cumulative cost versus time for this program is shown in Fig. 3.12-1. Annual funding requires a peak of $1.82 billion as shown in Fig. 3.12-2. Net present value at 10 percent discount totals $5.05 billion as shown in Fig. 3.12-3.

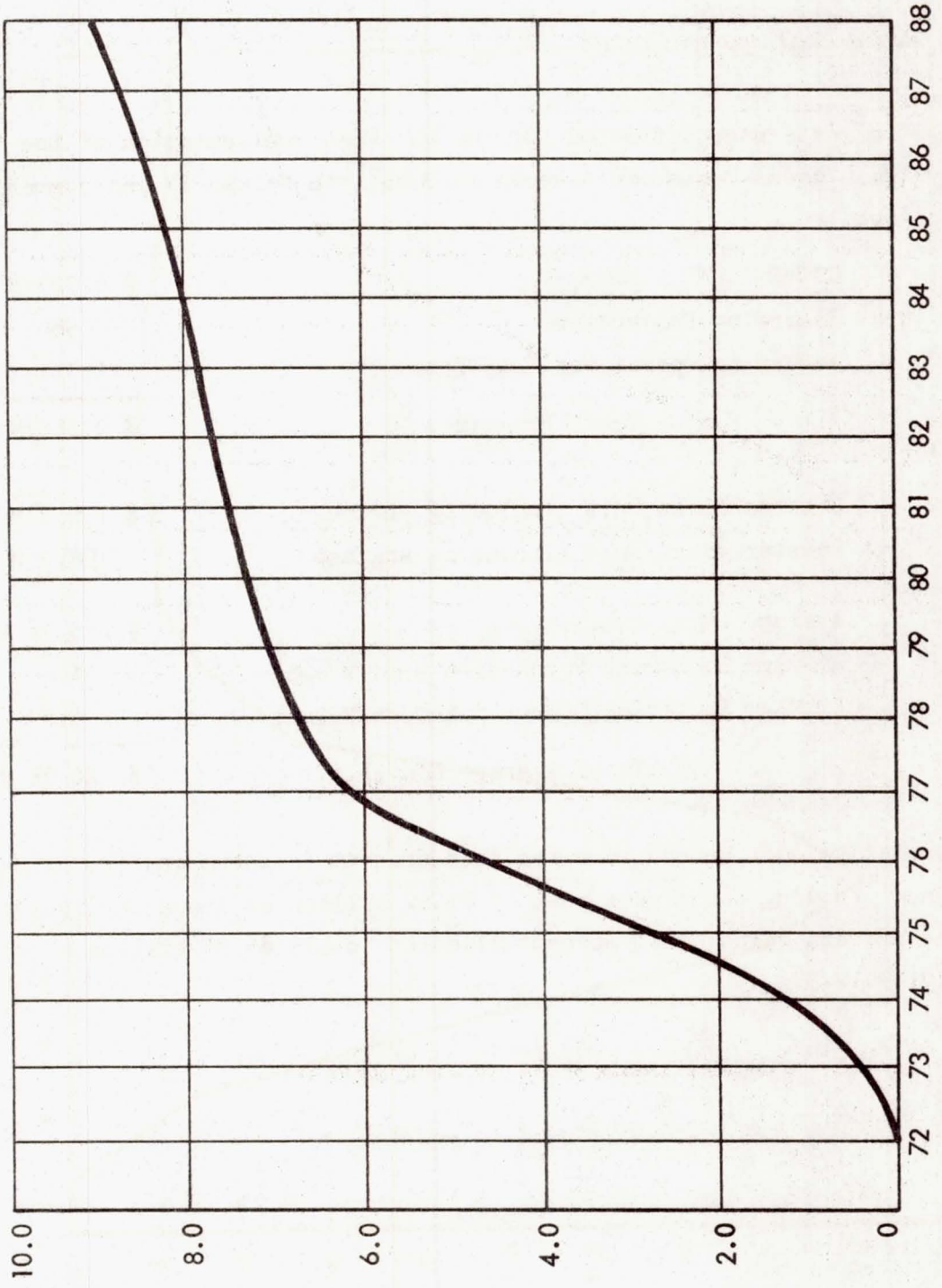

Fig. 3.12-1 Two-Stage Cumulative Cost

Fig. 3.12-1

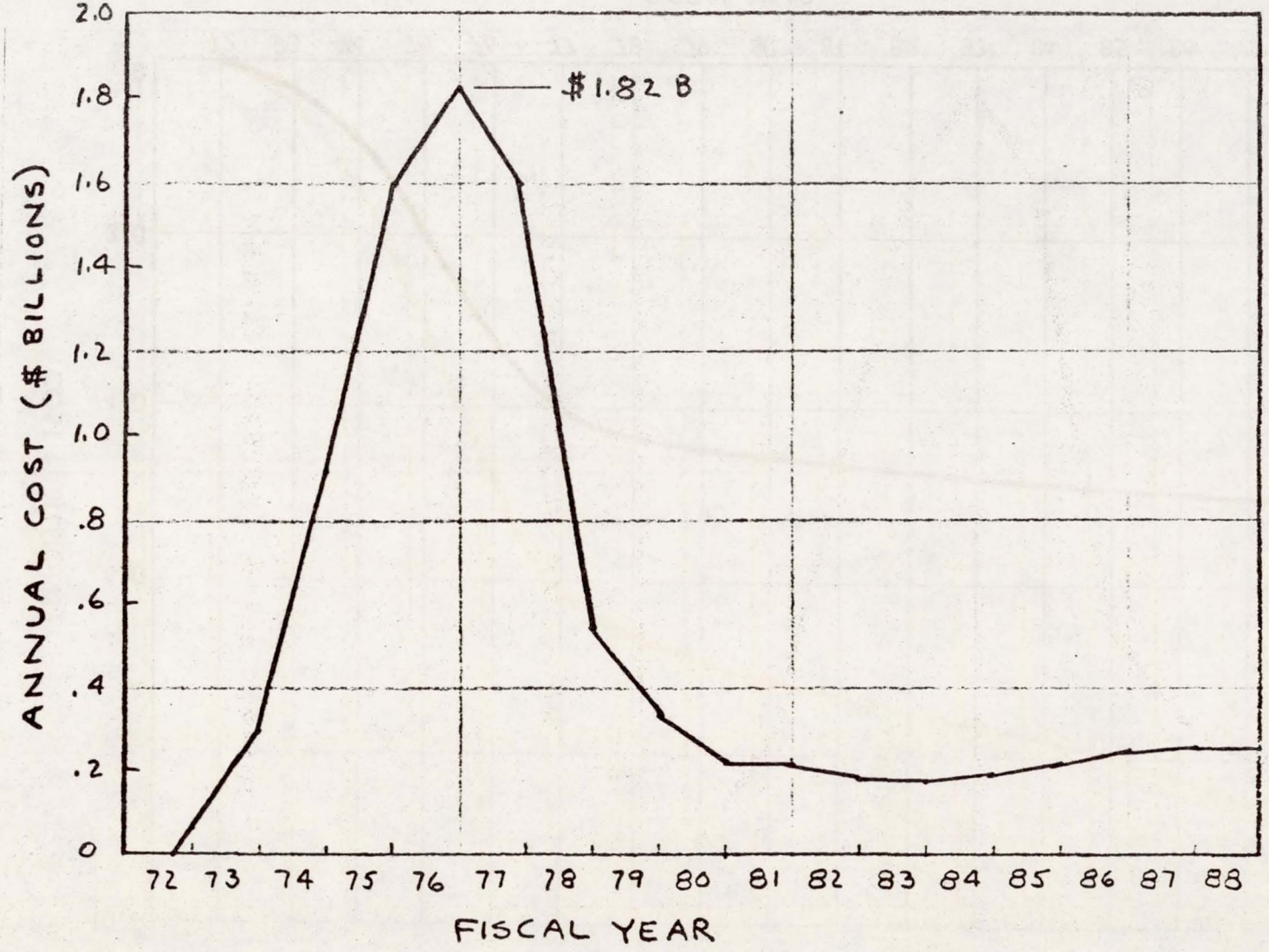

Fig. 3.12-2 2-Stage Annual Cost

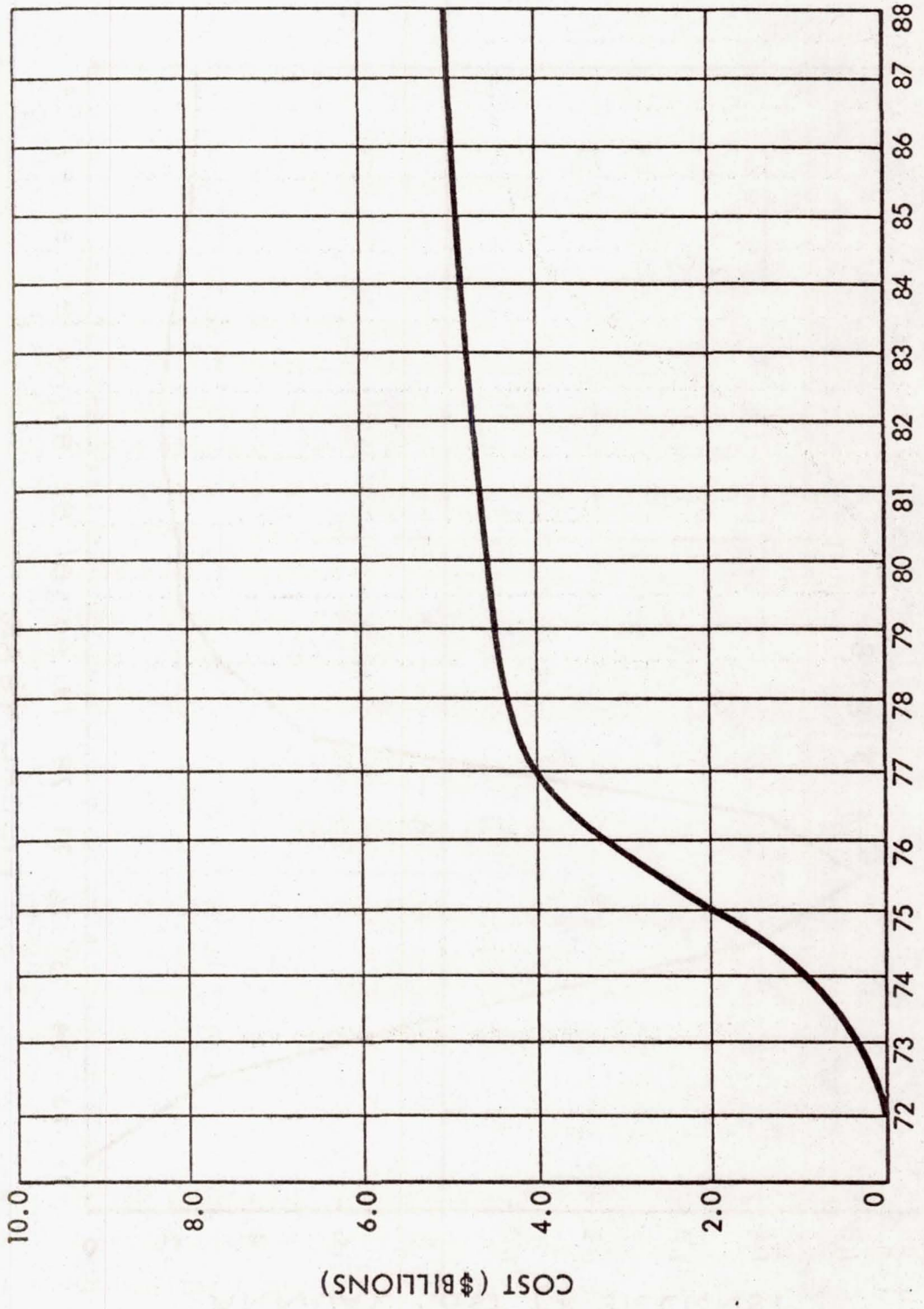

Fig. 3.12-3 Two-Stage NPV at 10%

Fig. 3.12-3

Section 4
CONVERSION SYSTEM

The primary purpose of Task 2, Growth to Two-Stage, is to establish the feasibility and cost of developing a stage-and-one-half space shuttle system which can later be converted to a two-stage, fully reusable system. The reason for interest in such a system is that it can provide a sufficiently early operational capability within the projected $1.25-billion peak annual funding limitation, and it can subsequently be converted to the more desirable two-stage, fully reusable system with the attendant advantage of lower operating costs. The intent of the study is to determine the technical feasibility of the conversion system and to identify the total costs and annual funding requirements for such a system as a function of the time-phasing of the conversion. The study was also to determine the performance penalties associated with the conversion system.

4.1 REQUIREMENTS, GROUNDRULES, AND ASSUMPTIONS

The requirements for the convertible system are essentially the same as for the stage-and-one-half and the two-stage system as defined in Section 2.1. An additional requirement was that, whenever possible, necessary performance penalties were to be placed on the stage-and-one-half system rather than on the two-stage system so that the eventual two-stage system could be near optimum.

Program planning and costing groundrules and assumptions for the convertible system are also as defined in Section 2.1 for the other systems. Phase C/D start is assumed to be January, 1972, FMOF of the stage-and-one-half vehicle is July 1978, and the 445-flight, ten-year mission model applies to the total program. The time-phasing and consequently the number of flights by each of the two configurations is to be determined on a cost basis.

4.2 GENERAL CONSIDERATIONS

4.2.1 Alternatives

For providing the technical, performance, and operational information required for the evaluation of the three alternative programs, it was necessary to establish and to define four different vehicle systems. They are characterized and identified by Code designations as:

- Stage-and-one-half reference system, established solely on the basis of space shuttle mission and program requirements

  Identification Code: $1\frac{1}{2}$-B

- Interim stage-and-one-half system, designed under the constraint of permitting conversion into a two-stage system

  Identification Code: $1\frac{1}{2}$-C

- Two-stage system, designed under the constraint resulting from conversion of the interim stage-and-one-half system

  Identification Code: 2-C

- Two-stage reference system, established solely on the basis of space shuttle mission and program requirements

  Identification Code: 2-B

The definition of the characteristics of the vehicles involved in the conversion program was completed before design improvements and changes reflecting results from wind tunnel test programs were made on the $1\frac{1}{2}$-stage and 2-stage reference designs which led to the final baseline designs presented in the foregoing sections. Consequently the reference $1\frac{1}{2}$-stage and 2-stage systems, $1\frac{1}{2}$-B and 2-B, defined by this study are not fully identical with the $1\frac{1}{2}$-Stage Baseline LS 200-10 and 2-Stage Baseline LS 400-7A presented in Sections 2 and 3 of this report, and these baseline configurations cannot be directly compared with the respective $1\frac{1}{2}$-C and 2-C configurations used for the convertible program. Incorporation of these later changes in the conversion study would further reduce

the performance penalties, listed in Table 4.4-1 for the systems used in the conversion program in comparison with the reference systems.

4.2.2 Vehicle Design, Requirements and Approach

General. In order to ensure the validity of the evaluation, an effort was to be made to pursue the required supporting studies, as far as possible within the scope of the study, to an equal level of depth and detail for these four systems. The driving general considerations for arriving at the individual vehicle design and configuration solutions were: (1) to accommodate the required propellant load in a minimum number of tanks of simple design and in a vehicle of minimum size, and (2) to attain a configuration providing trimmable and stable flight conditions with acceptable aerodynamic performances over the entire flight range. For the 1½-C and 2-C configurations, there was an additional general requirement of obtaining with regard to overall program cost and effectiveness, an optimum balance between design commonality and resulting performance losses.

All four systems were defined on the basis of the revised Level I requirements as established on 27 January 1971, which are essentially identical with the requirements directing the design of the LS 200-10 1½-Stage Baseline and LS 400-7A 2-Stage Baseline systems discussed in Sections 2 and 3 of this report.

The orbital ΔV requirements for the Due East Design Mission and for the South Polar Reference Mission, required for determining the impulse propellant load of the orbiters, were assumed from preliminary analysis for this study to be, (1) for the 2-stage configurations 650 ft/sec for orbit circularization and retro maneuver plus 300 ft/sec for abort-through-orbit capability under the one-engine-out condition, and (2) for the 1½-stage configurations 650 ft/sec for orbit circularization and retro maneuvers.

As the basic configuration for the orbiter vehicles of all four systems, the modified delta-body form was selected. Based on extensive LMSC studies, this configuration is found to be definitely superior to the delta-wing design for both the stage-and-one-half and two-stage concepts with regard to propellant packaging efficiency, cg balancing capability and lightweight structure design. Use of the common configuration, in addition, reduced the scope of the study considerably without impairing the validity of results, and placed all designs on the same level of confidence with regard to the available extensive background of aerodynamic performance data. For the latter reason, detail modifications of the basic aerodynamic configuration were kept to a minimum.

For the one-and-one-half-stage configurations, the number of main engines was first established by preliminary analysis. With this number given, the vehicle characteristics, including payload capability were determined assuming a fixed liftoff thrust-to-weight ratio of 1.25 and, consequently, a fixed GLOW corresponding to the selected number of engines. This value for the thrust-to-weight ratio was selected in accordance with previous detailed one-and-one-half stage studies where it was found to be compatible with engine installation requirements and to provide both adequate stability during the ascent phase, and a sufficient margin for permitting adjustment to weight increases later in the development. For the 2-stage configurations, the liftoff thrust-to-weight ratio was considered a variable within limits of approximately 1.3:1 to 1.42:1, in accordance with the results of current Phase-B studies. Consequently, the vehicles could be designed for a constant payload weight (25 Klb for the 270 nm 55-deg inclination Reference Mission) by adjusting the number of booster engines and the resulting GLOW.

For the two-stage systems, since to perform the design of the booster was not within the scope of the study, the required booster characteristic for establishing orbiter interface and system performance were determined under a subtask by scaling the characteristics of a baseline booster. For this baseline, the High-Crossrange, Canard Booster configuration documented by MacDonnell-Douglas, ICD dated 18 December 1970, and by Mass Properties Status Report 7,

dated 12 January 1971, was selected. The assumptions made regarding the booster-orbiter attach points, other interface characteristics, and for the scaling of the booster for compliance with the vehicle system were the same as applied to the definition of the booster for the 2-Stage Baseline LS 400-7A system, reported in Subsection 3.2.2 of this report.

A rationale could be developed for assuming that the stepped-development approach would provide a final 2-stage system with superior overall performance than attainable by direct development. There are valid reasons for assuming that such improvement could result from the phasing of the orbiter and booster development under the sequential approach. Delay of the booster development after conclusion of the major phases of the orbiter development should permit incorporation of advanced technology obtained from the orbiter development as well as relaxation of growth/uncertainty factors, resulting in better booster design and overall system improvement. It was felt, however, that such considerations should not be introduced at this point in the analyses since quantitative data validating these assumptions could not be developed within the scope of this study.

4.3 ALTERNATIVE VEHICLE SYSTEM CHARACTERISTICS

4.3.1 Reference Configurations

1½-B Reference System Design. The design was obtained by updating the LS 200-5 configuration (Section 2.3) to the new requirements. The escalating effects of inert weight increases, resulting from

- change from the previously employed 415 klb sea level thrust main rocket engines to the new 530 klb engines, having a lower engine thrust/weight ratio and requiring increased system installation weight increments
- change from LH_2 to JP-4 fuel for the airbreathing engines
- addition of insulation to the droptanks for assuring intact reentry

made it necessary to increase the overall vehicle system size for maintaining the specified payload capability. A satisfactory solution was found by installing 9 main rocket engines in the orbiter which, generating a total sea level thrust of 4.77 Mlb increased the GLOW to 3.816 Mlb pound. Correspondingly, the orbiter ascent impulse propellant was increased from 192 klb to 239 klb.

In order to integrate these new design requirements, secondary design changes were required.

- The installation of the new main rocket engines moved the vehicle cg back beyond the permissible limit for maintaining flight stability, particularly in the hypersonic flight regime. This was counteracted by the addition of lower fuselage surface aft of the engines, moving the lower flap ft aft and increasing overall orbiter length from 151.75 ft to 159 ft. (1)
- As a consequence of the growth in vehicle dry weight, the number of jet engines was increased from five to six.
- The aft landing gear assembly was changed by using an outboard design, creating sufficient space for the installation of the increased tank volume.

(1) Including modification in lower flap configuration for the accommodation of six base-installed cruise engines.

In addition,

- The jet engines were moved from their central position in the LS 200-5 design vehicle base. This was made possible by the use of JP-4, which permitted the fuel tank to be moved forward to counterbalance the engine mass during the critical hypersonic flight phase.

Finally, modifications in primary and secondary vehicle structures, tanks and tank support structures, and minor modifications in aft fuselage surfaces and upper and lower flap configurations became necessary for the accommodation of the above changes into an integrated vehicle system.

For the subsystems, with the exception of the above listed changes in the main propulsion and airbreathing propulsion systems, and subsequent changes in tankage requirements for the orbit maneuver propulsion system, the same concepts and designs were used as for the LS 200-5 orbiter, with only adjustments made to match changed detail performance requirements.

The droptanks were adjusted without additional design effort by parametrically correcting dimensions to the new propellant load requirements.

The resulting vehicle configuration is shown in Fig. 4.3-1 for the orbiter (Model LS 200-7) and in Fig. 4.3-2 for the composite launch configuration.

The major design characteristics of the vehicle system are summarized in Table 4.3-1, and a summary of mass properties is presented in Section 4.9 of this report.

2-B Reference System Design. For this system, a new orbiter design was established. Previous Phase B Proposal studies were updated for compliance with changes in requirements and configuration, and for incorporating the experience gained in the design of the $1\frac{1}{2}$-stage orbiter. Most subsystem concepts and designs could also be adopted from the LS 200-5 design with only minor modifications.

Table 4.3-1

VEHICLE SYSTEMS CHARACTERISTICS SUMMARY

Program	Direct to 1-1/2-Stage	Conversion From 1-1/2-Stage to 2-Stage		Direct to 2-Stage
Vehicle Concept	1-1/2-Stage		2-Stage	
Vehicle Configuration	1-1/2-B	1-1/2-C	2-C	2-B
Model No.	LS 200-7	LS 200-8	LS 400-5	LS 400-6
1. ORBITER				
Size and Configuration				
Impulse Propellant, K lb				
Ascent	239	239	546	547
Orbit Maneuver	39	39	31	29
Cruise (Inc. Reserve)	12	12	9	9
Length, Total, ft	159	159	159	159
Planform Area, Actual, ft^2	6,846	6,846	6,846	6,846
Fuselage – Contour		96.8% same surface →		
Ramp Lower Surface	No	Yes	Yes	Yes
Upper Surface End Section	Designed to accommodate engine (9) installation		Designed to accommodate engine (2) installation	
Upper Flap	← "	" →	← "	" →
Lower Flap				
Configuration	← Same →			
Retractable	Yes	Yes	No	No
Fins	← Same →			
Rudders and Auxiliary Control Surfaces	← Same →			
Forward Structures				
Primary Structure Bulkheads and Frames		Designed for common load envelope →		
Tanks and Tank Attach Structures	← Same →		← Same →	

Table 4.3-1 (Cont'd)

Vehicle Configuration	1-1/2-B	1-1/2-C	2-C	2-B
Aft Structures				
Basic Structure		Designed for module exchange		
Thrust Structure	← For 9 main engines →		← For 2 Main Engines →	
Carry Through Structure		← Same →		
Upper Surface	Heavy aft section for reacting point loads of thrust structure attachments into skin		Lighter aft section Extended skirt	
Lower Surface – Basic		Designed for Convertibility		
Flap Track Assembly	Integral	Added	No	No
Booster Attach Carry-Through Structure	No	No	Added	Integral
Booster Attach Fitting	No	No	Added	Integral
Droptank Attach Fittings	Integral	Added	No	No
Fin Support Structure		Designed for load envelope		
Jet Engine Mounts	For 6 engines	For 4 to 6 engines		For 4 engines
Landing Gear Mounts		Designed for common envelope		
Base Heat Shield	For 9 engines and specific thrust structure		For 2 engines and specific thrust structure	
Main Propulsion System				
Engine No.				
Total	9	9	2	2
Gimballed	5	5	2	2
Post Staging	2 of 3	2 of 3	2	2
Engine – Type	← ICD 13M15000B powerhead →			
Expansion Ratio	53:1	53:1	150:1	150:1
Nozzle Type	← Fixed →		← Retractable →	
Manifolding	← Same →		← Same →	

Table 4.3-1 (Cont'd)

Vehicle Configuration	1-1/2-B	1-1/2-C	2-C	2-B
Orbit-Maneuver Propulsion System				
Engine Type	RL10A-3-3A			
Number – Installed	2	2	2	2
– Used	1	1	1	1
Attitude Control Propulsion System				
Type	High pressure GO_2 and GH_2			
Thrusters				
Pressure, psia	250			
Number	39	39	34	34
Installation	Same			
Cruise Propulsion System				
Fuel	JP-4			
Jet Engines				
Number	6	6	4	4
Location	Vehicle base			
Inlet Duct Locations	Upper vehicle surface			
	Combined with droptank Attach doors			
		Same location		
Thermal Protection System				
Type				
Lower Surfaces	LI-1500			
Upper Surfaces	Titanium skin			
Design Condition				
Cruise Range, nm	1100			
Planform Loading	1-1/2-Stage		2-Stage	
Trajectory Temperature Limit, °F	2300	Lower		2300

Table 4.3-1 (Cont'd)

Vehicle Configuration	1-1/2-B	1-1/2-C	2-C	2-B
Landing Gear				
Design Condition				
Landing Weight	← 1-1/2-Stage →			2-Stage
CG Locations		← Envelope →		
Avionics				
Hardware		Minor modification required		
Software	← 1-1/2-Stage →		2-Stage →	
Other Subsystems	← Basically unchanged →			
Weights, K lb				
Liftoff	628	628	844	834
Dry	291	300	212	204
2. DROPTANKS				
Propellant Weight, M lb	3.065	3.065		
Liftoff Weight, M lb	3.188	3.188		
3. BOOSTER				
Number of Engines			11	10
Propellant Weights, M lb			2.713	2.595
Liftoff Weight, M lb			3.364	3.209
4. LAUNCH CONFIGURATION				
Length, ft	189	189	258	258
GLOW, M lb	3.82	3.82	4.21	4.04
Thrust/Weight Ratio	1.25	1.25	1.44	1.36
Staging Velocity (Rel.), ft/sec	17,900	17,900	9.400	9,200

The principal changes in vehicle design from the $1\frac{1}{2}$-stage configuration are caused by

- The reduction in number of main rocket engines from nine to two
- The requirement for the accommodation of an ascent propellant load increased by about 130 percent
- The change in interface requirements resulting from the replacement of the droptank module by the booster

Additional primary changes in the vehicle system were introduced by

- The change in main rocket engine expansion ratio from 53:1 to 150:1, and
- The requirement to start the main rocket engines during flight at a high altitude environment

Secondary changes, made necessary, or possible by the primary changes include changes in vehicle arrangement, primary and secondary structures, number of jet engines, and aero surface configuration details. In addition, the capability of the ACPS had to be increased to provide roll control for the one-engine-out operation mode of the main propulsion system in order to satisfy fail operation and abort-through-orbit requirements.

The increase in propellant load could be accommodated in the orbiter without changing the vehicle envelope by moving the remaining main rocket engines by approximately 20 ft aft. This was made possible without impairing the vehicle cg balance by the large reduction in the number of the main rocket engines. The resulting gain in length inside the vehicle in a region of maximal cross section provided the required space for the installation of the additional tank volume. It is realized that this approach of keeping the geometric dimensions of the 2-stage orbiter the same as for the $1\frac{1}{2}$-stage orbiter and sizing the entire vehicle system around the maximal propellant load which can be stored in this orbiter configuration is fundamentally not bound to result in an optimally sized vehicle system. However, in comparing the attained propellant capacity with desirable values obtained from parametric sizing studies, and the

attained vehicle performance with that quoted for current Phase B designs, the conclusion was drawn that the assumed orbiter size is close to the optimum and that therefore final adjustments to the results of more rigorous analyses would have an insignificant effect on the results and conclusions of this study.

The characteristics of the orbiter tankage and primary structure designs are the same as for the 2-Stage Baseline LS 400-7A. These were discussed in detail in Section 3.2 of this report and are not repeated here.

Resulting from these changes, the primary and secondary structures of the 2-B orbiter vary considerably from those of the $1\frac{1}{2}$-B configuration in the rear section of the vehicle (approximately from the aft bulkhead of the payload bay). Major differences occur in design, and location of the thrust structure; design and location of the upper surface (requiring for the one-and-one-half-stage configuration a heavy aft section leading the point-loads of the thrust structure attachments into the upper fuselage skin, and for the 2-stage configuration an extension covering the aft mounted engines); design of the base heat shield, accommodating different thrust structure contour; removal at the lower surface of lower flap track assembly since the aft movement of the main engines and their longer nozzle skirts remove the plume impingement interference encountered with the $1\frac{1}{2}$-stage design; incorporation of forward booster attach structures and reinforcements to the lower surface for carrying the longitudinal interface load into the thrust structure.

Affecting the fuselage configuration over its entire length, a ramp was added to the lower surface since, at the time this study was conducted, it was deemed advisable to improve the subsonic trim capability of the configuration for the condition of extreme forward cg position occurring at landing with a large payload and the jet engines removed. Otherwise, the entire orbiter section forward of the aft payload bay bulkhead is nearly identical with that of the $1\frac{1}{2}$-B design with regard to both internal arrangement and structural design, the latter only incorporating modifications for the accommodation of relatively insignificant changes in design loads.

Resulting from the substantial decrease in inert weight of the orbiter (a consequence of the removal of seven main rocket engines without increasing vehicle size), the number of cruise engines could be reduced from six to four. It was also determined by preliminary analysis that the reduction in vehicle inertia associated with the removal of the rocket engines would permit decrease of the number of ACP thrusters from 39 to 34 without impairing the full redundancy required for fail operational and fail safe operation.

With the exception of the discussed changes, the subsystems of the 2-stage orbiter are identical in concept and design with those of the $1\frac{1}{2}$-stage vehicle, with only adjustments made for changes in detail design requirements.

The resulting configuration is shown in Fig. 4.3-3 for the orbiter (Model LS 400-6) and in Fig. 4.3-4 for the composite launch configuration. (The drawings show six jet engines installed in the orbiter instead of the four finally selected.)

The major design characteristics of the vehicle system are summarized in Table 4.3-1, and a summary of mass properties is presented in Section 4.9 of this report.

4.3.2 Convertible System Configurations

The two vehicle configurations and systems, 1½-C and 2-C were obtained by modifications of the corresponding reference 1½-B and 2-B systems, made necessary by the introduction of features increasing their mutual commonality for reducing the development and cost for the conversion from the 1½-stage to the 2-stage concept. In this regard, the guideline was followed that:

- The aeroshape configuration should remain unchanged except for secondary surfaces
- The entire design forward of the payload-bay rear-end station should remain basically unchanged
- The changes should primarily be driven into the tankage, engine installation, and affected rear-end structures
- The performance of the final 2-C system should not be significantly penalized in comparison with the "uncompromised" 2-B configuration established for direct development. For the interim 1½-C configuration, larger performance losses in comparison with the "uncompromised" 1½-B configuration could be accepted as long as substantial performance potential for all Level 1 Missions was maintained.

The major design characteristics of the two resulting vehicle systems are summarized and compared with those of the corresponding B-configurations in Table 4.3-1. For the 1½-C configuration, the orbiter (Model LS 200-8) is shown in Fig. 4.3-5 and the composite launch configuration in Fig. 4.3-6.

For the 2-C configuration, the orbiter (Model LS 400-5), as well as the composite launch configuration, are adequately represented by Figs. 4.3-3 and 4.3-4 showing the 2-B configuration.

Summaries of the mass properties of both 1½-C and 2-C vehicle systems are presented in Section 4.9 of this report.

The rationale behind the decisions leading to the establishment of more important design characteristics and resulting differences affecting the desirable commonality between the C-configurations (enhancing low conversion cost) and between the corresponding B- and C-configurations (indicating potential performance penalties for the convertible systems) is briefly discussed in the following.

Aero Configuration Shape and Contour. For the four vehicle alternatives, the aero shape and contour are only insignificantly different. The aerodynamic surfaces remain basically unchanged, and only very small changes occur at the aft end of the fuselage as required by the different main engine location. One of the most significant aspects of this near-complete commonality is that it places the same level of confidence on the predicted aerodynamic characteristics of all the configurations. It is also a major factor in reducing the tooling costs associated with the conversion of the 1 1/2-C configuration into the 2-C configuration.

With the 2-stage configurations, the addition of a 3-deg body ramp was considered desirable for maintaining subsonic trim capability with the most forward cg location associated with this configuration. This design is also used with the 1 1/2-C configuration in order to maintain commonality of fuselage contour and structures.

Upper Body Surface and Flaps. The upper surface is pulled back farther with the 2-stage configurations in order to provide adequate thermal protection to the aft-located engines. Resulting from recent wind tunnel tests, the upper flaps also were changed to a rectangular planform in order to improve their effectiveness and to reduce roll-yaw coupling. Differences between the 1 1/2-stage and 2-stage designs are due to different contour fairing requirements around the respective 9- and 2-engine installations.

Lower Flap and Elevons. Configuration and surface area are basically the same for the four vehicle alternatives. However, in comparison with the LS 200-5 configuration, the width of the flap was reduced (with corresponding increase in length) to accommodate the installation of jet engines at the vehicle base. A major difference exists between the 1 1/2-stage and 2-stage designs because of the requirement to retract the flap in the 1 1/2-stage orbiter during engine operation. Since this involves substantial complication and weight penalty, this capability is not maintained in the 2-C configuration.

Fins, Rudders, and Auxiliary Aero Surfaces. The contour and total surface of this assembly remain unchanged from the LS 200-5 design for all four orbiter designs. However, for the 2-stage configurations and also for the 1-1/2-C vehicle, the areas of the rudders and auxiliary surfaces were increased at the expense of the fin area.

Structural Arrangement and Design. The differences in the structural arrangement between the 1-1/2-stage and 2-stage versions are shown in Fig. 4.3-7. As with Figs. 4.3-3 and 4.3-4, the drawing shows 6 jet engines installed in the two-stage version while finally their number was reduced to 4.

Forward Section. For the entire orbiter section forward of Fuselage Section 1271, the only major differences occur in the tankage. The entire arrangement and contour are identical with the 1½-stage and 2-stage configurations. Since differences in line loads are small, a common basic fuselage structure design for the common load envelope can be used, imposing only a small weight penalty in the 2-C design. The only other compromises on the structure of the C-configurations are that the frames are internally contoured to accomodate both the larger tanks in the 2-C vehicle and the large oxygen transfer lines used in the 1½-C vehicle.

Aft Section. The large dissimilarity in design requirments on the aft structures was discussed in detail in Section 4.3.1. Consequently, a considerable difference exists in basic and secondary structures between the 1½-stage and 2-stage configurations. However, still a substantial degree of commonality can be incorporated in the C-configurations by employing a common basic structure (carry-through structures and lower surface structure including landing gear and jet engine attach points and inlet ducts, fin support structures), which is designed for permitting incorporation of the required modifications by the exchange of major structural subassemblies.

Main Propulsion. Extensive and unavoidable differences exist between the 1 1/2-stage and 2-stage designs resulting from the different number and expansion ratio of the engines. However, with regard to the cost incurred for convertibility, the common use of the same engine powerhead, which is the highest development cost item is highly significant.

Airbreathing Propulsion. The change from LH_2 to JP-4 fuel for the airbreathing engines made it possible to counterbalance aft-mounted jet engines by the mass of their fuel for the critical hypersonic cg location condition with both 1 1/2-stage and 2-stage designs. Consequently, the functionally and operationally more satisfactory installation of the jet engines at the vehicle base was selected for all four vehicle alternatives. This arrangement, however, imposes the constraint that the number of jet engines can be changed only in increments of two engines.

Orbit Maneuvering Propulsion. The concept used for the LS 200-5 configuration, using a separate tank system (also storing the ACPS propellants) and two RL10A-3A-3A engines (providing full redundancy since only one engine is used for normal operation), is also adopted for the four orbiter alternatives, the only difference being in that tank sizes are changed in accordance with the variation in propellant load requirements.

Attitude Control Propulsion. For all four alternatives the system concept using high-pressure LO_2 and LH_2 gas obtained by the conversion of the propellants stored in liquid form in the OMPS tanks remains unchanged from the one used with the LS 200-5 configuration. Change to a more advanced system using even higher pressures and also utilizing the turbine exhaust energy for impulse generation, as currently proposed for Phase-B developments, was not considered advantageous with vehicle systems which uses a separate, high performance system for satisfying the large majority of the maneuvering ΔV requirements.

The only difference between the $1\frac{1}{2}$-stage and 2-stage systems is that for the latter five of the 39 thrusters used in the $1\frac{1}{2}$-stage orbiters are removed. This, however, does not require any significant changes in the arrangement, or installation (including manifolding) of the remaining 34 thrusters.

Thermal Protection System. All systems are designed for 1,100 nm hypersonic crossrange capability.

With the convertible configurations, the design of the C-2 vehicle is penalized by $1\frac{1}{2}$-C requirements in two ways:

(1) The system is designed for the higher planform loading of the $1\frac{1}{2}$-stage vehicle.

(2) The system is designed for a reentry trajectory resulting in lower surface temperatures than assumed for the LS200-5 design, and because of the resulting increased exposure duration to high temperature environment, the insulation thickness is increased. This requirement stems from the introduction of negative body camber in both C-configurations which reduces the hypersonic pitch stability in the $1\frac{1}{2}$-C configuration (far aft cg location). For maintaining trim, increased flap deflections would become necessary with the surface temperature exceeding the assumed 2300°F limit with the trajectory flown with the LS 200-5 configuration The possibility

of coping with this problem by using ablator insulation on the flap of the 1½-C configuration was discarded as not compatible with the minimum refurbishment requirement of the space shuttle concept. Consequently, the situation was remedied by flying a lower-temperature trajectory.

Landing Gear. All configurations use a tricycle landing gear. The length of the main gear remains unchanged. The forward gear, however, is lengthened with the configurations displaying a body ramp.

For the baseline 1 1/2-stage, a design was selected in which the wheel wells of the main gear are arranged far outboard in the vicinity of the fuselage leading edge. With this, a high degree of stability is obtained. For the 2-stage designs, a narrower arrangement, but still providing adequate stability was selected which is more compatible with the installation of the large diameter tankage required with these applications. This design was also maintained with the 1 1/2-C configuration for reasons of commonality.

Other design considerations (cg location and landing weight) cause additional differences between the 1 1/2-stage and 2-stage design conditions. A common, compromised design is used for both convertible orbiters, resulting in weight penalties with both design.

Avionics. Only a minor modification of hardware is required for the conversion from the 1 1/2-stage to the 2-stage orbiter. The introduction of the booster module, however, results in a substantial change in software requirements.

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LIMITED CALENDAR LIFE	LIMITED OPERATING LIFE

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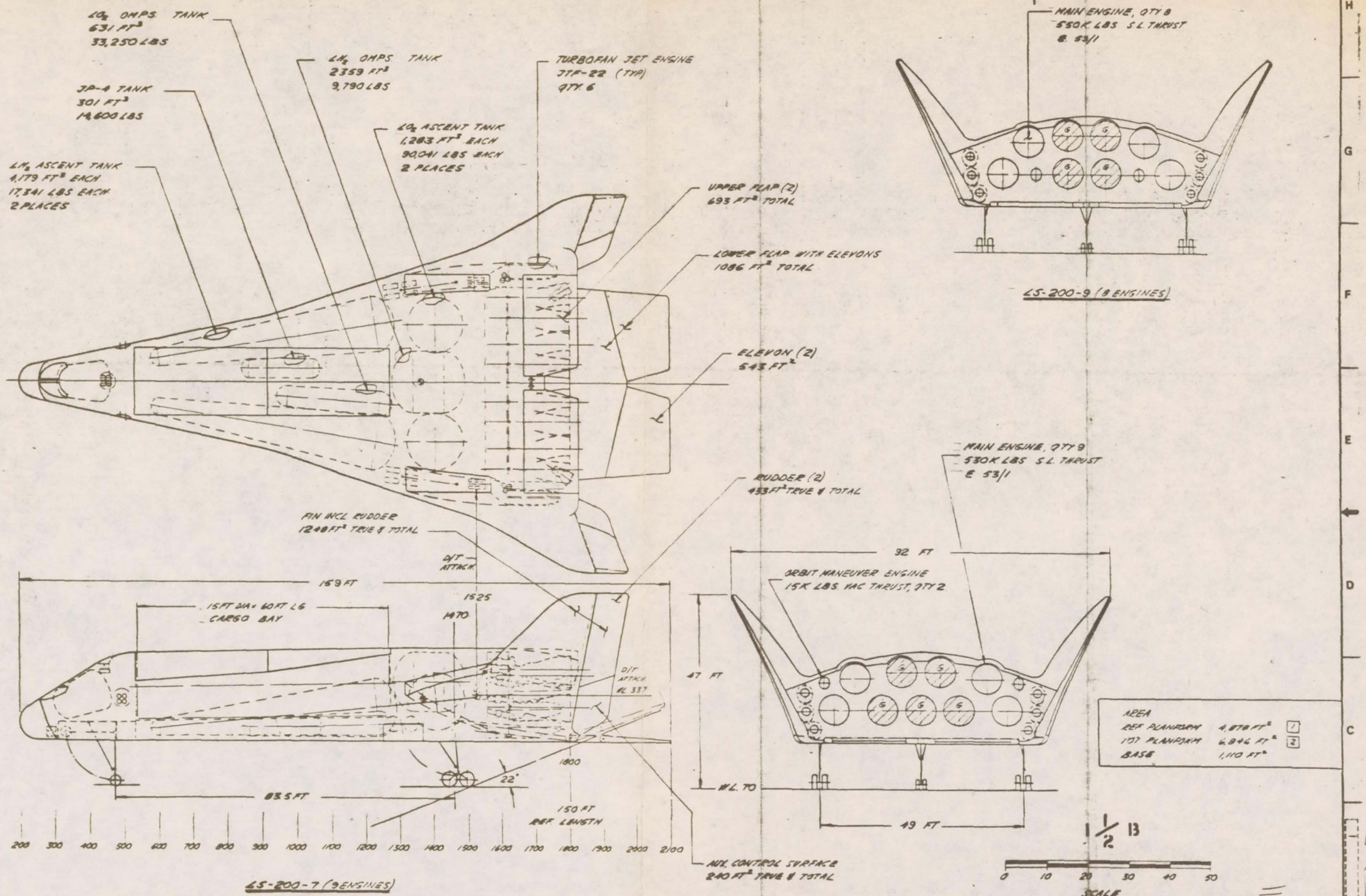

AREA		
REF. PLANFORM	4,870 FT2	1
TOT. PLANFORM	6,846 FT2	2
BASE	1,110 FT2	

1 1/2 B

0 10 20 30 40 50

SCALE

NOTES:

1. BASED ON TRIANGULAR AREA WITH APEX AT F.S 0 BASE AT F.S 1800 AND ALTITUDE OF 150 FT (REF. LENGTH)
2. BASED ON AREA OF TOTAL ORBITER AS SHOWN IN PLAN VIEW

Fig. 4.3-1

QTY REQD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO

PARTS LIST

UNLESS OTHERWISE SPECIFIED DIMS ARE IN INCHES TOLERANCES ON: FRACTIONS ± 1/16; DECIMALS: .X ± .1, .XX ± .03, .XXX ± .010; ANGLES ± 2 DEG	DATE	2-6-71	LOCKHEED MISSILES & SPACE COMPANY, A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION, SUNNYVALE, CALIFORNIA
	DWN	KARL TOLLO	
	APVD	M. HAWKING	GENERAL ARRANGEMENT (1) (2)
	APVD		DELTA 3001 - STAGE 1 1/2
	ENGRG		MODEL LS-200-7 & 9
	CHKR		
	APVD		SIZE E; CODE IDENT 14447; DRAWING NO SKT 100038; REV -
	CONTR		SCALE 1/2; SHEET
	APVD		
	CCA/CI/PD		

NEXT ASSY | USED ON | APPLICATION

4.3-17

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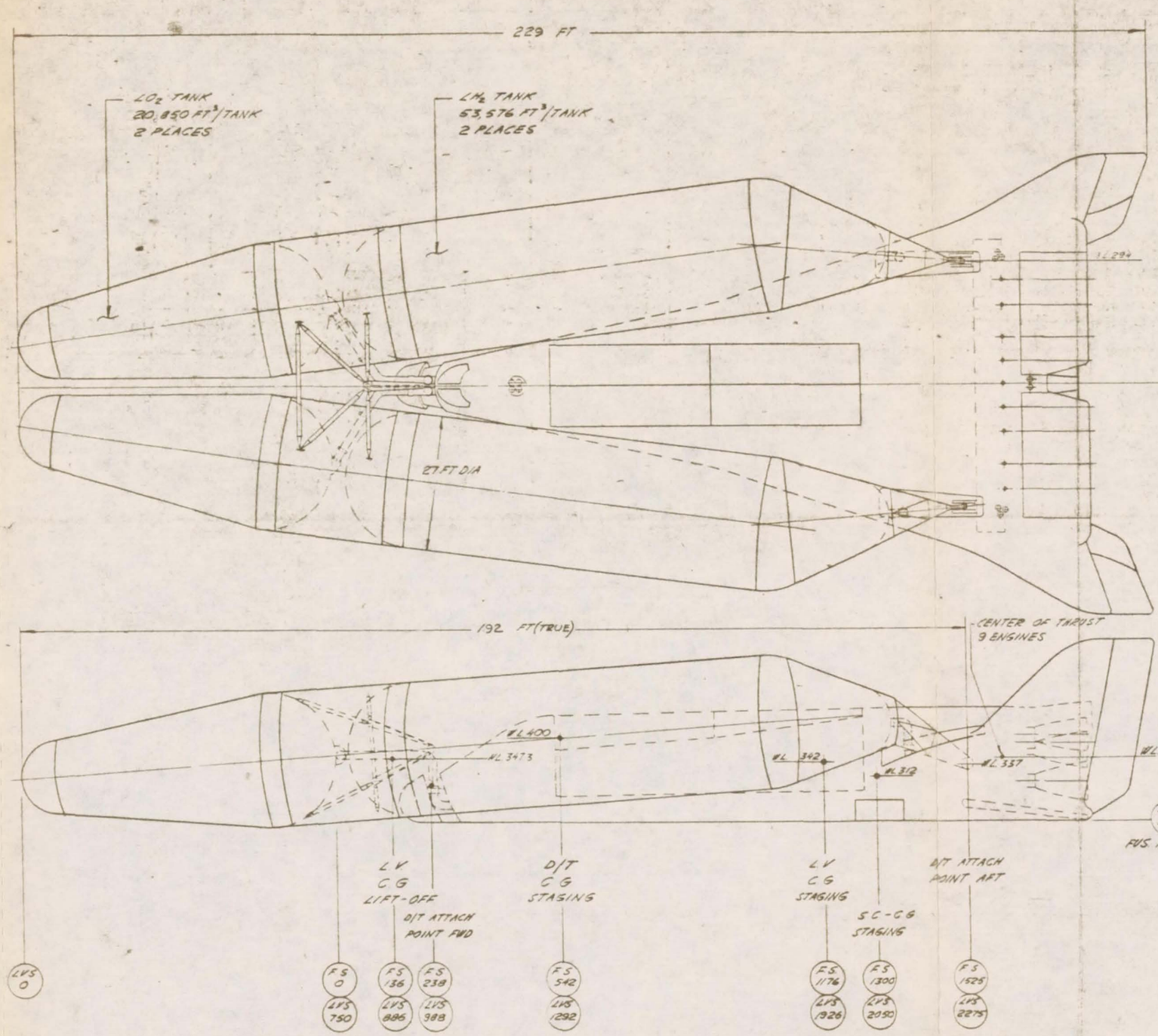
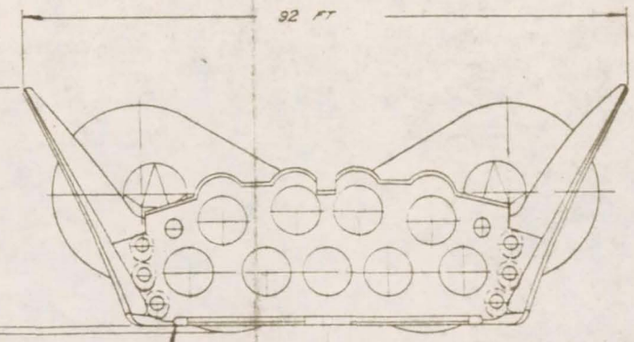
LIMITED CALENDAR LIFE	LIMITED OPERATING LIFE

REVISIONS			
ZONE	LTR	DESCRIPTION	DATE	APVD





ORBITER
MODEL LS-200-7#9
REF. DWG. SKT 100038

1 1/2 B

0 10 20 30 40 50 FT
SCALE

NOTES:
1. C G LOCATIONS ARE PRELIMINARY ESTIMATES.

FIG. 4.3-2

QTY REQD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO

PARTS LIST

INTERPRET DWG PER	UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES TOLERANCES ON FRACTIONS = ± 1/16 DECIMALS: .X = ± .1, .XX = ± .03, .XXX = ± .010 ANGLES = ± 2 DEG
NEXT ASSY / USED ON	CONTR
APPLICATION	CCA/CI/FO

DATE	2-10-71
DWN	KARL TOLLO
APVD	
APVD	
ENGRG	
CHKR	
APVD	
APVD	

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

LAUNCH VEHICLE (1&2)
DELTA BODY STAGE 1/2
MODEL LS-200-7#9

SIZE	CODE IDENT	DRAWING NO	REV
E	06997	SKT 100043	

SCALE 1/100 SHEET

4.3-19

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LIMITED CALENDAR LIFE	LIMITED OPERATING LIFE

REVISIONS			
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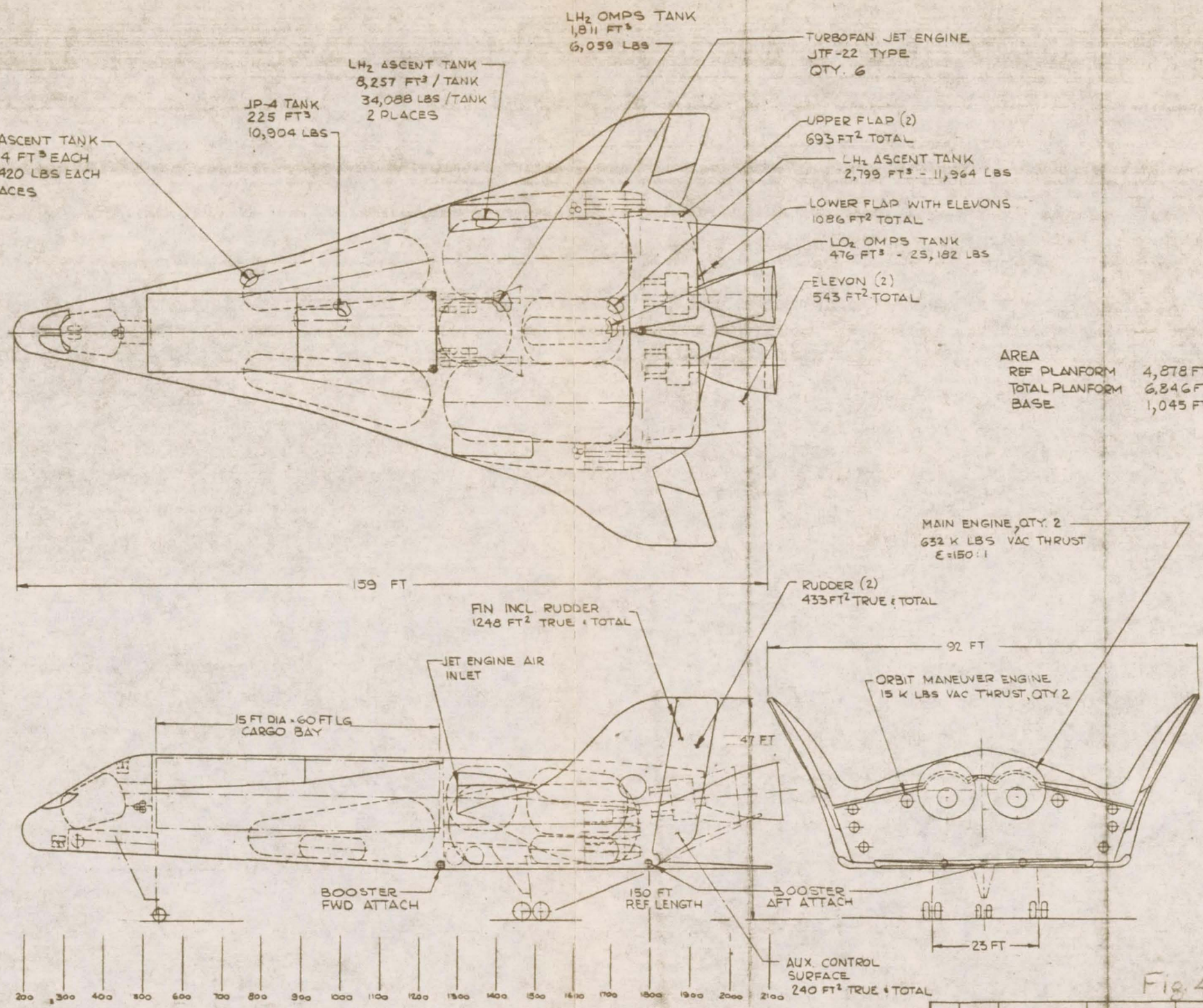

AREA

REF PLANFORM	4,878 FT²
TOTAL PLANFORM	6,846 FT²
BASE	1,045 FT²

Fig. 4.3-3

QTY REQD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO

PARTS LIST

INTERPRET DWG PER	UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES. TOLERANCES ON: FRACTIONS = ± 1/16; DECIMALS: .X = ± .1, .XX = ± .03, .XXX = ± .010; ANGLES = ± 2 DEG	DATE 7-17-71	LOCKHEED MISSILES & SPACE COMPANY, A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION, SUNNYVALE, CALIFORNIA
		DWN H. Schmidt	
		APVD Gen. Hardesty K.	GENERAL ARRANGEMENT (2B&2C)
		APVD	DELTA BOOY - 2 STAGE
		ENGRG	MODEL LS 400-5 & -6
		CHKR	
NEXT ASSY / USED ON	CONTR	APVD	SIZE E; CODE IDENT 06397; DRAWING NO SKS 100040; REV
APPLICATION	CCA/CI/FD	APVD	SCALE /--; SHEET /--

2 C
2 B

SKS 100040

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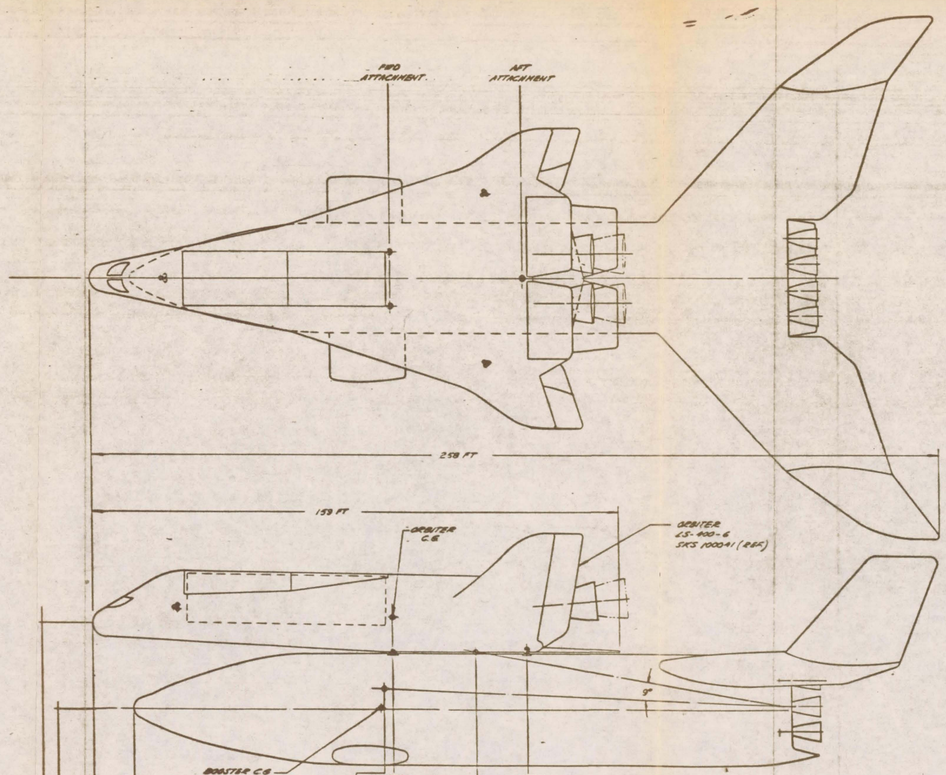

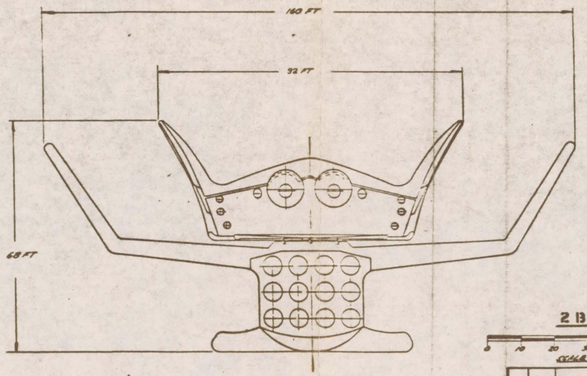

NOTE
C.G. LOCATIONS ARE PRELIMINARY ESTIMATES

2B

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Fig. 4.3-4

4.3-23

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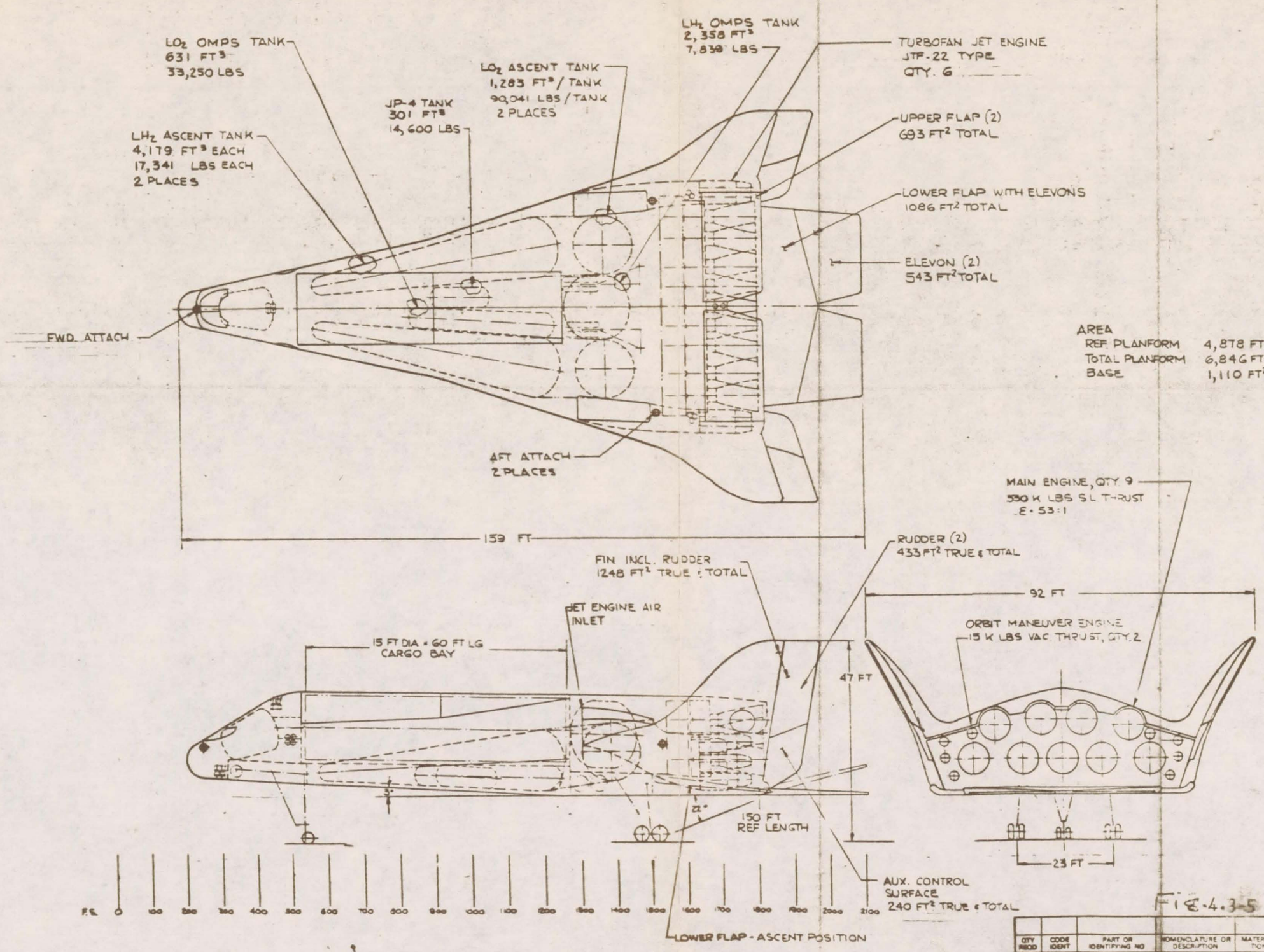

AREA

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TOTAL PLANFORM	6,846 FT2
BASE	1,110 FT2

FIG 4.3-5

1½C

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PARTS LIST

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LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

GENERAL ARRANGEMENT (1½C)
DELTA BODY - BASE 1/2
MODEL LS-200-B

SIZE E | CODE IDENT 18217 | DRAWING NO SKS100039 | REV

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4.3-25

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LIMITED CALENDAR LIFE | LIMITED OPERATING LIFE

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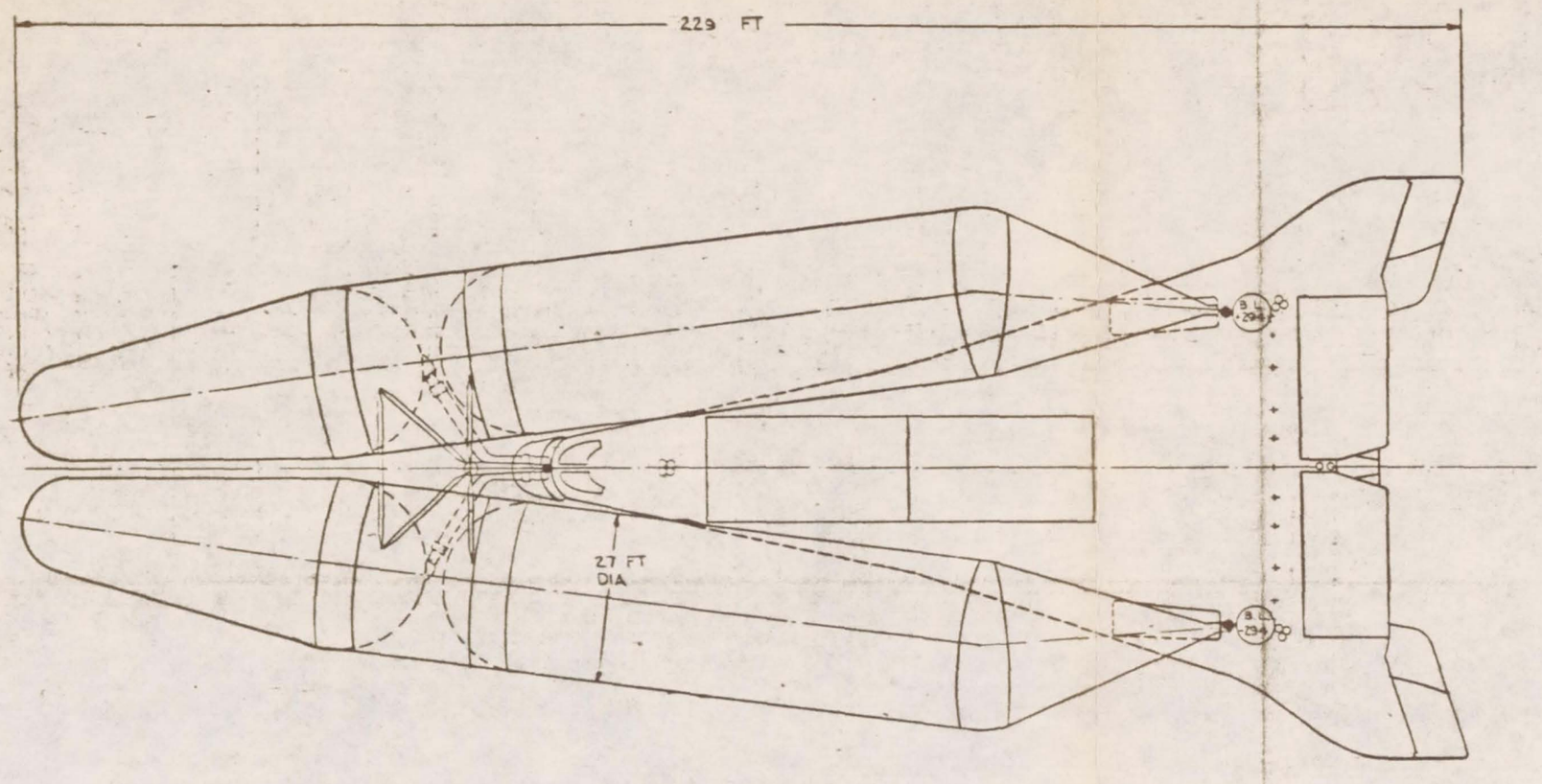

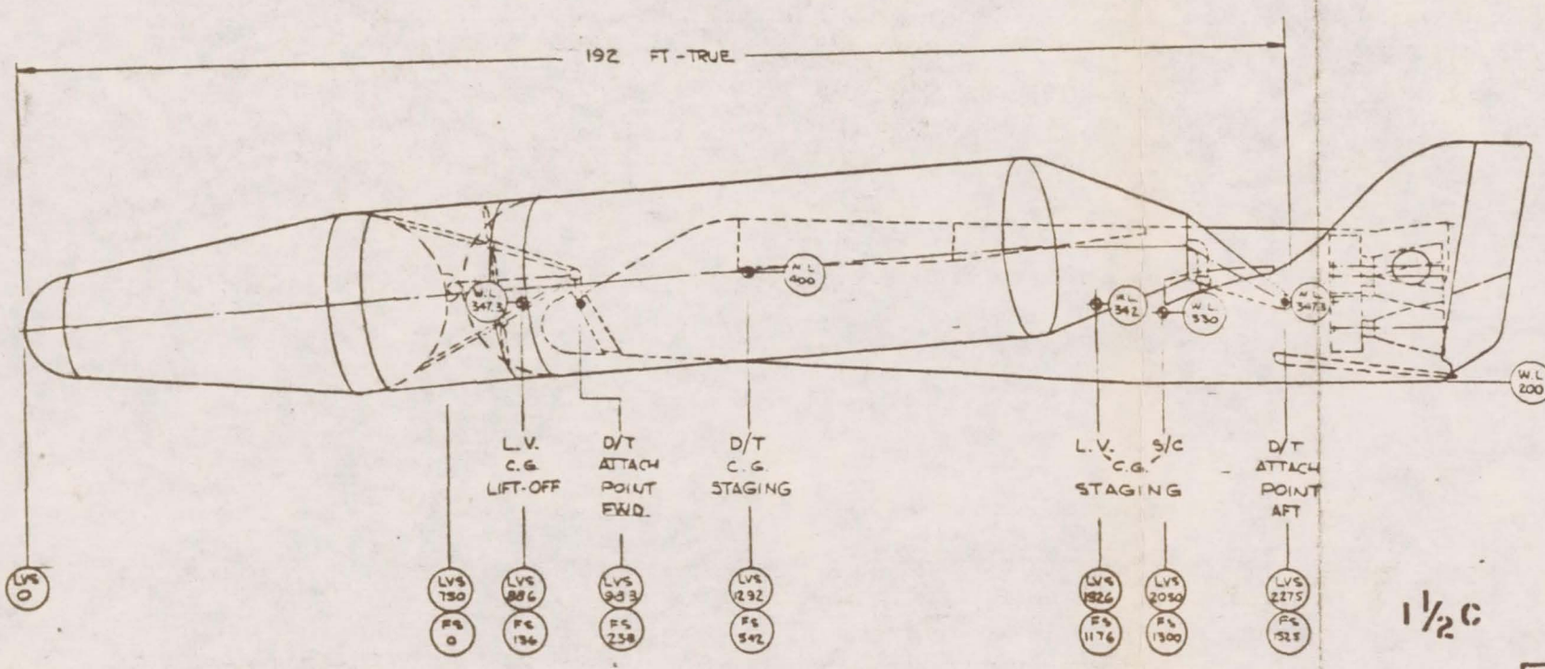

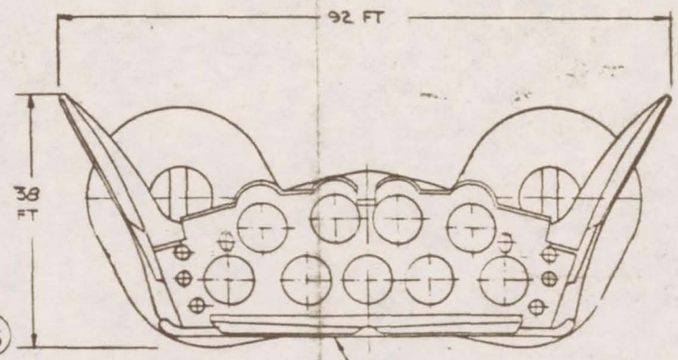

Note: C.G. LOCATIONS ARE PRELIMINARY ESTIMATES

1½C

Fig. 4.3-6

PARTS LIST

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INTERPRET DWG PER

UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES. TOLERANCES ON:
FRACTIONS = ± 1/16
DECIMALS: .X = ± .1
.XX = ± .03
.XXX = ± .010
ANGLES = ± 2 DEG

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CCA/CI/FD

DATE 2-5-71
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APVD
APVD
ENGRG
CHKR
APVD
APVD

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

LAUNCH VEHICLE (1½C)
DELTA BODY - STAGE 1½
MODEL LS-200-8

SIZE E | CODE IDENT 74897 | DRAWING NO SKS 100042 | REV L
SCALE | SHEET

4.3-27

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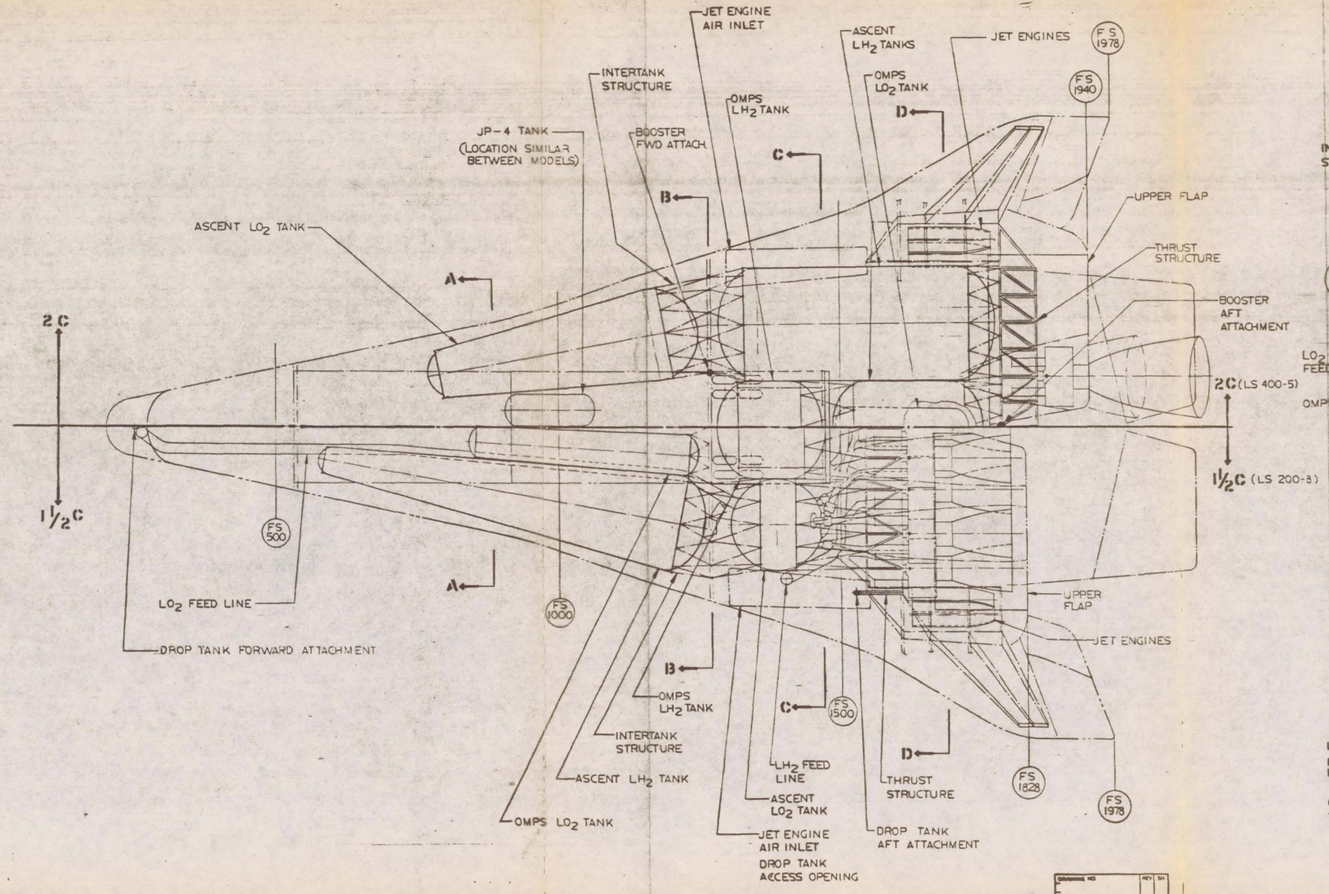

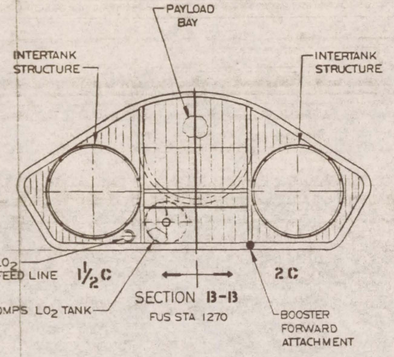

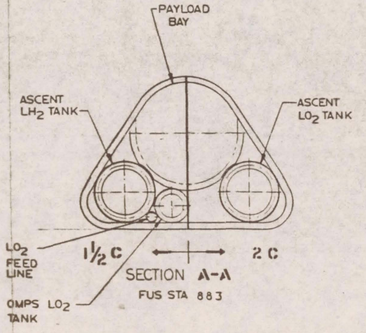

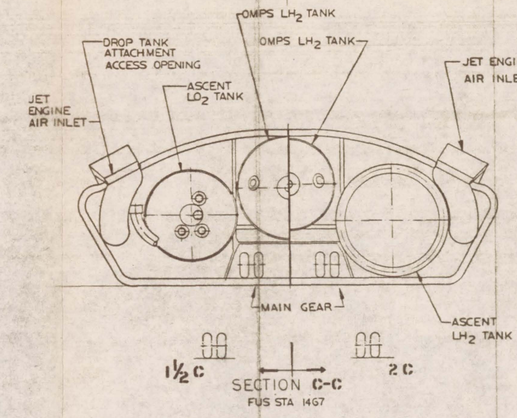

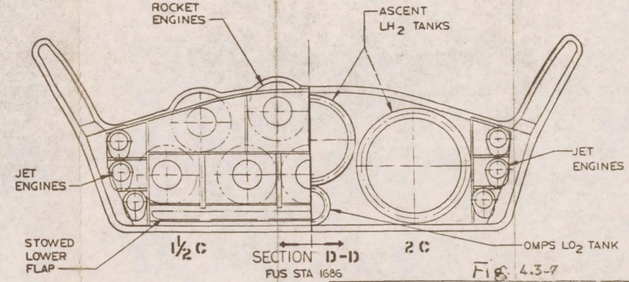

Fig. 4.3-7

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			PARTS LIST				

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		APVD	HUNTSVILLE, ALABAMA
	FRACTIONS = ± 1/16	APVD	COMPOSITE ARRANGEMENT
	DECIMALS .X = ± .1	ENGRG	DELTA BODY (2 STAGE/STAGE)
	.XX = ± .03	CHKR	MODELS LS 400-5/LS 200-8
	.XXX = ± .010	APVD	
	ANGLES = ± 2 DEG	APVD	SIZE CODE IDENT 00887 DRAWING NO SKS 100045 REV
NEXT ASSY	USED ON	CONTR	
APPLICATION		APVD	SCALE SHEET

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4.4 TECHNICAL CONVERSION FEASIBILITY

For the four alternative vehicle configurations, the design and performance characteristics are summarized in Table 4.4-1.

4.4.1 Mission Performance

The listed performance values reflect the assumed method of selecting a fixed liftoff thrust-to-weight ratio of 1.25 for the stage-and-one-half configurations. This makes the payload a variable. With the two-stage configurations, a fixed payload capability of 25,000 lb for the 270-nm, 55-deg inclination Reference Mission makes the liftoff thrust-to-weight as well as the GLOW variable. The quoted values for these characteristics are then obtained by selecting the number of booster engines such that the liftoff thrust-to-weight ratio is maintained within assumed permissble limits.

With the 1½-B configuration, as discussed in Section 4.3.1, the GLOW increased by 300K lb to 3.82 M lb in comparison with the LS 200-5 configuration, reflecting increases in orbiter liftoff and dry weight by 83,700 lb and 21,500 lb, respectively. The compromises made in the 1½-C orbiter design for reducing conversion costs to the 2-C design are reflected in an increase in dry weight of 9,000 lb in comparison with the reference 1½-B design, reducing the payload capability by an equal amount. This primarily affects the capability for the 270-nm, 55-deg inclination Reference Mission. The remaining capability of 13.5 Klb is considered adequate for the initial phase of the stepwise program.(1) Also, it could be increased beyond requirements by the removal of the jet engines. For the other Level 1 missions, full compatibility with the established requirements is still maintained.

With the two-stage configurations, the attained payload values demonstrate that the system defined by this study closely approaches the optimum for the specified capability. Very favorable gross liftoff weight values are

(1) See also addendum, page 4.4-5, quoting corrected payload capability of 17 klb for this mission.

obtained (4.04 Mlb for the reference 2-B design and 4.21 Mlb for the compromised 2-C design). They reflect the high volumetric and structural efficiency of the delta-body design. Also associated with these low GLOW values for the 2-B design is a reduction in the number of booster engines to 10 from the 12 used with the baseline MAC/DAC booster. An additional consequence of the efficient orbiter design is the relatively low-staging velocity which is bound to alleviate booster design requirements.

The weights penalty in the 2-C orbiter in comparison with the uncompromised 2-B orbiter amounts to 7,870 lb, resulting in an increase of GLOW by 170,000 lb. Concurrently, the number of booster engines has been increased from the 10 for the 2-B configuration to 11, since the liftoff thrust-to-weight ratio of 1.28 obtained with 10 engines is considered marginal.

A breakdown of the weight penalties imposed on the 2-C orbiter dry weight in comparison with the uncompromised 2-B design is shown in Table 4.4-2. The third column lists the relative weight penalty on the item in question, and the last column the resulting relative weight penalty on the entire dry weight of the orbiter. As seen, the total weight penalty corresponds to an increase in orbiter dry weight by 3.86 percent. It is interesting to note that the weight penalties from designing for maximal structural commonality – a primary consideration for reducing conversion production costs – amounts (including growth/uncertainty increment) to only 605 lb, or 0.3 percent of the total dry weight. If the common landing gear were replaced by different assemblies designed for the specific design requirements of the two vehicles, the total weight penalty could be reduced to 4,870 lb or only 2.4 percent of the total dry weight, which would reduce it to a rather unsignificant factor.

4.4.2 Flight Performance Characteristics

C.G. Balance Feasibility. Extreme forward and aft center-of-gravity locations are listed in Table 4.4-1 and quoted in fuselage stations. They can be converted to percentile values used in Sections 2.6 and 3.5 by relating them to a reference length of 1752 inches.

The extreme c.g. locations correspond to vehicle flight and load conditions defined as follows:

Condition	Payload (K lb)	ABES	Cruise Fuel
Extreme Aft C.G.			
Entry	0	In	In
Landing	0	In	Out
Extreme Forward C.G.			
Entry	40	Out	Out
Landing	40	Out	Out

Values shown for the extreme cg locations indicate that with each particular vehicle the magnitude of the cg shift between the extreme conditions stays within the capability for maintaining trim capability of each configuration. However, the extreme aft condition, obtained with the stage-and-one-half configurations, as well as the cg shift occurring among the two convertible C-configurations exceed the limits for trimmable configurations. Modifications in the aerodynamic surfaces of the two stage-and-one-half configurations will be required and, if commonality is necessary, modifications will be required also for the 2-C configuration. The effects of these modifications on vehicle sizes, payload capabilities, and conversion capability, are not expected to become sufficiently significant for changing the conclusions drawn from this study.

Maximum Lift-to-Drag Characteristics. The subsonic values listed are based on preliminary results of wind tunnel testing reported in Sections 2.6 and 3.5. These values show considerable improvement over those assumed previously for the LS 200-5 configuration.

Landing Characteristics. Landing characteristics of the two-stage configurations are comparable to those of space shuttle Phase B delta-wing configurations. Because of the increased landing weight, the similarly configured stage-and-one-half orbiter has higher landing speeds and attitudes. The listed speeds for both configurations are considered well within the limits of acceptable values inasmuch as lifting-body landing speed and attitude are bound to depend very markedly on pilot technique.

4.4.3 Addendum

As mentioned in Section 4.2, the study was concluded before completion of additional design studies on both the stage-and-one-half and two-stage orbiter configurations, and of wind tunnel test programs conducted to verify the aerodynamic characteristics of the delta-body configuration used for the orbiter in both systems.

From the now available final results of these design studies and test programs, it can be concluded with respect to the Conversion Study that:

a. A fully trimmable and stable stage-and-one-half orbiter configuration, using the assumed delta-body configuration, can be designed, satisfying the Level I Mission performance and operational requirements. This can be done without the necessity for increasing the launch configuration GLOW or for making any other modifications on the configurations, which would significantly affect operations or development and program costs.

b. The use of a ramp on the lower fuselage surface of the two-stage delta-body orbiter is no longer considered a requirement for assuring subsonic trim capability under all flight conditions.

The removal of the body ramp reduces the dry weight penalties of the convertible configurations, in comparison with the baseline configurations defined in this study, as follows:

- For the 1½-C and 2-C configurations by approximately 3,550 lb (~ 750 lb reduction in structural weight and ~2800 lb reduction in TPS weight for removing the requirement for flying a reduced-temperature reentry trajectory).
- For the 2-B configuration by approximately 750 lb (reduction in structural weight).

Consequently, the payload quoted above for the 1½-C configuration increases from 13.5 klb to 17.0 klb, and the GLOW values quoted for the 2-C and 2-B configurations reduce from 4.21Mlb and 4.04Mlb to 4.14Mlb and 4.024Mlb, respectively.

4.4.4 Conclusions

Space Shuttle vehicle systems can be designed and developed for a stepwise program approach by which an ultimate two-stage system is obtained by the conversion of an interim stage-and-one-half system. The primary reasons for the technical feasibility are:

1. The size, shape, and aerodynamic configuration of the orbiter using a delta-body configuration can be kept identical for the stage-and-one-half and two-stage versions.
2. An orbiter vehicle can be designed which will have a high degree of commonality between the interim stage-and-one-half and ultimate two-stage versions with regard to general arrangement, and primary and secondary structures. With the exception of the main propulsion system, the subsystems in both versions are nearly identical with only detail adjustments to changes in system requirements required.

3. The common size and aerodynamic configuration make it possible to reduce to a minimum additional flight testing for the development of the two-stage version.

4. In comparison with the two-stage system obtained by direct development, the performance penalty (expressed in GLOW increase) imposed on the two-stage system obtained under the conversion program cost, is insignificant.

5. The performance capability of the interim 1½-stage system is adequate for an initial phase of the Space Shuttle program. It is reduced from the Level I requirements only for the 270 nm, 55 deg inclination Reference Mission, and only when flown with the airbreather engines installed.

Table 4.4-1

PERFORMANCE CHARACTERISTICS SUMMARY

Program	Direct to 1-1/2-Stage	Conversion From 1-1/2-Stage to 2-Stage		Direct to 2-Stage
Vehicle Concept	1-1/2-Stage		2-Stage	
Vehicle Configuration	1-1/2-B	1-1/2-C	2-C	2-B
MODEL	LS 200-7	LS 200-8	LS 400-5	LS 400-6
MASS CHARACTERISTICS				
Weight, K lb(1)				
Orbiter – Dry	291	300	212	204
– Liftoff	628	628	844	834
Composite – Liftoff	3,820	3,820	4,210	4,040
Orbiter Extreme CG Fuselage Station, in.				
Aft – Entry	1,387	1,391	1,391	1,391
– Landing	1,405	1,409	1,375	1,375
Forward – Entry	1,330	1,335	1,282	1,282
– Landing	1,330	1,335	1,282	1,282
Planform Loading, (lb/ft^2)				
– Entry	851	851	659	625
– Landing	804	830	611	597
PAYLOAD CAPABILITY, K LB				
Design Mission, Due East				
Calculated Value(2)	94.0	84.6	74.5	74.2
Design Value	65.0	65.0	65.0	65.0
Reference Mission South Polar				
Calculated Value(2)	49.0	39.0	42.8	42.5
Design Value	40.0	40.0	40.0	40.0

Table 4.4-1 (Cont'd)

Vehicle Configuration	1-1/2-B	1-1/2-C	2-C	2-B
Reference Mission, 270-nm, 55-deg(3)				
Calculated Value	22.5	13.5	25.0	25.0
Design Value	25.0	25.0	25.0	25.0
FLIGHT PERFORMANCE CHARACTERISTICS				
Trimmed L/D_{Max}				
CG Max Aft – Entry	1.7	1.7	1.7	1.7
– Landing	5.85	5.85	5.85	5.85
CG Max Fwd – Entry	1.7	1.7	1.7	1.7
– Landing	5.7	5.7	5.45	5.45
Landing Speed, kn				
At 15 deg	180	180	155	155
At 22 deg(4)	150	150	130	130

NOTES: (1) For 270-nm Reference Mission
(2) ABES OUT
(3) ABES IN
(4) Tail Scrape

Table 4.4-2

2-C ORBITER WEIGHT PENALTY FOR COMMONALITY WITH 1-1/2-C ORBITER

Item	Weight, lb		Percent Penalty	
	2-B	2-C Increment	Item Weight	Total Dry Weight
Forward Structure				
Basic and Bulkheads	21,600	+250	1.2	0.123
Tanks and Attach	9,000	0	0	0
Aft Structure				
Basic	15,100	+300	2.0	0.147
Thrust Structure	4,100	0	0	0
Tanks and Attach	15,300	0	0	0
Fin Support	1,700	0	0	0
Base Heatshield	1,600	0	0	0
Jet Engine Support	1,100	0	0	0
Propulsion				
Main	43,800	0	0	0
ABPS	11,800	0	0	0
Inlet Ducts	700	0	0	0
TPS	36,100	+3,500	9.7	1.717
Landing Gear	8,900	+3,000	33.7	1.471
Other	16,300	+120	0.7	0.059
Growth/Uncertainty	16,800	+700	4.2	0.343
TOTAL DRY WEIGHT	203,900	+7,870		3.860

4.5 SYSTEM DESIGN

This system design was primarily concerned with the feasibility of converting a stage-and-one-half orbiter into an orbiter to be used in a fully-recoverable two-stage system. The design drawings that illustrate the approach to this conversion are as follows:

Fig. 4.5-1 General Arrangement ($1\frac{1}{2}$C) Delta-Body - Stage-and-Half Model LS 200-8, SKS-100039

Fig. 4.5-2 General Arrangement (2C) Delta-Body - Two-Stage Model LS 400-5, SKS-100040

Fig. 4.5-3 Launch Vehicle (2B) Delta-Body - Two-Stage Model LS 400-6, SKT-100044

Fig. 4.5-4 Launch Vehicle ($1\frac{1}{2}$C) Delta-Body - Stage-and-a-Half Model LS 200-8, SKS-100042

Fig. 4.5-5 Structural Arrangement Load-Path Concepts Study Baseline System (Two-Stage), SKG-100119

Fig. 4.5-6 Composite Arrangement Delta-Body (Two-Stage/Stage-and-a-Half) Models LS 400-5/LS 200-8, SKS 100045

Fig. 4.5-7 Structural Arrangement Load-Path Concepts Study Main Keel System, SKG-100115 (3 sheets)

Fig. 4.5-8 Baseline Thrust Structure Arrangement Delta-Body - Two-Stage Model LS 400-5, SKG-100118

Fig. 4.5-9 Volume Curve Delta-Body Orbiter Model LS 400-5, SKS-100046

The stage-and-one-half orbiter ($1\frac{1}{2}$C) conceived for this task is arranged as shown in Fig. 4.5-1. In this arrangement, the basic design approach was to compromise this orbiter for the two-stage version rather than the reverse if possible. This concept in general was the same as that of the $1\frac{1}{2}$-B system (baseline stage-and-one-half) except as follows:

a. Aerodynamic Shape and/or Geometry

- Incorporated the 3-degree lower-surface forward ramp

Landing Gear System

- Nose landing gear is longer because of ramp effect.
- Station-wise location of main landing gear is normal (worst aft center-of-gravity condition is dictated by stage-and-one-half orbiter), while buttline location of gear is compromised by two-stage location.

Figure 4.5-2 shows the converted two-stage orbiter (also previously discussed in Section 3.5). The difference (not shown) between the 2-B and 2-C systems would show-up only in the main landing gear location. The aft center-of-gravity location of the 2-B orbiter would determine this location, which would be more forward than that allowable for the convertible 2-C arrangement.

Launch vehicles associated with the conversion orbiters are shown in Fig. 4.5-3 (similar arrangement for 2-C) for the two-stage orbiter and in Fig. 4.5-4 for the convertible ($1\frac{1}{2}$C) orbiter. This arrangement differs from the $1\frac{1}{2}$B launch vehicle only with respect to the orbiter with its forward ramp resulting in a higher vertical location of the staging center-of-gravity.

Figure 4.5-5 is a structural arrangement of the baseline two-stage orbiter. This design indicates with the aid of isometric illustrations those primary structural areas effected by the convertibility concept; these areas are:

- Rocket-engine thrust structure
- Airframe-shell beams and thrust-structure attach points
- Upper-surface flaps
- Intertank strut frames
- Booster attach-fittings and backup structure
- Ascent tankage

A composite arrangement of both candidate orbiters is shown in Fig. 4.5-6. This drawing shows the relative positions of the major components of each system, including those structural areas considered for conversion. The packaging arrangement adopted for the stage-and-one-half ($1\frac{1}{2}$C) system was necessary to stay within the aft center-of-gravity limitation for this lifting-body concept. The two-stage (2-C) packaging is dominated by the propellant requirement; however, the arrangement of tankage is similar to the stage-and-one-half and a similar load path was assumed for the structural analysis. In addition, the two-stage orbiter provides for a required booster attachment system compatible with the presently known booster concept.

The major differences that effect the structural airframe between the two candidate concepts are:

a. Thrust Structure Area

Location of the $1\frac{1}{2}$C system is well forward of the 2-C system, thus requiring the thrust structure attach fittings to be in different airframe areas.

b. Airframe Shell

The spanwise aft shell area (top and bottom) acts as a beam and works in conjunction with the spanwise thrust structure beams to support the total air frame and propellant tanks.

- Loading intensity is higher on the $1\frac{1}{2}$C system
- Location is further forward on the $1\frac{1}{2}$C system
- Lower-surface flap system - because of the necessity to retract the lower-surface falp to an area directly under the rocket-engine thrust structure and attach fittings on the $1\frac{1}{2}$C, an additional surface, not the lower-fuselage shell, must be provided to accept the attach fittings.

c. Ascent Tankage Kick Loads

The kick loads are stabilized by the bulkhead located on each orbiter (payload bay aft end) with the 2-C system requiring a heavier bulkhead ring-frame to stabilize the effects of the forward-located ascent oxygen tanks.

d. Upper Surface Flaps

The flaps are attached directly behind the previously mentioned airframe spanwise shell beams; therefore, they are installed at different locations and, as a result, have different shapes both along contour lines and for different rocket engine clearances.

e. Booster Attachment System

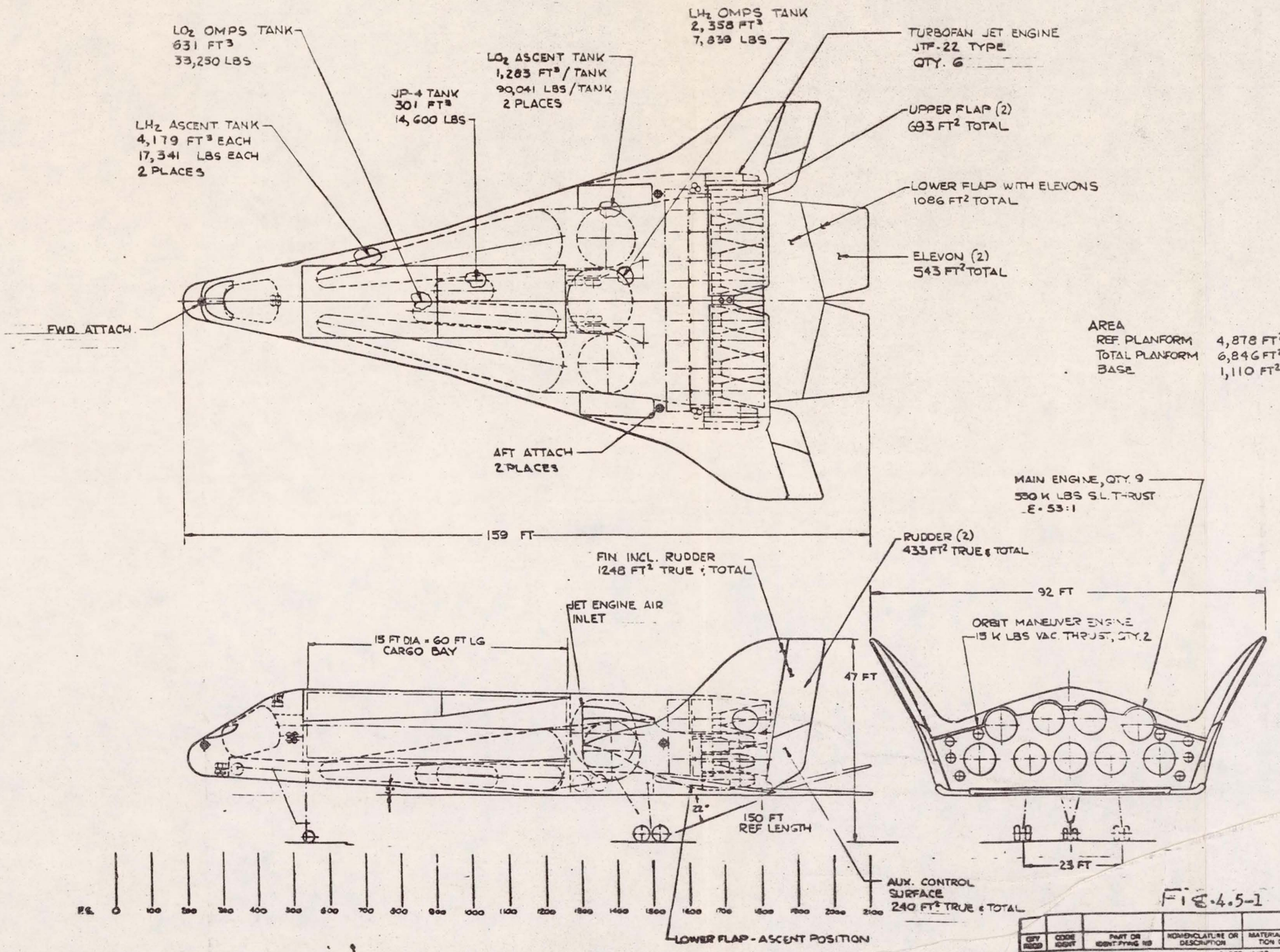
Because of the need for attach fittings and their chosen locations, the two-stage (2-C) orbiter requires the installation of specific longerons to backup the attachment loads as well as an aft fuselage lower-surface arrangement that will permit the transfer of orbiter loads to these fittings via the longerons system.

An alternate approach to the structural arrangement for both the convertible orbiter and converted orbiter is shown in Fig. 4.5-7. This drawing illustrates a concept in which two main keels (fore and aft bulkheads) are used as the structural backbone of each orbiter system. The keels provide the main load path between the orbiter (oxygen mass) and the booster attachments and between the orbiter (oxygen mass) and orbiter rocket engines during orbiter operation.

To provide a more detail description of the two-stage orbiter baseline thrust structure, an arrangement drawing showing the spanwise horizontal beam and vertical trusses necessary to support the rocket engines and the orbiter propellant tanks was made and is indicated as Fig. 4.5-8. Also shown on this

drawing are the booster attachments, the ascent tankage intertank-support kick-ring frame, and the base-mounted jet engines indicating the use of the three-fin spars as the front and rear engine-mount bulkheads and the aft spar/aft bulkhead forming the aft jet-engine-compartment seal.

Indications of the usable volumes for both candidate orbiters as well as their wetted areas are shown in Fig. 4.5-9.



AREA

REF. PLANFORM	4,878 FT2
TOTAL PLANFORM	6,846 FT2
BASE	1,110 FT2

FIG. 4.5-1

1½C

REVISIONS				
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ITEM NO	ZONE	MATERIAL SPECIFICATION	MATERIAL DESCRIPTION OR NOTE	NOMENCLATURE OR DESCRIPTION	PART OR IDENT. NO	CODE IDENT	QTY REQD

PARTS LIST

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DATE 2-17-71
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APVD
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CHKR
APVD
APVD
APVD

CONTR
CONTR NO

LOCKHEED MISSILES & SPACE COMPANY
A DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

GENERAL ARRANGEMENT (1½C)
DELTA BOOSTER - STAGE 1½
MODEL LS-299-E

SIZE E | CODE IDENT 32431 | DRAWING NO SKS 100030 | REV

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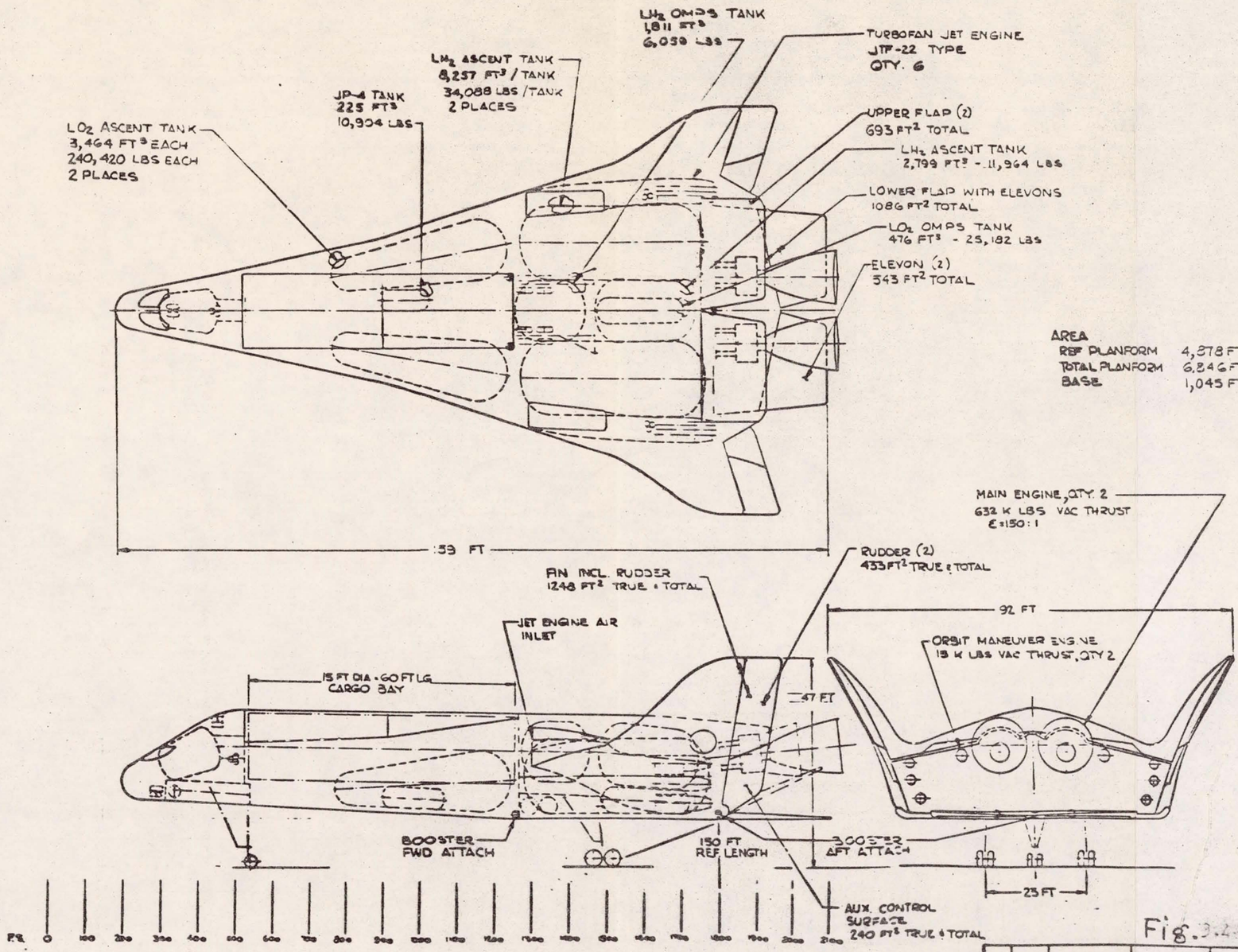

Fig. 4.5-2

2C
2B

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PARTS LIST

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ASSY USED ON	CONTR		GENERAL ARRANGEMENT (2B&2C) DELTA BODY - 2 STAGE MODEL LS 400-5 E-G
APPLICATION	CCA O FB		SIZE E CODE IDENT 06097 DRAWING NO SKS 100040 REV
			SCALE — SHEET

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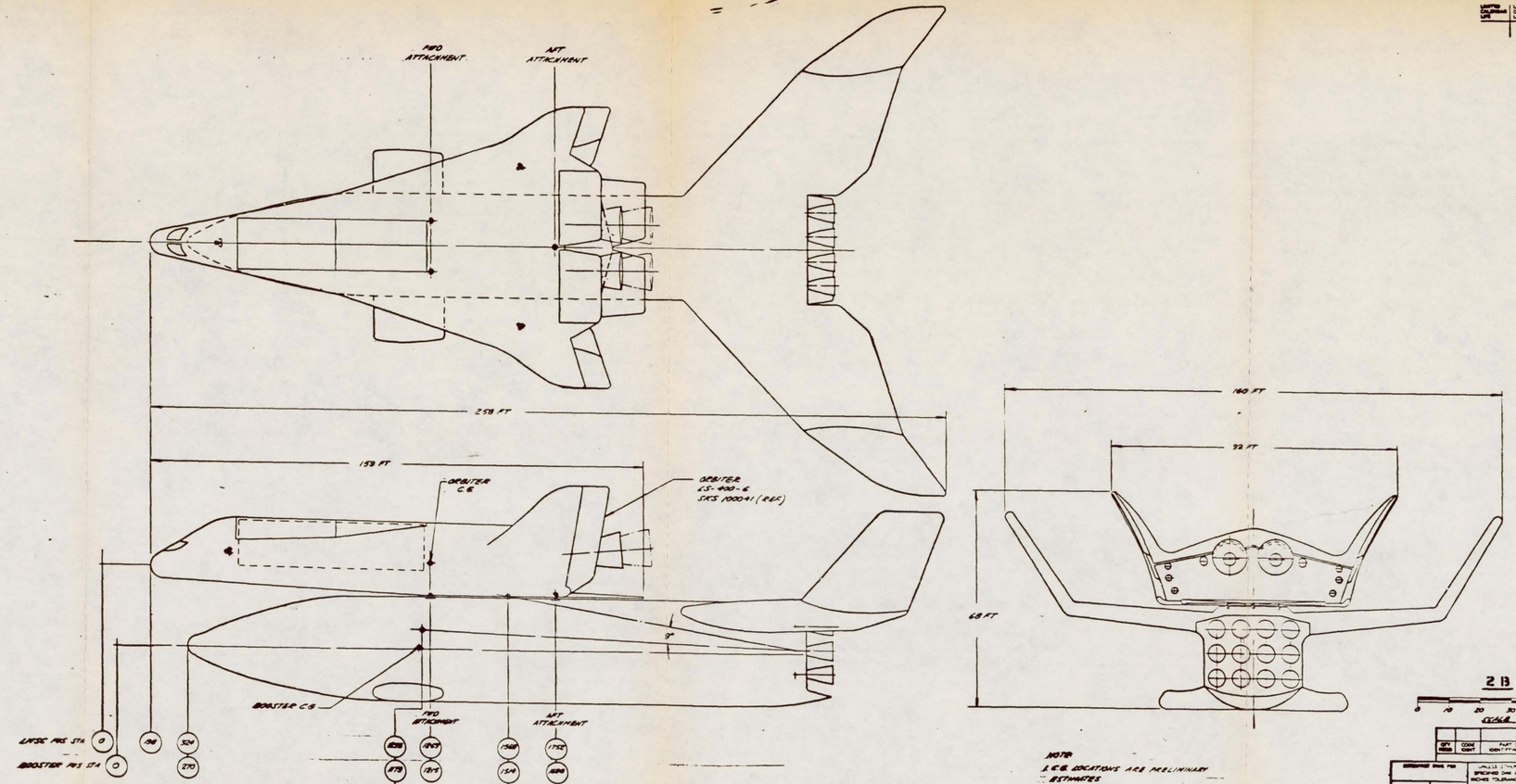
FWD ATTACHMENT
AFT ATTACHMENT
259 FT
159 FT
ORBITER C.G.
ORBITER
41-000-6
SKS 100041 (REF)
9°
BOOSTER C.G.
FWD ATTACHMENT
AFT ATTACHMENT
LAUNCH VEHICLE C.G.
STAGING C.G.
ORBITER FRS STA
BOOSTER FRS STA
0
298
504
270
1298
1304
1278
1284
1398
1404
1378
1384
140 FT
92 FT
68 FT
NOTE
C.G. LOCATIONS ARE PRELIMINARY ESTIMATES.
2 B
SCALE
0 10 20 30 40 50 FT

Fig. 4.5-3

4.5-11

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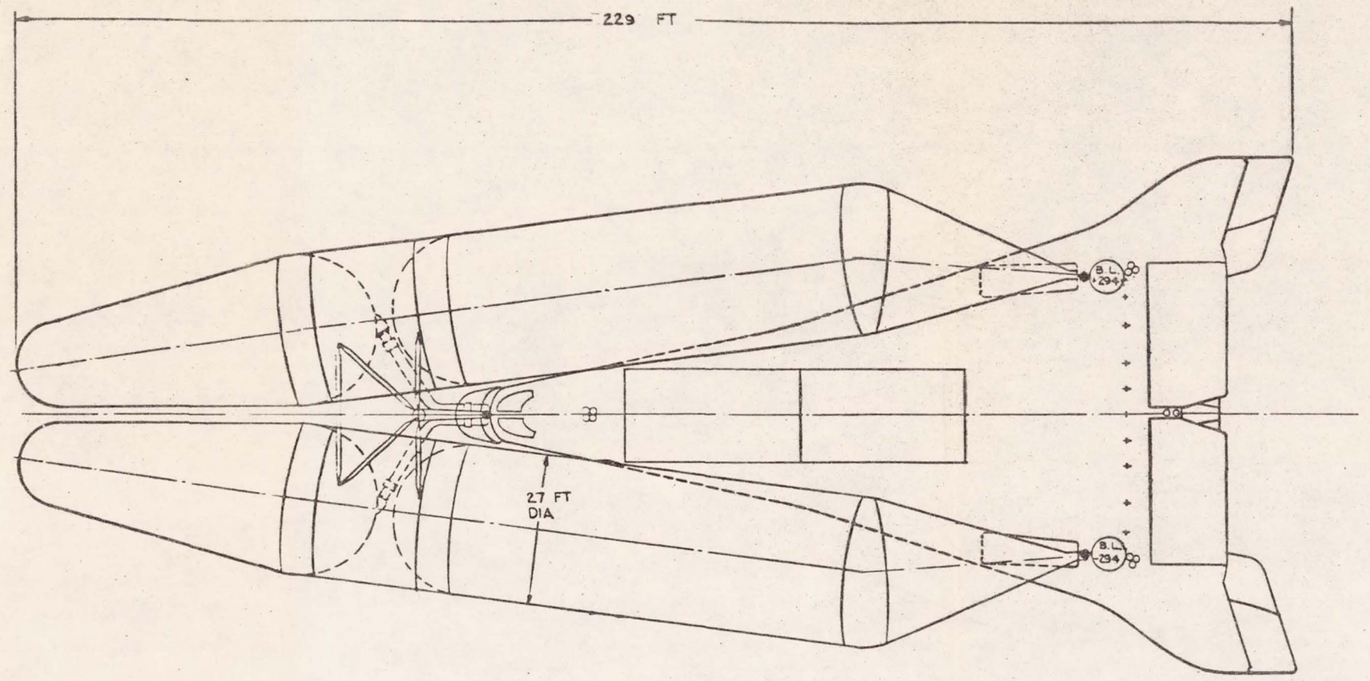
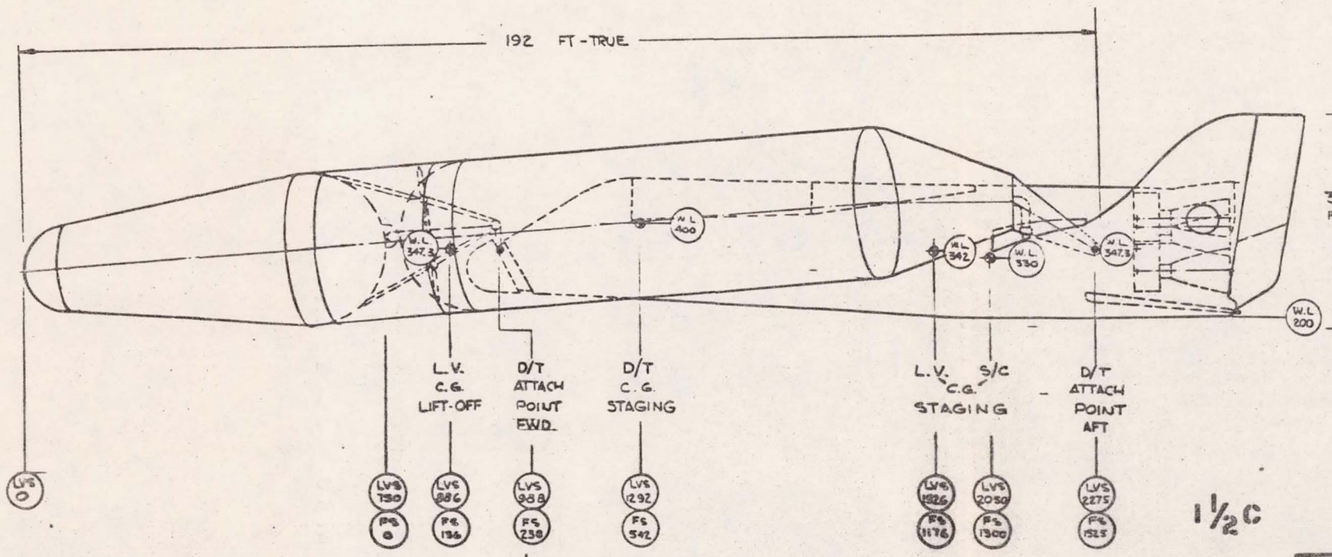
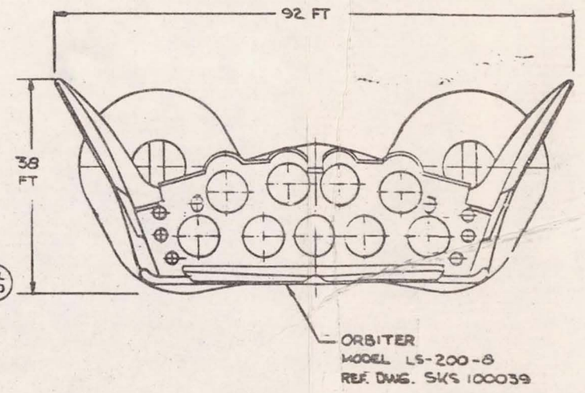
LIMITED CALENDAR LIFE | LIMITED OPERATING LIFE

REVISIONS			
ZONE/LTR	DESCRIPTION	DATE	APVD







1½C

Note: C.G. LOCATIONS ARE PRELIMINARY ESTIMATES

Fig. 4.5-4

CITY REGD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO

PARTS LIST

DESTROY DWG PER

UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES. TOLERANCES ON: FRACTIONS = ± 1/16; DECIMALS: .X = ± .1, .XX = ± .03, .XXX = ± .010; ANGLES = ± 2 DEG

DATE 2-3-71
DWN
APVD
APVD
ENGRG
CHKR
APVD
APVD

CONFR
CCA/DWD

NEXT ASSY APPLICATION | USED ON

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

LAUNCH VEHICLE (1EC)
DELTA BODY - STAGE 1 ½
MODEL LS-200-8

SIZE E | CODE IDENT 77287 | DRAWING NO SKS 100042 | REV

SCALE | SHEET

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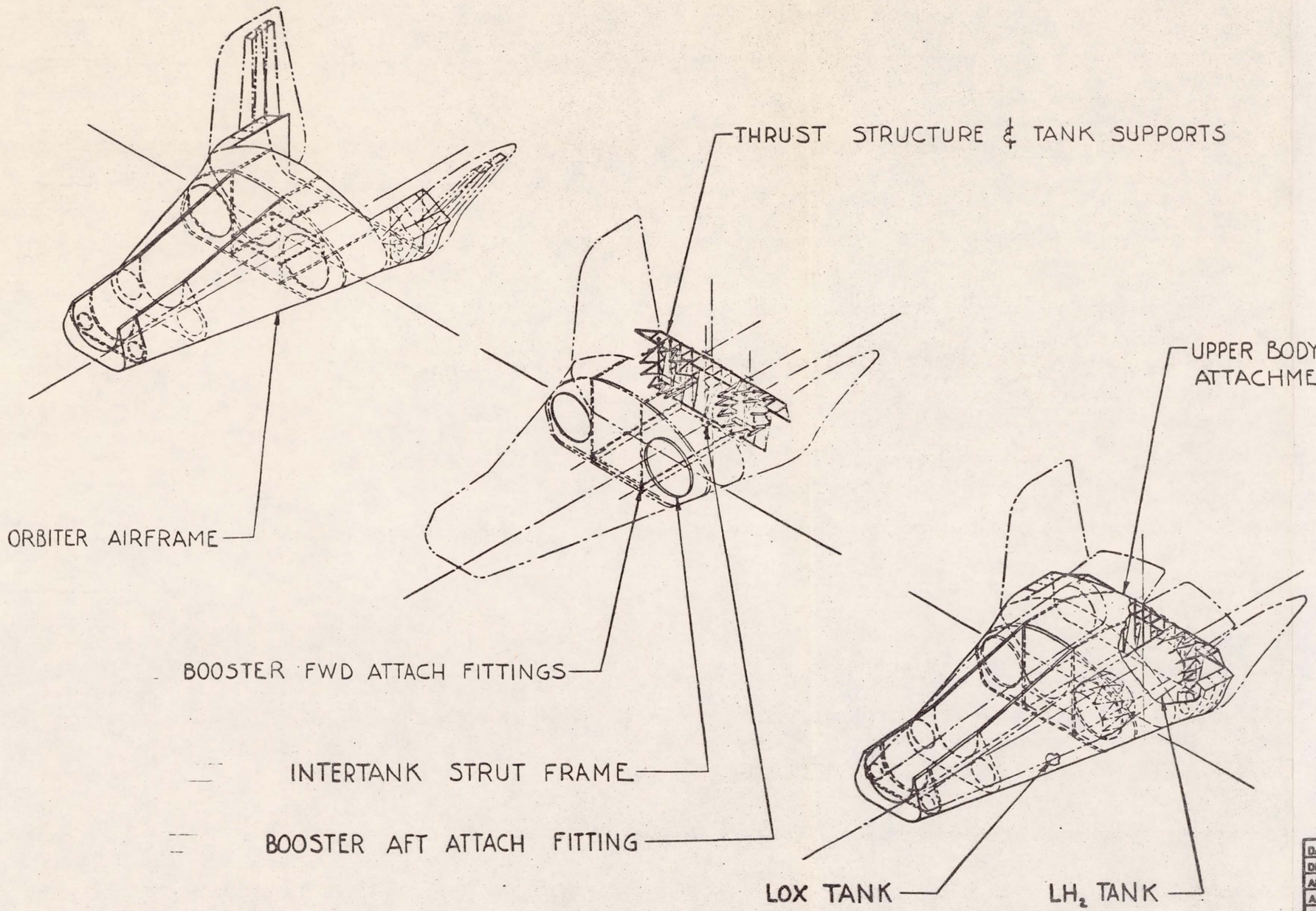

Fig. 4.5-5

DATE 2-24-71	LOCKHEED MISSILES & SPACE COMPANY A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION SUNNYVALE, CALIFORNIA	
DR MAS		
APPD Geo. Havrisik	STRUCTURAL ARRANGEMENT LOAD PATH CONCEPTS STUDY BASELINE SYSTEM (2-STAGE)	
APPD		
ENGRG		
CHK		
APPD	SIZE / CODE IDENT / DRAWING NO. SKG 100119 / REV -	
APPD	SCALE / SHEET	

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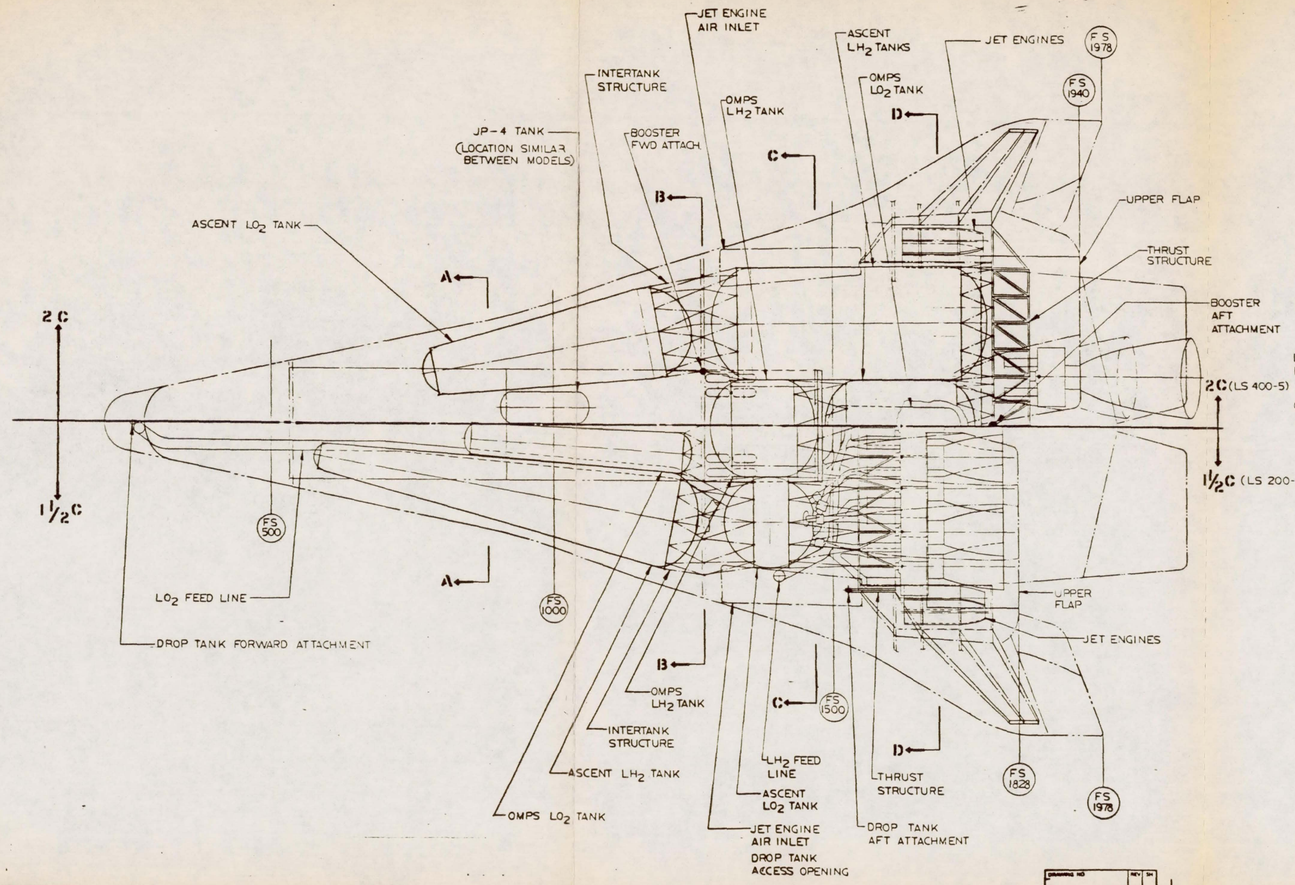

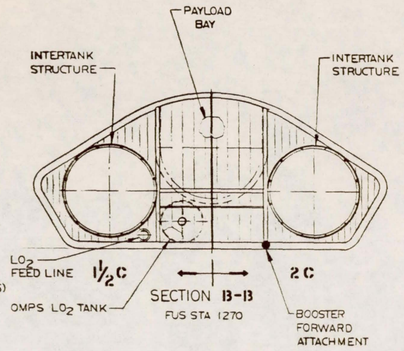

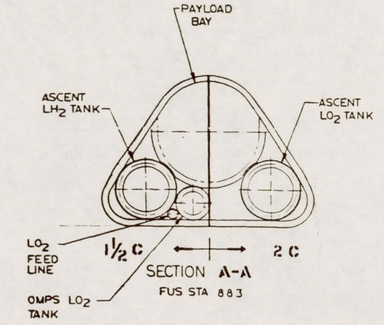

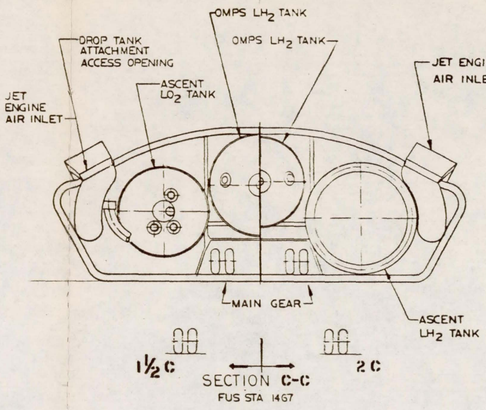

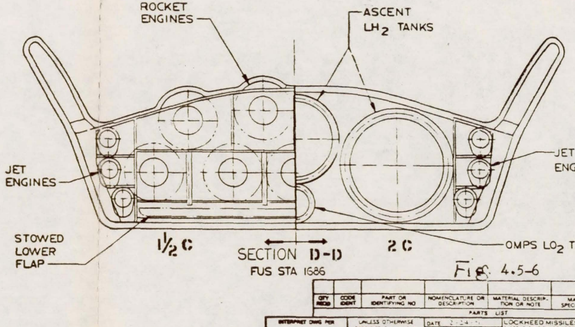

Fig. 4.5-6

QTY REQD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO
			PARTS LIST				

INTERPRET DWG PER	UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES TOLERANCES ON	DATE 2-24	LOCKHEED MISSILES & SPACE COMPANY A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION SUNNYVALE, CALIFORNIA
	FRACTIONS ± 1/16	DWN WS SCHULTZ	
	DECIMALS .X ± 0.1	APVD	COMPOSITE ARRANGEMENT
	.XX ± .03	ENGRG	DELTA BOOT (2 STAGE/STAGE½)
	.XXX ± .010	CHKR	MODELS LS 400-S/LS 200-S
	ANGLES ± ± 2 DEG	APVD	SIZE CODE IDENT DRAWING NO REV
NEXT ASSY USED ON	CONTR	APVD	J 09987 SKS 100045
APPLICATION	DO NOT SCALE DRAWING		SCALE NONE SHEET

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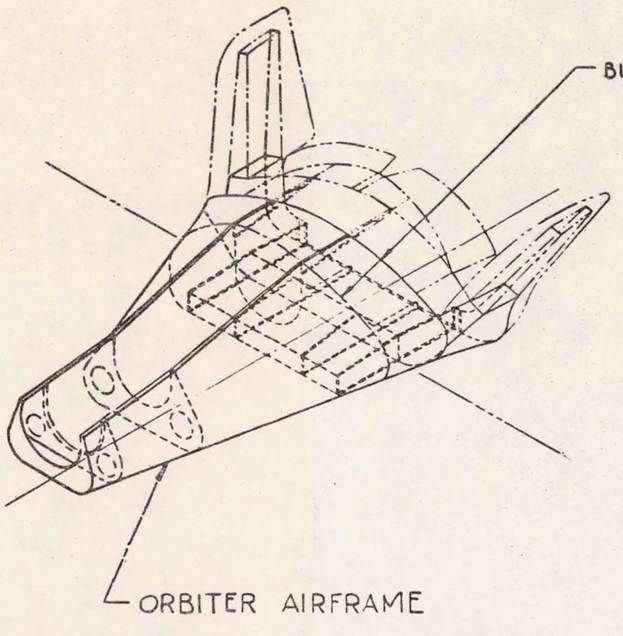

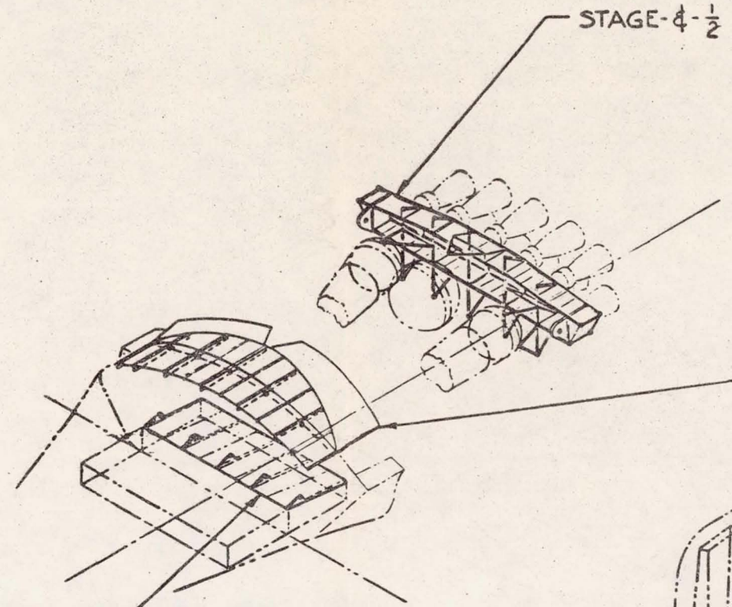

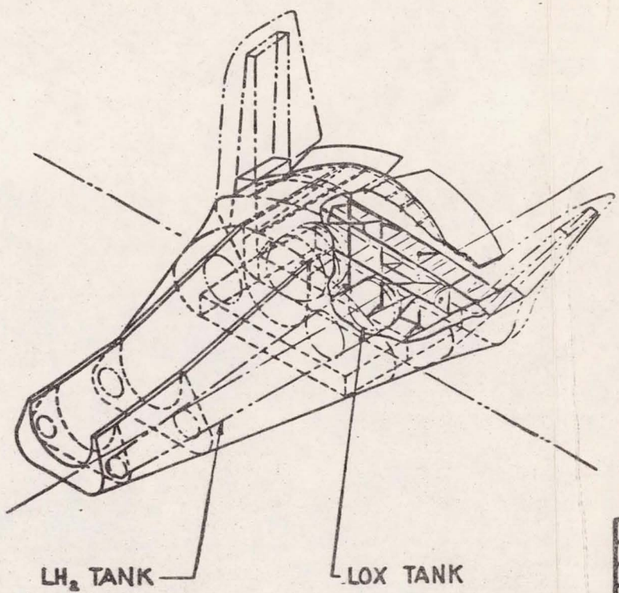

REF: LS-200-5 ORBITER
Fig. 4.5-7 SHT. 1

DATE 2-9-71	LOCKHEED MISSILES & SPACE COMPANY	
DR mac	A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION	
APPD Ea Haysik	SUNNYVALE, CALIFORNIA	
APPD	STRUCTURAL ARRANGEMENT	
ENGRG	LOAD PATH CONCEPTS STUDY	
CHK	— MAIN KEEL SYSTEM	
APPD	SIZE / CODE IDENT / DRAWING NO. SKG100115 / REV	
APPD	SCALE / SHEET 1 of 2	

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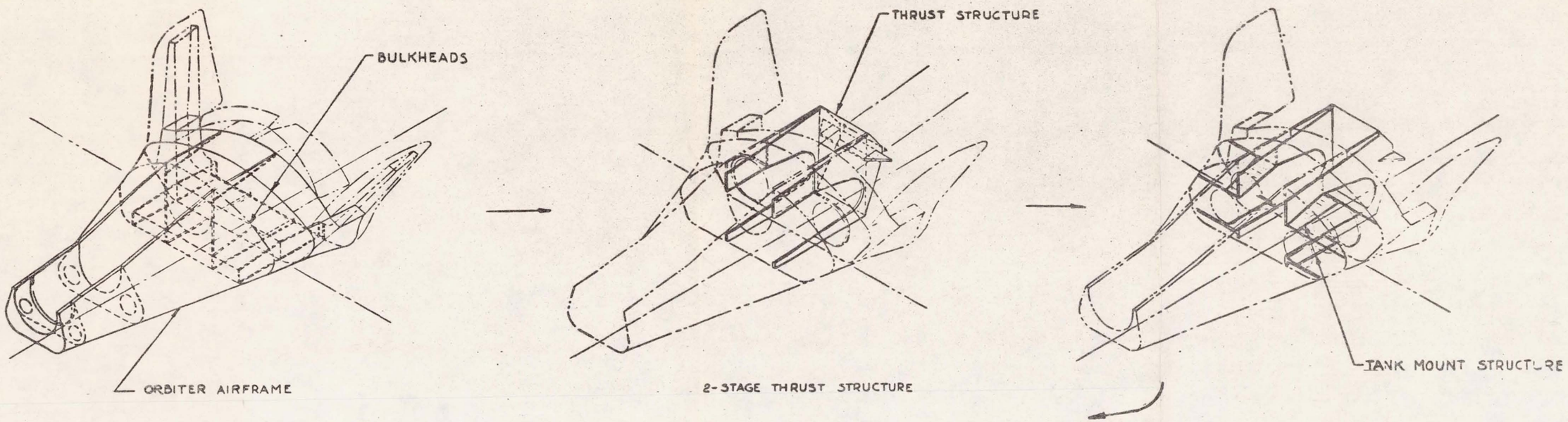

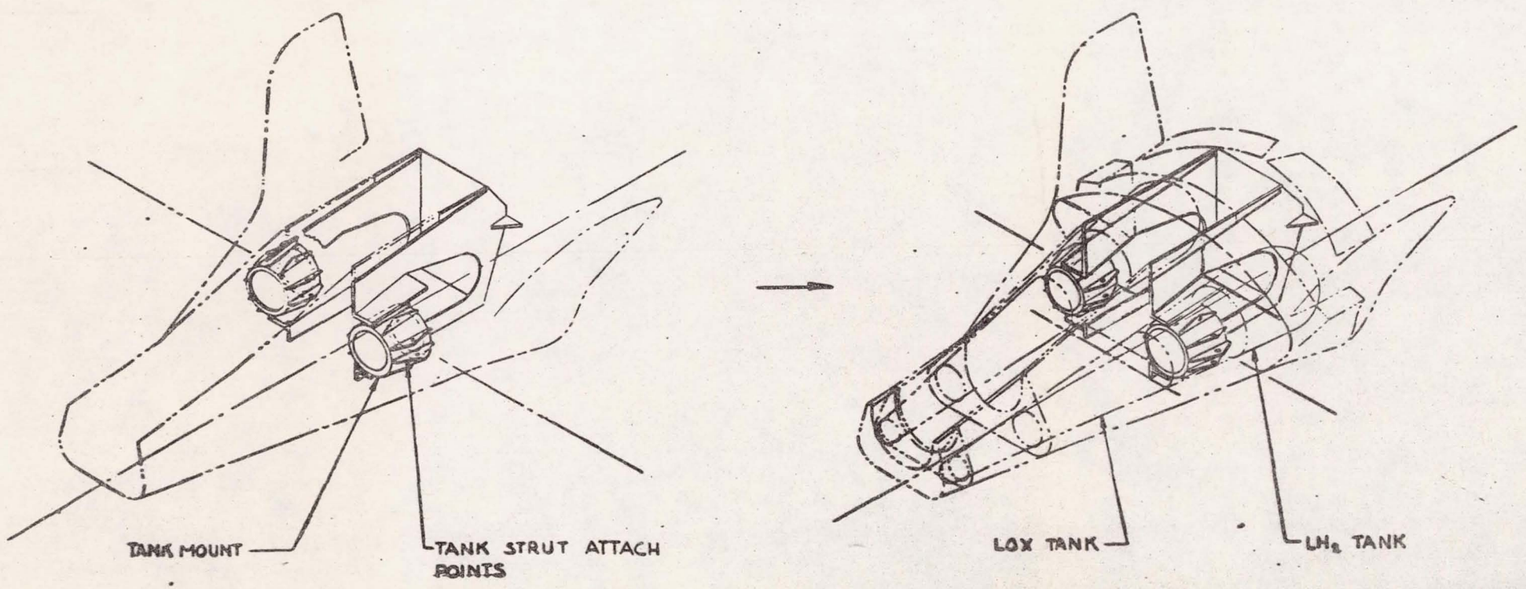

REF: LS-400-2 ORBITER
Fig. 4.5-7 SHT. 2

DATE 7-5-71	LOCKHEED MISSILES & SPACE COMPANY A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION SUNNYVALE, CALIFORNIA
DR mak	
APPD Geo. Havrilik	STRUCTURAL ARRANGEMENT LOAD PATH CONCEPTS STUDY— MAIN KEEL SYSTEM
APPD	
ENGRG	
CHK	
APPD	SIZE / CODE IDENT / DRAWING NO. LSKG100115 / REV
APPD	SCALE / SHEET 2 OF 3

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TANK MOUNT

BULKHEADS

GIMBAL

THRUST STRUCTURE

DROPTANK ATTACH

THRUST STRUCTURE

GIMBALS

TANK STABILIZATION POINTS

BULKHEADS

THRUST STRUCTURE

2-STAGE

1½-STAGE

THRUST STRUCTURE ATTACH POINTS

FLAP TRACK

Fig. 4.5-7 SHT. 3

DATE 2-10-71	LOCKHEED MISSILES & SPACE COMPANY A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION SUNNYVALE, CALIFORNIA
DR MAG	
APPD GEO HAVRISIK	STRUCTURAL ARRANGEMENT LOAD PATH CONCEPTS STUDY - MAIN KEEL SYSTEM
APPD	
ENGRG	
CHK	
APPD	SIZE / CODE IDENT / DRAWING NO. SKG1CC115 / REV
APPD	SCALE 1/55 / SHEET 3 OF 3

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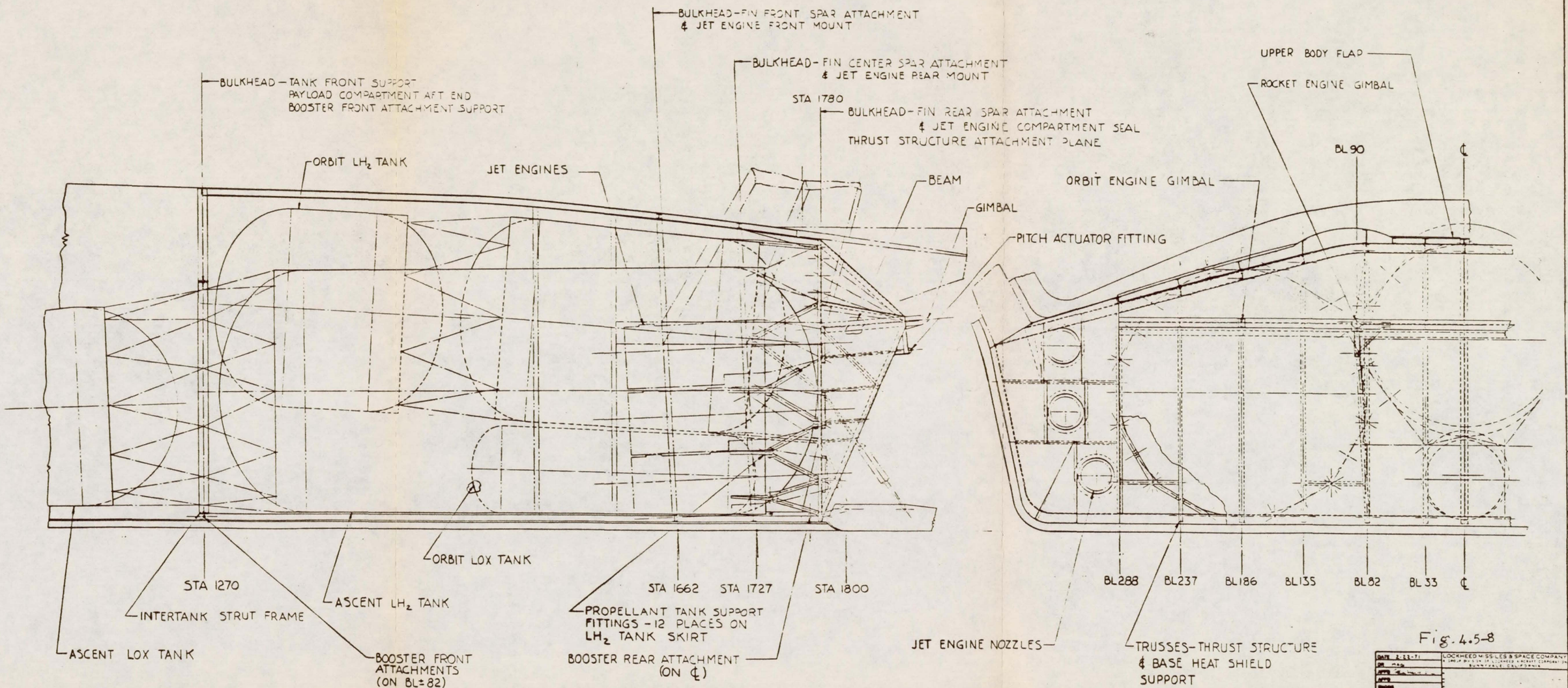

Fig. 4.5-8

LOCKHEED MISSILES & SPACE COMPANY
SUNNYVALE, CALIFORNIA

SIZE E | CODE IDENT ...657 | DRAWING NO. SKG100118 | REV
SCALE ... | SHEET

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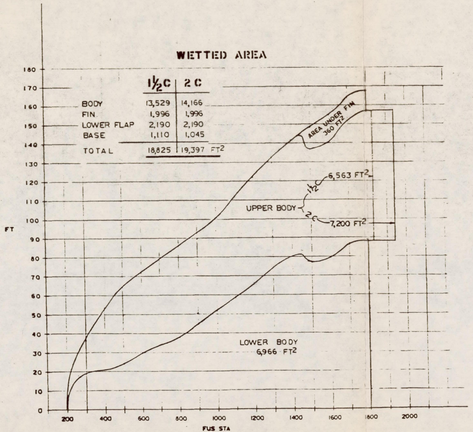
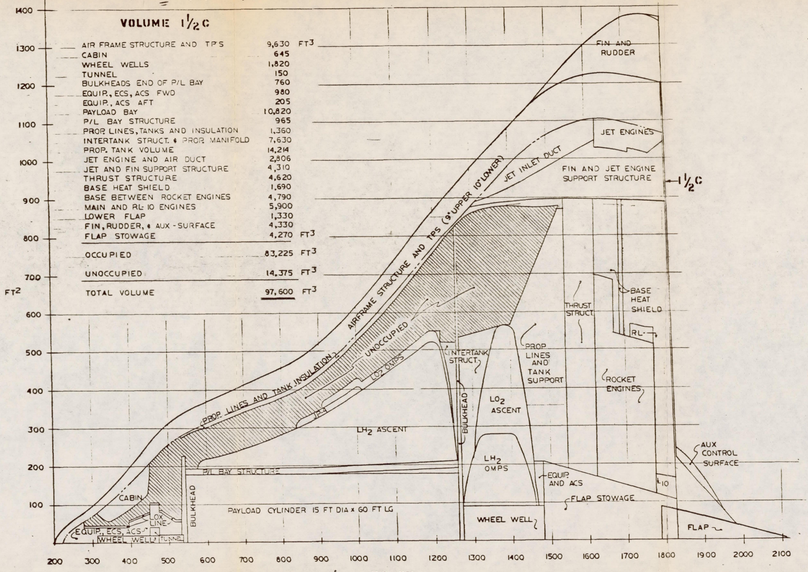
WETTED AREA

	1½C	2C
BODY	13,529	14,166
FIN	1,996	1,996
LOWER FLAP	2,190	2,190
BASE	1,110	1,045
TOTAL	18,825	19,397 FT²



VOLUME 1½C

AIR FRAME STRUCTURE AND TPS	9,630	FT³
CABIN	645	
WHEEL WELLS	1,820	
TUNNEL	150	
BULKHEADS END OF P/L BAY	760	
EQUIP., ECS, ACS FWD	980	
EQUIP., ACS AFT	205	
PAYLOAD BAY	10,820	
P/L BAY STRUCTURE	965	
PROP LINES, TANKS AND INSULATION	1,360	
INTERTANK STRUCT. & PROP MANIFOLD	7,630	
PROP. TANK VOLUME	14,214	
JET ENGINE AND AIR DUCT	2,806	
JET AND FIN SUPPORT STRUCTURE	4,310	
THRUST STRUCTURE	4,620	
BASE HEAT SHIELD	1,690	
BASE BETWEEN ROCKET ENGINES	4,790	
MAIN AND RL 10 ENGINES	5,900	
LOWER FLAP	1,330	
FIN, RUDDER, & AUX - SURFACE	4,330	
FLAP STOWAGE	4,270	FT³
OCCUPIED	83,225	FT³
UNOCCUPIED	14,375	FT³
TOTAL VOLUME	97,600	FT³



VOLUME 2C

AIR FRAME STRUCTURE AND TPS	9,980	FT³
CABIN	645	
WHEEL WELLS	1,820	
TUNNEL	150	
BULKHEADS END OF P/L BAY	760	
EQUIP, ECS, ACS FWD	970	
EQUIP., ACS AFT	200	
PAYLOAD BAY	10,820	
PAYLOAD BAY STRUCTURE	965	
PROP LINES, TANKS AND INSULATION	1,920	
INTERTANK STRUCTURE	5,570	
PROP. TANK VOLUME	28,753	
JET ENGINE AND AIR DUCT	2,807	
JET AND FIN SUPPORT STRUCTURE	4,310	
THRUST STRUCTURE	2,940	
BASE HEAT SHIELD	600	
BASE BETWEEN ROCKET ENGINES	1,330	
MAIN AND RL-10 ENGINES	5,070	
LOWER FLAP	1,330	
FIN, RUDDER, & AUX. - SURFACE	4,330	FT³
OCCUPIED	85,270	FT³
UNOCCUPIED	20,080	FT³
TOTAL VOLUME	105,350	FT³

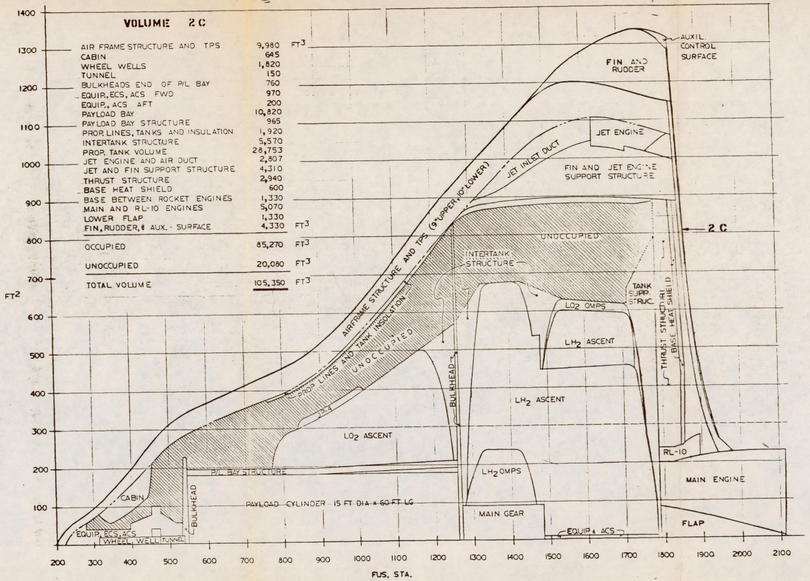


Fig. 4.5-9

VOLUME CURVE
SATURN V/ S-IVB TANK 500-510

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4.6 STRUCTURES

4.6.1 Introduction

Convertibility of the delta-body orbiter, designed as a stage-and-one-half vehicle system and designated as the $1\frac{1}{2}$C, to one serving as the second-stage of the two-stage reusable vehicle system and designated as the 2-C, is primarily a structural design and internal arrangement problem. To accomplish this task, the goal was to design a compromised $1\frac{1}{2}$C orbiter so that a minimum of structural modifications would be required to convert to a 2-C configuration. This task was completed only to the depth of establishing gross structural weights and to show feasibility.

4.6.2 Structural Design Criteria

The loads design criteria for the stage-and-one-half vehicle system ($1\frac{1}{2}$-C) is described in detail in Sec. 2.8.4. Design criteria used for loads analysis for the two-stage convertible orbiter (2-C) is described in detail in Sec. 3.8.2. The steady-state launch-release load factor is derived from MAC/DAC Phase B booster data. A summary comparison of the two loads criteria used for this study is shown in Table 4.6-1. The difference in the criteria used for 2-C and $1\frac{1}{2}$-C consists of a higher liftoff load factor and higher $\alpha q/\beta q$ characteristics for 2-C.

The booster thrust vector, based on the MAC/DAC geometry, is assumed to be directed through the composite center-of-gravity of the launch vehicle, resulting in a high normal acceleration as well as the maximum longitudinal load factor seen by the orbiter. This condition is considered to result in maximum booster-orbiter interface loads and in design loads for the aft section of the 2-C orbiter. A comparison of the maximum 2-C interface loads with the maximum $1\frac{1}{2}$-C interface loads is shown in Fig 4.6-1.

4.6.3 Stage-and-One-Half Line Load Distribution

The "booster" load path for the $1\frac{1}{2}$-C orbiter is primarily through the thrust structure droptank attachments (Fig 4.6-2). No in-plane bending is introduced

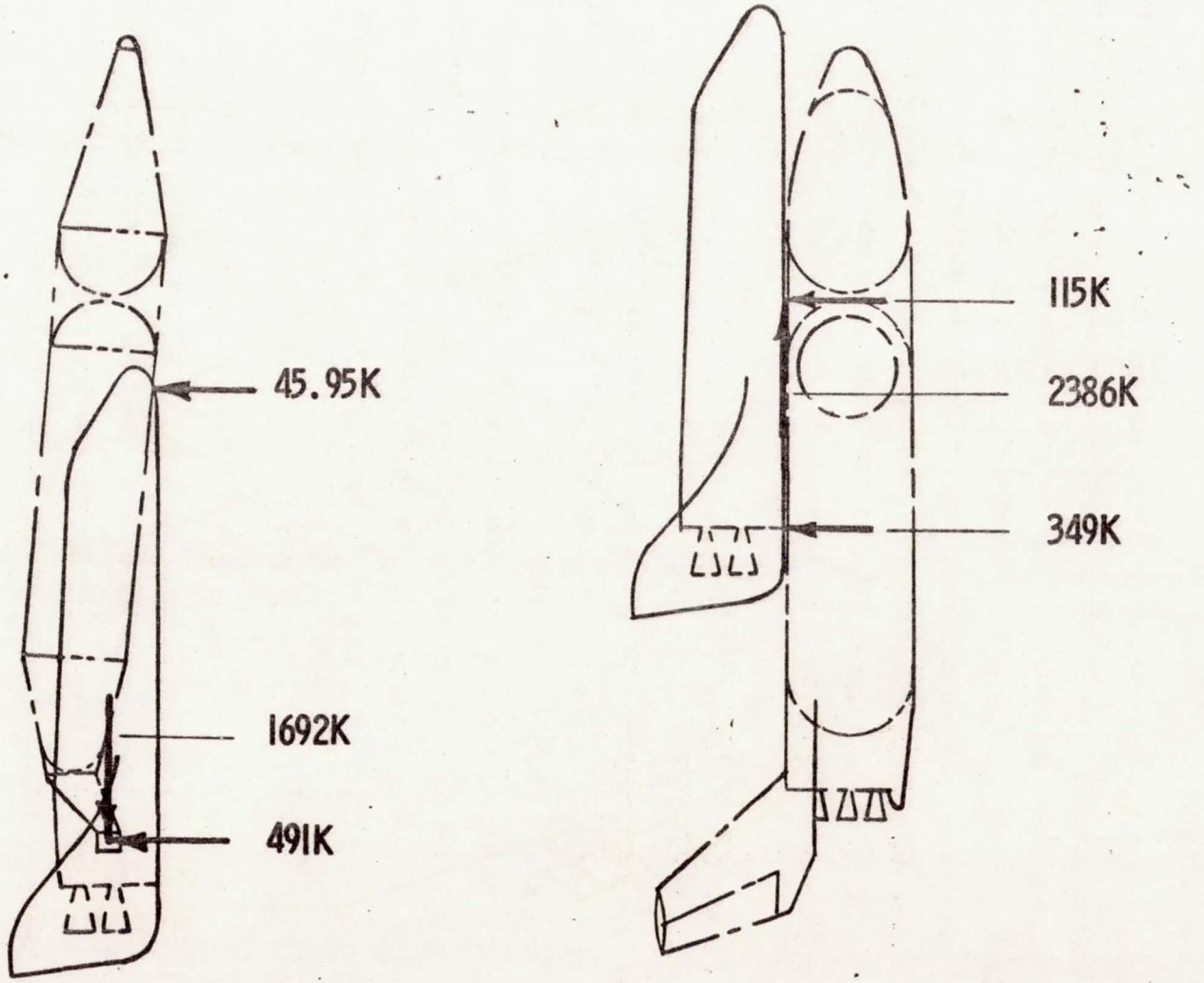


Fig 4.6-1 Orbiter Design Interface Loads

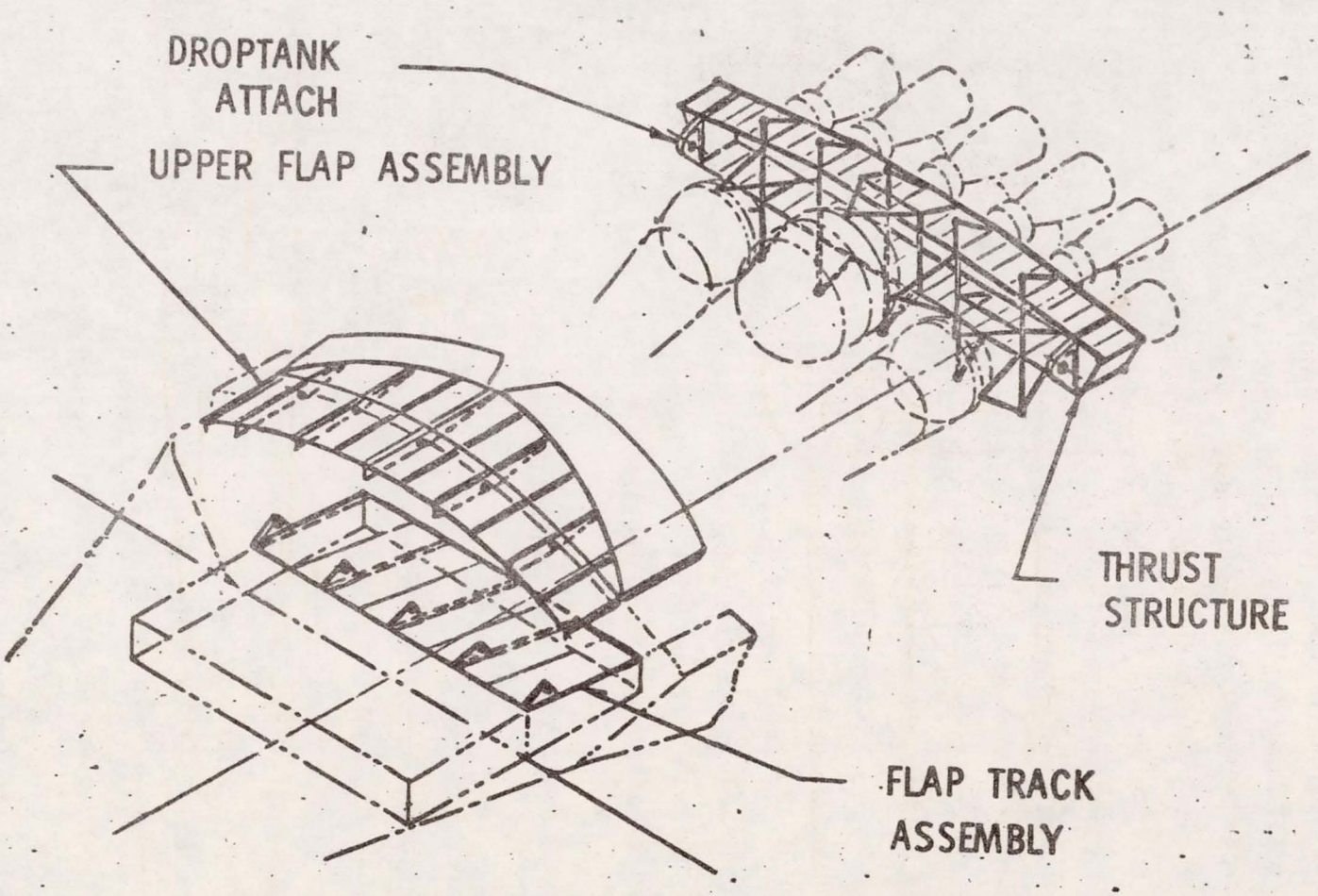


Fig 4.6-2 Delta-Body Orbiter Stage-and-One-Half Structure

into the orbiter fuselage from the droptank loads. The thrust structure equilibrates all loading. This is shown graphically in the figure where a heavily reinforced upper-flap assembly skirt, which attaches to the fuselage upper surface, distributes the load concentrations from the thrust structure. Attach fittings on the flap track assembly distribute the lower load concentrations from the thrust structure to the fuselage lower surface. The propellant tanks are supported off the thrust structure; they are mounted in a statically determinate manner by tying the forward end of the LH_2 ascent tank to a fuselage frame without restricting longitudinal deformations. Converting design bending moments and axial load for the $1\frac{1}{2}$-C orbiter to maximum line-load intensities, N_x, (lb per lineal in.) reveals that the maximum values occur between fuselage Sta 600 and Sta 1200, which constitutes the payload compartment section of the fuselage. Figures 4.6-3 and 4.6-4 show maximum compression (-) and tension (+) stress resultants for the upper and lower surfaces, respectively. A complete discussion of this analysis is presented in the Sixth Letter Progress and Status Report, Section 2.4 (L2-01-01-M1-3).

4.6.4 Two-Stage Line Load Conversion

The conversion from the $1\frac{1}{2}$-C load path to the 2-C load path begins with establishing common structure between the two orbiters. This structure is shown in Fig. 4.6-5 as solid and dashed lines. Structure that is removed from the $1\frac{1}{2}$-C includes the thrust structure, flap track assembly, and all of the internal tankage, shown in Fig. 4.6-2

The structure that is replaced is shown in Fig. 4.6-6. It includes the new thrust structure, the booster attach reinforcement (to the lower surface of the fuselage), booster attach fittings, and tankage.

The booster load path for the 2-C orbiter is quite different from that of the $1\frac{1}{2}$-C system. A severe eccentric load path results between the two stages at the structural connection. The load most severe to the rear portion of the orbiter, discussed previously in Section 3.8, is the booster maximum acceleration condition.

Table 4.6-1

LOADS CRITERIA FOR DESIGN CONDITIONS

	Stage-and-one-Half	Two-Stage
• Liftoff		
Launch Release - All Engines Operating		
Steady-State Load Factor	1.25	1.35
Total Load Factor Including Dynamic Amplification	1.50	1.70
• Maximum Dynamic Pressure		
Maximum αq, psf-deg	2000	2800
Maximum βq, psf-deg	N/C	2800
Long Load Factor	1.7	1.7
Lateral Aero Load Factor (Aeroelasticity and Buffeting)	1.3	1.3
• Maximum Longitudinal Acceleration, g	3.0	3.0
• Landing Impact Condition		
Sinking Speed, ft/sec	10	10
Aerodynamic Life, g	1.0	1.0
Main Gear Vertical Load (At Payload = 40K lb)	1.7	1.7

As discussed previously in Section 3.8, to reduce the disparity in loads between the two orbiters, the ascent tankage for the two-stage orbiter is designed to be a primary-load-carrying structure. The fuselage line loads for the 2-C orbiter for the various load conditions are superimposed on the $1\frac{1}{2}$-C line loads in Figs. 4.6-3 and 4.6-4 for comparison. It is noted that $1\frac{1}{2}$-C upper-surface compression line loads are more severe than the 2-C line loads (Fig. 4.6 3), while the lower-surface compression line loads are equivalent from Sta 550 to Sta 950. The $1\frac{1}{2}$-C loads are more severe forward of Sta 550, and the 2-C loads are more severe aft of Sta 950. It is also interesting to note the difference in response between orbiters for + αq and - αq. For the $1\frac{1}{2}$-C orbiter, + αq causes compression on the upper surface, and - αq causes compression on the lower surface, while the exact opposite is true for the 2-C orbiter. The reason for this apparent anomaly is simply because of the way each orbiter is supported to its "booster". The $1\frac{1}{2}$-C is essentially a "simply-supported" beam between the nose and aft end, while the 2-C orbiter is a "propped-cantilever" beam between Sta 1270 and the aft end.

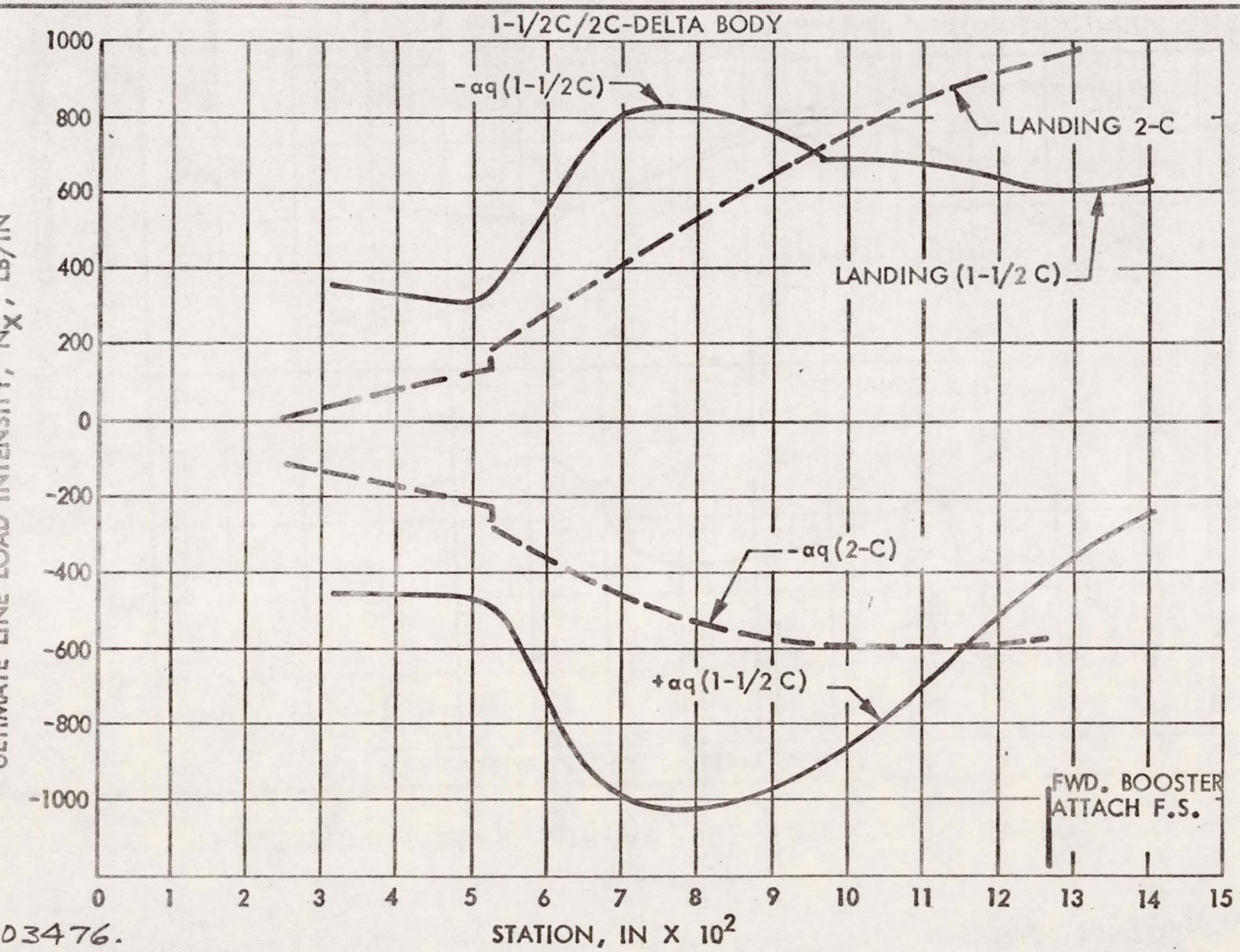
The $1\frac{1}{2}$-C orbiter has relatively light line loads throughout both upper surfaces, while the 2-C orbiter has a severe load concentration to distribute aft of Sta 1270, as discussed previously in Section 3.8. There are two possibilities for reacting this tension load: the first is to provide two tension tie-rods from the thrust structure to the booster attach fitting, and the alternative is to react the tension load with the bottom fuselage surface and to shear-lag the load into tapered longerons leading to the booster fittings. As discussed in Section 3.8, the latter approach was selected for the present study.

4.6.5 Conclusions

The conclusions gained from this study are discussed elsewhere from a general program-cost approach. Structurally, the results from this study establish that it is feasible, but the weight penalties associated with the approach are too complex to truly evaluate in such a brief study. The $1\frac{1}{2}$-C fuselage forward of Sta 1270 appears to be quite adequate to serve as a 2-C orbiter. Structural changes aft of Sta 1270 are required.

Fig. 4.6-3

MAXIMUM UPPER SURFACE DESIGN LOAD ENVELOPE



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IMSC-A989142
Vol II

Fig. 4.6-4

LOCKHEED

MAXIMUM LOWER SURFACE DESIGN LOAD ENVELOPE

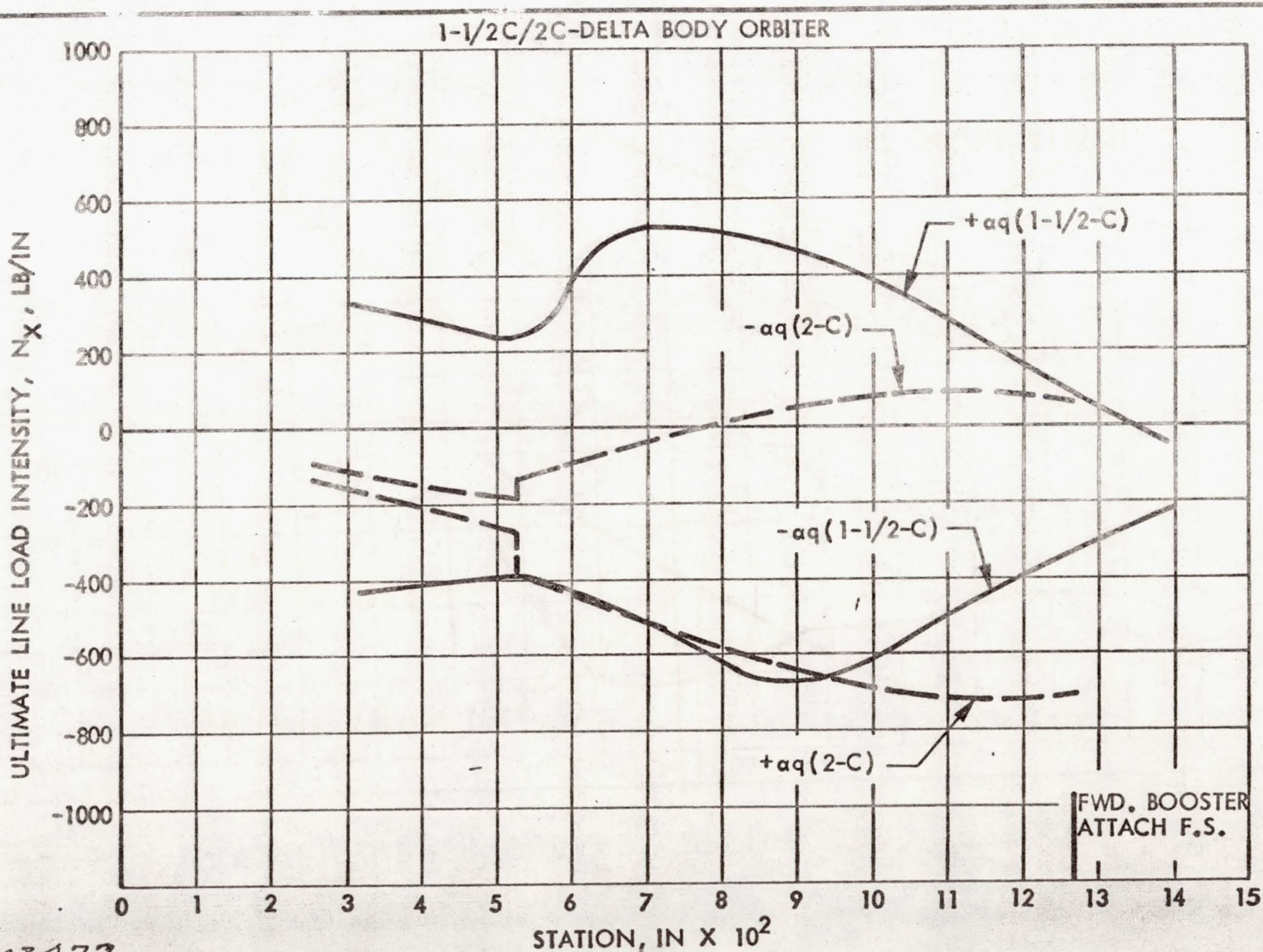

Fig. 4.6-4

4.6-8

D03477

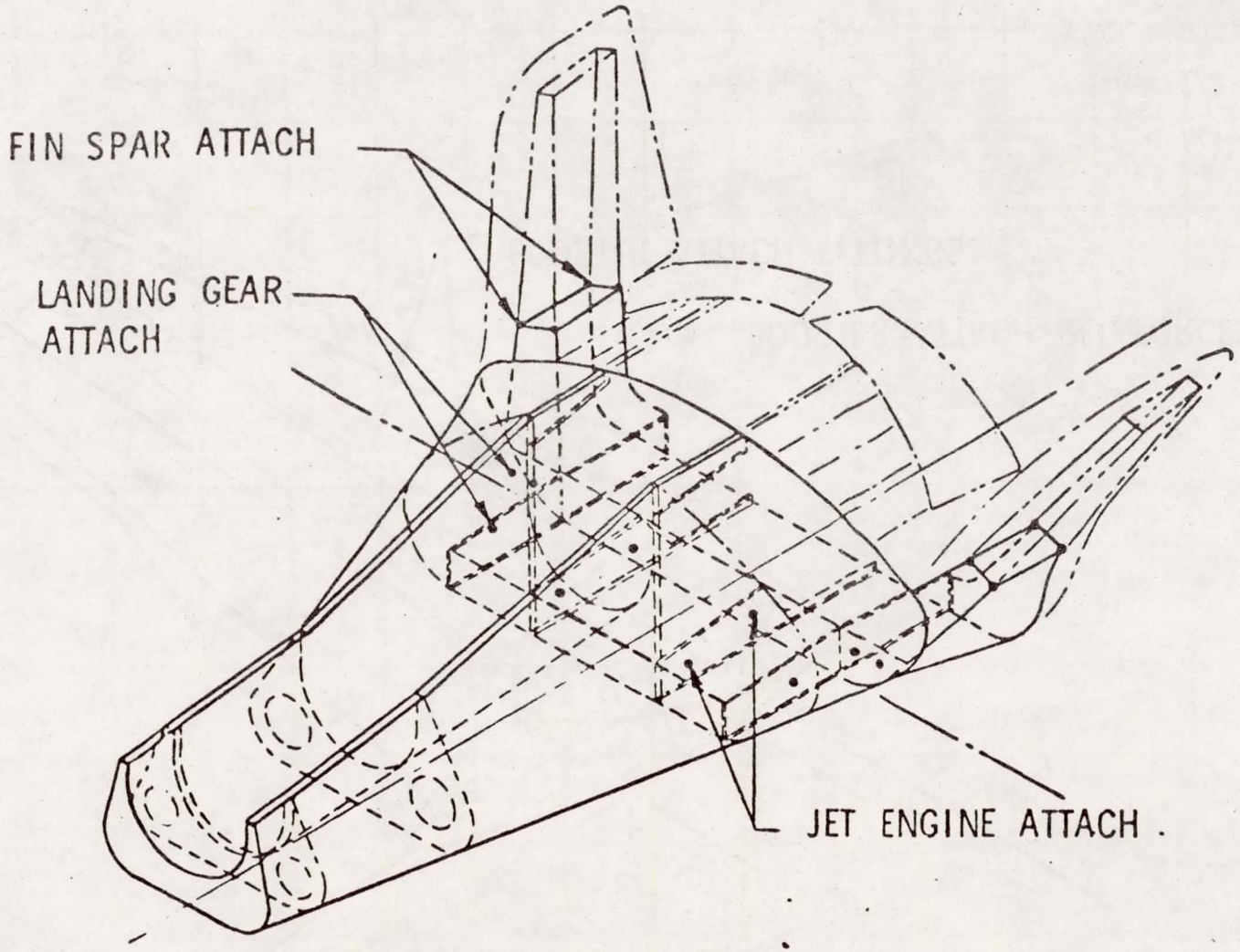

Fig. 4.6-5 Delta-Body Orbiter Stage-and-One-Half/Two-Stage Common Structure

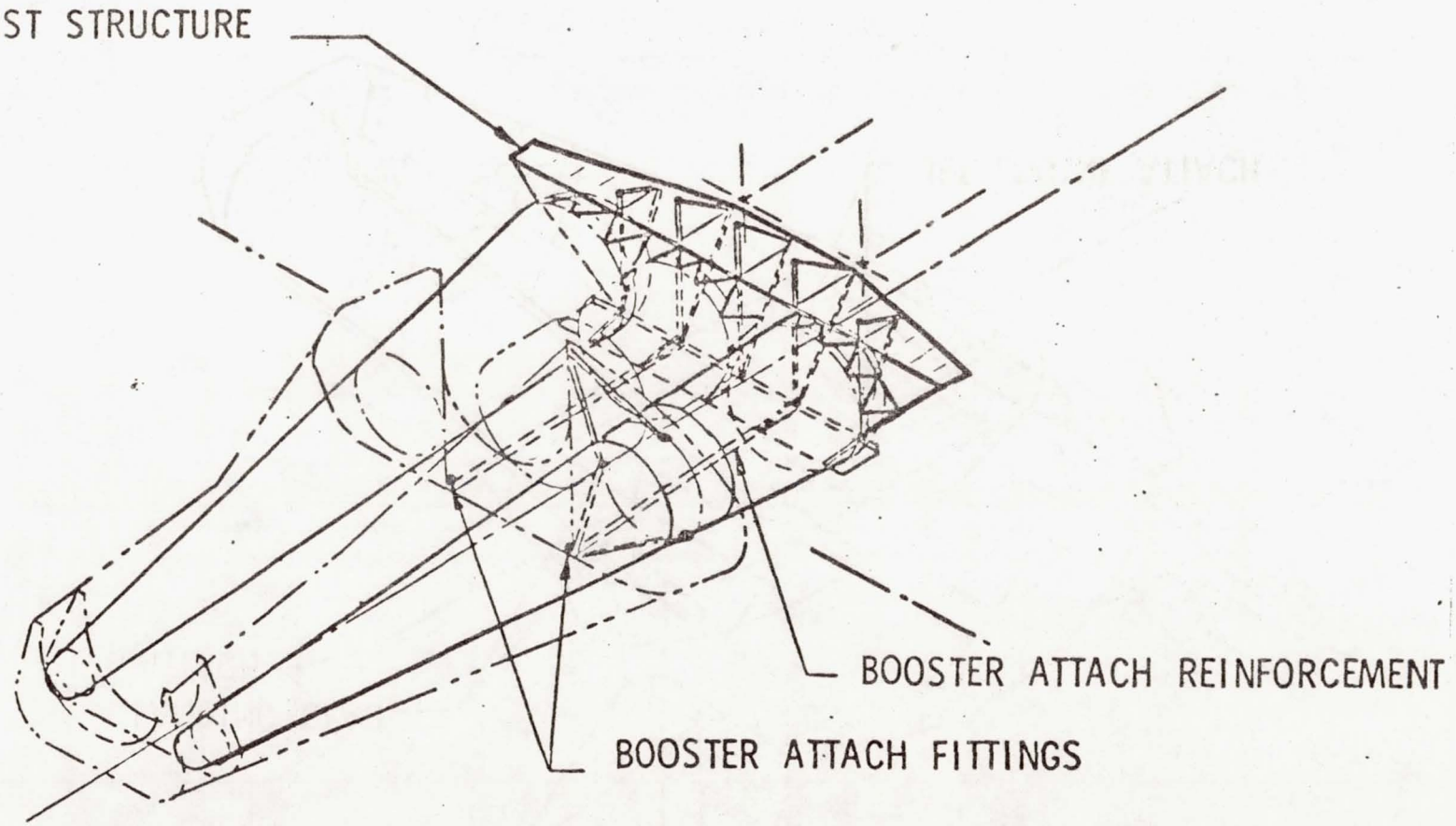

Fig. 4.6-6 Delta-Body Orbiter Two-Stage Structure

4.7 PROPULSION

Propulsion systems investigations have been performed to evaluate the extent of conversion required or the degree of commonality between the four major propulsion systems on the stage-and-one-half and the two-stage orbiters. The principal conclusion reached in these investigations of an incremental development approach is that the main propulsion system (MPS) will require a nearly complete revision, while the other three propulsion systems could be limited to resizing of their respective propellant tanks. As an aid to assessing the propulsion system changes anticipated in the conversion of a stage-and-one-half orbiter to a two-stage orbiter, an abbreviated set of characteristics for the four major propulsion systems, applied to the baseline configurations (labelled"B") and the convertible systems (labelled "C"), is presented in Table 4.7-1.

In the main propulsion system, the use of nine rocket engines with an on-board propellant load of 240K lbm to supplement propellant carried in the droptanks for stage-and-one-half orbiter is grossly different from the nearly reciprocal arrangement of two rocket engines with an on-board propellant load of 546K lbm for the two-stage orbiter. Rocket engines for this propulsion system are defined by NASA document 13M15000B, dated 1 March 1971, except that the stage-and-one-half configuration would employ a nozzle expansion ratio of 53:1 to provide a nearly optimum performance throughout its ascent into the 50-nm x 100-nm orbit. To more clearly illustrate the extent of changes in tankage, plumbing, and engine arrangement, Figs. 4.7-1 and 4.7-2 present simplified MPS schematics for each of these two vehicles. In the stage-and-one-half orbiter concept, LO_2 from the droptanks in cascaded into the spherical ascent tanks that directly feed the rocket engines. The LH_2 droptank propellant supplies all rocket engines through a manifold arrangement until droptank separation, after which the number of operating engines is reduced to two and they are supplied LH_2 from the orbiter tanks. Droptanks are sized so

TABLE 4.7-1

PROPULSION SYSTEMS CHARACTERISTICS

ORBITER STAGE	1-1/2-B	1-1/2-C	2-C	2-B
MPS				
NO. ROCKET ENGINES	9	9	2	2
THRUST LEVEL (LBF)	530K (SL)	530K (SL)	632K (VAC)	632K (VAC)
NOZZLE AREA RATIO	53	53	150	150
PROPELLANT WEIGHT*	239,200	239,200	547,200	546,600
OMPS				
NO. ROCKET ENGINES	2	2	2	2
THRUST LEVEL	15K	15K	15K	15K
PROPELLANT WEIGHT*	34,100	34,100	27,000	25,300
ACPS				
NO. ROCKET ENGINES	39	39	34	34
THRUST LEVEL	1.5K	1.5K	1.5K	1.5K
PROPELLANT WEIGHT*	4,890	4,890	4,130	3,940
ABPS				
NO. JET ENGINES	6	6	6	4
THRUST LEVEL	18K	18K	18K	18K
JP FUEL WEIGHT*	12,200	12,200	9,700	9,100

*IMPULSE PROPELLANT

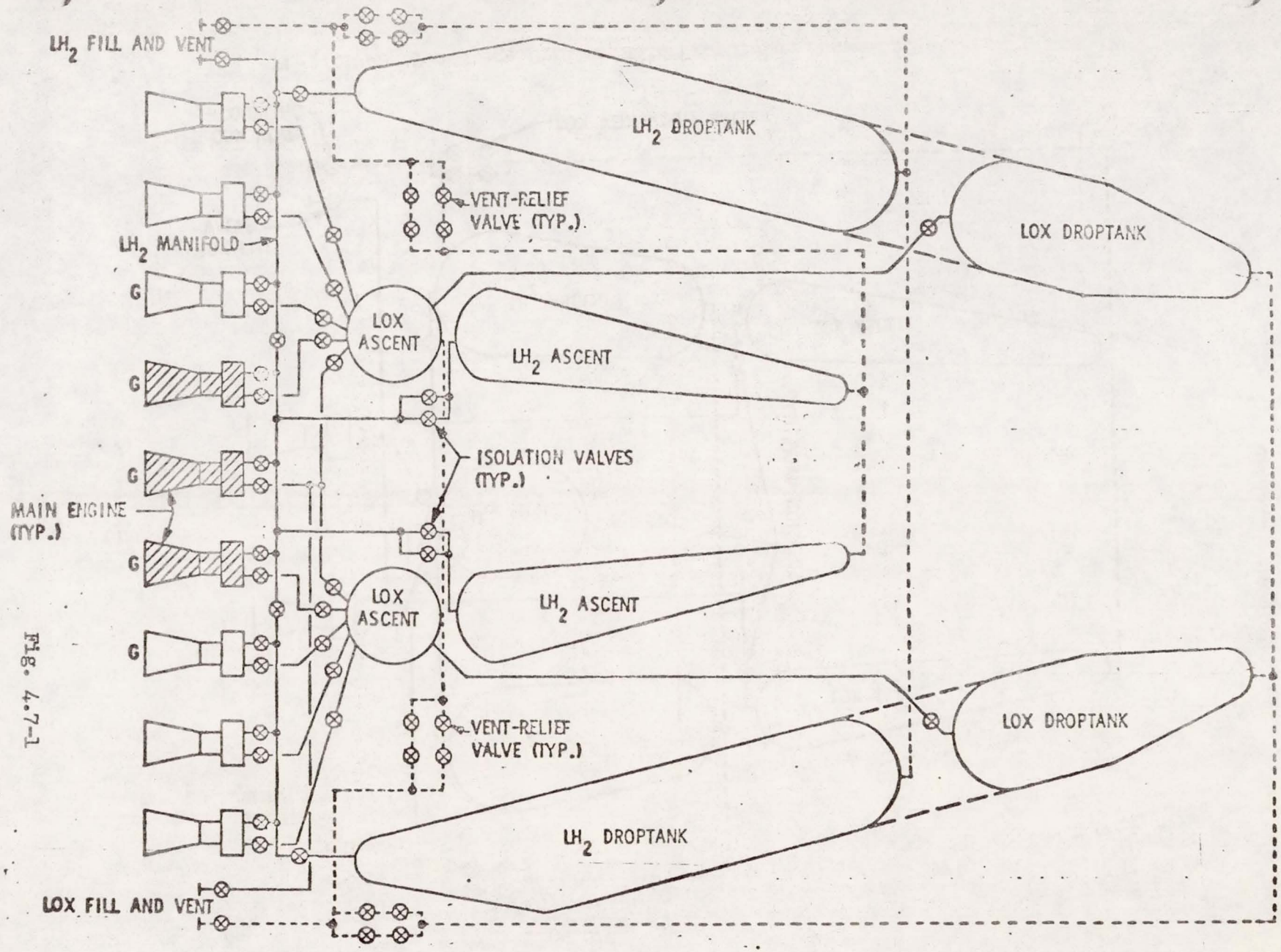

Fig. 4.7-1 Stage-and-One-Half Orbiter Main Propulsion Schematic

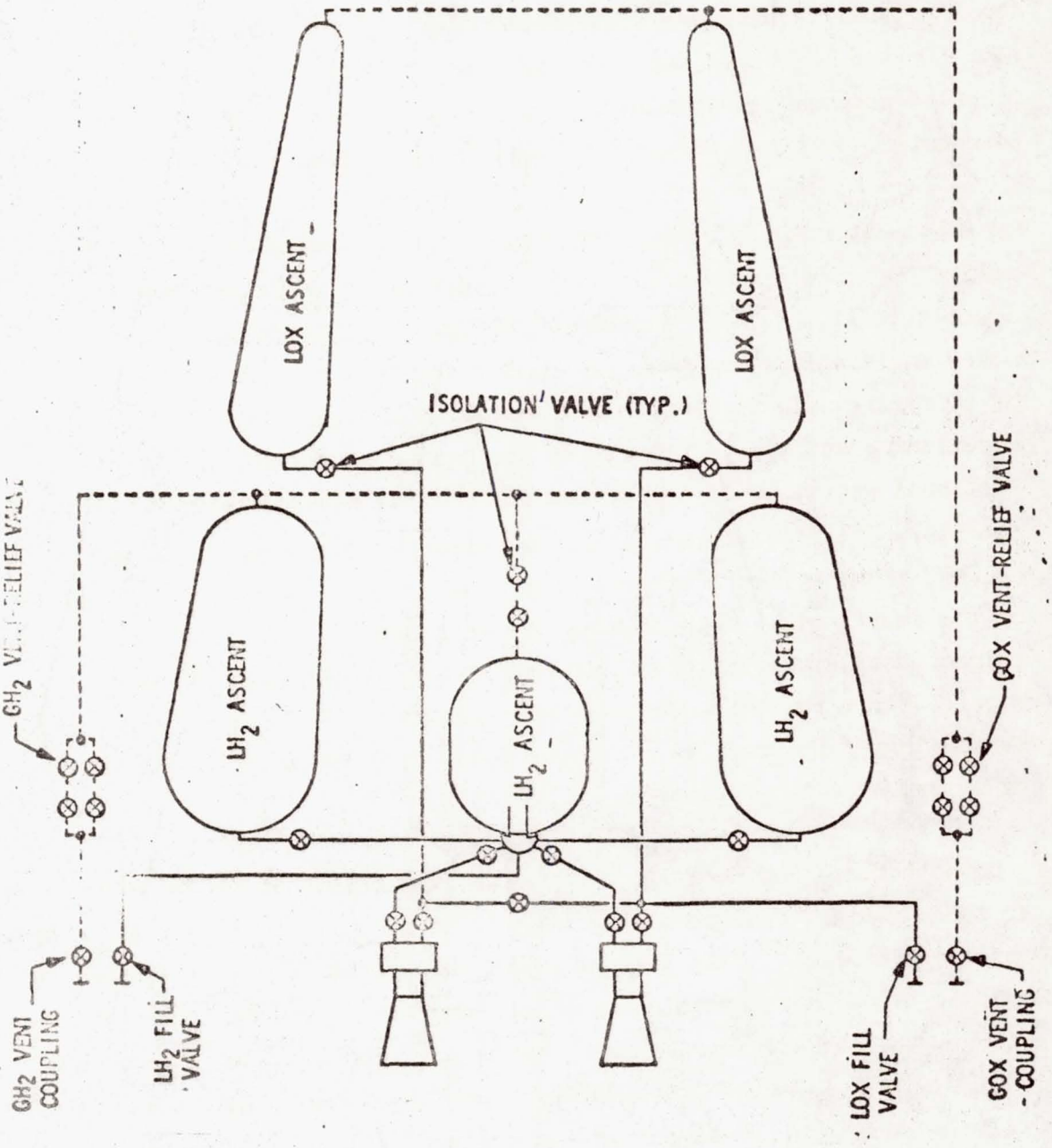

Fig. 4.7-2 Two-Stage Orbiter Main Propulsion Schematic

Fig. 4.7-2

that LO_2 droptank depletion constitutes the signal for initiation of droptank separation. The MPS schematic for the two-stage orbiter depicts the two-engine installation with the two outer LH_2 tanks cascading into the central tank and the LO_2 tanks feeding the engines directly. Since the shape and the size of both LO_2 and LH_2 tanks change along with the number of rocket engines, the only items of commonality between the two MPS arrangements are a number of valves associated with flow to-and-from individual engines and the rocket engine powerhead.

The orbit-maneuvering propulsion system (OMPS) considered for these vehicle applications utilizes two RL10 rocket engines in a system arranged schematically as shown in Fig. 4.7-3. Obviously, the specific installation points for the rocket engines and the tanks will alter the length and routing of plumbing. The differing weights of the vehicles while in orbit also will dictate different propellant quantities. Otherwise, the number of valves, their size, and the functional operation of the system will not be altered between the two vehicle concepts. In the 2-stage orbiter, a capability to transfer propellant from the OMPS tanks to the MPS tanks (not shown on the schematic) is required during the abort mode. This commonality will persist even if alternate OMPS design concepts are employed such as (1) using ACPS pumps to provide liquid propellants to remotely mounted thrust chambers or (2) operating these remotely mounted thrust chambers in a gas-gas mode using the ACPS propellant conditioning system as a supply source.

As noted before, the attitude control propulsion system (ACPS) concept employed for these vehicle designs shares its liquid-propellant storage with the OMPS propellant storage. This high-pressure thrust chamber concept utilizes separate propellant-conditioning subsystems for oxygen and for hydrogen to generate the high-pressure gases. Since the system, illustrated schematically in Fig. 4.7-4, is based on providing a continuous gas supply for operation of any four thrusters, these subsystems are not affected by the removal or addition of thrusters to the overall vehicle installation.

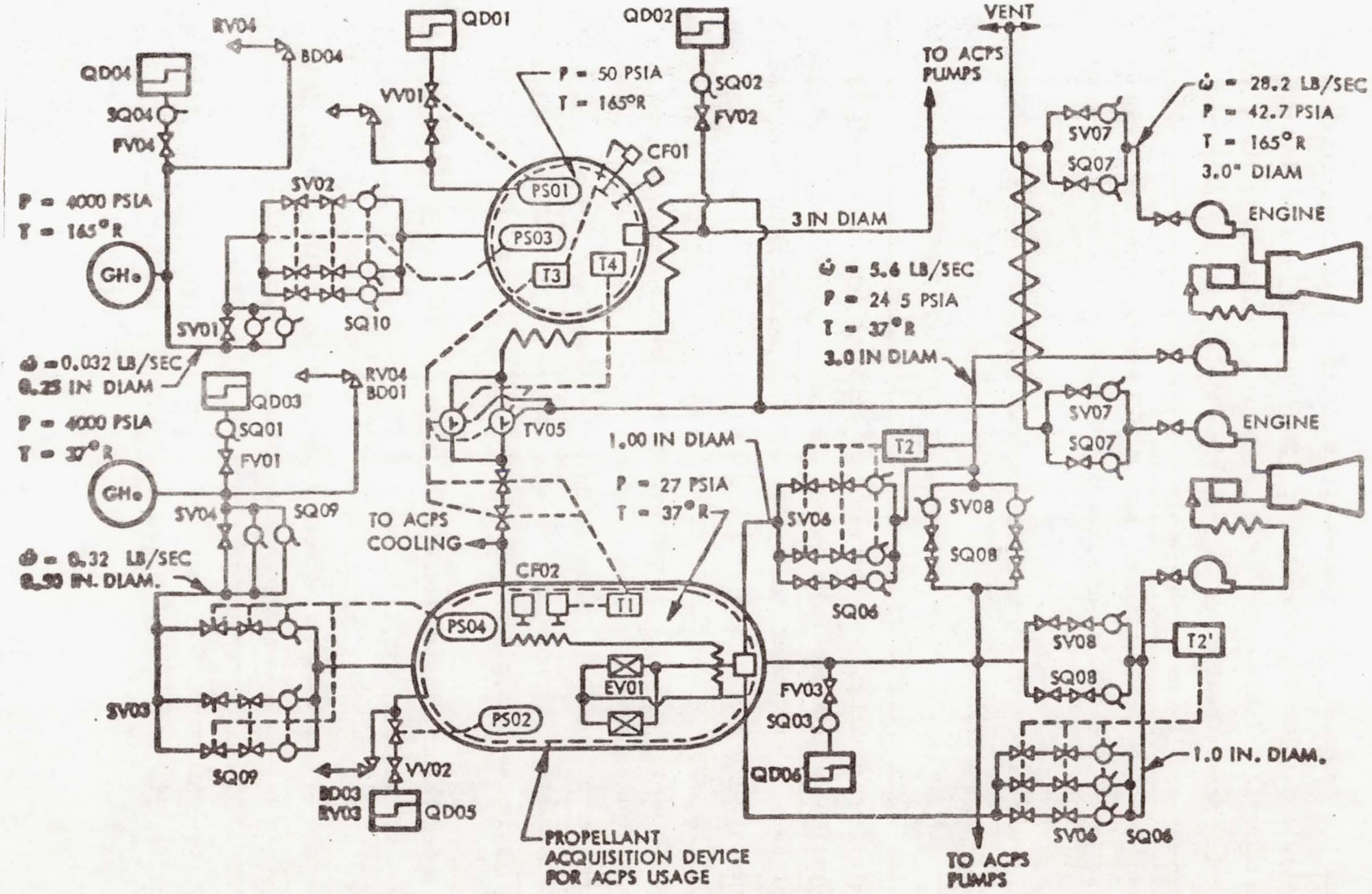

Fig. 4.7-3 Orbit Maneuver Propulsion Schematic

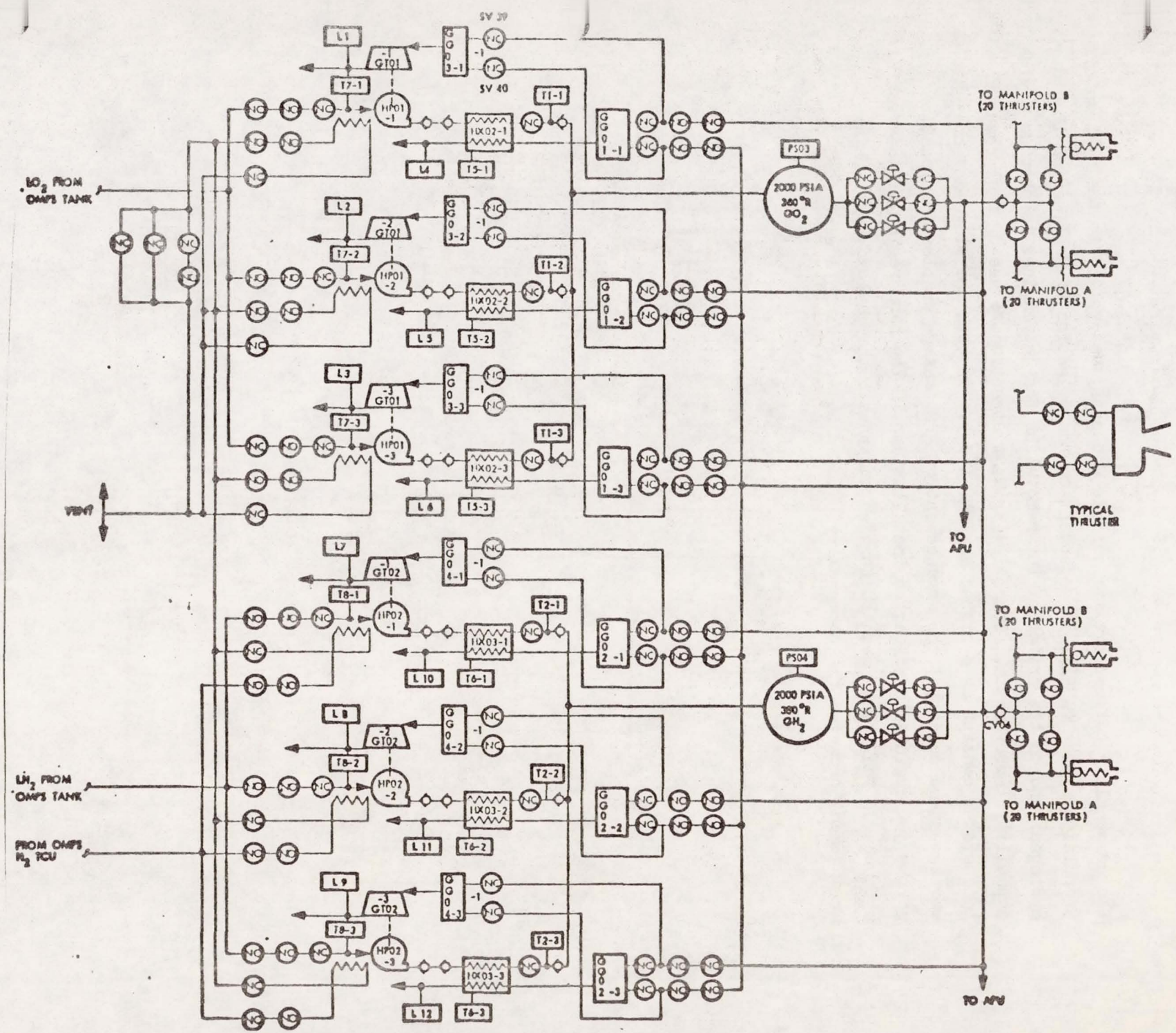

Fig. 4.7-7 Attitude Control Propulsion Schematic

The commonality between orbiter vehicle concpets, as applied to the airbreathing propulsion system (ABPS), is dependent on the selection of fuel (JP or liquid hydrogen) and on the criterion for removal of the jet engines to provide payload capability improvements. The existing NASA directive to use JP fuel results in a system concept as depicted in Fig. 4.7-5. In both the stage-and-one-half and two-stage concepts, the arrangement of MPS tankage enables the location of the jet engines in submerged installations in the base region of both vehicles, with the JP fuel tanks mounted forward underneath the payload bay. This arrangement promotes the commonality between the two vehicle concepts.

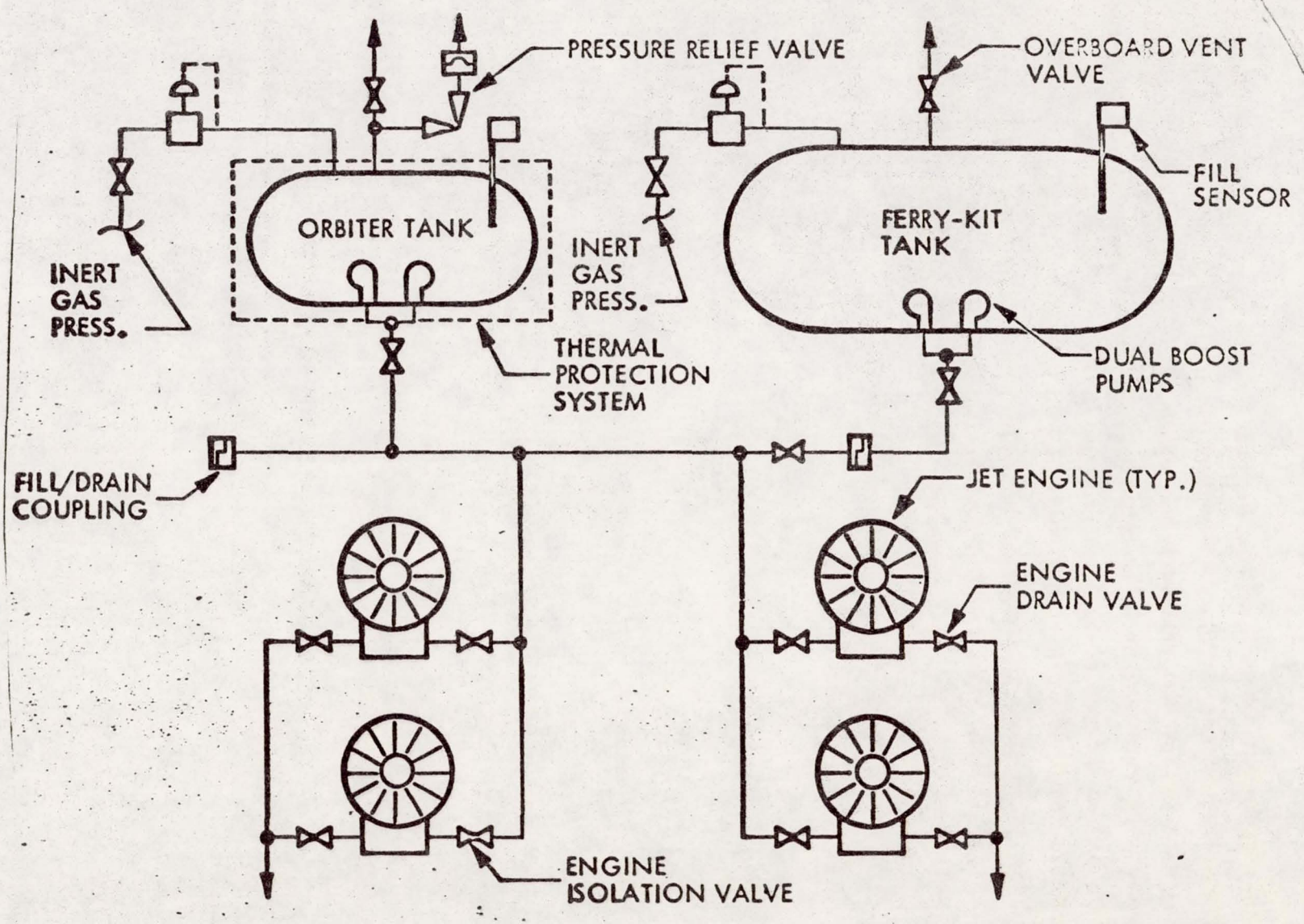

Fig. 4.7-5 Airbreathing Propulsion System (JP-Fuel)

4.8 SUBSYSTEMS

Assumptions and unique differences between the vehicles which impact the subject systems in the conversion process are:

(1) It is assumed in both the stage-and-one-half and two-stage versions that the orbiter is in command of the combined vehicles during launch and ascent until separation.

(2) The dimensions and configurations of the orbiter for the two versions are essentially the same as far as avionics installation factors are concerned.

(3) The major difference in the two versions is in the 9 main engines and 6 jet engines in the stage-and-one-half, and only 2 main engines and 4 jet engines in the two-stage; this will have minor effect upon the stage-and-one-half orbiter avionics in terms of additional hardware in certain subsystems. (This will be detailed in each subsystem discussion.)

(4) There are three attachment points for both vehicle verions. On the stage-and-one-half orbiter, the droptanks are attached at the nose and two points in the rear near the leading edge of the fins in the dorsal area. The two-stage orbiter is attached on the ventral surface with the forward attachment to the booster vehicle just aft of the payload bay and the two rear attachments again near the fins (but below). From the avionics standpoint, the two rear-attachment points can be considered identical (except for some minor wiring changes from top to bottom) in both versions; the forward attachment points probably will require replacement of a substation control (in the DMS) and wiring from the nose to the rear of the payload bay. This is a minor change and probably, by reconfiguring the DMS substation controller, it can be held to a minor wiring change between the two versions.

(5) The stage-and-one-half orbiter has fuel attachments from the droptanks to the vehicle. These are not needed on the two-stage; therefore, the avionics will have to be changed (removal) in going to the two-stage version. Again, this is minor and probably will employ a small amount of wiring change to remove functions (valve and fuel flow control).

(6) There must be a communication link between the orbiter and booster in the two-stage version which is not needed on the stage-and-one-half. In previous studies (pre-Phase B), this was envisioned as either a close-coupled RF link or a special wire connection through one of the mechanical attachment points. Either way, it is not considered a major impact in terms of design, since the data transfer rates will be well within the design of present day systems and the voice link is not a problem. Again, this is a minor change that will be accomplished during vehicle mechanical configuration change.

(7) The centers-of-gravity (cg) of the two orbiters are not identical. The deviation about the cg is approximately the same for each orbiting vehicle; however, insufficient data are available to evaluate the impact of this difference. This study assumes that the avionics impact of cg differences in within the performance envelope of identical hardware designs - i.e., the two versions can utilize the same avionics hardware by employing different software programs.

(8) The antenna placement for communication to the ground must be considered in light of the booster and droptank placement differences. The stage-and-one-half droptanks are essentially on the side and in front of the orbiter, whereas the two-stage is essentially on the bottom and rear of the orbiter. Since this physical constraint exists only during the launch and ascent phases within close proximity of the vehicle to the launch base, a major constraint is not placed on the avionics design. The original stage-and-one-half

concepts in the referenced Engineering Memos have antennas placed in redundancy on the top and bottom of the orbiter vehicle; therefore, no changes are necessary for this consideration (assuming booster antennas complete 360 deg coverage).

In summary, the orbiter avionics hardware is essentially the same for the two orbiter versions. The major operational differences will be during the launch and ascent phases of the mission, primarily in software programming of the onboard computers. Each avionic subsystem is evaluated under appropriate headings.

4.8.1 Orbiter Avionics Subsystems. Data Management Subsystem (EM L2-01-03-M1-1B, see Appendix B).

The number of substation controllers (SSC) in the two-stage orbiter will be reduced because of the smaller number of main engines and jet engines. All of the SSC interface with the vehicle control computer (VCC) where the major software impact will occur. A lesser software change will be required in the command control computer (CCC) and the control display subsystem. These changes are detailed as follows:

The SSCs will reduce by 7 for the main engines and 2 for the jet engines (a total of 9). Considering SSC unit redundancy and signal spares provided in the SSCs, the on-board quantity of SSCs will be reduced by 30. The total quantity of SSCs required for the two-stage orbiter will be 60 (90-30) with a decrease of weight of 130 lb consisting of 120 lb of equipment, and 10 lb of wire/installation hardware. Based upon the estimated 5 watts per SSC, power requirements will decrease by 115 watts during the Boost-Ascent phase and 35 watts during atmospheric operation.

Software modules for like engines are identical; therefore, modification will consist of deleting modules (a reduction in data management load) and a minor modification to the executive language. The magnitude of this change is

within the context of normally occurring growth provisions and should be neglected as a conversion cost (assuming a capability must be maintained during the vehicle lifetime to support software refinements and changing mission requirements).

The total software development for a two-stage orbiter is sufficiently near the stage-and-one-half in the context of the engine differences, to prevent identification of any cost differential between the two concepts.

In the related data management subsystem, differences to accommodate changes in external attach points (tanks vs booster) indicate a minor modification which is within the scope of anticipated in-service modifications required to prevent obsolescence.

4.8.2 Guidance and Navigation (EM L2-01-03-M1-2A, see ACS Fifth Monthly Report)

The guidance and navigation problem of the stage-and-one-half and two-stage orbiters is identical after staging: No hardware difference can be identified as unique to either approach.

4.8.3 Communication (EM L2-01-02-M1-3A, see ACS Fifth Monthly Report)

A communication link that is required between the orbiter and booster for the two-stage vehicle is not required for the stage-and-one-half. This communication link must carry both data (low-speed) and voice in a duplex installation. Adequate growth provisions can be designed into the stage-and-one-half avionics by inclusion of a data transfer IO in the command control computer and a line driver IO in the intercommunication subsystem. Implementation of the provision in service would entail addition of an interconnect cable between the vehicles (with suitable connector) and implementation of a pre-planned software module.

No antenna changes are necessary.

4.8.4 Control and Display (EM L2-01-03-M1-4, see ACS Fourth Monthly Report)

No hardware changes are identifiable as required between the stage-and-one-half and two-stage orbiters. It may be assumed that if an overlay keyboard identification technique is employed, that additional or different overlay designators are required; however, this is a trivial hardware item.

4.8.4.1 Conversion Software. Two conversion software problems have been identified. First are the relatively minor changes in configuration represented by fewer engines and changes in signals caused by replacing booster attach interfaces for droptank interfaces. Considering the anticipation of these modifications during design of the original software, an estimated 2,000 instruction changes whould suffice for the modification. At 0.3 man-months per thousand instructions for utility programming, the cost is trivial.

A second software problem revolves around the impact, in the Vehicle Control Computer (VCC) and Command Control Computer (CCC), expected from different center-of-gravity envelopes between the stage-and-one-half and two-stage orbiters.

The CCC impact may be rationalized as only reflecting differing data transfer rates between the Guidance and Navigation Computer, VCC, and Control Display; this is a relatively minor problem. The impact on the VCC includes a possible requirement for new flight control algorithms, and a new validation and flight test program. At this time, an assumed 80 percent redevelopment of the VCC off line word bank and 40 percent redevelopment of the contained memory is used to scope the task.

Costs for the four major factors of conversion are:

• DP & T	$ 555.8K
• Documentation	60.0K
• Engineering Definition	210.0K
• Integration	280.0K

Some software changes will be required in programmed display formats and content to support changes in checkout and sequence routine which result from the software changes discussed under Data Management, G & N, and Communications. Also, additional film/magnetic tape instruction packs may be required to support the altered configuration. Existing film and tape on-board technical-order-type data will require revision, even if the changes are not sufficient to require additional capacities. The magnitude of this software change is equatable to a TCTO compliance during a schedule overhaul on an in-service airplane (e.g., C-5A with MADAR).

4.8.5 Electrical Power System (EM L2-01-06-M1-1B, see Appendix B)

The two-stage orbiter requires less electrical power during the boost ascent phase than the stage-and-one-half orbiter because of the difference in main engines. No decrease in the shuttle system requirements is realized, since the engines removed from the orbiter must be used on the booster.

The two-stage orbiter requires a peak of 5.6 kw during ascent for main engine electrical power. This reduces the ac subsystem peak load to 11.4 kw, the peak occurring during atmospheric operation instead of during boost ascent for the stage-and-one-half. The electrical power subsystem design can be considered unchanged except for employing smaller ac generators (30 KVA) on the auxiliary power units required for hydraulic power.

4.9 MASS PROPERTIES

The mass properties data contained in this section reflect the impact on weight which resulted from design modification stated in Section 4.1. The four vehicle designs evaluated were: (1) stage-and-one-half reference system, weighed on the basis of mission and program requirements ($1\frac{1}{2}$-B); (2) an interim stage-and-one-half, permitting conversion into a two-stage system ($1\frac{1}{2}$-C); (3) a two-stage, which resulted from the conversion of an interim stage-and-one-half (2-C); and (4) a two-stage reference system based on program and mission requirements (2-B). All of the above weight summaries are shown in Tables 4.9-2 through 4.9-7. Table 4.3-1 reflects the major design commonality and differences between the four vehicles.

The design processes that converted the previous stage-and-one-half baseline designated LS 200-5 to the initial conversion point of $1\frac{1}{2}$-B per Level I requirements has been discussed in some detail in the Eighth Monthly Progress Report in Section 2.3. Basically, the changes necessary to the LS-200-5 system to conform to the Level I requirements can be summarized as follows:

(1) adoption of the 550K engine
(2) replacement of LH_2 by JP-4 for cruiseback system
(3) addition of insulation to droptanks

Common droptank systems were used for both $1\frac{1}{2}$-B and $1\frac{1}{2}$-C and are shown in Table 4.9-1. These tanks were sized for a system GLOW of 3,816,000 lb at a T/w = 1.25 and using a $\lambda' = 0.9615$. By fixing droptank and propellant weight, the orbiter gross payload Δ weights were dependent upon the change in orbiter dry weight. The loss in payload due to converting the $1\frac{1}{2}$-B into the $1\frac{1}{2}$-C is 9004 lb.

The major dry weight differences occur in: (1) body structure, where 5456 lb of additional structure were required in the $1\frac{1}{2}$-C to accommodate higher booster line loads and minor contour modifications; (2) thermal protection system of the $1\frac{1}{2}$-C increased 2617 lb and is discussed in detail in Section 4.3.14, and

(3) contingency and miscellaneous weight changes amounting to 931 lb.

The major emphasis in this study was on converting a stage-and-one-half ($1\frac{1}{2}$-C) into a two-stage (2-C) with minimum impact on two-stage weight, performance, and design, with low conversion cost. The weight differences between the $1\frac{1}{2}$-C and the 2-C are as follows: (1) weight of aero surfaces remain the same, except for removing 1793 lb of flap retraction mechanism; (2) body structures for both $1\frac{1}{2}$-C and 2-C forward sections are essentially the same due to comparable line loads on the body forward of station 1271; the aft sections have large dissimilarities in design and structural arrangement which are discussed in detail in Section 2.3.2; (3) the TPS weights are essentially the same for both designs in that the thicknesses of LI-1500 are common; however, due to body extension of the upper surface on the 2-C, an additional 246 lb of Titanium/Dyna Flex was required; (4) landing gear weight was defined for the $1\frac{1}{2}$-C and not modified for the 2-C, as shown in Table 4.3-1; (5) the main propulsion system required major revision in tankage, plumbing, and engines; due to additional ascent propellant definition, the tank weights increased 10,420 lb; the reduction from 9 main 530K thrust engines to two 632K thrust engines reduced the weight 55,063 lb; plumbing and the subsystems accounted for the remaining 16,744 lb; (6) the cruise propulsion system weights for both $1\frac{1}{2}$-C and 2-C orbiter were evaluated for 6 JTF 22B JP-4 burning engines with associated installation equipment and support structure. Both engine system for $1\frac{1}{2}$-C and 2C were located in the orbiter base and utilized a common ducting system for air intake. Following the weight analysis, it was determined that only four A/B engines would be required for the 2-C which would save over 3500 lb per engine. This correction was not incorporated in the 2-C weight statement but is shown in the summary Table 4.3-2 and Table 4.4-2. (7) the auxiliary propulsion system is of the same design for both $1\frac{1}{2}$-C and 2-C. It is shown in the Fifth Monthly Progress Report and with the exception of tank weight and quantity of thrusters (39 for $1\frac{1}{2}$-C and 34 for 2-C) are equivalent for both designs; and (8) all other subsystems are the same except for surface control weights which reflect slight modification due to slightly reduced upper hinge moments.

The 2B System is shown in Tables 4.9-6 and 4.9-7 and reflect the first in a series of two-stage orbiter and booster weight, and performance evaluations. The 2B orbiter weights were developed in the same manner as those of the 2C with the exception of Body Structure, Thermal Protection System, Landing Gear, Cruise propulsion, Prime Power, and miscellaneous items. The weights of the Body Structure are based upon the actual loading conditions for an ideal two-stage system and show a 1120 lb weight reduction from the 2C. The TPS weight reduction from the 2C of 3496 lb results from reduced wing loading and minor area changes. The landing gear is 3.7 percent of landing weight and is not penalized for stage-and-one-half landing loads. The 2B Cruise Propulsion System consists of 4 JTF 22B-A4 type engines with their associated systems. The Prime Power System was reduced in dry weight due to reduced fuel cell LO_2 - LH_2 requirement. A 10 percent of dry weight less main engines was applied for growth and contingency. Both the 2C and 2B designs carried a 25K payload and used a scaled MAC/DAC MP-7 Booster. The booster scaling and discussion is shown in Section 3.2.2.

Weights which are shown in this section are for the 55 deg/270 NM mission. This mission was selected due to its ease in comparing the two-stage to the one-and-one-half-stage dry and propellant weights; also two-stage engine out abort was still being evaluated at this time.

A detailed discussion and chart depicting weight and potential payloads for the due east, south polar, and 55 deg/270 nm missions are presented in Volume I, Section 5.3 The payloads result from using a booster or droptank designed for 55 deg/270 nm missions; however, the payload weight is added to the orbiter weight until the total mission ΔV requirement is met.

LOCKHEED

Table 4.9-1

MISSION WEIGHT SUMMARY

CONFIGURATION	DROP TANKS (LS200-7 & LS200-8)	BY	Broadhead	DATE	29 April 1971

CODE	SYSTEM	A	B	C	D	E	F	G	H
1	WING GROUP	N/A							
2	TAIL GROUP	N/A							
3	BODY GROUP - Structure	88,460							
4	INDUCED ENVIR PROTECTION	7,150							
5	LANDING, RECOVERY, DOCKING	N/A							
6	PROPULSION - ASCENT Plumbing	6,106							
7	PROPULSION - CRUISE	N/A							
8	PROPULSION - AUXILIARY	N/A							
9	PRIME POWER	N/A							
10	ELECT CONVER & DISTR. Instr.	325							
11	HYDRA CONVER & DISTR	N/A							
12	SURFACE CONTROLS	N/A							
13	AVIONICS	N/A							
14	ENVIRONMENTAL CONTROL	N/A							
15	PERSONNEL PROVISIONS	N/A							
16	RANGE SAFETY & ABORT	N/A							
17	BALLAST	0							
18	GROWTH/UNCERTAINTY - 10% Dry	10,204							
19									
	SUBTOTAL (DRY WEIGHT)	112,245							
20	PERSONNEL	N/A							
21	CARGO	N/A							
22	ORDNANCE	N/A							
23	RESIDUAL FLUIDS	10,563							
24									
	SUBTOTAL(INERT WEIGHT)	122,808	122,808						
25	RESERVE FLUIDS								
26	Losses - Prelaunch	(31,122)	0						
27	PROPELLANT - ASCENT	3,065,024	0						
28	PROPELLANT - CRUISE	N/A							
29	PROPELLANT-- MANEUV/ACS	N/A							
30									
	Drop Tank Total Wt. Lb.	3,187,832	122,808						
	TOTAL(GROSS-WEIGHT)LB.								

DESIGNATORS:

EVENTS
A LAUNCH WEIGHT
B IGNITION (EFFECTIVE) SEPARATION
C
D
E
F
G
H

NOTES & SKETCHES:

$\lambda' = .9615$

LOCKHEED

Table 4.9-2

1-1/2-B	MISSION WEIGHT SUMMARY								
CONFIGURATION	STAGE-AND-ONE-HALF ORBITER/DROPTANKS			BY	Broadhead	DATE	29 April 1971		
CODE	SYSTEM LS 200-7	A	B	C	D	E	F	G	H
1	WING GROUP								
2	TAIL GROUP	17914							
3	BODY GROUP	56529							
4	INDUCED ENVIR PROTECTION	36769							
5	LANDING, RECOVERY, DOCKING	11914							
6	PROPULSION - ASCENT	105160							
7	PROPULSION - CRUISE	19140							
8	PROPULSION - AUXILIARY	6871							
9	PRIME POWER	1620							
10	ELECT CONVER & DISTR	3704							
11	HYDRA CONVER & DISTR	2073							
12	SURFACE CONTROLS	4270							
13	AVIONICS	3762							
14	ENVIRONMENTAL CONTROL	1274							
15	PERSONNEL PROVISIONS	210							
16	RANGE SAFETY & ABORT								
17	BALLAST								
18	GROWTH/UNCERTAINTY	19725							
19									
	SUBTOTAL (DRY WEIGHT)	290935							
20	PERSONNEL	725							
21	CARGO	22520							
22	ORDNANCE								
23	RESIDUAL FLUIDS	3474							
24									
	SUBTOTAL(INERT WEIGHT)	317654	317654	317654	317654	317654	317654	317654	317654
25	RESERVE FLUIDS	10609	10609	10609	10609	2550	2550	2550	2550
26	IN FLIGHT LOSSES	9497	9169	4653	4571	2329	492	467	0
27	PROPELLANT - ASCENT	239229	239229	239229	0	0	0	0	0
28	PROPELLANT - CRUISE	12234	12234	12234	12234	12234	12234	12234	0
29	PROPELLANT -- MANEUV/ACS	38946	38946	38946	38946	16737	12486	465	0
30									
	TOTAL(GROSS-WEIGHT)LB.	628169	627841	623325	384019	351504	345416	333370	320204
	DROPTANKS - DRY	112245							
	DROPTANKS PROP - RESID.	10563							
	SUBTOTAL (DROPTANK INERT)	122808							
	DROPTANK PROP. - ASCENT	3065023							
	SUBTOTAL (DROPTANK GROSS)	3187832							
	TOTAL (COMPOSITE VEHICLE)	3816000							

DESIGNATORS:

EVENTS

A	ORBITER/DROPTANKS LAUNCH
B	PRIOR SEPARATION
C	AFTER SEPARATION
D	INJECTION 50 nm x 100 nm
E	ON-ORBIT 260 nm x 260 nm
F	PRIOR RETRO
G	AFT RETRO
H	LANDING

NOTES & SKETCHES:

Condition	Center of Gravity (% of 1800 in.)
Payload Out, ABPS in (Entry)	77.1
Payload Out, ABPS in (Landing)	78.1
40K Payload, ABPS Out (Entry)	73.9
40K Payload, ABPS Out (Landing)	73.9

Table 4.9-3

MISSION WEIGHT SUMMARY

CONFIGURATION: 1-1/2-C STAGE-AND-ONE-HALF | BY: SGB | DATE: 29 April 1971

CODE	SYSTEM LS 200-8	A	B	C	D	E	F	G	H
1	WING GROUP	N/A							
2	TAIL GROUP	17941							
3	BODY GROUP	61985							
4	INDUCED ENVIR PROTECTION	39386							
5	LANDING, RECOVERY, DOCKING	11914							
6	PROPULSION - ASCENT	105160							
7	PROPULSION - CRUISE	19140							
8	PROPULSION - AUXILIARY	6871							
9	PRIME POWER	1620							
10	ELECT CONVER & DISTR	3704							
11	HYDRA CONVER & DISTR	2073							
12	SURFACE CONTROLS	4270							
13	AVIONICS	3762							
14	ENVIRONMENTAL CONTROL	1274							
15	PERSONNEL PROVISIONS	210							
16	RANGE SAFETY & ABORT								
17	BALLAST								
18	GROWTH/UNCERTAINTY	20629							
19									
	SUBTOTAL (DRY WEIGHT)	299939							
20	PERSONNEL	725							
21	CARGO	13516							
22	ORDNANCE	3474							
23	RESIDUAL FLUIDS								
24									
	SUBTOTAL(INERT WEIGHT)	317654	317654	317654	317654	317654	317654	317654	317654
25	RESERVE FLUIDS	10609	10609	10609	10609	2550	2550	2550	2550
26	IN FLIGHT LOSSES	9497	9169	4653	4571	2329	492	467	0
27	PROPELLANT - ASCENT	239229	239229	239229	0	0	0	0	0
28	PROPELLANT - CRUISE	12234	12234	12234	12234	12234	12234	12234	0
29	PROPELLANT--MANEUV/ACS	38946	38946	38946	38946	16737	12486	465	0
30	SUBTOTAL (ORBITER GROSS)	628169	627841	623325	384014	351504	345416	333370	320204
	DROPTANK - DRY	112245							
	TOTAL GROSS RESIDUAL	10563							
	SUBTOTAL (DROP TANK INERT	122808							
	DROPTANK PROP-ASCENT	3065023							
	SUBTOTAL DROPTANK GROSS	3187832							
	TOTAL (COMPOSITE VEHICLE) LB	3816000							

DESIGNATORS:

EVENTS

A	LAUNCH WEIGHT
B	PRIOR SEPARATION
C	AFTER SEPARATION
D	INJECTION 260 nm x 260 nm
E	ON-ORBIT 260 nm x 260 nm
F	PRE-RETRO
G	ENTRY
H	LANDING

NOTES & SKETCHES:

Condition	Center of Gravity (% of 1800 in.)
Payload Out, ABPS in (Entry)	77.4
Payload Out, ABPS in (Landing)	78.3
40K Payload, ABPS Out (Entry)	74.2
40K Payload, ABPS Out (Landing)	74.2

Table 4.9-4

WEIGHT SUMMARY (LAUNCH CONDITION)

CONFIGURATION	2C Two-Stage		BY	Broadhead	DATE	29 Apr 71

CODE	SYSTEM LS 400-5	ITEM OR MODULE BOOSTER	EST	CAL	ACT	ORBITER	EST	CAL	ACT
1	Wing Group	49,316	100			N/A	100		
2	Tail Group	13,935				16,324			
3	Body Group	136,861				46,360			
4	Induced Envir Protection	64,831				39,632			
5	Landing, Recovery, Docking	21,590				11,914			
6	Propulsion - Ascent	*111,627				43,773			
7	Propulsion - Cruise	35,957				19,036	**		
8	Propulsion - Auxiliary	3,680				6,418			
9	Prime Power	1,028				1,620			
10	Elect Conver & Distr	1,235				3,704			
11	Hydra Conver & Distr	5,040				2,073			
12	Surface Controls	4,910				4,270			
13	Avionics	3,808				3,678			
14	Environmental Control	495				1,274			
15	Personnel Provisions	147				210			
16	Range Safety & Abort	0				0			
17	Ballast	0				0			
18	Growth/Uncertainty	39,128	100			18,138	100		
19									
	SUBTOTAL (Dry Weight)	493,488				218,427			
20	Personnel	400	100			725	100		
21	Cargo	0				25,000			
22	Ordnance	0				0			
23	Residual Fluids	8,810	100			3,541	100		
24									
	SUBTOTAL (Inert Weight)	502,698				247,693			
25	Reserve Fluids	41,251				8,187	100		
26	In Flight Losses	20,319				8,280			
27	Propellant - Ascent	2,768,187				546,268			
28	Propellant - Cruise	89,169				9,542			
29	Propellant - Maneuv/Acs	5,038	100			30,702	100		
30									
	TOTAL (Gross-Weight) LB	3,426,662				850,672			

DESIGNATIONS:

Class of Weight
EST - Percent Estimated Weight
CAL - Percent Calculated Weight
ACT - Percent Actual Weight

NOTES & SKETCHES:

Gross Launch Wt = 4,277,334

* 11 Main Engs of 550K S.L. Thrust ea.

** 6 Cruise Engines (finally reduced to 4)

Table 4.9-5

MISSION WEIGHT SUMMARY

CONFIGURATION: 2C Two-Stage Orbiter | BY: Broadhead | DATE: 29 April 1971

CODE	SYSTEM LS 400-5	A	B	C	D	E	F	G	H
1	WING GROUP	N/A							
2	TAIL GROUP	16324							
3	BODY GROUP	46360							
4	INDUCED ENVIR PROTECTION	39632							
5	LANDING, RECOVERY, DOCKING	11914							
6	PROPULSION - ASCENT	43773							
7	PROPULSION - CRUISE	19036							
8	PROPULSION - AUXILIARY	6418							
9	PRIME POWER	1620							
10	ELECT CONVER & DISTR	3704							
11	HYDRA CONVER & DISTR	2073							
12	SURFACE CONTROLS	4270							
13	AVIONICS	3678							
14	ENVIRONMENTAL CONTROL	1274							
15	PERSONNEL PROVISIONS	210							
16	RANGE SAFETY & ABORT	0							
17	BALLAST	0							
18	GROWTH/UNCERTAINTY	18138							
19									
	SUBTOTAL (DRY WEIGHT)	218427							
20	PERSONNEL	725							
21	CARGO	25000							
22	ORDNANCE	0							
23	RESIDUAL FLUIDS	3541							
24									
	SUBTOTAL(INERT WEIGHT)	247693	247693	247693					
25	RESERVE FLUIDS	8187	2148	2148					
26	IN FLIGHT LOSSES	8280	0	0					
27	PROPELLANT - ASCENT	546268	0	0					
28	PROPELLANT - CRUISE	9542	9542	0					
29	PROPELLANT — MANEUV/ACS	30702	0	0					
30									
	TOTAL(GROSS-WEIGHT)LB.	850672	259383	249841					

DESIGNATORS:

EVENTS

A LAUNCH WEIGHT - ORBITER
B ENTRY
C LANDING
D
E
F
G
H

NOTES & SKETCHES: Center of Gravity

Condition	% of 1800 IN.
Payload Out, ABPS In (Entry)	74.9%
Payload Out, ABPS In(Landing)	75.8%
40K Payload, ABPS Out (Entry)	70.8%
40K Payload, ABPS Out (Landing)	70.8%

Table 4.9-6

WEIGHT SUMMARY (LAUNCH CONDITION)									
CONFIGURATION	2B TWO-STAGE		BY	SGB	DATE	29 April 1971			
CODE	SYSTEM LS 400-6	ITEM OR MODULE							
		BOOSTER	EST	CAL	ACT	ORBITER	EST	CAL	ACT
1	Wing Group	46000	100			N/A	100		
2	Tail Group	12998				16324			
3	Body Group	128371				45240			
4	Induced Envir Protection	61156				36134			
5	Landing, Recovery, Docking	20261				8917			
6	Propulsion - Ascent	101605*				43776			
7	Propulsion - Cruise	35459				13569			
8	Propulsion - Auxiliary	3680				6418			
9	Prime Power	1028				1511			
10	Elect Conver & Distr	1235				3704			
11	Hydra Conver & Distr	5040				2073			
12	Surface Controls	4910				4270			
13	Avionics	3808				3672			
14	Environmental Control	495				1274			
15	Personnel Provisions	147				210			
16	Range Safety & Abort	0				0			
17	Ballast	0				0			
18	Growth/Uncertainty	36925	100			16820	100		
19									
	SUBTOTAL (Dry Weight)	463128	100			203918	100		
20	Personnel	400				725			
21	Cargo	0				25000			
22	Ordnance	0				0			
23	Residual Fluids	8187	100			3515	100		
24									
	SUBTOTAL (Inert Weight)	471715	100			233158	100		
25	Reserve Fluids	37888				7735			
26	In Flight Losses	19109				8285			
27	Propellant - Ascent	2595357				546617			
28	Propellant - Cruise	80756				8082			
29	Propellant - Maneuv/Acs	5038	100			28987	100		
30									
	TOTAL (Gross-Weight) LB	3209863				833760			

DESIGNATIONS:

Class of Weight
EST - Percent Estimated Weight
CAL - Percent Calculated Weight
ACT - Percent Actual Weight

NOTES & SKETCHES:

Gross Launch Wt - 4043623

*10 Main engines of 550K S.L. Thrust Each

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Table 4.9-7

MISSION WEIGHT SUMMARY

CONFIGURATION	2B TWO-STAGE ORBITER	BY	Broadhead	DATE	29 April 1971

CODE	SYSTEM LS 400-6	A	B	C	D	E	F	G	H
1	WING GROUP	N/A							
2	TAIL GROUP	16324							
3	BODY GROUP	45240							
4	INDUCED ENVIR PROTECTION	36134							
5	LANDING, RECOVERY, DOCKING	8917							
6	PROPULSION - ASCENT	43776							
7	PROPULSION - CRUISE	13569							
8	PROPULSION - AUXILIARY	6418							
9	PRIME POWER	1511							
10	ELECT CONVER & DISTR	3704							
11	HYDRA CONVER & DISTR	2073							
12	SURFACE CONTROLS	4270							
13	AVIONICS	3672							
14	ENVIRONMENTAL CONTROL	1274							
15	PERSONNEL PROVISIONS	210							
16	RANGE SAFETY & ABORT	0							
17	BALLAST	0							
18	GROWTH/UNCERTAINTY	16820							
19									
	SUBTOTAL (DRY WEIGHT)	203918							
20	PERSONNEL	725							
21	CARGO	25000							
22	ORDNANCE								
23	RESIDUAL FLUIDS	3515							
24									
	SUBTOTAL(INERT WEIGHT)	233158	233158	233158					
25	RESERVE FLUIDS	7735	1901	1901					
26	IN FLIGHT LOSSES	8285	0	0					
27	PROPELLANT - ASCENT	546617	0	0					
28	PROPELLANT - CRUISE	8982	8982	0					
29	PROPELLANT— MANEUV/ACS	28987	0	0					
30									
	TOTAL(GROSS-WEIGHT)LB.	833760	244041	235059					

DESIGNATORS:

EVENTS

A LAUNCH WEIGHT - ORBITER
B ENTRY
C LANDING
D
E
F
G

NOTES & SKETCHES:

Condition	Center of Gravity (% of 1800 in.)
Payload Out, ABPS in (Entry)	75.4
Payload Out, ABPS in (Landing)	76.4
40K Payload ABPS Out (Entry)	71.3
40K Payload ABPS Out (Landing)	71.3

4.10 OPERATIONS

This subsection presents the results of the analysis performed to determine the cost and complexity of converting to a two-stage shuttle operation after one-half the ten-year mission model flights (750 total) have been flown by stage-and-one-half shuttles.

4.10.1 Candidate Site and Operational Concepts

Concept I - KSC: Maximum use of existing facilities and equipment. Baseline stage-and-one-half operations, droptank manufactured at Michoud and shipped by barge to KSC, modular maintenance facility planned for booster growth, two new LUTs for stage-and-one-half and two new LUTs for two-stage, new pad flame deflector for stage-and-one-half and two-stage.

Advantage	Disadvantage
Lowest cost method	New LUTs

Concept II - KSC and New Site: Transition to new two-stage peculiar main base. Baseline stage-and-one-half operations, droptank manufactured at Michoud and shipped by barge to KSC. Baseline two-stage: Maintenance and Assembly building, horizontal mate, horizontal transporter, on-pad built-in erection, launch mount adapter.

Advantage	Disadvantage
Eventual reusable shuttle - peculiar main base	High-cost new facilities
Low initial shuttle facility cost	Two separate facilities required for program

Concept III - New Site: Shuttle-peculiar main base. Maintenance and Assembly Building designed for orbiter maintenance wing with provision for booster maintenance wing, vehicle mate area designed suitable for horizontal tank-to-vehicle mate or booster-to-orbiter mate. Launch pads common to either vehicle. Two-stage requires different launch mount adapter and additional swing arms on personnel access tower.

Advantage	Disadvantage
Reusable shuttle-peculiar main base	High initial facility cost First vertical droptank flight over land

Concept IV - KSC. Existing facilities for stage-and-one-half - new facilities for two-stage. Baseline stage-and-one-half operations droptank manufactured at Michoud and shipped by barge to KSC, modular maintenance facility planned for booster growth and located at new site area, refurbished stage-and-one-half vehicle delivered to VAB on connecting roadway for vertical tank mate.

Two-stage conversion - MAB facility expanded to include horizontal vehicle mating area and booster maintenance wing; two new KSC launch pads with built-in erection, horizontal pad delivery.

Advantage	Disadvantage
Eventual reusable shuttle-peculiar main base Low initial shuttle facility cost	Cost of new KSC launch pads

4.10.2 Conversion Cost Analysis

The facility and equipment costs for each of the candidate concepts are shown in Fig. 4.10-1. The costs shown for the baseline stage-and-one-half were taken from the facility cost analysis included in Section 2.17.1 of this report. The cost increases shown are rough estimates for the additional required facilities or modifications to provide ground support operations for a two-stage shuttle vehicle. The increased costs for recovery facilities cover additional hangar and ground handling equipment. The increased costs for maintenance facilities are for new booster maintenance facilities and for either Vertical Assembly Building (VAB) modifications (KSC) or for new horizontal mate facilities (New Site). The increased costs for launch facilities cover items such as new LUTs, new flame deflectors, LCC modifications, and in some cases totally new pads.

Fig. 4.10-2 shows the turnaround manpower associated with the four operational concepts. This evaluation is based upon a fully operational fleet at a launch rate of 75 flights per year. The ratio of support personnel to on-line personnel remains constant at 6 to 1 for all concepts. The quantities shown for the baseline stage-and-one-half were taken from the manpower loading study included in Section 2.17.1 of this report. The manpower increases shown are required to support the additional booster vehicle which is not required for stage-and-one-half operation. The manpower decreases are the result of operating efficiencies associated with totally new shuttle-peculiar facilities. By combining the effects of the two parameters measured in this cost analysis (facility cost increases and manpower increases), a comparison of the four candidate concepts can be made. This comparison is shown in Fig. 4.10-3 by converting manpower to total program cost and adding it to the nonrecurring facility cost increases. The costs shown in this figure are for concept comparison only and are not meant to be the total costs associated with conversion.

	BASELINE 1 1/2-Stage AT KSC	INCREASES FOR 2-STAGE			COST OF NEW SITE ONLY	
		CONCEPT I	CONCEPT II	CONCEPT IV	CONCEPT III	
		Saturn/ Apollo	New Site 2-Stage	KSC New Pads	1 1/2-Stage New Site	2-Stage Increases
RECOVERY FACILITIES	$ 21.78 M	$ 1.5 M	$ 22.28 M	$ 1.5 M	$ 21.78 M	$ 1.5 M
Contingencies*	14.4 M	0.99M	14.7 M	0.99 M	14.4 M	0.99M
MAINTENANCE FACILITIES	$ 55.3 M	$ 22.0 M	$198.1 M	$ 41.5 M	$178.0 M	$ 20.6 M
Contingencies*	36.5 M	14.5 M	131.0 M	27.4 M	117.5 M	13.6 M
LAUNCH FACILITIES	$ 28.5 M	$ 25.5 M	$283.60 M	$220.1 M	$234.4 M	$ 49.2 M
Contingencies*	18.8 M	16.8 M	187. M	145. M	154.5 M	32.4 M
SUBTOTAL	$175.28 M	$+81.29 M	$+836.68 M	$+436.49 M	$720.58 M	$+118.29 M
TOTAL FOR CONCEPT		$256.57M	$1011.96 M	$ 611.77 M		$ 838.87 M

*Contingencies = Supervision, Design, Administration, Activation and Logistics = 66%

Fig. 4.10-1 Facility and Equipment Cost Comparison

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	BASELINE 1 1/2-Stage AT KSC	CHANGES FOR 2-STAGE			CHANGES FOR NEW SITE ONLY	
		CONCEPT I	CONCEPT II	CONCEPT IV	CONCEPT III	
		Saturn/ Apollo	New Site 2-Stage	KSC New Pad	1 1/2-Stage New Site	2-Stage Increase
ON-LINE						
LANDING FIELD	36	NC	NC	NC	NC	NC
PURGE AREA	28	+28	+28	+28	NC	+28
MAINT AREA	316	+173	+173	+173	NC	+173
MATING AREA	56	+28	-8	-8	-20	-8
LAUNCH PADS AND LCC	78	+39	NC	NC	-30	NC
SUPPORT						
RECOVERY AREA	226	+41	+41	+41	NC	+41
MAINT AREA	2,305	+1,560	+1,246	+1,246	-140	+1,246
LAUNCH AREA	550	+275	+64	+64	-210	+64
TOTAL	3 595	+2,144	+1,544	+1,544	-400	+1,544

*75 FLIGHTS/YEAR 5-DAY WEEK, SINGLE SHIFT

Fig. 4.10-2 Ground Crew Size Comparison*

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GROUNDRULES

MANYEAR = $20,000
75 FLIGHTS / YEAR, 5-DAY WEEK,
SINGLE SHIFT, 10-YEAR PROGRAM

CONCEPT	FACILITY COST	MANPOWER COST*	COMPARITIVE COST
I	$0.257 B	$0.933 B	$1.190 B
II	$1.012 B	$0.873 B	$1.885 B
III	$0.839 B	$0.833 B	$1.672 B
VI	$0.612 B	$0.873 B	$1.485 B

*5 YEARS OF 1 1/2 STAGE LAUNCHES & 5 YEARS OF 2-STAGE LAUNCHES

Fig. 4.10-3

Fig. 4.10-3 Stage-and-One-Half To Two-Stage Operations Study Results

Concept I has the lowest conversion cost, while Concept II shows a 40 percent increase over Concept I, Concept III shows a 30 percent increase over Concept I, and Concept IV shows a 20 percent increase over Concept I.

Converting from a stage-and-one-half operation to a two-stage operation using Concept I increases the facility cost by $81 million and the manpower costs by $214 million over a pure stage-and-one-half.

4.11 SYSTEM COST ANALYSIS

System costs were estimated for feasible alternate plans for implementing the conversion system. Plans were analyzed for phasings which allowed from 3 to 5 years of interim operations as a stage-and-one-half and for booster DDT&E spans of 6 to 7 years. The details of these analyses are described in Volume IV.

The annual funding profiles for the plans investigated are shown in Figs 4.11-1 through 4.11-6. The cost characteristics of all plans are summarized in Table 4.11-1. Plans 1 and 2 were rejected because of the number of years which they exceed a $1.2 billion funding level. Plans 5 and 6 were rejected because of their high total program cost and the fact that they require the production of additional stage-and-one-half orbiters to support the program. Of the two remaining plans, Plan 4 was selected as the approach offering the best possibility for achieving an improved funding profile. This plan was then adjusted by modifying the schedule at lower levels of the WBS to smooth out the funding profile. The smoothed Plan 4 arrived at is shown in Fig. 4.11-7. Peak funding level was reduced to $1.29 billion but the program requires funding levels in excess of $1.1 billion for 4 years.

Plan 4 allows 4 years of interim operations as a stage-and-one-half and 3 years of stage-and-one-half orbiter development before booster development starts. Summary cost data for this plan are as follows:

DDT&E	$ 8,386M
Recurring Production	561M
Recurring Operations	2,075M
Total Program	$11,022M
1-1/2-C Orbiter First Unit (Including engines)	$ 144.7M
2-C Orbiter First Unit (Including engines)	106.9M
2-C Booster First Unit (Including engines)	169.7M

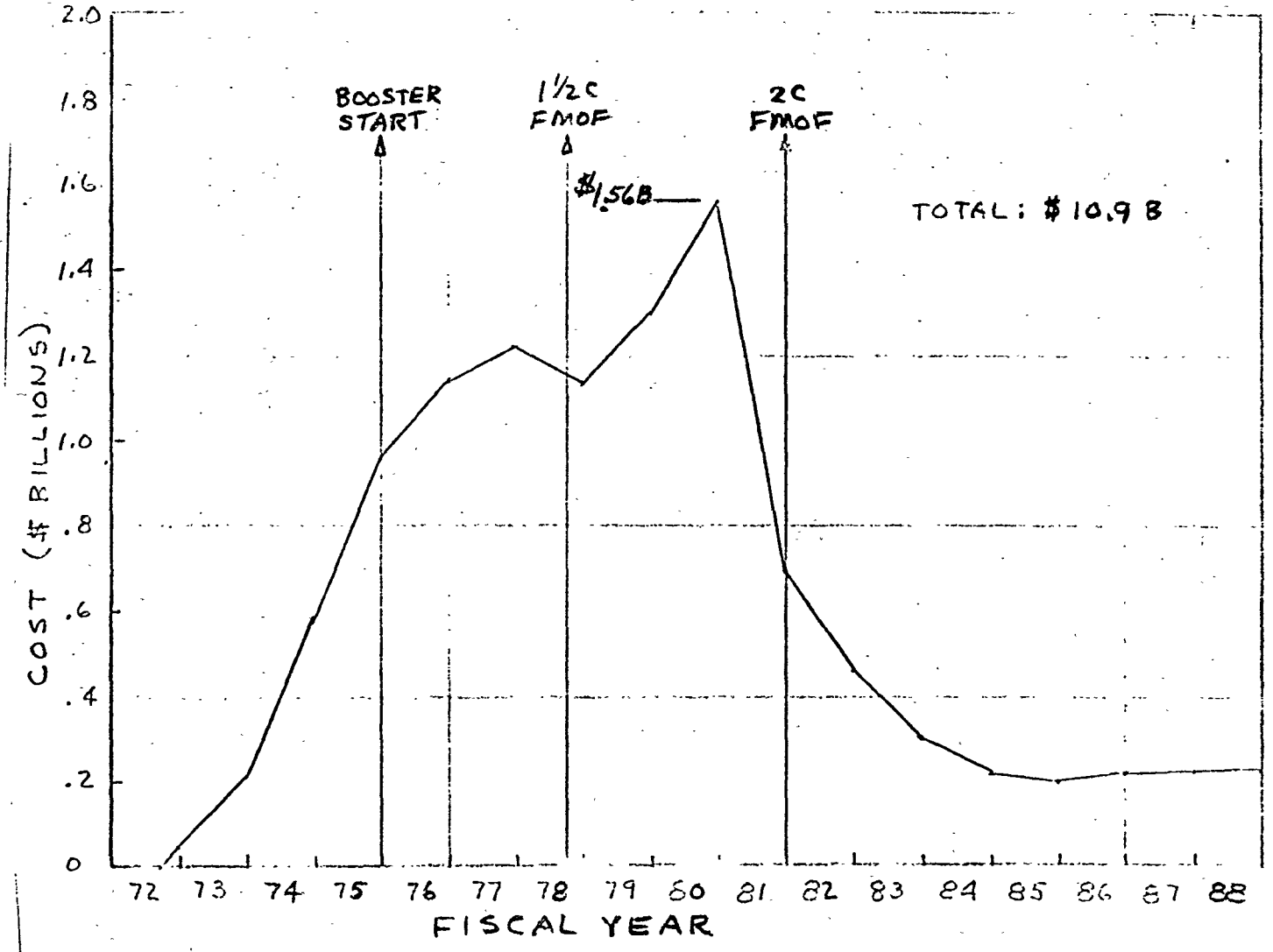

Fig. 4.11-1 Plan 1 Conversion System Annual Cost
3 Yr Interim Operations, 6 Yr Booster Development

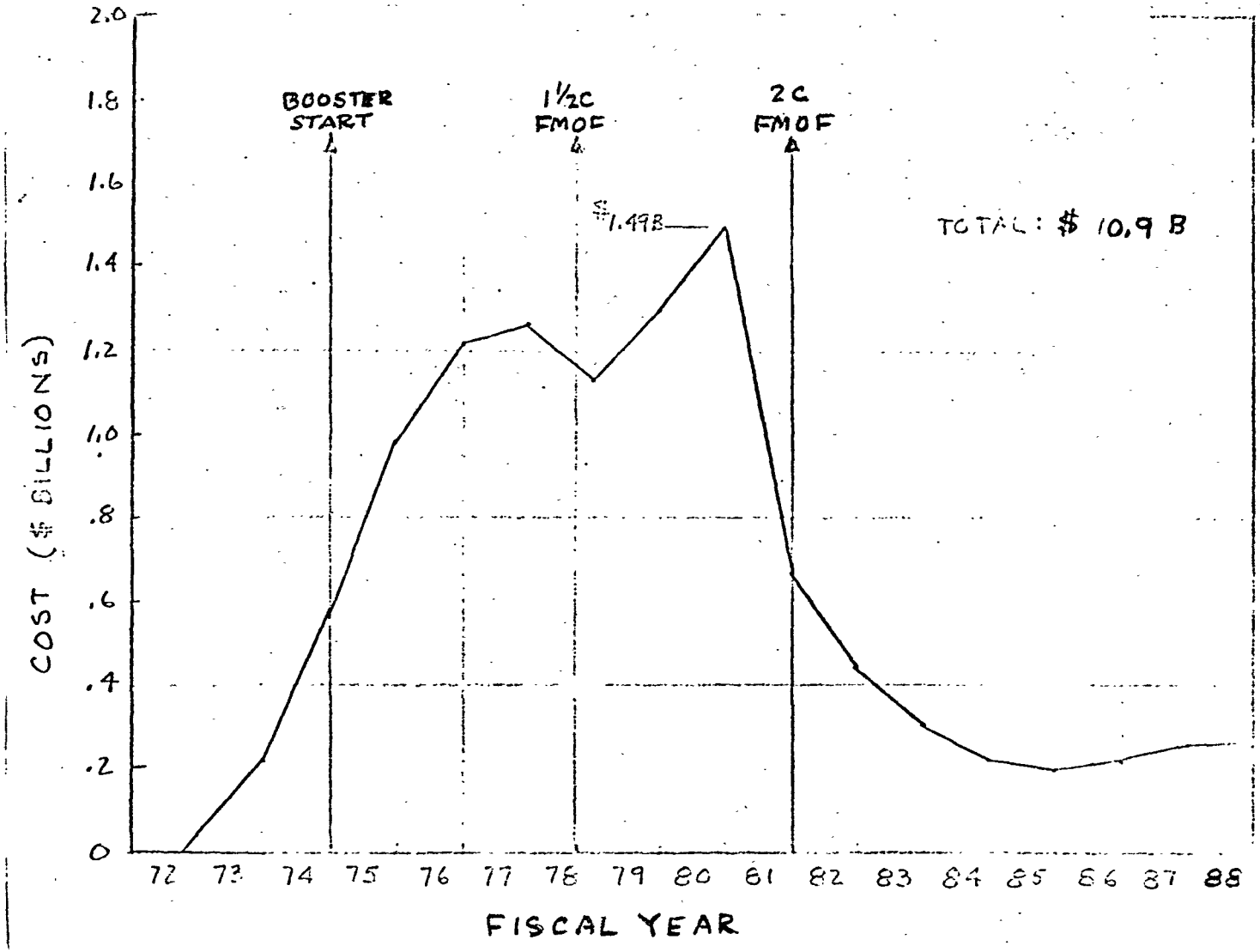

Fig. 4.11-2 Plan 2. Conversion System Annual Cost
3 Yr Interim Operations, 7 Yr Booster Development

Fig. 4.11-2

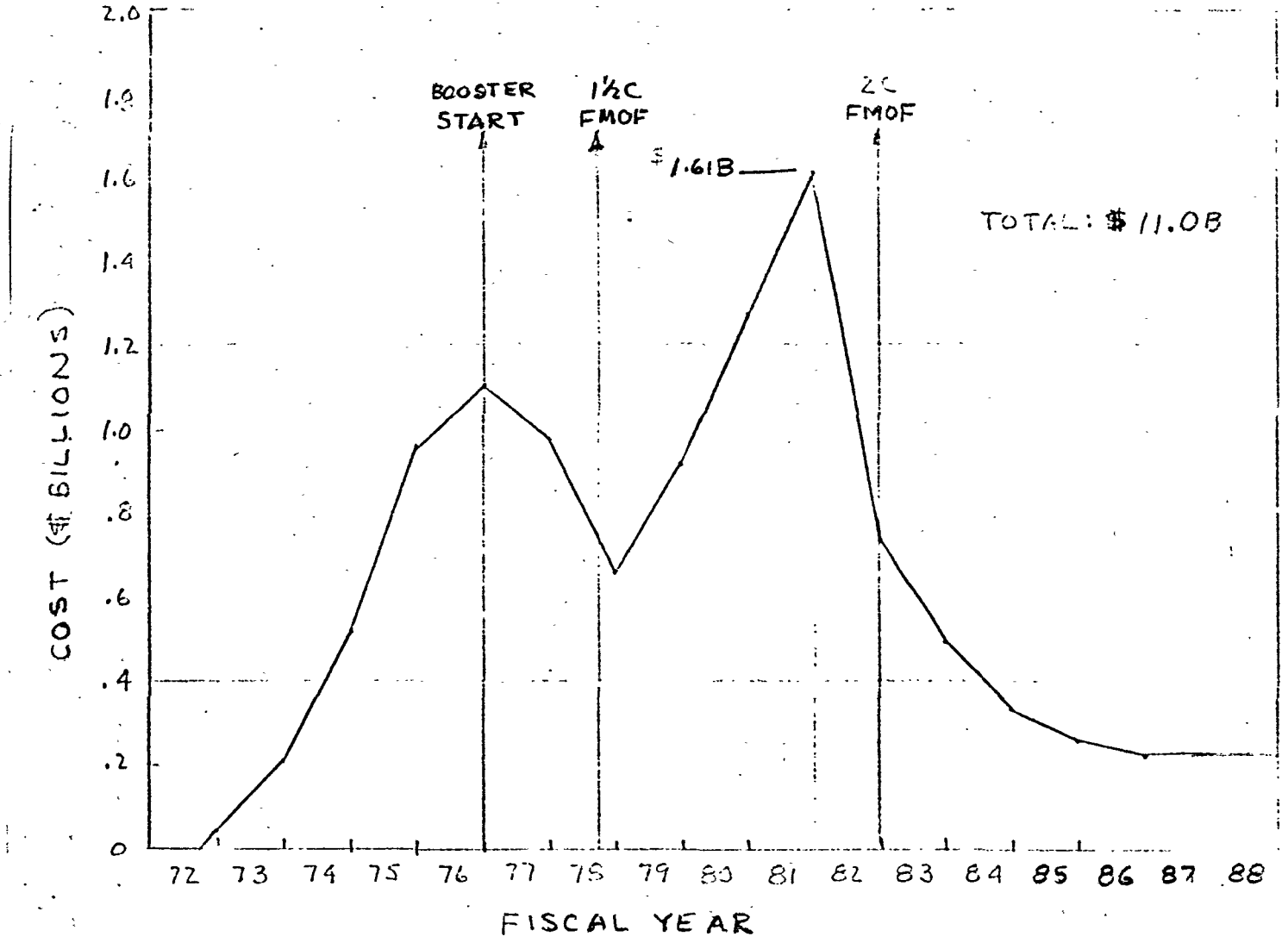

Fig. 4.11-3 Plan 3. Conversion System Annual Cost
4 Yr Interim Operations, 6 Yr Booster Development

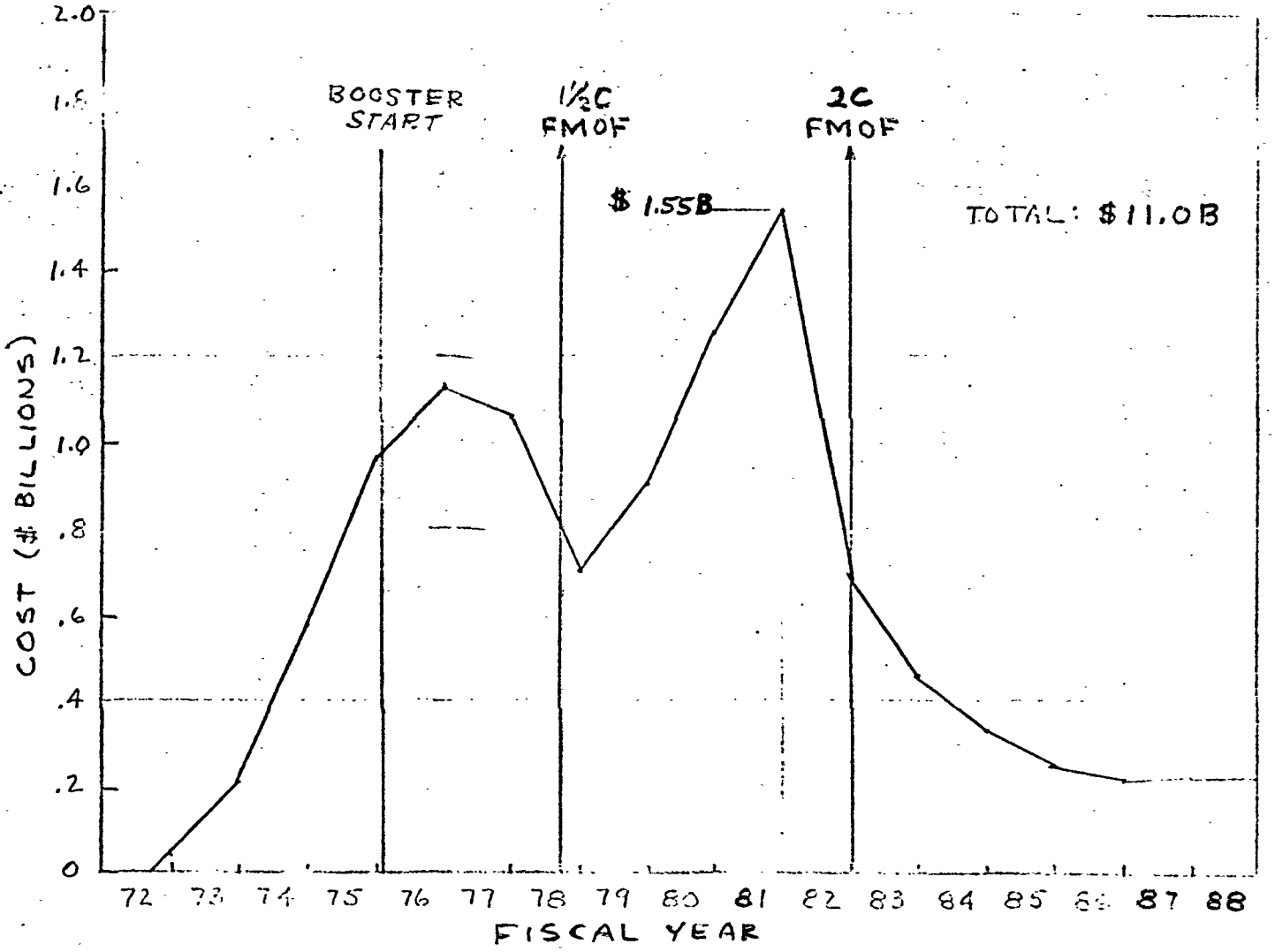

Fig. 4.11-4

Fig. 4.11-4 Plan 4. Conversion System Annual Cost
4 Yr Interim Operations, 7 Yr Booster Development

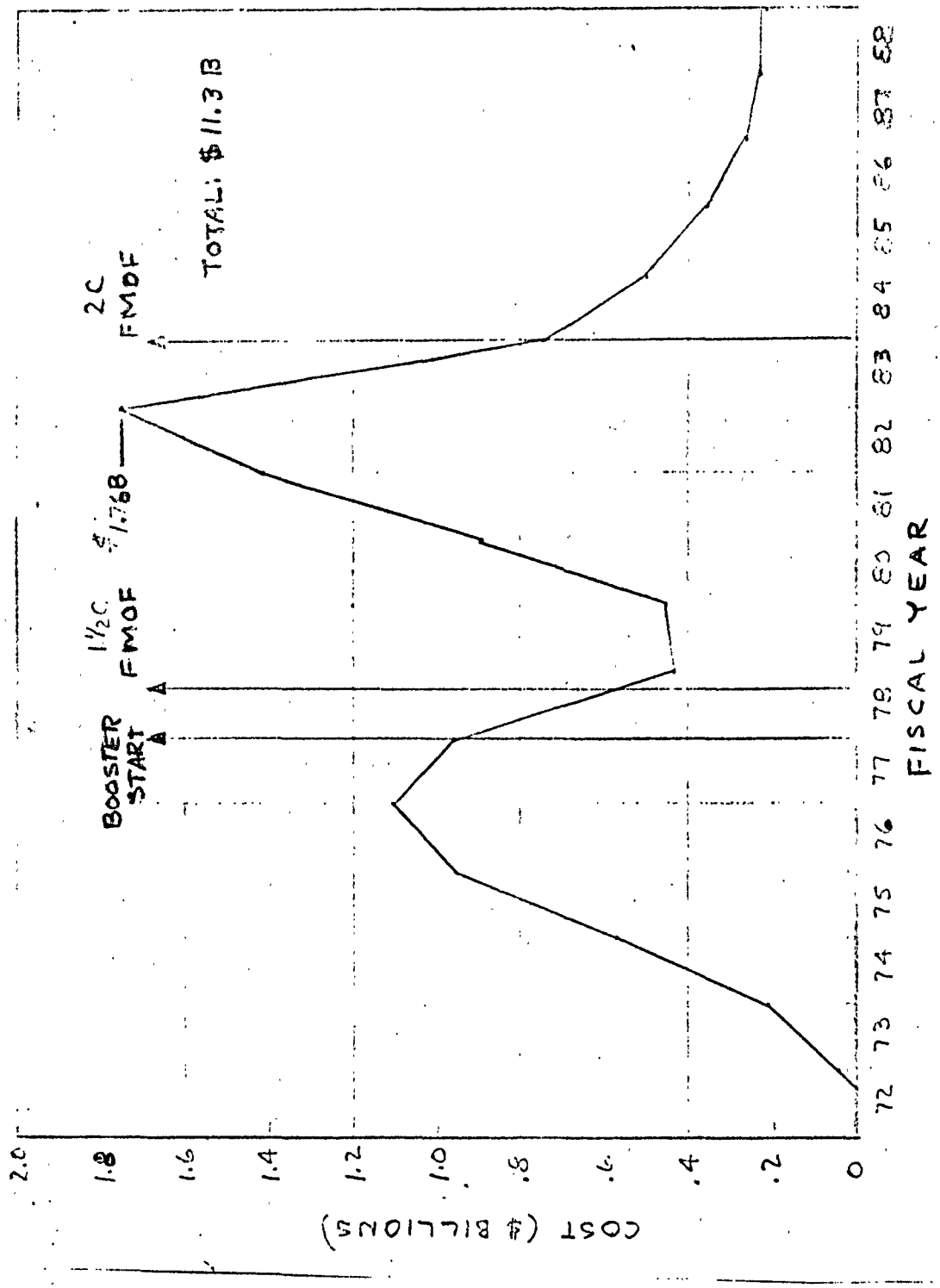

Fig. 4.11-5 Plan 5. Conversion System Annual Cost
5 Yr Interim Operations, 6 Yr Booster Development

Fig. 4.11-5

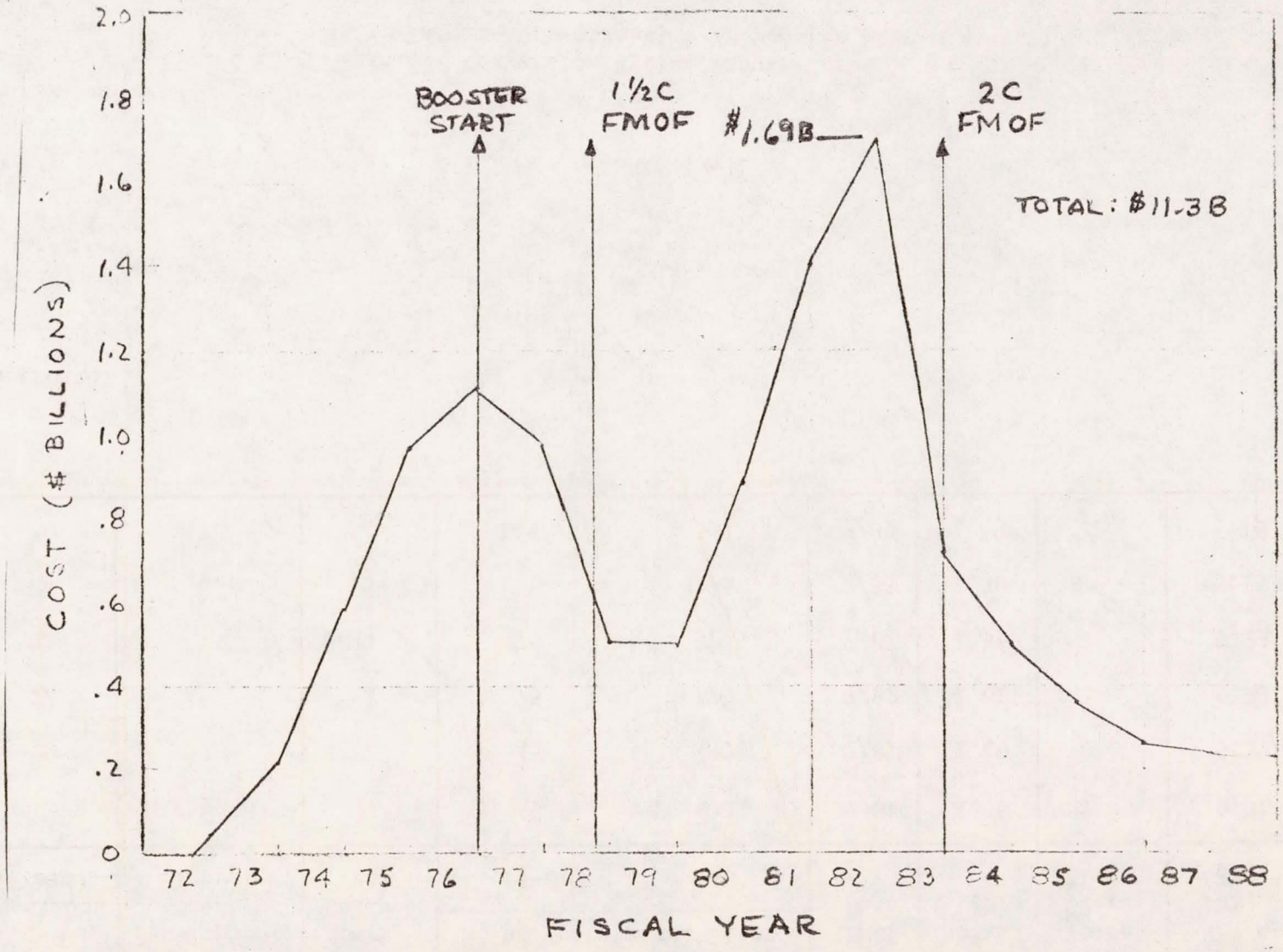

Fig. 4.11-6 Plan 6. Conversion System Annual Cost
5 Yr Interim Operations, 7 Yr Booster Development

Table 4.11-1

COMPARISON OF ALTERNATE CONVERSION PHASINGS

Plan	Interim 1½-C Operations (Years)	Booster Development Span (Years)	Booster Start Delay (Yrs)	No of Operational Flts.		2-C FMOF Date	Peak Funding ($B)	No. of Years Over $1.2 B	NPV at 10% ($B)	Total Program Cost ($B)
				1½-C	2-C					
1	3	6	3	45	400	7/81	1.56	3	5.34	10.9
2	3	7	2	45	400	7/81	1.49	4	5.37	10.9
3	4	6	4	75	370	7/82	1.61	2	5.20	11.0
*4	4	7	3	75	370	7/82	1.55	2	5.23	11.0
5	5	6	5	115	330	7/83	1.76	2	5.15	11.3
6	5	7	4	115	330	7/83	1.69	2	5.18	11.3

Table 4.11-1

* Best approach

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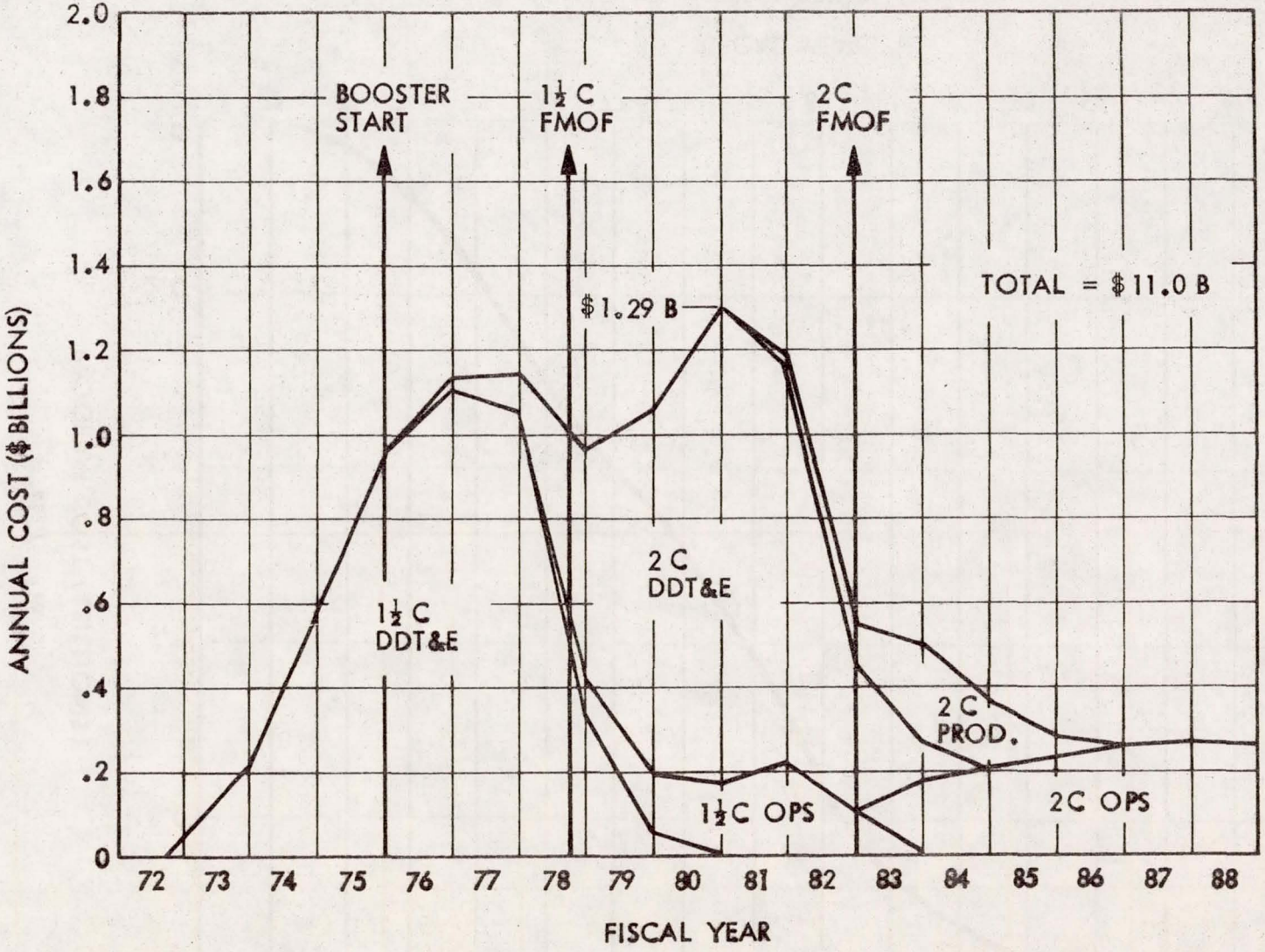

Fig. 4.11-7 Smoothed Plan 4. Conversion System Annual Cost 4 Yr Interim Operations, 7 Yr Booster Development

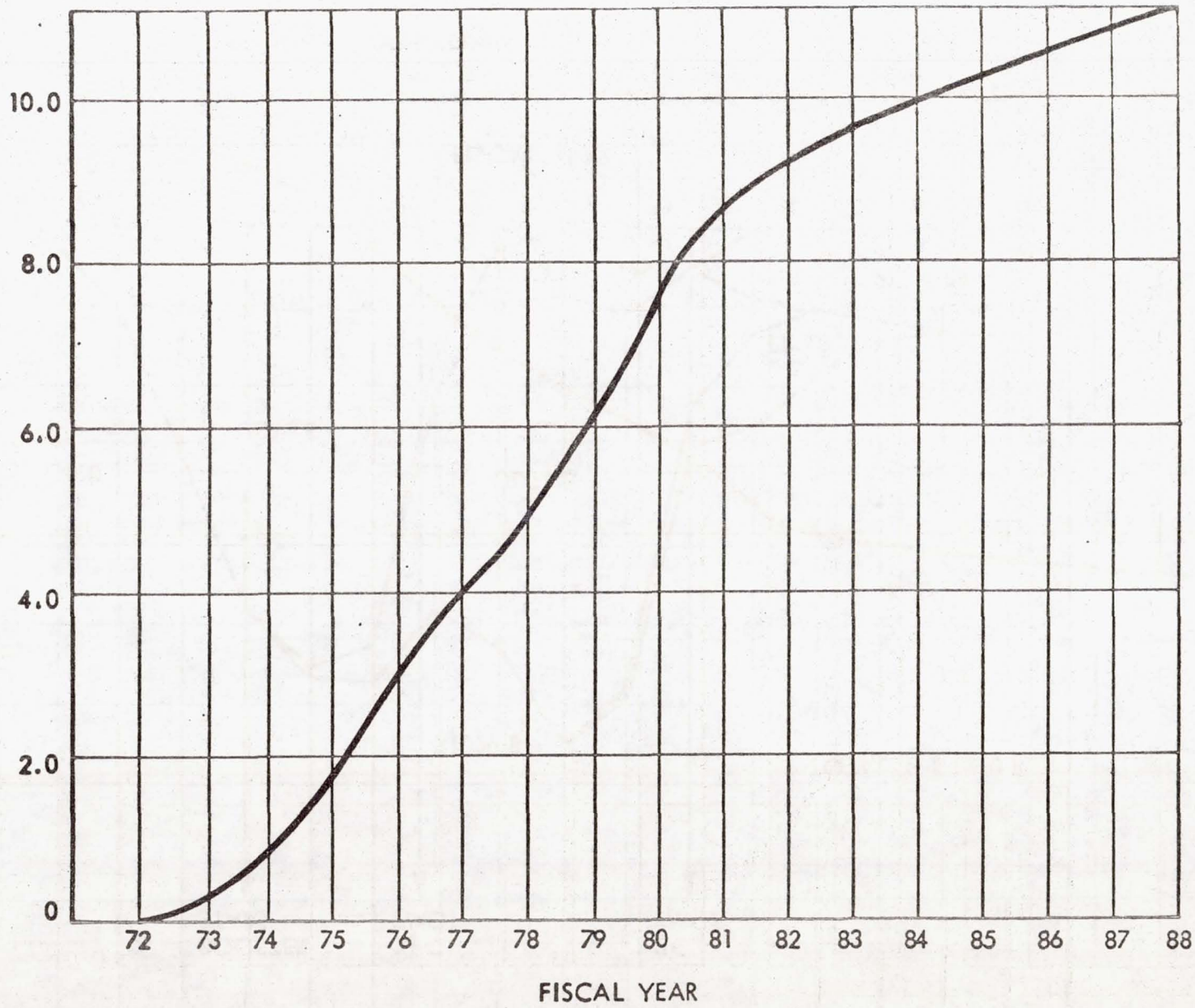

Fig. 4.11-8 Conversion System Cumulative Cost

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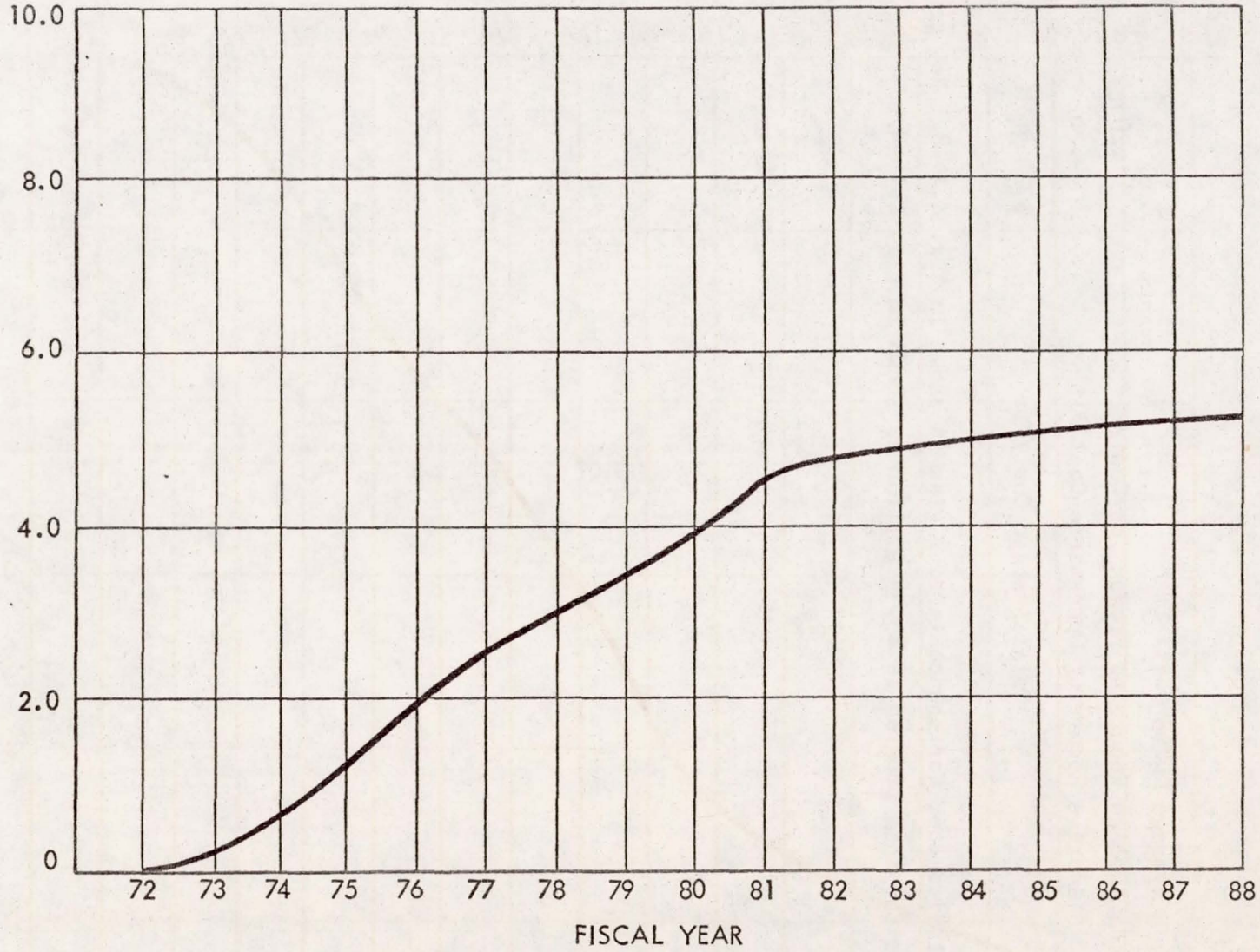

Fig. 4.11-9

Fig. 4.11- Conversion System NPV at 10%

Average DDT&E Cost/Flt	$ 18.84 M
Average Recurring Production Cost/Flt	1.26
Average Recurring Operations Cost/Flt	4.66
Total Average Cost/Flt	$ 24.76 M

Cumulative cost for the conversion plan is shown in Fig. 4.11-8. Net present value for this program is $5.34 billion as shown in Fig. 4.11-9.