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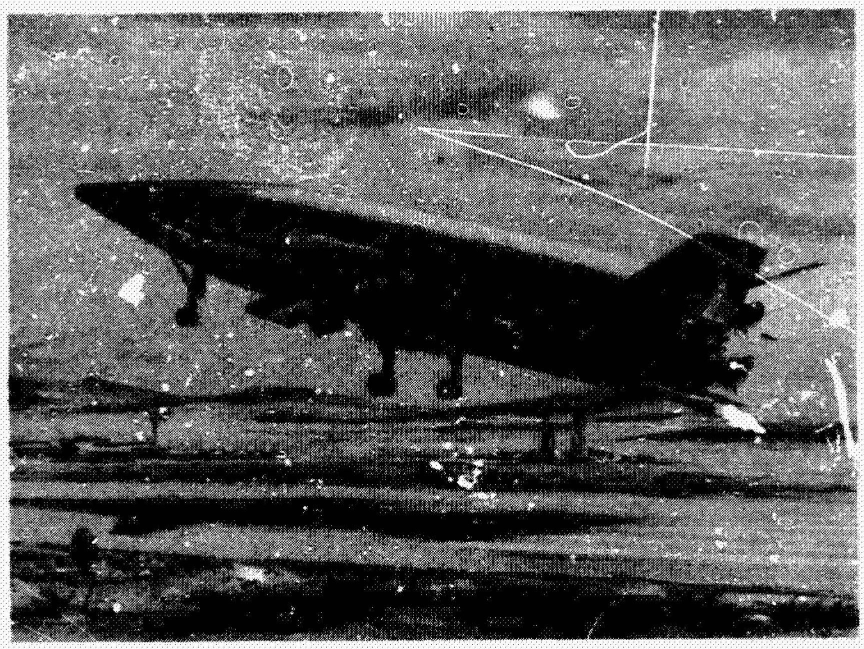

SPACE SHUTTLE FINAL TECHNICAL REPORT

VOLUME II • FINAL VEHICLE CONFIGURATIONS

GENERAL DYNAMICS
Convair Division

REPORT NO. GDC-DCB69-046

SPACE SHUTTLE
FINAL TECHNICAL REPORT

VOLUME II ✦ FINAL VEHICLE CONFIGURATIONS

31 October 1969

Prepared by
CONVAIR DIVISION OF GENERAL DYNAMICS
San Diego, California

FOREWORD

This volume of Convair Report No. GDC-DCB 69-046 constitutes a portion of the final report for the "Study of Integral Launch and Reentry Vehicles." The study was conducted by Convair, a division of General Dynamics Corporation, for National Aeronautics and Space Administration George C. Marshall Space Flight Center under Contract NAS 9-9207 Modification 2.

The final report is published in ten volumes:

Volume I	Condensed Summary
Volume II	Final Vehicle Configurations
Volume III	Initial Vehicle Spectrum and Parametric Excursions
Volume IV	Technical Analysis and Performance
Volume V	Subsystems and Weight Analysis
Volume VI	Propulsion Analysis and Tradeoffs
Volume VII	Integrated Electronics
Volume VIII	Mission/Payload and Safety/Abort Analyses
Volume IX	Ground Turnaround Operations and Facility Requirements
Volume X	Program Development, Cost Analysis, and Technology Requirements

Convair gratefully acknowledges the cooperation of the many agencies and companies that provided technical assistance during this study:

NASA-MSFC	Aerojet-General Corporation
NASA-MSC	Rocketdyne
NASA-ERC	Pratt and Whitney
NASA-LaRC	Pan American World Airways

The study was managed and supervised by Glenn Karel, Study Manager, C. P. Plummer, Principal Configuration Designer, and Carl E. Crone, Principal Program Analyst (all of Convair) under the direction of Charles M. Akridge and Alfred J. Finzel, NASA study co-managers.

ABSTRACT

A study was made to obtain a conceptual definition of reusable space shuttle systems having multimission capability. The systems as defined can deliver 50,000-pound payloads having a diameter of 15 feet and a length of 60 feet to a 55-degree inclined orbit at an altitude of 270 n.mi. The following types of missions can be accommodated by the space shuttle system: logistics; propellant delivery; propulsive stage delivery; satellite delivery, retrieval, and maintenance; short-duration missions, and rescue missions.

Two types of reusable space shuttle systems were defined: a two-element system consisting of a boost and an orbital element and a three-element system consisting of two boost elements and an orbital element. The vehicles lift off vertically using high pressure oxygen/hydrogen rocket engines, land horizontally on conventional runways, and are fully reusable. The boost elements, after staging, perform an aerodynamic entry and fly back to the launch site using conventional airbreathing engines. Radiative thermal protection systems were defined to provide for reusability. Development programs, technology programs, schedules, and costs have been defined for planning purposes.

During the study, special emphasis was given to the following areas: System Development Approaches, Ground Turnaround Operations, Mission Interfaces and Cargo Accommodations/Handling, Propulsion System Parameters, and Integrated Electronics Systems.

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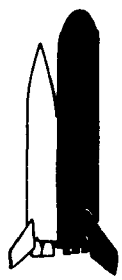
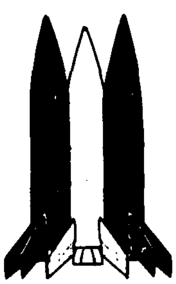
SUMMARY

This volume covers the work done on the final vehicles assigned to Convair by NASA for investigation in the final period of Phase A of the Integral Launch and Reentry Vehicle (ILRV) System Study.

Two vehicle systems were examined. The vehicles are fully reusable and no hardware is jettisoned. The systems are:



a. FR-3. This is a two-element system. The configuration is necessarily asymmetric, the booster element being considerably larger than the orbiter in order to achieve the minimum system weight. There is no basic commonality except for the rocket engines and flyback engines.



b. FR-4. This is a three-element system. The central orbiter element is located between the two booster elements. The booster elements are identical. They are also larger than the orbiter and have no commonality except for the rocket engines and flyback engines.

The FR-3 and FR-4 systems are both sequentially staged. That is, the orbiter element engines are ignited at the staging point. The vehicle systems are launched vertically and pitch into a ballistic trajectory to the staging point. After separation, the orbiter proceeds to an initial 43-n.mi. injection point and then finally to a 270-n.mi., 55 deg mission orbit, to rendezvous and dock with a space station. After a stay time not exceeding seven days the orbiter leaves the station, retrofires, and reenters. After entry the wings and turbofan landing engines are deployed and the orbiter arrives at the launch site making a standard airplane landing. After separation from the orbiter the booster element(s) proceed through a series of energy management maneuvers to depress their downrange distance, and then fly back to the launch site in a subsonic configuration, using turbofan engines for cruise.

The elements of both systems consist of simplified flat bottomed bodies with constant cross-sections built to accommodate state-of-the-art cylindrical tanks for the LO_2/LH_2 propellants. The vehicle noses are shaped to give the required aerodynamic characteristics. Vee tail stabilizers are added to give stability and control during hypersonic reentry, the transonic range and for subsonic flight and

landing. A low, stowable wing is incorporated in booster and orbiter elements which is deployed subsonically. The basic structure utilizes existing materials technology, being primarily aluminum alloy. Radiation thermal protection systems are employed for reusability and ease of turnaround. All the configurations use 400,000-lb sea level thrust, high chamber pressure bell nozzle rocket engines as stipulated by the NASA. The orbiter element attitude control propulsion subsystem utilizes LO_2/LH_2 propellants for compatibility with the main propellant systems and for easier turnaround. All elements incorporate standard aircraft type landing gear, and land tangentially on 10,000-ft long runways with landing speeds around 180 knots.

All orbiters are required to accommodate a 50,000-lb payload of 15-ft diameter by 60-ft long to orbit and return. An additional excursion was made to establish the effect of a 22-ft diameter payload.

The major characteristics of the baseline vehicles examined are summarized below

Vehicle System:	FR-3		FR-4	
Gross Liftoff Weight Overall Length (Launch)	4.33 Million Lb 235.5 Feet		4.92 Million Lb 219 Feet	
Element	Booster	Orbiter	Boosters (x 2)	Orbiter
Payload	—	50K	—	50K
Gross Weight at Liftoff	3.40M	0.93M	1.88M	1.16M
Landing Weight, lb	517K	287K	325K	322K
Gross Volume, ft^3	236K	89K	122K	107.5
Number of Boost Engines	15	3	9	3

The vehicle mass properties, performance, sensitivities, aerodynamics, aerothermodynamics, loads, thermostructural design, boost control, and costs are summarized in this volume. Greater depth on these is provided in the other volumes of this final report.

An examination of the total program costs shows no significant difference between the FR-3 and FR-4 at traffic rates of 20 to 50 launches per year. Above this rate, the FR-3 becomes increasingly more economical than the FR-4.

SECTION 1

INTRODUCTION

This volume gives a detailed description of the final vehicles analyzed by Convair during the latter part of the Integral Launch and Reentry Vehicle Study. These vehicles were assigned by NASA after review of the configuration spectrum described in Volume III of this final report. Section 2 contains the vehicle requirements used during the study. Section 3 describes the approach used in establishing the basic shape, size, and design characteristics of the vehicles. Section 4 documents, in summary form, the final Phase A vehicle configurations. The remaining Sections (5 through 12) expand the description of major subsystems and other important items such as safety, mission analysis, and ground operations. While the reasoning behind the current vehicle configuration design features is given in this volume, the detailed analyses are presented in other volumes of this report.

The final vehicles are designated as follows:

<u>FR-3</u>. This is a two-element system with the booster considerably larger than the orbiter in order to arrive at minimum system weight. The configuration is necessarily asymmetric and is shown in Figure 1-1. There is no propellant crossfeed. This is a sequentially staged system with the orbiter element rocket engines ignited at staging.

<u>FR-4</u>. This is a three-element system with the orbiter element centrally located between the booster elements as indicated in Figure 1-2. There is no commonality between booster and orbiter, except for the rocket engines and flyback engines. There is no propellant crossfeed between elements. This is also a sequentially staged system.

A typical flight profile is shown in Figure 1-3. After liftoff and vertical rise, the combined booster and orbiter element commence a programmed pitch schedule followed by a gravity turn to the staging point. After staging, the orbiter element proceeds to the initial injection point and the booster utilizes a series of energy management maneuvers at maximum lift coefficient in order to depress the apogee and reduce the downrange distance and flyback range. The booster enters at maximum lift, achieves subsonic velocity in the region of 50,000 ft altitude, at which point the vehicle's wings are extended for subsonic flight, a 180-degree turn is followed by cruise back to the launch site.

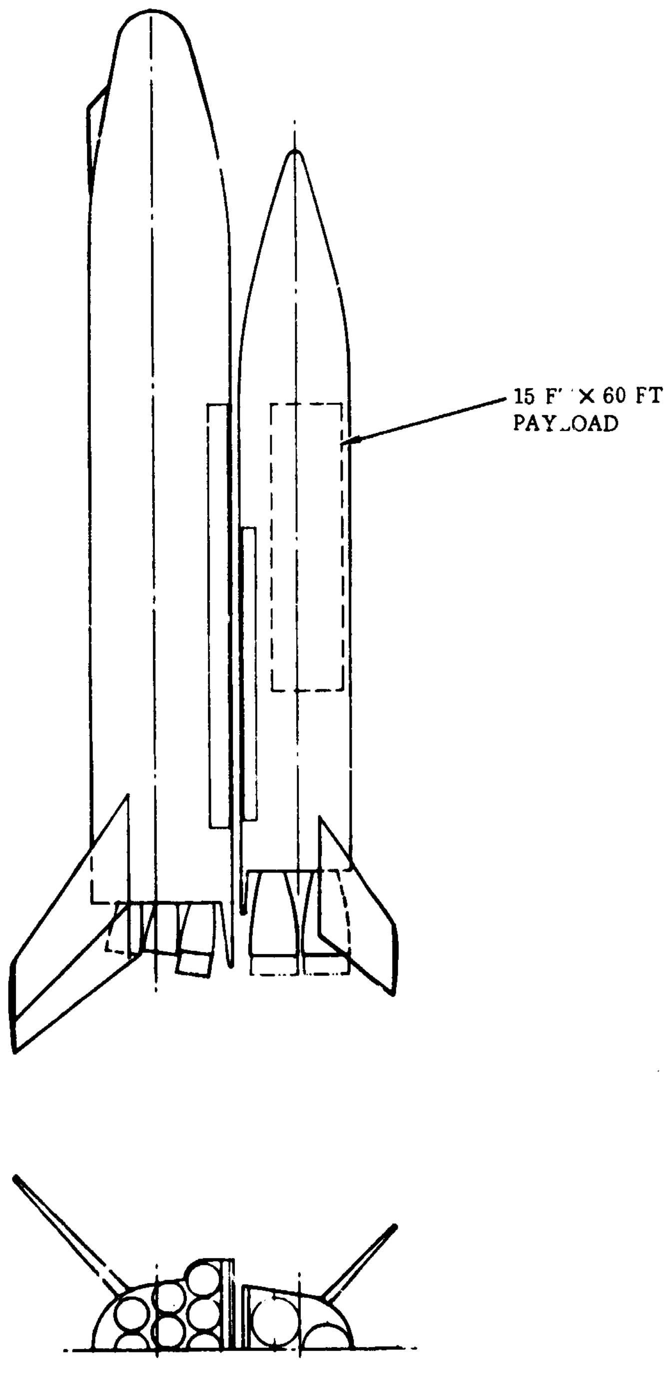


Figure 1-1. Typical FR-3 Launch Configuration

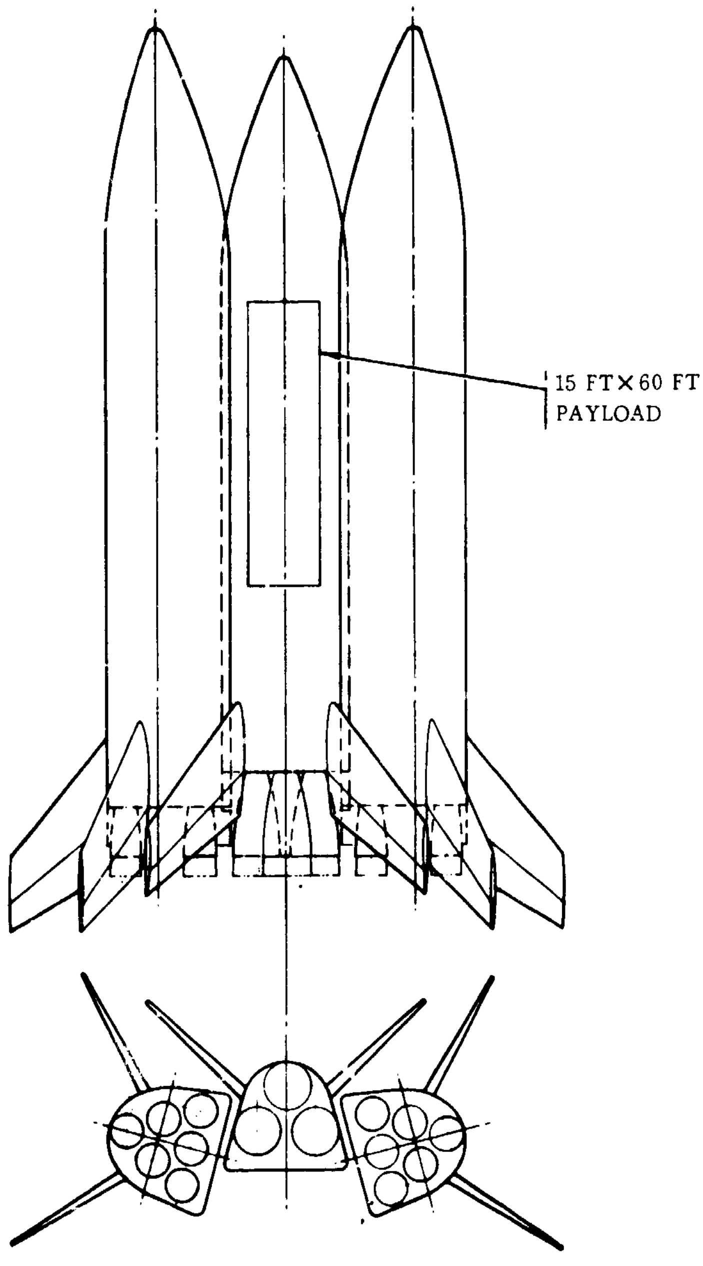


Figure 1-2. Typical FR-4 Launch Configuration

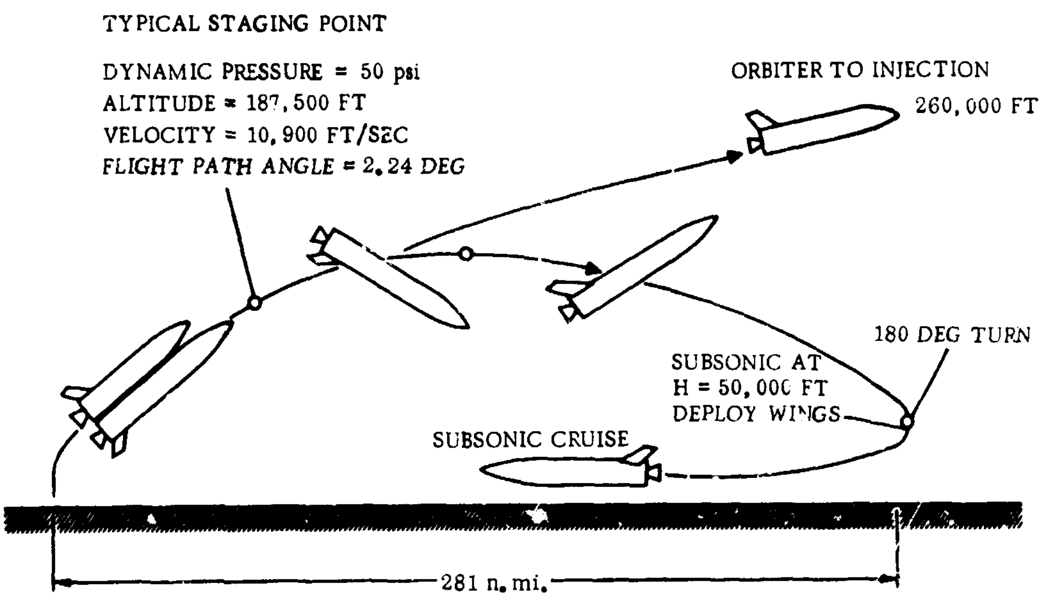


Figure 1-3. Typical Flight Profile -- FR-3

SECTION 2

FR-3 AND FR-4 MISSION AND VEHICLE REQUIREMENTS AND DESIGN GROUND RULES

The ground rules applied to the final vehicle configurations consist of specific NASA requirements stated at the beginning of the study with later modifications, plus certain additional specifications made by Convair where the need arose.

2.1 DESIGN PHILOSOPHY

The design philosophy in the ILRV study is aimed at fully reusable space shuttle systems with minimum recurring costs.

2.2 REUSE FACTOR

The minimum number of missions of the major airframe components of any vehicle element in the system is 100.

2.3 NOMINAL TRAFFIC RATE

The nominal traffic rate is 100 flights per year for 10 years.

2.4 PAYLOAD

The basic up and down payload weights are 50,000 lb. The payload envelope is an uninterrupted space 15 ft in diameter by 60 ft long. A single unit payload of these dimensions must be removable from the orbiter vehicle.

The effect of incorporating a payload bay similar in size to the above with a 22 ft diameter by 30 ft long section superimposed upon it was to be examined, but due to the late introduction of this payload bay size all the FR-3 and FR-4 basic vehicle work considers only the 15 ft diameter payload bay. The 22 ft diameter payload bay is treated as an excursion from the nominal. In the case of the FR-3, new orbiter lines were required to accommodate the larger diameter.

The payload weights include all cargo, passengers, associated environmental control and life support subsystem and transfer equipment. The passengers are to be physically located in the payload bay region of the orbiter element and intravehicular access is required between the payload bay and the forward-located crew compartment. (Convair studies shown later indicate the desirability of locating up to four passengers forward, adjacent to the crew compartment for certain missions. (See Volume VIII.)

2.5 CREW

The crew in each element shall consist of two pilots. A shirtsleeve crew environment is required at a cabin pressure of 10 psia. A two-gas subsystem is to be used.

2.6 MISSION ORBIT AND ΔV REQUIREMENTS

The basic mission orbit is a 55-deg inclination at an altitude of 270 n.mi. with a parking orbit of 100 n.mi. as indicated in Figure 2-1.

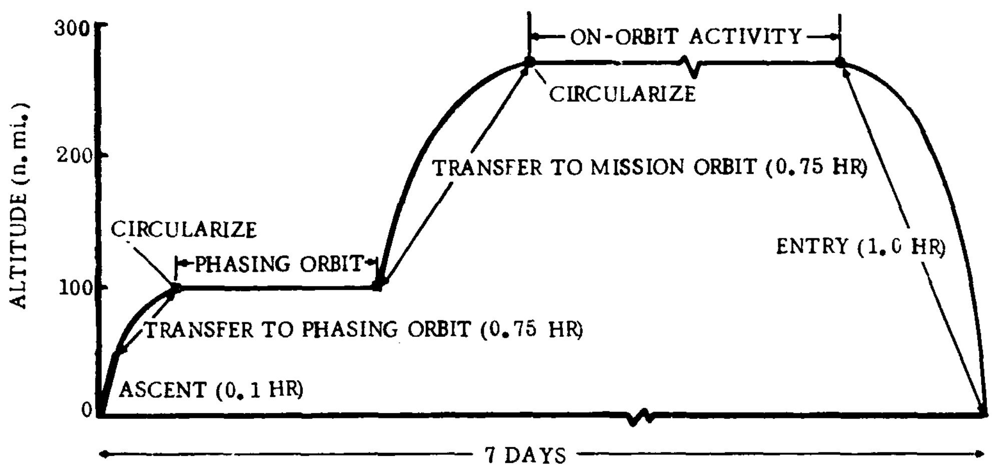


Figure 2-1. Mission Profile

A flight performance reserve of 3/4 of one percent of the total ideal ΔV, including back pressure losses, is to be applied to the main propulsion subsystem requirements.

The on-orbit main propulsion ΔV shall total 1800 fps, apportioned as follows:

Maneuver		ΔV (fps)
Circularize at 100 n.mi.		110
Transfer 100 n.mi. — 260 n.mi.		280
Circularize at 260 n.mi.		280
Retro from 260 n.mi.		450
Launch insertion dispersion		200
Contingency and reserve		480
	Total	1800

For attitude control on-orbit, 200 fps shall be provided, for a total ΔV on-orbit of 2000 fps. The 200 fps includes transfer to the 270-n. mi. orbit.

2.7 MISSION DURATION

The total mission duration is nominally set at seven days. For missions over the duration, the weights of additional expendables or equipment required for the additional time shall be subtracted from payload weight.

2.8 LAUNCH AND LANDING SITE

The launch site is ETR. The booster elements will return to a horizontal landing on a 10,000-ft runway at the launch site. The booster elements will have capability to return to the launch site with one flyback airbreathing engine out. The orbiter will normally return to the launch site for a horizontal landing. A nominal landing speed of 180 knots is desired.

2.9 ORBITER ABORT PHILOSOPHY

The Convair abort philosophy established for this study is as follows.

In event of abort, the objective will be to keep going. Staging will be effected as soon as the booster propellants are depleted, after which the booster will return to base in the normal fashion. The orbiter will proceed along the trajectory to low orbit, regardless of whether the abort decision is made before or after staging. Because of the added ΔV losses, the orbiter will be required to burn maneuver propellants in order to achieve a "once-around" orbit (once around the earth and return to base). In the FR-3 and FR-4 vehicles this requires a minimum thrust-to-weight ratio at orbiter ignition of 0.80 with one engine out.

With the once-around abort philosophy a crossrange of 800 n.mi. is required to return to the ETR launch site from a 55-deg inclination orbit.

2.10 LOAD FACTORS

An axial load limit of a 3g is applied to performance during ascent to limit forces on passengers. However, structural axial load factors shall be 4g for potential use in non-passenger flights. Also, a 50-fps sharp edge gust, as specified in MIL-SPEC-8861, subsonic gust condition. and a nominal landing load factor of 2g are used.

2.11 WINDS

Maximum launch winds adopted in Phase A were 99 percent probability Marshall synthetic winds.

The ground wind conditions adopted were for a surface wind speed of 99 percent at the ETR launch site.

2.12 FACTORS OF SAFETY

The following safety factors were adopted:

Aerodynamic and Associated Inertia Loads	=	1.40
Thrust and Associated Inertia Loads	=	1.25
Personnel Compartment Pressures	=	2.00
Reusable Propellant Tank Pressures	=	1.50

These factors will be refined in continued studies to reflect more sophisticated conditions such as crack propagation criteria.

2.13 MATERIAL TEMPERATURE CONSTRAINTS

The material temperature constraints adopted for the study were:

Titanium	To 800°F
Inconel 718	800°F to 1200°F
Rene 41	1200°F to 1400°F
L605	1400°F to 1900°F
TD Nickel Chrome	1800°F to 2200°F
Columbium	2200°F to 2500°F
Tantalum	2500°F to 3100°F

2.14 WEIGHT CONTINGENCY

A 10 percent contingency is applied to the total system dry weights after their nominal weight is calculated. The added contingency weight is reflected in all basic performance estimates.

2.15 SAFETY CRITERIA

Fail safe capabilities were incorporated in all subsystems (see Section 7).

2.16 PROPULSION REQUIREMENTS

2.16.1 MAIN PROPULSION. The baseline engine is required to be the high performance bell nozzle LO_2/LH_2 engine, with individual unit thrust levels of 400,000 lb at sea level. Booster engines with expansion ratios of 35/80 and orbiter engines with an expansion ratio of 160 were used on the final vehicles. Thrust uprating was not used. A nominal mixture ratio of 6.5 was used in both booster and orbiter elements.

2.16.2 FLYBACK PROPULSION. Existing turbofan engines were used for flyback and landing in all elements. The orbiter element is designed for one go-around at sea level with all turbofan engines running.

The booster element is designed for cruise back to the launch site with one turbofan engine inoperative.

2.16.3 ATTITUDE CONTROL PROPULSION SUBSYSTEM. The attitude control subsystem is required to use LO_2 /LH_2 propellants. During space maneuvers, angular accelerations of at least 2.5 deg/sec^2 about the three axes and linear accelerations between 0.03 g and 0.05 g in six directions are provided without crosscoupling; during entry, angular accelerations of 2.5 deg/sec^2 are provided without using downward or forward facing thrusters. The thrusters are located so that a thruster at any location can fail and accelerations about all axes and in all directions will still be provided without crosscoupling but at reduced accelerations. Accelerations about all axes and in all directions are provided, with crosscoupling, after the failure of two thrusters in any location.

2.17 AEROTHERMODYNAMICS

An entry transition Reynolds number of 10^6, based upon boundary layer edge properties, was used to define the onset of boundary layer transition. Transition was complete at a Reynolds number of 2×10^6.

2.18 ADDITIONAL DESIGN REQUIREMENTS

Additional design requirements are:

a. Electronic cockpit displays.

b. Onboard guidance and navigation.

c. Automatic approach and docking.

d. Hard docking with space station.

e. Cargo and personnel transfer.

f. Direct landing visibility.

g. Automatic landing capability.

h. Ferry capability.

i. Ground turnaround time: 80-hr work period.

2.19 TIME PERIOD

The logistics space vehicle system is to support earth orbital programs in the post-1974 time period.

SECTION 3

FINAL CONFIGURATION BASIS

3.1 ELEMENT SHAPES

The shape of the orbiter elements for the 15-ft-diameter by 60-ft-long payload bay FR-3 and FR-4 systems was adopted directly from the Convair FR-1 system (identified hereafter as the T-18 shape). This was a system of three identically-sized elements with all engines burning at liftoff and propellant crossfeed between elements. This shape had been developed during the first part of the ILRV study and was modified during parallel space transportation systems studies. In this respect, the orbiter vehicle shape has hypersonic L/D potential giving well over 1500 n.mi. crossrange. This can be achieved by increasing the thermal protection capability and weight over that used for the nominal 800 n.mi. crossrange. If 800 n.mi. crossrange were to be finally specified as the maximum requirement, the shape could be modified to a lesser fineness ratio than that shown in Figure 3-1.

Figure 3-1 shows the lines of the T-18 shape in non-dimensional form. All dimensional data have been converted to percentages of the element reference length, except such constant items as body nose radius and stabilizer leading edge radius.

The body, shown in Figure 3-1, consists of a constant section with a tapered forebody terminating in a hemispherical nose. The forebody shape is parabolic in planform with a partially straight lined lower surface ramp in the profile view. The constant section consists of a flat bottom with sides tapering inward at a 12-degree angle and a full upper radius. The upper radius allows maximum usage of state-of-the-art cylindrical or truncated conical cryogenic propellant tankage. The flat bottom improves the hypersonic lift/drag ratio and provides convenient stowage space for the switchblade type subsonic wings. The sides slope inward to improve the hypersonic L/D and to reduce entry heating on side surfaces. The vee tail is attached high up on the afterbody for subsonic stability reasons, and to provide hypersonic and transonic stability. Hypersonic roll control is obtained via differential deflection of the ruddervators or the vee tail.

The basic T-18 shape shown in Figure 3-1 was used for all FR-4 system orbiters and boosters, for both the basic 15-ft and 22-ft-diameter payload bays. The FR-3 system used the T-18 shape for the 15-ft-diameter payload bay orbiter. Considerable work had been invested in this shape during the earlier part of the study and the configuration was retained in order to derive the benefit of this work. Any future reconfiguration of the FR-4 would reduce the orbiter fineness ratio to reflect the required crossrange (when this is finally stipulated) and would cause redesign of the booster elements to suit the booster mission. The boosters need to fly at maximum angle of attack hypersonically to derive the highest drag and shortest downrange distance possible, which will reduce the flyback range to the launch site and thus the flyback fuel weight. Therefore,

LINES DATA

BASE LINE LENGTH (POINTED NOSE TO AFT END OF ELEVON LOWER SURFACE) = 100 %

NOSE RADIUS - CONSTANT	1 FT.
BODY L.E. RADIUS - CONSTANT	.5 FT.
SIDE SLOPE - CONSTANT	12°
NOSE TOP ℄ SLOPE - CONSTANT	16°
NOSE BOTTOM ℄ SLOPE - CONSTANT	15°

ALL BODY TOP RADII CENTER ON BODY ℄

BODY:

WIDTH			17 %
HEIGHT			14.56 %
CONSTANT CROSS-SECTION STARTS			36.1 %
CONSTANT SECTION TOP RADIUS			6.65 %
TOP ℄ COORDINATES:			
STRAIGHT 16° SLOPE TO			11.5 %
STA	14 %	ORD	+ 3.98 %
	18 %		+ 4.91 %
	22 %		+ 5.65 %
	26 %		+ 6.17 %
	30 %		+ 6.48 %
BOTTOM ℄ COORDINATES:			
STRAIGHT 15° SLOPE TO			18.87 %
STA	22 %		- 5.06 %
	26 %		- 6.75 %
	30 %		- 7.44 %
END OF CONSTANT SECTION AT			93 %

V-TAIL CHARACTERISTICS

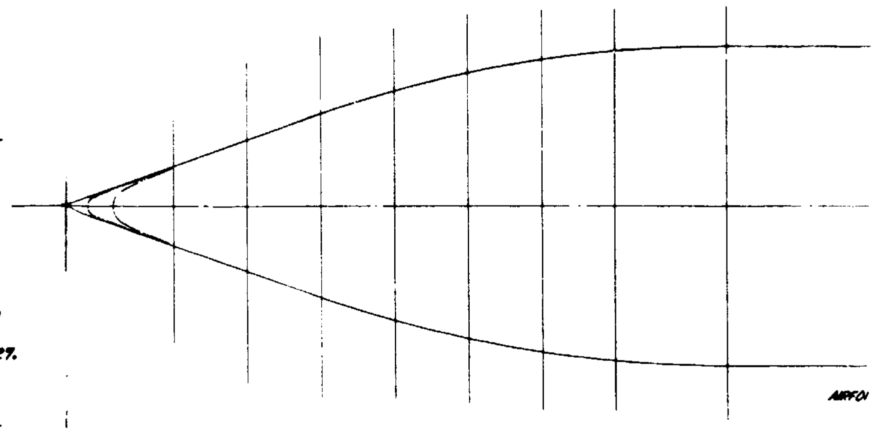
SWEEP L.E. TRUE	46°
ROLL OUT	45°
L.E. RADIUS - CONSTANT	.5 FT.
ROOT CHORD	15.45 %
TIP CHORD	9.28 %
SEMI SPAN - TRUE	17.04 %
LOCATE 60% ROOT CHORD AT	93 %
ASPECT RATIO	1.375
TAPER RATIO	.60
AIRFOIL ROOT	4412 MOD.
TIP	4410 MOD.
TIP CROSS-SECTION	HALF-ROUND
BOTTOM SURFACE & FUSELAGE ℄ VERTICAL PLANE INTERSECT AT	-3.62 %
INCIDENCE (FLAT BOTTOM)	0°

WING CHARACTERISTICS

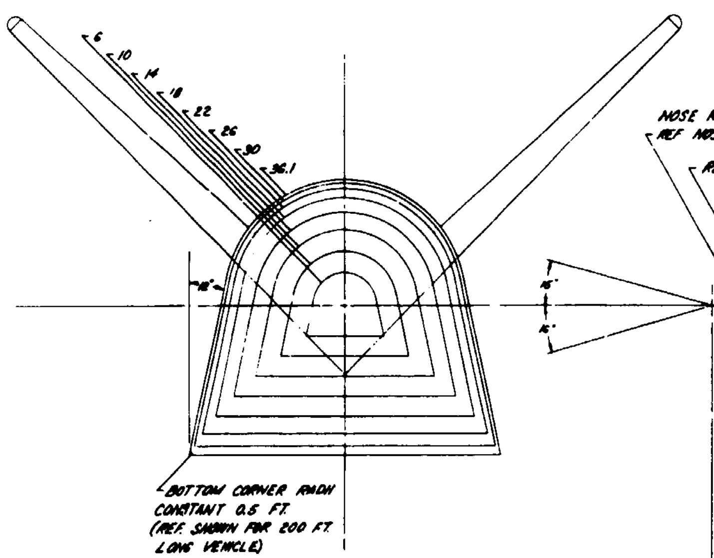
SWEEP L.E.	10°
ASPECT RATIO	9.25
TAPER RATIO	.80
ROOT CHORD	7.42 %
TIP CHORD	5.92 %
SEMI SPAN	30.8 %
M.A.C. LOCATION (outboard of root)	14.88 %
QUARTER CHORD M.A.C. AT	55.5 %
M.A.C.	6.72 %
AIRFOIL ROOT	4421
TIP	4418
DIHEDRAL (WING EXTENDED)	0°
INCIDENCE (CHORD PLANE) WING FULLY EXTENDED	+ 6°
WING ROOT T.E. LOCATED ABOVE FUSELAGE BOTTOM A CONSTANT	.75 FT.

EXPOSED AREA OF V-TAIL - DEFINED AS TRUE AREA OF FLAT BOTTOM SURFACE FROM INTERSECTION WITH CONSTANT FUSELAGE CROSS-SECTION TO TIP, EXCLUDING TIP FAIRING

EXPOSED AREA OF WING - DEFINED AS AREA OUTBOARD OF FUSELAGE MAXIMUM THEORETICAL WIDTH WITH WING FULLY EXTENDED (L.E. SWEPT 10°)



PLANFORM AREA (SQ. FT.) = 14.75 % REF. LENGTH (FT.) × 100 % REF. LENGTH (FT.)
WING AREA (EXPOSED, SQ. FT.) = 28 % PLAN AREA
V-TAIL AREA (EXPOSED, TRUE, SQ. FT.) 28.75 % PLAN AREA
BODY WETTED AREA = 3.04 TIMES PLAN AREA
BODY TOTAL VOLUME = PLAN AREA TIMES 10.1 % REF. LENGTH IN FEET.



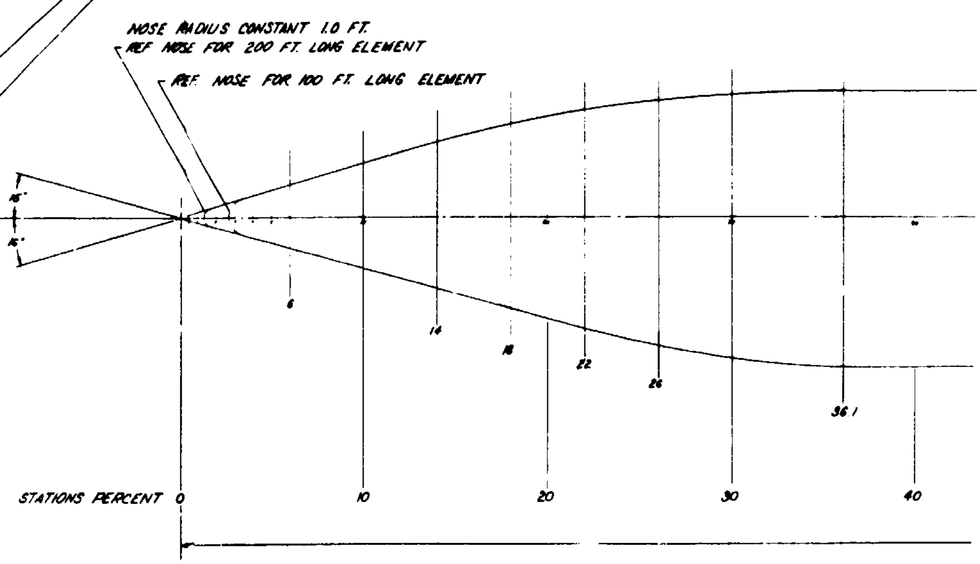


Figure 3-1. T-18 Configuration Lines

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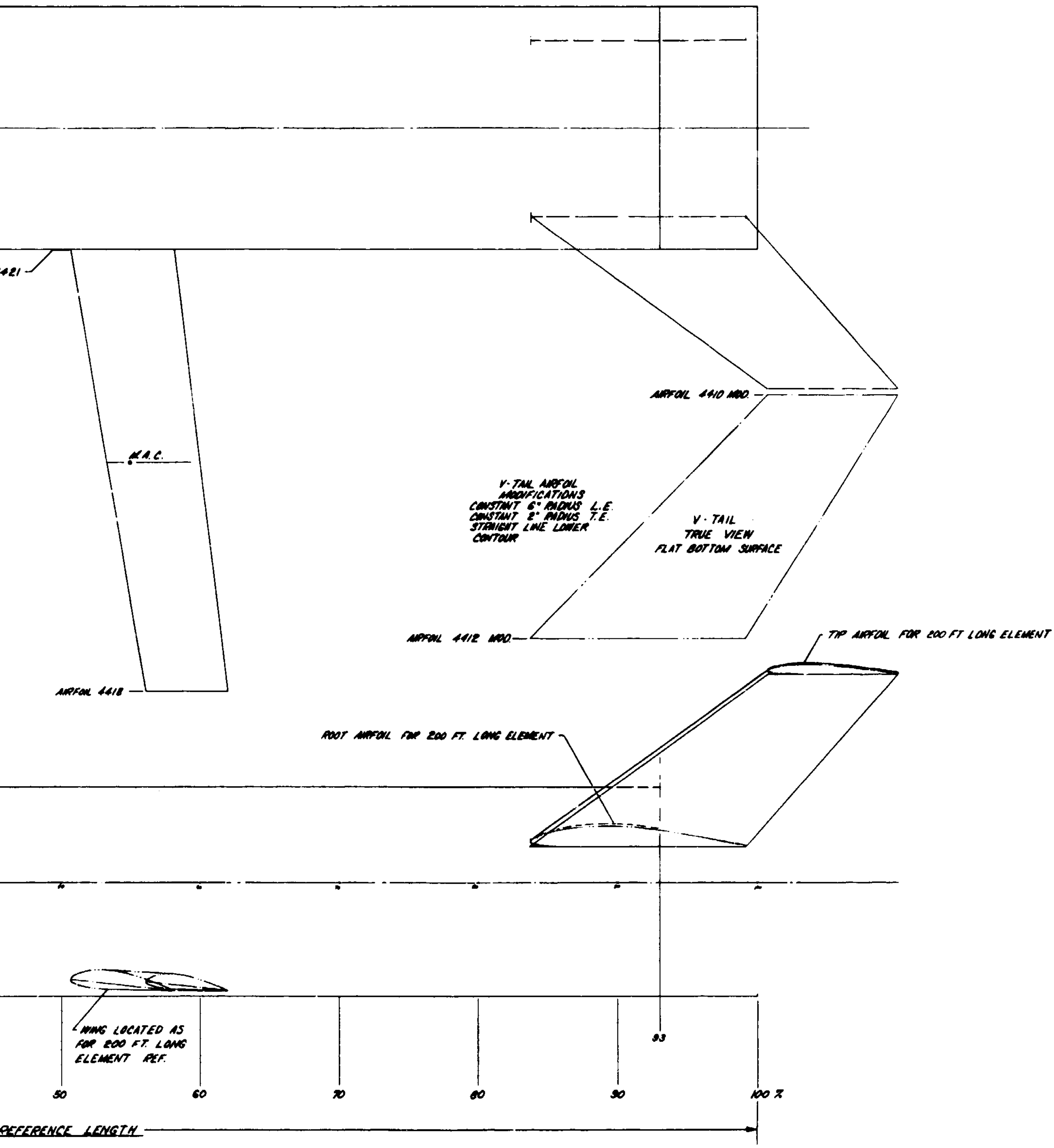


FOLDOUT FRAME 2

the boosters can have a low hypersonic L/D. This can conveniently be introduced by blunting the nose, which does not degrade the subsonic L/D for flyback. This improves the vehicle volume to wetted area ratio and because of the large radius nose, reduces nose temperatures.

3.1.1 FR-3 BOOSTER SHAPE. Typical lines for the FR-3 booster are shown non-dimensionally in Figure 3-2. All parameters are related to the vehicle reference length. This booster will be considerably larger than the FR-4 boosters since all of the boost function is now contained within a single vehicle in FR-3 as against the twin booster vehicles of FR-4.

The overall FR-3 booster shape derives generally from the previous two-stage vehicle boosters (see Volume III of this report) which in turn were similar in concept to the FR-1 shapes. The current FR-3 shape shown in Figure 3-2 has a well blunted nose and steeper side slopes (approximately 9 deg) than the previously shown lines. The cross-section is still flat bottomed with a full upper radius for maximum cylindrical tank accommodation. The blunting of the nose allows a cylindrical tankage of constant cross-section throughout the entire vehicle, as opposed to the tapered tank extensions on earlier configurations. The constant section is faired out into a large spherical radius nose, the forebody being lofted via tangential curves through single control points. The side slope is held constant and the body constant section leading edge radius is allowed to grow until it matches the nose cap radius as seen in the front view of Figure 3-2. Sufficient space is left within the nose to accommodate the turbofan flyback engines, flyback fuel, and subsystems. The two pilot crew compartment is elevated above the nose section to give a more compact installation with better visibility potential. A vee tail similar in design to the previous configurations is used. No elevons are incorporated, roll control is achieved with the ruddervators on the vee tail with yaw coupling taken out by thrusters in the booster nose during the few minutes of hypersonic entry flight.

In all the final vehicles of this study phase the capability to stow the wings has been retained. Fixed-wing vehicle excursions were made briefly (see Volume III) and the preliminary conclusion was that there was little overall weight difference between fixed and stowed wings, provided the cross-range was limited to a few hundred miles. Because of the potential for increasing cross-range, and because of the unknown transonic stability problem of the fixed-wing configuration, it was decided to retain the stowable wing configuration in all Phase A system final vehicle orbiter elements. The case for a fixed-wing booster element appears to be stronger, since hypersonic cross-range is not a factor. The transonic regime is still a factor, however. Delta wings, or double delta wings, would reduce the aerodynamic center shift, but they are heavy and inefficient for the purpose of subsonic cruise back to the launch site since their aspect ratio is low. In configurations with high density propellant combinations a delta wing can supplement the planform area to reduce entry heating. However, in a low density propellant vehicle, the planform is already generally sufficient for the body alone to reduce entry temperatures to a tolerable value. In this respect straight or slightly

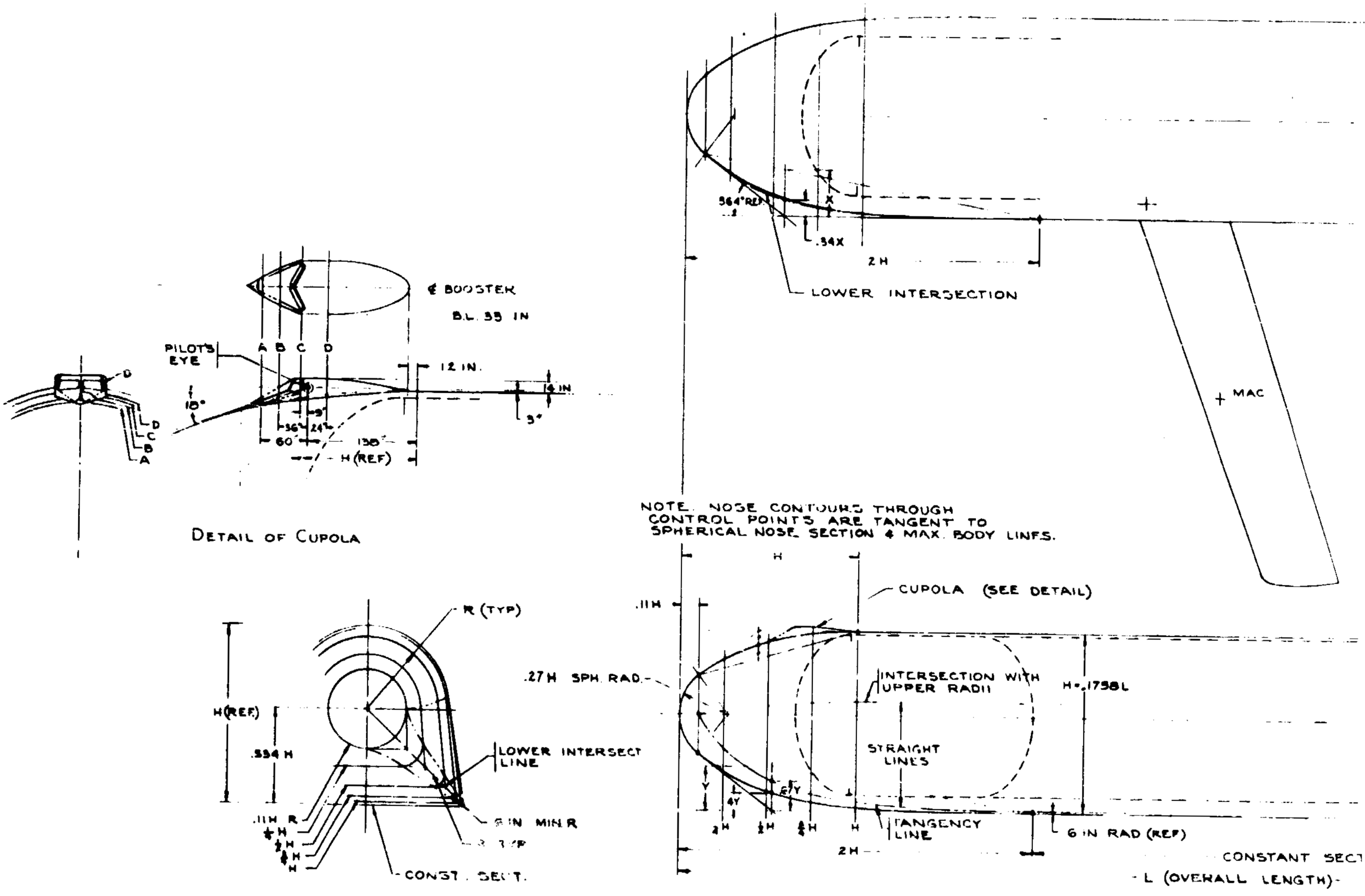
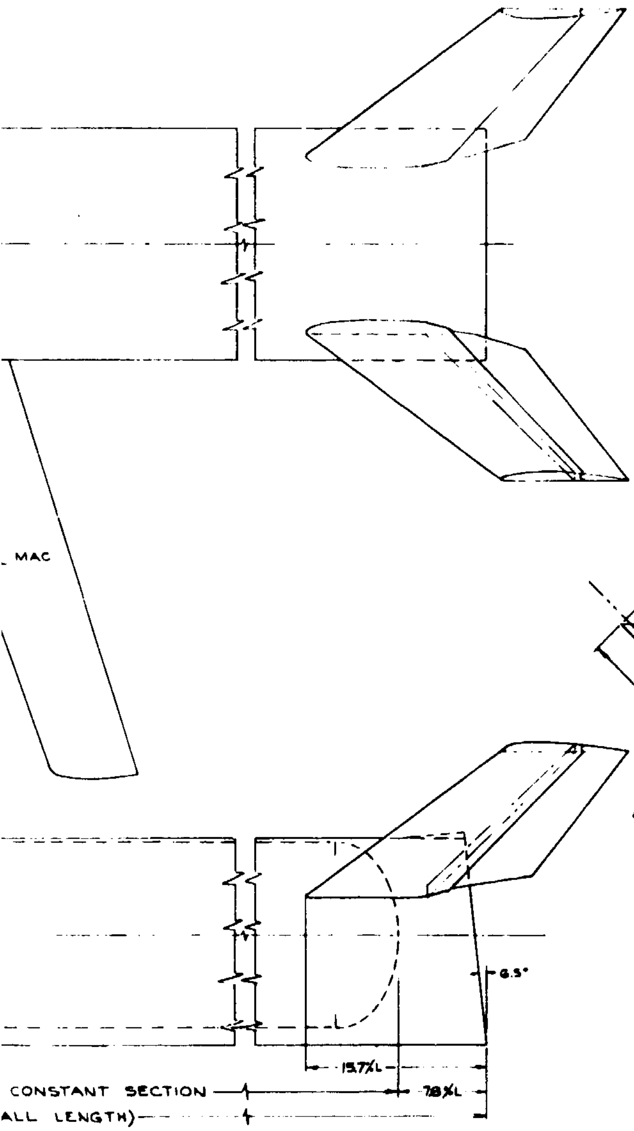


Figure 3-2. FR-3 Booster Lines



LINES DATA

VOLUMES AND AREAS

PLANFORM AREA	(SQ. FT)	= 0.1851 L^2
WING AREA EXPOSED	(SQ FT)	= 0.0623 L^2
V' TAIL AREA	(" ")	= 0.0519 L^2
BODY WETTED AREA	(" ")	= 0.6029 L^2
BODY TOTAL VOLUME	(CU. FT)	= 0.02545 L^3

"V"-TAIL CHARACTERISTICS

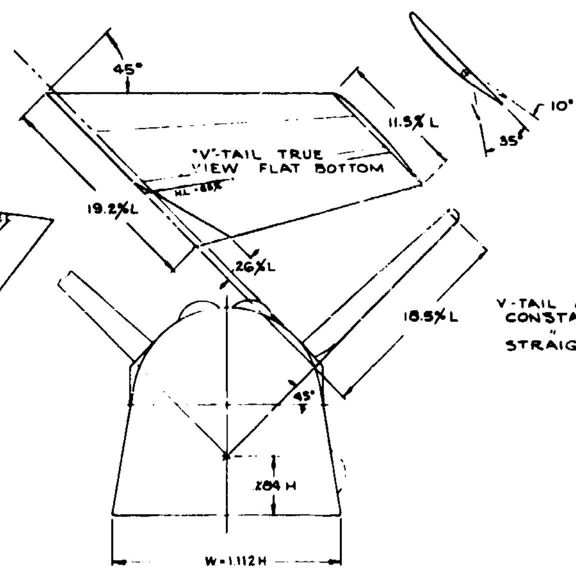
SWEEP L.E. TRUE	= 45°
ROLL OUT	= 45°
L.E. RADIUS CONSTANT	= 5 FT.
ROOT CHORD	= 19.2%L
TIP CHORD	= 11.5%L
SEMI-SPAN TRUE	= 18.5%L
ASPECT RATIO	= 1.26
TAPER RATIO	= .60
AIRFOIL ROOT	4412 MOD.
" TIP	4410 MOD.
TIP CROSS SECTION	HALF-ROUND
INCIDENCE (FLAT BOTTOM)	= 0°

NOTE: EXPOSED AREA OF V-TAIL IS DEFINED AS TRUE AREA OF FLAT BOTTOM SURFACE FROM INTERSECTION WITH CONSTANT SECTION OF FUSELAGE EXCLUDING TIP FAIRING.

WING CHARACTERISTICS

SWEEP L.E.	= 20°
DIHEDRAL (WING EXTEND)	= 0°
INCIDENCE (CHORD PLANE)	= +6°
ROOT CHORD	= 8.9%L
TIP CHORD	= 7.5%L
SEMI-SPAN	= 38%L
ASPECT RATIO	= 9.8
AIRFOIL ROOT	4421
" TIP	4418
WING PIVOT	= 46.8%L
1/4 M.A.C. LOC.	= 55%L

NOTE: EXPOSED WING AREA IS DEFINED AS AREA OUTBOARD OF FUSELAGE MAXIMUM THEORETICAL WIDTH WITH WING FULLY EXTENDED (20° SWEEP).



V-TAIL AIRFOIL MODIFICATIONS:
CONSTANT 6"R L.E.
" 2"R T.E.
STRAIGHT LINE LOWER CONT'R.

swept wings would be preferable, but the transonic stability picture is still not clear. Future booster investigations will cover the implications of fixed wings, but because existing wind tunnel data show that the current configurations with the wing stowed are stable throughout the transonic regime, the decision was made to retain the stowable wing on the Phase A final booster vehicles as well as for the orbiters. Preliminary excursions have been made into high wing boosters. They are still being investigated, but because of the more convenient stowage arrangement and the existing subsonic test data, low wing arrangements have been retained for the final FR-3 and FR-4 configurations.

3.1.2 FR-3 22-FT DIAMETER PAYLOAD BAY ORBITER SHAPE. Because of the large cross-section required by the payload and the relatively small size of the orbiter it was not possible to utilize the Figure 3-1 lines. A new set of lines developed for the orbiter included a blunt nose and a reduced ramp angle to reduce pitchup hypersonically. This is shown later in Section 4 of this volume.

The final vehicle elements and the shapes utilized are:

System	Booster Element	Orbiter Element
FR-3 (15-ft diameter payload)	Figure 3-2 lines	Figure 3-1 lines
FR-3 (22-ft diameter payload)	Figure 3-2 lines	New lines. See Figure 4-13
FR-4 (15-ft diameter payload)	Figure 3-1 lines	Figure 3-1 lines
FR-4 (22-ft diameter payload)	Figure 3-1 lines	Figure 3-1 lines

3.2 ELEMENT SIZING

The Convair two-stage recoverable synthesis program was used to size the final FR-3 and FR-4 vehicles. This program consists of a weight/volume sizing program integrated with an ascent trajectory program and a booster flyback program.

The vehicles generated in the earlier part of the study were used as the initial basis for the final iteration. The design process consisted of the various technical functions examining the configurations to create input data for the final synthesis. (Volume III, Section 1 indicates the process.) Once the vehicle shape was established the geometric parameters, volumes, areas, and dimensions were provided as input data to the program along with weight equations for major components such as structure and the thermal protection subsystem, and functional subsystems. These weight/volume relationships were utilized to size the vehicle system using inputs of performance mass ratio derived from the trajectory loop. The booster and orbiter elements were sized depending on the selection of staging velocity. Propulsion inputs were in the form of specific impulse values, numbers of engines in booster or orbiter, liftoff thrust-to-weight ratios (or fixed thrust as an option), mixture ratios, propellant densities, etc.

Booster/orbiter staging and orbital injection conditions were iteratively satisfied by the control logic using trajectory pitch control and element mass ratios. Booster fly-back fuel was determined and included in the total configuration sizing. Each run yielded a shuttle configuration sized to perform the specified mission for a given payload, within specified constraints and staisfying given staging and orbital injection conditions. Included in the output were the trajectory history, weight, volume, design data, flight sequence weights, and a summary page with significant weight, volume, geometry, propulsion, and trajectory data. The organization of the synthesis program is indicated in Figure 3-3. A more detailed schematic of the weight/volume subroutine is shown in Figure 3-4.

The program was also used for all parametric excursions and sensitivity runs where the payload value was held and the system gross liftoff weight and total system dry weight were allowed to vary.

3.3 THE BASIC ORBITER DESIGN

The orbiter design derived from the Figure 3-1 lines is presented in Figure 3-5, which shows the inboard profile of a typical orbit element with the 15-ft-diameter by 60-ft-long payload bay. This orbiter arrangement is typical of the final FR-3 and FR-4 vehicles which are described later in Section 4. The orbiter shown in Figure 3-5 was

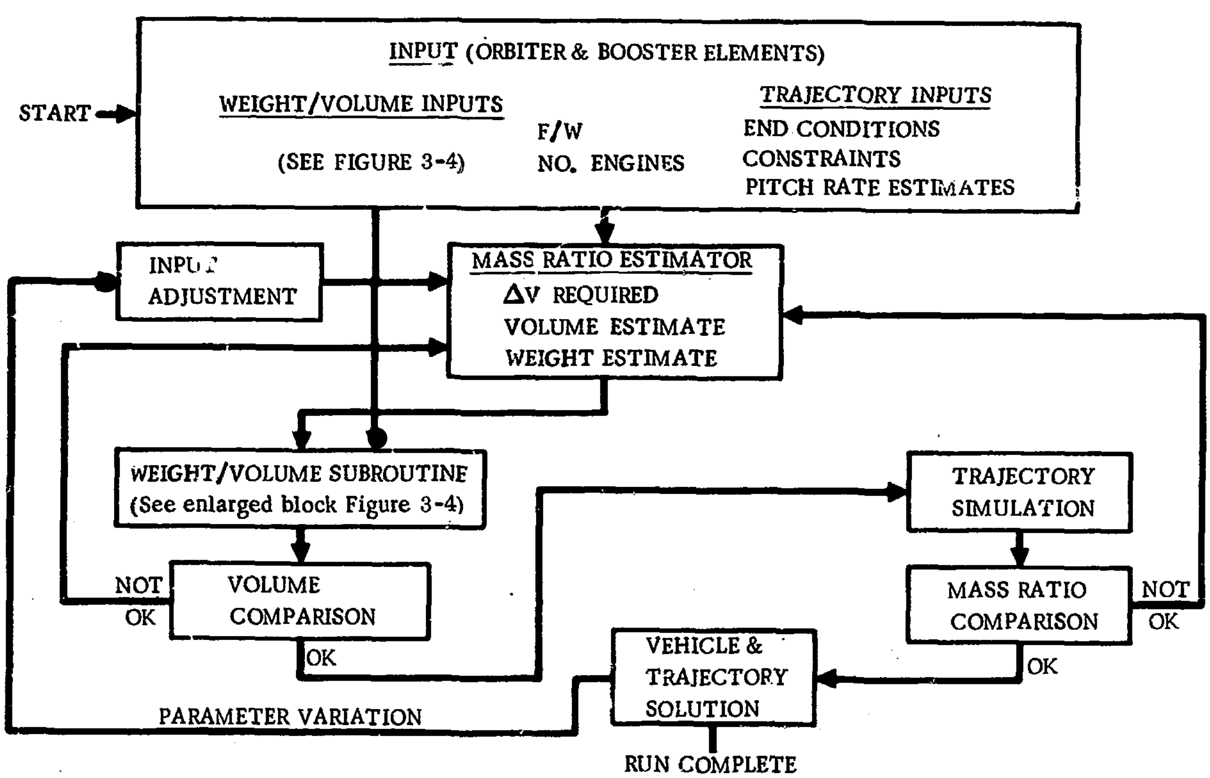


Figure 3-3. Typical Space Shuttle Synthesis Program Schematic

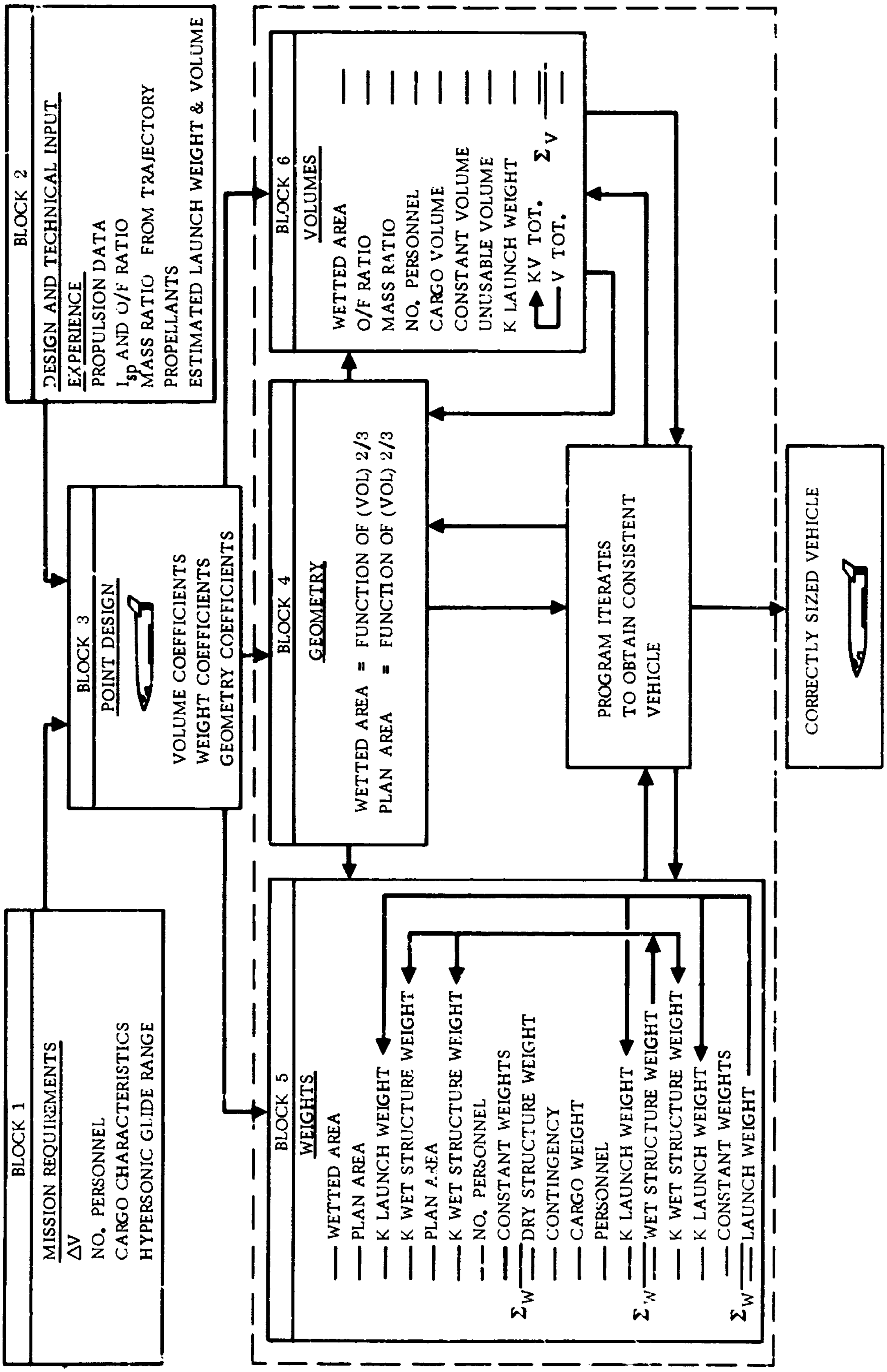


Figure 3-4. Weight/Volume Subroutine Flow Diagram

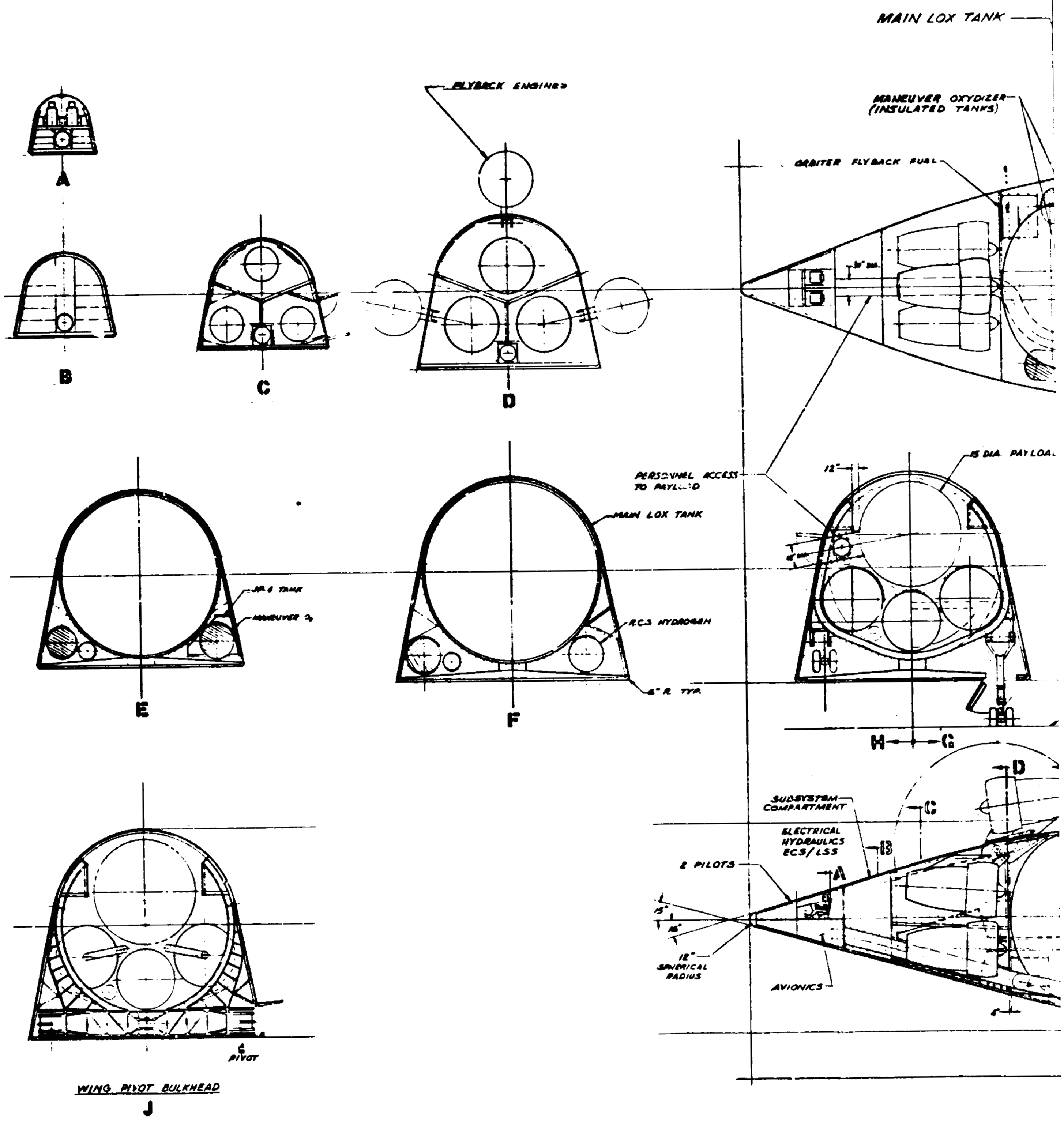


Figure 3-5. Typical T-18 Orbiter Element

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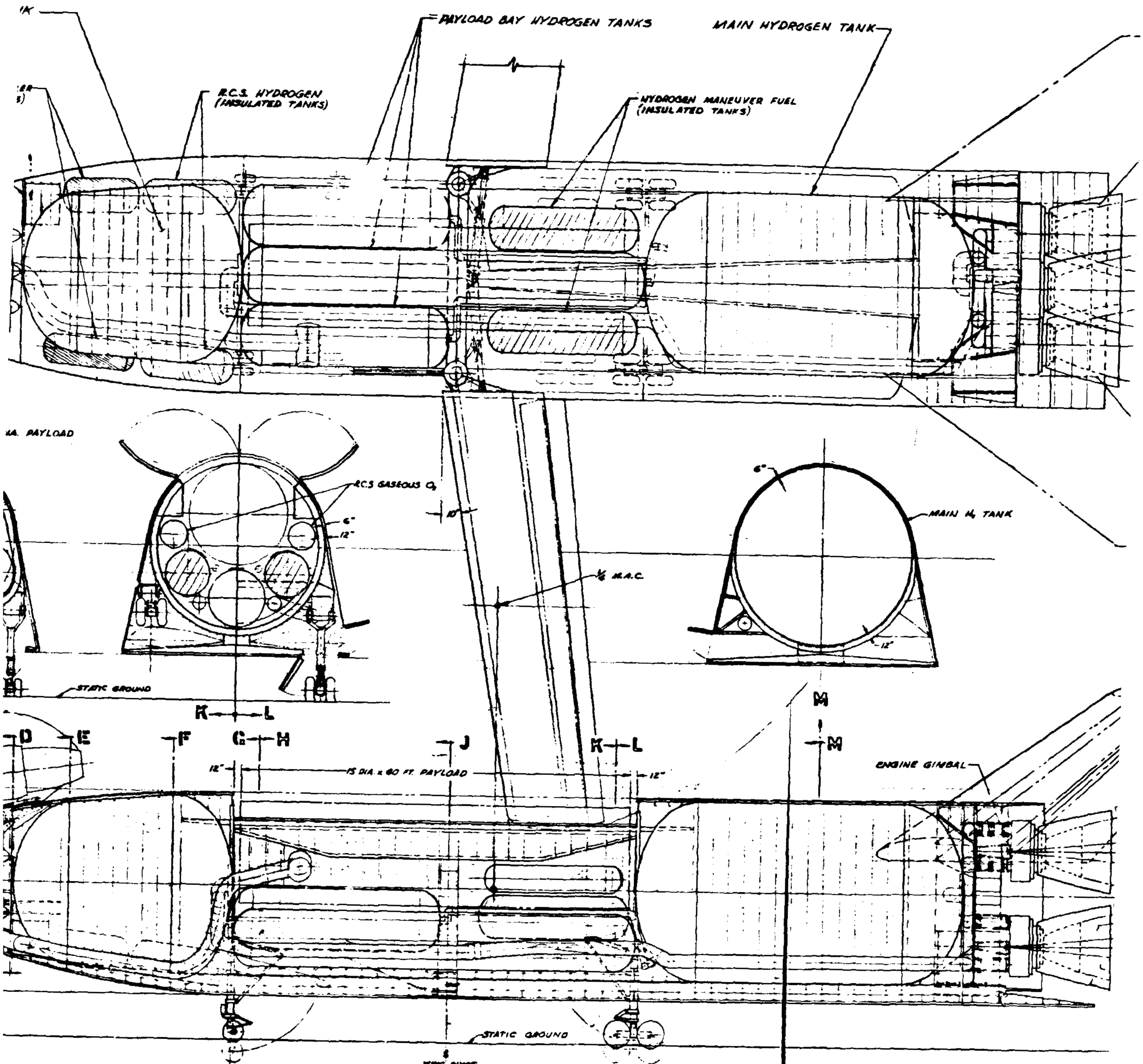
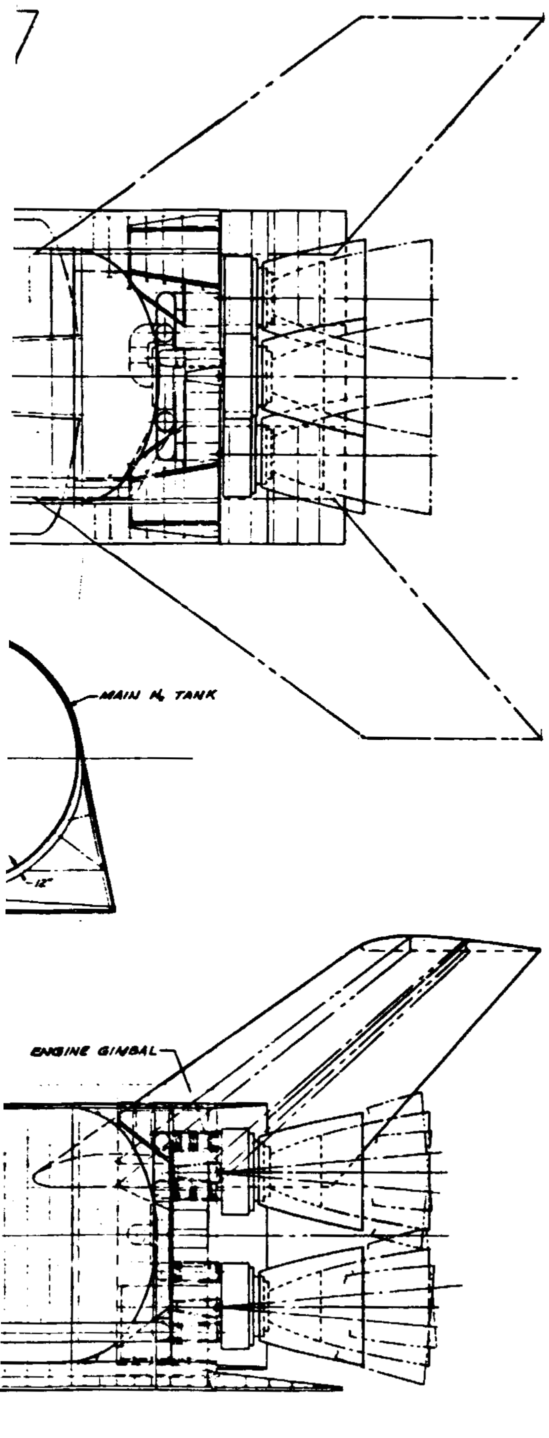
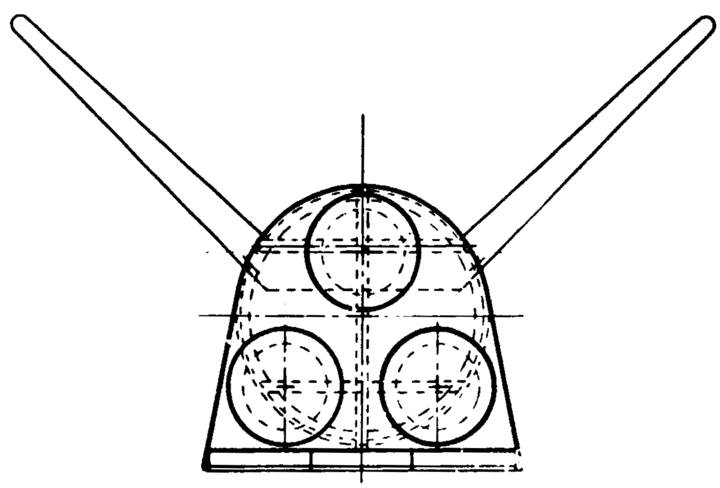


ORBITER INTERNAL ARRANGEMENT

FOLDOUT FRAME 2





ORBITER END VIEW

FOLDOUT FRAME 3

developed for the FR-1 configuration and is 204 ft long. This is larger than the final vehicles by percentages ranging from 7% in the case of the FR-4 15-ft diameter payload bay orbiter to 14% in the case of the FR-3 15-ft diameter payload bay orbiter. However, suitable internal volume packaging factors are introduced in the synthesis of these orbiters to allow for the difference, and the layout as shown in Figure 3-5 is described here in detail as typical of both of these orbiters.

The general arrangement of the orbiter consists of a nose compartment 14 ft long. This accommodates the crew, instruments, controls, and consoles in a cabin with conventional side-by-side seating. The seats are adjustable to provide crew support against axial loads during boost phase. Airline type visibility must be provided for landing and for ferry flight horizontal takeoff, requiring the design of retracting visors over the windshields for protection during the entry heat pulse. This forward cabin also includes the guidance, navigation, and communications equipment.

Immediately aft of the pressure bulkhead which terminates the crew compartment at station 14 ft aft of the nose is a subsystems compartment approximately 7 ft long. This compartment will house the following electro-mechanical subsystems:

a. Electrical power generation and distribution.

b. Environmental control and life support.

c. Hydraulic power generation and distribution.

They are described in Section 6 of this volume.

As shown in Figure 3-5, the flyback engine compartment is located aft of the subsystems compartment and forward of the main LO_2 tank. Off-the-shelf flyback engines from the existing range or immediately projected range of fan jet engines are installed in this compartment. The FR-4 orbiter requires two engines in the 40,000-lb-thrust class, such as the Rolls Royce RB 211-22. The method shown for rotating these engines to the flying position is based on a hydraulically-actuated double-acting mechanism principle employed in the wing-fold mechanism of carrier-based aircraft. The arrangement provides a means of readily exposing the engine for maintenance operations.

Engine compartment doors will be closed after the engines are extended, to maintain a clean aerodynamic shape for minimum drag in the flyback condition.

Fuel lines from the JP-4 tank and hydraulic lines will be routed through the pivot fittings using standard swivel type joints. Electrical wires will be bundled and installed to form a flexible goose neck to provide for the rotation.

The main LO_2 tank forms an integral part of the structure. It uses internal frames and external stringers with pure monocoque domes on the ends. The tanks do not have internal insulation. Access for inspection and maintenance is provided through a hatch on the forward bulkhead. This tank contains 10,075 ft^3 (FR-4) and 7618 ft^3 (FR-3) of LO_2 and is pressurized to 24.0 psig. The tank tapers along its entire length from the forward bulkhead to the aft bulkhead interface. All bulkheads have a radius-to-height ratio of 1.414 to eliminate compression.

The area below the LO_2 tank is used for storage of small propellant tanks for the orbiter vehicle. Two five-ft diameter superinsulated tanks are located on each side providing storage (up to seven days) for secondary LO_2 (maneuvering oxidizer). Two five-ft-diameter superinsulated tanks are located on each side in aft end of bay for LH_2 for the attitude control propulsion subsystem. A single tank is provided in the forward righthand side of the compartment shown in Figure 3-5 for JP-4 fuel for the orbiter flyback engines.

The payload compartment is shown in Figure 3-5. This compartment contains the 15-ft-diameter by 60-ft-long payload bay, which is mounted at the centerline in the fuselage. Access for deployment and retrieval of the payload is through hydraulically-actuated double doors, which form the top of the vehicle in this area. Two large longerons, located on each side of the payload, support the payload and the top access doors and carry structural loads through the compartment. They are tapered at each end and terminate at the propellant tank integral structure.

The primary structure below the longerons is made up of 12-inch-deep frames formed to the outside diameter and covered with skin and stringers. This diameter corresponds to the outside diameter of the integral LH_2 tank just aft of this compartment, providing structural continuity through the payload bay forward to the integral LO_2 tank. A slight transition is made to this circular shape in the forward portion of the bay to provide maximum space utilization for supplementary propellant tanks.

The wing pivot bulkhead is located approximately mid-bay and supports the variable-geometry wing with large clevis fittings at the outboard end of the carrythrough truss structure. The upper portion of the bulkheads is designed to clear the single propellant tank in the booster.

At the forward and aft ends of this payload compartment, heavy bulkheads support the two nose landing gear assemblies and the two main landing gear assemblies. The landing gear installations are completely outside the basic circular structure of the payload compartment. The nose and main landing gear concept is very similar to the B-52 arrangement. The fuselage of both vehicles remains level during ferry takeoff and landing operations (no rotation), allowing placement of the fore and aft gear assemblies at a convenient structural attachment point fore and aft of the vehicle center of gravity (in this case, the heavy structure at each end of the payload bay.) The wide tread of the T-18 (26 ft) negates the need for outriggers as used on the B-52 and is well within the maximum turnover angle specified by specification AFSCM 80-1.

If ground directional stability becomes a problem with this arrangement, a conventional tricycle gear arrangement can be adopted (although the turnover characteristics will be marginal).

In the landing gear arrangement shown, the wings must be deployed before the main gear can be extended. This is not considered a disadvantage since no attempt to make a gear-down landing with wings stowed would ever be made. For ground handling convenience, it is possible to sweep the wings aft until the trailing edge is adjacent to the main gear strut. This gives a wing span approximately equal to the fixed vee tail span, which would be the designing dimensions for factory bay widths, etc. For cycling of the wing during checkout the vehicle is put on jacks in the horizontal position, as all aircraft are for cycling of their landing gear, and the gear and wing mechanisms are checked at the same time. The landing gear is operated hydraulically via conventional actuators. The wing is deployed via screwjacks driven by hydraulic motors in similar fashion to the F-111 system.

Supplementary propellant tanks are located in the lower portion of the orbiter payload bay. Three tanks contain part of the main hydrogen fuel: one tank is located on the centerline running the full length of the bay, and two tanks are located on each side in the forward bay. Two superinsulated tanks are located on each side in the aft bay, providing storage (up to seven days) for secondary hydrogen (maneuvering fuel). In addition, two tanks are located below the longeron in the aft end for gaseous oxygen for the attitude control propulsion system.

The main LH_2 tank forms an integral part of the structure in this area. It uses external frames and stringers with pure monocoque, $r/h = \sqrt{2}$, domes on the ends. This tank is internally insulated. An access door for inspection and maintenance is located in the forward dome. The tank contains 19,140 ft^3 (FR-4) and 15,050 ft^3 (FR-4) of LH_2 and is pressurized to 28.5 psig. The tank has a constant diameter over its entire length.

The compartment aft of the main hydrogen tank contains the thrust structure, engine gimbal support, stabilizer attach structure, interstage connection, propellant lines, and pressure volume compensating ducts. An outer fairing surrounds the entire compartment.

Three high pressure bell nozzle rocket engines are supported at gimbal points on the thrust structure beam intersections. The engines are protected during entry by an extension of the lower surface. The aft end is described further in Section 4., where the point design vehicles are covered.

Additional orbiter items are:

a. Personnel Access Tube. A 30-in.-diameter pressurized tube is provided for access of personnel from the cockpit to the payload bay. The tube is routed from the pressure bulkhead at Station 14 aft along the bottom centerline through the subsystem compartment, to the bulkhead at the aft end of the flyback engine compartment. At this point, it swings out around the lower forward LO_2 tank dome, along the lower left side of the compartment between the lower structure and the LO_2 tank. It then swings up around the aft LO_2 tank dome and into the payload bay compartment under the upper longeron. The aft end is connected to a larger, 42-in.-diameter tube which has a flexible coupling for attaching to the airlock on the payload. (Location on the payload is assumed at this time since no actual payloads have been designed.) The outboard end of the larger tube is connected to an access door through the outside of the vehicle. This access can be used for ingress to the payload bay just prior to launch.

b. Attitude Control Propulsion Subsystem. Propellant tanks for the ACPS have been discussed previously. The system uses 48 thrusters: 24 located forward in the flyback engine compartment, and 24 located aft in the thrust sturcture area.

c. Thermal Protection Subsystem (TPS) Supports. The lower TPS is supported on a pedestal structure below the basic integral vehicle structure, as shown in Sections E through M of the configuration drawing. The cross beams are cantilevered from the pedestal to support the TPS.

The upper TPS is supported by structural members between the integral structure and the outside frames, as shown typically in Section M.

d. Wing Stowage. The wing is stowed between the circular integral structure and the lower TPS support beams, as shown in Sections K and L. An access door extends from the wing pivot aft. The door will be split just aft of the extended wing trailing edge to allow the aft portion of the door to be closed when the wing is extended. The portion above the wing will be extended upward to lay flat against the fuselage side to minimize drag.

The basic structural materials are aluminum alloy for tanks and main body structure, titanium alloy for the lower heat shield support and thrust structure, and aluminum boron composite in longitudinal stress situations such as beam caps and stiffeners. The thermal protection subsystem consists of microquartz and dynaflex insulation external to the basic structure with post-supported cover panels, primarily of cobalt alloy on the lower surface and 811 titanium alloy on the upper and side surfaces. The vee tail is thermally protected by a similar radiation heat shield.

SECTION 4

FINAL VEHICLE DESIGNS

4.1 VEHICLE LAYOUT, CHARACTERISTICS, AND PERFORMANCE

The following section summarizes the final vehicle properties developed in Phase A.

4.1.1 FR-3 TWO-STAGE SEQUENTIAL-BURN 15-FT-DIAMETER × 60-FT-LONG PAYLOAD BAY SYSTEM. The FR-3 launch configuration is shown in Figure 4-1. The orbiter and booster elements are arranged with their flat lower surfaces adjacent, their nozzle exits aligned, and the booster engine nozzles extended. Initially, the orbiter and booster were located nose-to-nose on earlier configurations (see Volume III). Loads analysis, however, shows that booster loading will be considerably reduced in the tail-to-tail configuration and that orbiter loading can be slightly reduced. This is a major reason for the tail-to-tail longitudinal location in the final vehicles. A second reason is that wake effects on the booster from the blunt base of the orbiter present an unknown in the nose-to-nose configuration, which is avoided in the selected arrangement. Thirdly, the launch stack is more easily handled if the orbiter is located near the pad surface instead of a long way vertically up the booster length. The selected configuration also allows the interstage attachments to be made via the higher density thrust structure of both elements as well as at logical forward attach points such as the wing pivot or main landing gear bulkheads. The bottom surface to bottom surface launch configuration was selected over a piggyback arrangement because it reduced stage interconnect distances, provided a clean aerodynamic launch configuration, and reduced potential physical interference between the element stabilizers.

It is noted that the current FR-3 booster is shorter than the previous versions, so the disparity in length as shown in Figure 4-1 is not as great as it was. However, the design of future orbiter shapes with reduced crossrange/hypersonic L/D capability will tend to reduce the orbiter length also.

The asymmetric center of gravity of the two-element combination requires that the net thrust line of the booster be canted 6-1/2 deg to an averaged position such that this becomes the manufacturing zero position. The engines then gimbal either side of this 5 degrees to accommodate the offset c.g. at liftoff, maximum αq, and burnout.

Stage separation is currently envisioned as a longitudinal separation using a guide rail on the booster. Completely passive separation of the orbiter using the difference in ballistic coefficient between the two vehicles takes about 15 secs. This time can be reduced by using a retrorocket in the booster nose or by running the orbiter engines in the throttled mode, depending on the impingement problem effects on both vehicles.

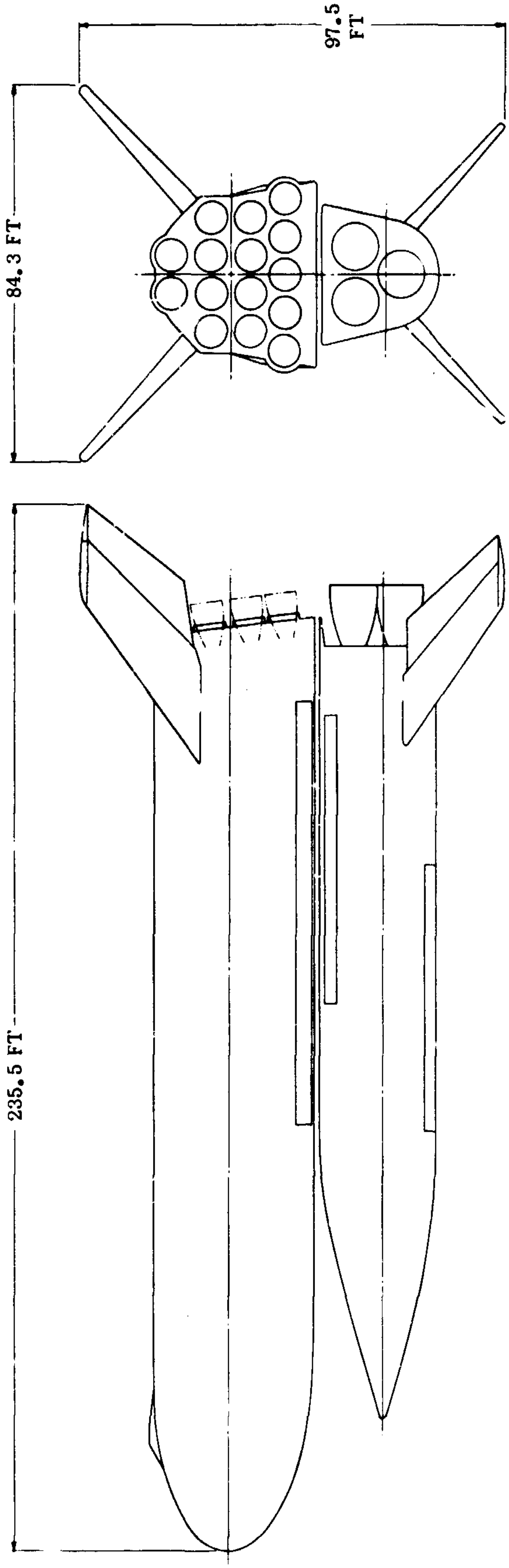


Figure 4-1. FR-3 Launch Configuration

The orbiter engines do not burn at liftoff. They are ignited at staging. Previous investigations in the ILRV and Space Transportation Systems studies have shown that orbiter burn with or without throttling is detrimental to system performance unless crossfeed is incorporated. It would be better to expend only a fraction of this weight in developing upper stage rocket ignition subsystems with a very high reliability.

The orbiter element, as seen in Figure 4-1 and again in a three-view drawing in Figure 4-2, is similar to that described earlier in Section 3 of this volume. It is smaller in external size, having a reference length of 179 ft (versus 204 ft) from the nose to the trailing edge of the lower body flap. The payload bay is, of course, the required 15-ft-diameter by 60-ft-long, and the internal design and volume factors have been adjusted to compensate for this.

The shape of the FR-3 booster has been shown previously in Figure 3-2. The three view drawing and general arrangement are shown in Figures 4-3 and 4-4. The FR-3 booster element has a constant cross-section with a blunt nose. Maximum usage of tankage space is made except that separate LO_2 and LH_2 tanks are incorporated as explained later in this section. The structural design has been modified to one of integral tanks of 2021 or 2022 aluminum alloy with a radiation protection subsystem of 811 titanium fairing cover panels on the upper surface, and HS188 cover panels over microquartz insulation for the lower surface. Only minimal insulation is required due to the short period heat pulse in the booster entry trajectory.

The forward compartment contains the four RB211-56 flyback turbofans together with their spherical JP-4 tank. The engines pivot outboard to the deployed position via large trunnions attached to engine support pylons. Engine feed and control lines pass through the pivot point. Engine compartment doors are closed immediately after deployment of the engines. The forward compartment also contains the electrical and hydraulic power generation subsystems and the nose landing gear. The nose landing gear incorporates twin 36 × 11 Type VII tires. The two-pilot crew compartment consists of a cupola above the nose compartment. This provides side-by-side crew seating and contains consoles, instruments, and displays. Guidance, navigation, and communications equipment are also installed within the nose compartment as well as the crew cabin pressure and life support subsystem, which is simplified by the booster mission flight time of just over one hour.

The liquid oxygen tank and liquid hydrogen tanks are both 33 ft in diameter and both have bulkheads with $r/h = \sqrt{2}$. Four LO_2 lines feed aft from the LO_2 tank, two dorsally and one each within the lower heat shield. The tank frames and stringers are all external and the intertank section is of similar construction. The body external shape is bulged slightly in the region of the stowable wing pivot point carryover bulkhead in order to increase the frame depth without interfering with tank skin continuity. The "bulge" in the external lines will have negligible effect on vehicle aerodynamics. It only affects the upper radius and is faired out before the body leading edge is reached so that no discontinuity appears in the plan view. This shape is carried over to include the increased depth of the main landing gear attach frames.

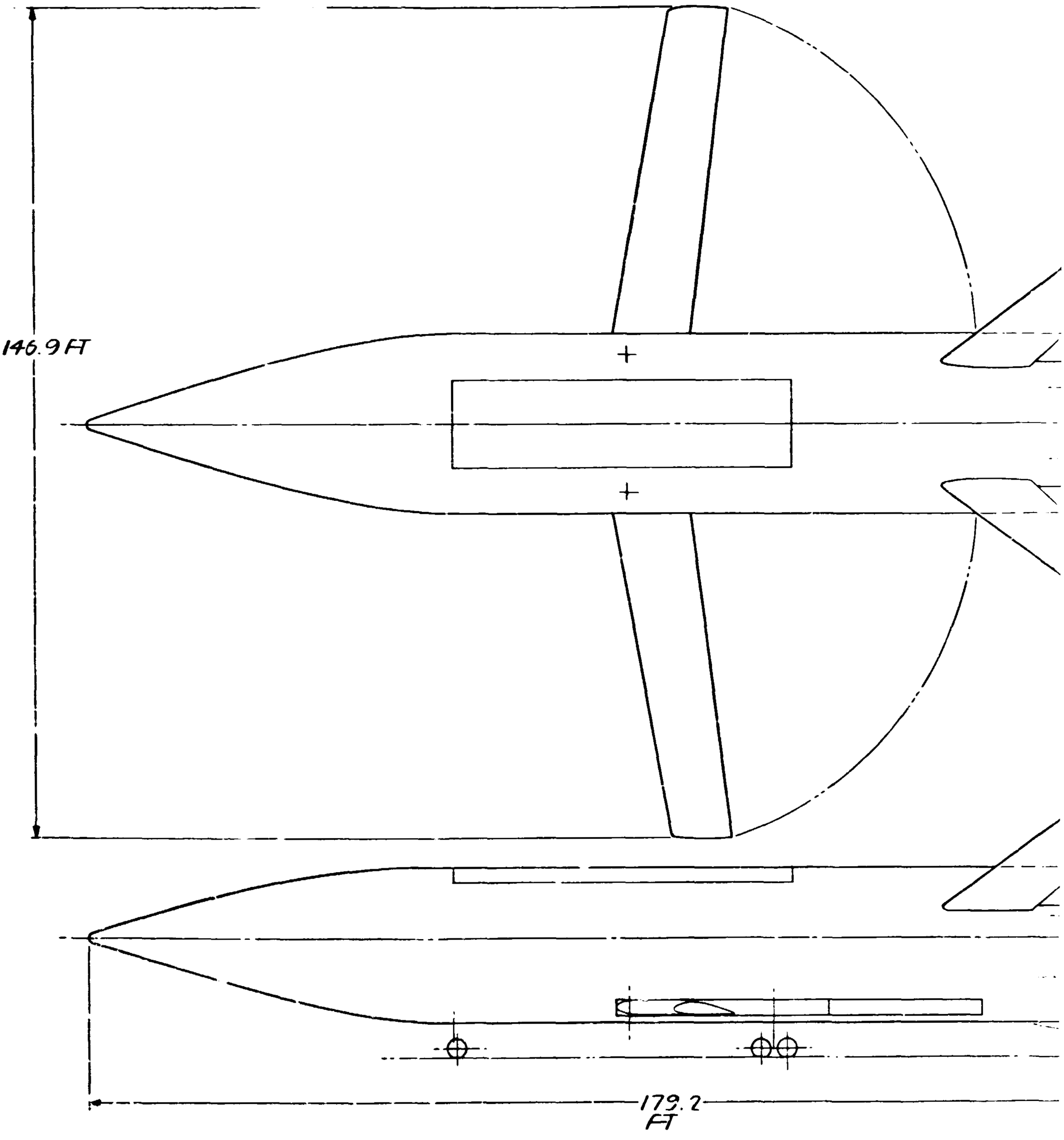


Figure 4-2. 'R-3 Orbiter

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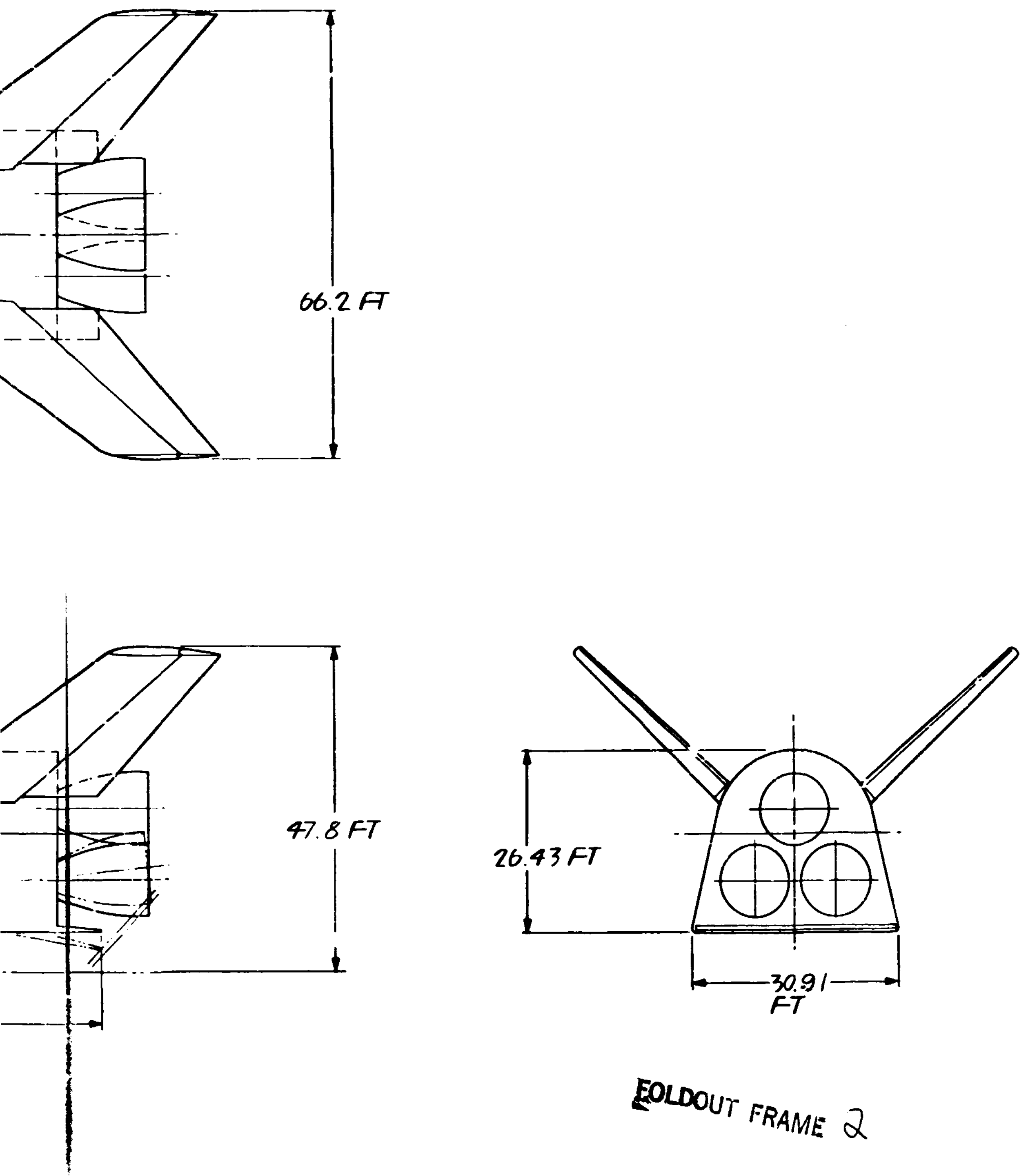

FOLDOUT FRAME 2

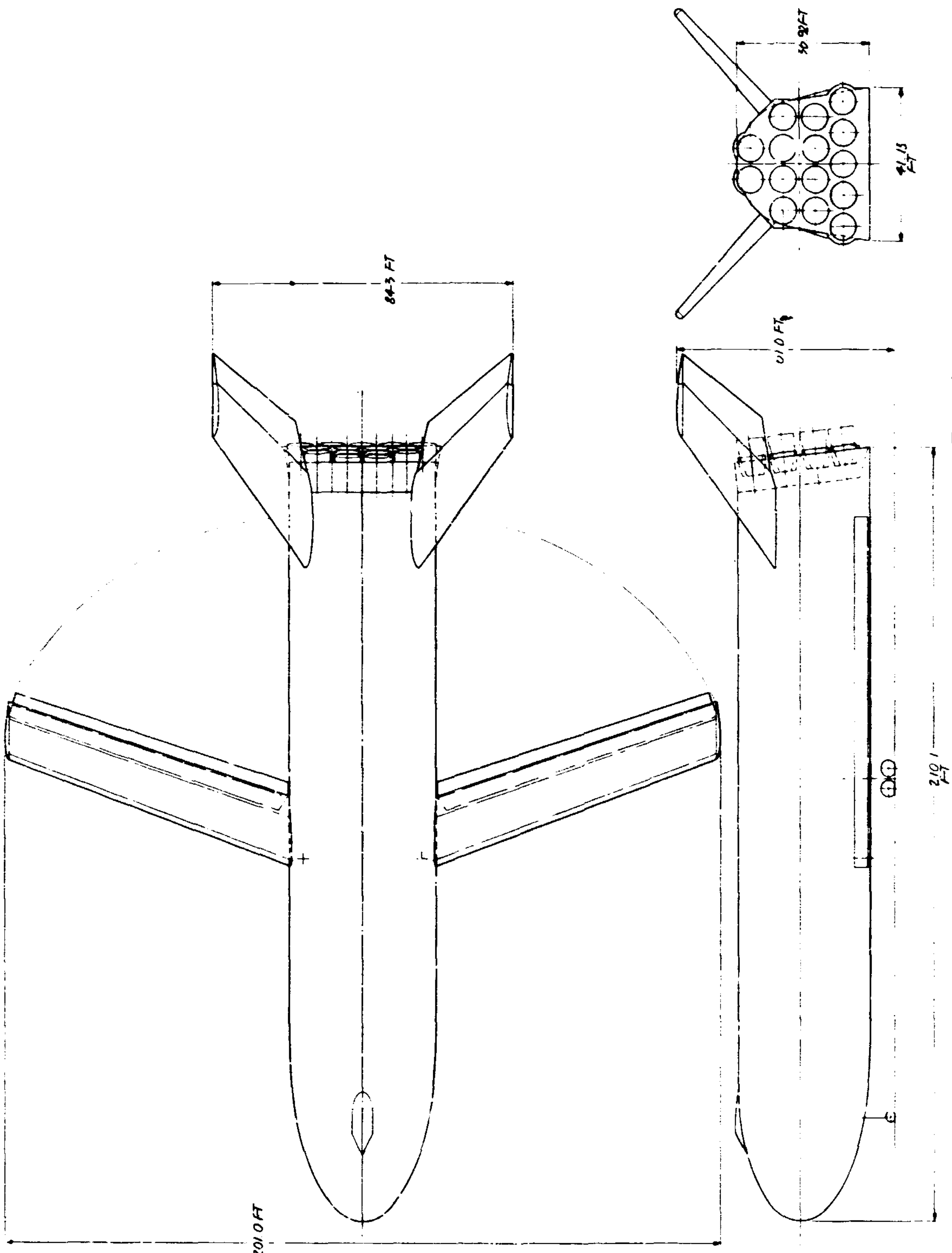

Figure 4-3. FR-3 Booster, Three-View Drawing

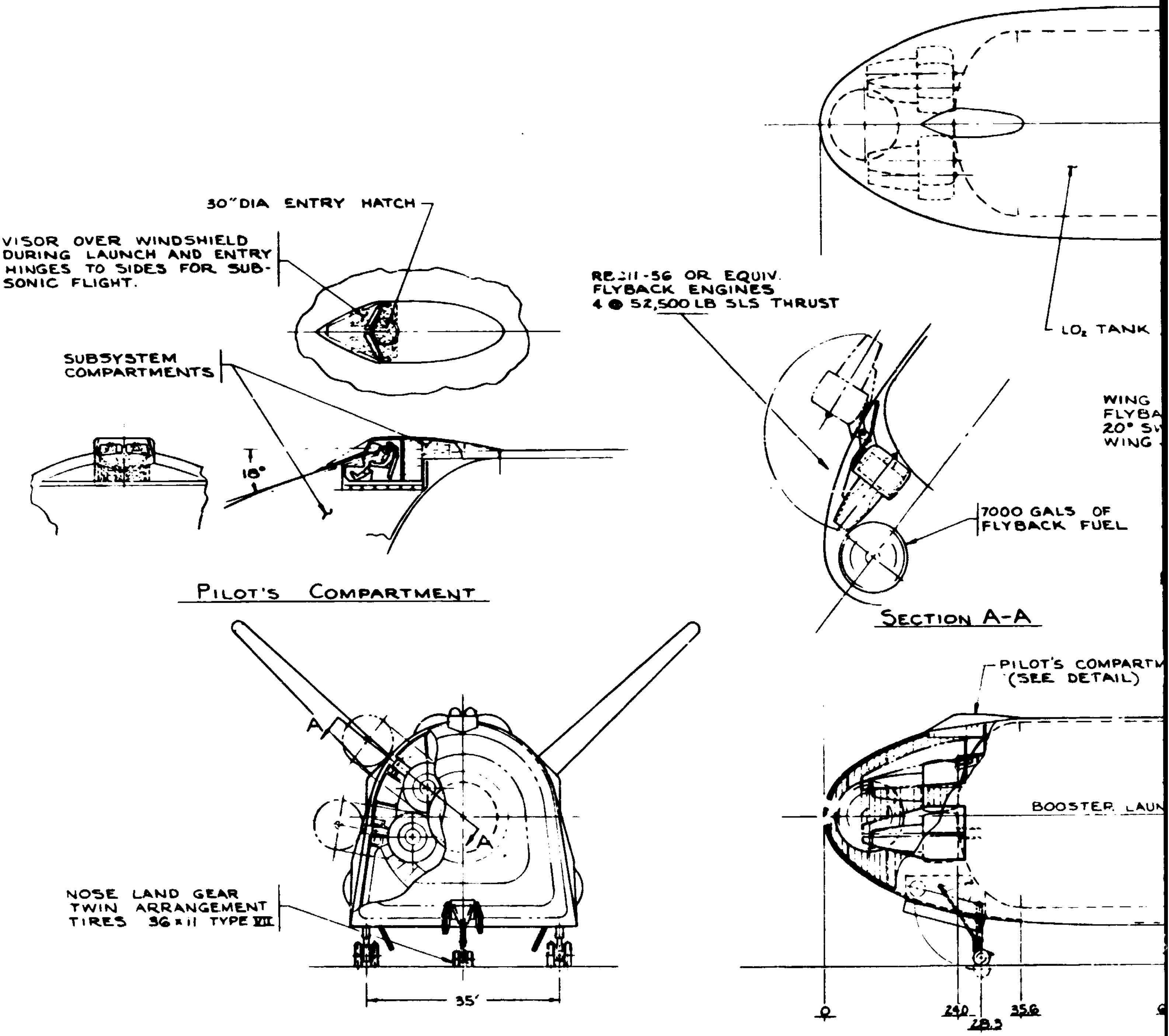

Figure 4-4. FR-3 Booster Basic Configuration

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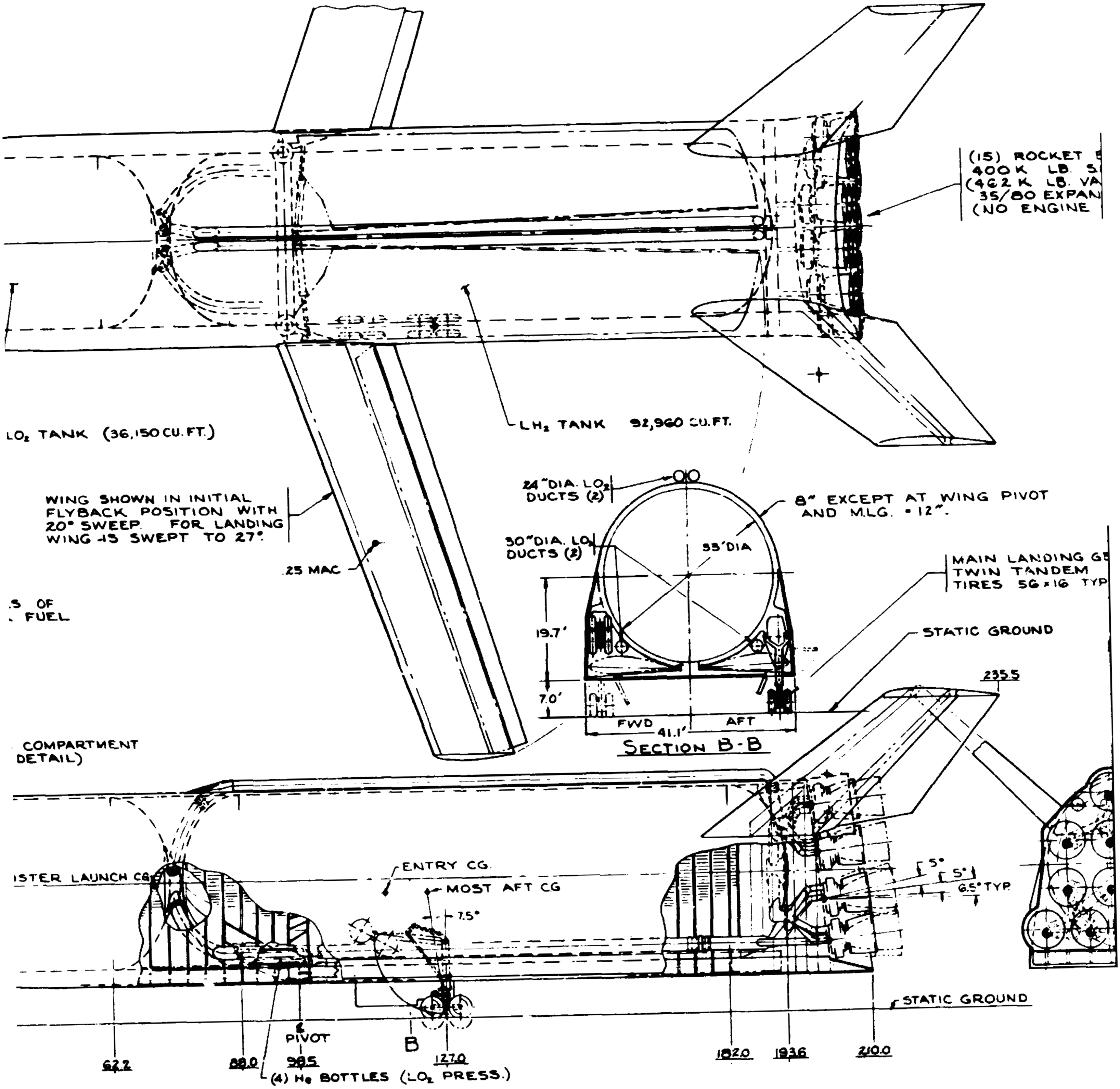
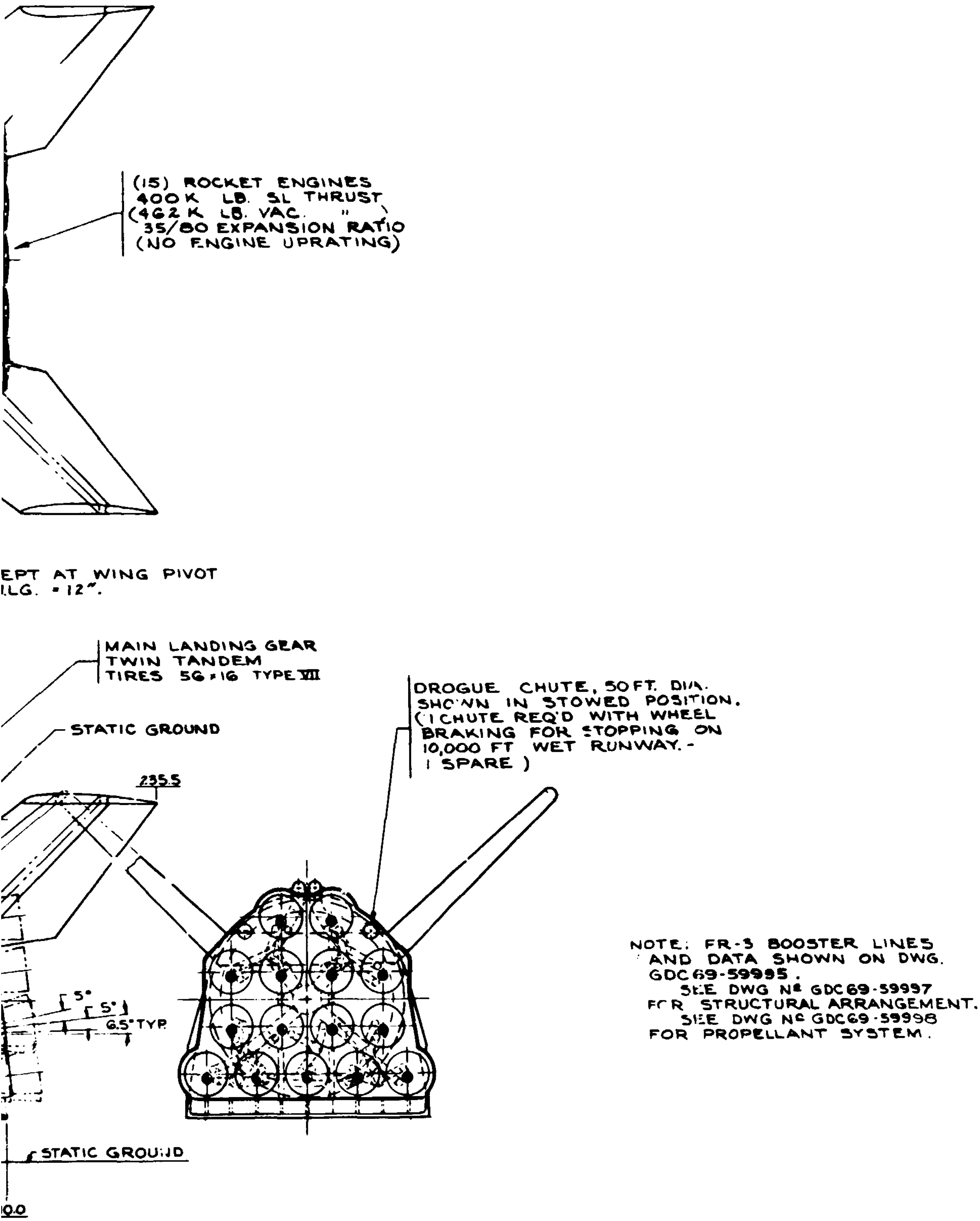


FOLDOUT FRAME 2



FOLDOUT FRAME 3

The lower heat shield and wing stowage arrangement is similar to that shown previously for the FR-4. The heat shield and side fairings are supported by the integral tanks, which form the primary load carrying structure. The wing doors are segmented to allow locking after passage of the wing on deployment. The wings are deployed by screw jacks driven hydraulically and slaved together for even deployment. The wings are a conventional structure with full span inverting flaps. At the start of cruise, the wings are deployed to about 20 deg of sweep, and as the fuel is used up in the nose, the wings are brought aft to 27 deg of sweep for landing. The entry c.g. is currently at 54.7% of the body length and the aftermost c.g. at landing is 58.9%.

Construction of the liquid hydrogen tank is similar to the liquid oxygen tank. Internal polyurethane insulation is used in the LH_2 tank, or open faced honeycomb, in order to reduce the problem of heat leaks and temperature differentials especially at such major structural tie-in points as the wing, landing gear, and thrust structure connections.

The LO_2 tank is located forward of the LH_2 tank for launch c.g. reasons. It is necessary to keep the c.g. as far forward as possible in order to reduce the offset thrust vector requirement. (In this respect, a nose-to-nose configuration has an advantage but compared to the previously mentioned disadvantages, it is not significant). The forward location of the LO_2 tank does incur a high head at the aft end of the LO_2 lines, but this is still acceptable.

The main landing gear consists of conventional four-wheel bogies per strut. Four 56 × 16 Type VII tires are used per side. The wheel spacing is sufficient for flotation on SAC type runways. Minimum weight braking is combined with a 50-ft-diameter drogue chute to provide landing over a 50-ft screen on a 10,000-ft wet runway. A backup drogue chute is carried, but only one at a time is utilized. The main landing gear retracts forward and is restrained in the down position by tension drag links. Landing gear doors are provided in the heat shield. Some local straight lining of the side slope angle is allowed to create a slight bulge in the contour to accommodate the main gear. The landing angle of the booster is very close to the horizontal and the main gear is kept as short as the 2 g nominal landing oleo compression will allow. The Type VII tire capability is well within the anticipated landing speed of 180 knots.

The thrust structure at the aft end of the hydrogen tank consists of a beam matrix with the ends supported from a circular skirt extension of the tank diameter. Fifteen of the required 400,000-lb sea level static thrust bell nozzle, two-position rocket engines are supported from gimbal point pads at the thrust structure beam intersections. The propellant lines are manifolded within the aft compartment to feed groups of engines. Pressure volume compensating ducts are used at each propellant feed inlet. The engine pattern is slaved electronically to operate together. The engines are spaced based on the design rule that they will be returned to the null position in case of failure. The expansion ratio of the 15 booster engines is 35 in the retracted position and 80 in the extended nozzle position. The vacuum thrust of the engines is 462,000 pounds each. The engines will be throttled starting with the outermost pair during boost to maintain the 3 g axial limit and at the same time to relieve the asymmetric thrust situation. The nozzle extensions are retracted for entry to remain within the protection of the lower surface heat shield extension.

The vee tail is set at 45 degs dihedral with a true sweepback on the leading edge of 45 degs. The ruddervators on the vee tail have their hinge li·· ·· at 65% chord with 35 deg up and 10 deg down deflection. A nickel alloy hot stru··ture empennage is proposed for the current design.

The characteristics of the FR-3 system are summarized in the synethesis program output of Table 4-1. The gross liftoff weight is ·.329 million lbs. Additional major weights, volumes, dimensions, propulsion and trajectory data are given. The flyback range of 281 miles is flown with subsonic L/D = 7.2. Note that in the propulsion section the nominal and the uprated outputs, coded NOM/UR, are the same since no uprated thrust is used. The orbiter element, which has three nominal 400, 000-lb sea level thrust engines each with an expansion ratio of 160, has a vacuum thrust of 472, 000 lb per engine.

Figure 4-5 shows volume and wetted area plots of the FR-3 booster. Additional geometry is given in the summary of all the vehicles given in the latter part of this section.

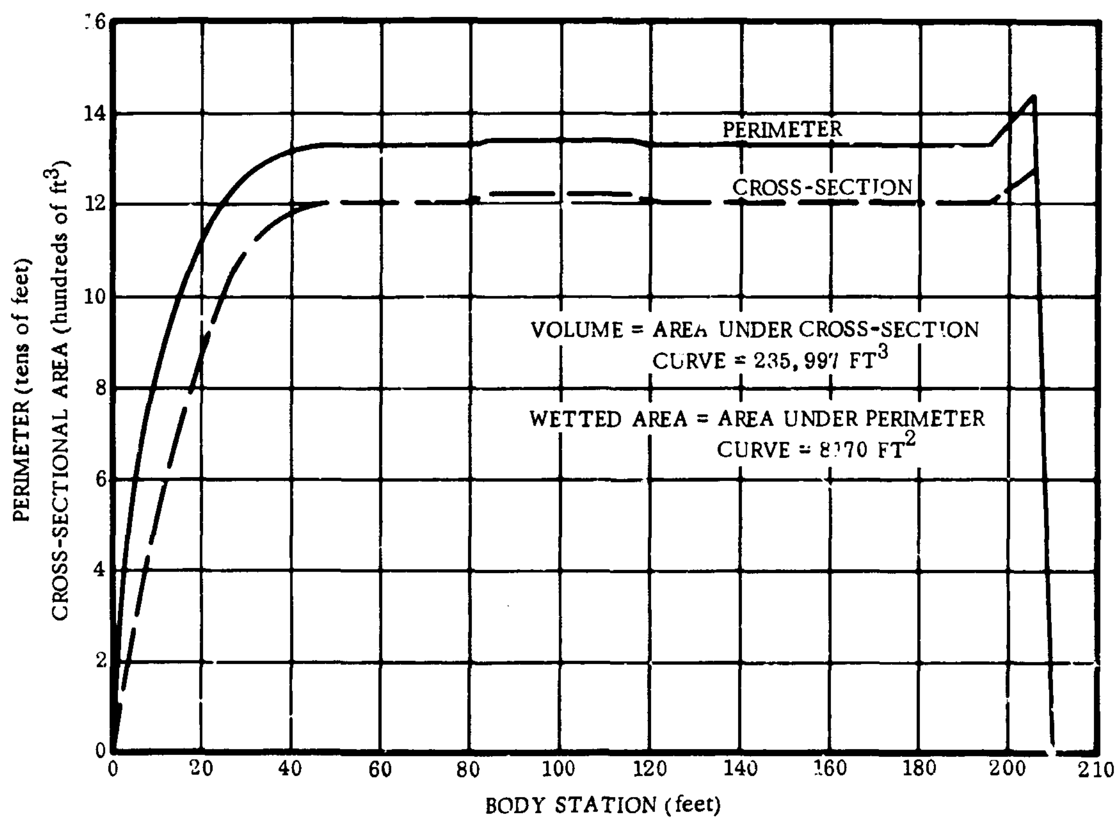


Figure 4-5. FR-3 Booster Wetted and Volume

Table 4-1. FR-3 15-Foot-Diameter Payload Baseline System Synthesis Summary

NASA FR-3, 50K L 15-3 ENGINES (400K) — 10/20/69 SEPARATE BULKHEAD

	BOOSTER ELEMENT	ORBITER	VEHICLE
WEIGHT			
PROPELLANT, ASCENT	2811879	592241	
PROPELLANT, ORBIT MANEUVER		37317	
PROPELLANT, TOTAL	2811879	629558	
FLYBACK FUEL	46916	2868	
PAYLOAD		50000	
STRUCTURE	469868	213150	683018
CONTINGENCY	46987	21315	
OTHER	26667	9793	
TOTAL	3402316	926685	4329000
IN ORBIT		339578	
RETURN CONDITION	590437	330445	
ENTRY	0	289655	
LANDING	517462	286787	
VOLUME			
FUEL	92957	15047	
OXIDIZER	36153	7618	
PROPELLANT	129111	22665	
PAYLOAD		10638	
OTHER	106886	55757	
TOTAL	235997	89060	
GEOMETRY			
LENGTH	210.1	179.2	
BODY WETTED AREA	26612.7	14904.8	
BODY PLANFORM AREA	8170.1	4910.0	
ENTRY PLANFORM LOADING	69.1	59.0	
PROPULSION			
THRUST-TO-WEIGHT		1.52774	1.38688
NO. OF ENGINES	15	3	
SL THRUST/ENG NOM/UR	400258/ 400258	238526/	6003864/ 600386
VAC THRUST/ENG NOM/UR	462152/ 462152	471911/	6932280/ 6932280
SL ISP NOM/UR	389.3/389.3	232.0/	389.3/389.3
VAC ISP NOM/UR	449.5/449.5	459.0/	449.5/449.5
TRAJECTORY			
MASS RATIO	2.85343	2.72922	
MAXIMUM DYNAMIC PRESSURE			670.2
STAGING DYNAMIC PRESSURE			50
STAGING VELOCITY (RELATIVE)			10912
STAGING ALTITUDE			187497
STAGING FLIGHT PATH ANGLE (RELATIVE)			2.244
INJECTION VELOCITY (INERTIAL)		25897	
INJECTION ALTITUDE		259999	
INJECTION FLIGHT PATH ANGLE (INERTIAL)		.000	
INJECTION INCLINATION, deg		54.90	
FLYBACK RANGE, n.mi.	260.9		

Note: Units not shown are ft, lb, sec.

Initial excursions on the FR-3 system were made to establish the number of engines required per element. The once-around abort philosophy requires that the orbiter have sufficient thrust-to-weight with one engine out so that by burning up the available on-orbit maneuver propellants the orbiter can overcome the additional misalignment losses incurred by the reduced thrust-to-weight ratio at staging, and can thus make a single orbit, entering to arrive at the launch site as shown schematically in Figure 4-6. The actual increase in ΔV required is plotted versus orbiter F/W in Figure 4-7. This indicates that a minimum F/W in the orbiter of about 0.80 will allow a once-around abort with some reserve considering the available 1800 fps of orbit maneuver velocity propellant stored on board. The thrust-to-weight ratio of the orbiter at ignition, given in Table 4-1, is 1.53, using three of the 400,000-lb sea level engines. With one engine out this reduces to 1.02, which is very adequate for the once-around abort.

Intuitively, it appears desirable to have as few main rocket engines as possible on the orbiter element because of the inert weight sensitivity on the order of 30-lb gross liftoff weight (GLOW) increase for 1-lb increase of orbiter inert weight. In light of this, some thought was given to a two-engine orbiter configuration.

With two engines having 472,000-lbs thrust each (400,000-lbs at sea level), and an orbiter thrust-to-weight requirement of 1.6 (F/W = 0.8 with one engine out) the maximum orbiter staging weight is 590,000 lbs. When this value is superimposed on extrapolated orbiter staging weights versus staging velocities sensitivity data of the design point configuration, as shown in Figure 4-8, a staging velocity on the order of 13,800 fps is required for a two-engine orbiter configuration. Comparing this value with the results of a GLOW versus staging velocity sensitivity for the design point, as shown in

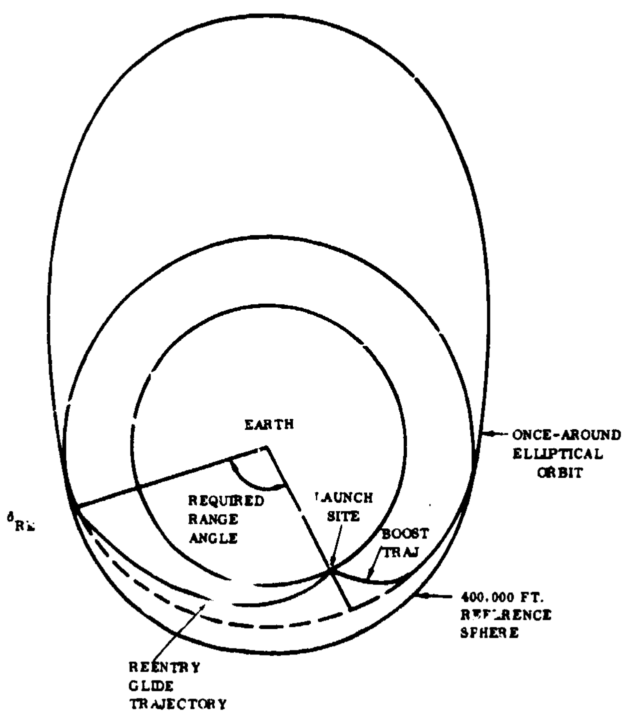


Figure 4-6. Once-Around Trajectory

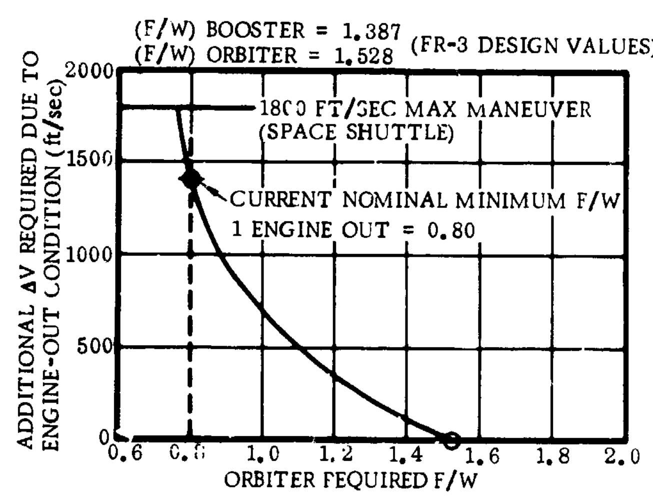


Figure 4-7. Orbiter One Engine Out Abort ΔV Requirement

Figure 4-9, the higher staging velocity value is well off the GLOW minimum point indicating increased GLOW or a need to relax engine thrust or F/W requirements.

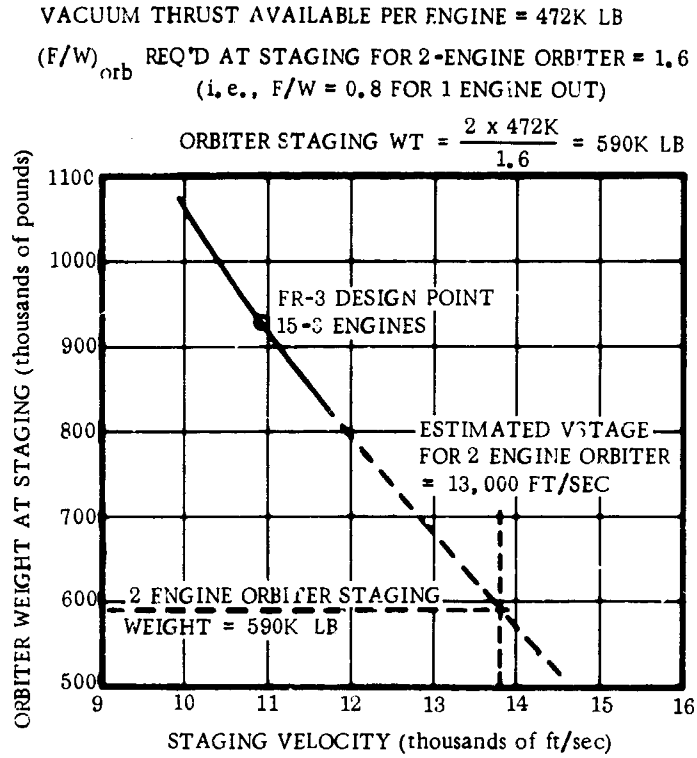


Figure 4-8. Staging Velocity vs Orbiter Staging Weight (FR-3)

Other considerations of increasing staging velocity are indicated in the design point synthesis results shown in Table 4-2. Of particular interest is the booster return weight, which if allowed to increase much beyond the 590,000-lb value would require one more tubrofan engine of the selected thrust level. Increasing flyback range with its attendant fuel increase is another consideration of increased staging velocity. A third consideration is the worsening propellant fraction of the orbiter design as its size decreases. This due to having a fixed payload bay size using a greater proportion of the total volume as the size decreases.

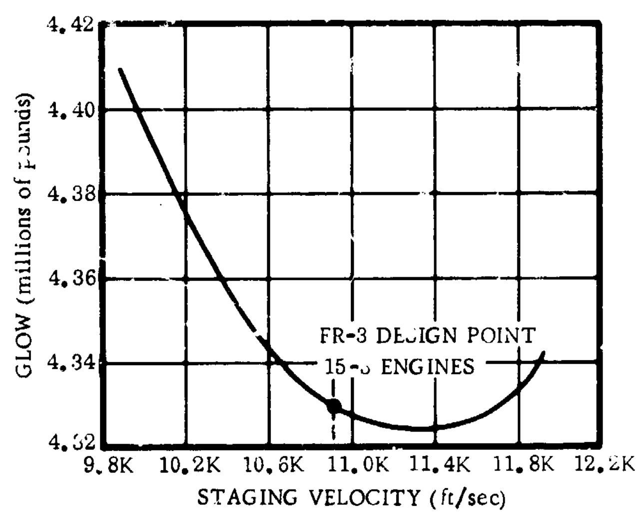


Figure 4-9. Staging Velocity versus GLOW Liftoff Weight (FR-3)

While it is still desirable to reduce orbiter weight as far as possible, results show that three 400,000-lb main engines are necessary to satisfy the design requirements.

Table 4-2. Staging Velocity Synthesis Results, 15-3 Engines

V_{Stage} FPS →	9902	10414	10912	11859
Orbiter Staging Weight, lb	1,07[illegible]02	996,085	926,685	8[illegible]3,798
GLOW, lb	4,408,[illegible]	4,356,658	4,329,000	4,337,128
Booster Liftoff Weight, lb	3,331,173	3,360,573	3,402,316	3,520,330
Booster Return Weight, lb	584,517	586,428	590,437	603,814
Flyback Range, n.mi.	25[illegible].0	2[illegible]0.0	280.9	301.7
Dry Weight (both Stages), lb	701,[illegible]5	690,639	683,018	675,636

Synthesis runs were made to investigate the effect of a 16-3 engine FR-3 system versus the 15-3 engine system. The results are presented in Table 4-3. It is seen that for approximately the same staging velocity, the 16-engine booster resulted in a lower gross liftoff weight. However, the dry weight was lower for the 15-engine FR-3 booster. Because 15 engines should therefore result in a lower system cost, and because the installation was slightly easier, and because vehicle loss due to catastrophic engine failure is statistically less for a lesser number of engines, this was the selected design point despite the increased gross weight.

Table 4-3. FR-3 Number of Engines Comparison

	16-3	15-3
GLOW, lb	4,296,874	4,329,000
Dry Weight (Orbiter + Booster), lb	695,615	683,018
Staging Velocity, fps	10,810	10,912
Max. Dynamic Pressure, psf	811.8	670.2
Booster Return Weight, lb	603,388	590,437
Flyback Range, n.mi	283.1	280.9
Orbiter Stage Weight, lb	939,5[illegible]	926,653
Booster Liftoff Weight, lb	3,357,305	3,402,316

A further investigation was made to compare separate bulkheads versus common bulkheads in the FR-3 booster. A common bulkhead was used between the forward-located LO_2 tank and the LH_2 tank. New geometric inputs were made to the synthesis program to reflect this and the weight inputs were adjusted to allow for the bulkhead change and elimination of the intertank section. The results are shown in Table 4-4.

Table 4-4. FR-3 Booster Element Data, Common versus Separate Bulkheads

	Separate Bulkheads		Common Bulkhead
GLOW, lb	4,329,000	ΔGLOW	4,214,660
	4,214,660	= 114,340	
V_{Stage}, fps	10,912		11,011
q_{max}, lb/ft^2	670.2		725
W_{Return} (Booster), lb	590,437		566,598
W_{Dry} (Booster + Orbiter), lb	683,018	ΔW Deg	661,414
	661,414	= 21,604	
$Length_{Boost}$, ft	210.1		198.38
$Width_{Boost}$, ft	41.13		40.77
$Height_{Boost}$, ft	36.92		36.60
$Length_{Orb}$, ft	179.2		178.03
$Width_{Orb}$, ft	30.91		30.71
$Height_{Orb}$, ft	26.43		26.26
No Engines	15-3		15-3

The separate bulkhead installation costs 114,000 lbs in gross liftoff weight, 21,600 lbs in dry weight and an increased length of 11-1/2 feet in the booster. The booster flyback weight increases to the point where there is little margin left in the four projected RB211-56 flyback engines. However, there is still a small margin, and bearing in mind the 10% contingency already in the system, a return weight of 590,000 pounds is still feasible. The simplified propellant line subsystem with the separate bulkhead (LO_2 lines do not penetrate the LH_2 tank) and the easier inspection potential of the separate bulkhead vehicle led to the adoption of separate bulkhead tanks in the design point FR-3 booster. Figure 4-10 compares the separate and common bulkhead FR-3 boosters.

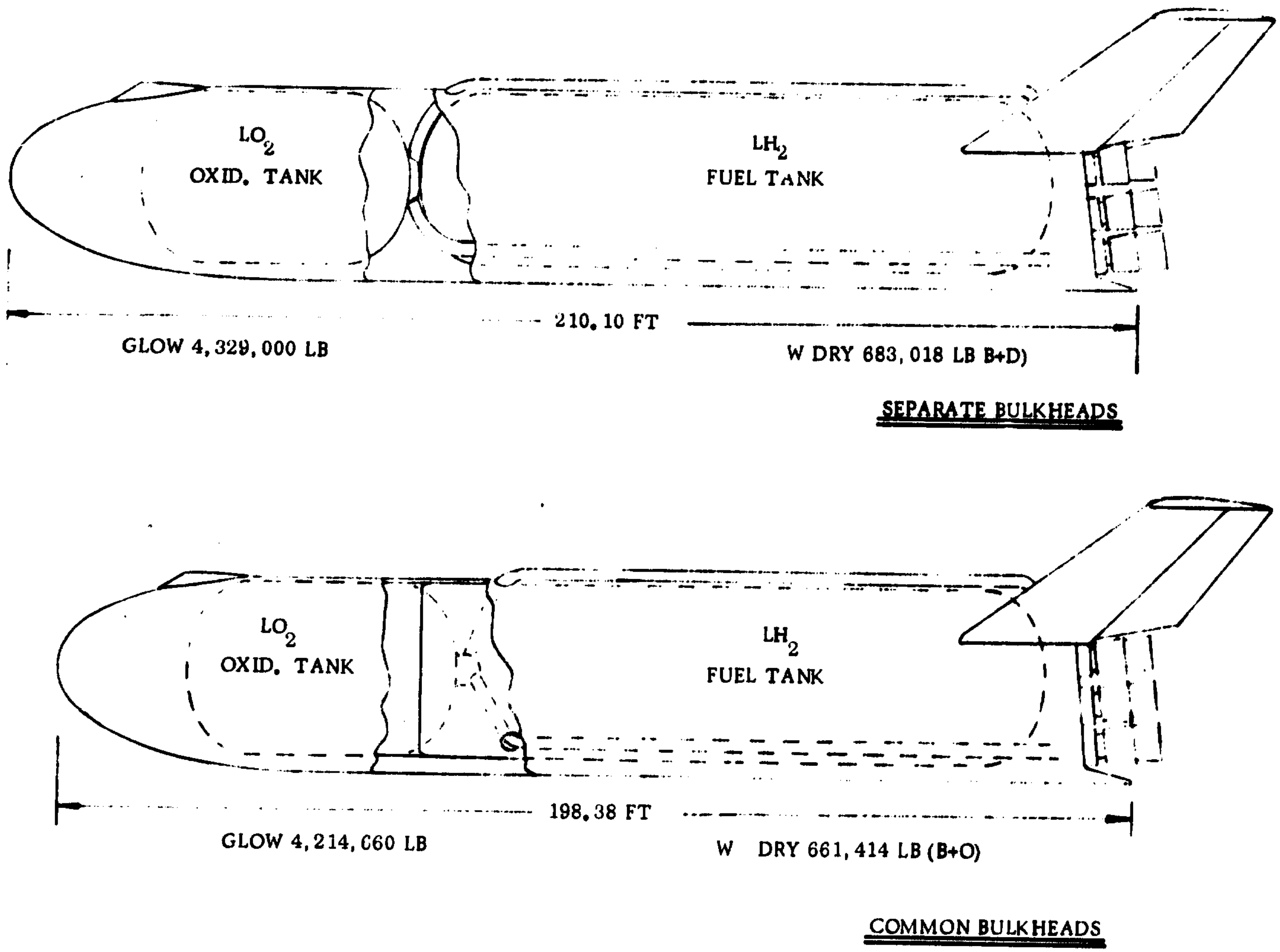


Figure 4-10. FR-3 Comparison of Separate versus Common Bulkheads

4.1.2 THE FR-3 TWO-STAGE SEQUENTIAL BURN 22-FOOT DIAMETER PAYLOAD BAY SYSTEM. This vehicle was treated as a perturbation of the baseline 15-ft-diameter by 60-ft-long payload bay system, since insufficient backup work existed to make it the baseline case. The orbiter vehicle was modified to incorporate the 22-ft-diameter by 30-ft-long bay superimposed upon the 15-ft-diameter by 60-ft-long bay. It was not possible to enclose the 22-ft-diameter within the cross-section of the basic orbiter lines (see Figure 3-1) and for this reason a new set of lines was generated to accommodate the payload bay for the size of orbiter anticipated. The cross-section was very similar to an early 22-ft-diameter payload bay vehicle with 50,000 pounds of payload. The major changes from the baseline orbiter apart from the new lines were:

a. The payload bay was increased to reflect the new diameter over 30 ft of the total 60-ft length.

b. No main propulsion LH_2 was stored beneath the payload bay, there being only space enough for the long-term storage propellants.

c. The larger payload bay door was reflected in increased weights for longerons and load carrying latches.

d. A three-engine orbiter was maintained with 400,000-lb sea level (472,000 lbs vacuum) thrust engines, but it was necessary to add another booster engine to give a comparable thrust-to-weight ratio at liftoff to the baseline FR-3. A 16-3 engine arrangement results.

The vehicle system was synthesized and the results are shown in the layout of Figure 4-11 and the synthesis summary of Table 4-5. The gross liftoff weight is 4.628 million lbs or about 300,000 lbs more than the baseline FR-3. The increase would be greater if the baseline orbiter had been used with its 12-deg side slope in the cross-section. The new cross-section has a side slope of approximately 10 deg giving the same orbiter element entry planform loading as that of the baseline orbiter of Section 4.1.1.

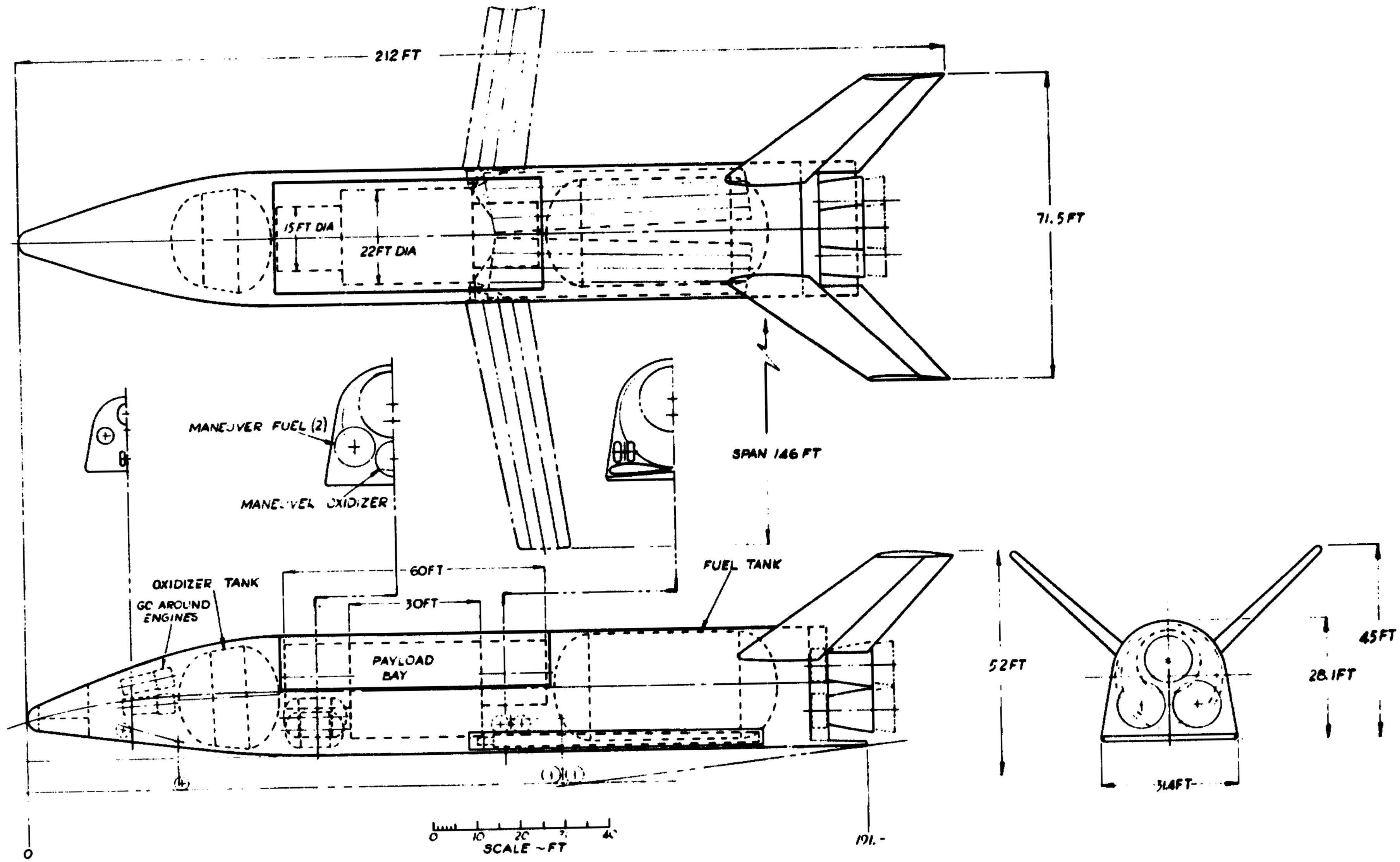


Figure 4-11. FR-3 22-Ft Dia Payload Bay Orbiter

Table 4-5. FR-3 22-Ft Dia Payload System Synthesis Summary

6-3 ENG,400K SL

	BOOSTER ELEMENT	ORBITER	VEHICLE
WEIGHT			
PROPELLANT, ASCENT	2951244	694423	
PROPELLANT, ORBIT MANEUVER		41148	
PROPELLANT, TOTAL	2951244	735572	
FLYBACK FUEL	47035	3164	
PAYLOAD		50000	
STRUCTURE	489350	240111	729461
CONTINGENCY	48935	24011	
OTHER	28482	10373	
TOTAL	3565047	1063231	4628278
IN ORBIT		374467	
RETURN CONDITION	613803	364368	
ENTRY	0	319598	
LANDING	538892	316434	
VOLUME			
FUEL	97569	22916	
OXIDIZER	37947	8933	
PROPELLANT	135516	31849	
PAYLOAD		16720	
OTHER	111832	62572	
TOTAL	247347	111141	
GEOMETRY			
LENGTH	213.4	191.8	
BODY WETTED AREA	27459.3	17002.9	
BODY PLANFORM AREA	8429.9	5416.1	
ENTRY PLANFORM LOADING	69.5	59.6	
PROPULSION			
THRUST-TO-WEIGHT		1.33066	1.38277
NO. OF ENGINES	16	3	
SL THRUST/ENG NOM/UR	399994/ 399994	238369/	6399899/ 6399899
VAC THRUST/ENG NOM/UR	461847/ 461847	471600/	7389557/ 7389557
SL ISP NOM/UR	389.3/389.3	232.0/	389.3/389.3
VAC ISP NOM/UR	449.5/449.5	459.0/	449.5/449.5
TRAJECTORY			
MASS RATIO	2.75986	2.84192	
MAXIMUM DYNAMIC PRESSURE			669.5
STAGING DYNAMIC PRESSURE			50
STAGING VELOCITY (RELATIVE)			10447
STAGING ALTITUDE			185103
STAGING FLIGHT PATH ANGLE (RELATIVE)			2.872
INJECTION VELOCITY (INERTIAL)		25897	
INJECTION ALTITUDE		260001	
INJECTION FLIGHT PATH ANGLE (INERTIAL)		-.000	
INJECTION INCLINATION		54.85	
FLYBACK RANGE	270.8		

4.1.3 THE FR-4 TWO-STAGE SEQUENTIAL-BURN 15-FT DIAMETER BY 60-FT LONG PAYLOAD BAY SYSTEM. The system was derived from previous FR-1 studies. The orbiter layout is described in datail in Section 3. The booster element of the FR-4 system is similar in shape to the orbiter except that it is larger in order to increase the staging velocity and to minimize overall system weight. The booster elements have common bulkheads in the FR-4 concept to try to maximize propellant capacity. Synthesis runs were made at various volume differences between the booster and orbiter elements, corresponding to different staging velocities. The results are shown in Figure 4-12 where the engine numbers were also varied.

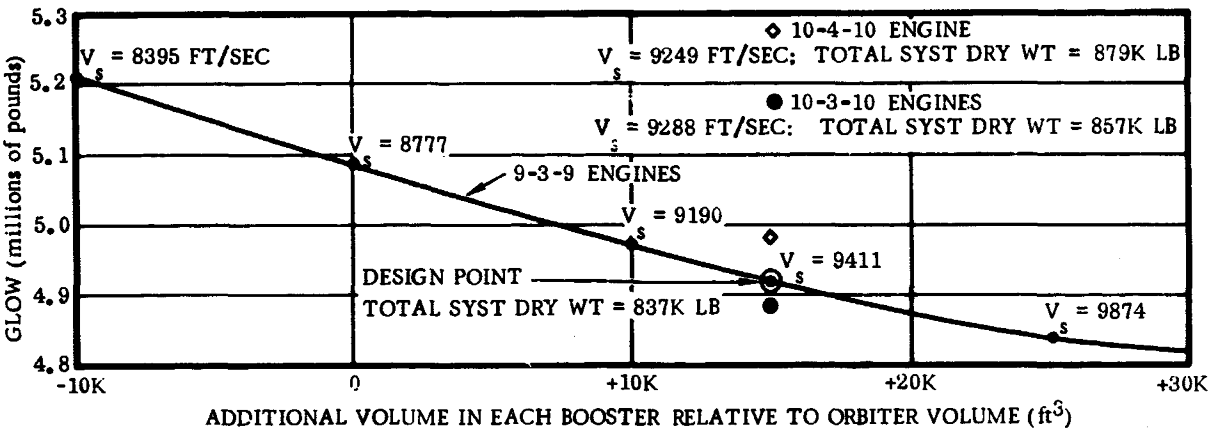


Figure 4-12. FR-4 Gross Lift Off Weight Versus Volume Difference Between Elements. 400K SL Thrust Engines

As shown in the figure a 9-3-9 engine arrangement (booster-orbiter-booster) was selected. The 10-3-10 arrangement had a lower gross weight but a higher total dry weight. The 10-4-10 arrangement was higher in weight in both respects. Three engines in the orbiter will just suffice to provide a once-around abort requirement of 0.80 with one engine out. With all three engines in the point design producing 472,000 lbs of thrust each, the F/W at staging is 1.22. The actual selection of additional volume in each of the booster elements was +15,000 cu ft each, corresponding to a staging velocity of 9411 fps. This is not at the minimum point of the curve as shown, there being some further reduction in proceeding to higher staging velocities. However, with increasing staging velocity and reduced orbiter size the lost volume geometric parameters in the program are optimistic to the right of the design point on Figure 4-12. The entire curve can be expected to rotate upward at the higher staging velocities since packaging of the propellant in the smaller orbiter, with its fixed payload bay size, is less efficient than indicated. The design point has been checked out by layout, and the volume distributions are correct.

The orbiter at the design point is shown on Figure 4-13 in a three view drawing. The booster three-view is shown in Figure 4-14 and the launch arrangement for full access

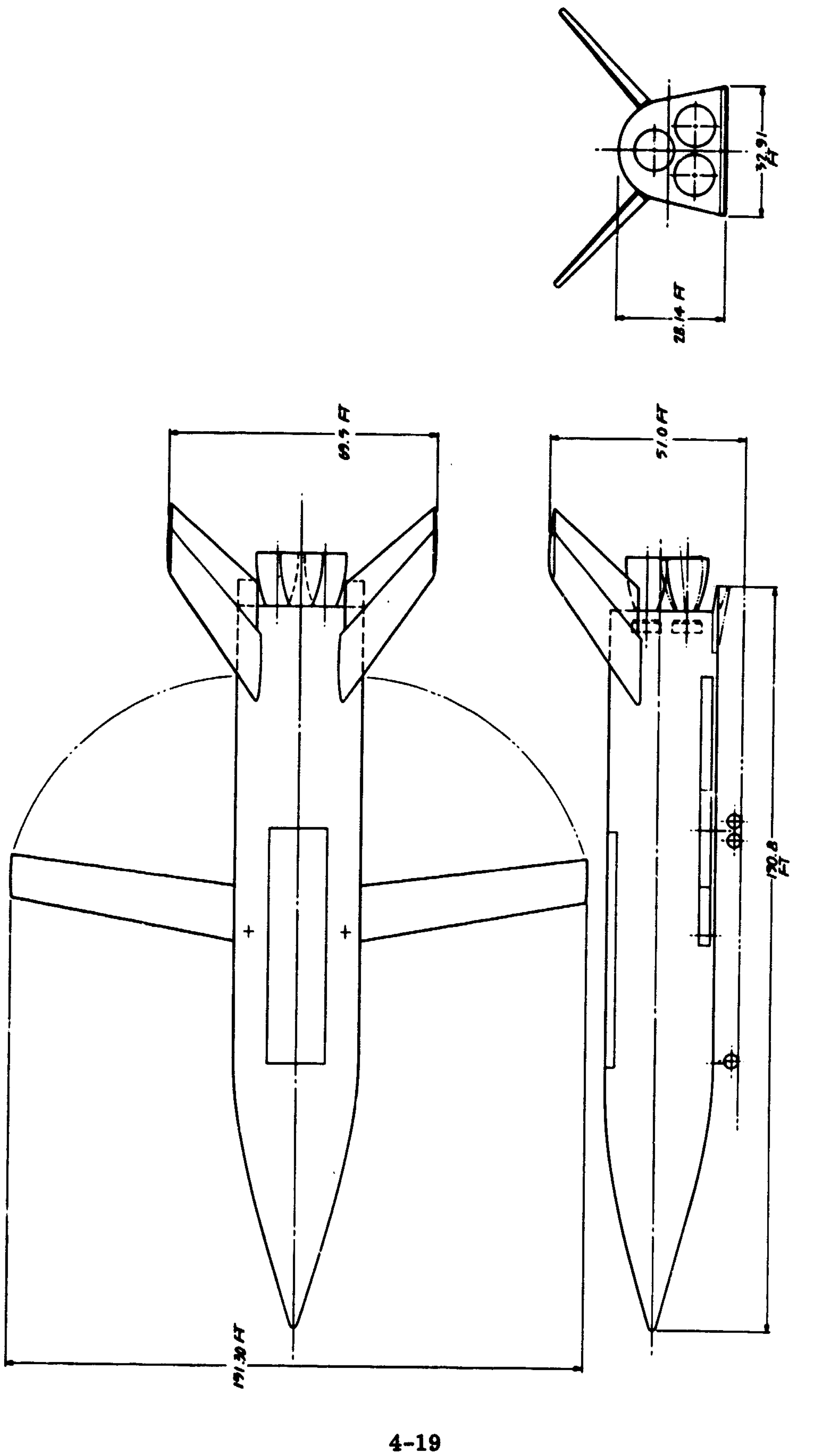


Figure 4-13. FR-4 Orbiter Configuration

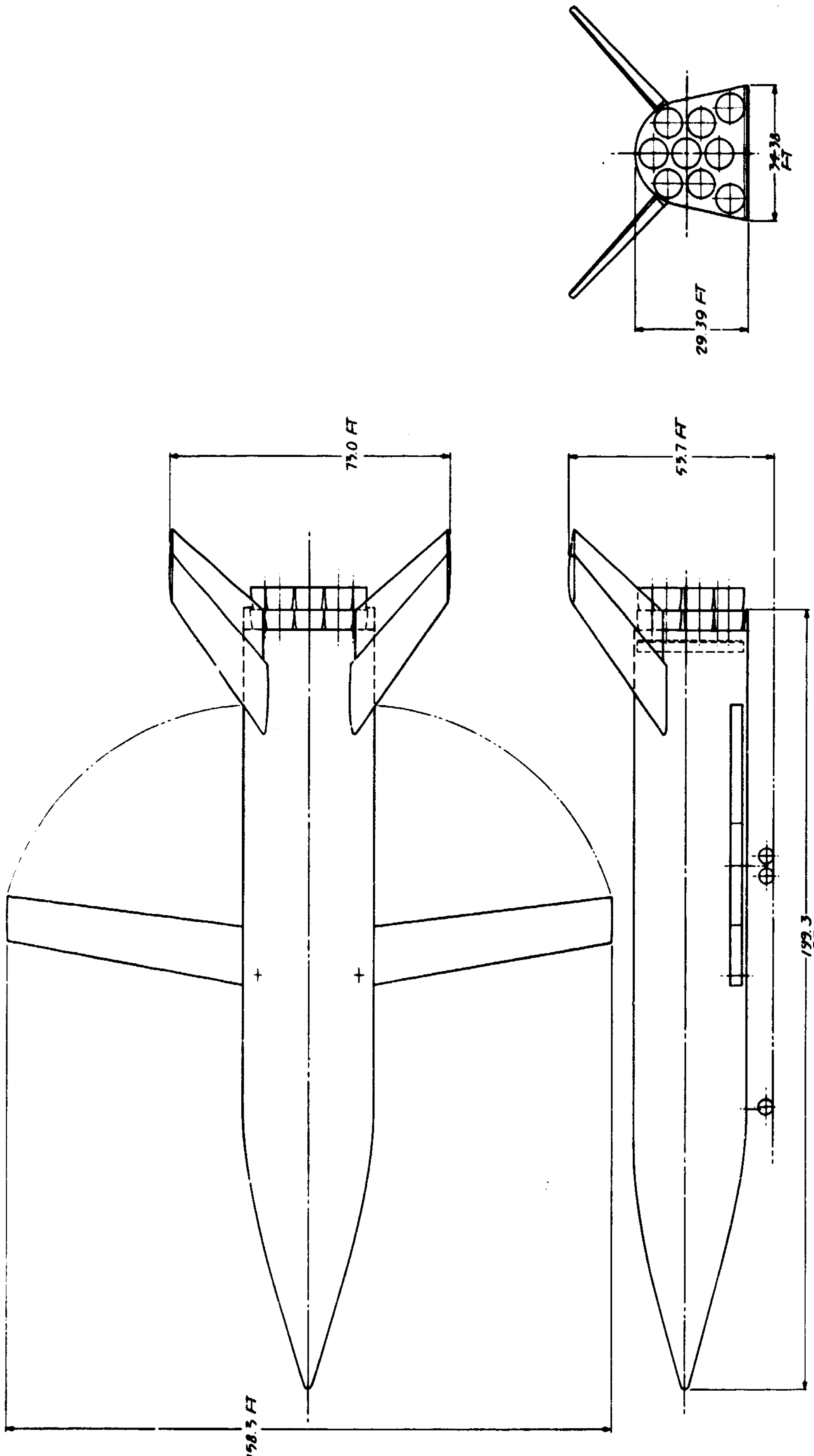


Figure 4-14. FR-4 Booster Configuration

to the payload bay is shown in Figure 4-15. The overall synthesis summary for the FR-4 is given in Table 4-6. The gross liftoff weight is 4.919 million lbs.

Figure 4-16 shows the profile arrangement of the FR-4 booster and orbiter elements.

Apart from the shape, there is little internal commonality between the booster and orbiter elements. The boosters have a two-pilot crew compartment with reduced displays and a simplified environmental control subsystems reflecting the short flight time and suborbital mission of the booster element. Three RB211 flyback engines are required in the booster for the current flyback weight of 356,000 lbs per element. The integral tankage is shown with a common bulkhead. It is necessary to run three LO_2 lines through insulated tunnels in the forward end of the LH_2 tank. These passages would route the LO_2 lines external to the LH_2 tank in as short a distance as possible. The booster element tanks are made of aluminum alloy with a nickel base super alloy lower heat shield and much reduced insulation below the stowable wing, and an upper and side surface fairing/heat shield of 811 titanium alloy with only a minimum of insulation required. Internal insulation is used in the LH_2 tank. Nine 400,000-lb sea thrust bell nozzle engines are required. The expansion ratio is 35/80 for the two-position nozzles.

The orbiter has three similar engines except that an expansion ratio of 160 is required. The basic nozzle is compromised for commonality of engine design to the expansion ratio 35 point. The orbiter engines only need be two position if entry protection requires retraction of the nozzle extensions. Otherwise, it can consist of a fixed nozzle extension from $\epsilon = 35$ to $\epsilon = 160$. The orbiter profile and secondary tankage arrangement is very similar to that previously described. Further dimensional data for the FR-4 system are given in a summary comparison of all vehicles at the end of this section.

4.1.4 THE FR-4 TWO-STAGE SEQUENTIAL-BURN 22-FT DIAMETER PAYLOAD BAY SYSTEM. This vehicle represents an excursion from the baseline. Because the orbiter element is sufficiently large, the basic orbiter lines of Figure 3-1 were used since the 22-ft-diameter payload could be accommodated within the developed shape. This does not give a direct comparison with the FR-3 orbiter, which required new lines due to its smaller size, but it does compare more directly with the baseline FR-4.

Excursions were made to determine the element trends for size and numbers of engines. The fixed-thrust engines, of course, complicate the Phase A analyses by introducing step functions into the data as engine increments change. To avoid this, the ground was covered initially in the synthesis program by varying F/W at liftoff and the delta volume in each booster element relative to the orbiter. The results are plotted in Figure 4-17 for 10-4-10 and 10-3-10 engine arrangements showing weights against Δ volume. Figure 4-18 shows the weights versus F/W at liftoff. The final design point

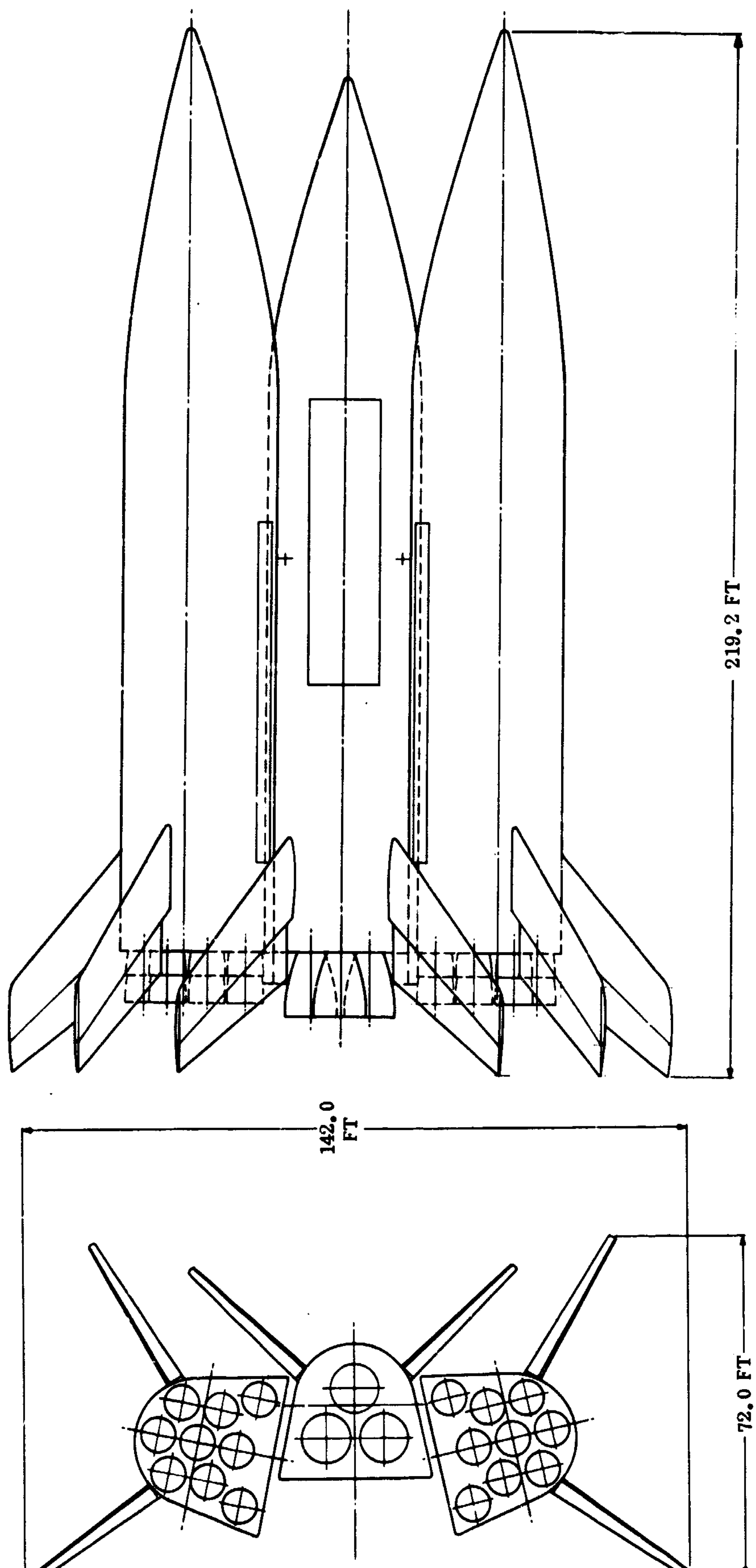


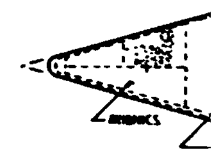
Figure 4-15. FR-4 Launch Configuration

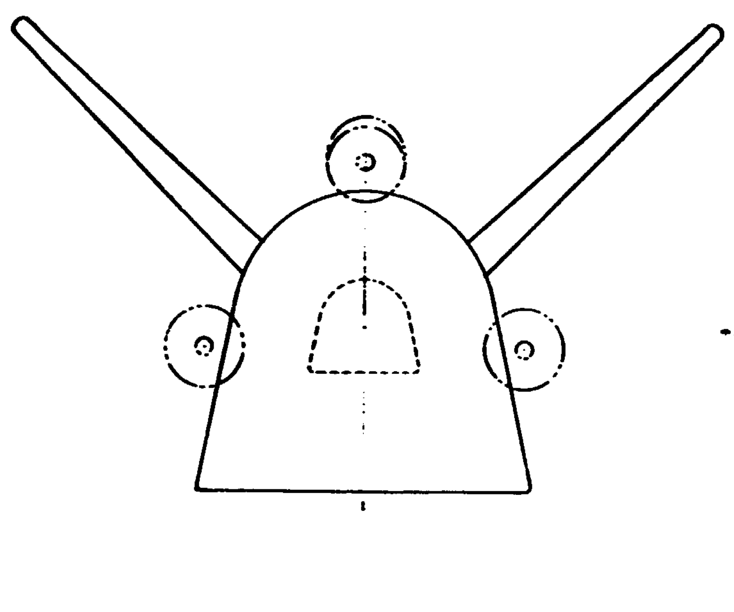
Table 4-6. FR-4 15-Foot-Dia Payload Baseline System Synthesis Summary

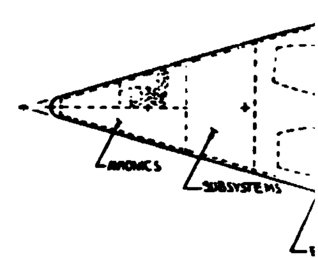
10/13/69

NASA FR-4 50K P/L, 9-3-9 ENG TWLO = 1.465, DELTA VOL = +15K

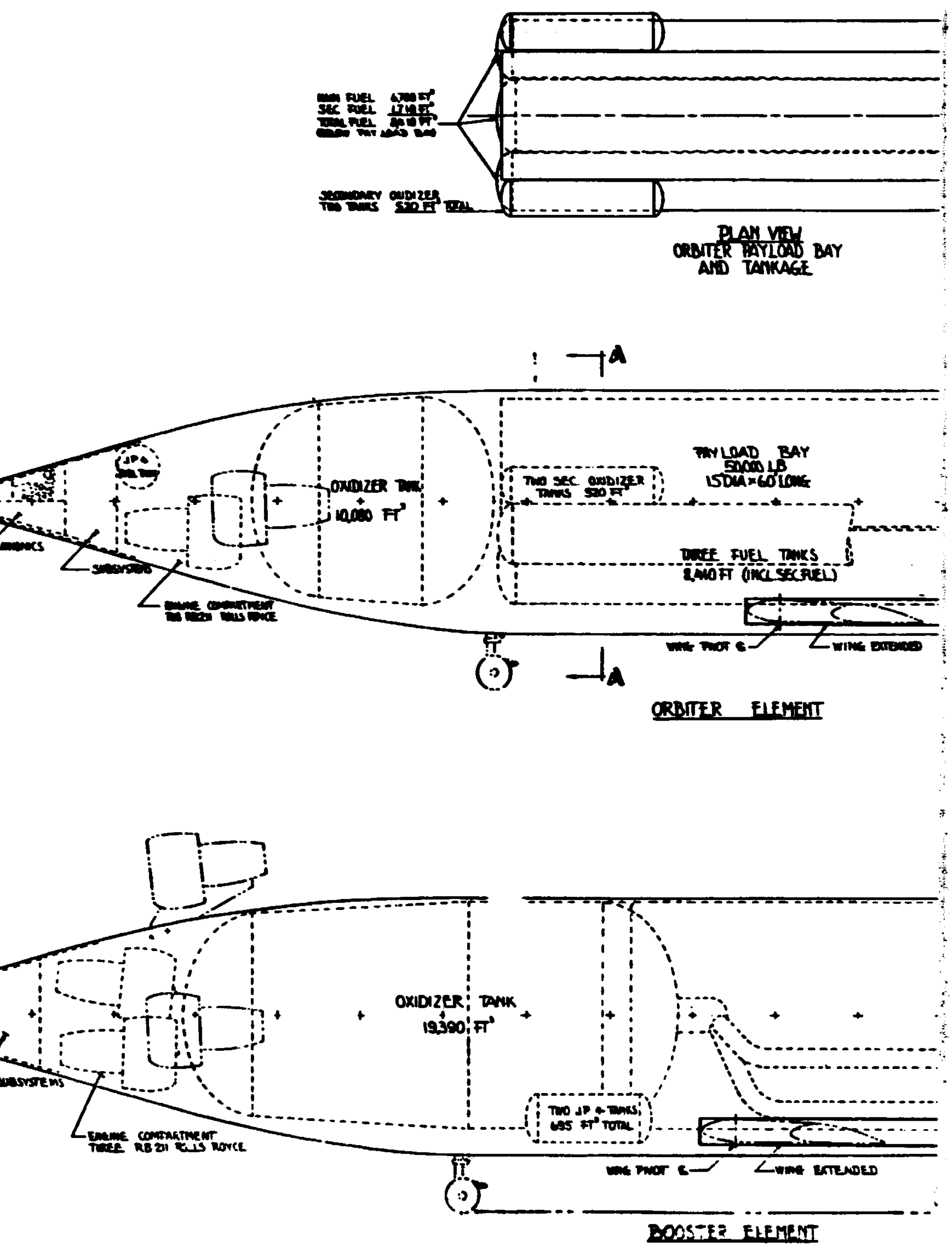
	BOOSTER ELEMENT	ORBITER	VEHICLE
WEIGHT			
PROPELLANT, ASCENT	1508915	783234	
PROPELLANT, ORBIT MANEUVER		42117	
PROPELLANT, TOTAL	1508915	825351	
FLYBACK FUEL	30711	3225	
PAYLOAD		50000	
STRUCTURE	294954	246935	836843
CONTINGENCY	29495	24693	
OTHER	14973	10792	
TOTAL	1879048	1160996	4919092
IN ORBIT		383560	
RETURN CONDITION	370133	372950	
ENTRY	355767	327176	
LANDING	324789	322475	
VOLUME			
FUEL	49899	19143	
OXIDIZER	19407	10075	
PROPELLANT	69306	29219	
PAYLOAD		10638	
OTHER	53179	67596	
TOTAL	122485	107452	
GEOMETRY			
LENGTH	199.3	190.8	
BODY WETTED AREA	18432.9	16892.0	
BODY PLANFORM AREA	6072.2	5564.6	
ENTRY PLANFORM LOADING	58.6	58.8	
PROPULSION			
THRUST-TO-WEIGHT		1.21951	1.46470
NO. OF ENGINES	9	3	
SL THRUST/ENG NOM/UR	400280/ 400280	238544/	7205049/ 7205049
VAC THRUST/ENG NOM/UR	462178/ 462178	471948/	8319212/ 8319212
SL ISP NOM/UR	389.3/389.3	232.0/	389.3/389.3
VAC ISP NOM/UR	449.5/449.5	459.0/	449.5/449.5
TRAJECTORY			
MASS RATIO	2.58728	3.02659	
MAXIMUM DYNAMIC PRESSURE			657.8
STAGING DYNAMIC PRESSURE			50
STAGING VELOCITY (RELATIVE)			9411
STAGING ALTITUDE			179381
STAGING FLIGHT PATH ANGLE (RELATIVE)			5.797
INJECTION VELOCITY (INERTIAL)		25897	
INJECTION ALTITUDE		259992	
INJECTION FLIGHT PATH ANGLE (INERTIAL)		.000	
INJECTION INCLINATION		54.95	
FLYBACK RANGE	255.4		







FOLDOUT FRAME 2



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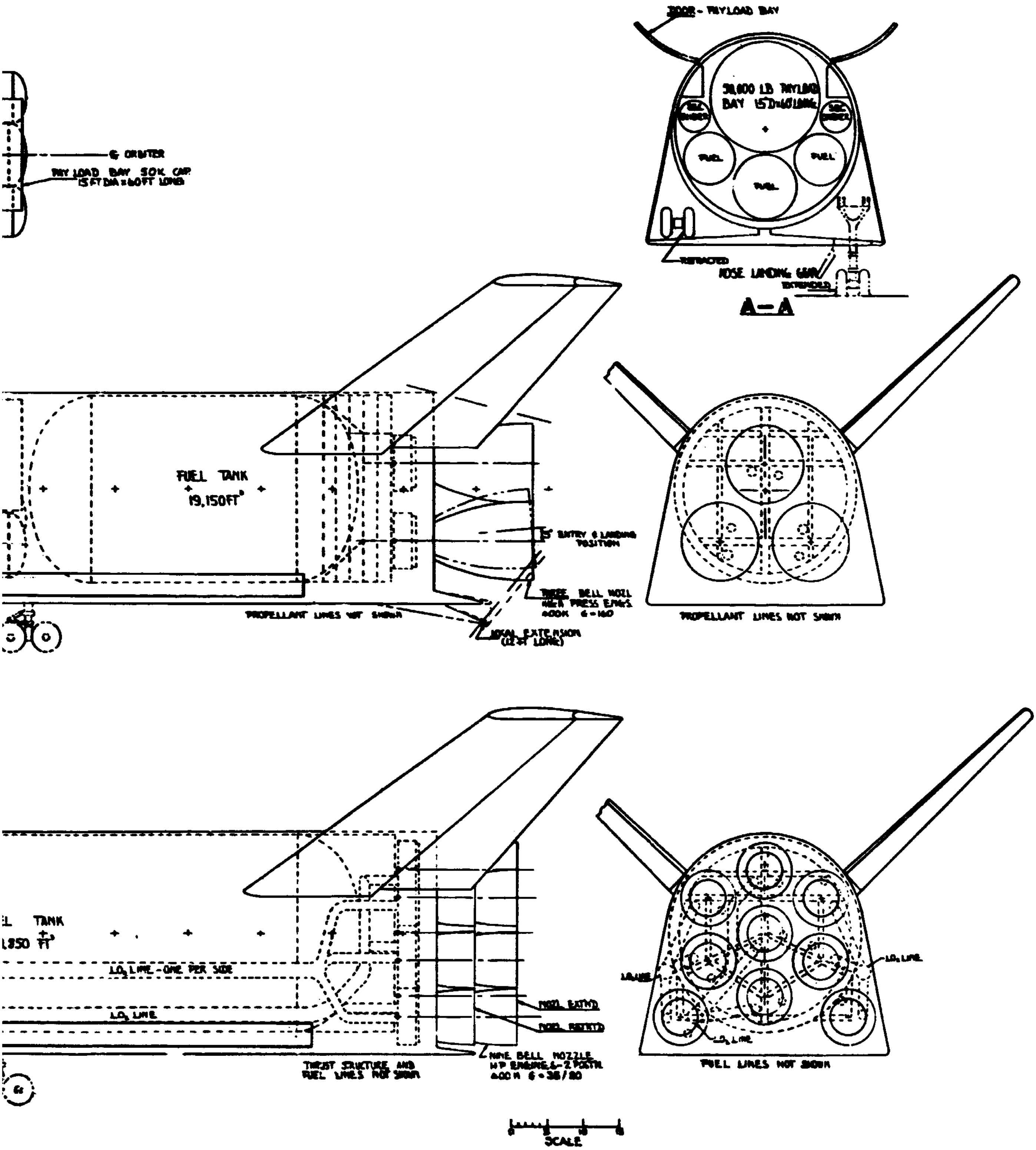


Figure 4-16. FR-4 Element Profiles and General Arrangement

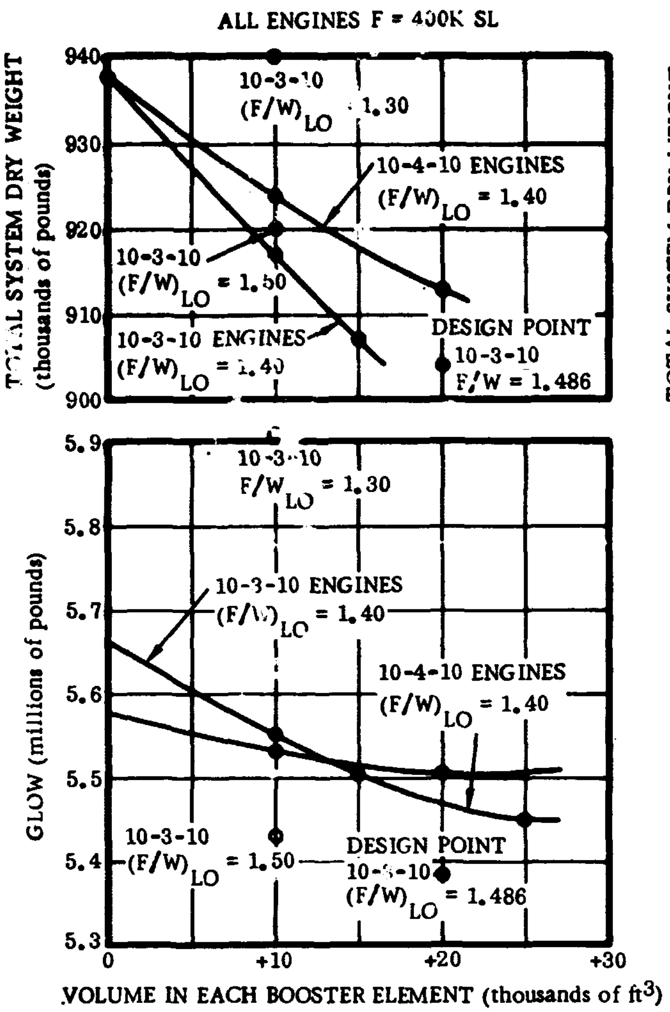


Figure 4-17. FR-4 22 Dia. PL Vehicle Weight Variations vs Δ Volume in Booster Elements

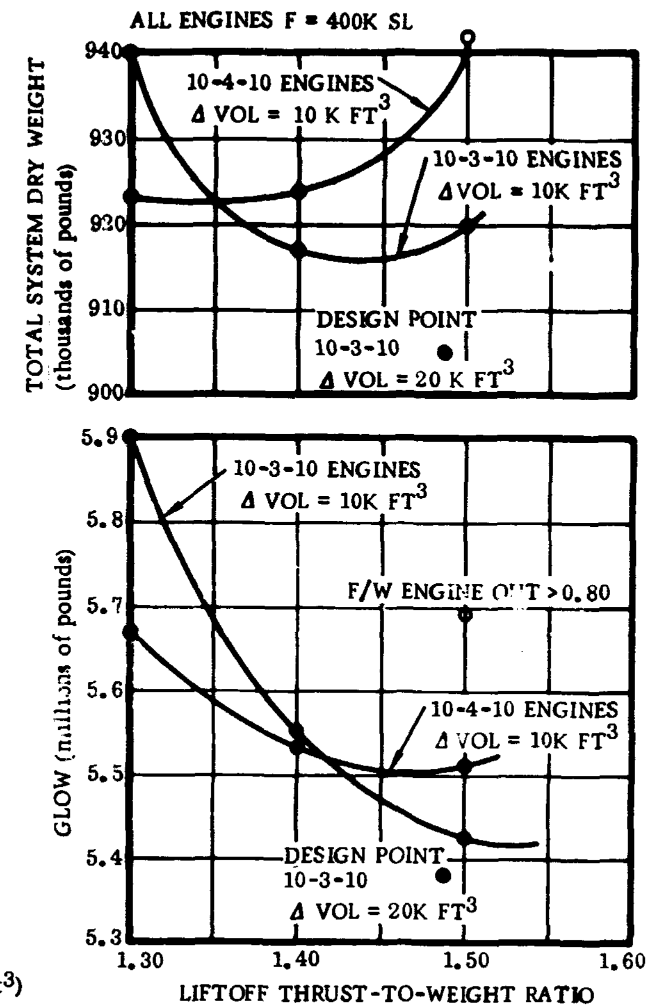


Figure 4-18. FR-4 22-Ft-Dia. Payload Vehicle Weight Variations vs F/W at Liftoff

is shown. A three-engine orbiter, being lighter and having a F/W at staging of 1.25, still has a one-engine-out once-around abort capability. With one engine out, the F/W is about 0.84, which gives a small positive margin over the estimated 0.80 minimum. Note that the Δ volume minimum appears to occur with each booster element 25,000 cubic feet larger than the orbiter. Because of the geometric "lost" volume problem with the smaller orbiter, and because of the minimum cross-section requirements, the design point was selected at Δ volume = 20,000 cu ft. A thrust-to-weight ratio of 1.486 resulted when the 10 fixed-thrust 400,000-lb sea level thrust engines (per each of two boost elements) were included in the point design run. This is near the optimum gross liftoff weight. Further adjustment of delta volume (staging velocity) is possible to match vehicle element sizes with fixed increments of thrust, but this becomes a lengthy procedure where a spectrum of vehicles is involved. The design point staging velocity is 10,400 fps.

Table 4-7 presents the synthesis summary data for the FR-4 22-foot-diameter payload system. The gross liftoff weight is 5.38 million lbs. A difference of 460,000 lbs in gross liftoff weight is the penalty for going to the 22-ft-diameter payload.

The layout of the system is shown in Figure 4-19. Tank and bay volumes are given. Only secondary fuel is stored below the payload bay envelope. The 10-engine arrangement is shown, again using the requirement to slave all engines with failure in the null position as the spacing criteria, with ± 5 degrees gimbal in the Z direction and ± 3 degrees in the Y direction.

A summary of the vehicle dimensions is given in Table 4-8. A summary of the major characteristics is given in Table 4-9. Tables 4-10 and 4-11 present the data on the FR-3 and FR-4, 15 ft diameter × 60 ft payload bay vehicle in the NASA-required format.

Table 4-7. FR-4 22-Ft-Dia Payload System Synthesis Summary

NASA FR-4, 22FT D P/L, 10-3-10 ENG T-1A LINES — 10/16/69 POINT DESIGN, VOLDIF

	BOOSTER ELEMENT	ORBITER	VEHICLE
WEIGHT			
PROPELLANT, ASCENT	1717355	739562	
PROPELLANT, ORBIT MANEUVER		44136	
PROPELLANT, TOTAL	1717355	783698	
FLYBACK FUEL	36399	3381	
PAYLOAD		50000	
STRUCTURE	321408	261137	903954
CONTINGENCY	32141	26114	
OTHER	16897	10678	
TOTAL	2124200	1135007	5383408
IN ORBIT		401516	
RETURN CONDITION	406846	390822	
ENTRY	390555	342954	
LANDING	353889	338097	
VOLUME			
FUEL	56788	24404	
OXIDIZER	22086	9513	
PROPELLANT	78874	33917	
PAYLOAD		16720	
OTHER	60022	68309	
TOTAL	138896	118946	
GEOMETRY			
LENGTH	207.8	197.3	
BODY WETTED AREA	20044.6	18076.1	
BODY PLANFORM AREA	6603.1	5954.6	
ENTRY PLANFORM LOADING	59.1	57.6	
PROPULSION			
THRUST-TO-WEIGHT		1.24651	1.48598
NO. OF ENGINES	10	3	
SL THRUST/ENG NOM/UR	399986/ 399986	238369/	7999710/ 7999710
VAC THRUST/ENG NOM/UR	461838/ 461838	471600/	9236758/ 9236758
SL ISP NOM/UR	389.3/389.3	232.0/	389.3/389.3
VAC ISP NOM/UR	449.5/449.5	459.0/	449.5/449.5
TRAJECTORY			
MASS RATIO	2.76257	2.82814	
MAXIMUM DYNAMIC PRESSURE			691.4
STAGING DYNAMIC PRESSURE			50
STAGING VELOCITY (RELATIVE)			10401
STAGING ALTITUDE			184876
STAGING FLIGHT PATH ANGLE (RELATIVE)			4.035
INJECTION VELOCITY (INERTIAL)		25897	
INJECTION ALTITUDE		260000	
INJECTION FLIGHT PATH ANGLE (INERTIAL)		-.000	
INJECTION INCLINATION		54.82	
FLYBACK RANGE	276.8		

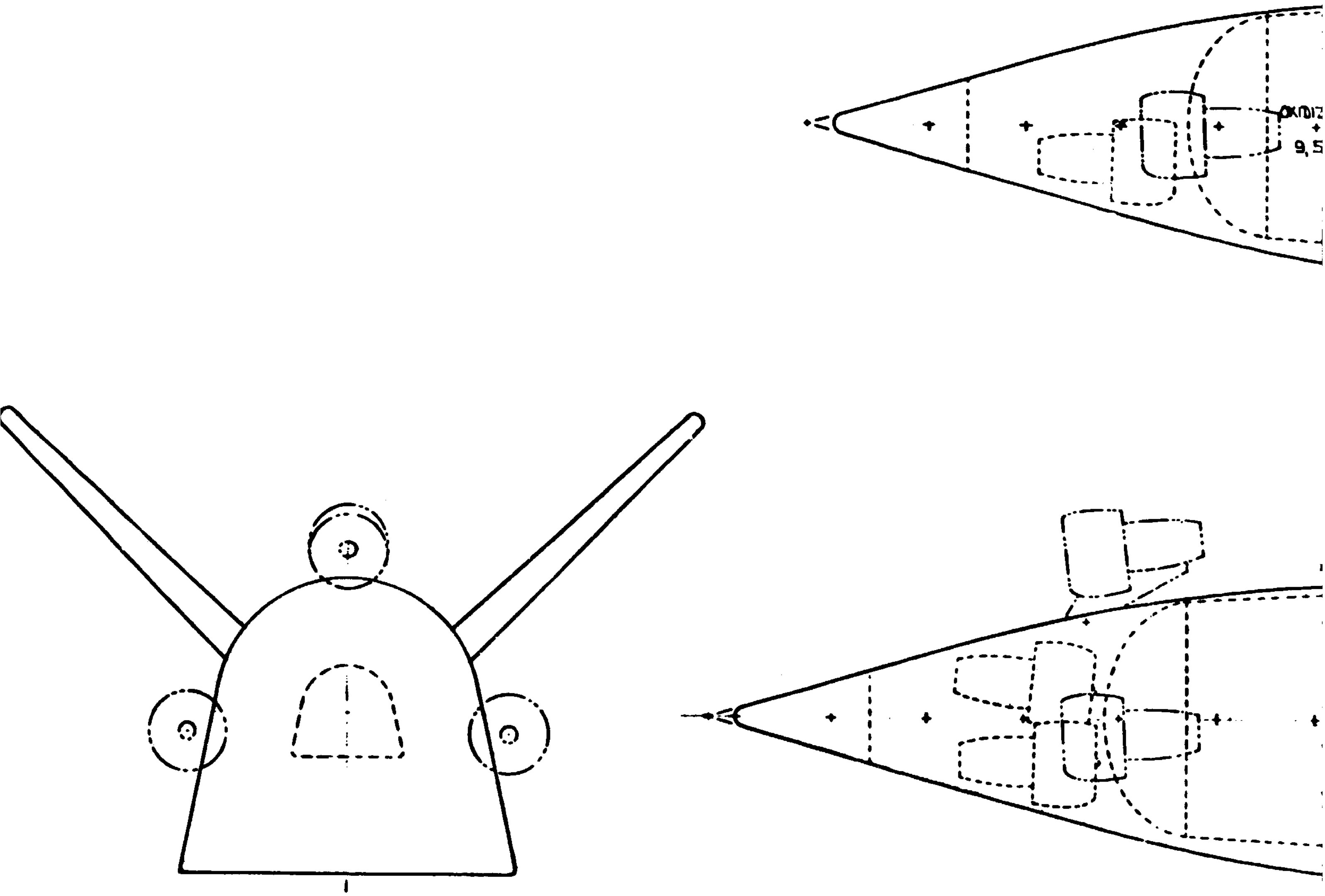
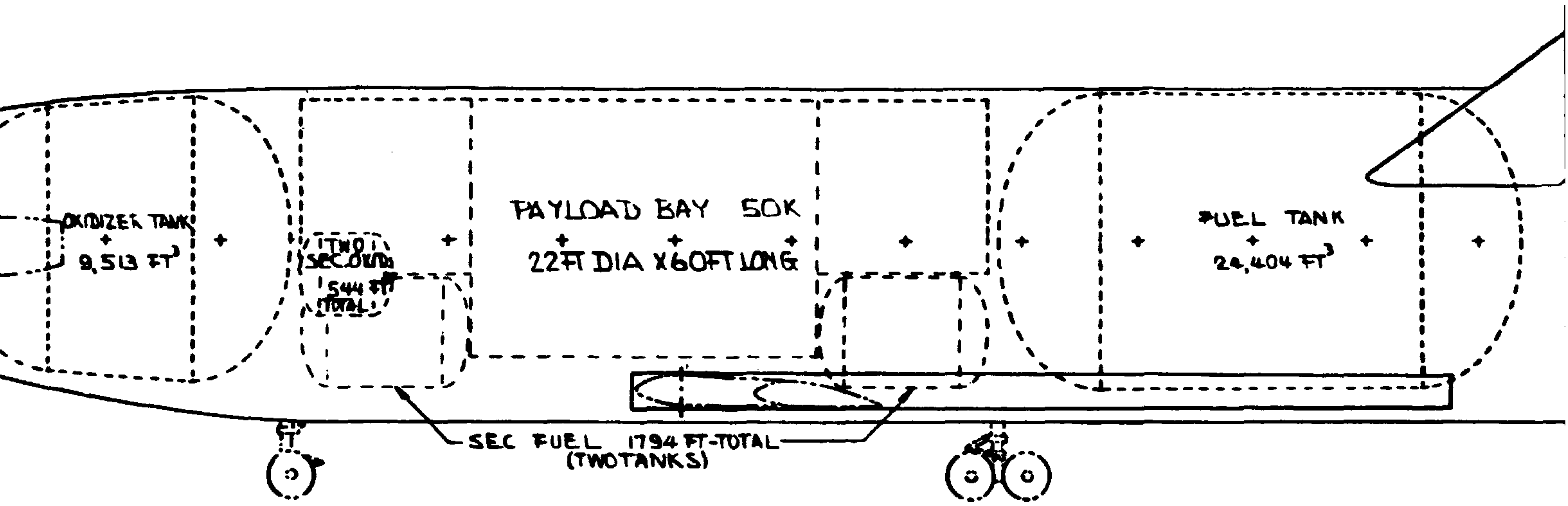
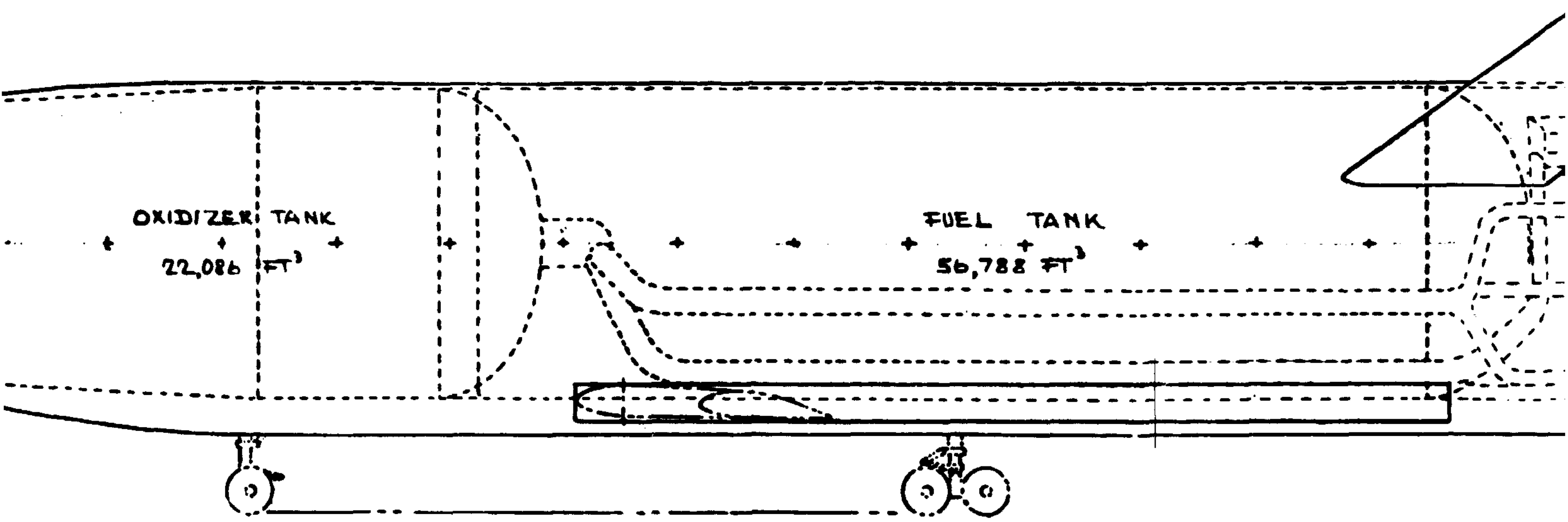


Figure 4-19. FR-4 22-Ft-Dia P/L System Elements FOLDOUT FRAME 1

FOLDOUT FRAME

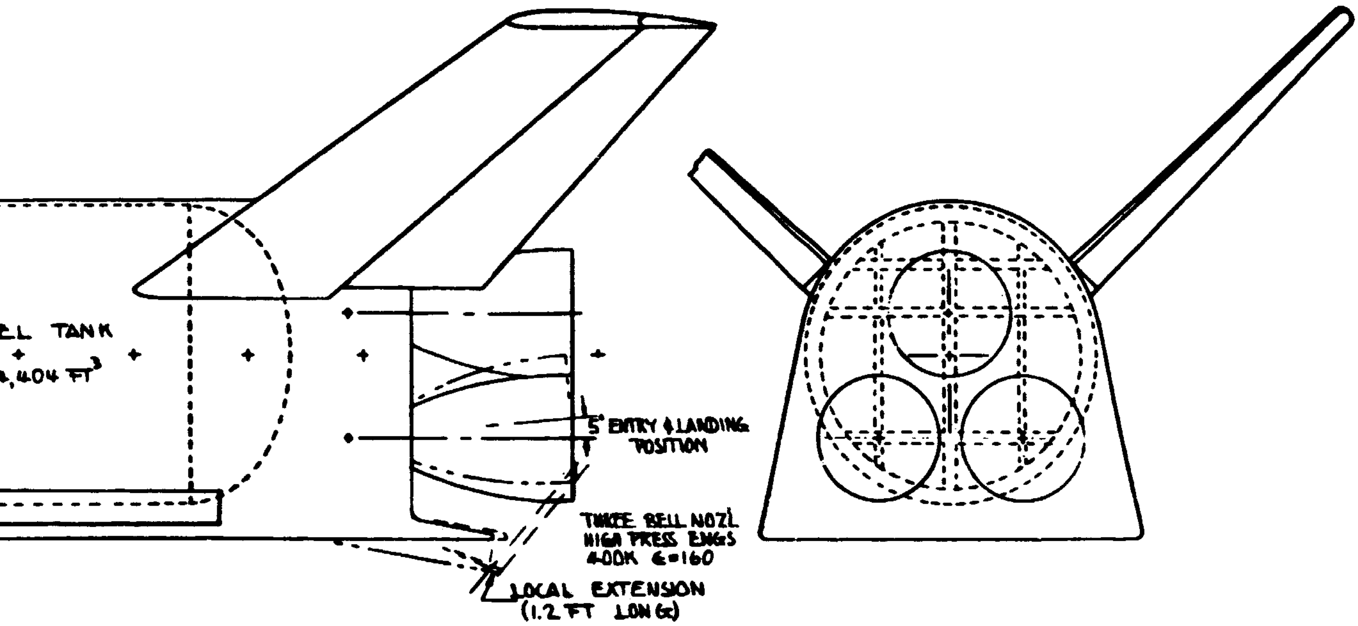


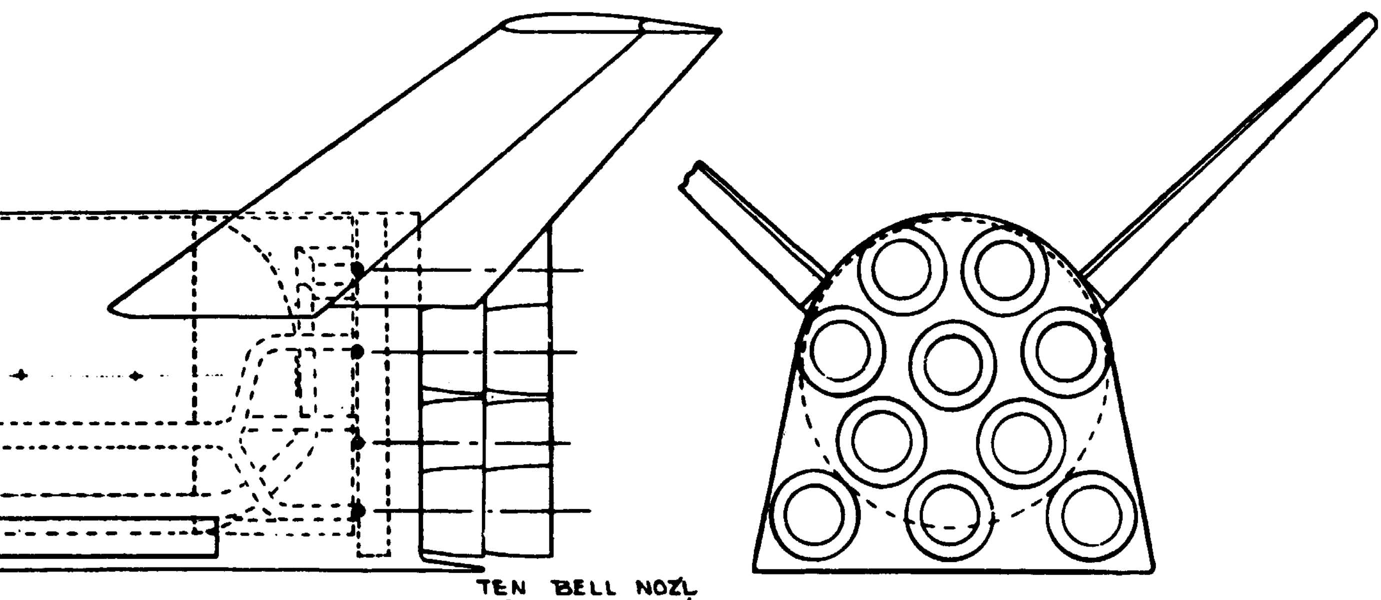
ORBITER ELEMENT
197.4 FT LONG



BOOSTER ELEMENT
207.80 FT LONG

FOLDOUT FRAME 2





FOLDOUT FRAME 3

Table 4-8. Vehicle - Dimensional Data Summary

	FR-3 Vehicle				FR-4 Vehicle			
	15 by 60 Payload Bay		22 by 60 Payload Bay		15 by 60 Payload Bay		22 by 60 Payload Bay	
	Booster	Orbiter	Booster	Orbiter	Booster	Orbiter	Booster	Orbiter
Elements								
Volume (ft^3)	236,000	89,060	247,350	111,140	122,370	107,470	138,900	118,950
Body Wetted Area (ft^2)	26,610	14,900	27,460	17,000	18,420	16,890	20,040	18,080
Body Planform Area (ft^2)	8,170	4,910	8,430	5,420	6,070	5,570	6,600	5,960
Base Area (ft^2)	1,200	632	1,250	703	777	712	885	765
Body Length (L) (ft)	210.13	179.20	213.45	191.82	199.3	190.8	207.80	197.30
Width (ft)	41.13	30.91	41.77	31.35	34.38	32.91	35.85	34.0
Height (ft)	36.92	26.43	37.51	28.07	29.39	28.14	30.65	29.11
Dorsal R (ft)	17.17	12.10	17.42	13.01	12.94	12.37	14.00	13.30
Wing Span (ft)	201.0	146.9	204.0	146.0	158.30	151.30	165.00	156.66
Area (ft^2)	2901.0	1336.0	2993.0	1490.0	1607.0	1514.0	1747.0	1620.0
Sweep L.E. (deg)	20/27	10.0	20/27	10.0	10.0	10.0	10.0	10.0
Empennage Span (ft)	84.3	66.2	85.5	71.5	73.0	69.5	76.1	71.8
Area (ft^2)	2283.0	1397.0	2355.0	1618.0	1728.0	1583.0	1879.0	1694.0
Sweep L.E. (deg)	45.0	45.0	45.0	45.0	45.0	45.0	45.0	45.0
Height (ft)	61.0	47.8	62.0	52.0	53.7	51.0	56.0	52.7
cg Location								
Entry (%)	54.7	57.0	54.7	57.0	60.8	67.0	60.8	57.0
Landing (%)	58.8	57.3	58.8	57.3	64.4	56.7	64.4	56.7
Reference Computer Run	10/20/69		10/20/69		10/14/69		10/16/69	
Launch Configuration								
Overall Length (ft)	235.5		240.0		219.2		228.6	
Width (ft)	84.3		85.7		72.0		75.1	
Height (ft)	97.5		99.1		142.0		148.0	

Table 4-9. Vehicle — General Characteristics Summary

	FR-3 Vehicle				FR-4 Vehicle			
	50 by 60 Payload Bay		22 by 60 Payload Bay		15 by 60 Payload Bay		22 by 60 Payload Bay	
GLOW (million lb)	4.33		4.63		4.92		5.38	
Total Dry Weight (k lb)	683		729		837		904	
Relative Staging Velocity (ft/sec)	10,912		10,447		9,411		10,401	
Altitude (ft)	187,497		185,103		179,381		184,876	
Maximum q (lb/ft^2)	670.2		669.5		657.8		691.4	
Rocket Eng Arrangement	15-3		16-3		9-3-9		10-3-10	
Element	Booster	Orbiter	Booster	Orbiter	Booster	Oribter	Booster	Orbiter
Thrust/Weight	1.386	1.527	1.383	1.331	1.465	1.220	1.486	1.247
Contingency (k lb)	47	21	49	24	29 each	25	32 each	26
Propellant (million lb)	2.81	0.630	2.95	0.736	1.51 each	0.825	1.72 each	0.784
Flyback Fuel (k lb)	46.9	2.87	4.70	3.16	30.7 each	3.23	3.64 each	3.38
Weight Landing (k lb)	517	287	539	316	325	322	354	338
Flyback Eng-R R								
RB211-22	4	—	4	—	3	2	3	2
P&W TF 33-P-3	—	3	—	3	—	—	—	—
ΔV Losses Total ft/sec)	3782	707	3765	841	3862	730	3820	723
Gravity (ft/sec)	3360	151	3362	173	3235	222	3180	182
Drag (ft/sec)	422	158	403	173	627	89	640	123
Misalignment (ft/sec)	—	398	—	495	—	419	—	418
Ideal Velocity	14,690	29,518	14,208	29,633	13,268	29,622	14,217	29,570

Table 4-10. FR-3 Vehicle Summary

SPACECRAFT DESIGN DATA STATEMENT

CONFIGURATION FR-3 ORBITER (15' × 60' P/L BAY)	BY	DATE

REFERENCE MIL-M-38310A, SP-6004, AN-9103-D

SYSTEM/AIRFRAME DESIGN					
1. Aerodynamic Surfaces			Wing	H. Tail	V. Tail
Volume, Wetted, Outer Mold Line (ft^3)			-		-
Surface Area of Above Volume (ft^2)			-		-
Gross Area (ft^2)			1,336		1,397
Span (ft)			146.9		66.2
Mean Aerodynamic Chord Length (ft)			12.2		23.0
Theoretical Root Chord Length (in.)			162		338
Theoretical Root Chord Max. Thickness (in.)			34.1		40.5
Theoretical Tip Chord Length (in.)			129.0		203.0
Theoretical Tip Chord Max. Thickness (in.)			23.2		20.3
2. Body Structure		Cabin	Cargo	Remain	Main Tanks
Volume, Wetted, Mold Line (ft^3)			10,638	89,060	
Pressurized Volume (ft^3)		350			22,665
Surface Area, Wetted, (ft^2)				14,900	
Maximum Length (ft)				179.2	
Maximum Depth (ft)				26.4	
Maximum Width (ft)				30.9	
Total Wetted Outer Mold Line Volume (ft^3)				89,0[illegible]	
Total Wetted Outer Mold Line Surface Area (ft^2)				14.[illegible]	
3. Induced Envir. Prot.	Wing	H. Tail	V. Tail	[illegible]	Fuel Tanks
Volume Delta Within					
1.0 & 2.0 (ft^3)			580	2[illegible]	304
Surface Area Delta					
Within 1.0 & 2.0 (ft^2)			2,[illegible]	14,900	3,657
Windward Unit Wt. (lb/ft^2)			2.43	2.45	
Leeward Unit Wt. (lb/ft^2)			1.90	1.88	
4. Launch Recovery and Docking			Main	Nose	Other
Landing Gear, Max Vert. Load/Gear (lb)			230,000	96,000	
Length, Oleo Extended (in.)*			180	[illegible]30	
Oleo Travel (in.)**			24	24	
Structural				Stress G. W.	Ult. L. F.
Flight (Subsonic Gust)				287,[illegible]	1.4
Landing				287,000	1.4
Limit Landing Sinking Speed (ft/sec)					12
Pressurized Cabin Ult. Dsn. Press. Diff. Flt. (psi)					20
Airframe Weight (as Defined in AN-W-11) (lb)					

SYSTEM DESIGN		
5. Main Propulsion	Liftoff	Landing
Number of Engines	3	3
Max SL Static Thrust per Engine (lb)	400,000	21,000
Total Propellant Tankage Volume (ft^3)	22,665	-
Total JP Fuel Tankage Volume (ft^3)	-	63
6. Through 16.		
Maximum Electrical Power (KVA)		23
Pressurized Surface Area (ft^2)		5,507
Maximum Mission Duration (days)		7
No. of Crew		2
17. Through 21.		
Maximum Number of Personnel Including Crew		2
Maximum Cargo (lb)		50,000
Maximum Cargo Density (lb/ft^3)		4.7
22. Through 27.		
Capacity Propellant Weight (lb)		629,558
Capacity JP Fuel Weight (lb)		2,868
GENERAL/MISSION DESIGN		
Maximum Design Gross Weight (lb)		926,690
Maximum Boost Load Factor 4 g at 926,690 G. W.		
Maximum Boost Dynamic Pressure (lb/ft^2)		670.2
Main Engine Specific Impulse (lb-sec/lb) (Vacuum)		459.0
Delta Velocity Available (ft/sec) Ideal		29,518
Entry Velocity (ft/sec)(Alt = 400,000 ft $\gamma = -1.0°$)		25,970
Entry Weight (lb)		[illegible]9,655
Ballistic Coefficient, W/C_DA (lb/ft^2)		-
Maximum Heating Rate (Btu/ft^2-sec) Stagnation = 58.6 Body L. E. = 46.3, Upper Surf = 0.35, Lower Surf = [illegible]1.3		
Factors of Safety (Define) See Volume II, Section 2		
Margins (Define)		
Contingency (Define) - Uncertainty Allowance		8,097
Weight Growth (Define) - 10% Dry Structure		21,315

*℄ Axle to ℄ Trunnion **Fully Extended to Fully Collapsed

Table 4-10. FR-3 Vehicle Summary, Contd

SPACECRAFT DESIGN DATA STATEMENT

CONFIGURATION FR-3 BOOSTER (15' × 60' P/L BAY)	BY	DATE

REFERENCE MIL-M-38310A, SD-6004, AN-9103-D

SYSTEM/AIRFRAME DESIGN					
1. Aerodynamic Surfaces			Wing	H. Tail	V. Tail
Volume, Wetted, Outer Mold Line (ft^3)			-		-
Surface Area of Above Volume (ft^2)			-		-
Gross Area (ft^2)			2,901		2,283
Span (ft)			201.0		84.3
Mean Aerodynamic Chord Length (ft)			17.3		32.9
Theoretical Root Chord Length (in.)			224		484
Theoretical Root Chord Max. Thickness (in.)			47		58
Theoretical Tip Chord Length (in.)			189		290
Theoretical Tip Chord Max. Thickness (in.)			34		29
2. Body Structure		Cabin	Cargo	Remain	Main Tanks
Volume, Wetted, Mold Line (ft^3)				235,997	
Pressurized Volume (ft^3)		350			129,111
Surface Area, Wetted, (ft^2)				26,613	
Maximum Length (ft)				210.1	
Maximum Depth (ft)				36.9	
Maximum Width (ft)				41.1	
Total Wetted Outer Mold Line Volume (ft^3)				235,997	
Total Wetted Outer Mold Line Surface Area (ft^2)				26,613	
3. Induced Envir. Prot.	Wing	H. Tail	V. Tail	Body	Fuel Tanks
Volume Delta Within					
1.0 & 2.0 (ft^3)				3,320	1,150
Surface Area Delta					
Within 1.0 & 2.0 (ft^2)				26,610	13,790
Windward Unit Wt. (lb/ft^2)				1.92	
Leeward Unit Wt. (lb/ft^2)				1.36	
4. Launch Recovery and Docking			Main	Nose	Other
Landing Gear, Max Vert. Load/Gear (lb)			495,000	72,000	
Length, Oleo Extended (in.)*			200	160	
Oleo Travel (in.)**			24	24	
Structural				Stress G. W.	Ult. L. F.
Flight (Subsonic Gust)				517,000	1.4
Landing				517,000	1.4
Limit Landing Sinking Speed (ft/sec)					12
Pressurized Cabin Ult. Dsn. Press. Diff. Flt. (psi)					20
Airframe Weight (as Defined in AN-W-11) (lb)					

SYSTEM DESIGN		
5. Main Propulsion	Liftoff	Landing
Number of Engines	15	4
Max SL Static Thrust per Engine (lb)	400,000	52,500
Total Propellant Tankage Volume (ft^3)	129,111	-
Total JP Fuel Tankage Volume (ft^3)	-	1,027
6. Through 16.		
Maximum Electrical Power (KVA)		23
Pressurized Surface Area (ft^2)		19,260
Maximum Mission Duration (days)		7
No. of Crew		2
17. Through 21.		
Maximum Number of Personnel Including Crew		2
Maximum Cargo (lb)		Orbiter
Maximum Cargo Density (lb/ft^3)		-
22. Through 27.		
Capacity Propellant Weight (lb)		2,811,879
Capacity JP Fuel Weight (lb)		46,916
GENERAL/MISSION DESIGN		
Maximum Design Gross Weight (lb)		3,402,316
Maximum Boost Load Factor 4 g at ______ G. W.		
Maximum Boost Dynamic Pressure (lb/ft^2)		670.2
Main Engine Specific Impulse (lb-sec/lb) Uprated = 449.5		389.3
Delta Velocity Available (ft/sec) Ideal		14,690
Entry Velocity (ft/sec)		10,912
Entry Weight (lb)		590,437
Ballistic Coefficient, W/C_DA (lb/ft^2)		-
Maximum Heating Rate (Btu/ft^2-sec) Stagnation = 6.9		
Lower Surf = 9.2, Upper Surf = 0.8		
Factors of Safety (Define) See Volume II, Section 2		
Margins (Define)		
Contingency (Define) - Uncertainty Allowance		19,487
Weight Growth (Define) - 10% Dry Structure		46,987

*℄ Axle to ℄ Trunnion **Fully Extended to Fully Collapsed

Table 4-11. FR-4 Vehicle Summary

SPACECRAFT DESIGN DATA STATEMENT

CONFIGURATION FR-4 ORBITER (15' × 60' P/L BAY)	BY	DATE

REFERENCE MIL-M-38310A, SP-6004, AN-9103-D

SYSTEM/AIRFRAME DESIGN

1. Aerodynamic Surfaces	Wing	H. Tail	V. Tail
Volume, Wetted, Outer Mold Line (ft^3)	-		-
Surface Area of Above Volume (ft^2)	-		-
Gross Area (ft^2)	1,514.0		1,583.0
Span (ft)	151.3		69.5
Mean Aerodynamic Chord Length (ft)	13.0		-
Theoretical Root Chord Length (in.)	172.2		359.0
Theoretical Root Chord Max. Thickness (in.)	36.2		43.1
Theoretical Tip Chord Length (in.)	137.3		215.5
Theoretical Tip Chord Max. Thickness (in.)	24.7		21.5

2. Body Structure	Cabin	Cargo	Remain	Main Tanks
Volume, Wetted, Mold Line (ft^3)		10,638	107,470	
Pressurized Volume (ft^3)	350			29,219
Surface Area, Wetted, (ft^2)			16,890	
Maximum Length (ft)			190.8	
Maximum Depth (ft)			28.1	
Maximum Width (ft)			32.9	
Total Wetted Outer Mold Line Volume (ft^3)			107,470	
Total Wetted Outer Mold Line Surface Area (ft^2)			16,890	

3. Induced Envir. Prot.	Wing	H. Tail	V. Tail	Body	Fuel Tanks
Volume Delta Within					
1.0 & 2.0 (ft^3)			660	3,280	358
Surface Area Delta				16,890	
Within 1.0 & 2.0 (ft^2)			3,180		4,294
Windward Unit Wt. (lb/ft^2)			2.4	2.78	
Leeward Unit Wt. (lb/ft^2)			1.88	2.13	

4. Launch Recovery and Docking	Main	Nose	Other
Landing Gear, Max Vert. Load/Gear (lb)	210,000	90,000	
Length, Oleo Extended (in.)*	180	180	
Oleo Travel (in.)**	30	36	
Structural		Stress G. W.	Ult. L. F.
Flight (Subsonic Gust)		322,000	1.4
Landing		322,000	1.4
Limit Landing Sinking Speed (ft/sec)			12
Pressurized Cabin Ult. Dsn. Press. Diff. Flt. (psi)			20
Airframe Weight (as Defined in AN-W-11) (lb)			

SYSTEM DESIGN

5. Main Propulsion	Liftoff	Landing
Number of Engines	3	2
Max SL Static Thrust per Engine (lb)	400,000	40,600
Total Propellant Tankage Volume (ft^3)	29,219	-
Total JP Fuel Tankage Volume (ft^3)	-	73

6. Through 16.	
Maximum Electrical Power (KVA)	23
Pressurized Surface Area (ft^2)	6,537
Maximum Mission Duration (days)	7
No. of Crew	2
17. Through 21.	
Maximum Number of Personnel Including Crew	2
Maximum Cargo (lb)	50,000
Maximum Cargo Density (lb/ft^3)	4.7
22. Through 27.	
Capacity Propellant Weight (lb)	825,351
Capacity JP Fuel Weight (lb)	3,225
GENERAL/MISSION DESIGN	
Maximum Design Gross Weight (lb)	1,160,996
Maximum Boost Load Factor 4 g at 1,160,996 G. W.	
Maximum Boost Dynamic Pressure (lb/ft^2)	657.8
Main Engine Specific Impulse (lb-sec/lb) (Vacuum)	459.0
Delta Velocity Available (ft/sec) Ideal	29,622
Entry Velocity (ft/sec) Alt = 400,000 ft $\gamma = -1.0°$	25,970
Entry Weight (lb)	327,176
Ballistic Coefficient, W/C_DA (lb/ft^2)	-
Maximum Heating Rate (Btu/ft^2-sec) Stagnation = 58.6	
Lower Surf = 11.3, Upper Surf = 0.35, Body L.E. = 46.3	
Factors of Safety (Define) See Volume II, Section 2	-
Margins (Define)	-
Contingency (Define) - Uncertainty Allowance	10,109
Weight Growth (Define) - 10% Dry Structure	24,694

*℄ Axle to ℄ Trunnion

**Fully Extended to Fully Collapsed

Table 4-11. FR-4 Vehicle Summary, Contd

SPACECRAFT DESIGN DATA STATEMENT

CONFIGURATION	BY	DATE
FR-4 BOOSTER (15' × 60' P/L BAY)		

REFERENCE MIL-M-38310A, SP-6004, AN-9103-D

SYSTEM/AIRFRAME DESIGN

1. Aerodynamic Surfaces		Wing	H. Tail	V. Tail
Volume, Wetted, Outer Mold Line (ft^3)		-		-
Surface Area of Above Volume (ft^2)		-		-
Gross Area (ft^2) (Exposed)		1,607.0		1,728.0
Span (ft)		158.3		73.0
Mean Aerodynamic Chord Length (ft)		13.6		-
Theoretical Root Chord Length (in.)		205.0		374.0
Theoretical Root Chord Max. Thickness (in.)		41.0		44.8
Theoretical Tip Chord Length (in.)		143.5		225.0
Theoretical Tip Chord Max. Thickness (in.)		25.8		22.5
2. Body Structure	**Cabin**	**Cargo**	**Remain**	**Main Tanks**
Volume, Wetted, Mold Line (ft^3)			122,370	
Pressurized Volume (ft^3)	350			69,306
Surface Area, Wetted, (ft^2)			18,420	
Maximum Length (ft)			199.3	
Maximum Depth (ft)			29.4	
Maximum Width (ft)			34.4	
Total Wetted Outer Mold Line Volume (ft^3)			122,370	
Total Wetted Outer Mold Line Surface Area (ft^2)			18,420	

3. Induced Envir. Prot.	Wing	H. Tail	V. Tail	Body	Fuel Tanks
Volume Delta Within					
1.0 & 2.0 (ft^3)				2,300	590.0
Surface Area Delta				18,420	
Within 1.0 & 2.0 (ft^2)					7,075.0
Windward Unit Wt. (lb/ft^2)				2.52	
Leeward Unit Wt. (lb/ft^2)				1.16	

4. Launch Recovery and Docking	Main	Nose	Other
Landing Gear, Max Vert. Load/Gear (lb)	210,000	90,000	
Length, Oleo Extended (in.)*	180	180	
Oleo Travel (in.)**	30	36	
Structural		Stress G. W.	Ult. L. F.
Flight (Subsonic Gust)		325,000	1.4
Landing		325,000	1.4
Limit Landing Sinking Speed (ft/sec)			12
Pressurized Cabin Ult. Dsn. Press. Diff. Flt. (psi)			20
Airframe Weight (as Defined in AN-W-11) (lb)			

SYSTEM DESIGN

5. Main Propulsion	Liftoff	Landing
Number of Engines	9	3
Max SL Static Thrust per Engine (lb)	400,000	40,600
Total Propellant Tankage Volume (ft^3)	69,306	-
Total JP Fuel Tankage Volume (ft^3)	-	695
6. Through 16.		
Maximum Electrical Power (KVA)		23
Pressurized Surface Area (ft^2)		9,825
Maximum Mission Duration (days)		1.5 hr
No. of Crew		2
17. Through 21.		
Maximum Number of Personnel Including Crew		2
Maximum Cargo (lb)		Orbiter
Maximum Cargo Density (lb/ft^3)		-
22. Through 27.		
Capacity Propellant Weight (lb)		1,508,915
Capacity JP Fuel Weight (lb)		30,717
GENERAL/MISSION DESIGN		
Maximum Design Gross Weight (lb) (Each)		1,879,048
Maximum Boost Load Factor 4 g at ______ G. W.		
Maximum Boost Dynamic Pressure (lb/ft^2)		657.8
Main Engine Specific Impulse (lb-sec/lb) (Uprated = 499.5)		389.3
Delta Velocity Available (ft/sec) Ideal		13,268
Entry Velocity (ft/sec)		9,411
Entry Weight (lb)		355,767
Ballistic Coefficient, W/C_DA (lb/ft^2)		-
Maximum Heating Rate (Btu/ft^2-sec) Stagnation = 8.5		
Factors of Safety (Define) See Volume II, Section 2		-
Margins (Define)		
Contingency (Define) - Uncertainty Allowance		11,703
Weight Growth (Define) - 10% Dry Structure		29,495

*℄ Axle to ℄ Trunnion **Fully Extended to Fully Collapsed

4.2 WEIGHT AND BALANCE SUMMARY

The overall system weights are covered in Volume V. The vehicle weight estimates for the FR-3 and FR-4 systems are summarized to the required NASA format in Tables 4-12 and 4-13, respectively. The mass properties data for the elements of the systems are given in Tables 4-14 and 4-15 for the FR-3 and FR-4 systems, respectively.

4.3 PERFORMANCE

This section discusses the baseline logistics mission. The mission profile is presented in Section 4.3.1, followed by a detailed discussion of each mission phase: ascent, on-orbit operations, retro, cruise, and landing. Section 4.3.7 discusses the performance requirements for abort.

4.3.1 MISSION PROFILE. The space shuttle transports cargo and personnel to and from a manned orbital space station and subsequently to a larger space base in low-altitude earth orbit. The cargo includes food, liquids, and gases in addition to both experiment modules and operational equipment. Personnel include trained astronauts and individuals who conduct specific scientific and technology experiments and operations. The shuttle logistics missions include long-lead-time scheduled resupply and crew rotations as well as discretionary flights.

The routine logistics mission is defined as a 55-deg inclined circular orbit at a 270-n.mi. altitude, with rendezvous within 24 hours of launch. The main propulsion subsystem on-orbit ΔV design requirement is 1800 fps and the attitude control propulsion subsystem ΔV design requirement is 200 fps.

Figure 4-20 shows the mission profile and the main flight phases. Each phase is discussed in detail in the following sections.

4.3.2 ASCENT TRAJECTORIES. The FR-3 15-ft diameter payload system ascent trajectory is shown in Figure 4-21. The FR-4 15-ft diameter payload system ascent trajectory is shown in Figure 4-22. Specific major trajectory data have already been shown in the vehicle synthesis summary runs in Section 4.1.

The FR-4 design point has an injection point into the initial orbit of 260,000 ft and a staging dynamic pressure of 50 psf. These were the selected conditions based on previous FR-1 work. Figure 4-23 shows that this is near the optimum dynamic pressure for minimum gross liftoff weight. While a lower trajectory appears to reduce gross liftoff weight, the implications of increasing boost aerodynamic heating have not been accounted for in the figure, which is based on synthesis runs using the point design thermal protection subsystem inputs. Further work is needed on the trajectory shaping to include detailed thermostructural weight effects.

Table 4-12. FR-3 Summary Weights

SPACECRAFT SUMMARY WEIGHT STATEMENT

CONFIGURATION: FR-3 POINT DESIGN (15' × 60' P/L BAY)
BY:
DATE:

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	U
1.0	AERODYNAMIC SURFACES	82,798						41,084	
2.0	BODY STRUCTURE	191,443						69,156	
3.0	INDUCED ENVIR PROT	47,388						43,052	
4.0	LNCH RECOV & DKG	30,269						14,020	
5.0	MAIN PROPULSION	141,261						47,539	
6.0	ORIENT CONTROL SEP & ULL	15,740						11,836	
7.0	PRIME POWER SOURCE	1,383						1,827	
8.0	POWER CONV & DISTR	4,092						2,443	
9.0	GUIDANCE & NAVIGATION	336						512	
10.0	INSTRUMENTATION	330						407	
11.0	COMMUNICATION	181						181	
12.0	ENVIRONMENTAL CONTROL	748						1,430	
13.0	(RESERVED)								
14.0	PERSONNEL PROVISIONS	297						615	
15.0	CREW STA CONTRL & PAN	308						308	
16.0	RANGE SAFETY & ABORT	275						55	
SUBTOTALS (DRY WEIGHT)		516,855						234,465	
17.0	PERSONNEL	476						540	
18.0	CARGO							50,000	
19.0	ORDNANCE								
20.0	BALLAST								
21.0	RESID PROP & SERV ITEMS	25,594						4,381	
SUBTOTALS (INERT WEIGHT)		542,925						289,386	
22.0	RES PROP & SERV ITEMS								
23.0	INFLIGHT LOSSES	159						4,829	
24.0	THRUST DECAY PROPELLANT	437						43	
25.0	FULL THRUST PROPELLANT	2,858,795						632,426	
26.0	THRUST PROP BUILDUP								
27.0	PRE-IGNITION LOSSES								
TOTALS (GROSS WEIGHT) (LB)		3,402,316						926,684	
DESIGN ENVELOPE VOLUME (FT^3)		236,000						89,060	
PRESSURIZED VOLUME (FT^3)									
DESIGN ENVEL SURF AREA (FT^2)		26,610						14,900	
PRESSURIZED SURF AREA (FT^2)									
DESIGN q, MAX (LB/FT^2)		670						670	
DESIGN n, MAX		4						4	
DESIGN POWER, MAX (KW)									
DESIGN NO. MEN/DAYS		2/,1						2/7	

DESIGNATIONS:

CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE

- A - BOOSTER
- B
- C
- D
- E
- F

SPACECRAFT

- M MANNED LAUNCH - ORBITER
- U UNMANNED LAUNCH

GROSS WEIGHT 4,329,000

NOTES & SKETCHES:

TANKS ARE OVERSIZED TO ACCOUNT FOR THRUST BUILD-UP AND PREIGNITION LOSSES.

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Table 4-13. FR-4 Summary Weights

SPACECRAFT SUMMARY WEIGHT STATEMENT									
CONFIGURATION: FR-4 POINT DESIGN (15' × 60' P/L BAY)			BY			DATE			
CODE	SYSTEM	ITEM OR MODULE						SPACECRAFT	
		A	B	C	D	E	F	M	U
1.0	AERODYNAMIC SURFACES	52,988	52,988					46,608	
2.0	BODY STRUCTURE	114,428	114,428					82,709	
3.0	INDUCED ENVIR PROT	36,092	36,092					55,361	
4.0	LNCH RECOV & DKG	15,835	15,835					15,830	
5.0	MAIN PROPULSION	88,464	88,464					49,717	
6.0	ORIENT CONTROL SEP & ULL	10,603	10,603					13,304	
7.0	PRIME POWER SOURCE	908	908					1,921	
8.0	POWER CONV & DISTR	2,656	2,656					2,670	
9.0	GUIDANCE & NAVIGATION	336	336					512	
10.0	INSTRUMENTATION	330	330					407	
11.0	COMMUNICATION	181	181					181	
12.0	ENVIRONMENTAL CONTROL	748	748					1,430	
13.0	(RESERVED)								
14.0	PERSONNEL PROVISIONS	297	297					615	
15.0	CREW STA CONTRL & PAN	308	308					308	
16.0	RANGE SAFETY & ABORT	275	275					55	
SUBTOTALS (DRY WEIGHT)		324,449	324,449					271,628	
17.0	PERSONNEL	476	476					540	
18.0	CARGO							50,000	
19.0	ORDNANCE								
20.0	BALLAST								
21.0	RESID PROP & SERV ITEMS	14,109	14,109					5,195	
SUBTOTALS (INERT WEIGHT)		339,034	339,034					327,363	
22.0	RES PROP & SERV ITEMS								
23.0	INFLIGHT LOSSES	159	159					5,014	
24.0	THRUST DECAY PROPELLANT	229	229					43	
25.0	FULL THRUST PROPELLANT	1,539,626	1,539,626					828,576	
26.0	THRUST PROP BUILDUP								
27.0	PRE-IGNITION LOSSES								
TOTALS (GROSS WEIGHT) (LB)		1,879,048	1,879,048					1,160,996	
DESIGN ENVELOPE VOLUME (FT^3)		122,370	122,370					107,470	
PRESSURIZED VOLUME (FT^3)									
DESIGN ENVEL SURF AREA (FT^2)		18,420	18,420					16,890	
PRESSURIZED SURF AREA (FT^2)									
DESIGN q. MAX (LB/FT^2)		658	658					658	
DESIGN g. MAX		4	4					4	
DESIGN POWER, MAX (KW)									
DESIGN NO. MEN/DAYS		2/.1	2/.1					2/7	

DESIGNATIONS:

CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE

- A - BOOSTER
- B - BOOSTER
- C
- D
- E
- F

SPACECRAFT

- M MANNED LAUNCH - ORBITER
- U UNMANNED LAUNCH

GROSS WEIGHT 4,919,092

NOTES & SKETCHES:

TANKS ARE OVER-SIZED TO ACCOUNT FOR THRUST BUILD-UP AND PREIGNITION LOSSES.

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Table 4-14. FR-3 Mass Properties Summary

SPACECRAFT SEQUENCE MASS PROPERTIES STATEMENT											Page Of
Configuration: FR-3 POINT DESIGN (15' × 60' P/L BAY)			By						Date		
No.	Mission Event	Weight (lb)	Center of Gravity (feet)			Moment of Inertia (slug-ft^2 10^6)			Product of Inertia (slug-ft^2 × 10^6)		
			x	y	z	I_{x-x}	I_{y-y}	I_{z-z}	I_{xy}	I_{xz}	I_{yz}
	GROSS LIFT-OFF	4,329,000	[illegible].8	0	92.5	36.39	356.69	357.39	-	1.093	-
	BOOSTER (℄ = 210.0)										
	LIFT-OFF	3,402,316	70.7	0	99.5	8.09	252.14	252.53	-	0.676	-
	MAX. αq	2,338,760	82.2	0	99.3	6.73	202.28	202.68	-	1.230	-
	BURNOUT	590,437	117.3	0	97.4	4.75	98.64	99.14	-	2.940	-
	ENTRY	564,247	114.9	0	97.2	4.74	96.02	96.51	-	2.820	-
	FLYBACK (INITIAL)	564,247	113.4	0	97.0	9.84	92.90	98.51	-	3.580	-
	LANDING	517,331	123.6	0	97.1	9.78	74.66	80.04	-	2.800	-
	BOOSTER PROPELLANTS	2,811,879	61.3	0	100.0	5.72	107.36	107.36	-	0	-
	MAX. αq PROPELLANTS	1,748,323	70.3	0	100.0	1.88	73.24	73.24	-	0	-
	FLYBACK PROPELLANTS	46,916	8.0	0	92.0	0.002	0.11	0.11	-	0	-
	ORBITER (℄ = 179.2)										
	LIFT-OFF	926,684	73.6	0	99.3	2.78	51.84	52.15	-	0.394	-
	BURNOUT	334,444	95.6	0	98.3	2.24	27.80	28.13	-	0.823	-
	ENTRY (PAYLOAD IN)	287,873	102.1	0	99.1	2.04	23.55	23.80	-	0.479	-
	FLYBACK (INITIAL)	287,873	101.8	0	98.9	3.05	22.66	23.83	-	0.670	-
	LANDING	285,005	102.7	0	98.6	3.12	22.29	23.38	-	0.626	-
	PAYLOAD	50,000	94.0	0	105.0	0.04	0.49	0.49	-	0	-
	ORBITER PROPELLANTS	592,240	61.2	0	99.8	0.52	16.14	16.14	-	NEGL.	-
	FLYBACK PROPELLANTS	2,868	23.0	0	92.0	NEGL.	NEGL.	NEGL.	-	0	-
			*		**						

NOTES:
* VEHICLE NOSE = STA 0 (+ AFT)
**GEOMETRIC CENTERLINE = WL 100
GROSS LIFT-OFF C. G. BASED ON BOOSTER REFERENCE SYSTEM

Ref. MIL-M-38310A or SP-6004

Table 4-15. FR-4 Mass Properties Summary

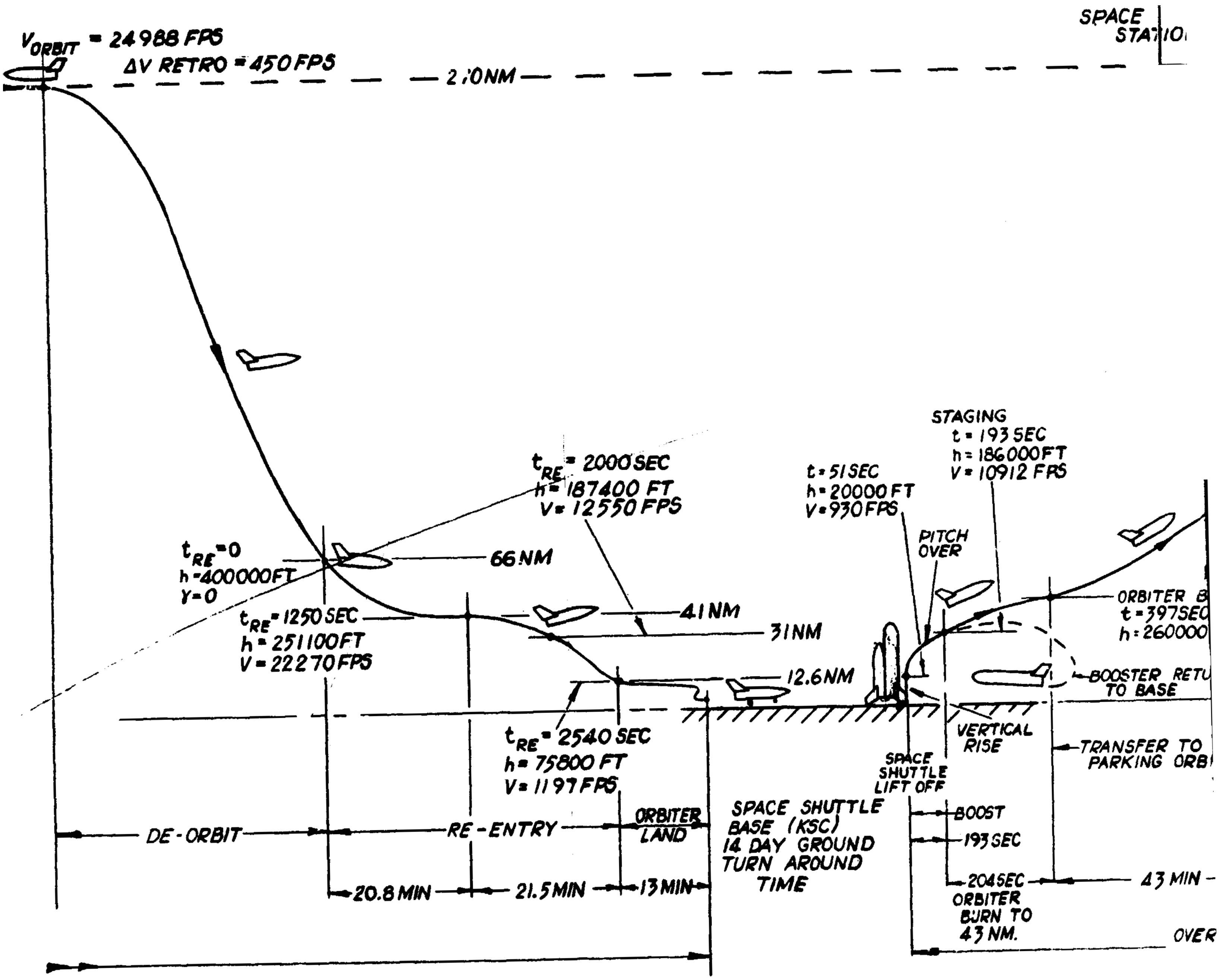
SPACECRAFT SEQUENCE MASS PROPERTIES STATEMENT

Configuration: FR-4 POINT DESIGN (15' × 60' P/L BAY) | By | Page | Of | Date

No.	Mission Event	Weight (lb)	Center of Gravity (feet) x	y	z	Moment of Inertia (slug-ft² 10⁶) I_{x-x}	I_{y-y}	I_{z-z}	Product of Inertia (slug-ft² × 10⁶) I_{xy}	I_{xz}	I_{yz}
	GROSS LIFT-OFF	4,919,092	84.0	0	99.3	130.54	408.95	289.94	-	1.049	-
	BOOSTER (ℓ = 199.3)										
	LIFT-OFF	1,879,048	85.9	0	99.5	3.84	108.85	108.97	-	-0.408	-
	MAX. αq	1,249,786	98.4	0	99.3	3.36	80.52	80.65	-	0.769	-
	BURNOUT	370,133	122.1	0	97.5	2.39	46.96	47.14	-	1.460	-
	ENTRY	355,636	121.3	0	97.4	2.38	45.97	46.15	-	1.430	-
	FLYBACK (INITIAL)	355,636	119.8	0	97.2	3.66	44.38	45.73	-	1.900	-
	LANDING	324,926	128.4	0	97.3	3.62	36.37	37.59	-	1.400	-
	BOOSTER PROPELLANTS	1,508,915	81.0	0	100.0	2.38	43.69	43.69	-	0	-
	MAX. αq PROPELLANTS	879,653	88.4	0	100.0	0.91	24.33	24.33	-	0	-
	FLYBACK PROPELLANTS	30,711	40.0	0	92.0	0.001	0.03	0.03	-	0	-
	ORBITER (ℓ = 190.8)										
	LIFTOFF	1,160,996	78.0	0	99.2	3.32	69.98	70.27	-	0.119	-
	BURNOUT	377,763	104.3	0	98.1	2.61	32.47	32.79	-	0.804	-
	ENTRY (PAYLOAD IN)	325,393	108.7	0	98.6	2.39	29.72	29.95	-	0.559	-
	FLYBACK (INITIAL)	325,393	107.4	0	98.7	3.65	28.33	29.74	-	0.889	-
	LANDING	322,168	108.3	0	98.4	3.73	27.78	29.10	-	0.817	-
	PAYLOAD	50,000	97.0	0	105.0	0.04	0.49	0.49	-	0	-
	ORBITER PROPELLANTS	783,233	65.4	0	99.8	0.69	25.50	25.50	-	-0.159	-
	FLYBACK PROPELLANTS	3,225	23.0	0	92.0	0	0	0	-	0	-
			*		**						

NOTES:
- * VEHICLE NOSE = STA 0 (+ AFT)
- **GEOMETRIC CENTERLINE = WL 100 (+ UP)

Ref. MIL-M-38310A or SP-6004



FOLDOUT FRAME 1

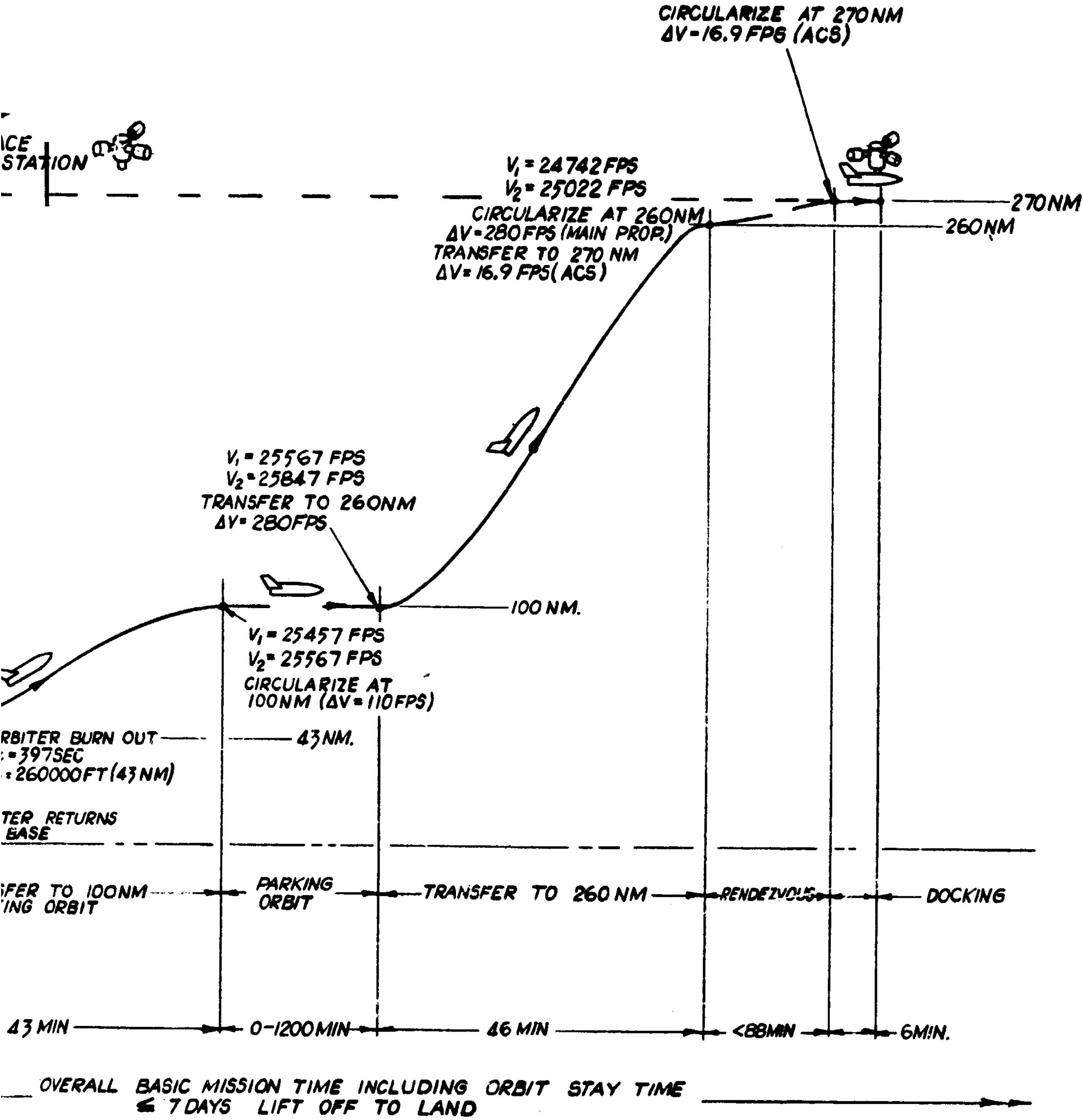


Figure 4-20. Mission Profile and Velocity Requirements - FR-3

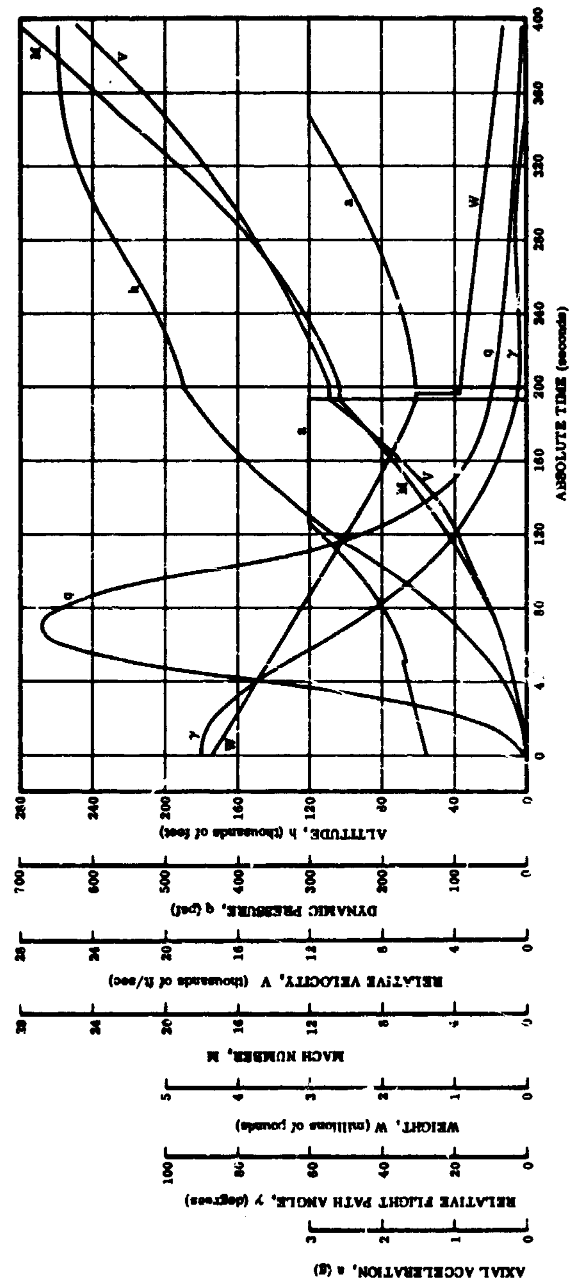


Figure 4-21. FR-3 Ascent Trajectory

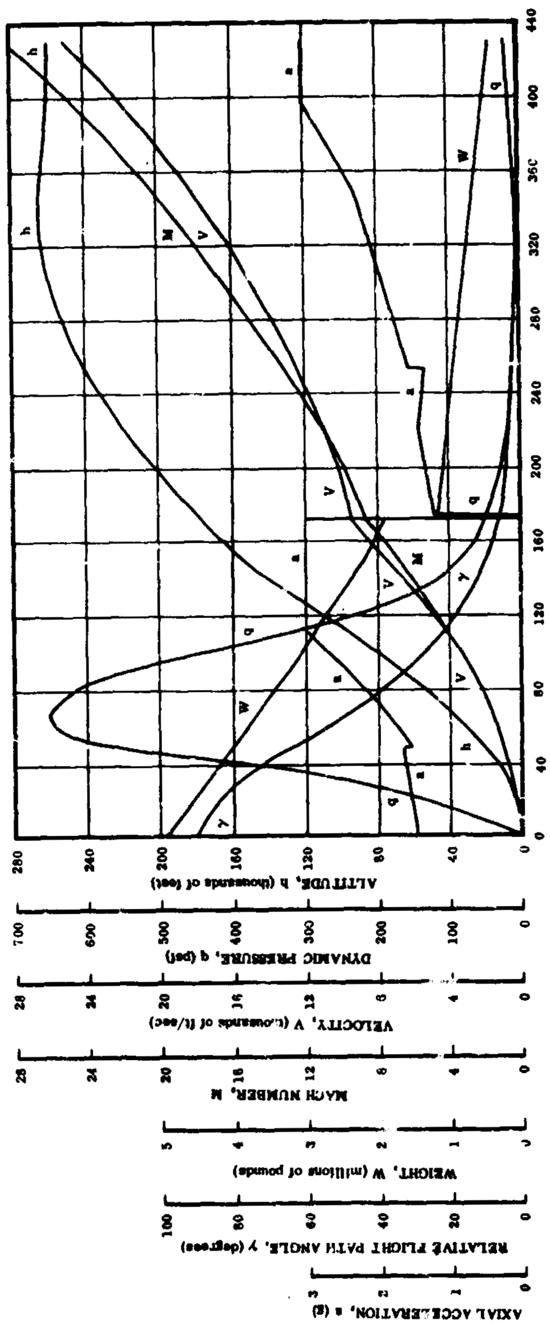


Figure 4-22. FR-4 Ascent Trajectory

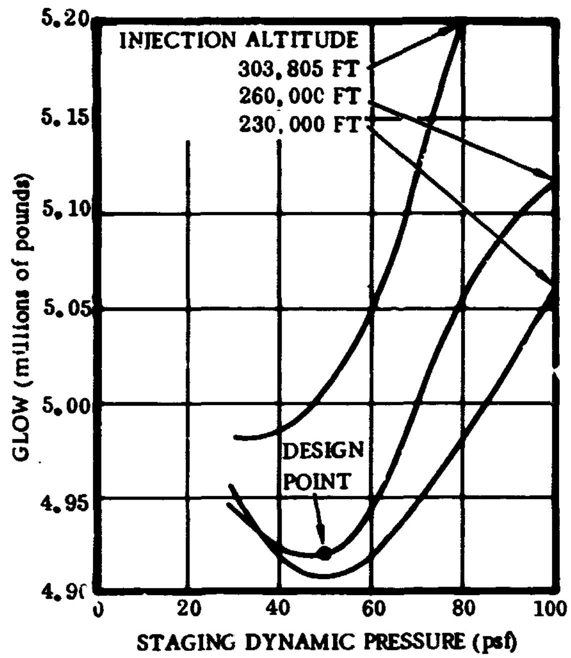


Figure 4-23. NASA FR-4 Point Design Minimum GLOW

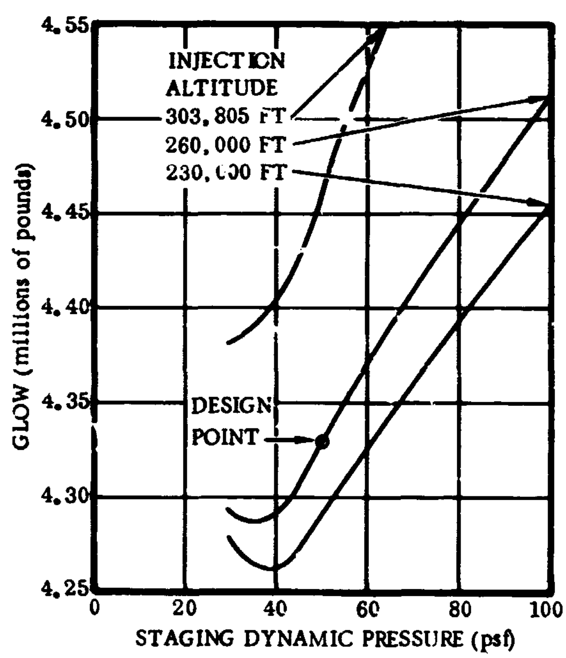


Figure 4-24. NASA FR-3 Point Design Minimum GLOW

The FR-3 point design was held at the same staging dynamic pressure of 50 psf (although the staging velocity is higher) and injection point of 260,000 ft (approximately 43 n.mi.). However, subsequent synthesis runs indicate that for the present system assumptions a lower staging dynamic pressure of 35 psf would result in a slightly lower gross liftoff weight (Figure 4-24).

4.3.3 ON ORBIT. The on-orbit activities begin with a transfer ellipse from 45 to 100 n.mi. and end with entry.

The main propulsion subsystem ΔV requirements are:

Maneuver	Main Propulsion ΔV (fps)
Circularize at 100 n.mi.	110
Transfer to 260 n.mi.	280
Circularize at 260 n.mi.	280
Entry	450
FPR and Contingencies	480
Insertion Dispersions and Out-of-plane Errors	200
Total System ΔV	1800

The ACPS furnishes limit cycle attitude control to +45 deg while in-orbit hold or during orbit transfers; orientation to ± 5 deg prior to each orbit maneuver burn and during rendezvous; roll control to ± 5 deg during each maneuver burn; ΔV to transfer from 260 n.mi. to 270 n.mi. and to rendezvous, dock, and undock; and orientation to ±2 deg during entry. The total ACPS ΔV for the mission is 157.9 fps. The ΔV requirements for each mission phase are given in Table 4-16. The design requirement of 200 fps indicates a reserve and dispersion allowance of 42.1 fps.

4.3.4 ENTRY PERFORMANCE

4.3.4.1 Orbiter Entry. Orbiter entry is established by the retro conditions used. Following undocking, the orbiter is oriented in such a way that the thrust axis is aligned with the flight path. The engines are fired for a duration sufficient to provide a retro ΔV of 380 fps, which results in entry at a flight path angle of -1 deg at 400,000 ft altitude from a 270-n.mi. orbit. This entry angle is sufficiently above the skip limit to avoid skip possibilities, and yet yields an entry flight path which is less stringent from an aerodynamic heating standpoint than a steeper entry would be.

Entry flight path characteristics were determined in a point mass trajectory computer program. In this program, the vehicle is flown at zero bank angle during pull-out until the flight path angle reaches zero. At this point, the vehicle rolls to the point which permits flight at constant altitude. As speed is reduced, the vehicle rolls back towards wings level until that roll angle is reached which is desired for purposes of gaining lateral range. The entry continues at this bank angle until a lower limit velocity cut-off is reached. Two such entry flight paths are shown, one for an 800 n.mi. crossrange in Figure 4-25, and the other for a 300 n.mi. crossrange in Figure 4-26. The 800 n.mi. crossrange case requires a 20 deg bank, while the 300-n.mi. case involves only the bank involved in the constant altitude portion of the entry described above, with a final bank angle of zero.

4.3.4.2 Booster Return. Following staging, an analysis of the booster return was made using the point mass entry trajectory program. The program was modified in such a way as to permit the following sequence of events. The vehicle flies inverted while pulling as much downward aerodynamic force as the combination of angle of attack and dynamic pressure will permit. When the flight path angle reaches zero, the vehicle is rolled to a bank angle which accomplishes the turn maneuver necessary to return to base. This bank angle must not exceed the normal acceleration limit for the vehicle. For the FR-4 vehicle, this limit of 4 g limited the bank angle to 55 deg, and for the FR-3 vehicle, a bank angle of 60 deg could be used without exceeding the g limit. A time history of the FR-3 booster return is presented in Figure 4-27, and the FR-4 booster return data are shown in Figure 4-28.

4.3.5 CRUISE.

Following staging and subsequent booster entry, the booster cruises approximately 300 n.mi. to return to base. The wings are deployed when the velocity during entry has been reduced to just subsonic, which occurs at an altitude of approximately 60,000 ft. The engines are extended at an altitude of

Table 4-16. ACPS ΔV Requirements

Task	Function	ΔV (fps)	Task	Function	ΔV (fps)
Limit cycle during transfer and orbit coast for 20 hours	Attitude Control to ±45 deg	9.7	Rendezvous and dock	Translate	7.2
	Drag Makeup	5.2		Attitude Control to ± 5 deg for 15 minutes	1.1
Orbit Maneuvers	Orientation to ± 5 deg for 20 minutes prior to maneuver. 4 maneuvers	5.9		Attitude Control to ± 0.5 deg for 20 seconds	1.7
	Control roll during maneuvers. ± 5 deg	1.0	Undock	Translate	8.3
Transfer from 260 to 270 n.mi.	Transfer ΔV	16.9		Attitude Control to ± 0.5 deg for 20 seconds	1.7
	Attitude Control to ± 0.5 deg for 22 seconds before and during burn	3.1	Limit cycle for 24 hours	Attitude Control to ± 45 deg	11.6
	Attitude Control to ± 5 deg during 0.75 hour transfer	3.3	Entry	Control to ± 2 deg with 2.5-3 deg/sec^2	23.1
Circularize at 270 n.mi.	Transfer ΔV	16.9		Control to ± 2 deg with 1.9-2.25 deg/sec^2	26.0
	Attitude Control to ± 0.5 deg for 22 seconds before and during burn	3.1		Control to ± 2 deg with 0.75-1.25 deg/sec^2 for 750 sec	11.5
			Total ACPS ΔV Requirements		157.9

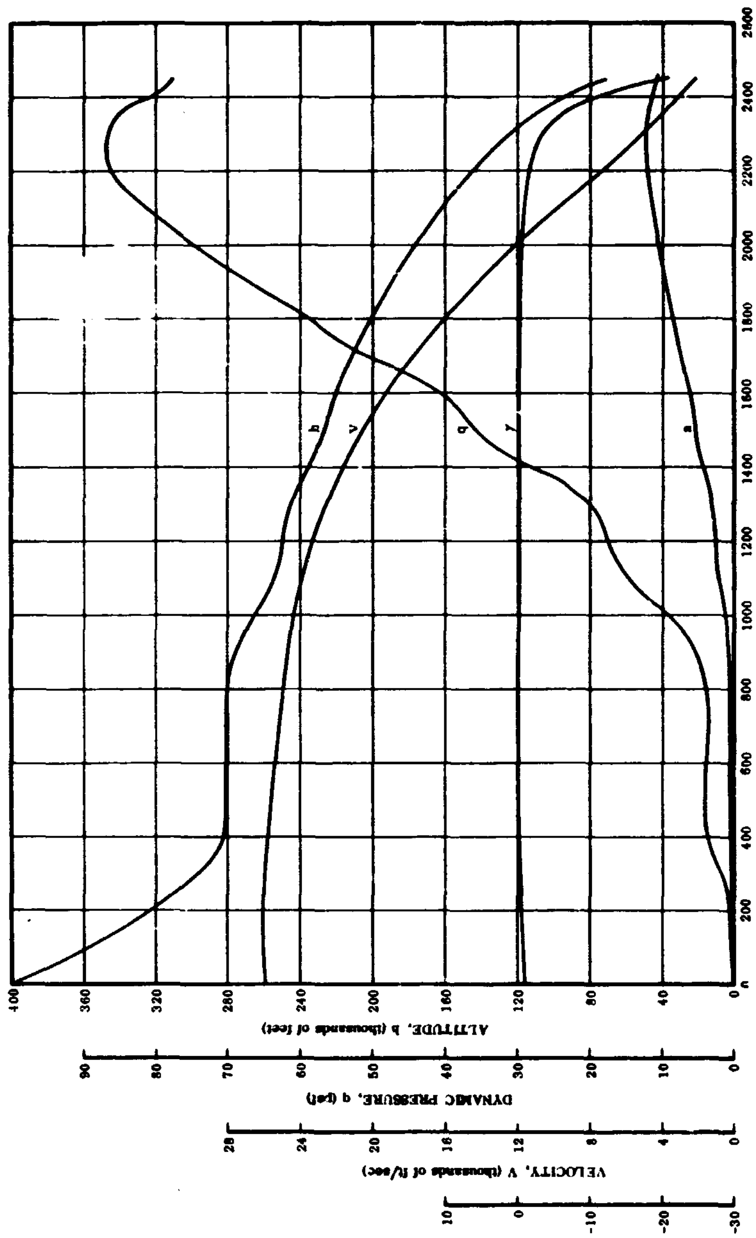


Figure 4-25. 800-n.mi. Crossrange Time History

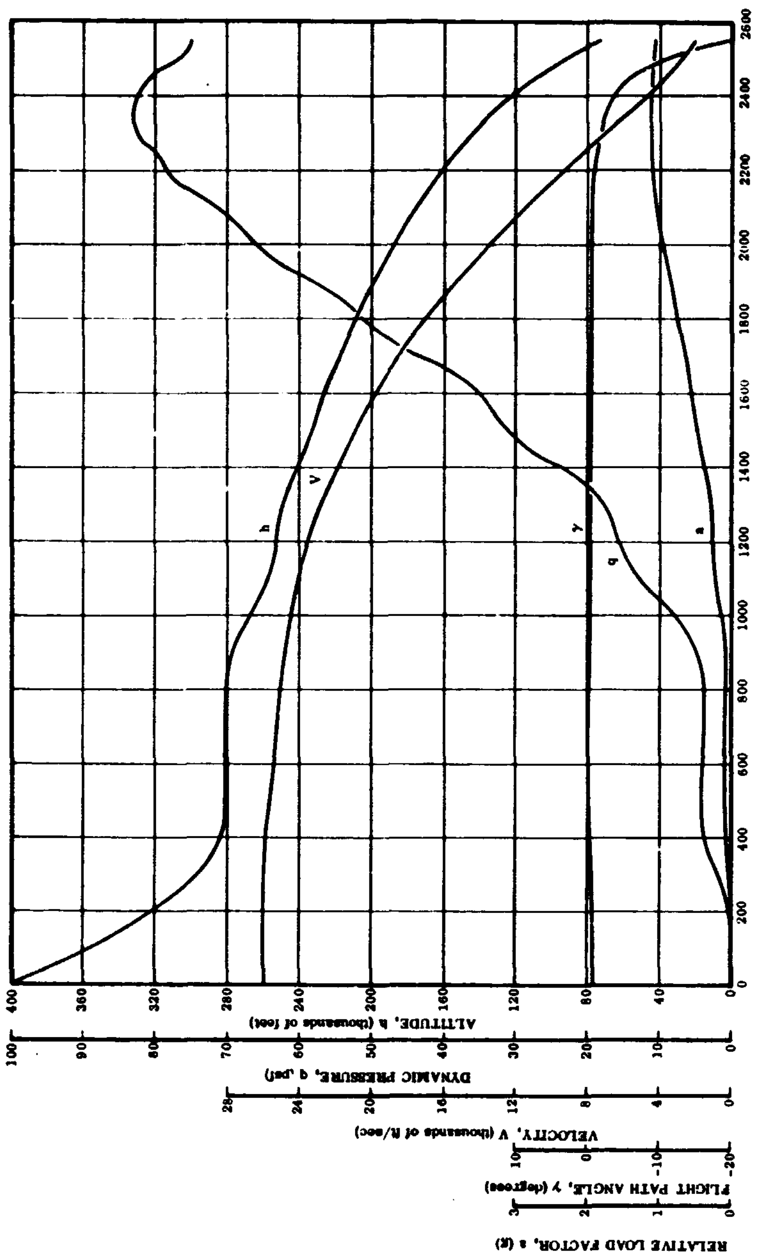


Figure 4-26. 300-n.mi. Crossrange Time History

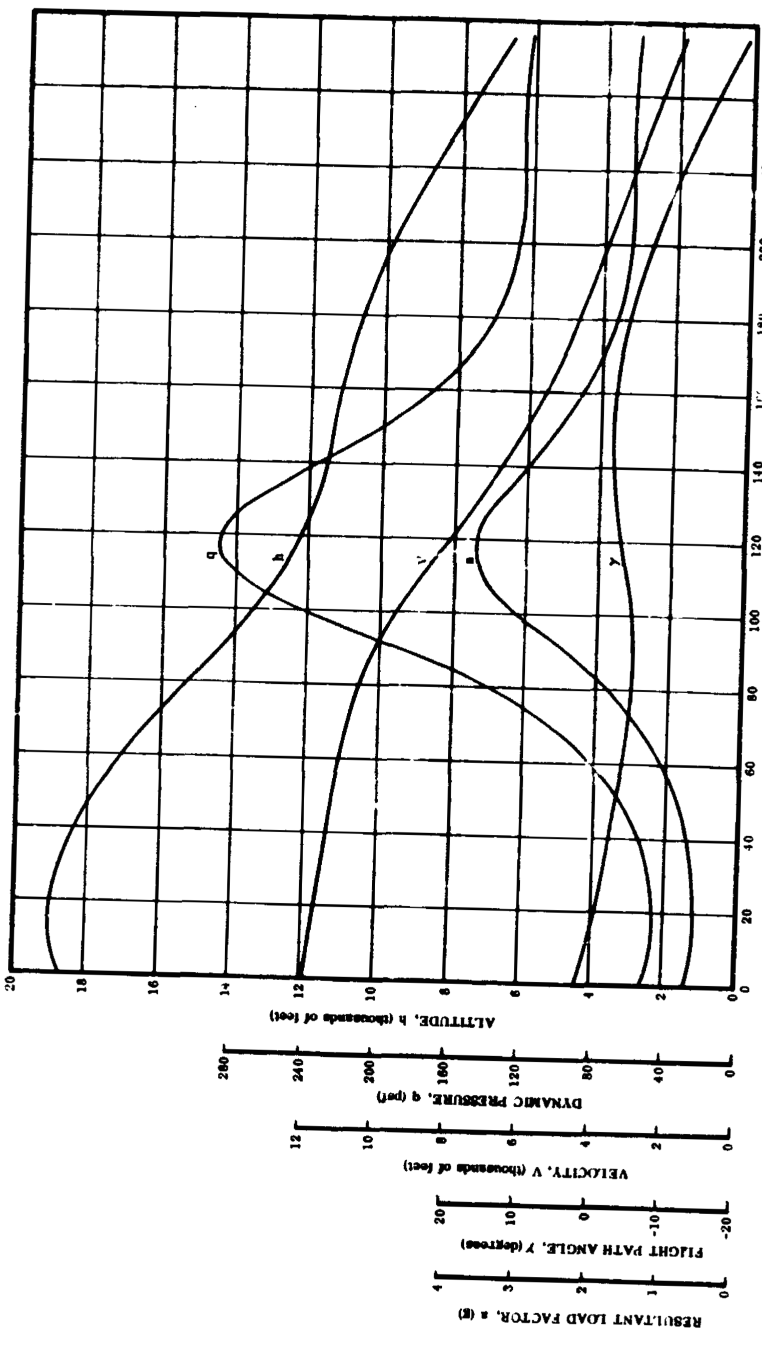


Figure 4-27. FR-3 Booster Entry

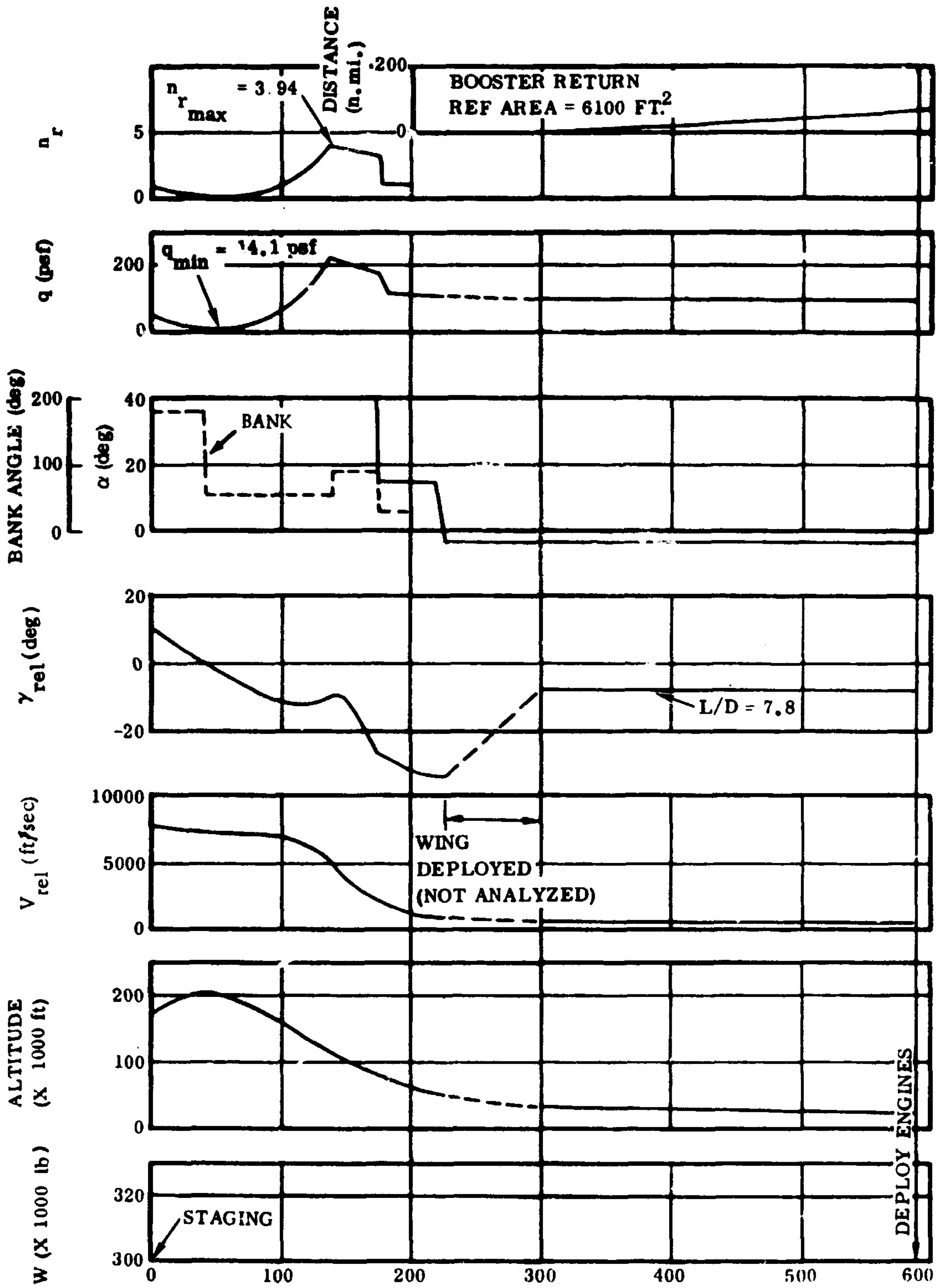


Figure 4-28. FR-4 Booster Return Trajectory - Staging to Engine Deployment

approximately 25,000 ft, started, and idled until the 15,000 ft altitude point is reached. Power is then applied, and the cruise back continued for the return to base. Figure 4-29 presents the FR-4 booster cruise history. FR-3 is very similar. The fuel requirements for this phase were computed from the Breguet range expression:

$$\text{Range (n. mi.)} = \frac{(L/D)\,V}{SFC}\ \text{Log}_e\ (W_{init}/W_{final})$$

where

V	=	cruise speed in knots
L/D	=	cruise lift/drag ratio
W_{init}	=	vehicle gross weight at start of cruise
W_{final}	=	vehicle gross weight at end of cruise
SFC	=	airbreathing engine specific fuel consumption — lb fuel/hr/lb thrust

The cruise speed was selected as the speed for L/D_{max}. A speed slightly higher than this will yield a small gain in the range parameter (L/D)V/SFC, but the installed engine weight must increase to yield the higher thrust required by this higher speed. Previous studies have indicated that a cruise altitude of 15,000 ft is reasonable from the standpoint of the combined engine plus fuel weight. The variation of cruise range with cruise fuel for a typical booster and orbiter is shown in Figure 4-30.

4.3.6 LANDING AND GO-AROUND. While flight tests conducted at the Flight Research Center at Edwards Air Force Base have indicated that, under ideal conditions of weather, terrain, basing, and pilotage, it is possible to land an unpowered low L/D vehicle, there are clear indications that improvement in landing capabilities can be obtained reasonably and should be incorporated in the vehicles under consideration in the space shuttle program. The power-off rate of sink, the difficulty in performing the flare, the landing visibility, and the touchdown speed can all be improved with the use of wings. The rate of sink and the landing flare can be further improved with the use of airbreathing engines for the orbiter (the booster has fly-back engines) used in the landing phase. For all-weather capability, it is mandatory that engines be used to permit the approach glide at the standard 3-deg glide slope. Engine plus fuel weight were computed for a number of possible engine installations used for powered landing approach of a typical orbiter element having a weight of 350,000 lb at the start of cruise. These engines were: (1) XJ-99 VTOL lift turbojet type, (2) Rolls Royce turbofan, (3) LO_2-H_2 attitude control propulsion system with added thrust chambers, and (4) JATO, all "rubberized" to provide the required thrust levels. These engine plus fuel data were generated for installed thrust sufficient to (1) yield a

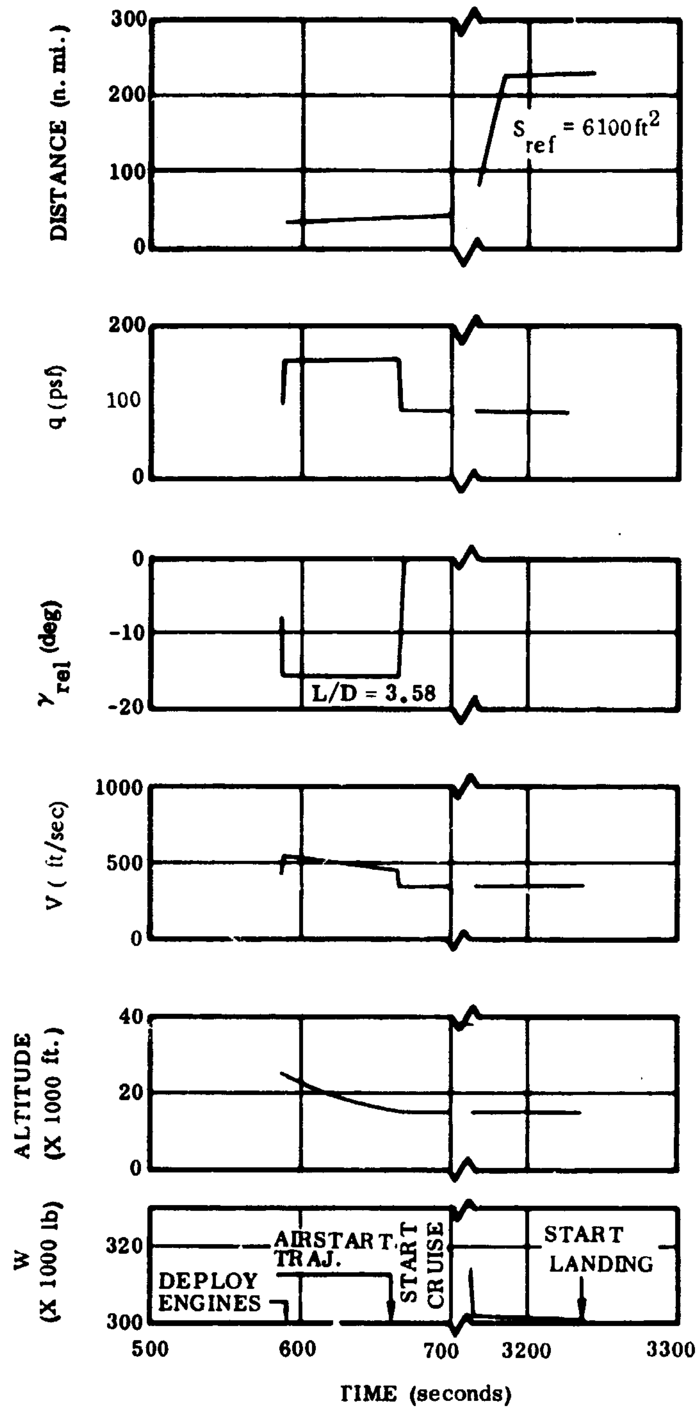


Figure 4-29. FR-4 Booster Return Trajectory - Engine Deployment to Landing

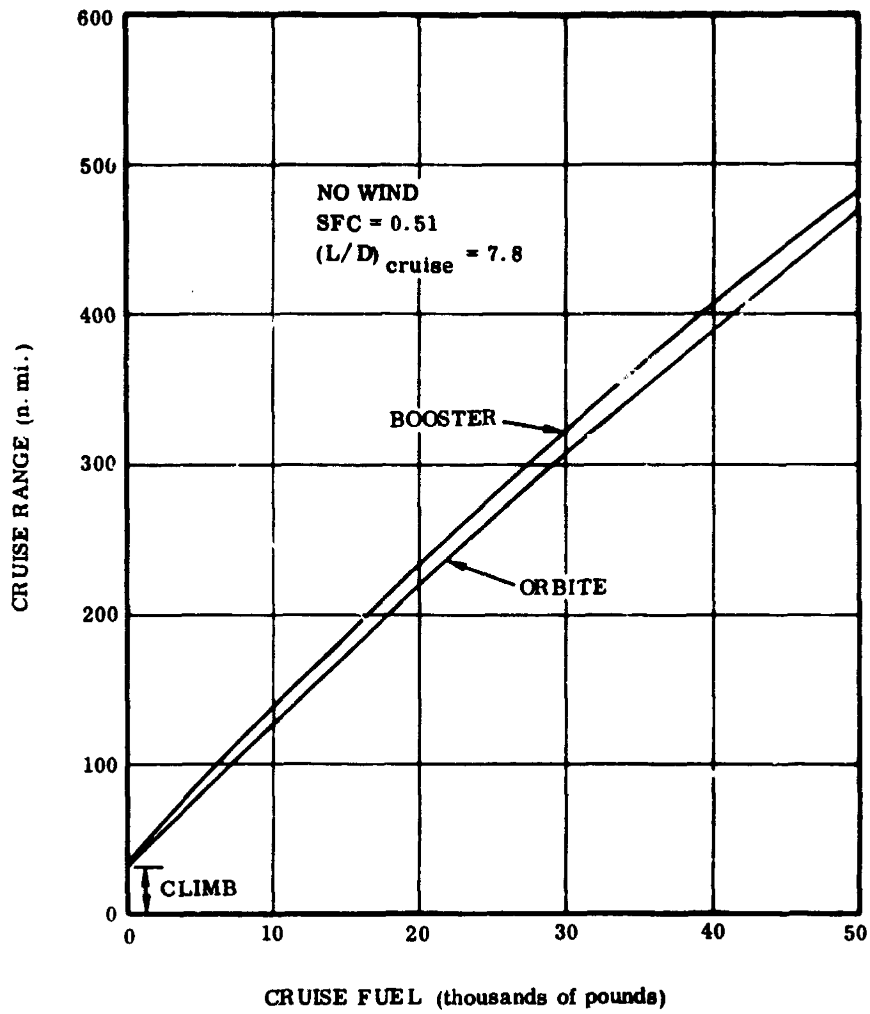


Figure 4-30. FR-4 Cruise Performance

powered glide down a 3-deg glide slope, and (2) level flight for standard sea level conditions for landing approach only (not for go-around). These data are presented in Figure 4-31. The upper curve is for the level flight case wherein engine plus fuel weight is plotted against level flight range (which is range gained over the unpowered case). The lower case is plotted against this same range gained over unpowered flight and also against range along the flight path. Consider an example case: Suppose it is desired to obtain the weight of the engine plus fuel for the XJ-99 type installation for the glide case where the range gained over the unpowered case is an arbitrary 4 n.mi. The lower curve indicates that this weight is 4200 lb, while the upper curve indicates that this weight would be 6000 lb if the level flight capability is desired.

The level flight capability has the advantage of being able to "stretch" the landing approach to a greater degree than the powered glide if in the powered glide case the decision to apply power comes late in the approach. The curves of Figure 4-31 were generated using the following data:

Propulsion Mode	Engine Thrust At 170 knots (lb)	SFC	I_{sp}
R/R TF	4.32	0.49	—
XJ-99 Type	9.94	1.13	—
LO_2-H_2 ACPS	60	—	393
JATO	Engine Weight 0.25 Fuel	—	245

(Fuel for all cases includes a 20% reserve for engine start and residuals)

The landing characteristics were examined using a three-deg-of-freedom landing trajectory program, both powered and unpowered, for the typical orbital vehicle. A standard -3 deg flight path approach was assumed for the powered landings. Figure 4-32 presents a landing history with an approach speed of 1.2 times the power-off stall speed; the flare was initiated at 50 ft, and the sink rate at touchdown was 2.7 fps. This approach speed corresponds to a lift coefficient of 0.486, which is less than that of maximum L/D. Touchdown speed is 186 knots.

Figure 4-33 presents a maximum L/D approach, which resulted in a 4.2-fps sink rate at touchdown. The flare maneuver requires larger angle of attack changes because of the decline of lift curve slope above maximum L/D. The touchdown speed for this maneuver is 165 knots.

Figure 4-34 presents a power-off approach at 320 fps, which results in a -8.1-degree glide slope. The flare maneuver was initiated at 150 ft and the sink rate at touchdown was 7.5 fps. The oscillations during this landing show that further work is needed on the stability augmentation system. The power-off approach must be made at this speed to allow reasonable maneuverability without thrust.

The booster has go-around capability using the engines installed for the return following staging. For the orbiter, however, it is questionable whether the go-around capability is really required in view of the weight penalty involved and the probability of a go-around really ever being required. Data and information regarding airline go-around experience were obtained from Pan American Airlines. Landing attempts by Pan American result in go-around being required for 1/2 of 1% of the landings (5 go-arounds for 1000 flights). The reasons for these are: (1) unscheduled traffic on the runways, and (2) aircraft in the landing phase too close to aircraft ahead. For the space shuttle vehicle, it would appear that neither of these is likely.

If it is desired that the orbiter have go-around capability, calculations indicate that about 3500 lb of fuel are required for this go-around, assuming that turbofan engines are used in a 356,000-lb orbiter. Engine weight has been indicated by a lower limit of 4500 lb (Figure 4-31) for the XJ-99 type engine plus additional engine installation weight to provide for some climb capability.

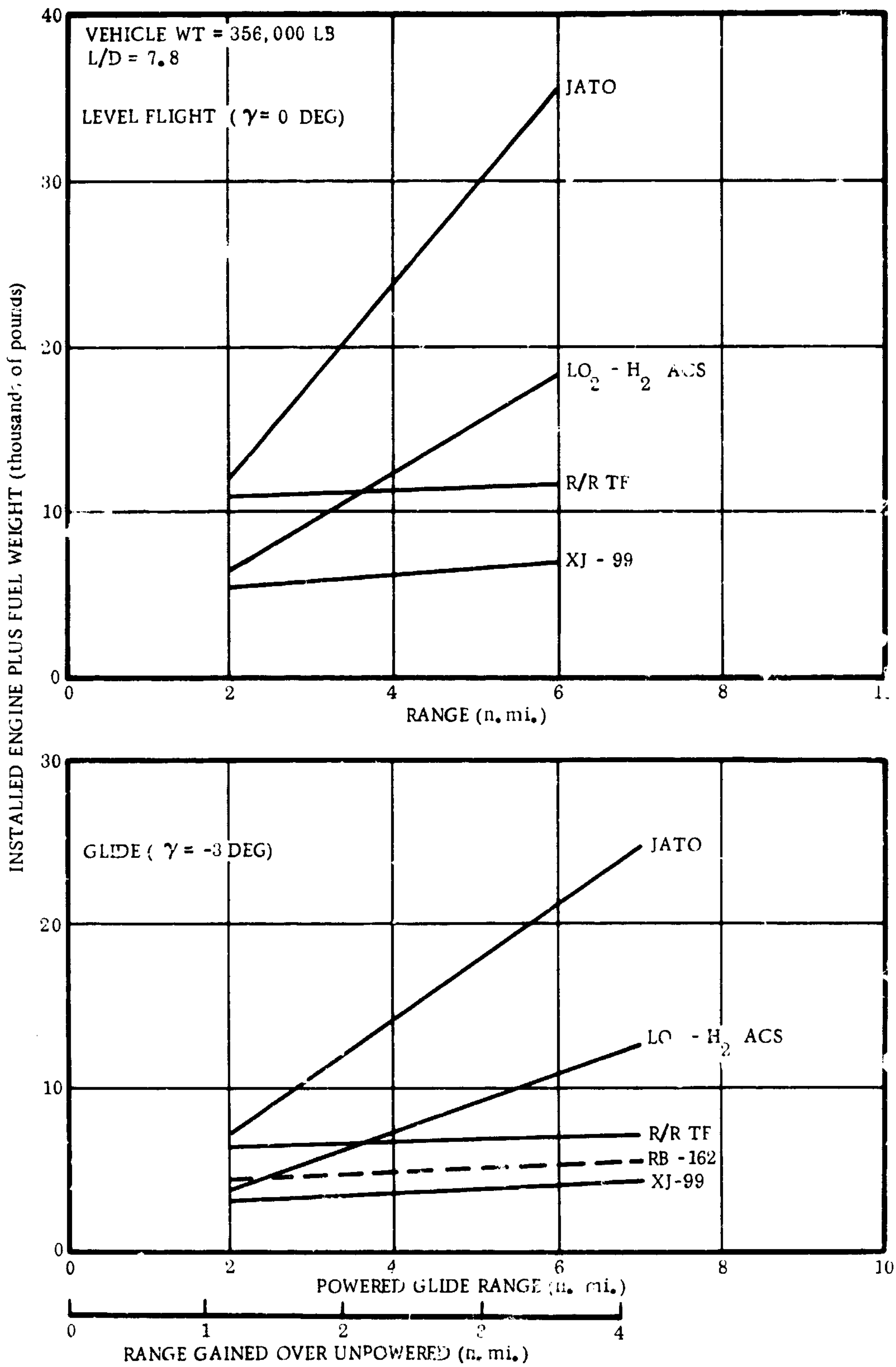


Figure 4-31. Landing Engine Plus Fuel Weight

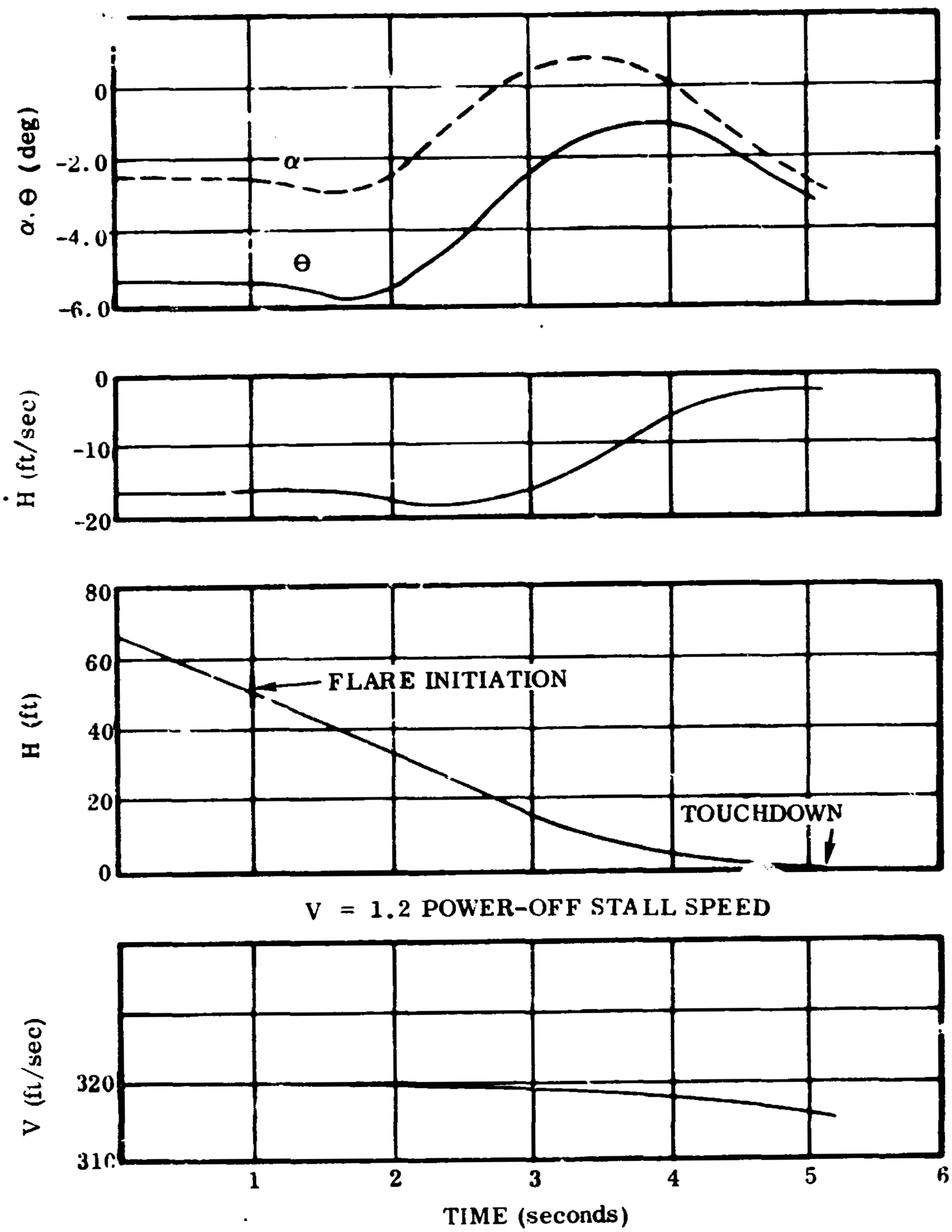


Figure 4-32. FR-4 Flare (V = 320 ft/sec, γ = -3 degrees)

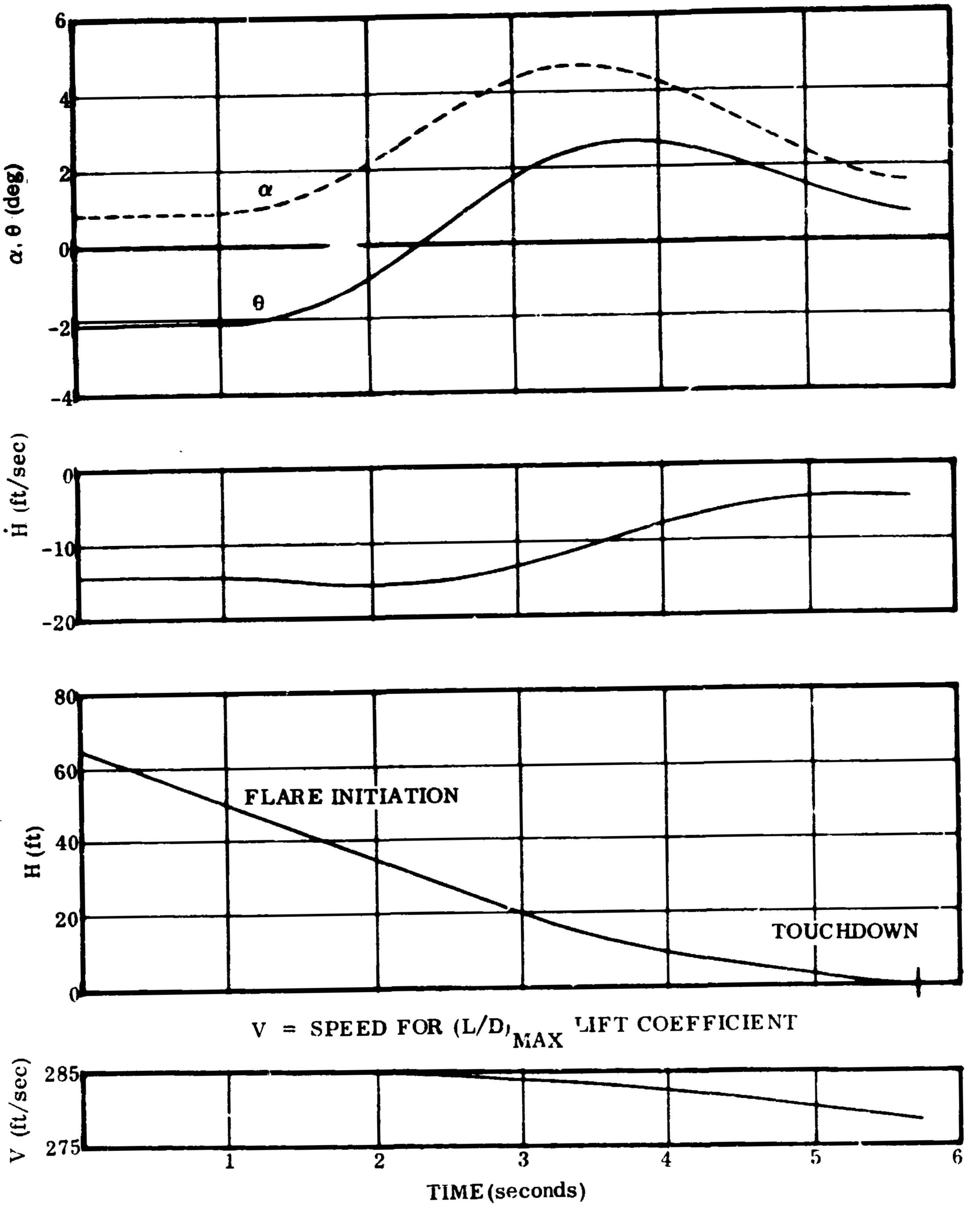


Figure 4-33. FR-4 Flare (V = 285 ft/sec, γ = -3 degrees)

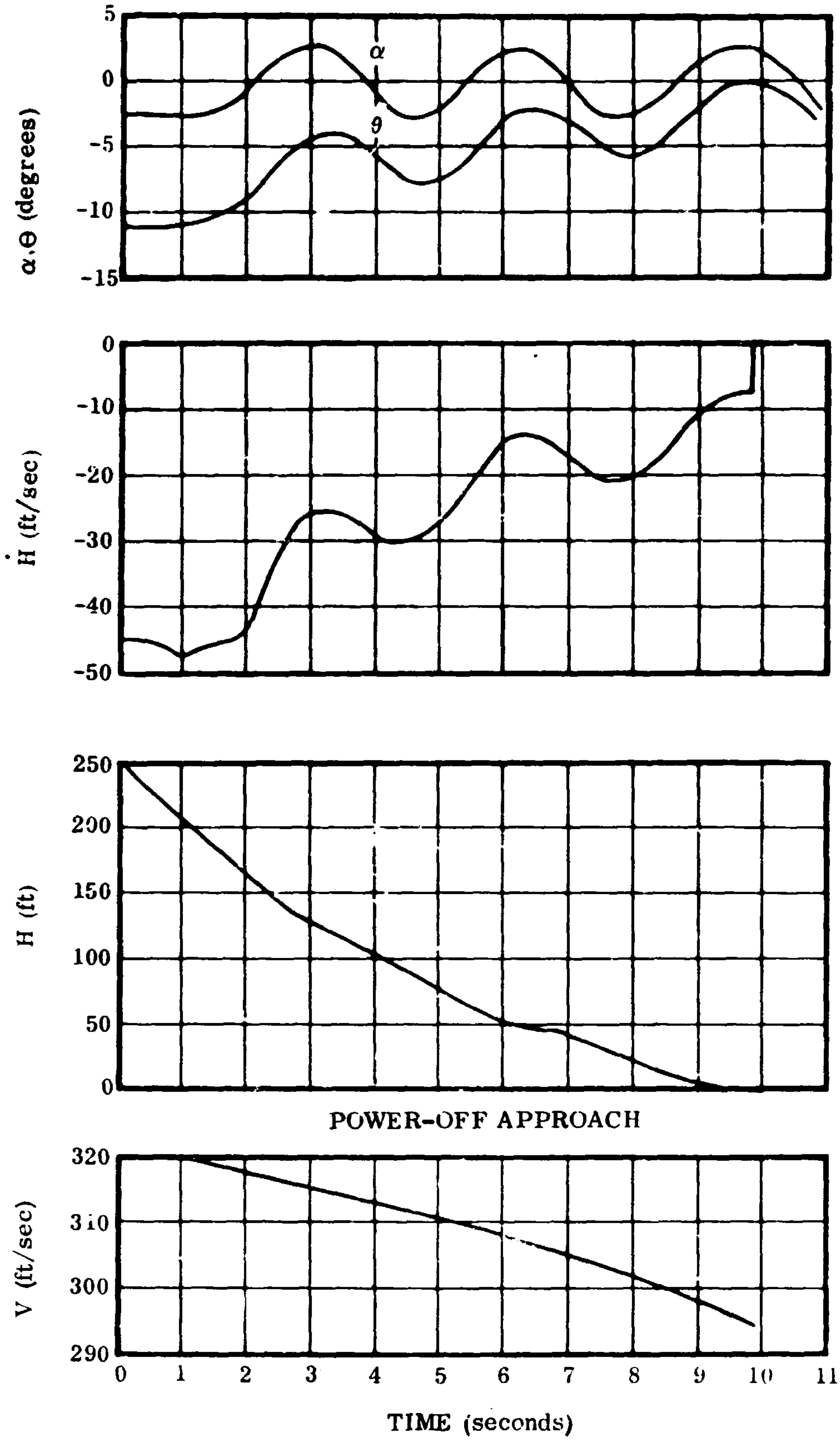


Figure 4-34. FR-4 Flare ($V = 320$ ft/sec, $\gamma = -8.1$ degrees)

4.4 AERODYNAMICS

4.4.1 FR-3. The subsonic aerodynamic characteristics of the FR-3 booster were determined using incremental wing, tail, and body contributions from the IPD tests corrected for configuration differences such as new nose shape and relative surface sizes using empirical methods similar to those of the USAF Stability and Control Datcom.

The FR-3 booster is less stable than the IPD configuration about the same reference moment center in terms of vehicle length for two reasons: the tail moment arm is shorter, and the body bluntness is de-stabilizing. However, the actual cg of the blunter FR-3 vehicle is somewhat farther forward than for the other vehicles. Figure 4-35 presents subsonic data for the FR-3 booster configuration. Note that the vehicle is still stable at a wing sweep as low as 16 degrees. The deployable wing allows the sweep to increase to maintain trim throughout the cruise as the cg moves aft.

The hypersonic characteristics of the FR-3 booster were determined using the hypersonic aerodynamic program (HAP). These data are presented in Figure 4-36.

4.4.2 FR-4. The subsonic aerodynamic characteristics of the FR-4 configuration were determined from wind tunnel testing in the Convair low speed wind tunnel and the low turbulence tunnel at Langley Research Center. The longitudinal stability and control for a center of gravity position 0.55 times the vehicle length are seen to be satisfactory. Pitch control using the ruddervators is very effective, yielding trim capabilities over a wide range of angle of attack. The static directional stability is good at all angles of attack. The low speed L/D_{max} with flaps retracted is seen to be 6.2 model scale. Figure 4-37 presents these low speed data. With flaps extended, the L/D_{max} increases to 7.2 model scale, or 7.8 full scale. One reason for the increase in L/D with flap extension lies in the fact that the flap action used results in a substantial wing area increase as well as probably providing a better effective wing incidence.

The hypersonic aerodynamic characteristics were determined from wind tunnel tests conducted in Tunnel C at AEDC on the IPD configuration combined with data generated using the Convair hypersonic aerodynamic program. The resulting data about a moment reference center at 0.55 times the vehicle length are shown in Figure 4-38.

Transonic wind tunnel tests conducted at Cornell Aeronautical Laboratory indicate that the static longitudinal and directional stability increased markedly in the transonic regime over the other speed regimes. The vehicle trims down transonically rather than showing any pitch-up tendency.

Details of the aerodynamic characteristics for the FR-1, FR-4, and FR-3 configurations are presented in Section 2 of Volume IV.

4.4.3 CENTER OF GRAVITY PROBLEMS. The aerodynamic data presented in these sections were based on early vehicle center of gravity estimates. The vehicle weights data shown in this volume show that the center of gravity is farther aft than that used in the aerodynamic analysis and the design loop has not been closed to bring these analyses together. While this center of gravity problem is serious, it is not uncommon during this stage of configuration definition.

Potential aerodynamic solutions which should not add any weight include:

a. Increasing V-tail rollout.

b. Deflecting bottom surface trimming surface.

c. Reducing nose camber.

d. Reducing body side slope at V-tail.

The FR-3 and FR-4 orbiters can be retrimmed by further deflection of the 10 degree trimming surface. The V-tail sizing analysis (Volume IV) showed that the vertical tail area could be reduced so that increasing the V-tail rollout can be done to improve longitudinal stability and control. The FR-4 booster has the more severe problem, and some reshaping of the body to reduce nose camber pitch-up and side slope reduction to increase V-tail effectiveness might have to be done. Close examination of the possibilities of moving equipment forward would be performed at the same time to see if the center of gravity can be moved back toward its initial location.

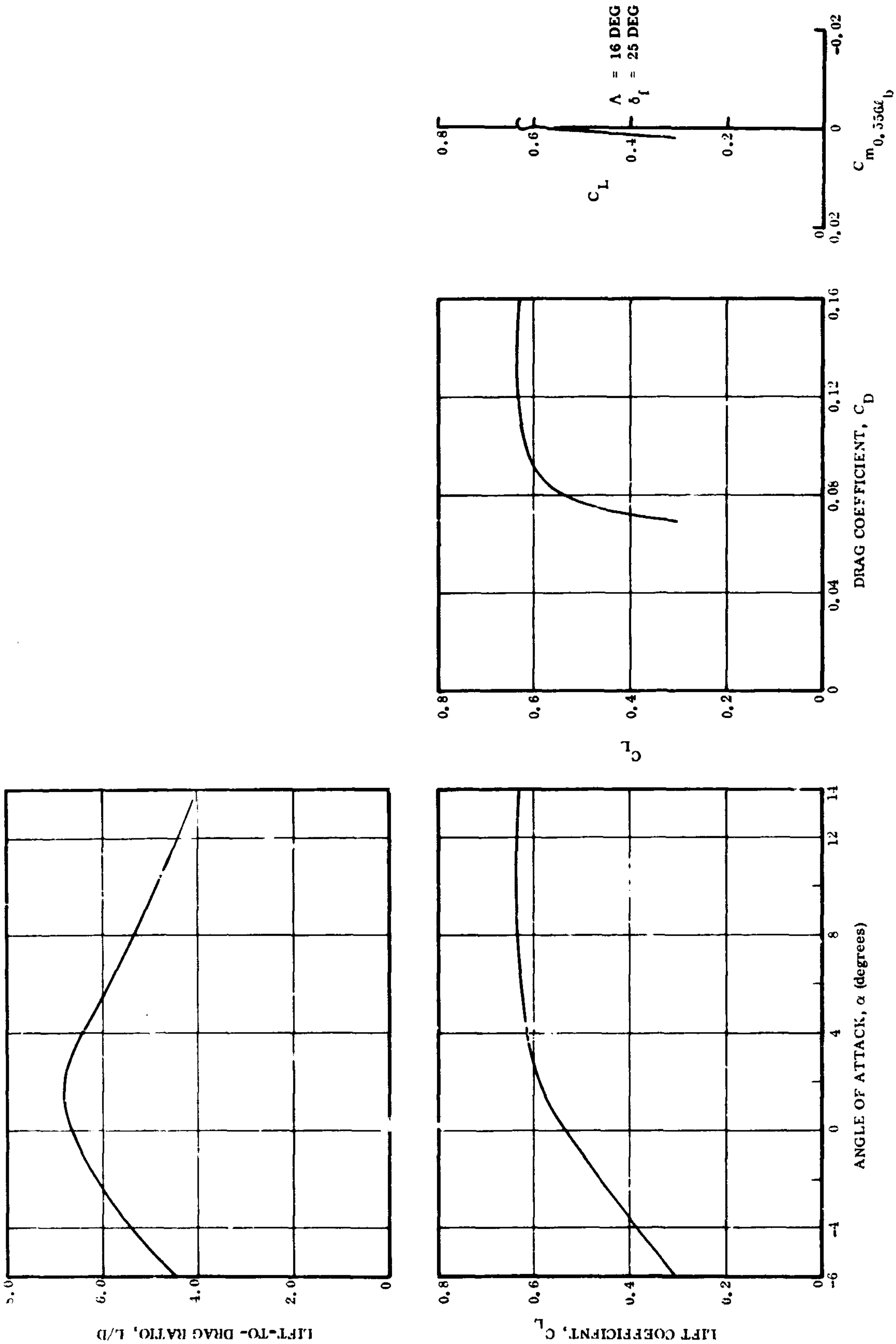


Figure 4-35. FR-3 Subsonic Longitudinal Aerodynamic Characteristics

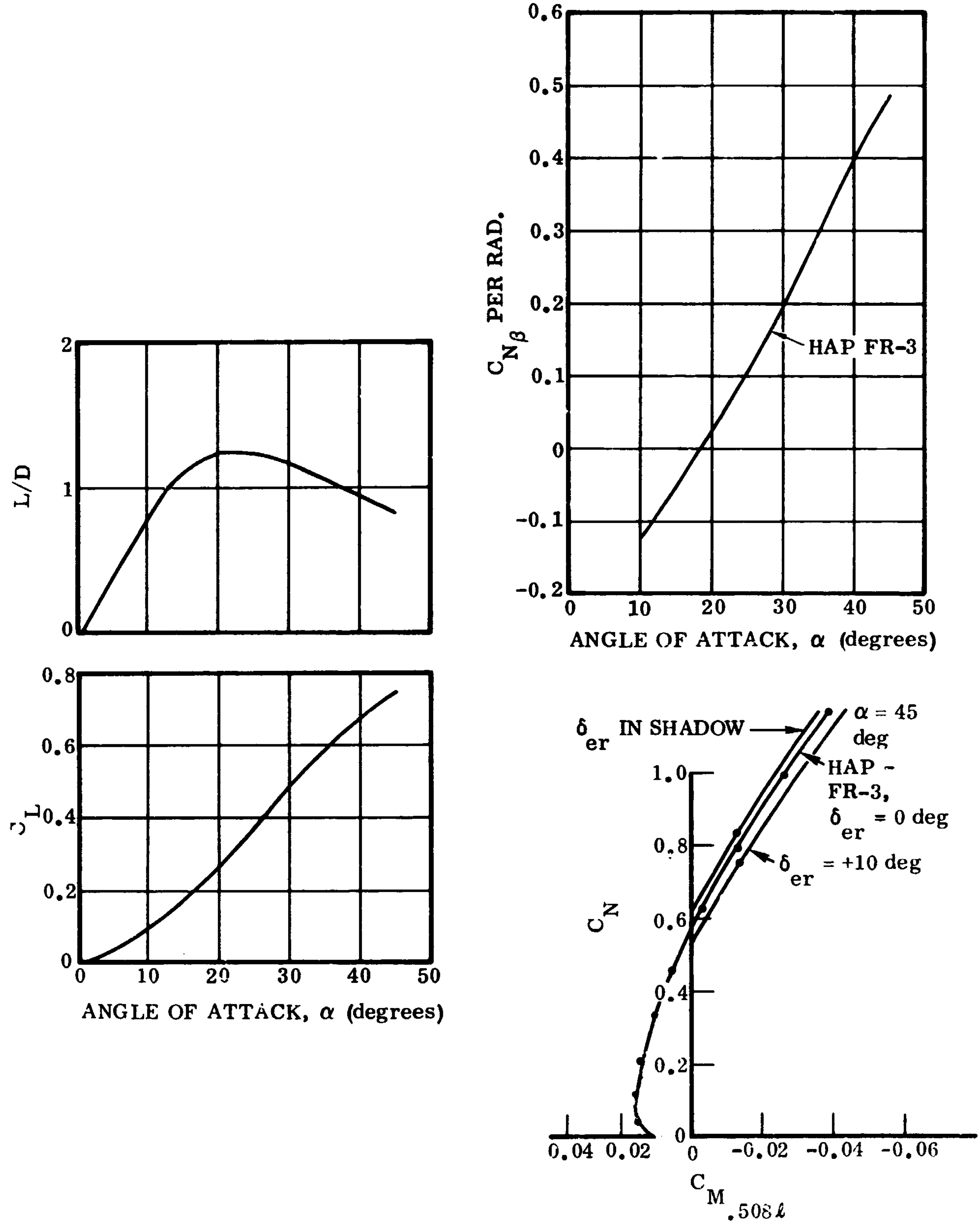


Figure 4-36. FR-3 Hypersonic Characteristics

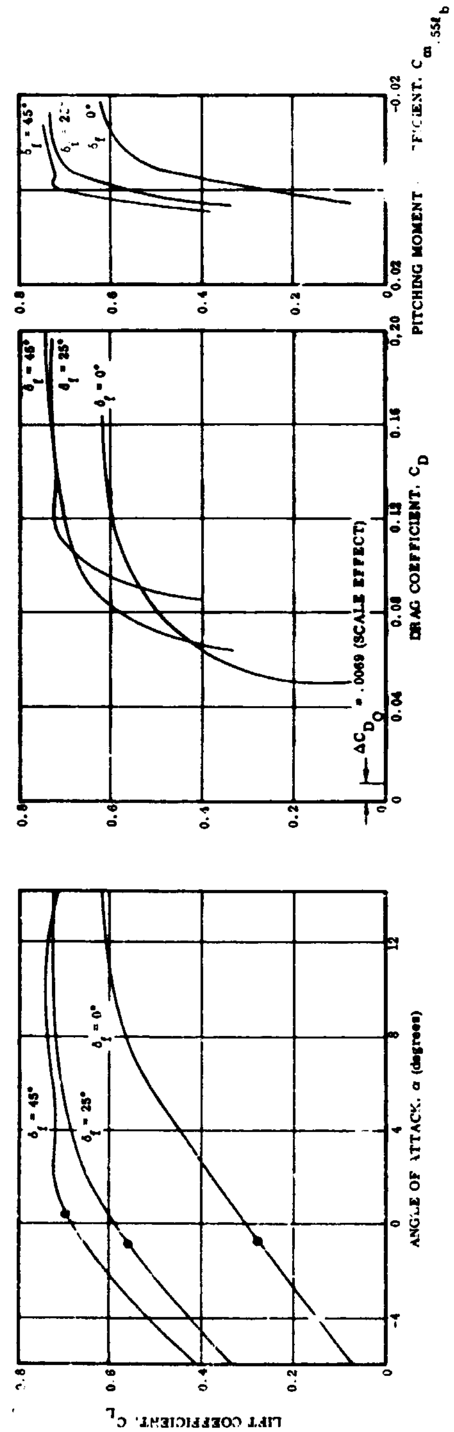


Figure 4-37. FR-4 Subsonic Longitudi.al Aerodynamic Characteristics (With Flaps)

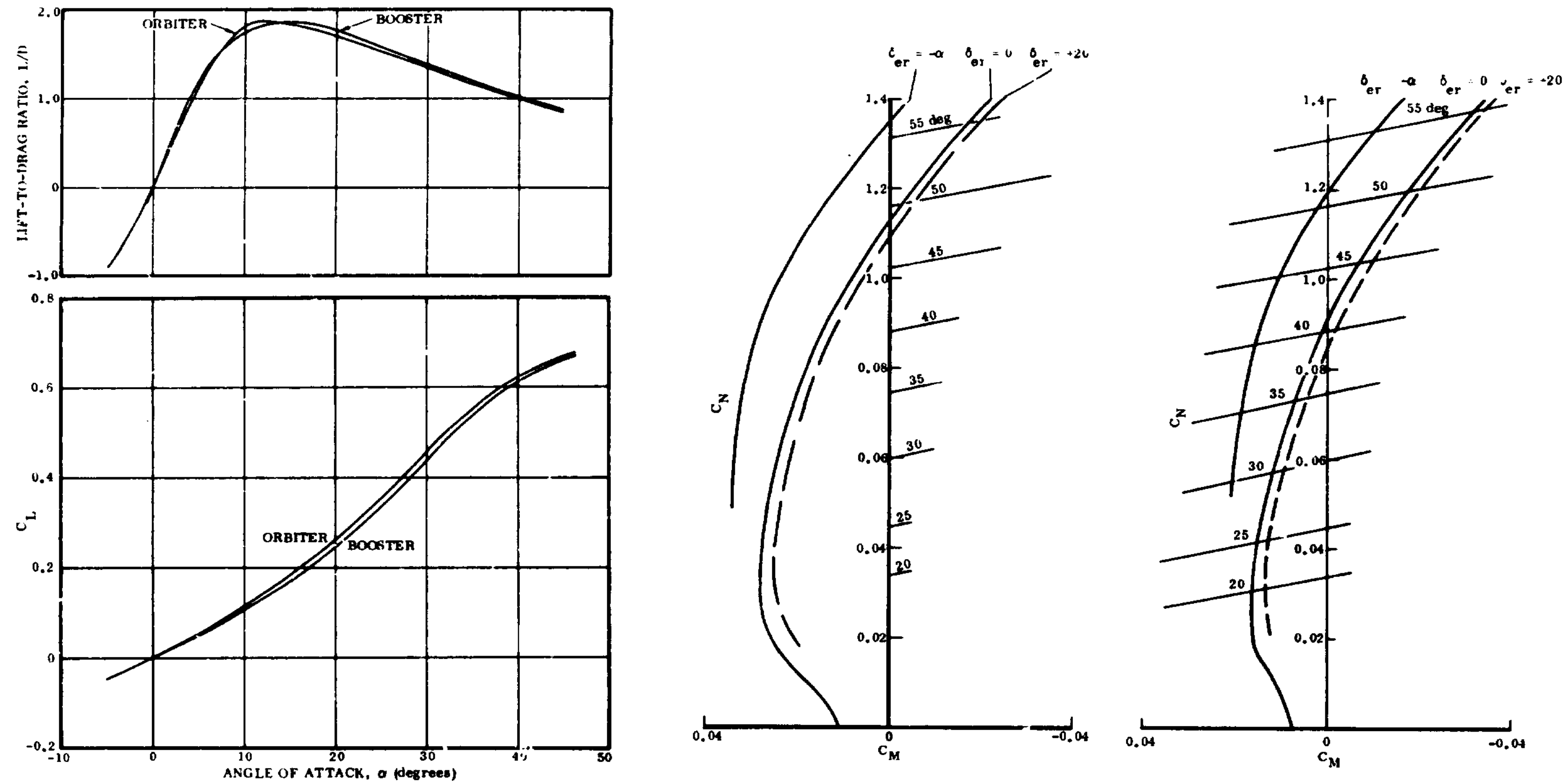


Figure 4-38. FR-4 Hypersonic Characteristics

4.5 PROPULSION DESIGN FOR FINAL VEHICLES

4.5.1 FINAL FR-3 VEHICLE. The booster and orbiter elements of the FR-3 have three propulsion subsystems, i.e., main propulsion, the attitude control propulsion subsystem (ACPS), and airbreathing (flyback) propulsion, which are described in Sections 4.5.1.1, 4.5.1.2, and 4.5.1.3 respectively.

4.5.1.1 Main Propulsion. Main propulsion for the FR-3 is provided by high Pc hydrogen-oxygen bell-nozzle engines. The main engines operate in a pump fed mode during boost and on orbit to provide orbit maneuvers.

Propellants are supplied to the engines by insulated propellant lines from internally insulated tanks. The tanks are pressurized with warm gas to minimize residuals, using gaseous hydrogen for the liquid hydrogen tanks for the booster and orbiter. High pressure helium is used to pressurize the oxygen tank in the booster, and gaseous oxygen is used to pressurize the orbiter liquid oxygen tank.

Booster engine area ratios are 35/80, operating retracted at an area ratio of 35 to an altitude of 30,000 feet, where all nozzles are extended to take advantage of the higher area ratio. The orbiter engine has an area ratio of 35/160, the higher area ratio being utilized to provide the higher I_{sp} desired by the upper stage. Very recent studies indicate that expansion ratios greater than 160 (e.g., 200) will result in increased payload.

A base nozzle contour defined by Pratt and Whitney is used. Using this contour instead of the maximum performance contour increases launch weight less than 0.3 percent, yet decreases engine length by three feet. The shorter nozzle is easier to protect from aerodynamic loads during ascent and from heat during entry. Engine performance for the FR-3 vehicle is summarized in Table 4-17.

The nominal engine mixture ratio is 6.5. This mixture ratio selection was based on a tradeoff considering the lower structural weight of higher mixture ratios and the higher specific impulse and therefore lower propellant weight at lower mixture ratios. During ascent, the engine mixture may be controlled to minimize residual (i.e., propellant utilization). A mixture ratio range of ±0.35 around the nominal of 6.5 is sufficient for control. This results in operation of the engine well within the engine MR operating constraints of 6 and 7.

During the initial phase of ascent, the fifteen booster engines operate at maximum nominal thrust providing a liftoff F/W of 1.387. Should an engine become inoperative (outside specification limits) the other engine will be operated at 108 percent of maximum nominal thrust. This rating provides maximum payload consistent with propellant utilization control range requirements. When the vehicle reaches a

Table 4-17. FR-3 Engine Performance

	Booster	Orbiter
Number of Engines	15	3
Sea Level Thrust (lb)	400,000	-
Vacuum Thrust (lb)	462,000	472,000
Specific Impulse, Sea Level (sec)	389	-
Vacuum (sec)	449.5	459
Area Ratio, Sea Level	35	35
Vacuum	80	160
Mixture Ratio	6.5	6.5
LH_2-NPSH 100%F	60	60
20%F	0	0
LO_2-NPSH 100%F	16	16
20%F	5	5
Engine Weight	4400	4600
Type Nozzle	Base	Base

maximum acceleration of 3g the engines are throttled to maintain 3g. Engine throttling is initiated at approximately 100 seconds after liftoff. Approximately 5 seconds prior to shutdown, the engines are throttled from approximately 60 percent thrust down to 10 percent thrust to minimize residuals. The booster stages at approximately 190 seconds after liftoff.

A few seconds prior to booster engine shutdown the three orbiter engines are started. The orbiter engine start sequence results in the orbiter vehicle achieving full thrust nominally at stage separation. The orbiter thrust to weight ratio after stage separation ,52. The orbiter accelerates until it reaches 3 g at approximately 330 seconds. The orbiter begins to throttle during the remainder of the orbiter solo phase. The orbiter is throttled to 70% by 390 seconds, just prior to propellant depletion. Within the last five seconds prior to shutdown the engine is further throttled to 10% to achieve minimum residuals.

After shutdown, the orbiter coasts from 40 n. mi. to 100 n. mi. During this time liquid residuals tend to keep the pump cool. When the orbiter reaches 100 n. mi. the first orbit maneuver is accomplished to circularize the orbit.

For orbit maneuvers, one orbiter engine is used in a pump-fed mode at 10 percent of the maximum throttle setting, i.e., 47,000-lb thrust. Propellants for the orbit maneuvers are provided to the engine from maneuver tanks through lines that are insulated to minimize boiloff.

Before the first orbit-maneuver firing, the oxygen-hydrogen ACPS thrusters in the rear of the vehicle are fired for approximately two minutes, providing a thrust of 10,000 pounds to settle the propellant, i.e., to provide liquid to the engine pumps. At the initiation of settling the prevalve of the secondary (maneuver) propellant subsystem is opened, allowing the propellants to flow to the main engine through the secondary lines.

During settling for the circularization maneuver at 100 n. mi., a ΔV of approximately 75 fps is provided by the ACPS. The main engine therefore provides only 37 fps for the first maneuver. The main engine burns approximately 10 seconds at 10% thrust to provide this ΔV.

After the first orbit maneuver, gaseous propellant is fed to a port downstream of the pump discharge. The gas forces liquid in the pump and secondary propellant line back into the tank. After all the liquid is pushed into the maneuver tank a prevalve is closed, maintaining the liquid in the highly insulated tanks.

After circularization, the orbiter may be required to remain in the 100 n. mi. orbit for up to 18 hours to permit phasing. During this time the engines and their pumps will receive heat from the sun, earth, and vehicle, causing the pump temperatures to increase.

After the coast phase a second orbit maneuver is required to transfer the vehicle from 100 n. mi. to 260 n. mi. The second maneuver starts by operating the ACPS to settle the propellant in the maneuver propellant lines. ACPS settling is approximately three minutes prior to main engine start.

Two minutes prior to main engine start, chilldown flow through the main engine pumps cools the pumps to near liquid-propellant temperature. Approximately 420 pounds of oxygen and 170 pounds of hydrogen are used to chill down the pumps. Main engines are started during the last five seconds of the attitude control settling phase. The main engine maneuver is accomplished in approximately 100 seconds. After main engine shutdown, the propellant is again pressurized out of the secondary line back into the maneuver tanks.

The third and fourth maneuvers, i.e., the maneuver to circularize at 260 n. mi. and the maneuver to deorbit the vehicle, are accomplished in a manner similar to the second maneuver. Main engine operating times for these maneuvers are 110 and 130 seconds, respectively.

4.5.1.1.1 Booster Propellant Feed. The propellant feed subsystem consists of all propellant ducting, pressurization subsystems, and venting subsystems required to condition propellants and supply them to the booster and orbiter engines, both for boost and orbit-maneuver firings. Figure 4-39 is a detail layout of the FR-3 booster design.

4.5.1.1.1.1 Liquid Oxygen Subsystem. The LO_2 tanks are located forward so that the vehicle is aerodynamically stable during ascent. Long main LO_2 lines are required to convey LO_2 to the engine manifold ducting. LO_2 is withdrawn from a common central sump under a flat-plate baffle which prevents vortexing and minimizes residuals at depletion. Multiple main LO_2 lines are used to reduce the magnitude of vehicle/engine development (integrated testing) and to simplify manifolding.

For the 16-engine design, five main LO_2 supply lines were selected. Prior to the selection, several different line configurations were examined, including 16 separate supply lines both internal and external to the LH_2 tank, five supply lines, and four supply lines. The designs were evaluated on the basis of minimizing line propellant residuals (which meant reducing or eliminating the horizontal line length at the aft end of the vehicle) and allowing engine throttling or shutdown of the outboard engines prior to termination of booster burn. The minimum-residual requirement eliminated propellant lines crossing the vehicle centerline. Structural clearance limitations allowed a 30-in. diameter line to be brought down both sides from the tank. These were large enough to supply all nine engines on the lower half. Single 14-in. diameter lines brought through the same space would supply only six engines.

Toward the end of booster burn, the lateral cg shifts toward the orbiter vehicle and it is desired to shift the thrust line by throttling outboard engines to minimize gimbal angles. This is accomplished by providing three lines for the upper seven engines. The four outboard engines are throttled to shut down completely and the remaining three center engines throttled as required, maintaining maximum allowable vehicle acceleration, with the lower nine engines running at full thrust. Thermal protection of the three external lines is provided. For a 15-engine design, these three lines would be combined into two lines.

Individual propellant lines to each engine are desirable from a development testing viewpoint if all the lines are dynamically similar. This would permit the bulk of propellant feed subsystem development testing to be performed with one engine on one set of ducting. This similarly may be obtained if the lines are routed through the LH_2 tank. There are problems inherent in such an approach, including heat transfer, leak detection, maintenance, and reliability that warrant further study. Because such a study has not been made, this approach was not taken for the final study vehicles.

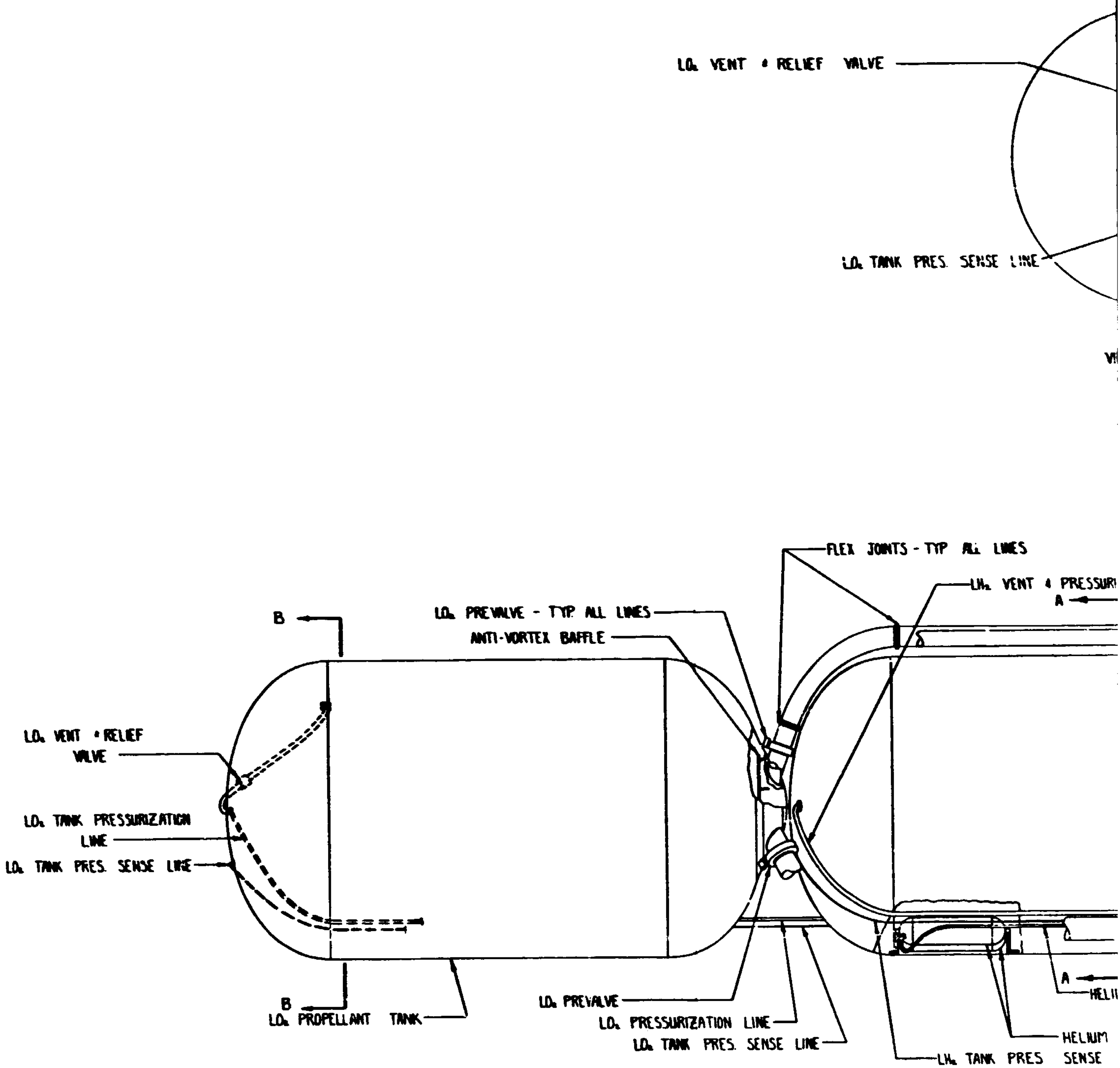
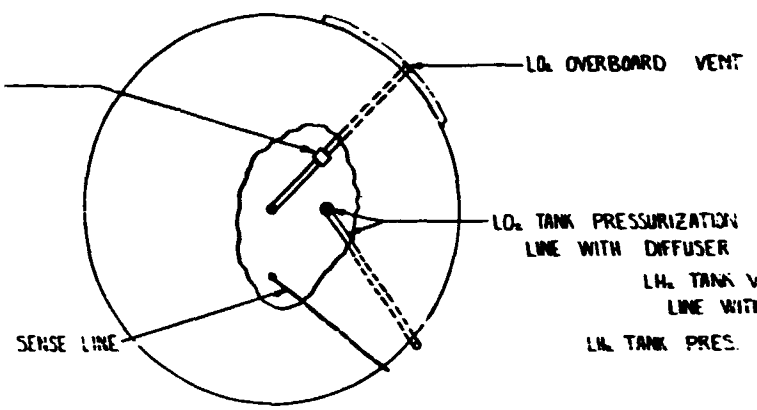


Figure 4-39. FR-3 Booster Propulsion Installation



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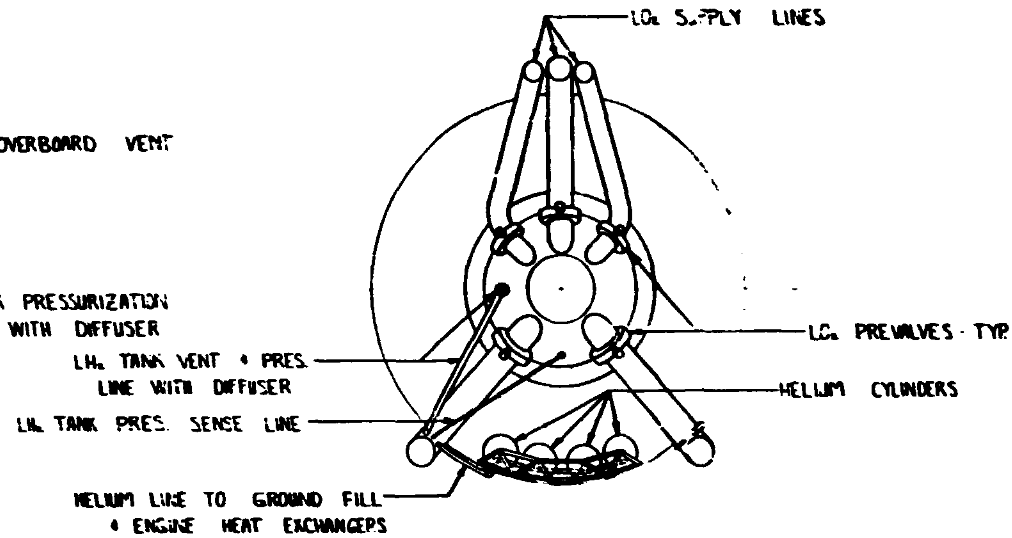
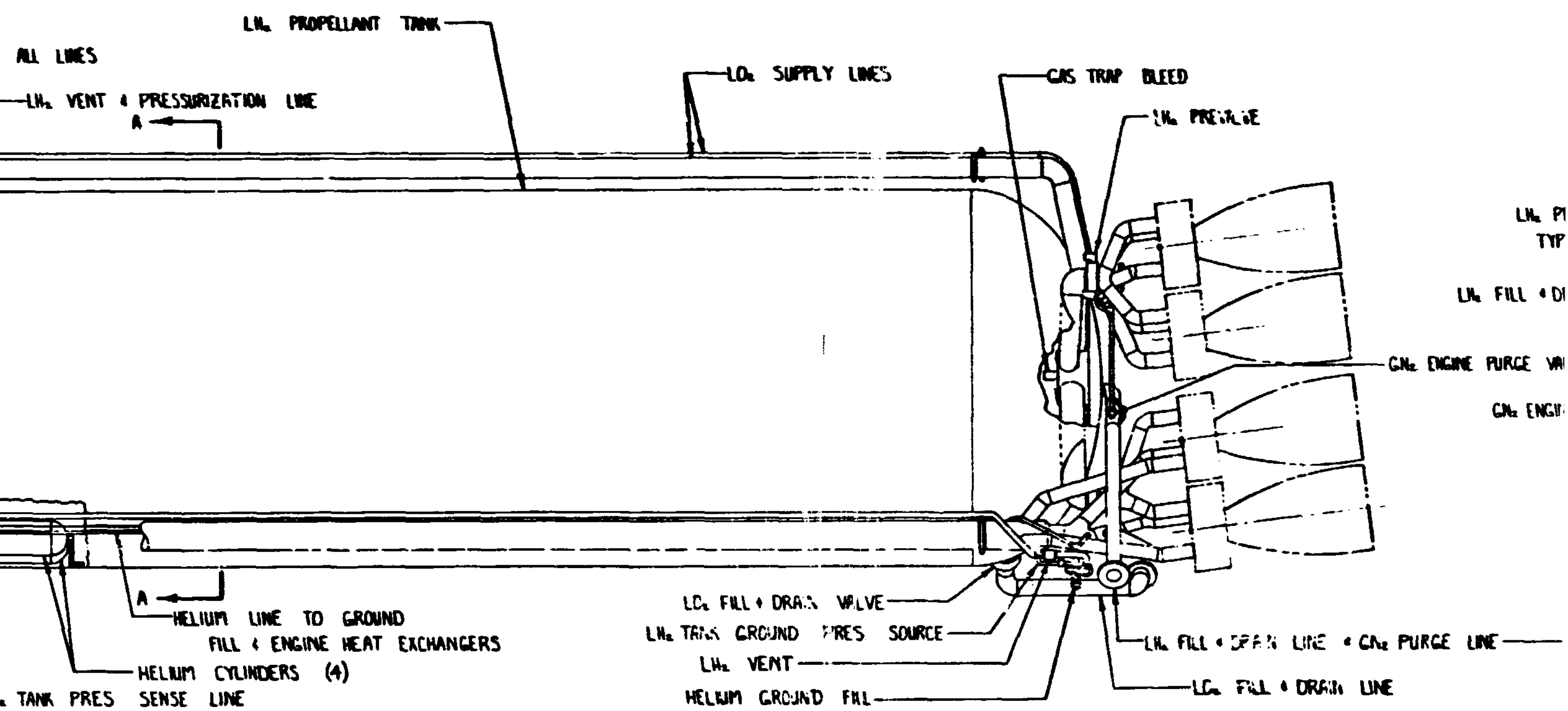
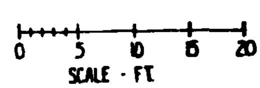
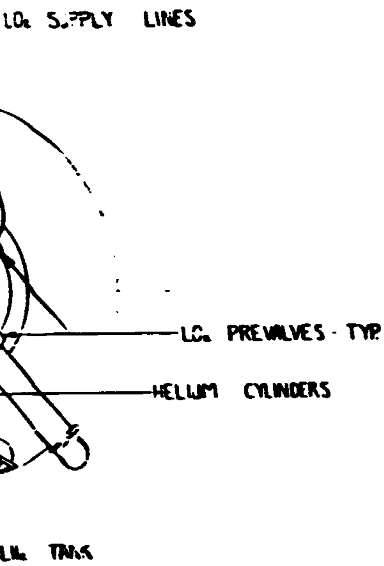


VIEW AA WITH LH₂ TANK REMOVED

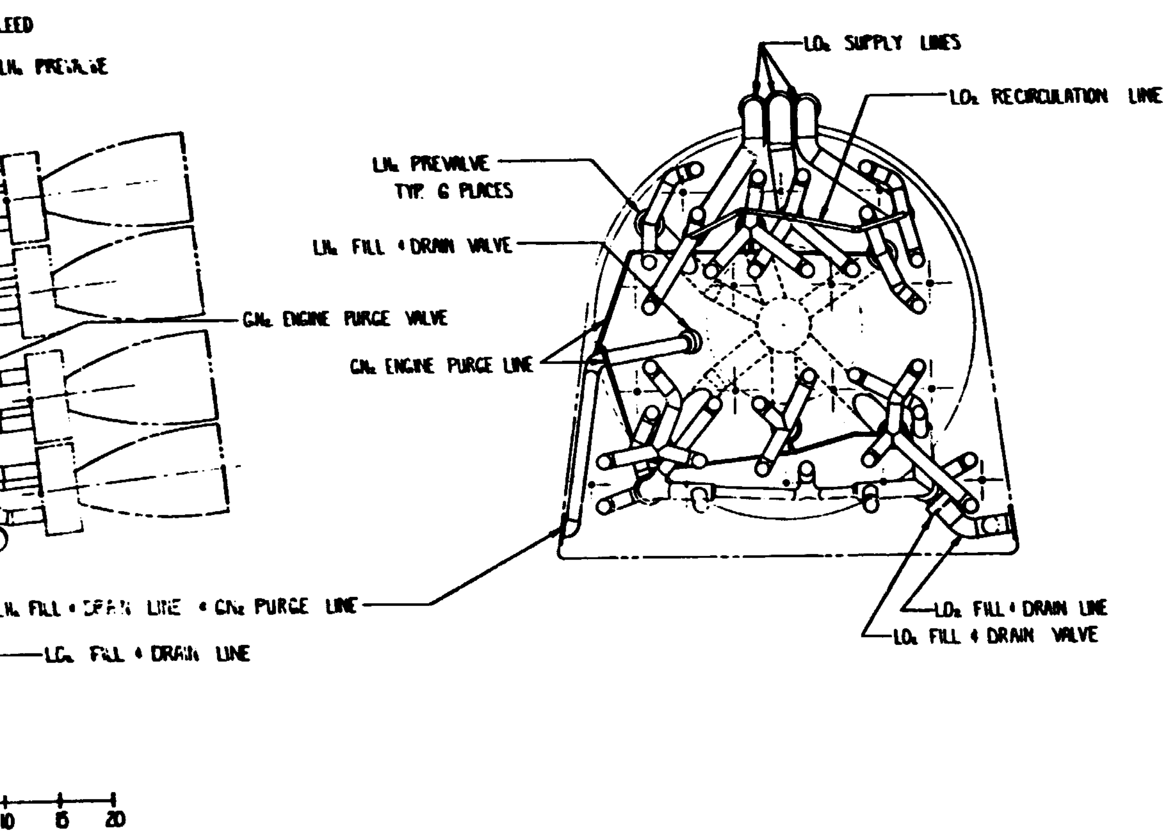




FOLDOUT FRAME 2



LO_2 SUPPLY LINES
LO_2 PREVALVES - TYP
HELIUM CYLINDERS
LH_2 TANK



LEED
LH_2 PREVALVE
LH_2 FILL & DRAIN LINE & GN_2 PURGE LINE
LO_2 FILL & DRAIN LINE
LO_2 SUPPLY LINES
LO_2 RECIRCULATION LINE
LH_2 PREVALVE
TYP. 6 PLACES
LH_2 FILL & DRAIN VALVE
GN_2 ENGINE PURGE VALVE
GN_2 ENGINE PURGE LINE
LO_2 FILL & DRAIN LINE
LO_2 FILL & DRAIN VALVE
10 15 20
FT

Bellows in the LO_2 ducting provide for required motion due to thermal and structural effects. Multi-ply double-wall bellows are used to reduce cycle failure due to flow-induced vibration and to provide safety should one wall fail. Duct prevalves provide for servicing and checkout, and pressure-relief check valves provide insurance against duct overpressurization failure during engine checkout.

A system integral with the engine (Figure 4-40) provides the required gimbal motion.

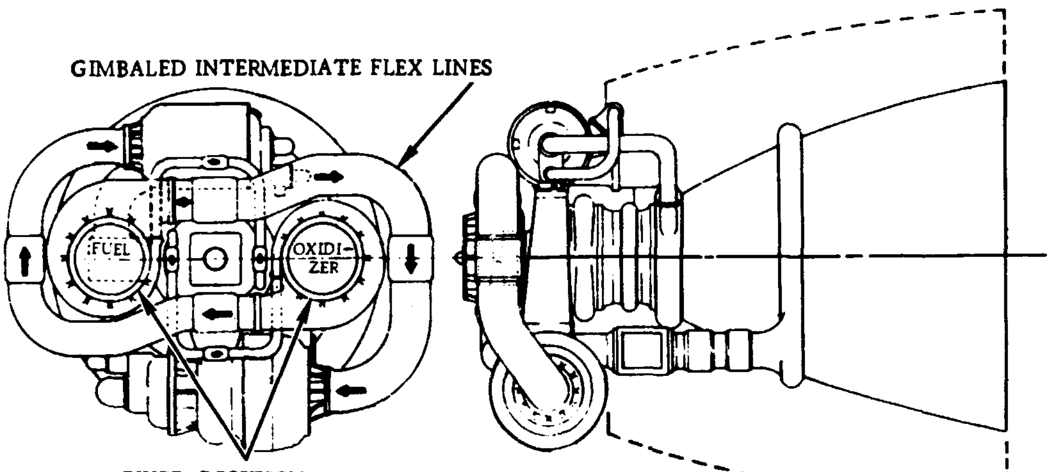


Figure 4-40. Engine Gimbal Provisions

The low-speed inducers (pump inlets) are fixed in relation to the vehicle structure. Intermediate pressure lines between the low-speed inducer and the second-stage pumps (fixed to thrust chamber) are provided with flex joints.

Overheating and geysering of the LO_2 in the long longitudinal (vertical) feed lines is prevented by utilizing natural thermal recirculation of LO_2 through the multiple vertical duct system. Recirculation lines connect appropriate lines at the manifold inlets to provide the recirculation path.

During engine start, the engine pumps accelerate rapidly, incurring additional (inertia) pressure losses above the nominal design friction losses. With the lines sized so that flow velocity throughout is equal to the engine pump inlet flow velocity, the required 16 ft NPSH is satisfied throughout the start transient, without special provisions (added tank pressure or, alternately, propellant subcooling).

With the LO_2 tanks placed forward, the LO_2 in the main longitudinal ducting must be used to achieve low residuals (see Figure 4-41). Following tank depletion, engine cutoff is initiated when the liquid level in the duct falls to the minimum LO_2 NPSH

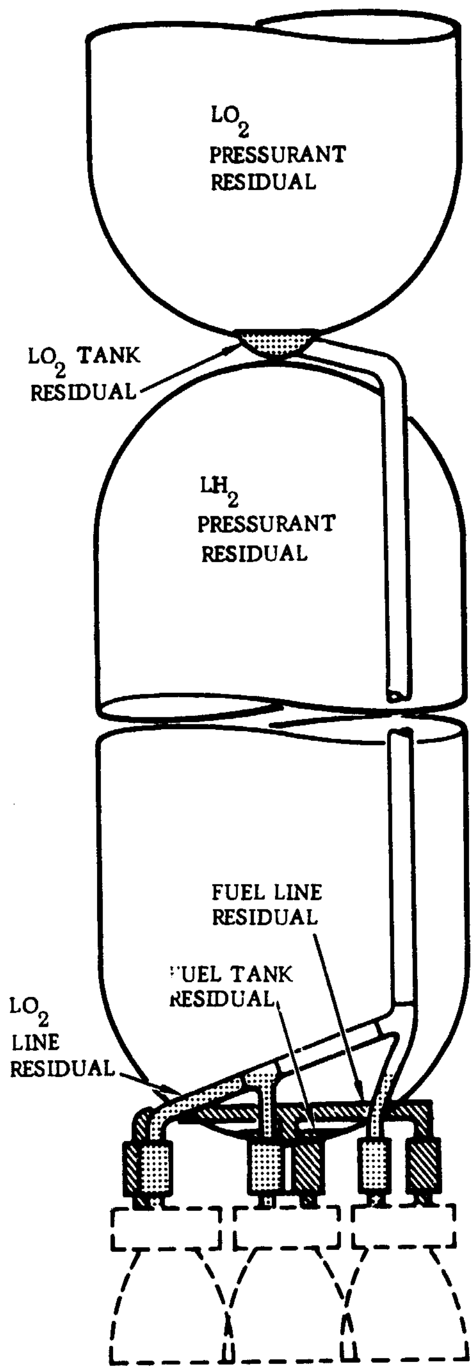


Figure 4-41. Residuals

(before gas entrainment into the engine). Throttling to 20% is used to reduce the NPSH requirement and resulting residuals.

With the long LO_2 lines, pressure surges at engine shutdown can be quite high for LO_2 valve-closure times approaching the time required for a pressure wave to travel from the valve to the tank and back again. To limit surge pressures to values below the design proof pressure of the ducting, valve closure times of 1.5 seconds or longer are required.

The tank is pressurized to prevent two-phase flow into the tank outlet and to minimize development of a large stratified or boiling layer at the top of the tank. The hydrostatic head in the long propellant lines provides the required NPSH at the pump. Helium is used as pressurant to minimize unusable pressurant residual weight. The helium is stored at 1000 psia in bottles within the LH_2 tank, and is heated to 500°R in an engine-mounted heat exchanger before use. The estimated tank pressure schedule is shown in Figure 4-42.

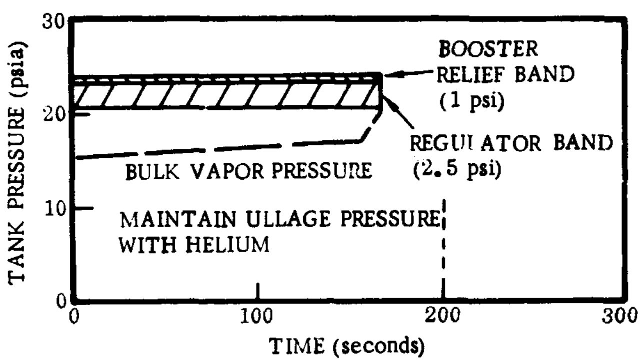


Figure 4-42. FR-3 LO_2 Tank Pressure Requirements

(U) The LO_2 tank is uninsulated but protected from wind and moisture condensation by a dry nitrogen purge in the space between the entry heat-shield and the tank. Prior to launch the tank is pressurized to about 20 psia from a ground helium supply.

(U) Filling and draining is accomplished through a connection to one of the main propellant feed ducts, as shown in Figure 4-39. Topping (replenishment of boiloff) is also accomplished through this duct until two minutes prior to engine start, at which time the fill-and-drain valve is closed and the tank is pressurized. The fill line below the fill-and-drain valve is drained of residual propellants before launch, and the fill-and-drain connection is retracted during liftoff. Chilldown gases from the ground fill lines are vented through an overboard facility vent. Gases generated during vehicle chilldown are vented through the vehicle boiloff/vent valve into the atmosphere.

4.5.1.1.1.2 Liquid Hydrogen (Fuel) Subsystem. Aft positioning of the fuel tank, resultant short lines, and relatively low operating pressures establish the design of the fuel system. Fuel is withdrawn from a central sump under a flat-plate baffle which prevents vortexing and minimizes residuals at depletion. Ducts are sized for pump-inlet flow velocities. Internal manifolding (inverted outlet duct) allows the engines to be mounted near the tank bottom, eliminates the need for insulating the portion of the ducting inside the tank, and simplifies routing of the LO_2 manifold ducting. Multiply double-wall bellows and double-wall external ducting are used to reduce life-cycle failure due to flow-induced vibration, to provide safety should one wall fail, and to prevent air liquifaction. Gimbaling flexibility is provided within the engine as shown in Figure 4-40. Duct prevalves provide for servicing and checkout, and pressure-relief check valves provide insurance against duct overpressurization failure during engine checkout.

Because of the short LH_2 lines, no geysering problem is expected. A bleed is provided for the high point (trap) in the inverted fuel outlet duct, preventing formation of a gas pocket during no-flow conditions.

The propellant will reach saturated conditions in the ducting during tanking. After pressurization for engine start, the added heating and resulting vapor pressure rise is small compared to the added tank ullage pressure of 24 psi.

The short-coupled fuel system with large lines has very low transient loss during start. The 60-ft NPSH requirement is satisfied by the engine prestart pressure requirement of 40 psia.

The short-coupled lines and high flow velocities require that residuals be determined at the time pull-through into the tank outlet is initiated in order to prevent gas entrainment into the engines. Shortly before tank depletion, the 60-ft NPSH requirement is satisfied by tank pressure. At tank depletion, the engines are shut down from 20% thrust and the liquid pull-through level is minimized by the outlet-baffle design, resulting in low residuals.

With valve closure time regulated to greater than 1.5 seconds for the LO_2 system, there is no expected pressure-surge problem in the fuel system.

Autogenous H_2 bleed pressurant at 300°R is provided by the main propulsion system to supply the necessary pump NPSH. A pressure of 26.3 psia is required prior to burnout. The estimated tank pressure scheduled is shown in Figure 4-43. The tank is insulated with either fiber-reinforced 3D foam (0.62 lb/ft^2) or open-cell insulation (0.78 lb/ft^2).

The tank will be pressurized prior to launch with ground helium to provide the required prestart pressure of 40 psia.

Filling and draining are accomplished through a duct into the bottom of the fuel tank, separate from the engine fuel-inlet ducts. Operation is similar to the LO_2 system previously described, except that all chilldown gases are vented through the vehicle boiloff/vent valve into the facility vent stack for disposal. A ground helium purge supply is also provided prior to tanking and in case of launch abort.

4.5.1.1.2 Orbiter Propellant Feed. Figure 4-44 is a detail layout of the FR-3 orbiter design, which is similar to the FR-3 booster just described, except as specifically discussed in the following paragraphs.

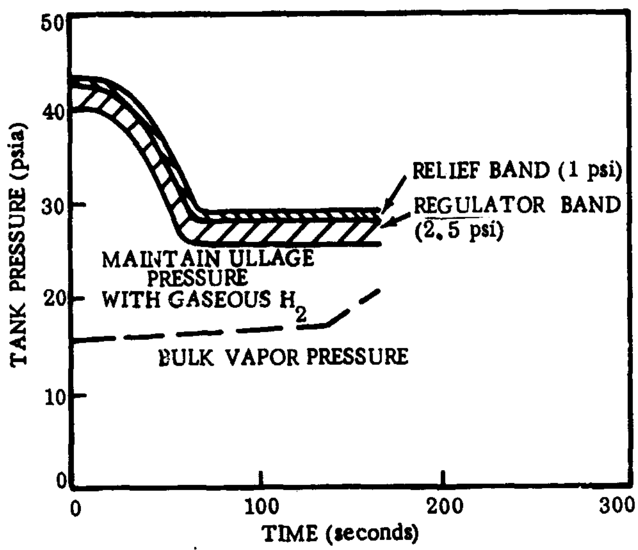


Figure 4-43. FR-3 LH_2 Tank Pressure Requirements Booster

4.5.1.1.2.1 Liquid Oxygen Subsystem. An inverted LO_2 tank outlet line is used to prevent intrusion of ducts into the payload area. An increase in tank pressure of 2-3 psi (equal to the liquid head in the inverted duct above the tank liquid level, times the vehicle acceleration, plus the baffle and line pressure drop) is required to prevent cavitation in the outlet system. The tradeoff of this added tank and pressurant weight versus the effect of a longer vehicle will determine the final selection of inverted vs. conventional (down inlet) design. This tradeoff has not yet been performed.

Dual ducts are provided from the common sump/baffle. One line is sized for nominal flow for the three engines. The other line is sized for orbit maneuver propulsion operation (one engine at 20%) and connects to both the main tank (thus providing a prestart recirculation flow path) and the orbit maneuver tank. Isolation valves are provided for orbit maneuver phase.

The orbiter engines are started before booster cutoff, while sufficient hydrostatic head is available to meet the NPSH requirement of 16 ft during the start transient.

Autogenous O_2 bleed from the engine is used since the pressurant residual is used for ACS propellant. The tank is pressurized to about 20 psia with ground helium prior to launch and is vented as required during the booster phase to maintain 20 psia. The estimated tank pressure schedule is shown in Figure 4-45. After burnout, the tanks are maintained at 25 psia. Residuals and boiloff are used to pressurize the maneuvering tanks and to operate the ACS engines.

4.5.1.1.2.2 Liquid Hydrogen (Fuel) Subsystem. The fuel tank remains at 40 psia during boost phase in order to provide the required prestart pressure (Figure 4-46).

Separate, highly insulated tanks located in the payload bay provide propellant for the four orbit-maneuver firings of the main engines. The tanks are sized for approximately 39,000 lb of propellant at MR = 5:1. The orbit-maneuver tanks are connected to the main propellant manifolds by separate lines sized to provide flow for 20% thrust of one engine. This sizing is predicated on the use of orbit-maneuver propellants to provide additional velocity required for once-around abort with one orbiter engine out. During orbit-maneuver firings, maximum engine thrust is 10% of nominal (one engine only). Between each firing, residual line propellants are returned to the tanks to

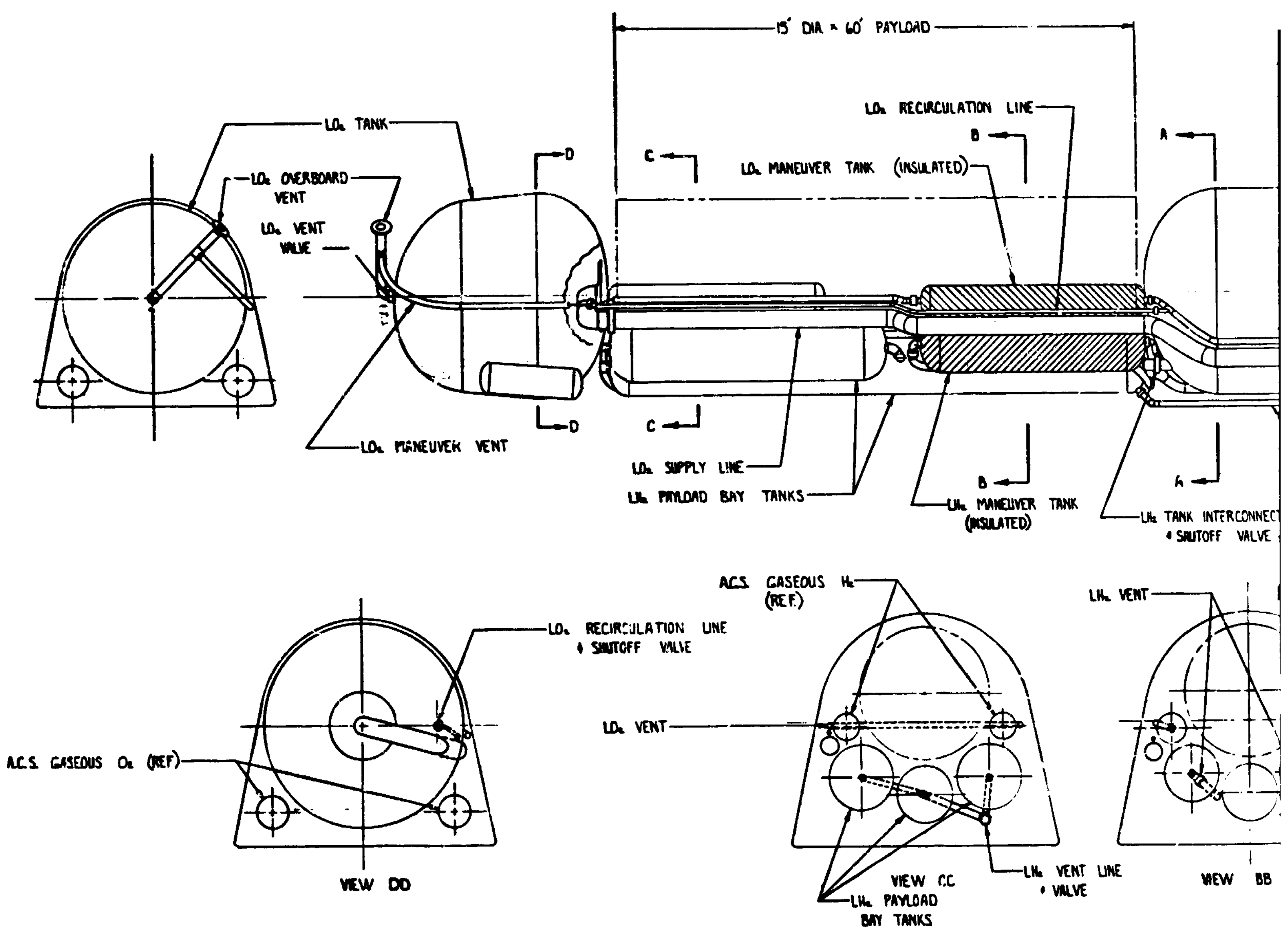
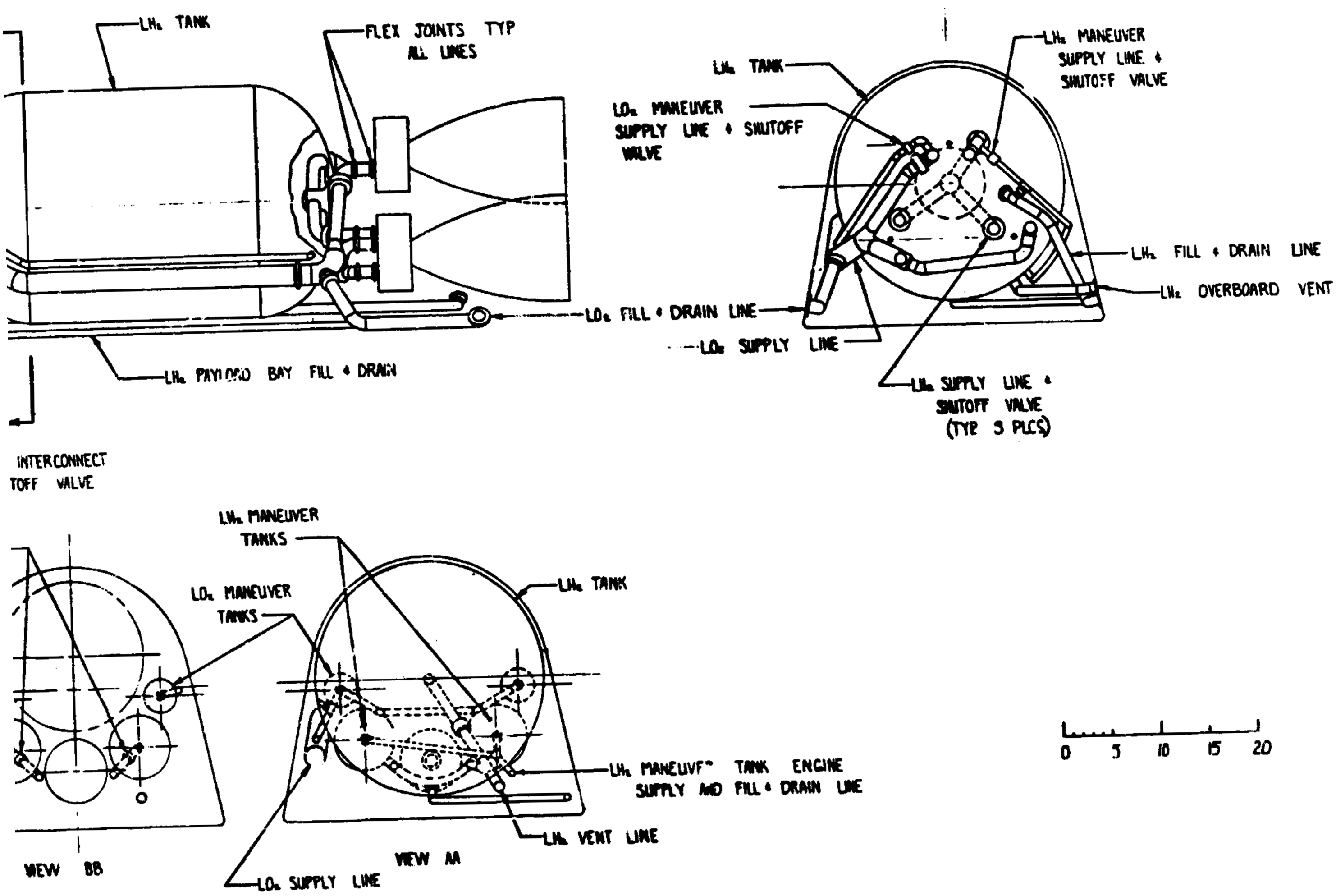


Figure 4-44. FR-3 Orbiter Propulsion Installation

FOLDOUT FRAME 1



FOLDOUT FRAME 2

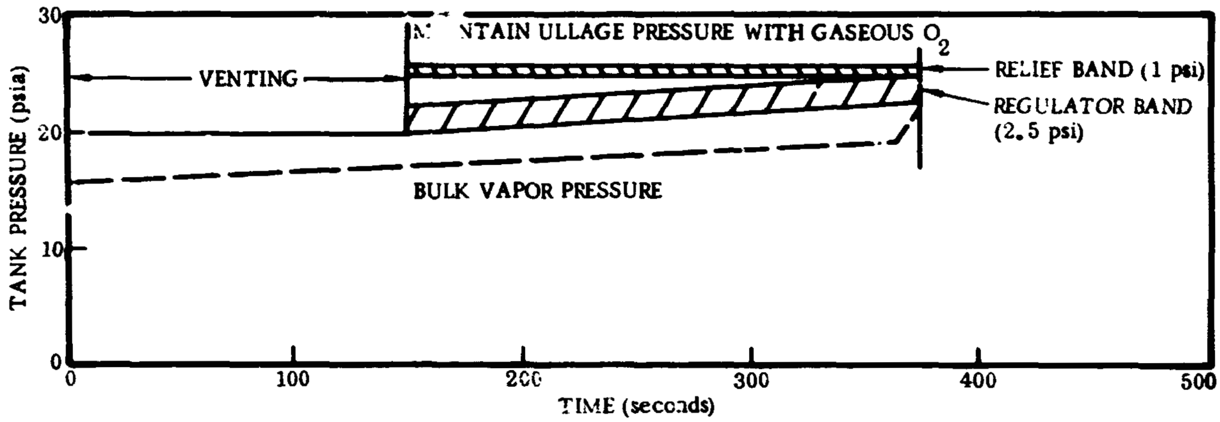


Figure 4-45. FR-3 Orbiter LO_2 Tank Pressure Requirements

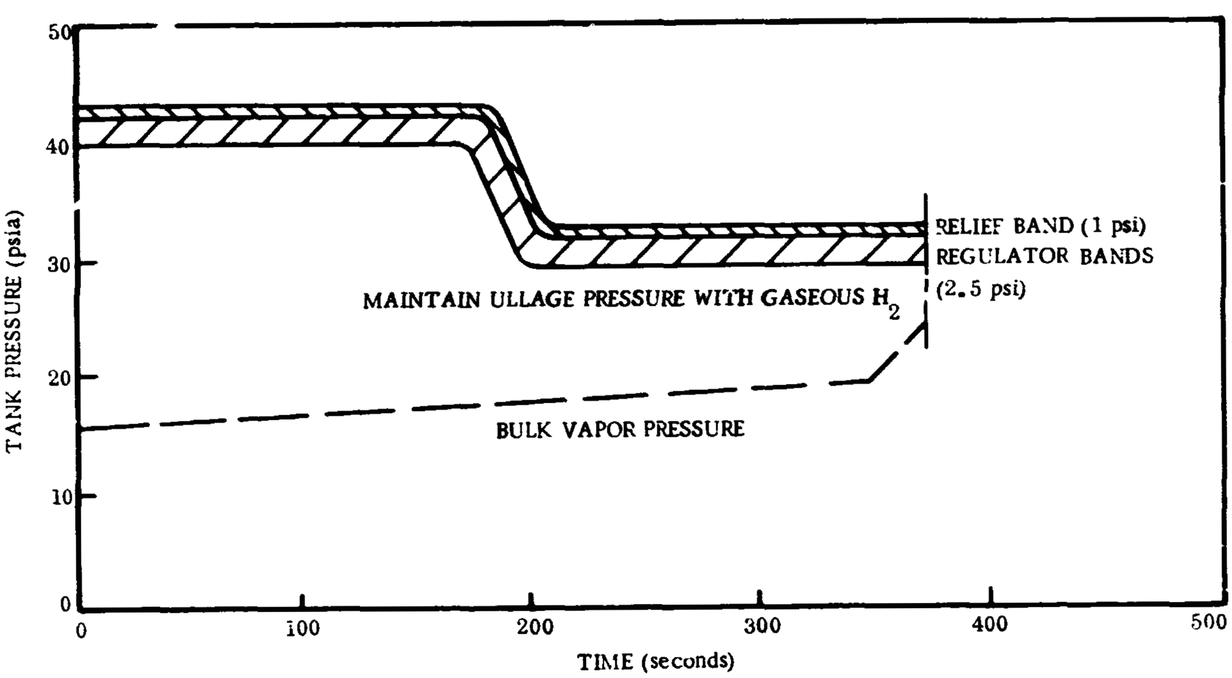


Figure 4-46. FR-3 Orbiter LH_2 Tank Pressure Requirements

eliminate propellant boiloff loss which would be incurred if the main engine were kept wet (heat soak-back and solar input). Attitude control propulsion combined with helium pressurization is used ' empty the lines of propellants. Tank pressurization is provided from the main tanks. A zero-g vent on the hydrogen maneuver tank is provided, with the ventage routed to cool the LO_2 maneuver tank to prevent LO_2 boiloff.

The LO_2 maneuver tanks were located in the aft portion of the payload bay to utilize existing available space. It is desirable to minimize the distance between the orbit maneuver tanks and the engines. By locating the engines further aft relative to the main tank, it may be possible to close-couple the tanks to the engines, thus reducing line and insulation weight, cooldown and boiloff propellant losses, and start transient effects. Moving the engines aft adds structural weight and moves the vehicle cg further aft.

4.5.1.2 Attitude Control. The attitude control propulsion subsystem consists of 48 attitude control thrusters with nominal thrusts of 2500 pounds. The thrusters are supplied with gaseous oxygen and hydrogen from high-pressure accumulators, which are sized to provide the propellant required for entry control. These accumulators may be charged on the ground, with main-engine bleed gas, auxiliary pumps, or compressor. If compressors are used, residual gases and liquids may be used to minimize or eliminate the need for additional propellant for attitude control. The engines may be operated at low pressure directly from the main tanks during certain orbital phases where high thrusts are not required and low impulse bits are desired.

4.5.1.3 Airbreathing (Flyback) Engines. The FR-3 space shuttle vehicle has airbreathing flyback engines in the booster element and the orbiter element. Booster engines will be deployed at an altitude of 25,000 feet or higher and windmill-started during glide to a normal cruise altitude of about 15,000 feet. The booster element will then fly back 250 to 300 n.mi. and land at the landing site. Orbiter engines will be deployed at an altitude of 15,000 feet or higher and windmill-started during glide toward the landing site. The engines will then operate at idle to an altitude of about 1500 feet when they will be brought to the thrust required for a powered approach and landing.

Two final FR-3 vehicle designs are being presented: one with a 15-ft-diameter payload bay and one with a 22-ft-diameter payload bay. For the final designs actual flyback engines were chosen and the airbreathing subsystem weights in the final synthesis computer runs reflect the installation of these engines.

The FR-3 is a two-element vehicle with the booster element having a flyback weight approximately 60 percent heavier than the FR-4 boosters, and the orbiter element having a flyback weight about 10 percent lighter than the FR-4 orbiter.

The booster element requires more and larger engines than the FR-4 booster while the size of the 40,000-pound high-bypass turbofan engines creates installation problems in the orbiter element.

Preliminary analysis of the booster element indicated a maximum sea level static thrust (MSLST) requirement of greater than 200,000 pounds. The 40,000-pound class candidate engines listed in Table 4-1 have growth versions in the 50,000-pound class that are projected for manufacture. A four-engine installation of one of these growth version engines appears the most reasonable.

The Rolls Royce RB211-56 engine was selected for the booster final design. It has a 52,500-lb MSLST. The installed thrust-to-weight ratio was assumed to be the same as for the RB211-22. The engine performance used to evaluate the flyback capabilities of the final FR-3 designs is the RB211-56 specification. Ten percent intake recovery and power takeoff losses were assumed. FR-3 final synthesis computer run parameters used in evaluating the booster flyback performance are:

	15-ft-dia Payload	22-ft-dia Payload
Flyback weight, lb	564,378	585,927
Flyback range, n.mi.	281	271
Planform area, ft^2	8,170	8,430
Max L/D	7.2	7.2
C_L at Max L/D	0.45	0.45

The flyback conditions, performance requirements, and capabilities of the FR-3 boosters under normal conditions are:

	15-ft-dia Payload	22-ft-dia Payload
Cruise Altitude, ft	15,000	15,000
Cruise Velocity, knots	269	269
Required Cruise Thrust, lb	78,386	81,379
Available Thrust at Max Cruise Rating, lb	79,625	79,625
Available Thrust at Emergency Max Cont., lb	85,586	85,586

Thrust available from the four turbofan engines at maximum cruise rating meets the requirements of the 15-ft-diameter payload vehicle but is not sufficient for the 22-ft-diameter payload vehicle. It will be necessary, therefore, for the 22-ft-diameter payload vehicle to fly at a slightly lower altitude or for the engines to operate between the maximum cruise rating and the emergency maximum continuous rate.

The booster is required to have the capability to fly back to the landing site with one engine inoperative. Comparisons of final performance and synthesis run values are:

Engine Out Conditions	15-ft-dia Payload	22-ft-dia Payload
Cruise Altitude, ft	7,900	6,400
Cruise Velocity, knots	239	234
Synthesis Input Velocity, knots	243	243
Required Fuel, lb	47,169	47,787
Synthesis Run Fuel, lb	46,916	47,035
Fuel Shortage, lb	253	752

The fuel available in the synthesis run is slightly less than that required to fly back. However, there is 9,000 pounds of ballast provided in the nose of both configurations for balance during hypersonic entry. This is not required during subsonic cruise, so a portion of the ballast weight could be converted to flyback fuel to provide the required fuel plus a reserve (in case of headwinds). This would neither impose a weight penalty nor compromise the stability of the vehicle.

Preliminary analysis of the orbiter element flyback requirements indicated a total MSLST requirement of about 60,000 pounds. The Pratt and Whitney TF33-P-7 turbofan engine has a maximum sea level static rating of 21,000 pounds, so three of these engines would match the requirements. This older engine has a bare thrust-to-weight ratio of about 4.5 and so would weigh more than an advanced turbofan engine of the same thrust rating but there are no advanced turbofan engines in this thrust range under development. An alternate selection would be two of the 40,000-pound thrust advanced engines. The installation weight would be approximately the same; however, the size of these larger engines would cause installation difficulties. The three-engine TF33-P-7 configuration was chosen for the final design. Synthesis computer run parameters and climb and go-around capabilities of the two FR-3 orbiter final designs are:

		15-ft-dia Payload	22-ft-dia Payload
Flyback weight, lb		289,655	319,207
Planform area, ft^2		4,910	5,416
Max L/D		7.8	7.8
C_L at Max L/D		0.55	0.55
Cruise velocity, knots		178	178
Required cruise thrust, lb		37,135	40,923
Max available thrust, lb		47,520	47,520
Climb capability:	Rate, ft/min	645	372
	Angle, deg	2.1	1.2

In the 15-ft-diameter payload FR-3 design the four 52,500-pound engines in the booster and three 21,000-pound engines in the orbiter provide satisfactory flyback and landing capability. In the 22-ft-diameter payload FR-3 design, the performance requirements are met but the capability is marginal.

The engines use JP-4 fuel for flyback and landing. Hydrogen is being considered as a replacement, however. Section 4.6 of Volume VI describes the potential weight saving to be gained by using hydrogen in the FR-3 flyback engines.

4.5.2 FR-4 FINAL SEQUENTIAL BURN VEHICLE. FR-4 propulsion subsystems and their modes of operation are similar to the ones in the FR-3.

4.5.2.1 Main Propulsion. Each booster element in the FR-4 has nine engines instead of the 15 engines of the FR-3. The 400K-lb-thrust engines are identical to the engines used in the FR-3. The FR-4 orbiter propellant feed subsystem is essentially identical to the FR-3 orbiter just described.

The FR-4 booster (two elements) is similar to the FR-3 booster except for fewer engines on each element, resulting in simpler propellant feed design. No detailed design layout drawings were made.

4.5.2.2 Attitude Control. The attitude control propulsion subsystem consists of forty-eight 3500-pound thrusters with associated propellant subsystems.

4.5.2.3 Airbreathing (Flyback) Engines. As with the FR-3 vehicle, two final FR-4 vehicle designs are being presented. One has a 15-ft diameter by 60-ft long payload bay. The other has a 22-ft by 60-ft payload bay. For these final designs an actual engine that will be available was chosen. Airbreathing subsystem weights in the final synthesis computer runs reflect the installation of this engine.

Preliminary analysis of the FR-4 vehicle indicated that total maximum sea level static thrust required for the booster element to cruise back at 15,000 ft altitude is in the 120,000-pound range. Installation studies discussed in Section 4.5 concluded that a three-engine configuration is preferred if engines are available with thrusts that match requirements. Several of the advanced turbofan engines listed as candidate engines are in the 40,000-pound class. The Rolls Royce RB211-22 engine has a slightly better thrust-to-weight ratio than do the other engines and was therefore selected as the engine for the FR-4 final design. Figure 4-8 shows an installation arrangement of this engine. Bare engine weight of the RB211-22 is 6353 pounds. Installation weights including cowls, inlet duct, tailpipes, etc., were estimated to be 1293 pounds, resulting in a total weight per engine of 7646 pounds

The final airbreathing propulsion subsystem configuration for the FR-4 vehicle is three RB211-22 engines on each booster element and two RB211-22 engines on the orbiter element. Engine performance values used to evaluate flyback capabilities of the final FR-4 designs are from the RB211-22 specification. Estimated performance values listed in the specification are for 100% intake recovery and no offtake losses. To account for these losses, the values were decreased by 10% in making the evaluation. FR-4 final synthesis computer parameters used in the evaluation are:

	15-ft-dia Payload	22-ft-dia Payload
Booster Flyback Weight, lb	355,500	390,288
Orbiter Flyback Weight, lb	325,700	341,478
Booster Flyback Range, n.mi.	255	276
Booster Planform Area, ft^2	6,072	6,603
Orbiter Planform Area, ft^2	5,565	5,955
Max L/D	7.8	7.8
C_L at Max L/D	0.55	0.55

Flyback conditions, performance requirements, and capabilities of the FR-4 boosters are:

Normal Conditions	15-ft-dia Payload	22-ft-dia Payload
Cruise Altitude, ft	15,000	15,000
Cruise Velocity, knots	224	224
Required Cruise Thrust, lb	45,577	50,037
Available Thrust at Max Cruise, lb	51,127	51,127

Thrust available from the three turbofan engines at maximum cruise rating is greater than that required for both the 15-ft-diameter and the 22-ft-diameter payload vehicles. Therefore, the booster flyback engines can operate at partial power or the booster could cruise at a higher altitude and velocity if this is desirable.

The booster is required to have the capability to fly back to the landing site with one engine inoperative. Since in both designs the engines provide more than enough thrust normally to fly at 15,000 feet, the booster can fly with one engine out at a higher altitude and velocity than if the engines furnished just enough thrust under normal conditions. Comparisons of final performance and synthesis run values are:

Engine-Out Conditions	15-ft-dia Payload	22-ft-dia Payload
Cruise Altitude, ft	7,300	3,800
Cruise Velocity, knots	197	188
Synthesis Input Velocity, knots	178	178
Required Fuel, lb	28,543	35,061
Synthesis Run Fuel, lb	30,711	36,399
Reserve Fuel Available, lb	2,168	1,338

Some reserve fuel for booster flyback is available. Additional fuel would be needed if strong head winds were encountered. In both FR-4 designs, 11,000 pounds of ballast are required in the nose of the booster to provide the proper balance during hypersonic entry, but is not required during subsonic cruise. A portion of this ballast could be converted to flyback fuel for a reserve in case of headwinds, therefore, without either increasing the weight of the vehicle or compromising its stability.

The two turbofan engines on the orbiter element are to provide the thrust for a powered approach and landing with the additional capability to climb, go-around, and make a second approach and landing. Performance capabilities of the two FR-4 designs are:

	15-ft-dia Payload	22-ft-dia Payload
Cruise Velocity, knots	178	175
Required Cruise Thrust, lb	41,756	43,779
Available Max Takeoff Thrust, lb	56,585	56,765
Climb Capability		
Rate, ft/min	817	675
Angle, deg	2.6	2.2

In both FR-4 designs, the three 40,600-pound turbofan engines on the booster element provide more than enough thrust to meet the flyback requirements. The two engines on the orbiter element provide thrust for approach and landing as well as limited climb capability for go-around if necessary.

4.6 AEROTHERMODYNAMICS

The results presented here are for the final configurations studied. Volume III, Section 3.6 and Volume IV, Section 4 provide background leading to final configurations and aerothermodynamic methods.

4.6.1 ORBITER ELEMENTS. The FR-3 and FR-4 orbiter elements were considered to be identical. The peak temperatures and the required thermal protection system insulation thicknesses were determined by the once-around abort 800 n.mi. lateral range entry trajectory, Section 4.2.1 of Volume IV. Figure 4-47 presents the 800 n.mi. lateral range peak temperature distributions and Figure 4-48 presents the configuration with selected peak temperatures. Figure 4-49 presents the insulation distribution on the lower and upper surface for the once-around entry.

The proposed structural temperature control after landing was to supply cooling air from ground support equipment after landing. This cool air would pass the backface of the insulation, as discussed further in Section 4.8.1 of Volume IV.

The main propellant tank cryogenic insulation selected was an internal type. Two types were considered: one was an open cell honeycomb and the other was the S-IVB 3D foam, Section 4.6 of Volume IV. A dry nitrogen purge was assumed to eliminate moisture condensation. The orbital maneuvering propellant tanks were insulated with superinsulation to control boiloff for the seven-day mission requirement, Section 4.6.3 of Volume IV.

Analysis of the aerodynamic heating of the nozzles indicated no significant temperature problem (Section 4.7 of Volume IV). The peak launch temperature was calculated to be 1290°R (830°F) which for an emissivity of 0.8 corresponds to a heat transfer rate of 1.1 Btu/ft^2-sec. During entry a peak temperature of 1520°R (1060°F) was calculated. The above temperatures were calculated for the 400,000-lb-thrust level engines where the nozzles do not protrude outside the base. An earlier analysis indicated that a peak temperature of 2760°R (2300°F) could be experienced on the larger nozzles of higher thrust engines which extended aft of the lower surface and hence the lower nozzle was exposed to the flow.

Figure 4-50 presents the insulation thickness variation as a function of lateral range at several selected locations. These were used to establish the variation of total thermal protection system mass variation as a function of lateral range shown by Figure 4-51. The variation is flat for lateral ranges less than 400 n.mi. because the trim capability of the configuration did not permit reducing the lateral range to less than 400 n.mi. Hence in order to get a lateral range less than 400 n.mi. it was necessary to fly a path which tranversed 400 n.mi. and ended at the desired lateral range. The TPS weight could be reduced 2800 lb if the abort philosophy could be modified to require crossrange not in excess of 400 n.mi.

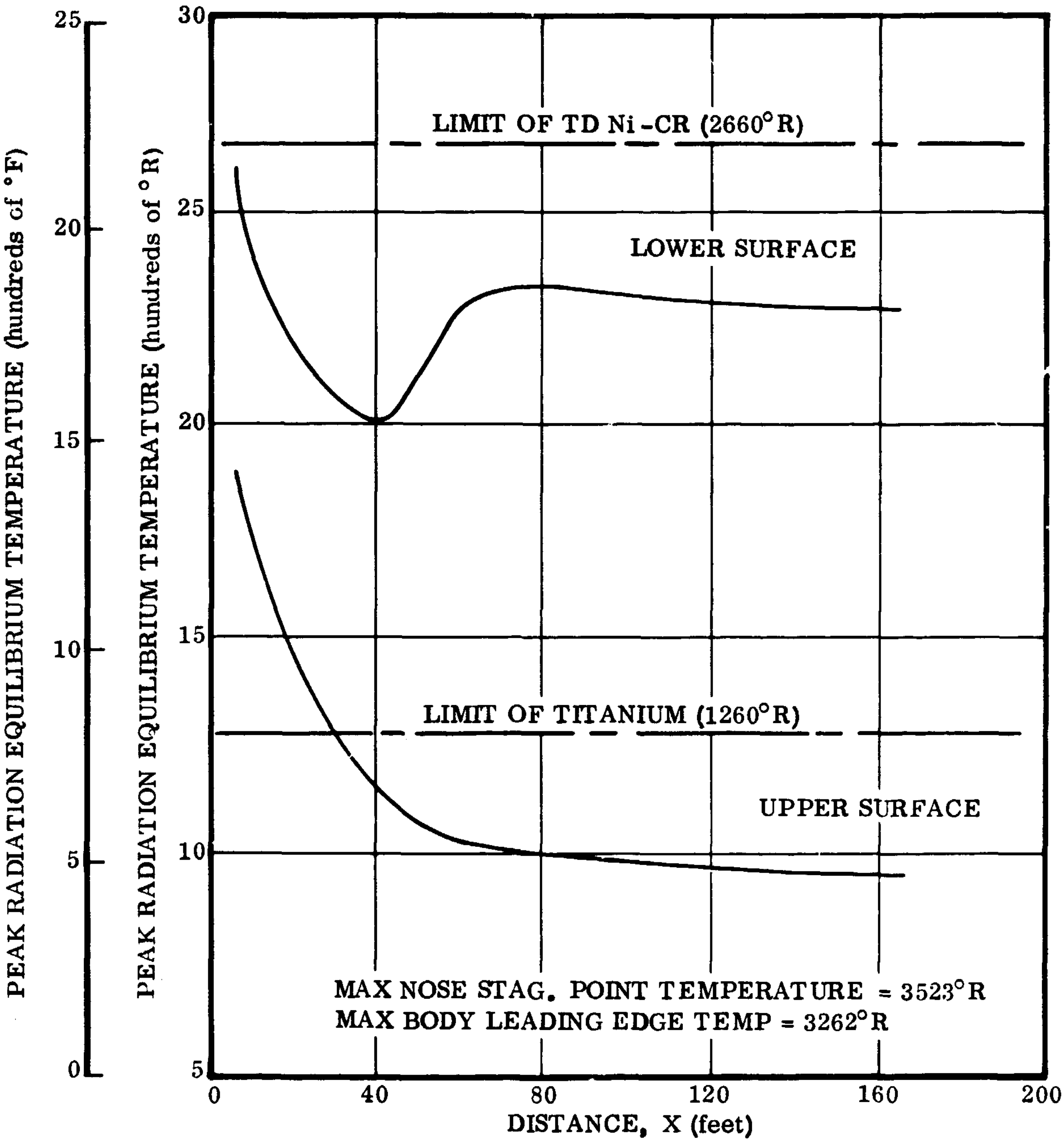


Figure 4-47. Peak Radiation Equilibrium Temperature for the FR-3 and FR-4 Orbiter, Trajectory No. 353 — 800 n.mi. Crossrange

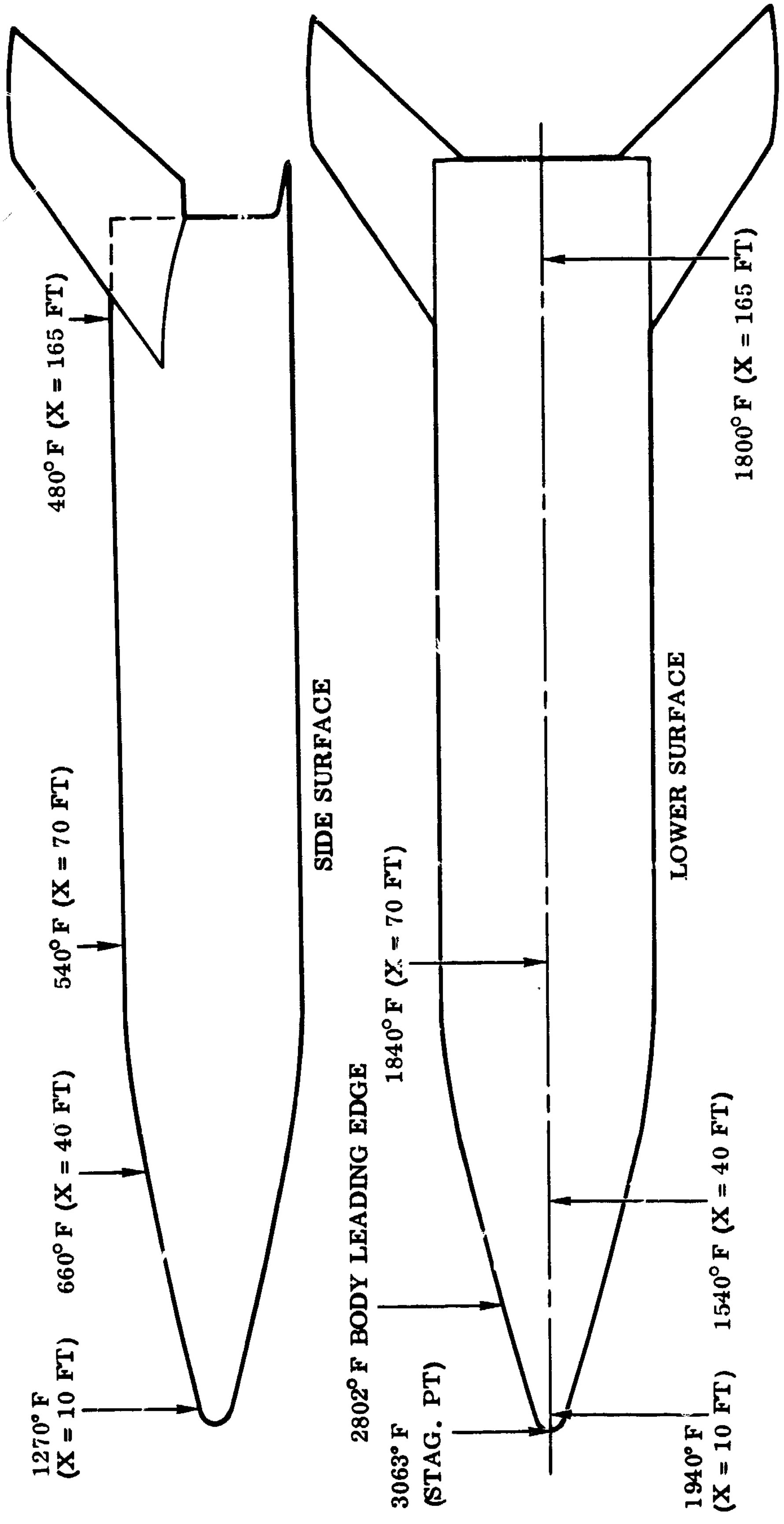


Figure 4-48. FR-3 and FR-4 Orbiter Peak Entry Temperatures

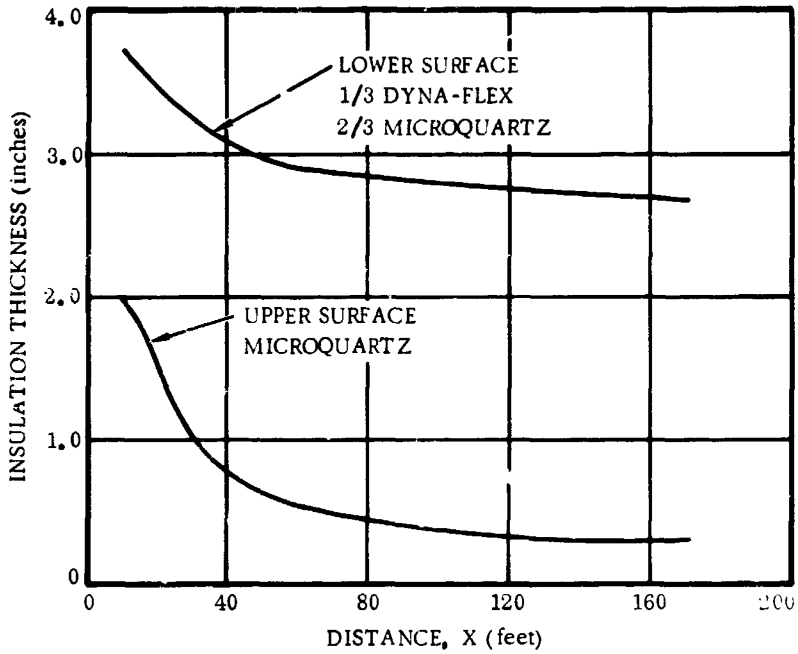


Figure 4-49. FR-3 and FR-4 Orbiter Lower and Upper Surface Insulation Requirements, Trajectory No. 353 — 800 n.mi. Crossrange

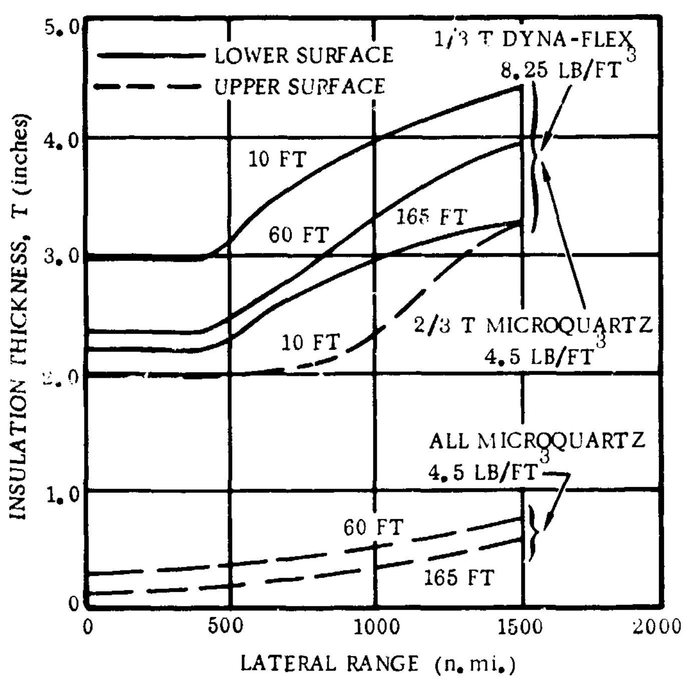


Figure 4-50. Local Insulation Thickness vs Lateral Range; FR-3 and FR-4

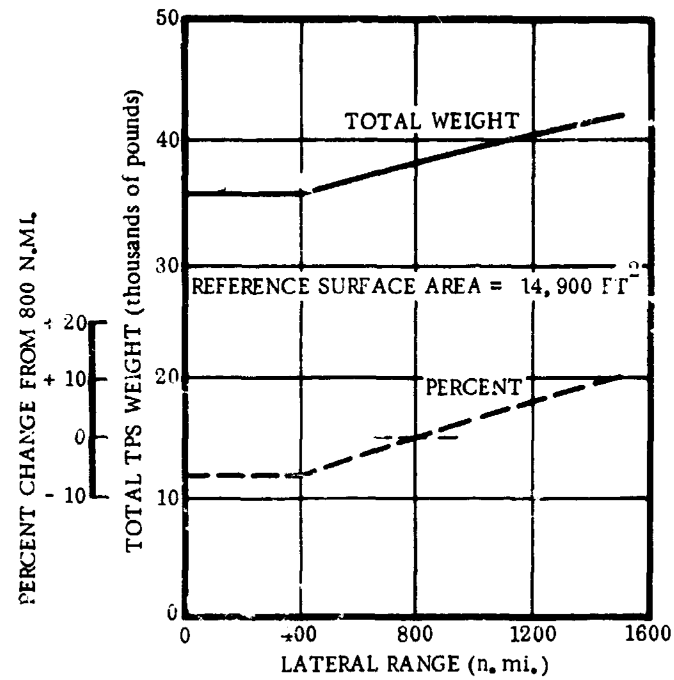


Figure 4-51. FR-3 Orbiter Total Thermal Protection System Weight vs Lateral Range

Figure 4-52 presents the variation of peak lower surface radiation equilibrium temperature as a function of planform loading. This curve was based on the results of the 800-n.mi. once-around abort and the variation determined during Space Transportation System studies. Note that the variation is rather [illegible] and the planform variations experienced during the ILRV study did not have much influence on lower surface entry temperatures.

Transition Reynolds numbers above 1×10^6 have no effect on the peak lower surface temperature for the 300 and 800 n.mi. trajectories. The peak lower surface temperature occurs on the first 10 ft during laminar flow, Figure 4-47. Increasing the transition Reynolds number will not change the peak temperature distribution over the first 40 ft. Also any temperature reduction aft of 40 ft will not cause a change in the cover panel material selection.

A heat transfer rate of 25 Btu/ft^2-sec was used as the base heating maximum value (see Volume IV, Section 4.10). It was determined to occur at 85 seconds after launch and was considered constant after this time.

4.6.2 BOOSTER ELEMENT. Two booster elements were analyzed. One was the initial configuration for the FR-3 and the second was the final FR-3 configuration. The initial FR-3 configuration was used to do a parametric study of staging conditions. One of the staging conditions corresponded to the staging point of the final FR-4 boost element.

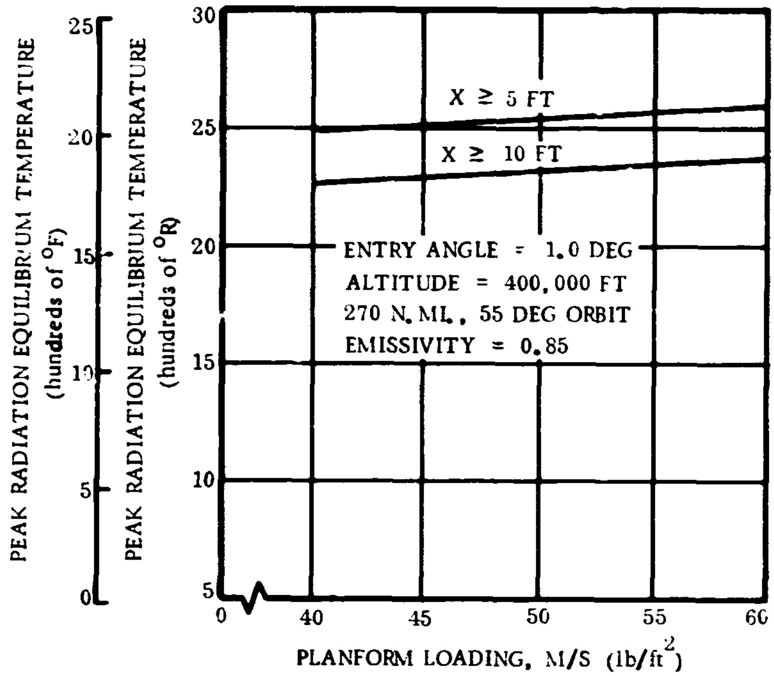


Figure 4-52. Peak Lower Surface Temperature vs Planform Loading; FR-3 and FR-4 Orbiters

4.6.2.1 FR-3 Boost Element. Figure 4-53 shows the final FR-3 boost element configuration. It was defined at the end of the contract study and hence only a cursory aerothermodynamic analysis was performed. Also shown on Figure 4-53 are the peak calculated radiation equilibrium temperatures. Figure 4-54 presents the temperature histories calculated for the recovery trajectory plotted on Figure 4-27. No structural temperature distributions were calculated. The short flight times would indicate the use of a hot structure. Insulation requirements are of the order of 0.1 to 0.2 inch of microquartz on the lower surface. The upper surface insulation requirement will be zero thickness. Therefore, the hot 811 titanium fairing over the aluminum alloy main structure was selected for the FR-3 booster in this phase. A hot structure approach using 718 nickel alloy or even 811 titanium is an alternate candidate (see Volume V, Section 8).

4.6.2.2 FR-4 Boost Element. The temperature distribution shown on Figure 4-55 was determined during the parametric staging velocity study performed on the initial FR-3 booster and reported in Volume III, Section 3.6.3. The FR-4 boost element staging velocity of 9411 fps and dynamic pressure of 50 psf corresponded to the temperatures plotted in Figure 3-90 of Section 3.6.3 of Volume III. Volume III Figures 3-83 and 3-84 present the recovery trajectory for the FR-4 boost element. Recovery angle of attack was 40 deg and the bank angle was 60 deg. Insulation requirements will be minimal.

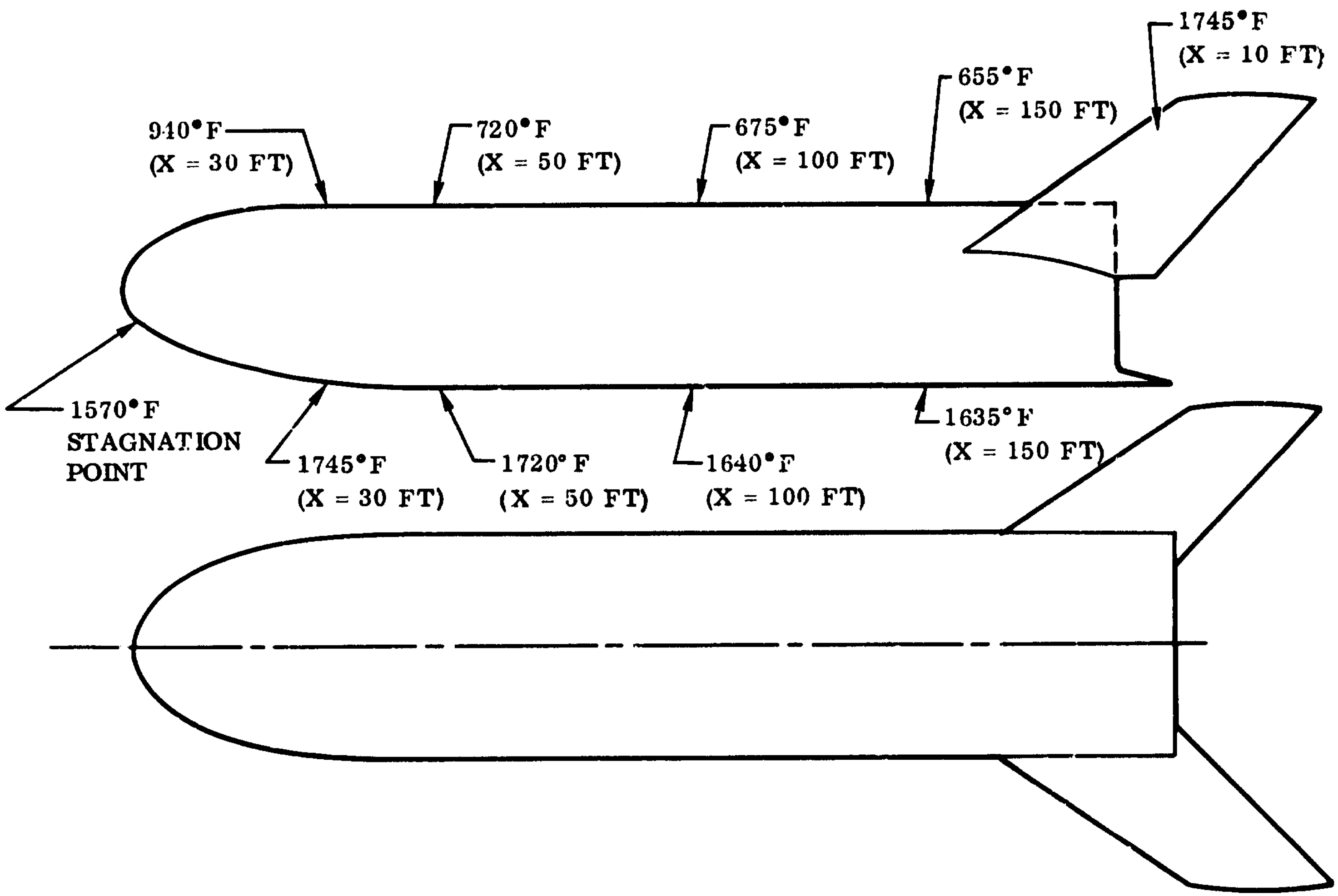


Figure 4-53. FR-3 Booster Peak Temperatures

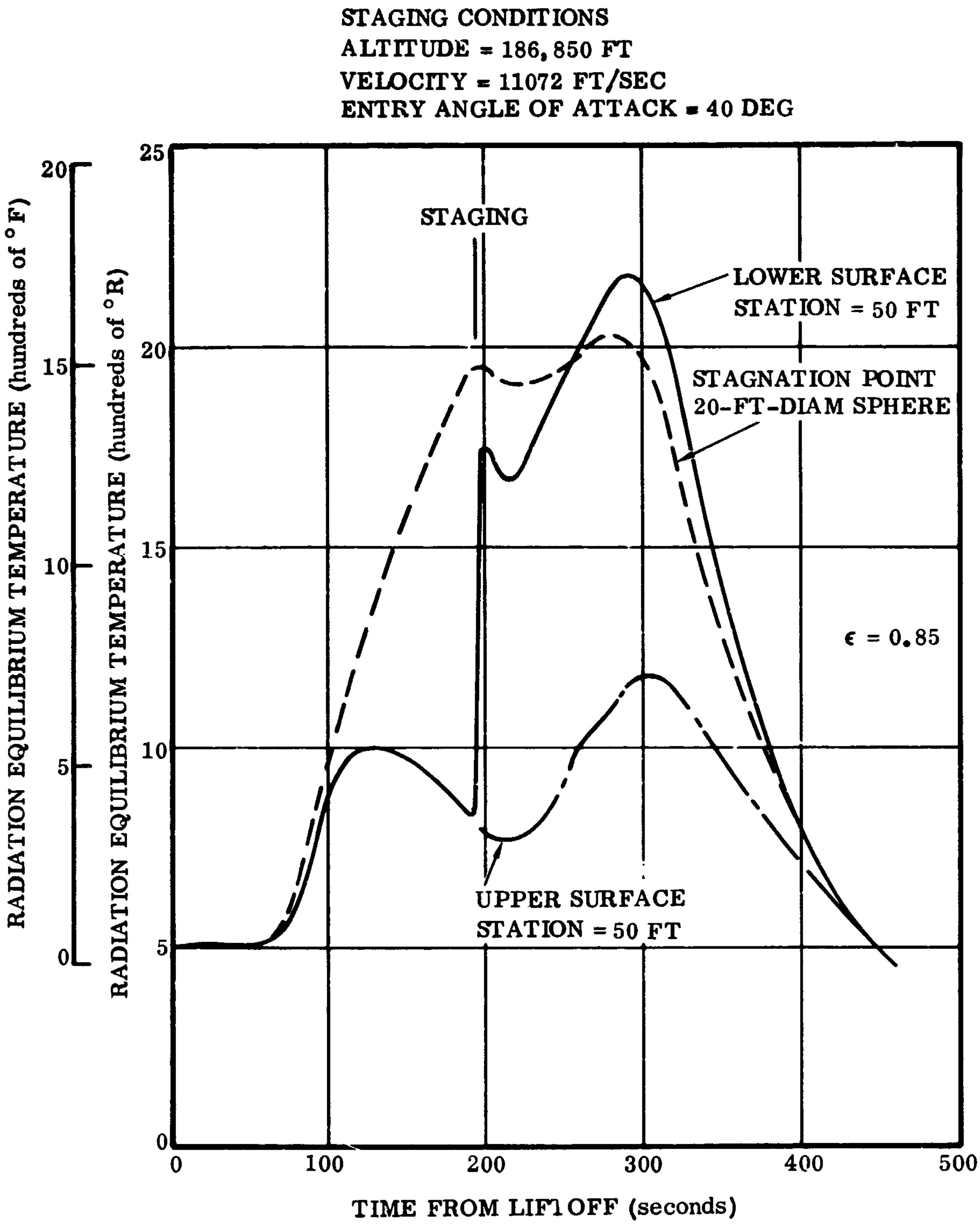


Figure 4-54. FR-3 Booster Ascent and Entry Radiation Equilibrium Temperature Histories

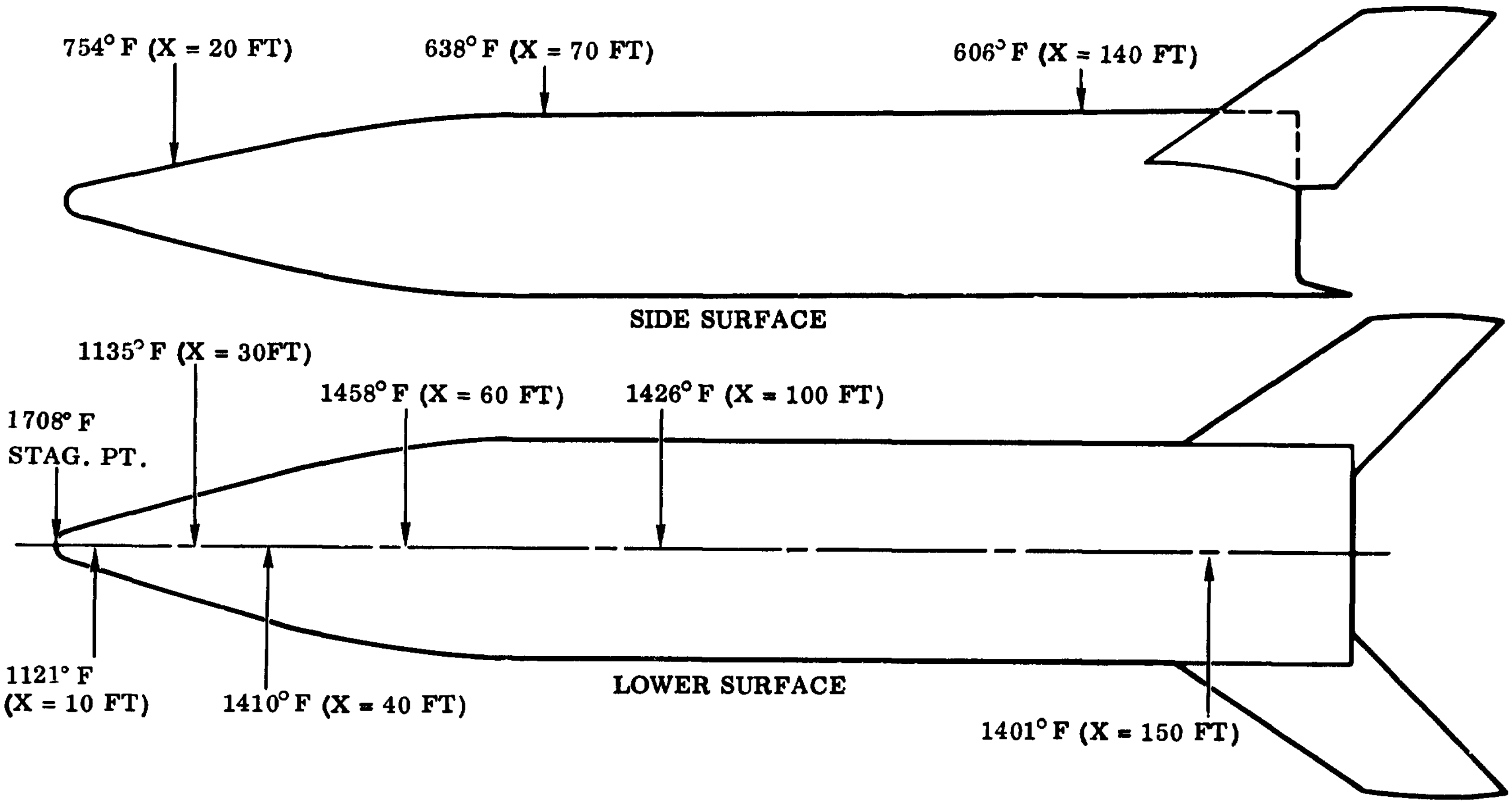


Figure 4-55. FR-4 Booster Peak Temperatures

4.6.2.3 Boost Element Base Heating. Aerodynamic heating of the main propulsion nozzles presents no problem for the 400,000-lb-thrust engines; see Section 4.C.1 and Volume IV, Section 4.7. Base heating by the propellant gas was determined to be a maximum of 25 Btu/ft^2-sec (Volume IV, Section 4.10).

4.7 THERMOSTRUCTURAL DESIGN AND ANALYSIS

During the sizing procedure that was used to develop the FR-3 and FR-4 configurations, one interim configuration was selected for developing into a point design of the thermostructure design. The configuration selected is identified as T-18. It was derived as a stage in the equal element FR-1 vehicle.

The T-18 configuration is shown in Figure 4-56 and the thermostructural design is discussed in Section 4.7.1. It presents a conceptual approach to airframe design that is generally valid for the FR-3 orbiter and FR-4 vehicles although there are differences in the configurations. These differences are discussed in Volume V, Section 8.1. Because these differences become significant in the FR-3 booster, a different structural arrangement was studied for this vehicle and is presented in Section 4.7.2 (and Volume V, Section 8.5). The design requirements, supporting analysis, and material data used in developing these structural arrangements as well as detailed discussion of the design concepts are presented in Volume V, Section 8.

4.7.1 LOADS

4.7.1.1 Typical FR-4 Net Loads. Net loads during ground and flight conditions were determined for various vehicle components. These included body, wing, fins, and landing gears. The loads were determined by computer programs that handle airload and mass distributions, cruise and booster thrust vectors, concentrated loads, and translational and rotational inertias. The vehicle is in quasi-static equilibrium in all cases. Rigid body analysis was used. Details relative to airloads, mass distributions, and net loads are given in Volume IV, Section 5.

A typical example of the results obtained by these analyses, Figure 4-57 and 4-58 show net body peak load intensities for various ground and flight conditions for the FR-4 orbiter and booster elements, respectively. It can be noted that critical loads for various areas of the body occur among the subsonic gust, maximum αq, booster burnout, and ground wind conditions combined with internal tank pressures.

4.7.1.2 Typical FR-3 Net Loads. A study similar to that for the FR-4 configuration was performed in determining FR-3 net loads. Detailed results of this study are given in Volume IV, Section 5.

Typical examples for net body peak load intensities are given in Figures 4-59 and 4-60 for the FR-3 orbiter and booster elements, respectively. Booster body net loads are for a configuration having non-integral tanks. For a configuration with integral tanks, the loads shown will be relieved by the effects of internal tank pressures.

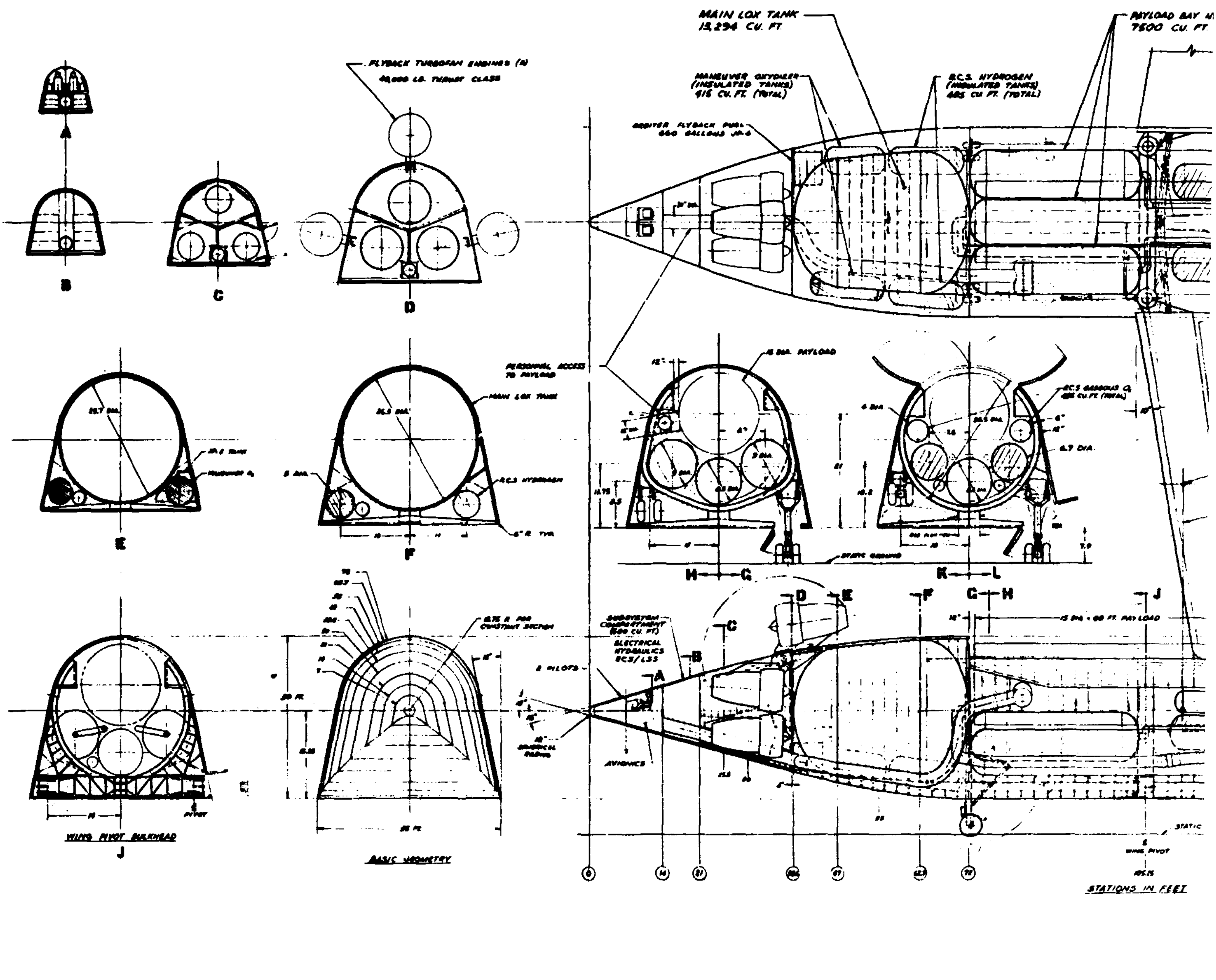
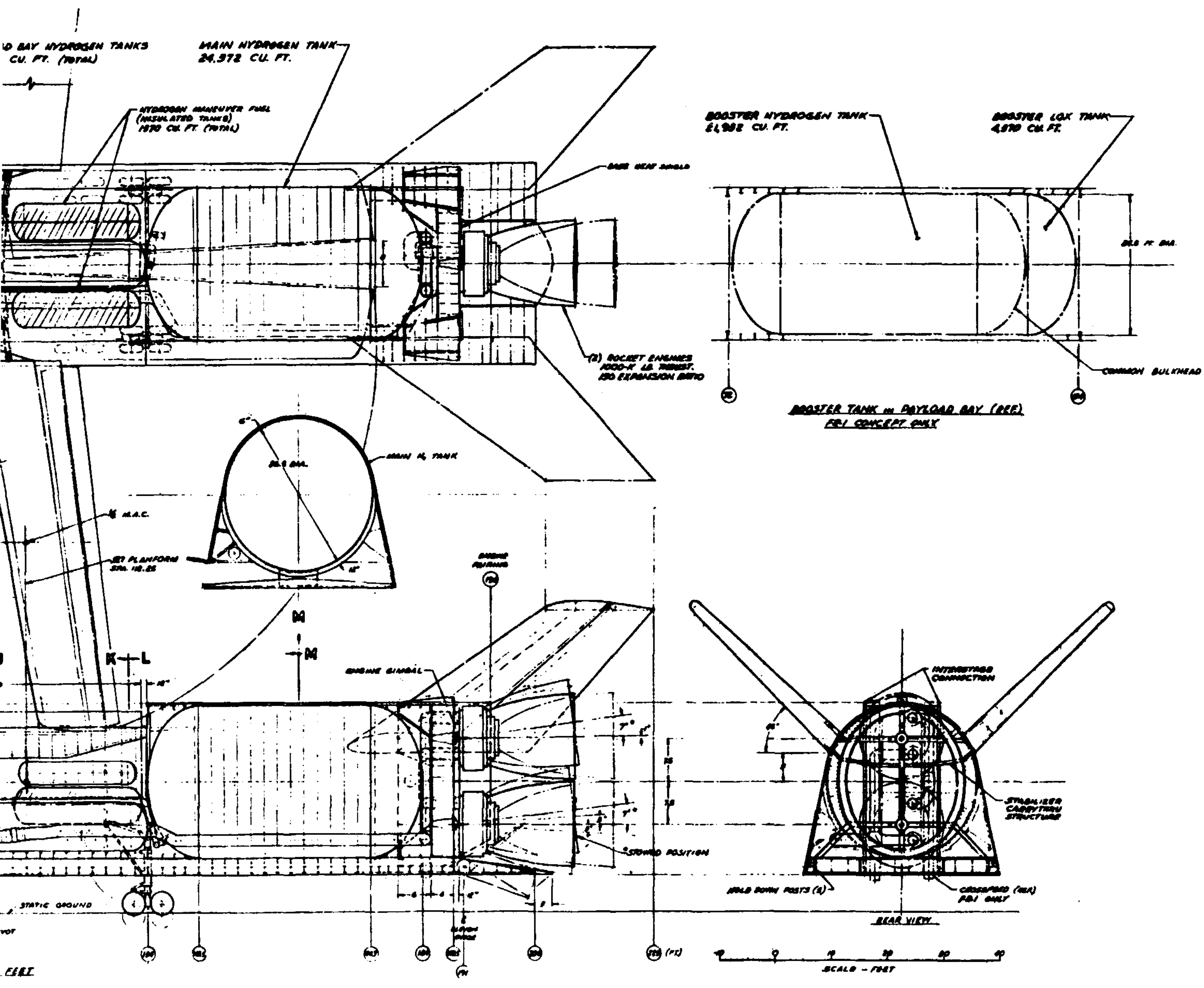


Figure 4-56. T-18 Vehicle Basic Configuration

FOLDOUT FRAME 1



FOLDOUT FRAME 2

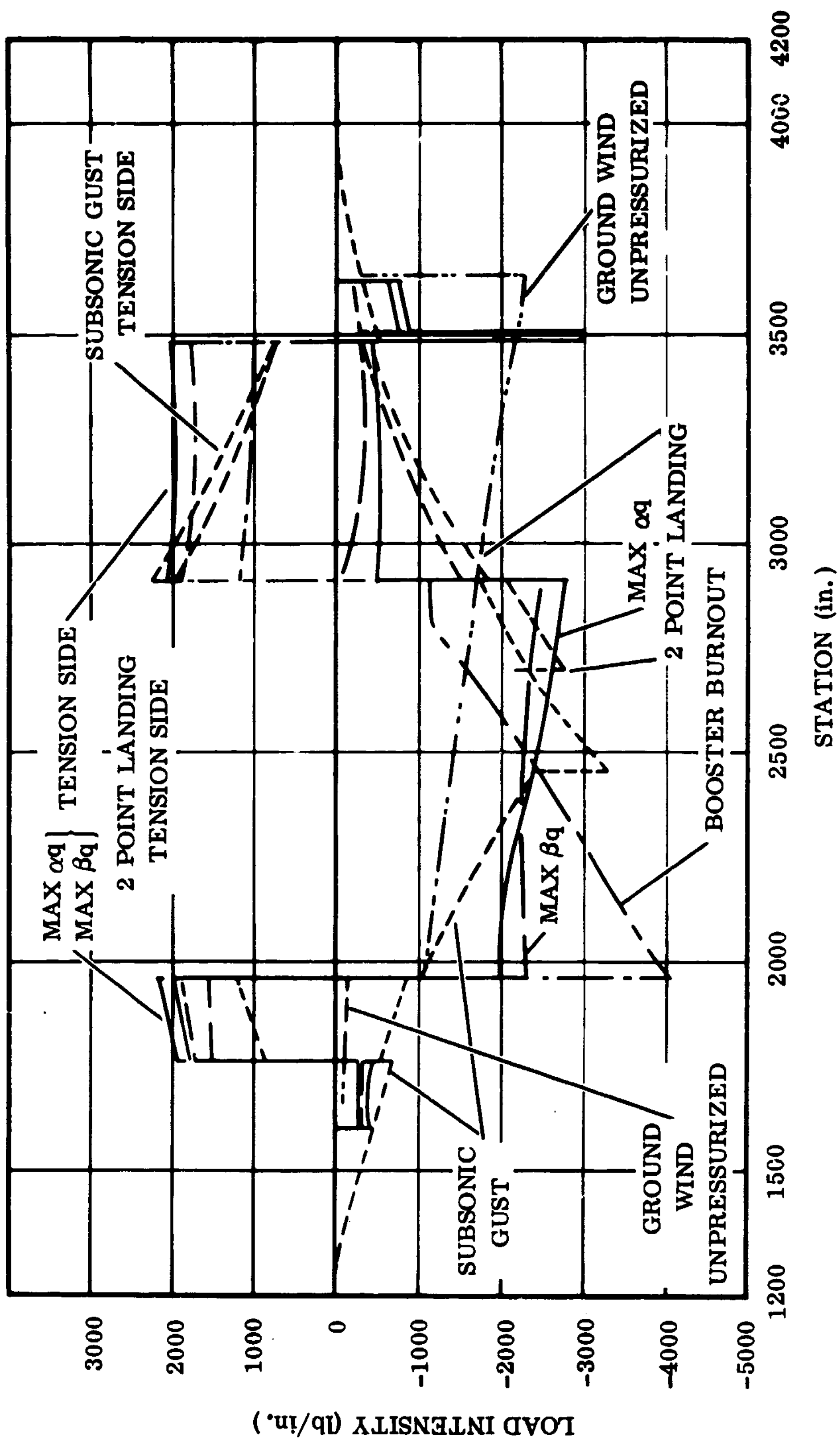


Figure 4-57. Comparison of Typical FR-4 Orbiter Peak Limit Load Intensities

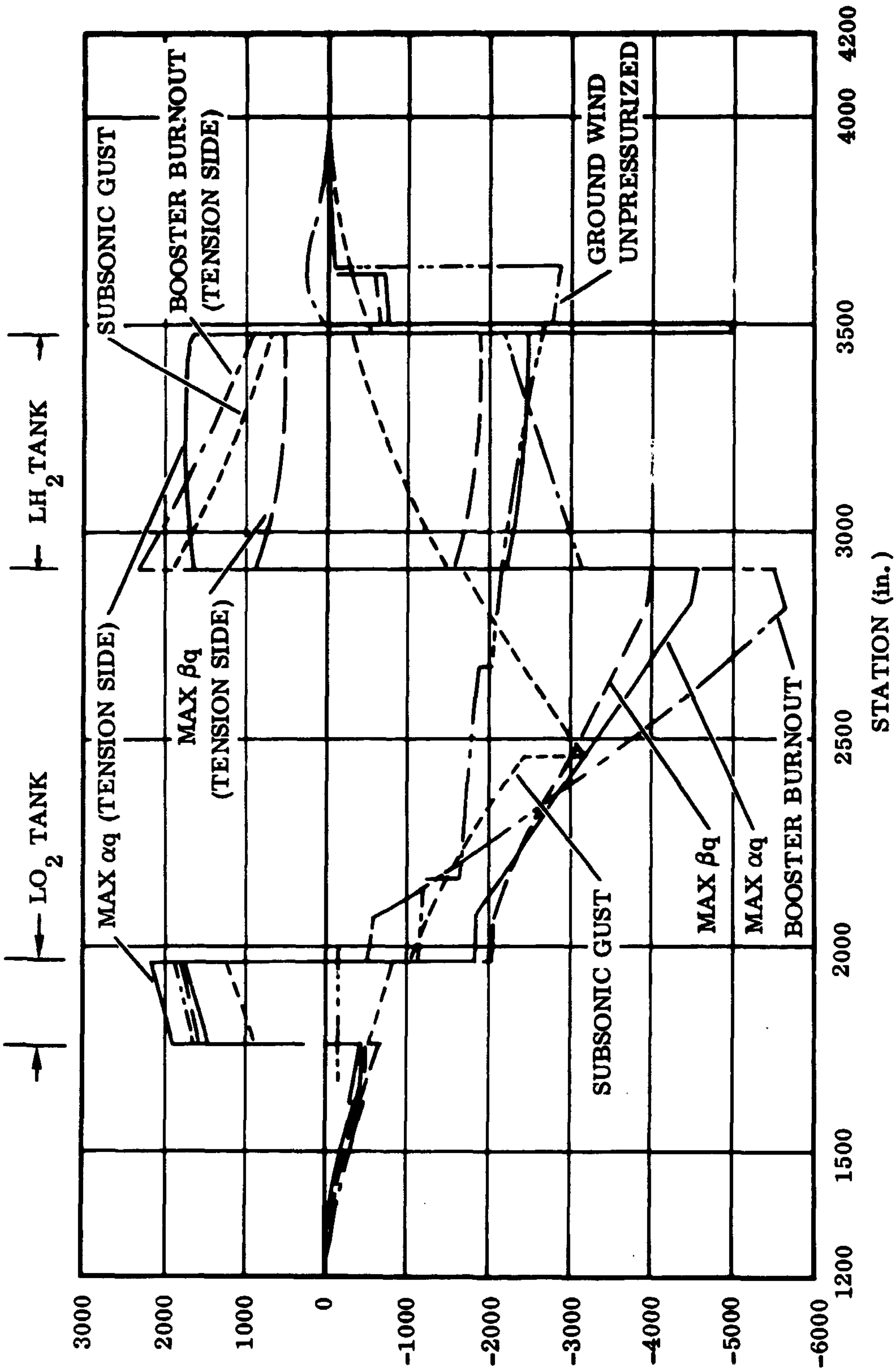


Figure 4-58. Comparison of Typical FR-4 Booster Peak Limit Load Intensities

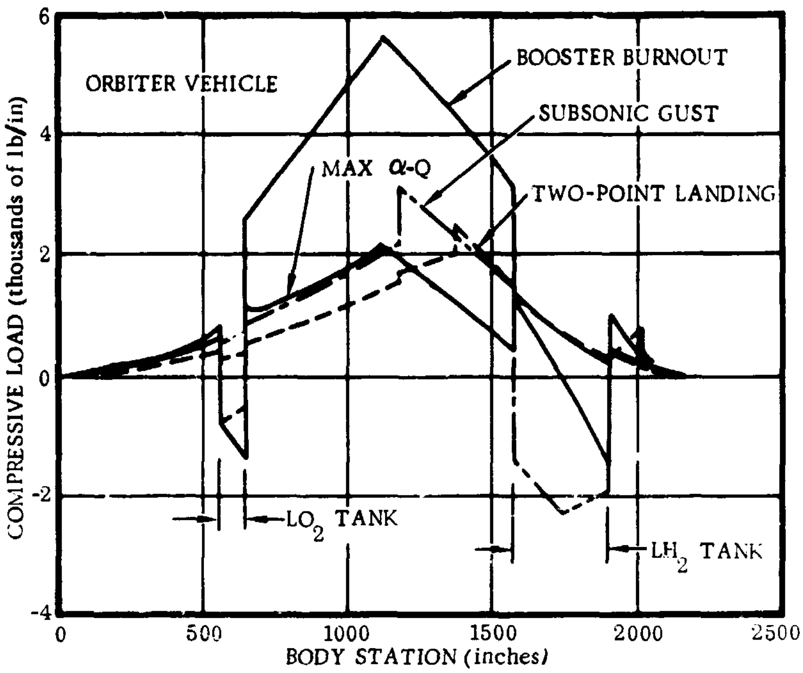


Figure 4-59. FR-3 Tail-to-Tail Configuration Peak Compression Loads (Limit)

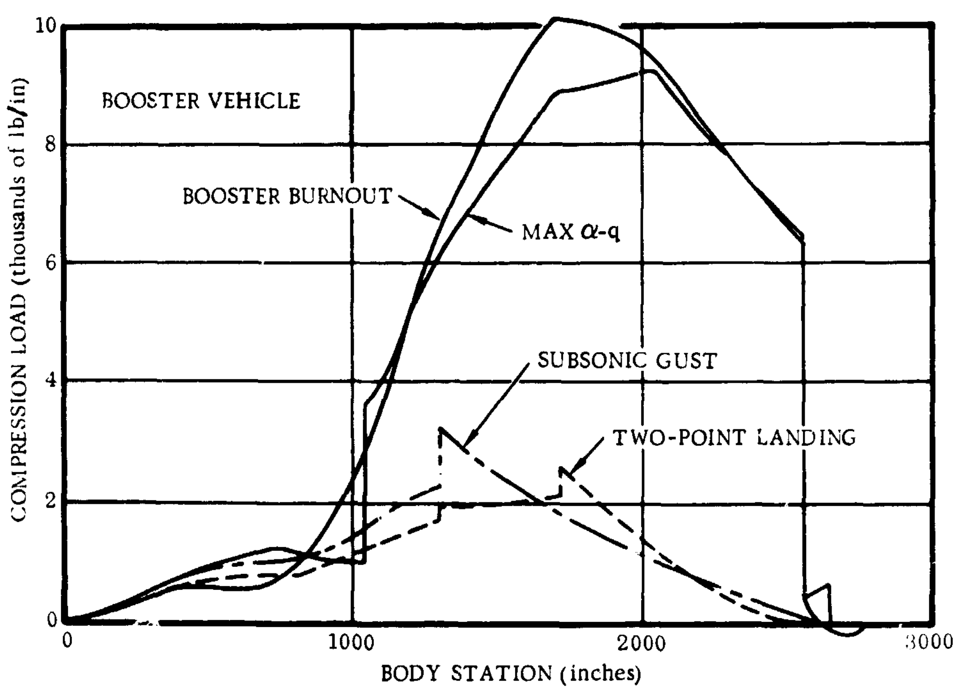


Figure 4-60. Tail-to-Tail Configuration Peak Compression Loads (Limit)

4.7.2 SELECTED THERMOSTRUCTURAL CONCEPTS. The approach to the structural design of the shuttle vehicles, both booster and orbitor, has been one of conventional structural arrangements and materials and state of the art fabrication methods. However, high-strength materials such as composites, titanium, and other heat-resistant alloys have been used where thermal and/or loading conditions showed them to be superior to other materials. The major structural assemblies shown in Figure 4-61 are:

a. Forward fuselage, including the crew compartment and turbofan engine bays.

b. LO_2 and LH_2 tanks.

c. Center section, including payload bay (orbiter only), wing pivot bulkhead, and landing gear bulkheads.

d. Thrust structure.

e. Tank transition structures.

f. Stabilizers and wings.

g. TPS and support structure.

Most structures, including the LO_2 and LH_2 tanks, have been designed for a thermal environment of 200°F or less. This temperature is maintained at the structural envelope during entry by the TPS. The exceptions are the aerodynamic stabilizer, wing, and orbiter payload bay doors. During entry, the swing wing is stowed in its compartment and is not subjected to elevated temperatures. The stabilizers and payload doors are designed from materials (such as Inconel and titanium alloy, respectively) that can withstand elevated temperatures and maintain their structural integrity at reduced but acceptable levels.

The fuselage section forward of the LO_2 tank is a semi-monocoque shell that contains the crew compartment, equipment bay, and turbofan engine compartments. Whereas the crew compartment is designed to be pressurized, the remainder of the forward fuselage is vented to ambient conditions. A major bulkhead at Station 38.3 supports the turbofan engine pivots and forms the structural joint for the transition to the LO_2 tank. The design of this fuselage section adheres to the classical methods of shell stiffening through the use of frames, bulkheads, and stiffeners. Longerons have been arranged to carry and redistribute concentrated loads that occur in the vicinity of the cockpit windshield and entrance and near the turbofan engine doors.

The LO_2 and LH_2 tanks are both designed to form an integral part of the load-carrying vehicle structure. They are fusion-welded assemblies of wide circular rings that form the tank skin and frames. At each end, ellipsoidal domes form the closure bulkheads. The tanks are joined to the other structural sections of the vehicle by means of transition skirts.

These transition structures between forward fuselage and LO_2 tank, LO_2 tank and payload section, and payload section and LH_2 tank are circular semi-monocoque shells that attach to the tanks with a bolted butt joint at the point of tangency of the tank dome and skin. Their design incorporates fittings, doublers, and other reinforcements to introduce and redistribute concentrated loads such as turbofan engine thrust and vehicle interconnect and staging forces.

The payload bay structure is located between the LO_2 and LH_2 tanks in the constant section of the vehicle. Its structural skins are stiffened by longitudinal stringers and fuselage rings. Major bulkheads in this section are the wing pivot bulkhead and the main landing gear and nose landing gear bulkheads. The structure, skin, rings, bulkheads, and stiffeners in the booster vehicle are continuous over the top of the section. In the orbiter vehicle, an opening (covered by two symmetrically arranged doors) is provided for insertion and removal of the payload. Built-up longerons of good column stability and high bending resistance provide load paths around the payload door.

The aft end of the LH_2 tank is joined to the main engine thrust skirt and the engine gimbal pads are mounted to a deep-section beam on the vertical centerline of the skirt. This beam, together with crosswise beams at the gimbal pads, transfers the main engine thrust loads into the skirt skin. The thrust skirt also incorporates a center tie box for the stabilizers. Beneath the thrust skirt and in line with the flat bottom of the vehicle, a multicell beam structure extends from side to side. Two holddown fittings, one on either side, are attached to this beam. A third holddown fitting is mounted to the upper end of the vertical main engine thrust beam.

The swing wings are of conventional design with "reverse" type flaps at the trailing edge. The wing box has two spars, ribs, and skins with integral stringers. The wing pivot fitting is a fusion-welded assembly of titanium alloy. Leading edge and fixed trailing edge assemblies are rib-stiffened contoured skins and honeycomb sandwich panels, respectively. The panels are supported on beams cantilevered from the rear spar.

Stabilizer construction of the booster differs from that of the orbiter because of a different thermal environment. The stabilizer of the booster is a hot structure throughout, made entirely from 718 nickel-base alloy. The orbiter stabilizer incorporates a TPS with a load-carrying structure fabricated from titanium alloy. Both stabilizers incorporate a two-spar box beam with rib and stringer stiffened skins. Leading edge and movable trailing edge surfaces are rib-stiffened contoured skin assemblies.

The TPS that covers the exterior of the entire vehicle, with the exceptions mentioned earlier, is mounted to its support structure and does not contribute to the ability of the basic vehicle structure to resist external and internal forces. The beams, membranes, and braces of the support structure are attached to the vehicle structure and transfer aerodynamic loads only.

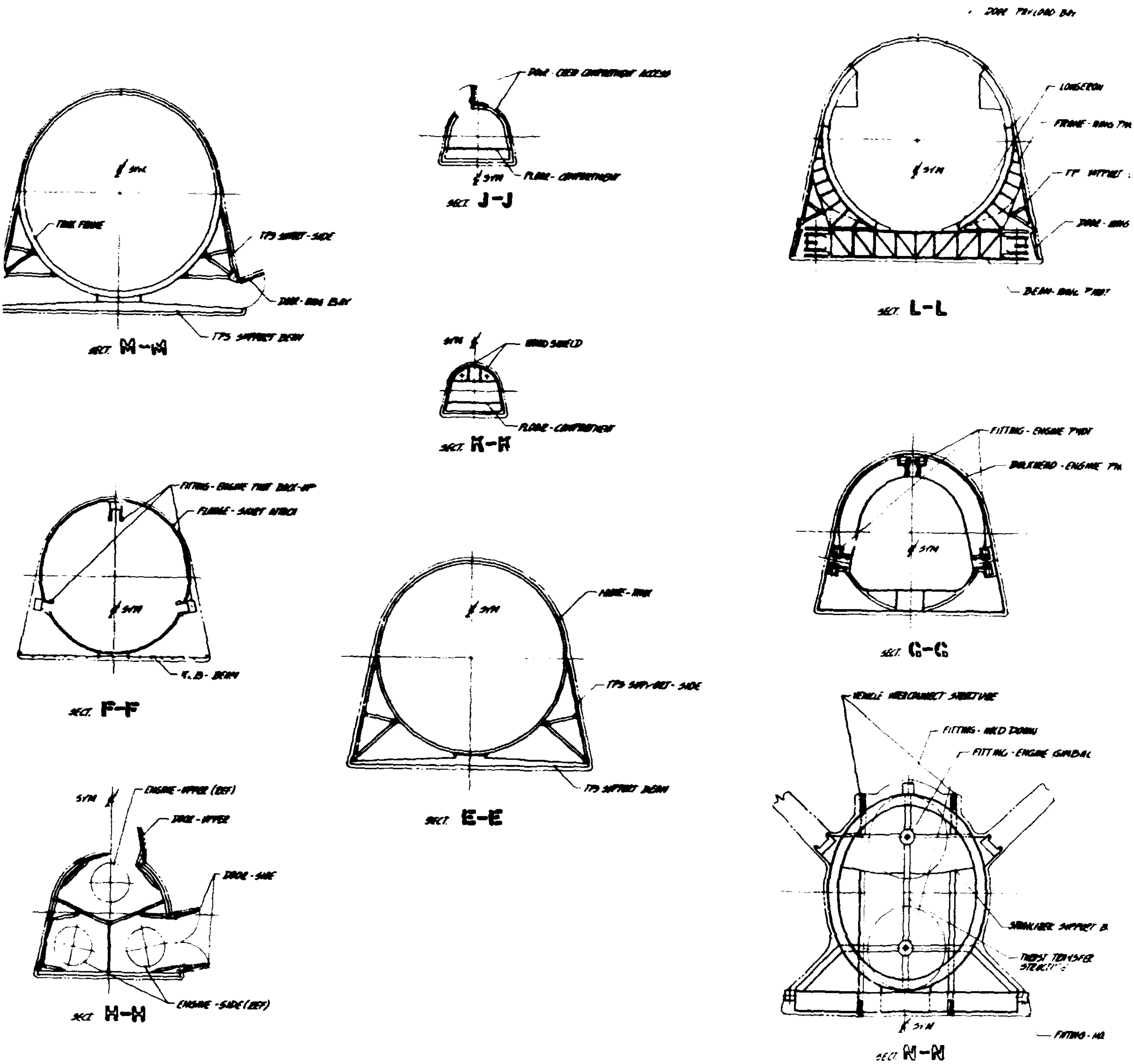


Figure 4-61. Structural Arrangement

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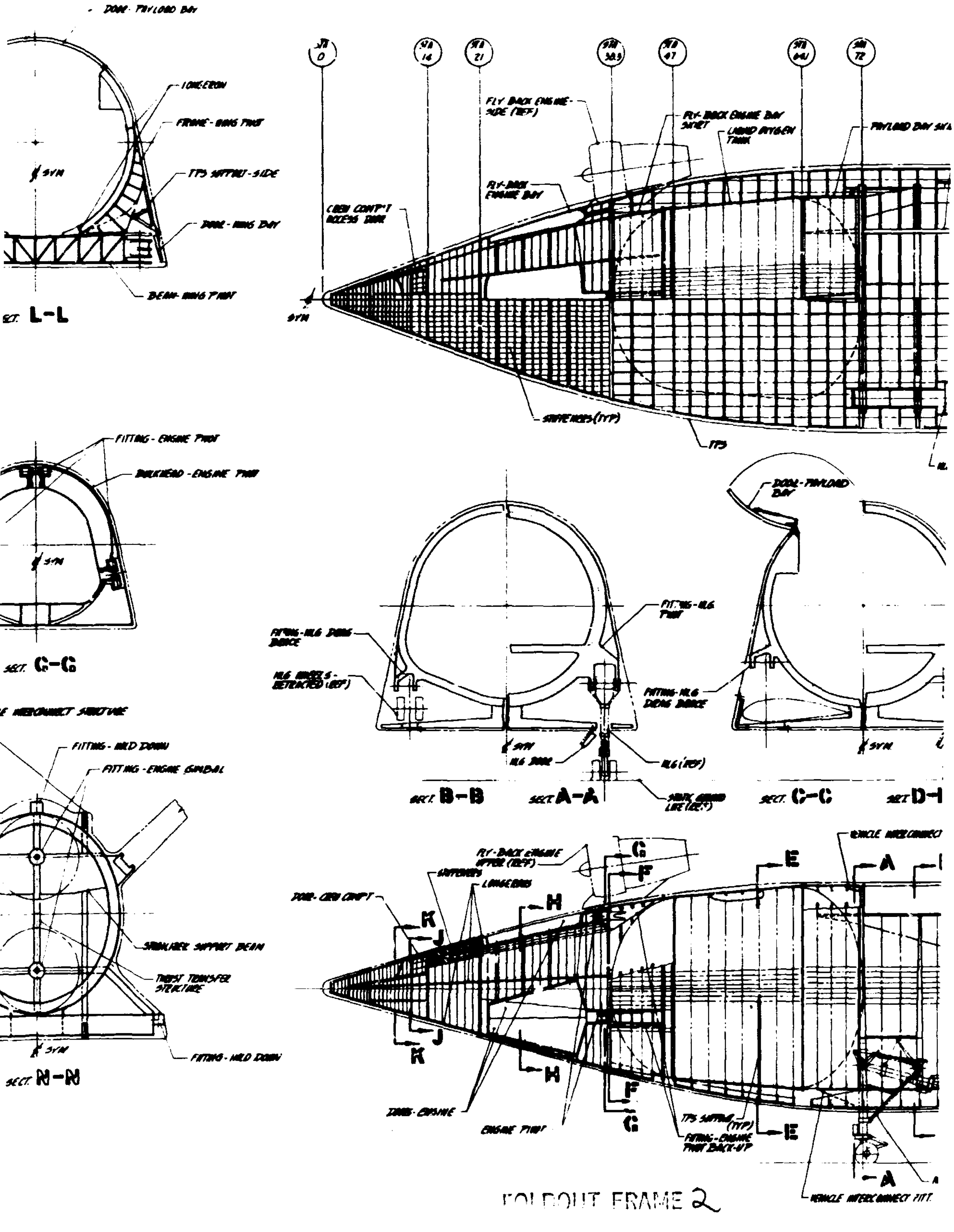
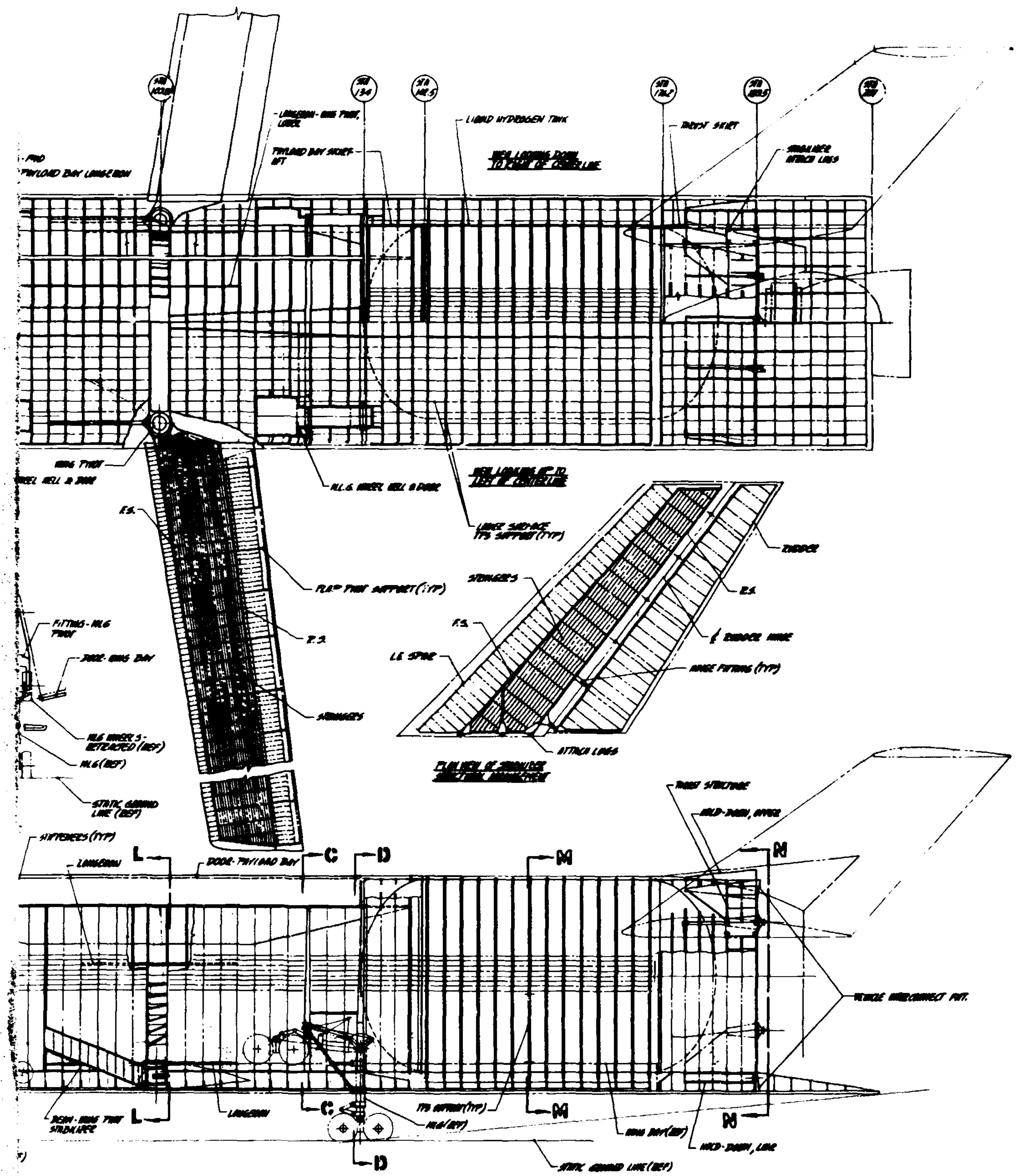


FOLDOUT FRAME 2



FOLDOUT FRAME 3

The overall structural arrangement shows an excellent adaptability to a logical manufacturing breakdown into major assemblies at the transition structures.

4.7.3 FR-3 BOOSTER THERMOSTRUCTURAL CONCEPT. The approach to the structural design of the booster element has been one of conventional structural arrangement, materials, and state-of-the-art fabrication methods. However, high-strength materials such as composites, titanium, and other heat-resistant alloys have been used where thermal and/or loading conditions showed them to be superior to other materials.

The FR-3 booster employs two propellant tanks 33 ft in diameter which are of similar construction as the Saturn S-IC stage. The forward LO_2 tank is 59.8 ft in length and separated from the aft LH_2 tank of 117.3 ft in length by an interstage adapter structure. Separate tanks were employed which provided a gross liftoff weight penalty of 2.6% compared to a vehicle configuration which employs tanks with a common bulkhead. The use of separate tanks, however, affords less developmental problems and cost.

The major structural assemblies; shown in Figure 4-62 are:

a. Forward fuselage, including the crew compartment, turbofan engine bays, and nose landing gear.

b. LO_2 tank.

c. Inter-tank adapter section.

d. LH_2 tank, including wing pivot bulkhead, and landing gear bulkheads.

e. Thrust structure, including vehicle hold down supports, base heat shield support, stabilizer carry-through structure, and vehicle separation supports.

f. Stabilizers and wings.

g. TPS and support structure

Most structures, including the LO_2 and LH_2 tanks, have been designed for a thermal environment of 200°F or less. This temperature is maintained at the structural envelope during entry by the TPS. The exceptions are the aerodynamic stabilizer and wing. During entry, the swing wing is stowed in its compartment and is not subjected to elevated temperatures. The stabilizers are designed from titanium alloy that can withstand elevated temperatures and maintain their structural integrity at reduced but acceptable levels.

The fuselage section forward of the LO_2 tank is a semi-monocoque shell that contains the crew compartment, equipment bay, turbofan engine compartments, and nose landing gear. While the crew compartment is designed to be pressurized, the remainder of the forward fuselage is vented to ambient conditions. A major bulkhead at Station 24 supports the turbofan engine pivots; another at Station 29 supports the nose landing gear. The design of this fuselage section adheres to the classical methods of shell stiffening through the use of frames, bulkheads, and stiffeners. Longerons have been arranged to carry and redistribute concentrated loads that occur in the vicinity of the cockpit canopy, the turbofan engine doors, and nose landing gear.

The LO_2 and LH_2 tanks are both designed to form an integral part of the load-carrying vehicle structure. They are fusion-welded assemblies of wide circular rings that form the tank skin and frames. At each end, ellipsoidal domes form the closure bulkheads. Major external frames in the LH_2 tank area are the wing pivot frame and the main landing gear frame. Through these areas the body was deepened to provide additional frame depth for increased load restraint.

Major wing support frames are located at Stations 98.5 and 82.5. With interconnecting stabilizer beams, they provide shear and moment restraint for the wings through the wing pivot fittings.

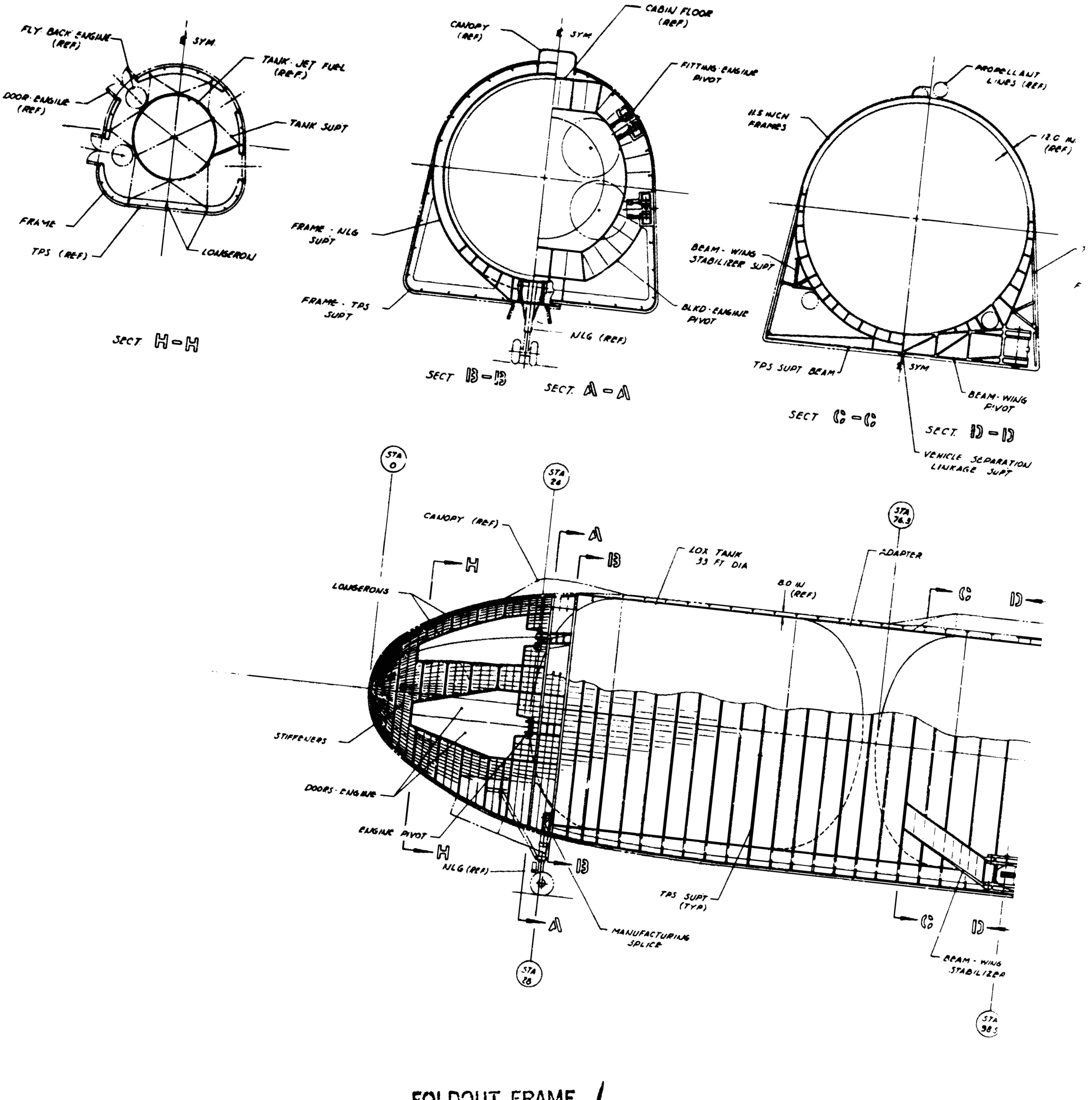
Main landing gear support frames located at Stations 120 and 127.5 restrain the main landing loads and drag loads, respectively.

The tanks are joined to adjacent structural sections of the vehicle by means of a bolted butt joint at the point of tangency of the tank dome and skin. Their design incorporates fittings, doublers, and other reinforcements to introduce and redistribute concentrated loads such as turbofan engine thrust and vehicle interconnect and staging forces.

The inter-tank adapter structure is located between the LO_2 and LH_2 tanks in the constant section of the vehicle. Its structural skins are stiffened by longitudinal stringers and fuselage rings.

The aft end of the LH_2 tank is joined to the boost engine thrust skirt which supports a matrix of deep-section beams. These intersecting beams support 15 gimbal pads and transfer the main engine thrust loads into the skirt skin. The thrust skirt also incorporates a center tie box for the stabilizers. Beneath the thrust skirt and in line with the flat bottom of the vehicle, a multicell beam structure extends from side to side. Two holddown fittings, one on either side, are attached to this beam. A third holddown fitting is mounted to the upper end of the vertical main engine thrust beam.

The vehicle employs a conventional swing wing design with reverse type flaps at the trailing edge. The wind box has two spars, ribs, and skins with integral stringers. The wing pivot fitting is a fusion-welded assembly of titanium alloy. Leading edge and fixed trailing edge assemblies are rib-stiffened contoured skins and honeycomb sandwich panels, respectively. The panels are supported on beams cantilevered from the rear spar.



FOLDOUT FRAME 1

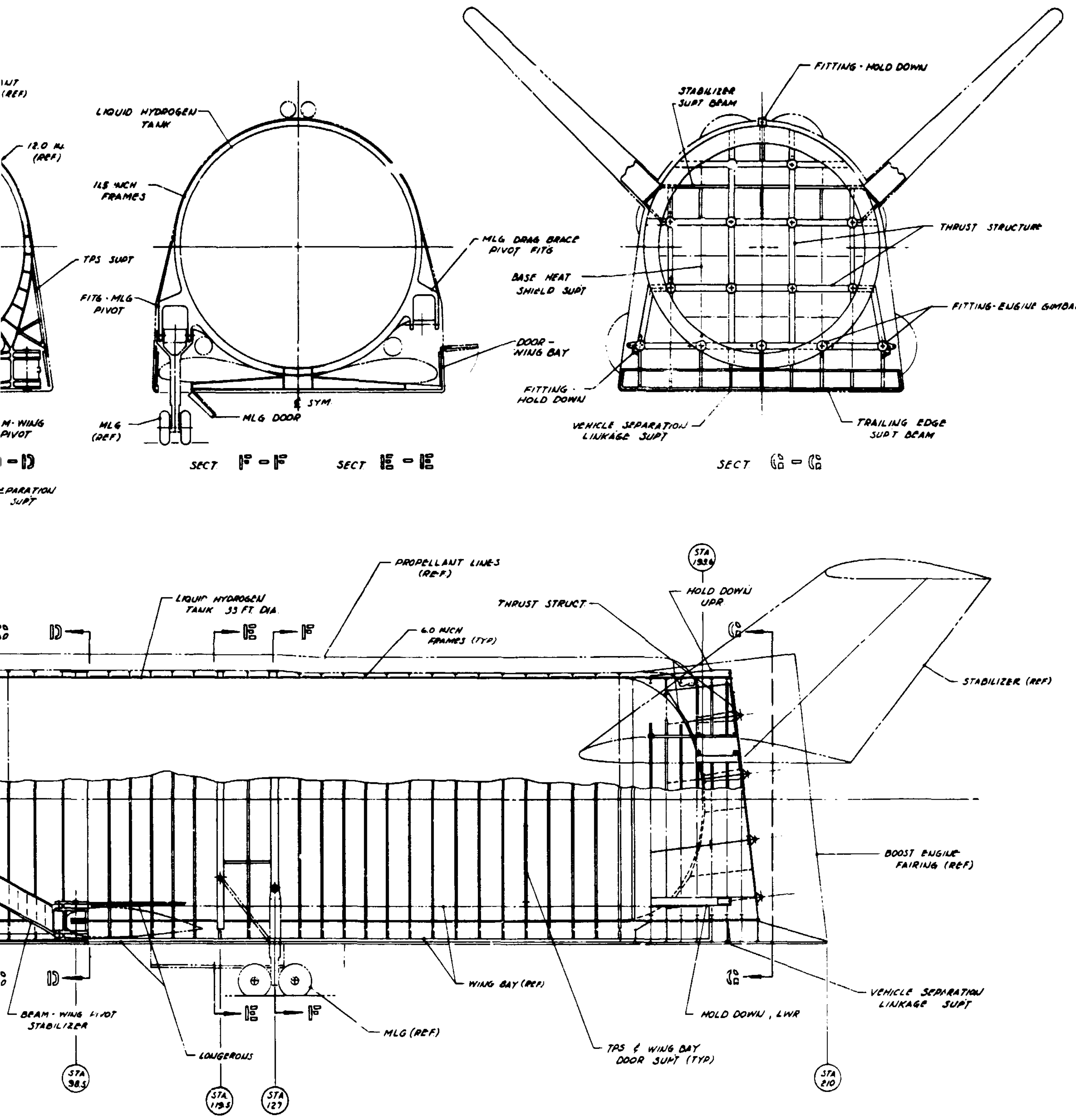


Figure 4-62. Structural Arrangement FR-3 Booster

FOLDOUT FRAME 2

Stabilizer construction of the booster differs from that of the orbiter because of a different thermal environment. The stabilizer of the booster is a hot structure throughout, made entirely from 718 nickel-base alloy and incorporates a two-spar box beam with rib and stringer stiffened skins. Leading edge and movable trailing edge surfaces are rib-stiffened contoured skin assemblies.

The TPS, mounted to its support structure on the outside of the vehicle, does not contribute to the ability of the basic vehicle structure to resist external and internal forces. The beams, membranes, and braces of the support structure are attached to the vehicle structure and transfer aerodynamic loads only.

An alternate structural arrangement for the FR-3 booster is shown in Figure 8-40 of Volume V. This arrangement presents a "hot" structure approach where the heat shield has been stiffened and supported with frames to carry the primary flight loads. The propellant tanks are installed within the airframe so they are isolated from thermal loads and deflections.

SECTION 5

BOOST PHASE CONTROL

The space shuttle vehicle boost phase will be similar to that of the Saturn or Atlas in that a vertical takeoff with pitchover to a gravity turn is an early trajectory requirement. As dynamic pressure builds up the vehicle must be controlled to a nearly zero angle of attack to keep the airloads at a minimum. Considerations were given to control forces generated by engine gimbaling, secondary injection, aerodynamic surfaces, and thrust modulation. Engine gimbaling was selected as the best means for producing control moments.

A unique feature that heavily influenced control requirements for the FR-3 and FR-4 vehicles is that both vehicles were aerodynamically stable throughout the boost phase of flight. With an aerodynamically stable vehicle, maximum αq loads can be relieved by limiting the control moment. For the limited control moment conditions at maximum αq the vehicle weathercocks (or rotates into the wind) to reduce the angle of attack, thereby reducing the airloads on the vehicle. The control moment limiting is unconventional when compared to the control systems on such aerodynamically unstable vehicles as Saturn. For these vehicles, a control limit can produce a catastrophic failure and load relief can only be provided by control system electronics.

Simulated flights using 99 percentile ETR Marshall synthetic winds have demonstrated that engine gimbaling is the preferred control technique dictated by engine-out, center-of-gravity offset, and maximum αq, conditions.

Non-linear control system analysis was performed to determine the minimum gimbal angular rates and accelerations for both configurations. An angular rate of 0.17 rad/sec and an acceleration of 10 rad/sec^2 were found to be the minimum acceptable for both configurations.

The boost phase control concept will be digital in nature and therefore all filtering, limiting, and stability augmentation will be by software. The three-degree-of-freedom simulation conducted for this study is a good representation of the actual flight software.

5.1 FR-3 GIMBAL REQUIREMENTS

Gimbal angle requirements for the FR-3 vehicle are primarily dictated by the center of gravity travels between launch and booster burnout. (See Figure 5-1.) Table 5-1 presents the overall gimbal angle requirements for conditions of all gimbaling engines and several combinations of fixed engines. The fixed engines are fixed at the indicated

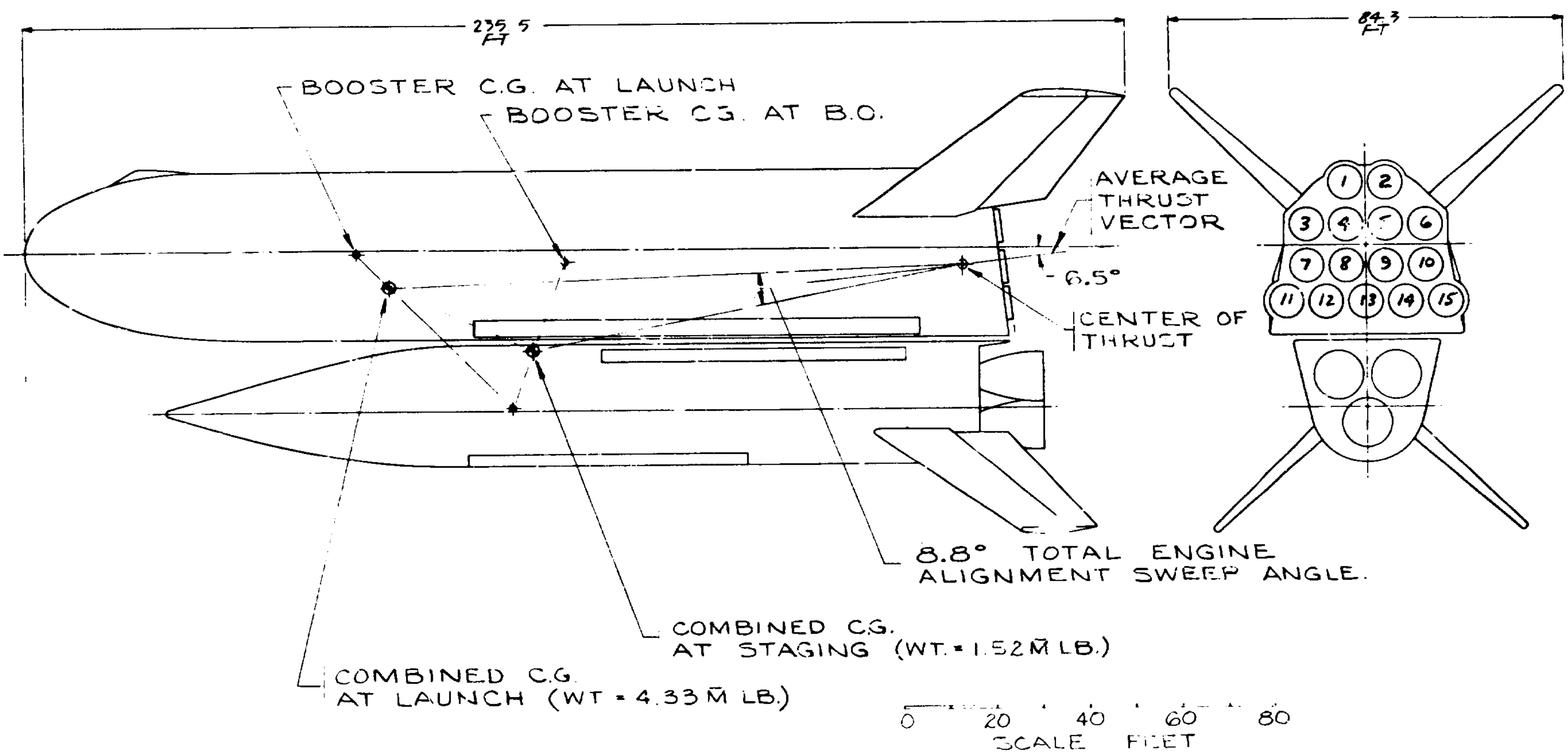


Figure 5-1. FR-3 Booster Thrust Alignment

cant angle. The total gimbal requirement for all cases is dictated by the engine out condition. The greater the number of engines which are fixed, the larger the gimbal angle requirement. This trend is due to the fact that less engines are available to cope with the center of gravity travel.

The cant angle was established by averaging the extremes to which the thrust vector must be oriented in order to pass through the center-of-gravity. In this manner gimbal angle requirements are minimized.

Table 5-1. FR-3 Gimbal Requirements

	Cant Angle (deg)	δ (deg)	δ_{Launch} (deg)	$\delta_{\alpha q}$ (deg)	δ_{BBO} (deg)	$\delta_{Engine\ Out}$ (deg)
All Gimballed	6.5	±5	-2.5	3	4.5	4.8
Engines 4, 5, 8, 9 Fixed	4.5	±6.5	-4.5	3	4.5	6.0
Engines 8, 9, 11, 12, 13 14, 15 Fixed	9	±7.5	-5.5	3	6.0	7.0
Engines 1, 2, 3, 4, 5, 6 Fixed	5	±7.0	-5.0	3	5.0	6.5

For the various combinations of fixed engines, the cant angle varies from 4.5 deg to 9 deg due to the variations in the effective center of thrust for both the fixed and gimbaled engine blocks. As the effective center of thrust moves away from the vehicle center of gravity, the thrust vector angle moves between large values. Since the cant angle is the average between the thrust vector extremes, the cant angle increases for an effective center of thrust which is moving away from the center of gravity.

5.2 FR-4 GIMBAL REQUIREMENTS

Gimbal requirements for the FR-4 vehicle are dictated by engine-out considerations since center of gravity variations from launch to burnout are negligible. The maximum αq gimbal requirements are for the limited minimum load conditions. The gimbal angle requirements for various flight conditions are:

Cant Angle	δ	$\delta_{Liftoff}$	$\delta_{\alpha q}$	$\delta_{Burnout}$	$\delta_{Engine\ Out}$
0 deg	±5 deg	±0.1 deg	3 deg	±0.1 deg	± 5 deg

SECTION 6

SUBSYSTEMS

6.1 MECHANICAL/ELECTRICAL SUBSYSTEMS

This section presents the types and characteristics of subsystems selected for the FR-3 and FR-4 launch and reentry vehicle systems. The subsystems are discussed in greater detail in Volume V of this report. The subsystems include electrical power generation and distribution, aerodynamic control, environmental control and life support, and hydraulic power generation and distribution. Similarity exists between the two and three-element systems (booster-to-booster and orbiter-to-orbiter) in the environmental control and life support subsystem and in the electrical power generation and distribution subsystem. Similarity of concept is retained for the aerodynamic control subsystem and hydraulic power generation and distribution subsystem but loads and equipment sizes are different because of the difference in vehicle mass properties.

6.1.1 ELECTRICAL POWER GENERATION AND DISTRIBUTION

6.1.1.1 Booster. Ascent and descent power is provided by rechargeable Ni-Cd batteries. The batteries were sized to provide 600 W-hr electrical energy based on a nominal power of 1300 watts and an energy density of 15 W-hr/lb. The batteries are removed after each flight and stored in a recharging mode. Power during cruise and landing is provided by engine-driven generators. Some of the power from the generators is converted to supply the baseline dc loads; the remainder 115/200 Vac, 400 Hz for aircraft mode peculiar loads. The generators supply 23 kVA, including a dc load of 2000 watts.

6.1.1.2 Orbiter. Ascent, orbital, and entry power are supplied by two 4.5 kW fuel cell modules operating in parallel throughout the flight. The peak orbital load of 3800 watts can be supplied by either fuel cell in the event of a fuel cell module failure. Should a second failure occur, a remotely-activated, Ag-Zn battery will provide power up to 2 hours to permit safe return. The fuel cell reactants (H_2 - O_2) are stored supercritically with 100 percent redundancy in reactant and tankage.

Power during powered flight following entry is supplied by engine-driven generators as in the booster case. Power requirements are essentially the same as for the booster.

Fuel cell powe tributed to using equipment through a central protected bus. Each fuel cell module is isolated by blc ing diodes. Feeder lines to using equipment are protected by remote reset, solid state circuit breakers located at the bus.

6.1.2 AERODYNAMIC CONTROL SUBSYSTEM. The aerodynamic control subsystem for both orbiters and boosters utilizes primary and secondary control subsystems. The primary subsystem includes elevons (or afterbody flaps), ruddervators, and wing spoilers. Secondary flight control is supplied by wing trailing-edge flaps. The control subsystem loads during reentry were based on a surface deflection of ±15 deg and a maximum deflection rate of 30 deg/sec at a dynamic pressure of 300 lb/ft^2. These are most conservative conditions and future study will probably permit a decrease in control subsystem requirements. Three independent hydraulic subsystems supply power to the primary flight controls. Three hydraulic actuators are used to position each control surface. This is a fly-by-wire subsystem and command signals include triple redundancy with monitoring to detect failures. Secondary control is provided by two of the three independent hydraulic subsystems.

6.1.3 ENVIRONMENTAL CONTROL/LIFE SUPPORT (EC/LSS). These subsystems are common for both the two and three-element systems and were designed for a baseline seven-day mission with excursions up to 30 days.

6.1.3.1 Booster. Because the basic booster flight mission is less than 1.5 hours duration, the EC/LSS can be of extreme simplicity. The proposed subsystem requires no active control of O_2 or CO_2. The cabin compartment, designed for low leakage, will be secured with sea level pressure and gas composition. During launch the cabin pressure regulator will allow cabin pressure to decay to 10 psia and to repressurize to atmospheric pressure on descent. Oxygen enrichment and/or use of gen masks will be required unless cruise-back is at an altitude less than 12,000 ft. Thermal control is by virture of thermal inertia of equipment and the water coolant loop during the nine-minute ascent and descent phase. The steady state thermal load of 5450 Btu/hr during cruise is rejected from the water coolant loop to atmosphere by a ram air heat exchanger.

6.1.3.2 Orbiter. This subsystem provides a shirtsleeve environment for a crew of 2 at 10 psia with an oxygen partial pressure of 2.7 psia. Pressurization and compositional control is provided from supercritically stored N_2 and O_2 and a two-gas sensing and control unit. Control of solids and odors is accomplished through the use of particulate and activated charcoal filters. No problem is anticipated with trace contaminant buildup because of the short mission duration. Lithium hydroxide beds are used for CO_2 control. Standard Apollo canisters are considered applicable. Relative humidity within the c v compartment is controlled to prevent condensation on cabin walls or to a maximum of 60% by use of a dehumidifying heat exchanger with centrifugal water separation.

The nominal steady state orbital heat load of 11,300 Btu/hr is transferred from the internal water coolant loop to an external freon loop and rejected through a space radiator. Sublimators are used to handle peak or abnormal loads. Thermal control during the ascent phase prior to obtaining use of the radiator is by thermal inertia of the equipment and water in the coolant loop. The jet engine fuel stored for the flyback engines is used as a heat sink during entry, descent, and landing.

The water management section collects fuel cell water, condensate, urine, and used wash water and provides temporary storage to permit choice of time of dump. Part of the fuel cell water is used for drinking and food reconstitution. The remainder can be used in the thermal control subsystem for added sublimator water or else dumped. The waste management subsystem provides air entrainment collection of urine and feces. The urine is passed to the water management subsystem. The feces are collected in a spin-type collector and vacuum dried. Whole body sponge bathing is provided by the personal hygiene section. Modular add-on features permit accommodation of missions in excess of thirty days.

6.1.4 <u>HYDRAULIC POWER GENERATION AND DISTRIBUTION</u>. Because of the wide disparity between aerodynamic control surface loads and vehicle electrical loads, a separate power generating source was selected for the hydraulic subsystem. An H_2 - O_2 fueled turbopump unit was selected with fuel supplied from main propulsion residuals. The fuel supply was integrated with the orbiter attitude control propulsion subsystem (ACPS) in a manner to assure delivery to the APU at 100 psi. A separate propellant pressurization subsystem is required for the booster since no ACPS is installed on the booster. For the three-element system, the unit was sized to provide 318 horsepower with an energy requirement of 707 and 4300 horsepower-minutes for the booster and orbiter respectively. The two element system requires 560 h.p. and 1260 h.p. min. for the booster. A 190 horsepower unit and 1680 horsepower-minutes are required for the orbiter. Three APUs are provided for each vehicle and each drives an independent hydraulic circuit. All three circuits provide power to the primary flight controls. Secondary controls are serviced by two of the hydraulic circuits. Each subsystem was sized to provide 50% full hinge moment at full rate to meet fail-operational/fail-safe-criteria.

6.2 INTEGRATED ELECTRONICS

The integrated electronics subsystem is covered in detail in Volume VII of this report. The system is configured with a goal of lowering operating costs. Autonomous operation from checkout through countdown, flight, and landing minimizes support required from extensive ground operations. This lowers cost but creates new requirements in the vehicle s electronics subsystem, including multipurpose displays, computers, and data transfer.

Computer-driven cathode ray turbe displays enable the crew to control and check out the vehicle. Conventional switches are replaced by fewer multifunction pushbuttons with computer control. Checkout, display generation, mission management, and autonomous navigation create computer and software requirements far exceeding those of Apollo. Multiprocessors, because of low weight and power, appear to achieve the objectives best.

To minimize wire bundle and connector complexities, a digital multiplexed data bus is used for most data transfer. Data bus concepts include uniform interfaces for all subsystems and provide flexibility, low weight, and high reliability.

Subsystems used for guidance, navigation, and control are conventional, with the choice between sensors made on the basis of development status, accuracy, and probable reliability. The need to land at any 10,000-foot runway requires an automatic landing subsystem first utilizing the present airline instrument landing system (ILS), and switching to a scanning beam system when it becomes available.

Onboard checkout was examined with particular emphasis on electrical power generation and the life support/environmental control subsystems. It was found that once a vehicle is structured with an integrated electronics subsystem containing powerful digital computers, flexible displays, and a multiplexed data subsystem, the inclusion of onboard checkout is quite feasible. It will not burden the vehicle with a large number of additional transducers, wire bundles, or special switching networks.

The weight bogies currently allowed for the Phase A vehicle avionics have been reflected in the Summary Weight Statements of Section 4.2.

6.3 LANDING GEAR

The landing gear of the FR-3 and FR-4 elements, covered briefly in this volume, is covered in more detail in Volume V.

All landing gears are state of the art, with tire sizes as used on present day aircraft. Ground flotation capability is for heavy load Z1 class landing fields at the design landing weights of the vehicles as summarized in Section 4.

Braking for landing on the 10,000-ft runway requires the use of a 50-foot diameter drogue chute in the FR-3 booster, allowing the use of minimum weight brakes.

6.4 WING ACTUATING SUBSYSTEM

The wing is operated by screwjacks driven by hydraulic motors via gear boxes. The system is a derivative of the F-111 system where the dual jackscrews have a synchronizing interconnect between gear boxes. The space shuttle wing actuators will be located aft of the pivot points for convenience of installation. Figure 6-1 shows a sketch of the typical arrangement.

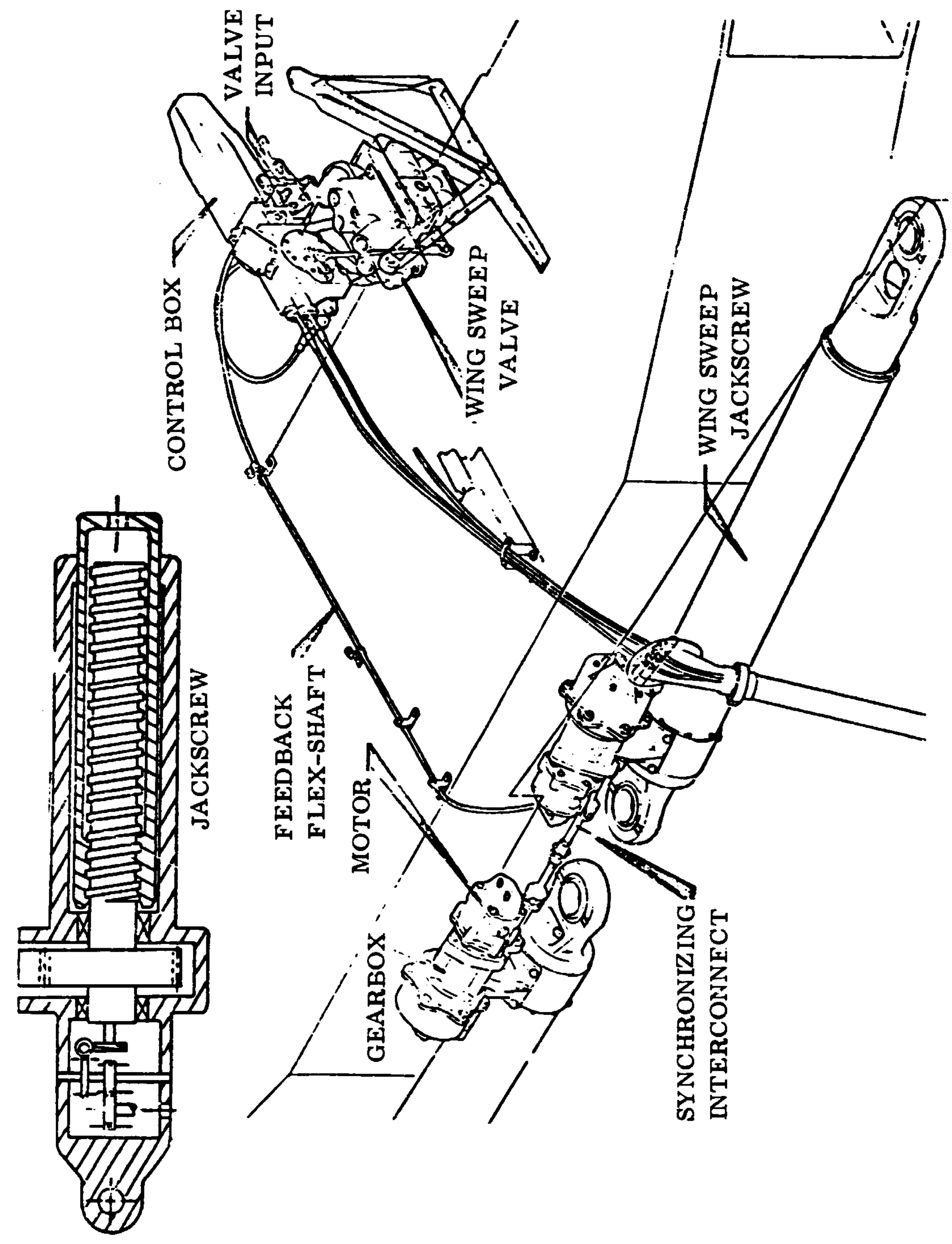


Figure 6-1. Dual Screw Wing Actuation Concept

6.5 STAGE SEPARATION

The final separation subsystems presented below for the FR-3 and FR-4 configurations were selected in preliminary tradeoffs between candidate subsystems. The paramount criteria used in the selection were reliability and safety.

6.5.1 FR-3 STAGE SEPARATION. The FR-3 separation subsystem was selected from the initial concept definitions presented in Volume III, Section 2.5. The longitudinal differential-drag concept was selected as being the simplest and most reliable. Initially a fully passive system was proposed; however, the addition of a nose jet on the booster element was deemed necessary in view of the weak aerodynamic environment at the nominal staging point. Preliminary calculations using analytically-obtained aerodynamics data indicate that the differential acceleration (less than 2 ft/sec/sec) was not sufficient to obtain the desired separation velocity at disengagement.

6.5.1.1 FR-3 Stage Separation System Description. As envisioned, the booster elements would thrust to propellant depletion and initiate engine cutoff.

Separation as currently conceived, is initiated when the booster thrust has reached a "commit" level by releasing the forward and aft attach points and firing a solid propellant jet located in the booster's nose. The booster is then free to slide aft along the orbiter's skinline by means of a pair of booster-mounted rails and orbiter-mounted slides.

As the booster moves aft under the combination of aerodynamic and jet-induced drag, the nose jet plume will begin impinging on the orbiter's thermal protected lower surface.* This additional impingement load will augment the interference aerodynamic loads in providing lateral-rotational clearance as the last slide leaves the rail. The nose jet thrust is terminated just prior to disengagement such that the separation velocity at disengagement (circa 40 fps) is sufficient to provide enough clearance to prevent post-disengagement collision and provide for sufficient clearance at orbiter engine start.

The booster, upon disengagement, initiates a reorientation maneuver and trims to high lift (belly up) in an attempt to minimize its apogee altitude. The orbiter stabilizes aerodynamically (or with its ACPS) and begins its main engine start sequence.

6.5.1.2 Parameter Selection. Table 6-1 was constructed utilizing the computed (but minimal) aerodynamic drag differential and the desired performance. The low-contamination jet I_{sp} was 220 sec and the installation mass fraction was 0.80.

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*The adverse effects of this impingement can be mitigated by selecting a low-contamination solid propellant and sacrificing I_{sp}.

Table 6-1. Selected Parameters, FR-3

Rail Length, ft	100
Disengagement Elapsed Time, sec	5
Separation Differential Acceleration, g	1/4
Disengagement Velocity, fps	40
Nose-to-tail Elapsed Time, sec	7-1/2
Jet Thrust, lb	150,000
Jet Firing Time, sec	5
Jet Propellant Weight, lb	3,400
Jet Installation, lb	4,250

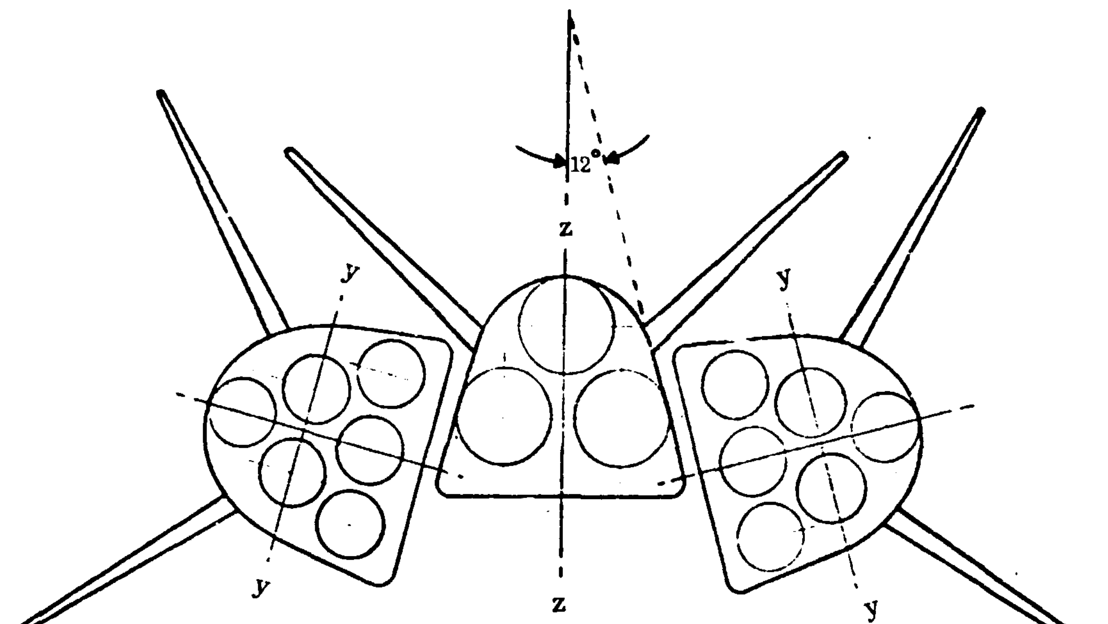
6.5.1.3 Conclusion. The parameters presented in Table 6-1 give adequate performance at the nominal separation point. Abort separation at much higher dynamic pressures was not examined.

6.5.2 FR-4 STAGE SEPARATION. Although not specifically analyzed, the FR-4 launch configuration was examined to determine the applicability of the FR-1 aft-hinge staging separation subsystem described in Volume V, Section 10. The lack of symmetry in the FR-4 vehicle* cluster will definitely produce degraded stage separation subsystem performance when compared with FR-1. Although stage separation at the nominal staging point appears adequate, abort separation under much higher aerodynamic pressures appears questionable.

6.5.2.1 Pertinent Configuration Differences. As configured, the FR-4 vehicle cluster consists of the two boosters nestled along the orbiter's sides as close as the orbiter's tail will allow (Figure 6-2). The booster y-y axis is aligned 12 deg to the orbiters z-z axis. The aft-hinge separation subsystem would attach to the orbiter in approximately the same manner as FR-1. This places the attach points approximately 114 ft aft and 13 ft below the orbiter's mass center.

Presuming an aft-hinge separation subsystem performance similar to FR-1 (Volume V, Section 10), the lack of configuration symmetry can be readily assessed. Assuming the configuration is in trim prior to the initiation of separation (boosters at zero thrust, orbiter at partial thrust), the following effects can be anticipated as aft-hinge rotation is accomplished:

*This lack of symmetry was dictated in response to a customer requirement for complete payload accessability up to within seconds of launch.



Note: Early Version Engine Configuration Shown

Figure 6-2. FR-4 Launch Configuration

a. The lift and drag components will increase with aft-hinge rotation (i.e. with booster angle of attack).

b. The increased drag component will produce additive axial components at 13 feet below the orbiter's center of mass and a nose down moment.

c. The increased lift component along the orbiter's y-y axis will cancel. However, the increased lift component times the sine (12 deg) is additive and acts at 114 ft aft of the orbiter's center of mass and produces an additional nose down moment.

d. The resulting nose down rotation will be only partially offset by the orbiter's control engines. Additional moments about the booster's z-z axis (above those anticipated for the FR-1 configuration) must be absorbed by the linkage. These moments operating over rather short moment arms are likely to produce large linkage forces.

e. Positive angles of attack result in angles of sideslip on the boosters and could result in the booster colliding with the orbiter's tail and thus must be avoided.

f. The downwash from the orbiter tail is likely to yaw and roll the boosters as they depart. (This effect is nearly cancelled in the FR-1 arrangement but results in different trajectories for the two boosters.)

g. Although separation clearance is enhanced by throttling the orbiter's engines during release, the resulting reduction in orbiter control will aggravate the nose down rotation.

6.5.2.2 Conclusions. Preliminary calculations for the FR-4 configuration indicate that the resulting performance at the nominal separation point is probably adequate; however, much higher linkage loads are likely. Since every effect enumerated above is aggravated by higher aerodynamic pressure, abort separation under much higher aerodynamic pressures appears questionable.

SECTION 7

SAFETY AND ABORT

Safety and cost both require that the crew, passengers, payload, and the vehicle be returned intact to the launch site following abort with a high probability of success (i.e., at least 0.999 [1 loss max/1000 flights]).

The above requirement can be met with vehicles exhibiting the following basic characteristics:

a. Ability to achieve a once-around earth and return to launch site maneuver by the orbiter, and normal flyback to the launch site for the booster, following: 1) subsystem failure which is cause for abort, 2) two engines failed in the booster and one engine failed in orbiter in the early portion of booster and orbiter ascent phases, or 3) up to three engines failed in the booster and up to two engines failed in the orbiter in the latter portion of these boost phases.

b. Provisions to supress potential fire or explosion by isolating compartments containing fuels and oxidizer tanks with sealed bulkheads and diaphragms and by purging critical compartments with an inert gas to limit oxygen concentration to prevent formation of combustible mixtures.

c. Rocket engine reliability of 0.99 with rocket engine catastrophic ratio < 1 percent of total engine failures.

Mission aborts are not critical from an operational cost standpoint when compared with vehicle losses; however, the number of mission aborts can be controlled to a limit of ~30/1000 flights with nominal weight and cost penalties, with vehicles exhibiting the following characteristics:

a. Ability to maintain performance thrust-to-weight ratio (F/W) with one booster engine out (i.e., fail-operational/fail-safe for booster engines and fail-safe only for orbiter engines).

b. Provision for fail-operational/fail-safe for all active mechanical subsystems, and fail-operational/fail-operational/fail-safe for avionics subsystems.

c. Provisions for fail-operational/fail-safe for orbiter engines for on-orbit maneuvers.

Safety in flight operations of the FR-3 and FR-4 vehicle concepts is achieved with intact abort.

The basic approach for treating the majority of failures is to provide redundancy. This redundancy can be provided to yield a fail-safe system or a fail-operational system. For the FR-3 and FR-4 vehicles, mechanical/electrical subsystems have fail-operational/fail-safe characteristics and electronic subsystems have fail-operational/fail-operational/fail-safe characteristics. When a failure occurs in a subsystem with fail-operational characteristics, the situation is not considered an abort because the mission can be completed.

When a failure occurs in a critical system at the fail-safe level, it is necessary to go to an abort procedure. The most critical abort situations are those occurring during ascent since the problem is basically one of finding suitable landing sites. The procedure for such abort situations is that the orbiter continue on to orbit and go once around the earth and perform a glide return to the launch site.

The FR-3 vehicle (with a 15-3 booster-orbiter engine arrangement) has a relatively low number of mission aborts because it incorporates fail-operational/fail-safe provisions for engines in the booster. The FR-3 can achieve staging with one engine out because of the 7% overthrust capability of the booster engines which allows the performance F/W to be maintained. The FR-3 can achieve intact abort with two engines out at liftoff. After liftoff, more engines can be out and intact abort is still possible.

The FR-3 and the FR-4 do not have fail-operational/fail-safe capability for engines from staging to orbit because the weight penalty to provide a 50% overthrust in the three engine orbiter is prohibitive. The FR-3 and FR-4 orbiters do have fail-operational/fail-safe capability for all on-orbit maneuvers.

The FR-4 (with a 9-3-9 booster-orbiter-booster engine arrangement) can achieve staging with one engine out; however, there is a weight penalty because a 13% overthrust is required. This amount of overthrust is outside the presently designed engine propellant utilization (PU) control capability and uprated or added engines are required with associated weight penalties. Because the FR-4 with the 9-3-9 arrangement does not have fail-operational booster engines, mission losses are higher than for the FR-3. The FR-4 has fail-safe provisions for booster and orbiter rocket engines and basically the same abort procedures as the FR-3. The intact abort success probability (safety) of the FR-4 is therefore approximately the same as the FR-3.

Both the FR-3 and FR-4 vehicle concepts incorporate inert gas purging provisions for fuel tank surrounds, rocket engine bay, and payload bay to suppress any potential fire or explosion resulting from leakage and subsequent vaporization of fuel (LH_2). Purging with an inert gas is provided during ascent and descent to an O_2 concentration < 2 % by volume for these areas.

Sealed (gas tight) bulkheads between compartments containing fuels and/or oxidizers keep them separated, and diaphragms seal off hot air and isolate hot surface ignition sources.

An intact abort success (safety) goal of 0.999 (1 loss/1000 flights) and a mission success goal of 0.97 (30 aborts/1000 flights) was established for the space shuttle system. These goals can be achieved or closely approached with specific abort procedures and design requirements incorporated in vehicle operations and design.

7.1 HAZARD ANALYSIS

Safety and cost in reusable launch vehicles are the real drivers leading to requirements for a high probability of successful abort. Again, the crew, passengers, payload, and the vehicles must be returned safely intact to make the reusable launch vehicle concept economically attractive. A safety and abort analysis conducted for the space shuttle was accomplished on this premise (i.e., intact abort).

The analysis sought the answers to the basic questions:

How safe should (can) the system be?

What makes it unsafe?

What action must be taken to change an unsafe situation into a safe operation?

How is safety improved?

What are the interfaces of safety with weight, cost, operations, and mission success?

These questions were answered by conducting a gross failure and mission termination analysis and hazards analysis with consideration given to:

a. Probability of occurrence of mechanical failures of subsystems, propulsion subsystems, and structure during the mission.

b. Abort options following these failures.

c. Availability of landing sites for aborted flights for several launch azimuths.

d. The hazard potential (fire or explosion) of the stored propellants.

intact abort and redundancy.

f. The interplay of safety, cost, and mission success.

The analysis led to:

a. An assessment of safety and mission success goals.

b. Definition of abort procedures for the various flight trajectory phases.

c. Design requirements for the vehicle.

d. Design requirements for safety (fire and explosion hazard).

The effect of engines on safety and missions was studied using different assumed engine reliabilities and the application of fail-operational/fail-safe to engines in both boosters and orbiter.

7.1.1 FAILURE AND MISSION TERMINATION ANALYSIS. The gross failure and mission termination analysis and a hazards analysis were conducted on an early FR-1 configuration. The basic objective was to establish, in a gross way, abort philosophies and mission termination procedures, subsystem design requirements, and requirements for redundancy. The data and information developed are qualitatively applicable to FR-3 and FR-4. The basic once around abort and immediate abort concepts were developed from this analysis.

The data and information developed were used to guide the vehicle and vehicle subsystem design tasks. The analysis also provided data and information to be used in investigation of propellant dumping.

The approaches used in the gross failure and mission termination analysis for mechanical/electronic failures were:

a. Establish failure rates from historical data and estimates.

b. Apply weighting factors to account for differences in mission phase stresses.

c. Determine points in the mission when failures are most likely to occur (distribute failures into the mission phases).

d. Determine the consequences of major element failures and investigate abort procedures which lead to successful intact recovery of vehicles.

e. Provide design improvements and develop operational abort procedures.

7.1.1.1 Consequences of Failures and Abort Philosophy. Each mission phase was examined and the failures were evaluated to determine the effect on the vehicle and what corrective action in the way of changes in operation or design (e.g., adding an engine) could be taken if the consequences of the failure meant an aborted mission or loss of life. Emphasis in the failure analysis was on crew and vehicle recovery (crew recovery from a safety standpoint and vehicle recovery from an economic standpoint).

Subsystems were examined to ensure that no single failure resulted in loss of life. This was done using backup subsystems to accomplish safe return. Investigations were also made to eliminate or reduce the number of time critical failures.

The abort philosophy used in the failure analysis is summarized below.

a. First priority: Save crew and passengers.

b. Second priority: Save reusable vehicles.

c. Landing site priority:

   1. Return to launch site.
   2. Once-around and return to launch site.
   3. Continental U.S. landing sites.
   4. Available landing sites.
   5. Survival/rescue.

d. Prior to docking: Return by earliest (low stress) route.

e. After docking: Continue to land.

7.1.1.2 Abort Glide Footprints. One of the consequences of failure is availability of landing sites.

Figure 7-1 shows the abort/glide footprints for the typical orbiter element when launching from ETR, for launch azimuths 0 to 90 deg. A hypersonic lift drag ratio of 1.9 is assumed. Example footprints are shown for abort velocities of 6,000, 10,000, and 15,000 fps for a direct injection into a 55-deg inclination orbit, launch azimuth about 37 deg. For this orbit, landing sites are available along the Eastern seaboard; however, more easterly launches are entirely over water. Landing site availability is therefore strongly dependent on launch azimuth and abort velocity.

The failure situation, effect, corrective action, and results were investigated for each major failure that has a bearing on safety and vehicle recovery. Specific abort procedures and design requirements necessary to provide a safe operational vehicle were defined. These are presented in Sections 7.2 and 7.3.

7.1.2 POTENTIAL HAZARD (FIRE OR EXPLOSION). The hazard of fire and explosion in the FR-1, FR-3, and FR-4 vehicles is greater than on a conventional JP-fueled aircraft. The reason for this is the large quantity of H_2 which has a very broad flammability range and a low ignition energy. There is always the possibility of a leak somewhere in the LH_2 subsystem which can generate hydrogen in gaseous form (GH_2). For the ascent phase of flight and entry, ambient air containing oxygen can enter the interstitial space where GH_2 could form.

Since the GH_2 concentration required to form a combustible mixture with air is very broad (4 to 75% by volume), it must be assumed that a small cryogenic leak of LH_2 could subsequently vaporize and form a mixture whose concentration is within the flammable range. The gaseous mixture is not hypergolic, however, because of the low ignition energy required (0.019 millijoules, 1/10 that for JP/air) it must be assumed that an ignition source in the form of electrical spark (chaffed wires/or static discharge) is always present. Sources of ignition in accidents reported (except Apollo) have not been identified conclusively; therefore, no potential source can be disregarded. During entry, stratified layers of hot air above the spontaneous ignition temperature could enter the fuel tank surrounds and ignite any combustible mixture that may be present.

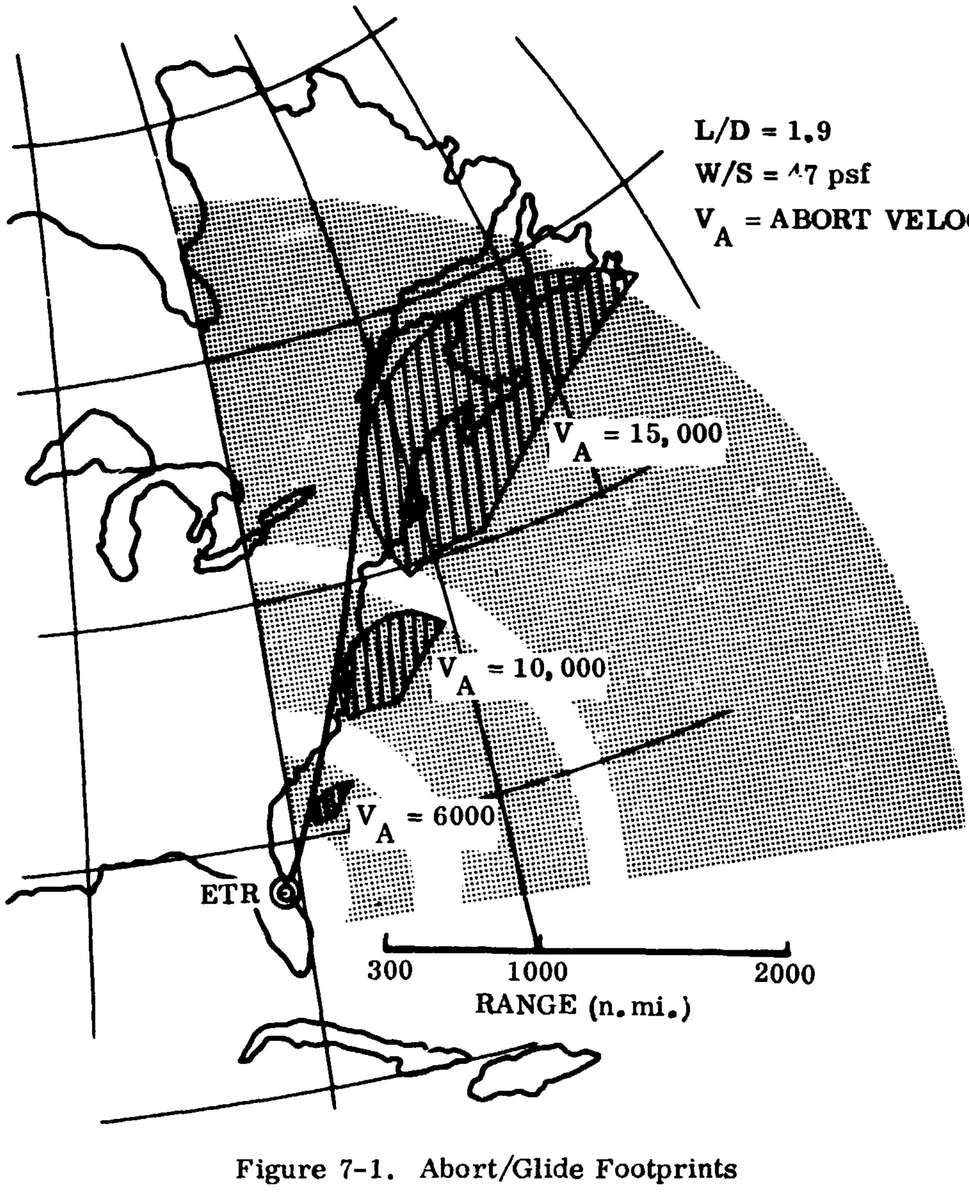


Figure 7-1. Abort/Glide Footprints

Table 7-1 shows a comparison of the relative hazard of a reusable launch vehicle and an airplane. The table shows that hazard is greater for the reusable vehicle because of the type of propellant, the amount of propellant, and the vehicle size compared with an F-106 airplane. Figure 7-2 shows schematically the hazard potential (sources of fire or explosion) in the vehicle for an early FR-1 design. The vehicle carries quantities of LO_2, JP-4, and LH_2. The major source of fire or explosion hazard is the stored hydrogen. Design approaches that will preclude, or minimize the probability of occurrence of hazardous conditions are discussed in Section 7.3.

Table 7-1. Hazards Comparison - Reusable Launch Vehicles/Airplanes

•Propellant and Fuel Characteristics (1 Atmos.)

	H_2/O_2	H_2/Air	JP-4/Air
Flammability Range (% by Volume)	4-94 (520°R)	4-75 (520°R)	0.7-4.8 (520°R)
Ignition Energy Required (Millijoules)	0.003	0.019	0.2

•Vehicle Characteristics

Propellant/Fuel	Reusable Launch Vehicle* (H_2/O_2 Propellant)	Aircraft (F-106) (JP-4 Fuel)
Weight at T.O.	* 2 1/2 Million lb	9,018 lb
ρ (Density)	At mixture ratio 7:1 average density = 24 lb/ft^3	48.5 lb/ft^3
Volume	≈100,000 ft^3	186 ft^3

*Early FR-1 vehicle

7.2 ABORT PROCEDURES

It is necessary to develop operating procedures (abort) to employ when a failure occurs at the fail-safe level. The basic procedure for such abort situtations is that the orbiter continue on to orbit and go once-around the earth and perform a glide return to the launch site. A schematic of this maneuver is shown in Figure 4-6. For the baseline 55 deg orbit, the crossrange required after entry to return to ETR is about 800 n.mi.

The booster and orbiter elements will complete the boost phase reaching a staging point with booster propellants depleted and orbiter propellant tanks full. The vehicles separate and the booster returns to the launch site in a normal manner while the orbiter continues once-around the earth and returns to the launch site. The once-around abort procedure reflects a high probability of successful intact abort from all failures of a mechanical nature, such as engine failure or gimbaling. Section 4.1 discusses thrust-to-weight requirements and ΔV requirements with engine out (see Figure 4-7). The figure shows that sufficient ΔV propellant is available for once-around with an orbiter engine out. In addition, there are rarely occurring failure situations which require immediate abort. Typical of these are structural failure,

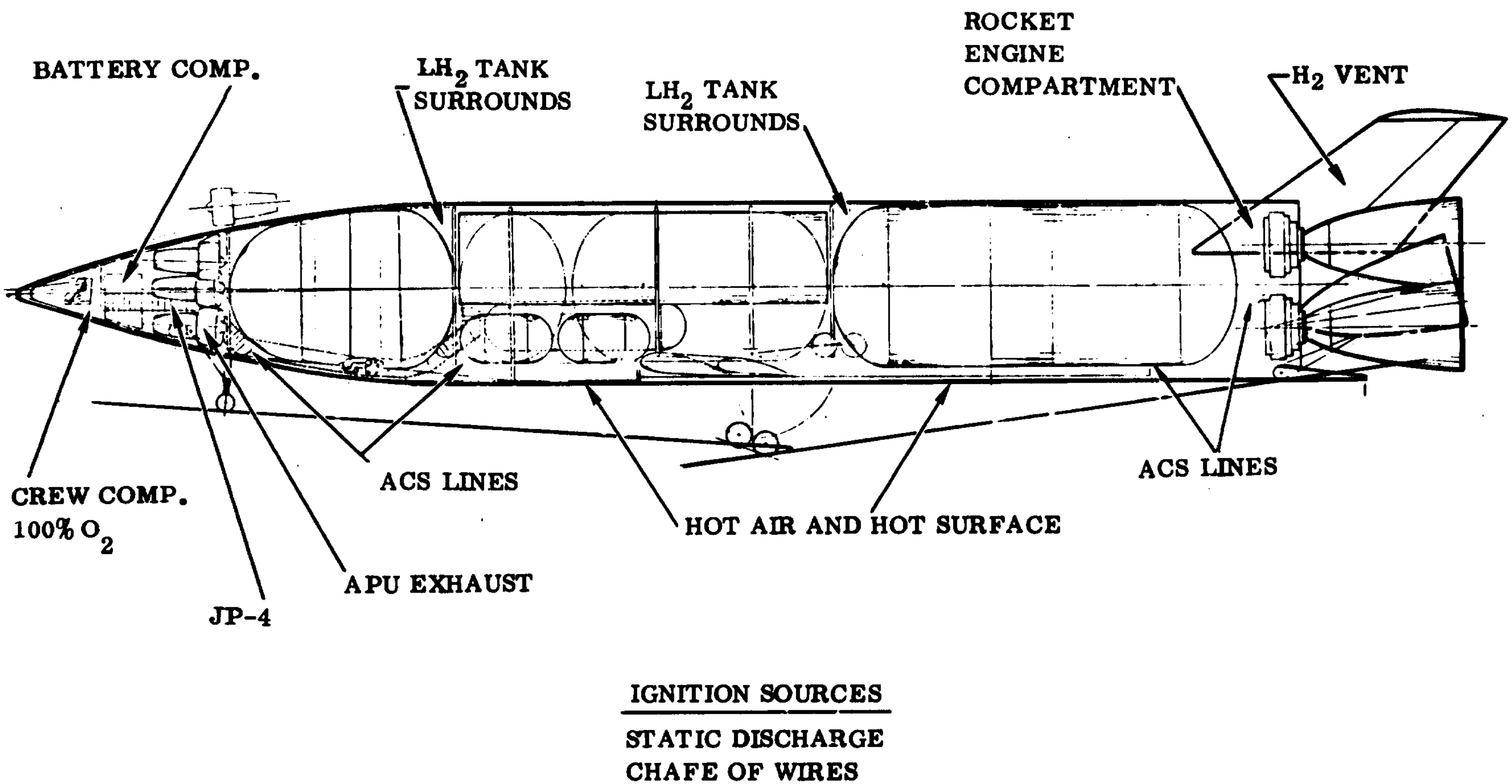


Figure 7-2. Hazard Potential

thermal protection system failure, time critical failures, hard-over gimbal, or catastrophic situations such as a fire which may require early separation.

A gross analysis of the requirement for propellant dumping established that the need for dumping is limited to special cases of low frequency of occurrence, due to a structural or thermal protection system failure where it is desirable to limit energy buildup (velocity) or due to time-critical failure (fire) where it is desirable to terminate boost (abort).

Ditching is not considered a satisfactory solution because of cost, even if the orbiter vehicle incorporated the penalties required for a safe water landing. Vehicle losses must be held to < 1/1000 to make the reusable concept economically attractive.

7.3 DESIGN REQUIREMENTS

7.3.1 SUBSYSTEMS AND MAIN PROPULSION. This section summarizes the vehicle design requirements resulting from the hazard analyses conducted to date. Vehicle and vehicle subsystem design requirements which will enable accomplishment of the abort procedures specified in Section 7.2 are:

Following failure of:

a. Any Rocket Engine:

    Design to keep going

    Design liftoff thrust/weight ≥1.16

    Design orbiter for once-around, land at launch site

b. Propellant Feed:

    Provide backup in one second (propellant feed tank pressurization)

    Design propellant feed backup for full thrust capability

    Design to deplete boosters equally before staging (for FR-4 throttling opposite booster engines)

    Design for once-around mission:

        Use main propellants

        Use or dump orbit maneuvering propellant

c. Thermostructure:

    Design for any engine out in any part of the trajectory

    Design for any turbojet out during flyback

    Design for failure transients

Provide redundant attachments

Provide multiload path design

d. Subsystems:

Design subsystems for fail-operational/fail-safe operation

Design provisions such that after failure:

Guidance Power Hydraulics Electrical ECS ACS Avionics	Switch to backup Complete mission After second failure - return to launch site

Function

Gimbal Aero control surfaces Wing deploy Turbojet deploy Landing gear deploy Inerting systems	Perform function - may be at reduced rate

7.3.1.1 Implications of Uprated Thrust Rating. The propellant utilization (PU) systems operating in the vehicle significantly reduces residual propellants which improves vehicle performance. This subsystem monitors tanked mixture ratio and regulates engine mixture ratio to give full utilization of available propellant (simultaneous depletion of fuel and oxidizer). The rocket engine is designed to provide a range of mixture ratios for this purpose. The engine, so designed, will actually have an increased thrust capability at some fixed mixture ratio. This is because of the pump and thermal limits required in the engine design. This situation is shown in Figure 7-3. For the 400K engine currently defined by Pratt and Whitney, this capability is 108 percent of nominal.

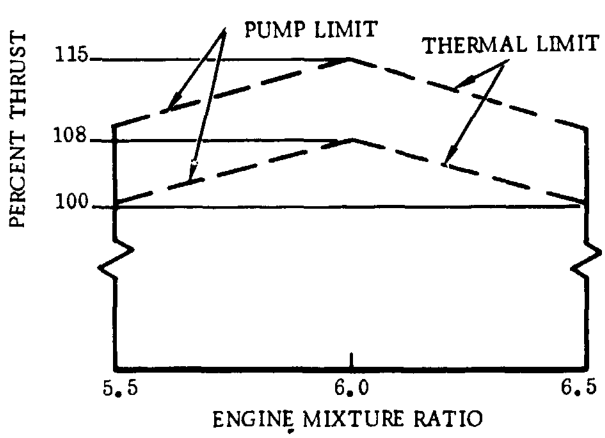


Figure 7-3. Engine Thrust vs. Mixture Ratio

Some overthrust capability is then inherent in the engine design. However, this inherent overthrust capability cannot be used for fail-operational purposes without significant performance penalty since no PU control is possible. To provide PU control capability at thrusts above nominal, the engine must be designed to operate beyond the nominal speed and temperature limits. Overthrust to 115 percent of nominal as defined in Figure 7-3 is thought by Pratt and Whitney to be feasible for a single flight. This capability would allow control of engine mixture ratio over the full design range at thrust up to approximately 7 percent. If engines are operated in this mode, some refurbishment is required after use.

The application of this principle (overthrust) to a 15-engine booster (FR-3) can be advantageous from a mission abort standpoint. For an engine MTBF = 15 (R = 0.99) mission aborts during liftoff to staging caused by engines alone can be reduced from 60/1000 flights to approximately two, if provision is made to reach staging with one engine out (fail-operational for booster engines). This is a significant gain since it can be accomplished with presently conceived engines. The amount of overthrust required is given by $\frac{1}{M - 1} = \frac{1}{15 - 1} = 7.2$ percent, which is approximately within the range of mixture ratios required for operation of the PU subsystem. The FR-3 incorporates this fail-operational provision for booster engines.

An alternate approach to operation of engines in an overthrust mode is to provide an extra engine and operate booster engines throttled. The fail-operational/fail-safe concept applied to the FR-3 15-3 engine arrangement can be accomplished by providing 16 rather than 15 booster engines and operating the 16 normally throttled to 93 percent. If one engine fails, the remaining 15 good engines are operated at 100 percent to maintain performance F/W.

7.3.1.2 Engine Effects on Safety and Mission Success. Engines introduce potential hazards due to loss of thrust and engine catastrophic failures. A quantitative assessment of the probability of engine catastrophic failure is not available, though Pratt and Whitney has estimated that one percent of all engine failures will be catastrophic, resulting in loss of the vehicle. Increasing the number of engines increases the probability of vehicle loss due to catastrophic failures, while performance failures (loss of thrust which may negate the once-around abort capability) diminish with increasing numbers of engines.

Table 7-2 presents a summary of the results of a study to determine the effect of rocket engines on safety and mission success in the FR-3. The table shows two cases which were investigated. One with fail-safe only provisions for booster and orbiter engines, and the other with fail-operational/fail-safe provided for the booster and fail-safe only provided for the orbiter.

As shown, mission aborts caused by engines alone are reduced from 72 to 14 with the application of fail-operational/fail-safe to booster engines. A 7 percent overthrust

Table 7-2. Effect of Providing Fail-Operational/Fail-Safe for Booster Engines for FR-3

Engine Configuration	Reliability Required	Vehicle Glow M (lb)	Engine Over Thrust Capability % Required	Losses /1000 Flights All Cases				
				Mission (Aborts)		Vehicle (Safety)		
				Calculated	Allowable	Calculated Engine Catastrophic Ratio 1%	Calculated Engine Catastrophic Ratio 0.1%	Allowable
15-3 Fail-Safe in Booster and Orbiter	0.99	4.33	0	92 (72)	30	1.33	0.70	.1
15-3 Fail-Operational in Booster; Fail-Safe in Orbiter	0.99	4.33	7*	33 (14)	30	1.33	0.70	1

*The presently conceived rocket engine provides for up to a 7 percent overthrust with full propellant utilization control

NOTE: Number in ( ) indicate losses caused by engines alone, remainder are for other causes which were determined from the failure analysis conducted for the FR-1, with gross estimates applied to account for updated subsystems.

level, some refurbishment is required. The cost of this refurbishment is yet to be determined, but it must be a low value to make the approach cost effective.

Table 7-2 shows that vehicle safety is influenced strongly by the engine catastrophic failure ratio. Reduction of this failure ratio to 0.1 percent is required to meet the safety goal.

An alternate approach to providing engine overthrust is to add an engine to the booster and normally operate booster engine throttled to 93 percent. There are weight and cost penalties associated with this approach.

A weight and cost comparison of fail-operational/fail-safe versus fail-safe and overthrust versus throttling is presented in Volume III. This comparison shows that fail-operational/fail-safe for the booster is cost effective, if engine refurbishment costs can be held down.

7.3.2 DESIGN FOR SAFETY (FIRE AND EXPLOSION HAZARD). Figure 7-4 shows schematically the requirements for design for safety. These design provisions are necessary in order to demonstrate vehicle operational safety which ensures that losses due to fire and explosion do not occur, or are minimal.

This initial assessment indicates that a solution to the fire and explosion hazard is to provide an inert gas to suppress any potential fire or explosion with a purge and positive pressure control system. The basic premise for the design evolves from consideration of several probabilities. The probability of occurrence of a small cryogenic leak of LH_2 or LO_2 is equal. The probability that both of these will occur at the same time (in any given flight) is remote. The probability of a line breakage resulting in a large flow of either LH_2 or LO_2 is remote. The probability of ambient air being present in the fuel tank surrounds is a certainty, unless a positive pressure is maintained in the tank surrounds so that ambient air containing O_2 cannot enter. If the pressurizing gas is inert and if the $(O_2) < 2$ percent by volume, combustion is not possibie even with a large H_2 concentration. The potential of fire or explosion from this cause is thus removed.

In summary, the requirements for design to minimize losses from fire or explosion are:

a. Provide sealed (gas tight) bulkheads between compartments containing fuels and/or oxidizers to keep them separated.

b. Provide diaphragms to seal off hot air and isolate hot surface ignition sources.

c. Provide purging with an inert gas during ascent and descent to an O_2 concentration < 2 percent by volume for LH_2 tank surrounds, rocket bay, and payload bay.

d. Provide O_2/N_2 crew and passenger compartment atmosphere.

e. Apply design practices to eliminate leakage potential and isolate electrical ignition sources.

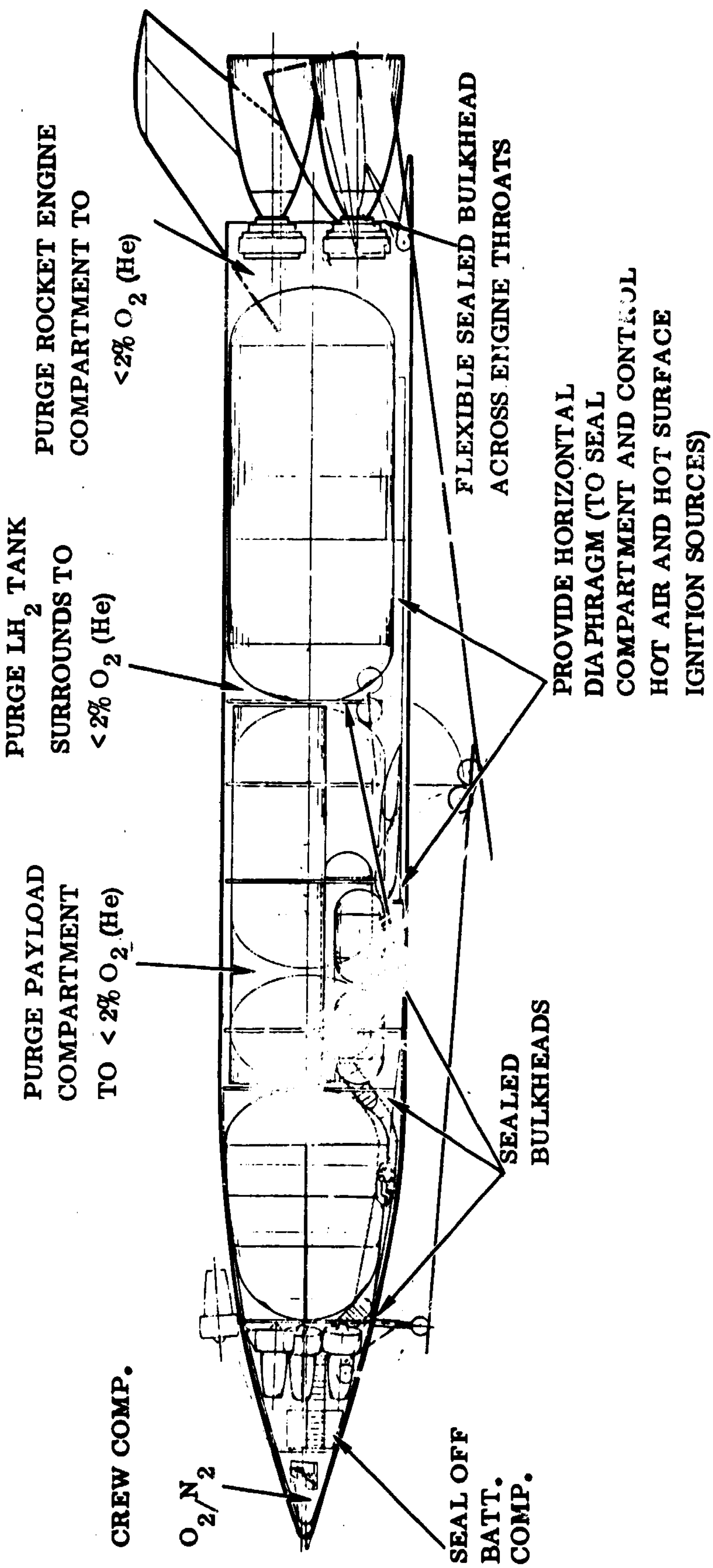


Figure 7-4. Design for Safety

SECTION 8

MISSION REQUIREMENTS AND ANALYSIS

This section presents the NASA mission traffic schedule, a description of each mission, and the performance capabilities of the FR-3 and FR-4 for each mission.

8.1 TRAFFIC MODEL

The Nominal Space Shuttle Traffic Model from the NASA Space Shuttle Task Group Report, Volume 1, 12 June 1969, is the basis for the model presented in Table 8-1. The time period of interest is 1975 through 1985. This model reflects an average annual launch rate of 51 flights per year. Table 8-2 summarizes the range of mission characteristics for the missions shown on Table 8-1. The frequency of mission type is:

Propellant delivery	44 (percent)
Personnel and cargo delivery	33
Propulsive stage and payload delivery	9
Experiment module delivery	6
Satellite missions	4
Short duration orbit missions Rescue missions	4

8.2 SPACE STATION/BASE LOGISTICS

The space shuttle transports cargo and personnel to and from a manned orbital Space Station and subsequently to a larger Space Base in low-altitude earth orbit. The cargo includes food, liquids, and gases in addition to both experiment modules and operational equipment. Personnel include trained astronauts and individuals who conduct specific scientific and technology experiments and operations. The shuttle logistics missions include long-lead-time scheduled resupply and crew rotations as well as discretionary flights. The routine logistics requirements for an orbital facility depend on the size of the facility and the type of experiments and operations being conducted at any given time. Typical requirements are summarized in Table 8-3 for a 12-man Space Station and a 50-man Space Base.

Table 8-1. NASA Mission Traffic Model

Mission	Year 75	76	77	78	79	80	81	82	83	84	85	Total Traffic
Logistics												
Space Station												
Personnel and cargo	4	4	4	4	4							20
Experiment Module	3	3	3	3	3							15
Space Base												
Personnel and cargo						20	20	20	20	20	20	120
Experiment module						3	3	3	3	3	3	18
Delivery of Propulsive Stages and Payload	7	1	8	3	4	6	5	2	7	5	3	51
Placement, Retrieval, Service, and Maintenance of Satellites	2	2	2	2	2	2	2	2	2	2	2	22
Delivery of Propellant												
LH_2				36	36	24	24	24	24	24	24	216
LO_2				6	6	4	4	4	4	4	4	36
Crew and Cargo (Lunar Mission				6	6	6	6	6	6	6	6	48
Short Duration Orbit												
Space Rescue	2	2	2	2	2	2	2	2	2	2	2	22
Total Flights Per Year	18	12	19	62	63	67	66	63	68	66	64	568

Table 8-2. NASA Mission Characteristics

Mission	Orbit Altitude (n.mi.)	Orbit Inclination (deg)	On-Orbit ΔV (fps)	Mission Duration (days)	Launch Azimuth (deg)
Space Station Space Base Logistics	200 to 300	28.5 to 90	1000 to 2000	7	90 to 180
Delivery of Propulsive Stage and Payload	100 to 200	28.5 to 55	1000 to 1500	7	90 to 41
Placement, Retrieval, Service, and Maintenance of Satellites	100 to 800	28.5 to Sun Synch	1000 to 5000	7 to 15	90 to 188
Delivery of Propellants	200 to 300	28.5 to 55	1000 to 2000	7	90 to 41
Short Duration Orbit	100 to 300	28.5 to 90	1000 to 2000	7 to 30	90 to 180
Space Rescue	270	55	2000 to 5000	7	41

Table 8-3. Routine Logistics Requirements

Requirements		Space Station (12 men)	Space Base (50 men)
Per Quarter			
Cargo Up	(lb)	12,000	48,000
Personnel Up	(men)	12	50
Cargo Down	(lb)	7,000	28,000
Personnel Down	(men)	12	50
Per Flight (Based on Traffic Model)			
Cargo Up	(lb)	12,000	9,600
Personnel Up	(men)	12	10
Cargo Down	(lb)	7,000	5,600
Personnel Down	(men)	12	10

The routine logistics mission is defined as a 55-degree inclined circular orbit at a 270 n.mi. altitude, with rendezvous within 24 hours of launch. The main propulsion system on-orbit ΔV design requirement is 1800 fps and the ACS ΔV design requirement is 200 fps.

Figure 2-1 Section 2 has presented the mission profile showing main engine burns. The main propulsion system ΔV requirements are shown in Table 8-4. The 1800 fps requirement shown contains an allowance of 200 fps for insertion dispersions and out-of-plane errors, and 480 fps for FPR and contingencies.

The ACS furnishes limit cycle attitude control to +45 deg while in orbit hold or during orbit transfers, orientation to ±5 deg prior to each orbit maneuver burn and during rendezvous, roll control to ±5 deg during each maneuver burn, ΔV to transfer from 260 n. mi. to 270 n.mi. and to rendezvous, dock, and undock and orientation to ±2 deg during entry. The total ACS ΔV for the mission is 157.9 fps. The ΔV requirements for each mission phase are shown on Table 8-5. The design requirement of 200 fps indicates a reserve and dispersion allowance of 42.1 fps.

8.3 DELIVERY OF PROPULSIVE STAGES AND PAYLOAD

The space shuttle delivers propulsion stages and payloads to low earth orbits to support a variety of missions within earth orbit and out of earth orbit. Such missions range from high altitude Earth satellites to unmanned planetary probes. In this

Table 8-4. Main Propulsion ΔV Requirements

Maneuver	ΔV (fps) Main Propulsion
Circularize at 100 n.mi.	110
Transfer to 260 n.mi.	280
Circularize at 260 n.mi.	280
Entry	450
Flight performance reserve and contingencies	480
Insertion dispersions and out-of-plane errors	200
Total Subsystem ΔV	1800

operational mode, the shuttle delivers both the payload package and the propulsive stage to orbit in a single launch. Upon achieving a low Earth orbit (100 to 200 n.mi. circular), the propulsive stage and payload are checked out and launched by the special two-man launch team carried on the orbiter.

On-orbit staytimes of up to seven days are required to allow for on-orbit checkout and launch window phasing.

The ΔV requirements for the highest payload capability orbit (100 n.mi. at 28.5 degrees orbital inclination) and the lowest payload capability orbit (200 n.mi. at 55 degrees) are shown in Table 8-6 as a function of mission profile. No parking orbit is assumed for phasing.

8.4 PLACEMENT, RETRIEVAL, REPAIR, AND MAINTENANCE OF SATELLITES

The space shuttle can place unmanned satellites into various earth orbits. It can also revisit certain high-priority or high-cost satellites and return them to earth if necessary. For such missions, the shuttle will be required to operate at altitudes up to 800 n.mi. and orbit inclinations from 28.5 degrees to polar. With this versatile operational capability, a wide variety of unmanned satellites will be prime candidates for space shuttle support.

These satellites are also logical candidates to be serviced and maintained by the space shuttle. The orbiter would then require the capability to revisit modules and satellites and bring them into an onboard facility where a service and maintenance crew could work in a shirtsleeve environment. The orbiter service and maintenance facility would contain equipment, instruments, and supplies that would allow trained personnel to conduct:

Table 8-5. ACS ΔV Requirements

Task	Function	ΔV (fps)	Task	Function	ΔV (fps)
Limit cycle during transfer and orbit coast for 20 hours	Attitude Control to ±45°	9.7	Rendezvous and Dock	Translate	7.2
	Drag Makeup	5.2		Attitude Control to ±5° for 15 minutes	1.1
Orbit Maneuvers	Orientation to ±5° for 20 minutes prior to maneuver, 4 maneuvers	5.9		Attitude Control to ±0.5° for 20 seconds	1.7
	Control roll during maneuvers. ±5°	1.0	Dedock	Translate	8.3
Transfer from 260 to 270 n.mi.	Transfer ΔV	16.9		Attitude Control to ±0.5° for 20 seconds	1.7
	Attitude Control to ±0.5° for 22 seconds before and during burn	3.1	Limit cycle for 21 hours	Attitude Control to ±45°	11.6
	Attitude Control to ±5° during 0.75 hour transfer	3.3	Entry	Control to ±2° with 2.5-3°/sec^2	23.1
Circularize at 270 n.mi.	Transfer ΔV	16.9		Control to ±2° with 1.9-2.25°/sec^2	26.0
	Attitude Control to ±0.5° for 22 seconds before and during burn	3.1		Control to ±2° with 0.75-1.25°/sec.^{2} for 750 sec.	11.5
			Total ACPS ΔV Requirements		157.9

a. Routine Servicing and Maintenance. These periodic functions would include such items as film changing and replenishment of attitude control propellants.

b. Repair. Although highly automated satellites are designed for long-term operations, a capability to visit such satellites in case of malfunctions is highly desirable. The orbiter could provide on-orbit replacement of instruments and components.

The payload capability range for the satellite placement mission is based on the ΔV requirements in Table 8-7. The two orbits defining the range are 100 n.mi. at 28.5 degrees inclination and 800 n.mi. at 90 degrees inclination. The satellite repair or retrieval missions will require more ΔV due to rendezvous requirements (within 24 hours of launch), as indicated in Table 8-8 for the same two trajectories.

Table 8-6. Delivery of Propulsive Stages and Payload ΔV Requirements

Orbital Altitude (n.mi.)	100		200	
Orbital Inclination (deg)	28.5		55	
	Main Propulsion ΔV (fps)	ACS ΔV (fps)	Main Propulsion ΔV (fps)	ACS ΔV (fps)
Circularize at 100 n.mi.	110	—	—	—
Transfer to 200 n.mi.	—	—	160	—
Circularize at 200 n.mi.	—	—	180	—
Drag Makeup	—	45	—	—
Undock	—	10	—	10
Entry	300	50	380	50
Dispersion	200	20	200	20
FPR and Contingency	400	—	400	—
Total ΔV	1010	125	1300	80
ΔV Difference from Baseline	-790	-75	-500	-120
ΔV Difference due to Launch Azimuth change from Baseline	-460	—	0	—

8.5 DELIVERY OF PROPELLANT

The space shuttle would operate as a propellant-delivery tanker in conjunction with a long-duration orbital propellant storage (OPS) facility. The OPS facility would act as a filling station to supply liquid hydrogen and oxygen propellants for high-energy,

Table 8-7. Satellite Placement ΔV Requirements

Orbit Altitude (n.mi.)	100		800	
Orbit Inclination (deg)	28.5		90	
	Main Propulsion ΔV (fps)	ACS ΔV (fps)	Main Propulsion ΔV (fps)	ACS ΔV (fps)
Circularize at 100 n.mi.	110	—	—	—
Transfer to 800 n.mi.	—	—	1105	—
Circularize at 800 n.mi.	—	—	1080	—
Drag makeup	—	45	—	—
Undock	—	10	—	—
Entry	300	50	1225	50
FPR and Contingency	400	—	400	—
Dispersions	200	20	200	20
Total ΔV	1010	125	4010	80
ΔV Difference from Baseline	-790	-75	+2210	-120
ΔV Difference from Baseline due to Launch Azimuth	-460	—	+ 880	—

large payload propulsive stages for interplanetary missions which could not be launched from Earth fully loaded with the space shuttle for space-based vehicles operating between Earth orbit and the Moon, and within Earth orbit. Propellants will also be delivered to the space base/station.

When operating as a propellant tanker, the orbiter payload bay would be configured differently depending on whether it is delivering all liquid hydrogen, liquid oxygen and liquid hydrogen, or a mix of dry cargo and propellants. The largest payload volume requirement will be for the liquid hydrogen deliveries. Including the tankage, insulation, and propellant transfer mechanisms, 45,000 pounds of liquid hydrogen would require about a 50,000-pound capability for the space shuttle. The volume corresponding to 45,000 pounds of liquid hydrogen is about 11,000 ft^3. With the volume required for the liquid hydrogen, sufficient capability will exist for combined liquid hydrogen and oxygen loads.

The space shuttle will rendezvous with the OPS facility and transfer propellant without crew EVA. Two men in addition to the crew will monitor the operation and provide manual override to the transfer systems.

Table 8-8. Satellite Service or Retrieve Mission
ΔV Requirements

Orbit Altitude (n.mi.)	100		800	
Orbit Inclination (deg)	28.5		90	
	Main Propulsion ΔV (fps)	ACS ΔV (fps)	Main Propulsion ΔV (fps)	ACS ΔV (fps)
Transfer to 250 n.mi.	260	—	—	—
Circularize at 250 n.mi.	360	—	—	—
Transfer to 100 n.mi.	260	—	—	—
Circularize at 100 n.mi.	260	—	—	—
Circularize at 100 n.mi.	—	—	110	—
Transfer to 800 n.mi.	—	—	1130	—
Circularize at 800 n.mi.	—	—	1080	—
Terminal Phase	—	20	—	20
Braking/Stationkeeping	—	90	—	90
Docking	—	10	—	10
Drag Makeup	—	90	—	—
Undocking	—	10	—	10
Entry	300	50	1225	50
FPR and Contingency	400	—	400	—
Dispersions	200	20	200	20
Total ΔV	2040	200	4145	200
ΔV Difference from Baseline	+240	+90	+2345	0
ΔV Difference due to Launch Azimuth	-460	—	+ 880	—

The lunar mission supply is composed of six men and 20,000 pounds of payload to be delivered to the OPS where the lunar tugs would be located, serviced, and fueled. Other payloads for interplanetary missions would also be delivered to the OPS for integration with the space tug or nuclear shuttle.

The mission duration is seven days. Table 8-9 presents the ΔV requirements for the highest payload mission (200 n.mi. at 28.5 degrees inclination) and the lowest payload mission (300 n.mi. at 55 degrees). All rendezvous phases are within 24 hours.

Table 8-9. Delivery of Propellant Mission ΔV Requirements

Orbit Altitude (n.mi.)	200		300	
Orbit Inclination (deg)	28.5		55	
	Main Propulsion ΔV (fps)	ACS ΔV (fps)	Main Propulsion ΔV (fps)	ACS ΔV (fps)
Circularize at 100 n.mi.	110	—	110	—
Transfer to 200 n.mi.	180	—	—	—
Circularize at 200 n.mi.	180	—	—	—
Transfer to 300 n.mi.	—	—	350	—
Circularize at 300 n.mi.	—	—	350	—
Transfer Phase	—	20	—	20
Braking/Station Keeping	—	90	—	90
Dock	—	10	—	10
Undock	—	10	—	10
Entry	380	50	500	50
FPR and Contingency	400	—	400	—
Dispersions	200	20	200	20
Total ΔV	1450	200	1910	200
ΔV Difference from Baseline	-350	0	+110	0
ΔV Difference from Baseline Launch Azimuth Change	-460	—	—	—

8.6 SHORT-DURATION ORBIT

The space shuttle will be capable of operation as a Short-Duration Orbital Station for up to 30 days to exploit man's capabilities as a selective sensor and decision maker.

The higher resolution obtained from a 100-n.mi. orbit as opposed to a 270-n.mi. orbit indicates a unique capability of the space shuttle short-duration orbital mission even in a 55-degree (or lower) orbital inclination. The large payload volume capability of the orbiter provides an ideal platform for the development of advanced equipment and instrumentation.

A module or modules is required containing appropriate instrumentation and provisions for a 10-man crew in a shirt-sleeve environment. This module can also be used as a flying test bed for sensor research, development, test, and calibration to support both manned and unmanned satellite missions, to develop and test complete experiment systems to verify their operational capabilities before being integrated into the Space Station, and to develop and flight-test systems components in support of a manned planetary program.

The ΔV requirements associated with these missions are presented in Table 8-10.

Table 8-10. Short-Duration Orbit ΔV Requirements

Orbit Altitude (n.mi.)	100		300	
Orbit Inclination (deg)	28.5		90	
	Main Propulsion ΔV (fps)	ACS ΔV (fps)	Main Propulsion ΔV (fps)	ACS ΔV (fps)
Circularize at 100 n.mi.	110	—	—	—
Transfer to 300 n.mi.	—	—	370	—
Circularize at 300 n.mi.	—	—	350	
Drag Makeup	—	190	—	—
Station Keeping	—	360	—	360
Entry	300	50	500	50
FPR and Contingencies	400	—	400	—
Dispersion	200	20	200	20
Total ΔV	1010	620	1820	430
ΔV Difference from Baseline	-790	+420	+ 20	+230
ΔV Difference due to Launch Azimuth Change	-460	—	+880	—

8.7 RESCUE

The space shuttle capability for space base/station rescue requires rendezvous within 24 hours of rescue request. The ΔV requirements in Table 8-11 reflect worst-case phasing requiring a 16-hour wait for the launch window, with 8 hours remaining for the flight operation to get to the base. These ΔV requirements result in no payload capability.

An increase in the allowable time to rendezvous from launch, or a better space base location at the time of rescue request will result in improved payload capability. The use of a main engine propellant tank in a portion of the payload bay will also increase payload capability as discussed in Section 2.3 of Volume VIII.

Table 8-11. Rescue Mission ΔV Requirements

Orbit Altitude (n.mi.)	270	
Orbit Inclination (deg)	55	
	Main Propulsion ΔV (fps)	ACS ΔV (fps)
Transfer to 550 n.mi.	750	—
Circularize 550 n.mi.	720	—
Transfer to 270 n.mi.	720	—
Circularize 270 n.mi.	750	—
Transfer Phase	—	20
Braking/Station Keeping	—	90
Dock	—	10
Undock	—	10
Entry	450	50
FPR and Contingency	400	—
Dispersion	200	20
Total ΔV	3990	200
ΔV Difference from Baseline	+2190	—

8.8 ALTERNATE MISSION CAPABILITY

Alternate mission capability was determined by holding the vehicle liftoff weights fixed and determining the payload to selected orbits, using the basic rocket performance equation. The linear partials are 30.1 lb payload fps for FR-3, and 29.6 lb payload/fps for FR-4. The results are shown in Tables 8-12 and 8-13 for the FR-3 and FR-4 vehicle systems respectively. The same sort of information was also generated generally on an orbit altitude and inclination basis for the baseline systems, as part of the overall sensitivity analysis which is covered in Section 10 of this report.

Note that any ascent payload in excess of 50,000 pounds cannot be returned unless the orbiter vehicle is specifically designed for such an overload, in which case, of course, the system gross liftoff weight must increase.

Table 8-12. FR-3 Mission Performance Summary

Mission	Orbit Altitude (n.mi.)		Orbit Inclination (deg)		On-Orbit ΔV (fps)		Payload Capability (lb up)	
Space Base/Station Logistics	270		55		1800		50,000	
Delivery of Propulsive Stage and Payload	100	200	28.5	55	1010	1300	87,600	65,000
Placement, Retrieval, Service, and Maintenance of Satellites	100	800	28.5	90	1010	2210	87,600	0
					2040*	2345*	56,520*	0*
Delivery of Propellants	200	300	28.5	55	1450	1810	74,400	46,700
Short Duration Orbit	100	300	28.5	90	1010	1820	87,600	22,900
Space Rescue	270		55		3990		0	

*Service, Maintenance, or Retrieve

Table 8-13. FR-4 Mission Performance Summary

Mission	Orbit Altitude (n.mi.)		Orbit Inclination (deg)		On-Orbit ΔV (fps)		Payload Capability (lb up)	
Space Base/Station Logistics	270		55		1800		50,000	
Delivery of Propulsive Stage and Payload	100	200	28.5	55	1010	1300	87,000	64,800
Placement, Retrieval, Service, and Maintenance of Satellites	100	800	28.5	90	1010	2210	87,000	0
					2040*	2345*	56,500*	0*
Delivery of Propellants	200	300	28.5	55	1450	1810	74,900	46,740
Short Duration Orbit	100	300	28.5	90	1010	1820	87,000	23,400
Space Rescue	270		55		3990		0	

*Service, Maintenance, or Retrieve

SECTION 9

TURNAROUND ANALYSIS

A ground turnaround analysis was made on a configuration of the reusable space vehicle. The FR-1 three-element vehicle with crossfeed was used as the baseline configuration. A detailed analysis of that vehicle is contained in Volume IX. Using the baseline vehicle analysis data for comparative purposes, a preliminary analysis has been made of the turnaround operations requirement for the FR-4 vehicle (three-element without crossfeed) and the FR-3 vehicle (two-element of different geometry). The ground turnaround cycle phases, which provided a functional flow of tasks for the baseline vehicle analysis, are equally applicable to the other two configurations being discussed. Figure 9-1 depicts the turnaround phases and their location as applicable to Complex 39, ETR. Additionally, this figure will provide a guide for facility location as discussed in Section 9.2. It was determined that configuration changes do not appreciably affect the basic turnaround cycle phases and only modify the elapsed time and manpower requirements for certain tasks within the phases. Size, shape, and number of elements do not have an effect upon facility requirements necessary to support the ground turnaround operations. Section 9.1 discusses the effects of configuration differences on maintenance and the ground turnaround tasks. Section 9.2 will present the facility requirements as they differ for the three-element or two-element reusable space vehicle configurations.

9.1 MAINTENANCE/REFURBISHMENT

The number of main propulsion subsystem engines, number of flyback engines, complexity of propellant subsystems and the wetted area involving thermal protective materials are the main sources of maintenance task differences when comparing vehicle configurations. Table 9-1 shows the vehicle design data which contribute to differences in elapsed phase times and required manhours. Table 9-2 presents a comparative summary, by phases, of the elapsed time to process the FR-4 and FR-3 reusable space vehicle through to the required ground turnaround cycle. It is well to mention here that each element is processed as an entity until mated to its other element or elements in preparation for launch.

Table 9-2 shows that the FR-3 booster element requires 64.0 hours to process in the maintenance phase, while each booster element of the FR-4 requires 60.4 hours. The 3.6 additional hours required for FR-3 booster maintenance is due to design configuration differences shown on Table 9-1. The fact that the FR-3 vehicle is composed of two-element versus three-elements for the FR-4 provides for a 5.0-

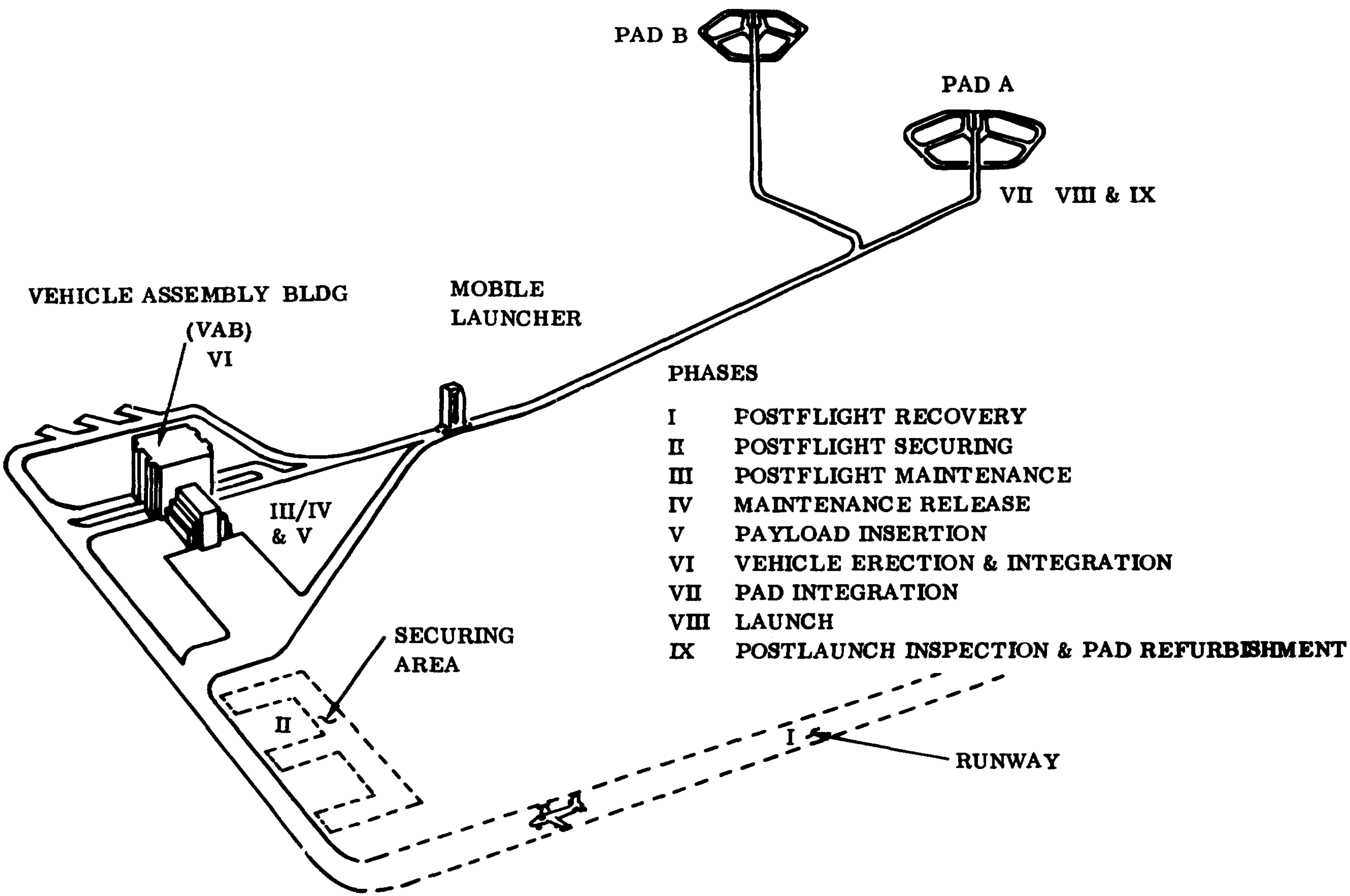


Figure 9-1. Turnaround Phases as Applicable to Complex 39, KSC

Table 9-1. Comparison of Vehicle Design Data Affecting Maintenance Tasks

Design Data	FR-4 (Three Element)	FR-3 (Two Element)
Propulsion Subsystem Main Boost Engines	9 rocket engines each booster. 3 rocket engines on orbiter.	15 rocket engines on booster. 3 rocket engines on orbiter.
Flyback Engines	3 jet engines each booster. 2 jet engines orbiter. (Both booster and orbiter engines equal in thrust.)	4 jet engines of 50,000-lb-thrust each on booster. 3 jet engines of 21,000-lb-thrust on orbiter.
Propellant Subsystem	Fuel lines and valves to feed 21 rocket engines. 3.84 million lbs propellant required to fuel vehicle.	Fuel lines and valves to feed 18 rocket engines. 3.44 million lbs propellant required to fuel vehicle.
Body Wetted Area	16,890 sq ft of TPS area to be inspected on orbiter 18,000 sq ft of area to be inspected on each of two boosters.	14,900 sq ft of TPS area to be inspected on orbiter. 26,610 sq ft of area to be inspected on one booster.

Table 9-2. Configuration Turnaround Cycles (Hr)

	Phases	FR-4 Booster	FR-4 Orbiter	FR-3 Booster	FR-3 Orbiter
I	Postflight Recovery	1.0	1.0	1.0	1.0
II	Postflight Securing	7.0	10.0	7.0	10.0
III & IV	Maintenance	60.4	67.9	64.0	68.0
V	Payload Insertion	—	6.5	—	6.5
VI	Erection & Integration	33.5	33.5	28.5	28.5
VII	Pad Integration	20.0	20.0	18.0	18.0
VIII	Launch	6.0	6.0	5.8	5.8
	Element Totals	127.9	144.9	124.3	137.8
	Vehicle Totals (Orbiter as Driving Factor)	144.9		137.8	

hour time decrease when processing the FR-3 as an integrated vehicle. The FR-3 booster element requires 124.3 hours total turnaround time. That time would only be valid if an orbiter element were standing by and ready to be mated with its booster element; otherwise, if the two elements were to commence ground turnaround processing at the same time, the orbiter element turnaround time would be the driving factor for vehicle turnaround since the booster would finish first but must wait for the orbiter before mating could occur.

Table 9-3 summarizes the manhour requirements for the two and three-element configurations. The fact that the FR-3 vehicle (two-element) has one less element than the FR-4 or FR-1 does not mean that one-third less time and one-third less manhours are required to process the vehicle through ground turnaround. However, with the increased booster element size and the increased number of main propulsion engines to be inspected (Table 9-1) the total ground turnaround time is still shorter by 7.1 hours and the manhours required are 2901 less (6203.7) than for the three-element vehicles (9104.7). It is concluded that although the FR-3 booster element is larger and more complex than the FR-4 booster elements, the fact that one less element is involved in the total vehicle turnaround cycle allows for a shorter vehicle turnaround time and expenditure of less direct manhours for ground operations.

9.2 FACILITIES

The impact of the different vehicle configurations (FR-1, FR-3, FR-4) on ground turnaround support facilities required is of some consequence. From a facility standpoint, the FR-1 and FR-4 vehicles are practically identical, the slight size variation has little effect on the size and type of facilities required. The facility requirement matrix shown in Table 9-4 shows major facility requirements presented on a configuration comparative basis. The matrix treats both FR-1 and FR-4 as similar vehicles for comparison with the FR-3.

It may be noted from Table 9-4 that the FR-3 vehicle has a somewhat more marked effect on facilities when compared to the FR-1 and FR-4 vehicles. The larger booster of the FR-3 requires a bigger postflight securing area. On the other hand, due to a lesser number of elements in the turnaround cycle, the area required for the main service building can be reduced by one-third. Similarly, propellant quantities can be reduced by approximately 25% and reductions made in propellant service line sizes and number of connections.

Should Complex 39 be considered for use as a launch facility, the existing Saturn launcher umbilical tower could be modified and used for the FR-3. It is also

Table 9-3. Manhour Requirements

	FR-4 (Three Element) Orbiter	FR-4 (Three Element) Booster	FR-3 (Two Element) Orbiter	FR-3 (Two Element) Booster
Vehicle Ground Operations	379.6	379.6	379.6	379.6
Vehicle Operations Support and Maintenance and Operations of Ground Support Facilities and Equipment	1955.2	1955.2	1955.2	1955.2
Vehicle Maintenance	789.5	655.4	784.4	749.7
Element Totals	3124.3	2990.2	3119.2	3084.5
Vehicle Totals	9104.7		6203.7	

Table 9-4. Facility Requirement Matrix, FR-1, FR-4, and FR-3 Vehicle

Facility	FR-1, FR-4 Vehicles	FR-3 Vehicle	Facilities as Applicable to Complex 39, KSC
Aircraft Runway	10,000 ft all weather.	10,000 ft all weather.	New construction required.
Revetted Securing Area	(2) 250 ft x 200 ft areas + magazine.	(1) 300 ft x 250 ft area (1) 250 ft x 200 ft area + magazine	New construction required.
Maintenance & Service Building	300,000 sq ft	200,000 sq ft	Addition to west side of existing VAB.
Logistics Building	250 ft x 180 ft	230 ft x 180 ft	Minor modification to existing low bay area of VAB.
Erection Facility	150-ton crane or lift platform.	250-ton crane or lift platform.	Existing in VAB for off pad erection.
Launch Pads (2)	3600 ft pentagon shape	3600 ft pentagon shape	Modify existing launch pads.
Fuel Storage Tank Farms (2)	1.3 million gal LH_2 800,000 gal LO_2	900,000 gal LH_2 550,000 gal LO_2	Existing facility at Complex 39 adequate for FR-3. Additional capacity required for FR-4.
High Pressure Gas Facility	700,000 gal LN_2 storage.	500,000 gal LN_2 storage.	Existing for FR-3. Additional capacity required for FR-4.
Launcher Umbilical Tower	New design. To accommodate 3-element vehicle. With 45 ft x 100 ft exhaust chamber	Existing Saturn LUT. Modified with new launcher mechanisms and umbilical connections.	Modify existing LUT for FR-3. New design required for FR-4
Flame Deflector	New design for 3-element vehicle.	Existing Saturn (modified) or new configuration to suit flame pattern.	Modify to accommodate flame pattern for both FR-3 and FR-4.

possible that the existing Saturn flame deflectors could be modified to suit the flame pattern of this vehicle. These two equipment modification possibilities for the FR-3 become impracticalities for the FR-1 and FR-4 vehicles due to overall vehicle size.

Other major facility changes are not substantial; for example, runway requirements are common to all vehicles, the size of the logistics building would vary by only 20 ft in length, and varying crane sizes for erection purposes can be readily met for all three configurations. If the Complex 39 VAB cranes are used, ample capacity currently exists for lifting each element regardless of vehicle configuration.

Some minor change is evident in aerospace ground equipment, principally in sizes of workstands, slings, and other handling equipment. The differences in AGE are due mainly to the disparity in size between the FR-3 booster (41-ft wide x 37-ft high) and the FR-4 booster (33-ft wide x 29-ft high).

SECTION 10

VEHICLE SENSITIVITIES

10.1 SENSITIVITY OF GROSS LIFTOFF WEIGHT AND TOTAL SYSTEM DRY WEIGHT TO SELECTED PARAMETER CHANGES

These sensitivities were generated using the synthesis program with variations from the design points for the FR-3 and FR-4 15-ft-diameter by 60-ft-long payload bay baseline vehicle systems as the reference point. The results are plotted in Figures 10-1 through 10-16 for the FR-3 system and in 10-17 through 10-32 for the FR-4 system. The figures are generally self-explanatory. Some explanation on volume sensitivity may be required as follows.

Variations in volume were made to reflect the system sensitivity to packaging efficiency. For instance, what happens if we need more or less volume in the thrust compartment for propellant ducting reasons, or more or less volume in the forebody. To some extent this parameter indicates the penalties incurred by payload bay volume requirements. Volume increases are costly in the orbiters.

Another volume variation in the FR-4 system consisted of making the booster elements differ in volume from the orbiter. This method is a holdover from the FR-1 system where the synthesis program logic was set to equate volumes. A discrete staging velocity resulted. To increase staging velocity in the FR-4 system the booster volumes are increased in the program input. This has been shown previously in Section 4. In the FR-3 synthesis the staging velocity is controlled directly by variation of the booster performance mass ratio.

10.2 SENSITIVITY OF THE PAYLOAD OF THE FIXED POINT DESIGN VEHICLES TO SELECTED PARAMETER CHANGES

These were made using the Convair general trajectory simulation module (GTSM) program with the FR-3 and FR-4 15-ft-diameter payload by 60-ft-long payload bay baseline vehicle systems point designs as fixed gross liftoff reference points. The ground rules for the analysis were:

a. ETR launch.

b. Launch azimuth = 37.65 degrees (inclination = 55 deg).

c. Perigee injection altitude = 260,000 ft and injection inertial velocity [illegible]5,897 ft/sec.

d. Axial load limit = 3g.

e. Staging at q = 50 psf.

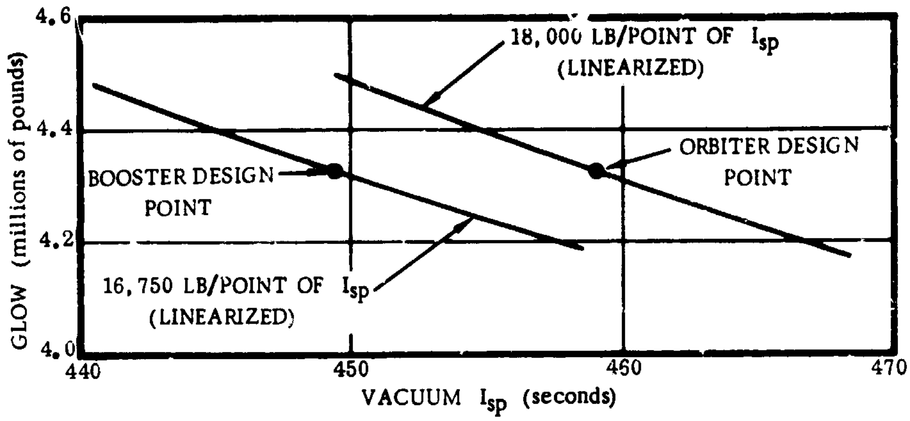


Figure 10-1. FR-3 GLOW Sensitivity to ISP

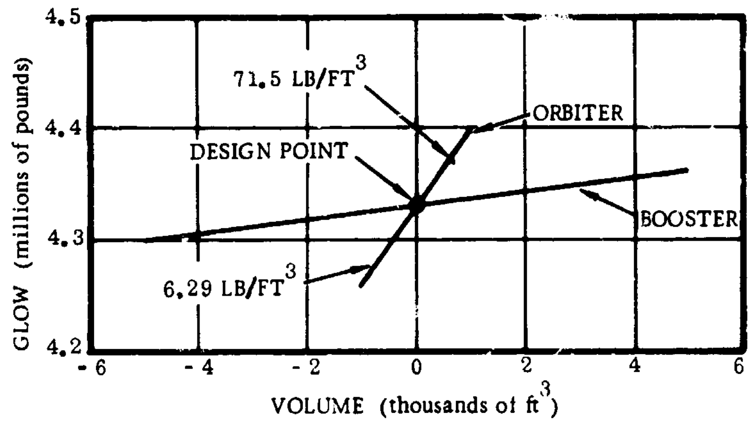


Figure 10-2. FR-3 GLOW Sensitivity to Volume

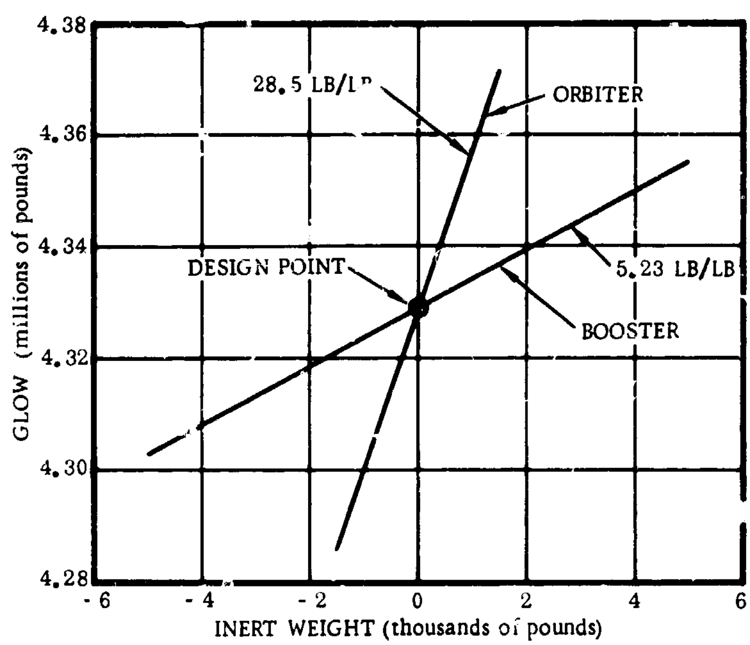


Figure 10-3. FR-3 GLOW Sensitivity to Inert Weight

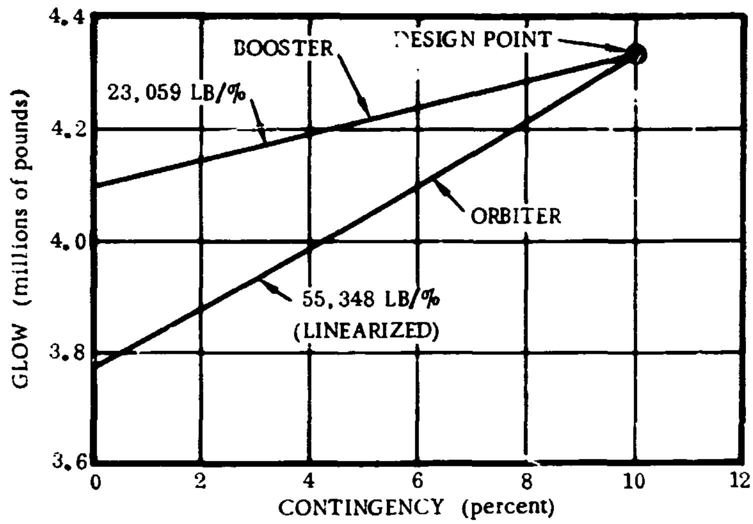


Figure 10-4. FR-3 GLOW Sensitivity to Contingency Percentage

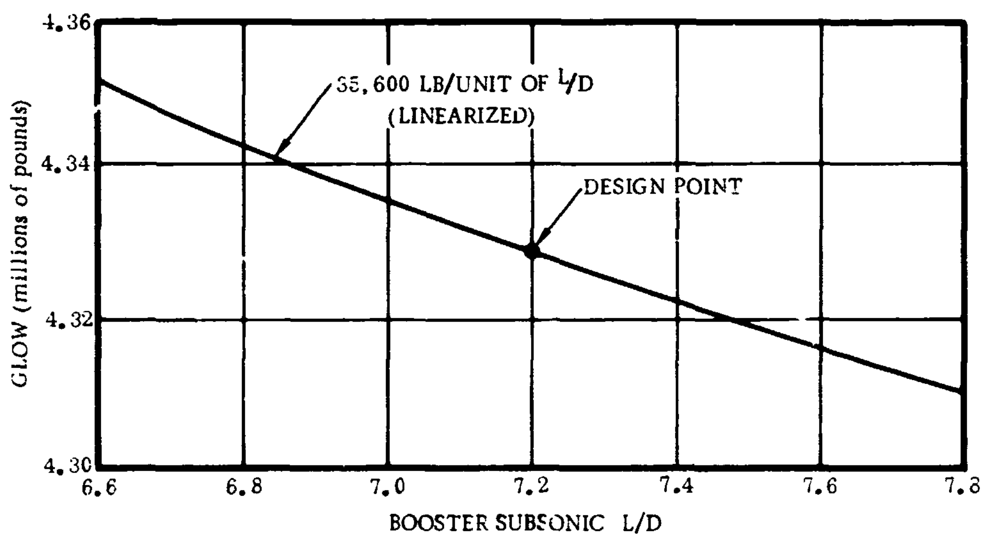


Figure 10-5. FR-3 GLOW Sensitivity to Booster Subsonic L/D

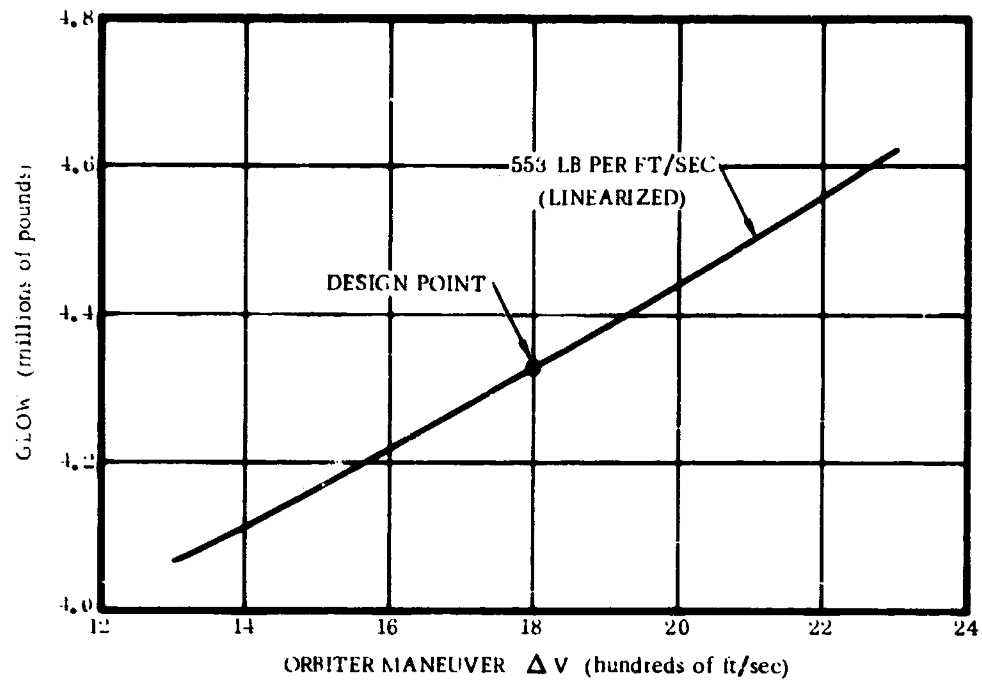


Figure 10-6. FR-3 GLOW Sensitivity to Orbit Maneuver ΔV

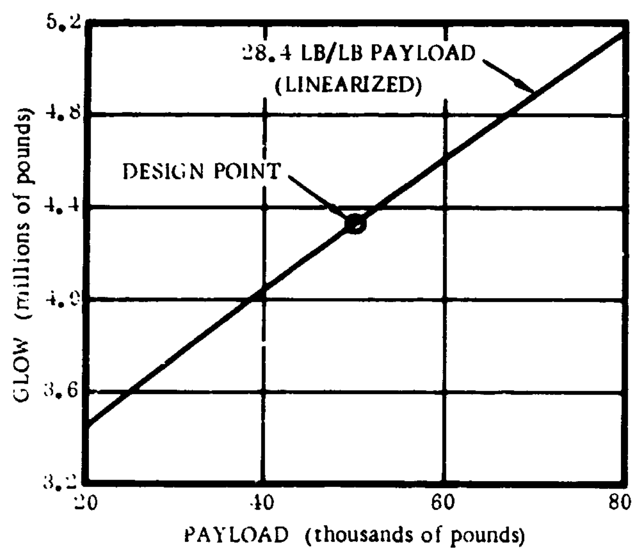


Figure 10-7. FR-3 GLOW Sensitivity to Payload

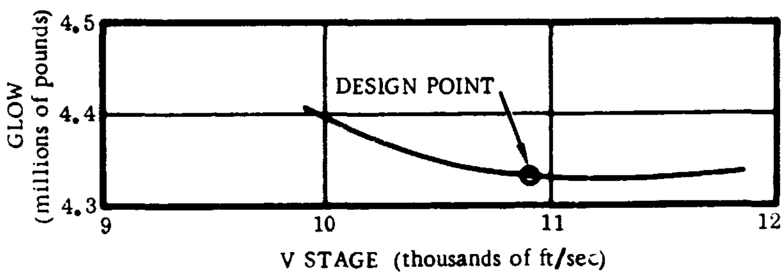


Figure 10-8. FR-3 GLOW Sensitivity to Staging Velocity

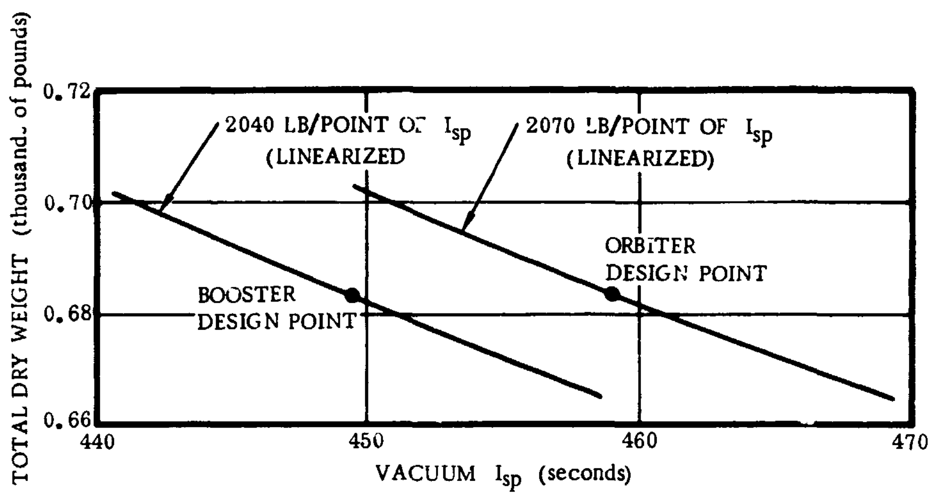


Figure 10-9. FR-3 Dry Weight Sensitivity to ISP

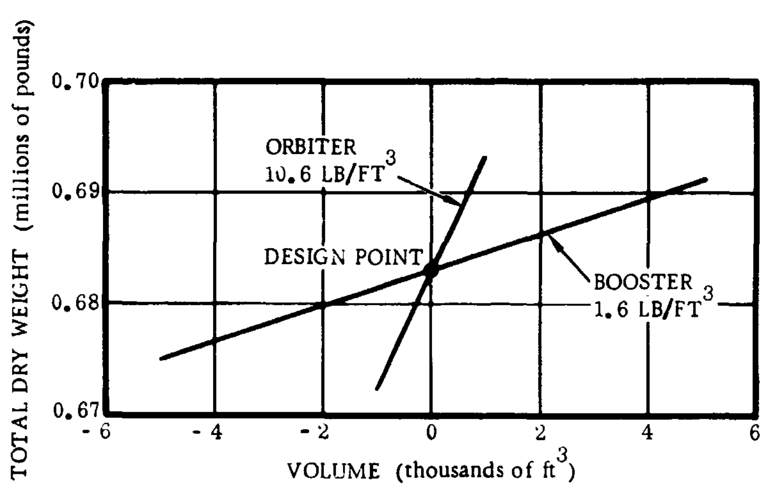


Figure 10-10. FR-3 Dry Weight Sensitivity to Volume

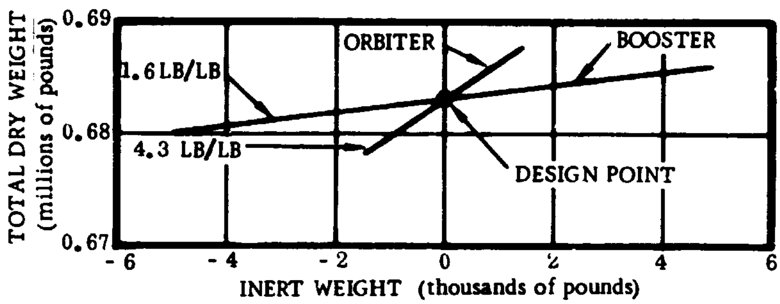


Figure 10-11. FR-3 Dry Weight Sensitivity to Inert Weight

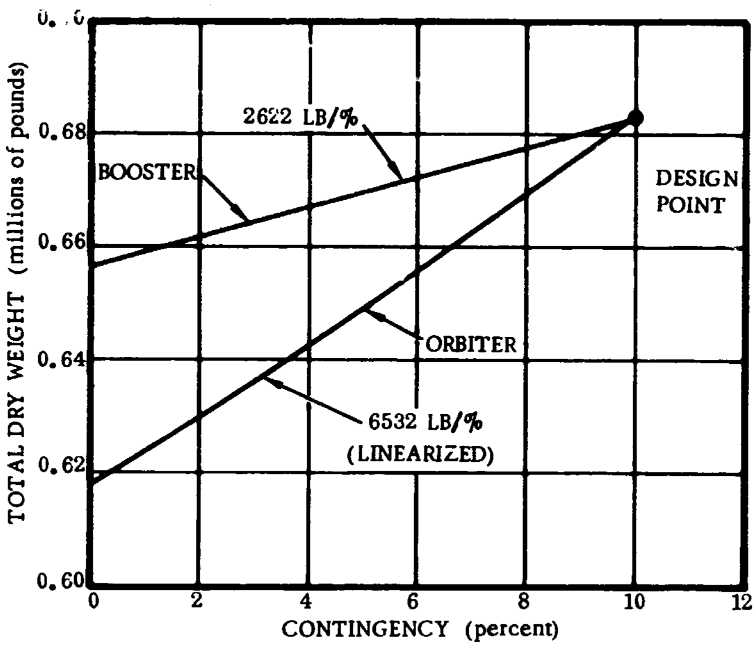


Figure 10-12. FR-3 Dry Weight Sensitivity to Contingency Percentage

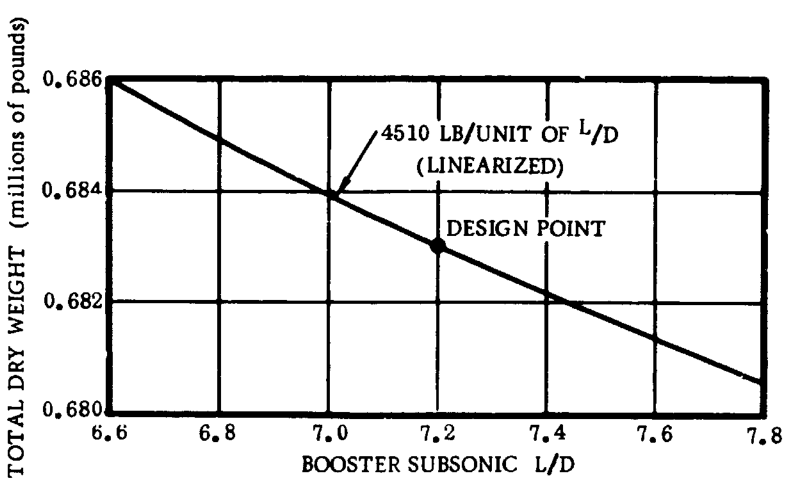


Figure 10-13. FR-3 Dry Weight Sensitivity to Subsonic L/D

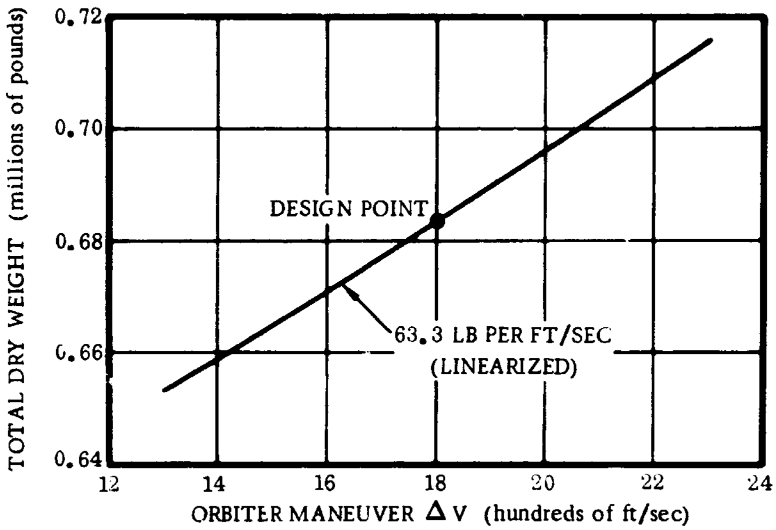


Figure 10-14. FR-3 Dry Weight Sensitivity to Maneuver ΔV

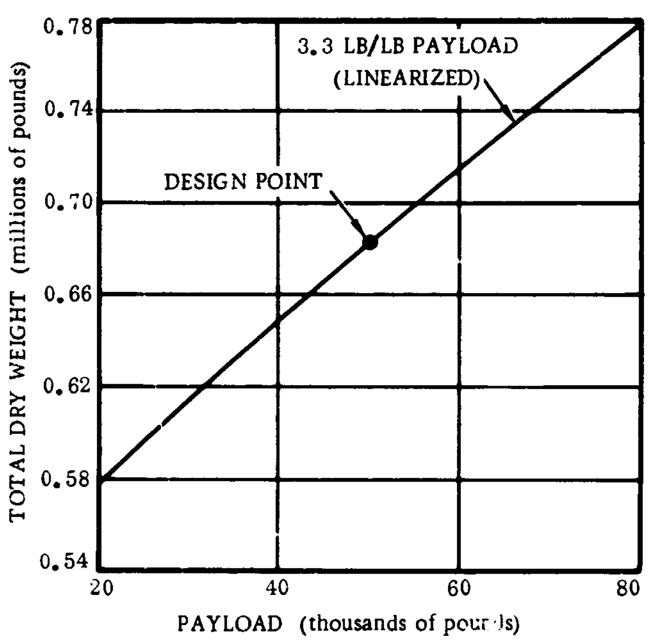


Figure 10-15. FR-3 Dry Weight Sensitivity to Payload

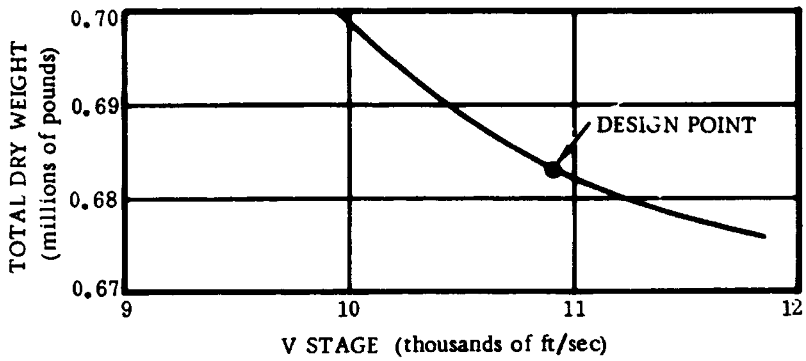


Figure 10-16. FR-3 Dry Weight Sensitivity to Staging Velocity

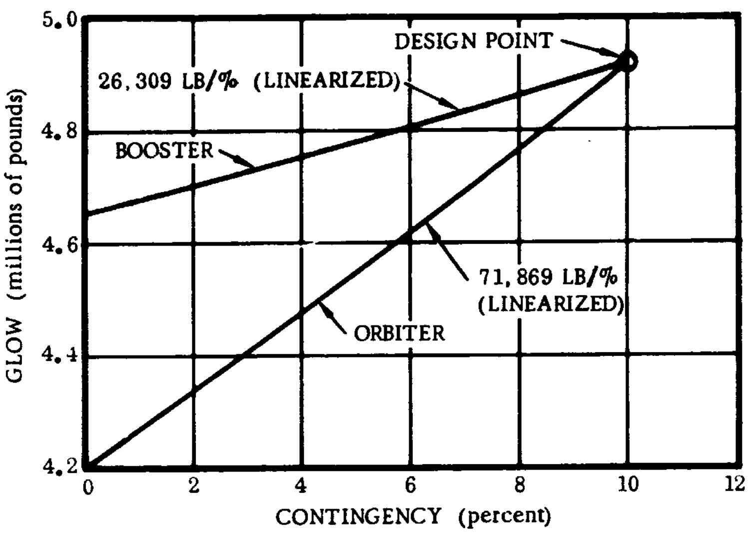


Figure 10-17. FR-4 GLOW Sensitivy to Contingency Percentage

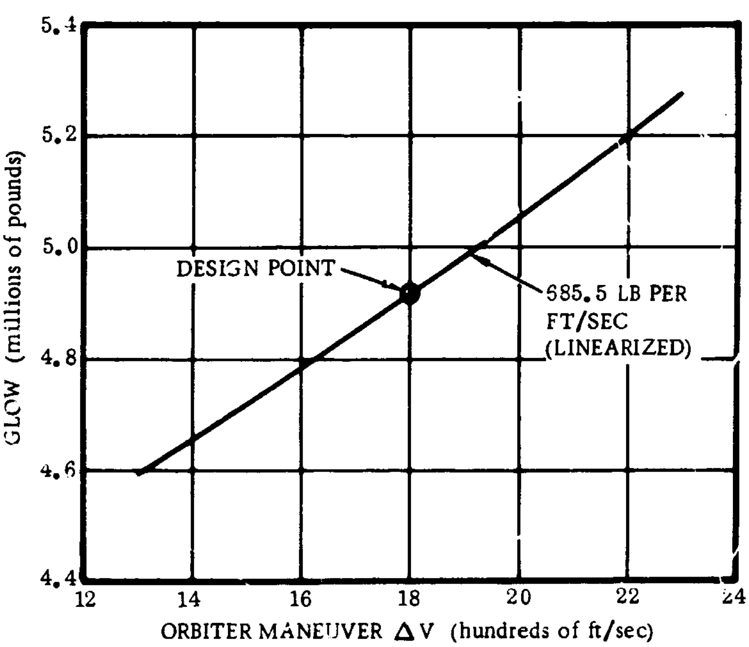


Figure 10-18. FR-4 GLOW Sensitivity to Maneuver ΔV

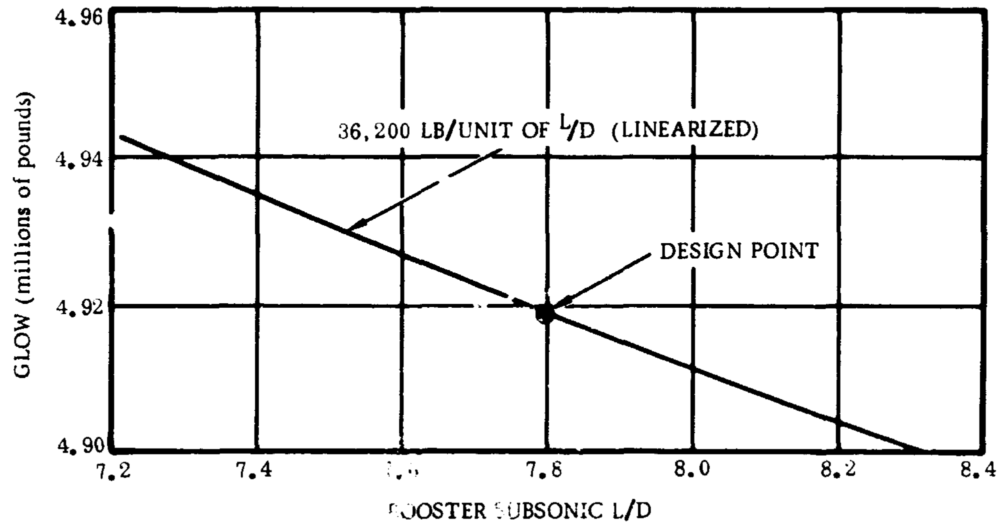


Figure 10-19. FR-4 GLOW Sensitivity to Booster Subsonic L/D

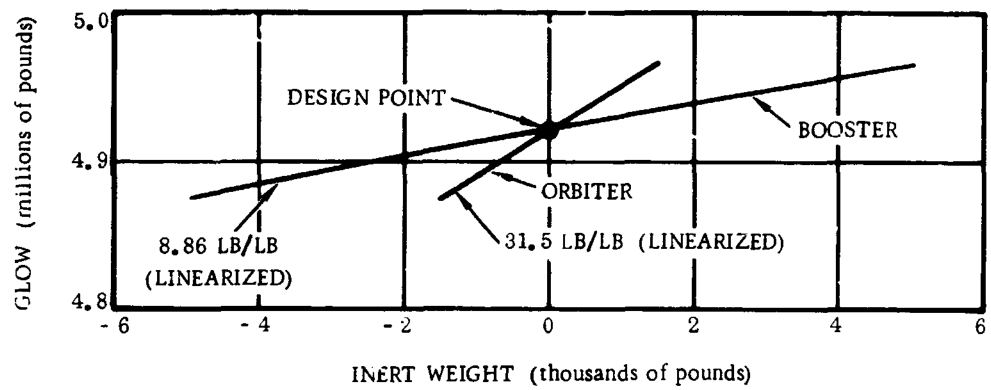


Figure 10-20. FR-4 GLOW Sensitivity to Inert Weight

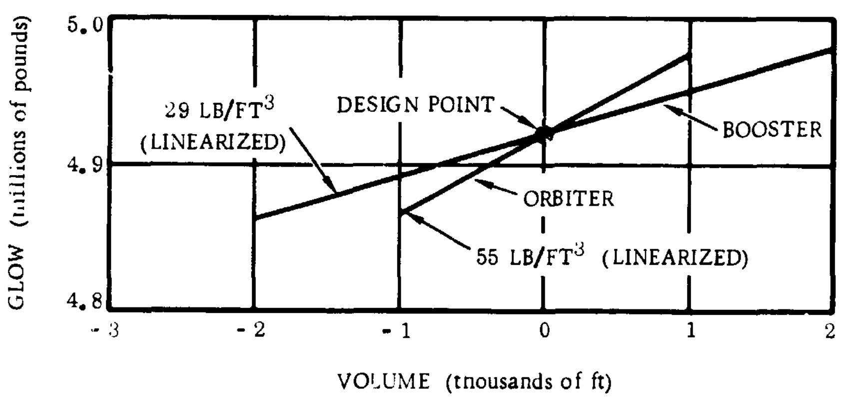


Figure 10-21. FR-4 GLOW Sensitivity to Volume

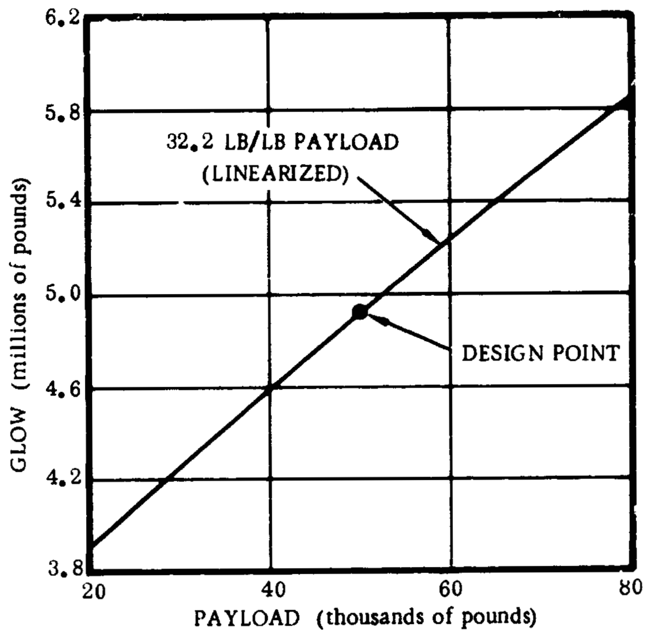


Figure 10-22. FR-4 GLOW Sensitivity to Payload Weight

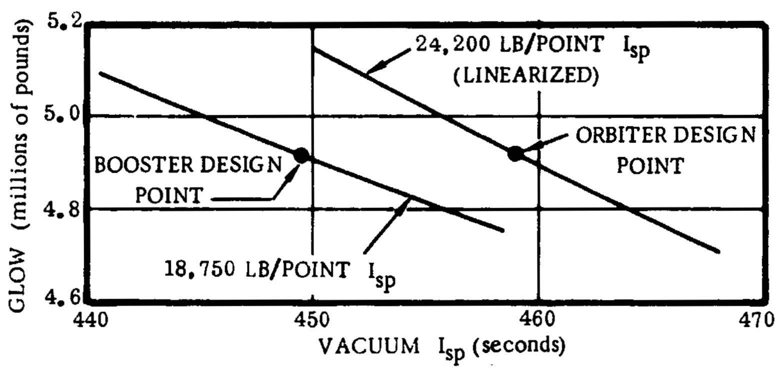


Figure 10-23. FR-4 GLOW Sensitivity to ISP

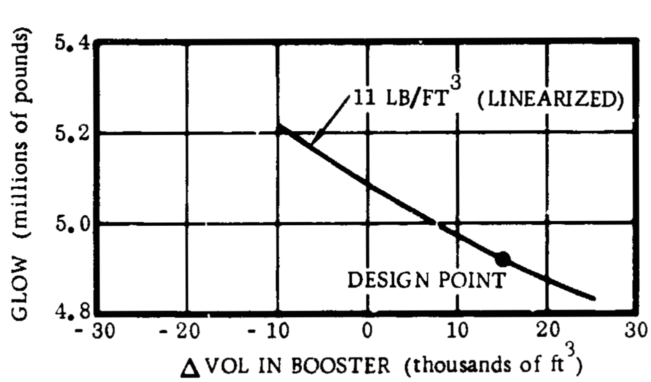


Figure 10-24. FR-4 GLOW Sensitivity to Booster ΔVolume

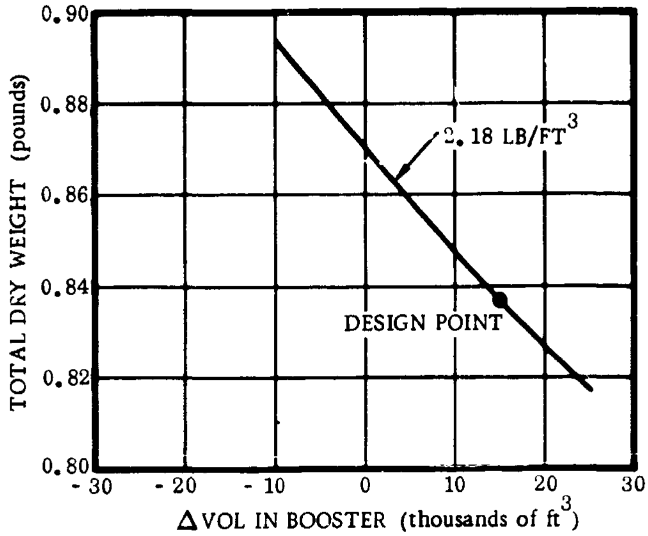


Figure 10-25. FR-4 Dry Weight Sensitivity to Booster ΔVolume

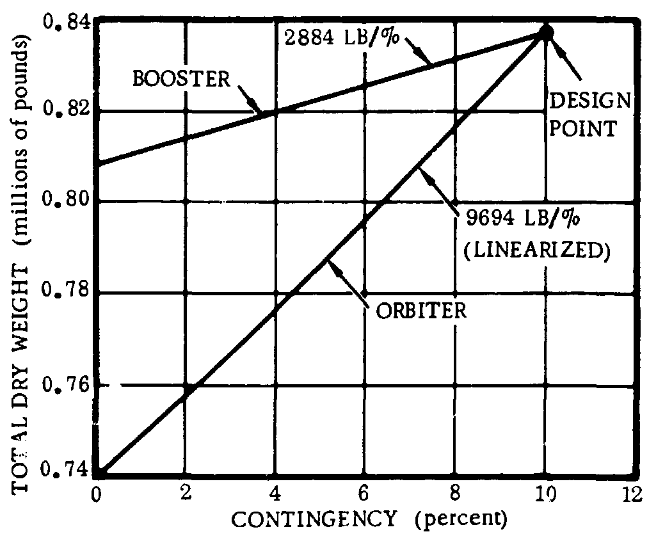


Figure 10-26. FR-4 Dry Weight Sensitivity to Contingency Percentage

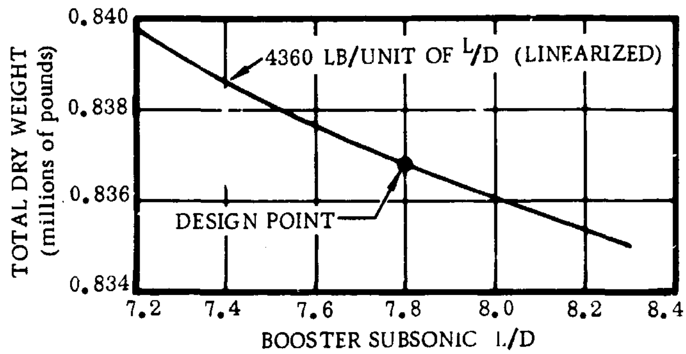


Figure 10-27. FR-4 Dry Weight Sensitivity to Booster Subsonic L/D

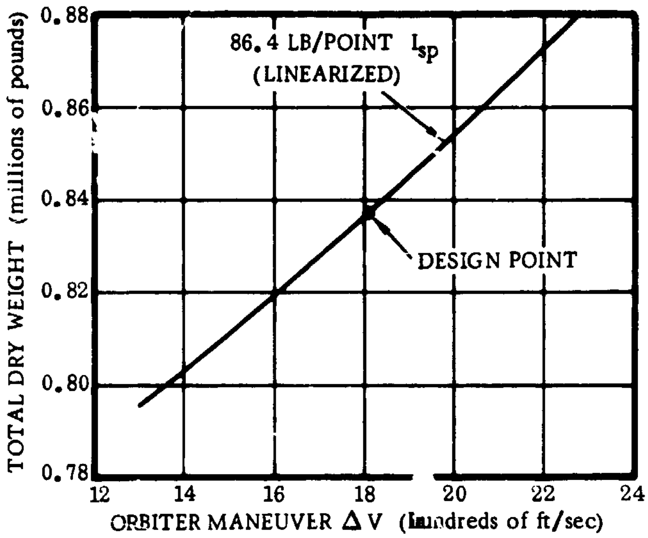


Figure 10-28. FR-4 Dry Weight Sensitivity to Maneuver ΔV

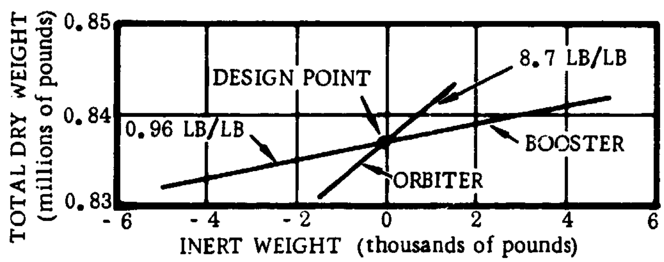


Figure 10-29. FR-4 Dry Weight Sensitivity to Volume

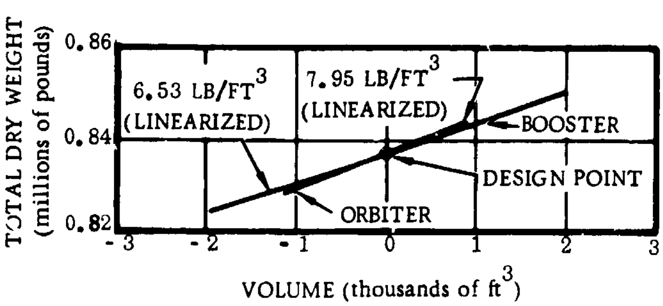


Figure 10-30. FR-4 Dry Weight Sensitivity to Inert Weight

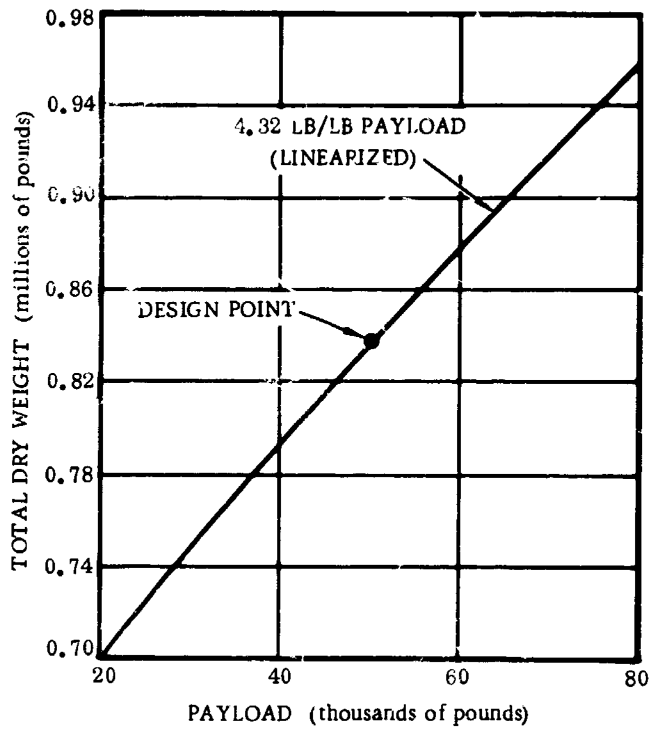


Figure 10-31. FR-4 Dry Weight Sensitivity to Payload

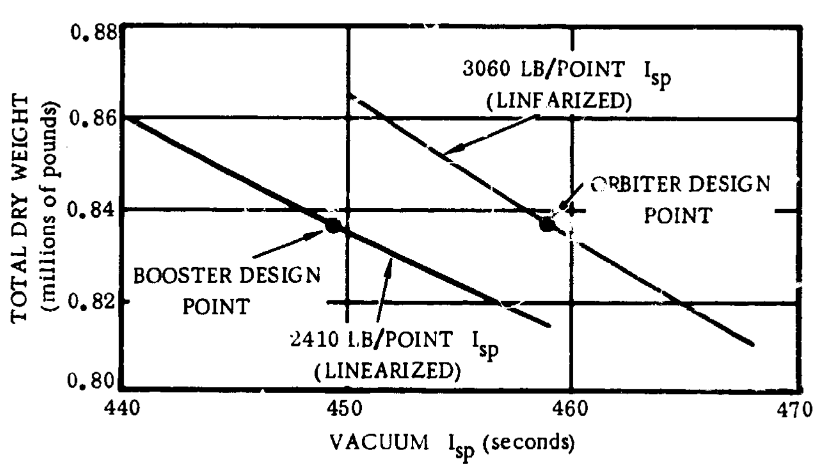


Figure 10-32. FR-4 Dry Weight Sensitivity to ISP

The analysis was made using a series of flight simulations with the independent variable perturbed a fixed percentage. The payload for the resulting weight-in-orbit was determined by comparison to the nominal trajectory simulation.

The payload sensitivities are presented in Table 10-1 for the FR-3 and Table 10-2 for the FR-4 in the form of partial derivatives for each independent parameter considered. The symbols used in the paylod sensitivities are:

W_{jB}	Booster jettison (inert) weight
W_{jo}	Orbiter jettison (inert) weight
ISP_B	Booster specific impulse (vacuum)
ISP_o	Orbiter specific impulse (vacuum)
T_B	Booster thrust (sea level)
T_o	Orbiter thrust (vacuum)
W_{PB}	Booster propellant
W_{Po}	Orbiter propellant
WIO	Weight-in-orbit
T	Thrust
$\dot{W}$	Flow rate
ISP	Specific impulse

10.3 FR-3 AND FR-4 ALTERNATE MISSION CAPABILITY

Three alternate missions were considered for the FR-3 and FR-4 point design vehicles. These are:

a. 90-deg inclination; 100-n.mi. circular orbit; on-orbit ΔV = 600 fps.

b. 30-deg inclination; 100-n.mi. circular orbit; on-orbit ΔV = 600 fps.

c. 30-deg inclination; 270-n.mi. circular orbit; on-orbit ΔV = 1800 fps.

The payload capability of each vehicle for the alternate missions was determined by computing the incremental velocity available above the baseline mission and converting this velocity to payload for a range of inclinations including the above.

Table 10-1. FR-3 Payload Senstivity Partial Derivatives

∂ PL/∂ W_{jB}	=	-0.188 lb/lb
∂ PL/∂ W_{jo}	=	-1.00 lb/lb
∂ PL/∂ $ISP_{B/T}$	=	849 lb/sec_{vac}
∂ PL/∂ $ISP_{o/T}$	=	912 lb/sec_{vac}
∂ PL/∂ $ISP_{B/\dot{W}}$	=	1184 lb/sec_{vac}
∂ PL/∂ $T_{B/\dot{W}}$	=	0.0857 lb/$lb_{S.L.}$
∂ PL/$ISP_{o/\dot{W}}$	=	979 lb/sec_{vac}
∂ PL/$T_{B/ISP}$	=	0.0238 lb/lb
∂ PL/$T_{o/ISP}$	=	0.212 lb/lb
ΔPL: 2.5g Limit	=	-13640 lb
ΔPL: 1.0g Limit	=	+7360 lb
∂ PL/∂ W_{PB}	=	0.060 lb/lb
∂ PL/∂ W_{Po}	=	0.230 lb/lb

Note: The slash after a subscript indicates the last quantity is held constant. For example $/\dot{W}$ means that flow rate is held constant.

The results of the parametric alternate mission analysis are presented in Figures 10-33 and 10-34 for the FR-3 and FR-4 vehicles, respectively. A specific alternate mission tabulation is shown in Section 8.8 of this volume.

10.4 MASS FRACTIONS

The mass fraction is defined as:

$$\text{Mass Fraction} = \lambda = \frac{\text{Weight of usable propellant in the element}}{\text{Total element weight - Element payload weight}}$$

This is plotted versus the usable propellant weight for the orbiter and booster elements of the FR-3 system in Figure 10-35 and for the FR-4 system in Figure 10-36.

Table 10-2. FR-4 Payload Sensitivity Partial Derivatives

$^*\partial PL/\partial W_{jB}$	=	-0.300 lb/lb (0.150 lb/lb for total of two boost elements)
$\partial PL/\partial W_{jo}$	=	-1.000 lb/lb
$\partial PL/\partial ISP_{/T}$	=	844 lb/sec
$\partial PL/\partial ISP_{o/T}$	=	1089 lb/sec_{vac}
$\partial PL/\partial ISP_{B/\dot{W}}$	=	1177 lb/sec_{vac}
$^*\partial PL/\partial T_{B/\dot{W}}$	=	0.147 $lb/lb_{S.L.}$
$\partial PL/\partial ISP_{o/\dot{W}}$	=	1198 lb/sec
$\partial PL/\partial T_{o/\dot{W}}$	=	0.389 lb/lb
$^*\partial PL/\partial T_{B/ISP}$	=	0.038 lb/lb
$\partial PL/\partial T_{o/ISP}$	=	0.035 lb/lb
ΔPL: 2.5g Limit	=	-14,100 lb
ΔPL: 4.0g Limit	=	+5600 lb
$^*\partial PL/\partial W_{PB}$	=	0.125 lb/lb
$\partial PL/\partial W_{Po}$	=	0.190 lb/lb

Note: *Derivatives are per booster element.

10.5 VARIATION OF THRUST-TO-WEIGHT RATIO AT LIFTOFF

Thrust-to-weight ratio at liftoff variations were made parametrically for the FR-3 and FR-4 systems. These assumed rubberized engines. The effect on gross liftoff weight and total system dry weight is shown in Figures 10-37 and 10-38 for the FR-3 and FR-4 vehicles, respectively. The fixed thrust engine combination points for the design point and other trial engine combinations are also shown. Note that an excursion to investigate a 14-3 FR-3 engine arrangement resulted in a glow of approximately 4.6 million pounds. It is off the plot of Figure 10-37. The liftoff thrust-to-weight ratio was approximately 1.2, which resulted in only a coarse convergence of the synthesis. Obviously the 14-3 configuration is not acceptable.

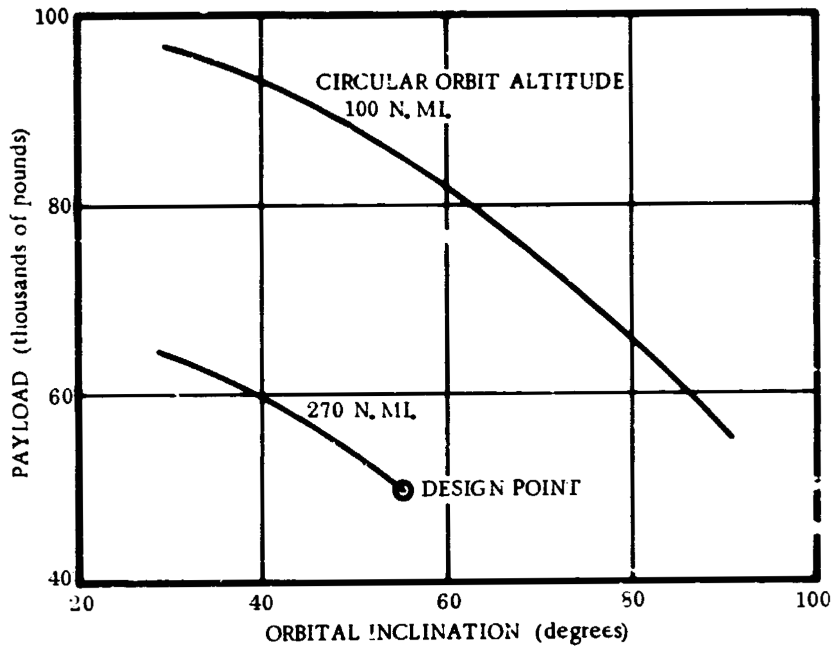


Figure 10-33. NASA FR-3 Point Design ~ Alternate Missions GLOW = 4.329 M lb

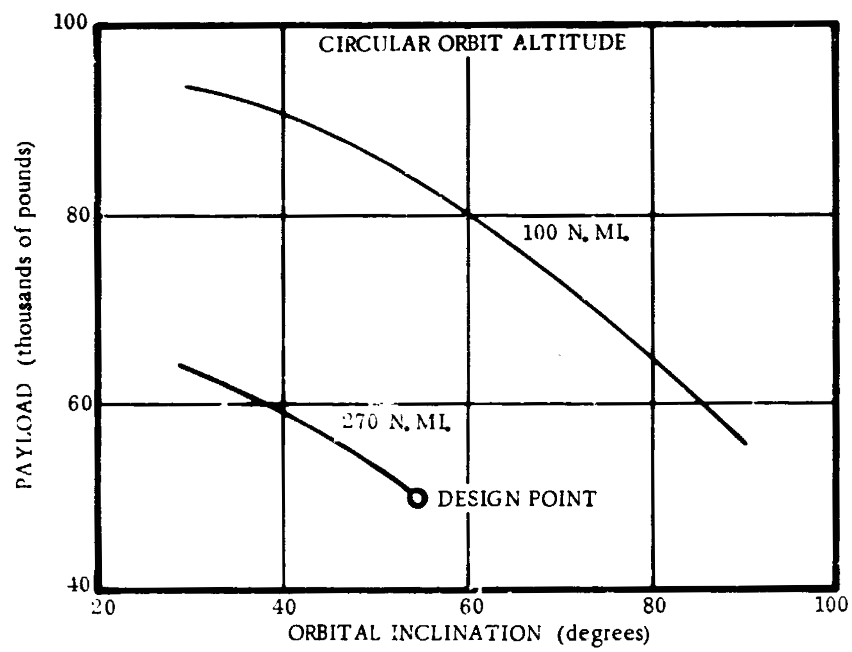


Figure 10-34. NASA FR-4 Point Design ~ Alternate Mission GLOW = 4.919 M lb

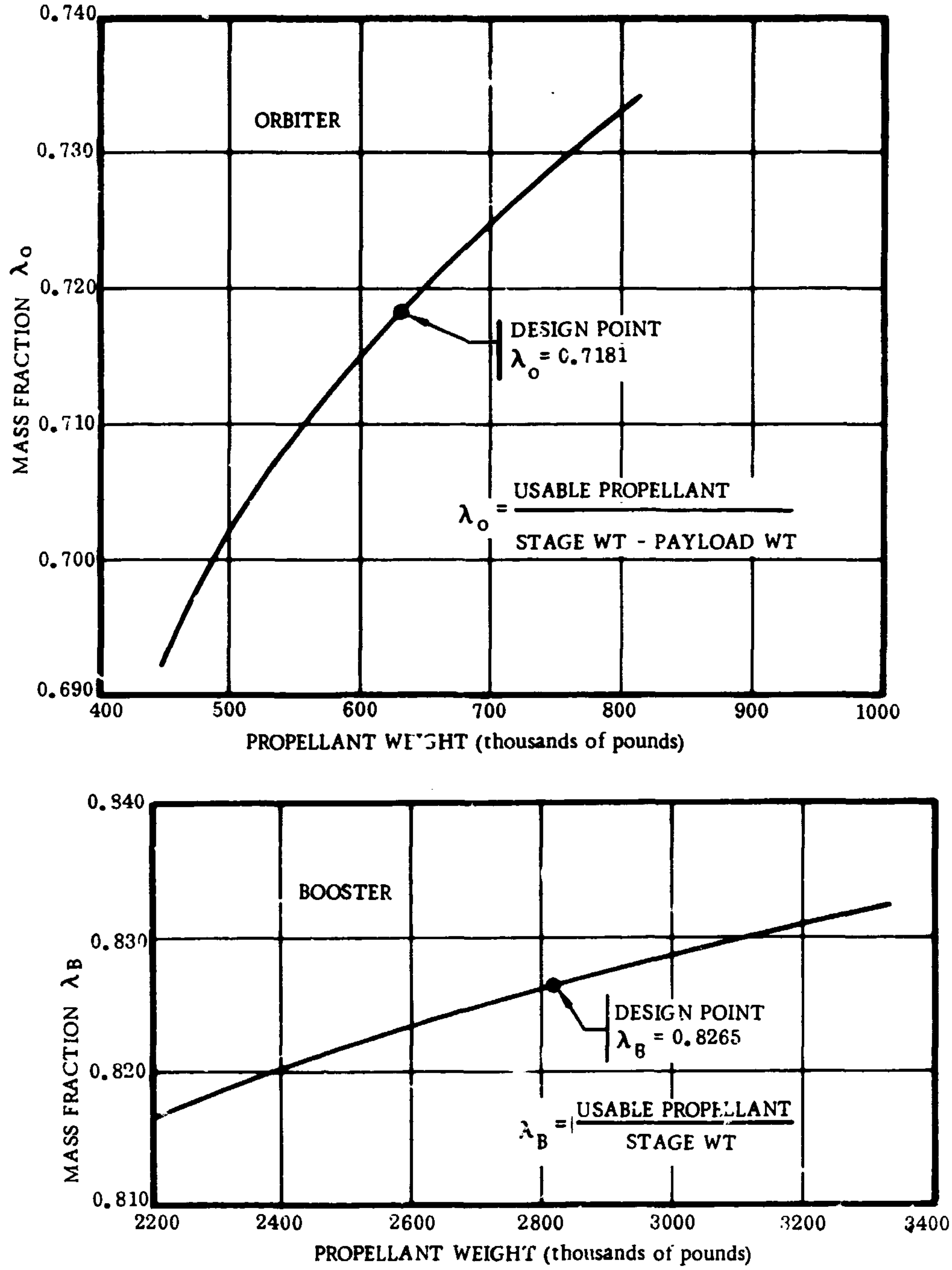


Figure 10-35. FR-3 Mass Fractions

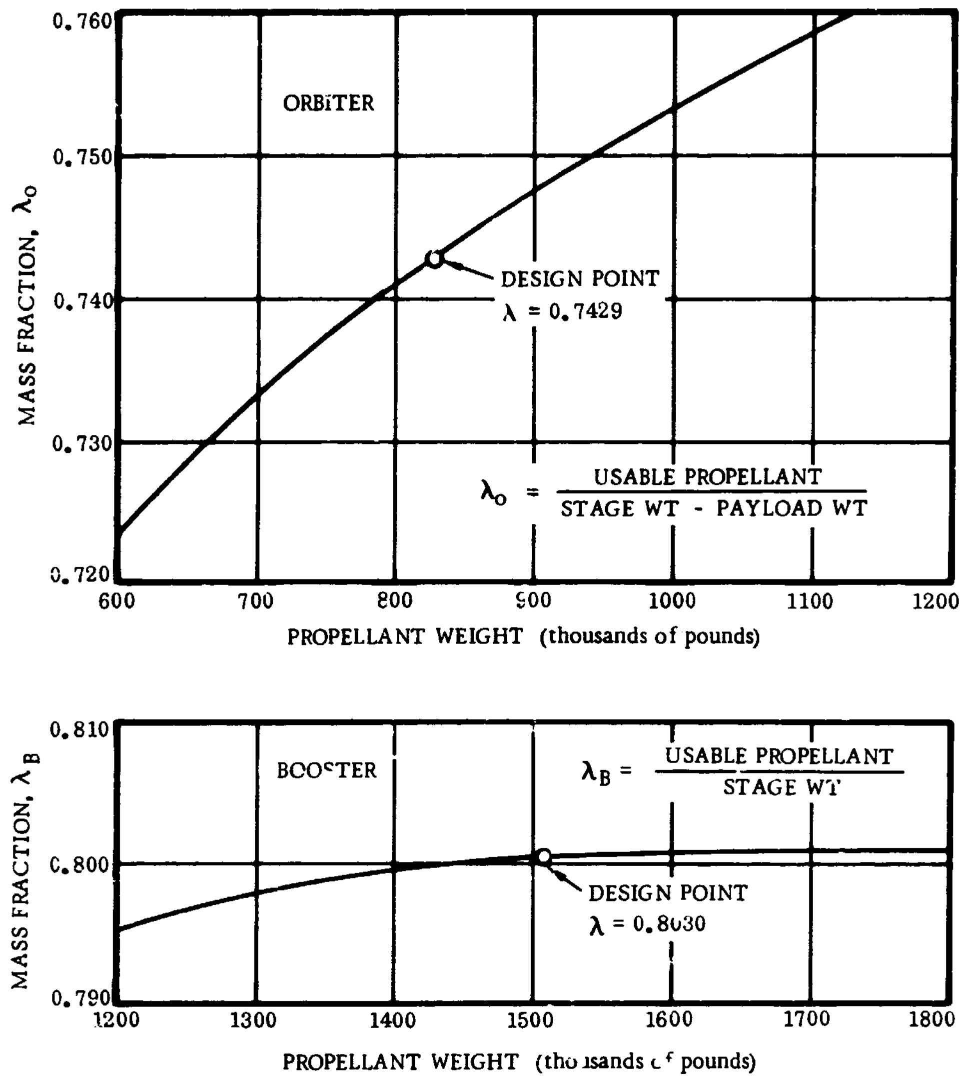


Figure 10-36. FR-4 Mass Fractions

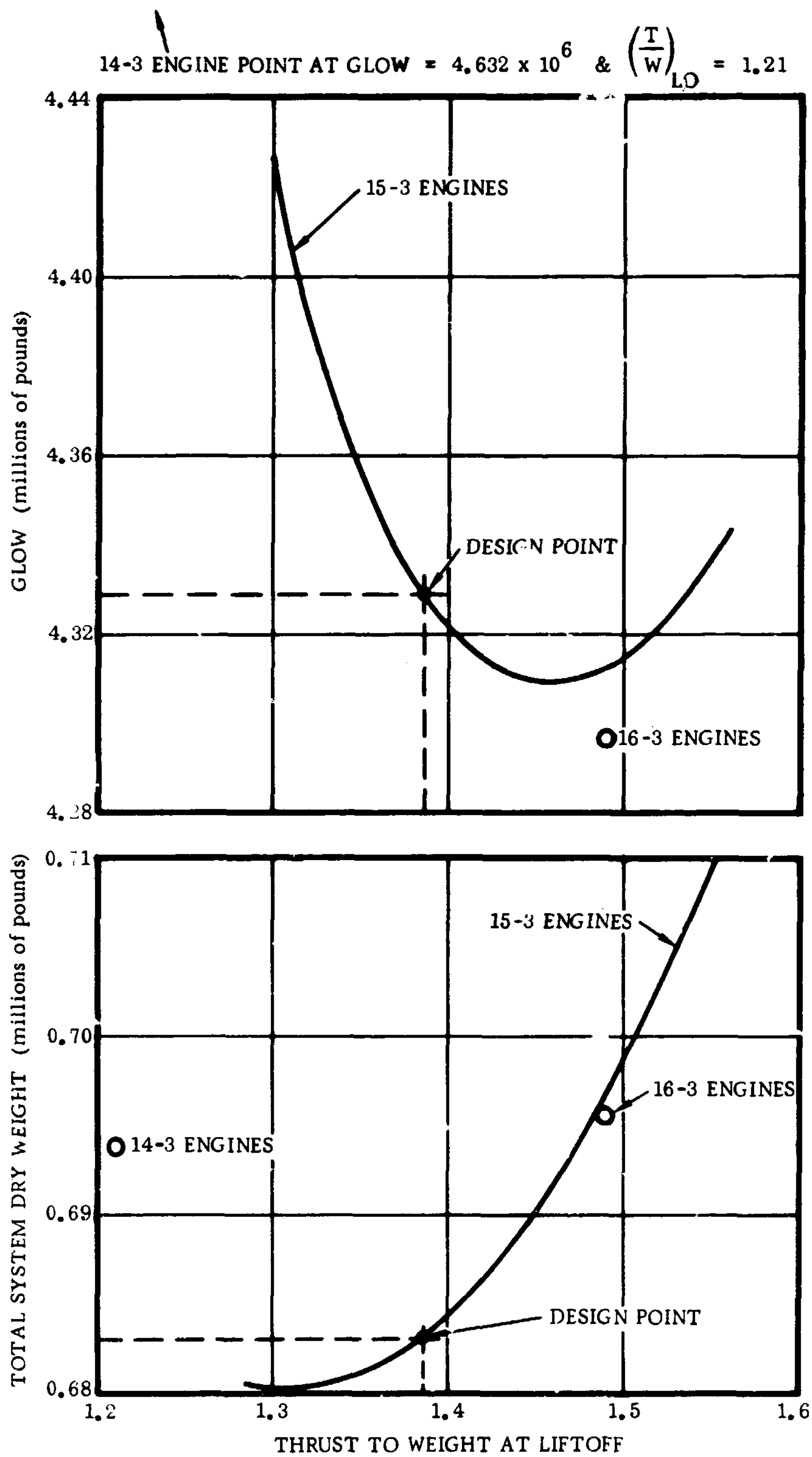


Figure 10-37. FR-3 System Weights Vs Thrust to Weight at Liftoff

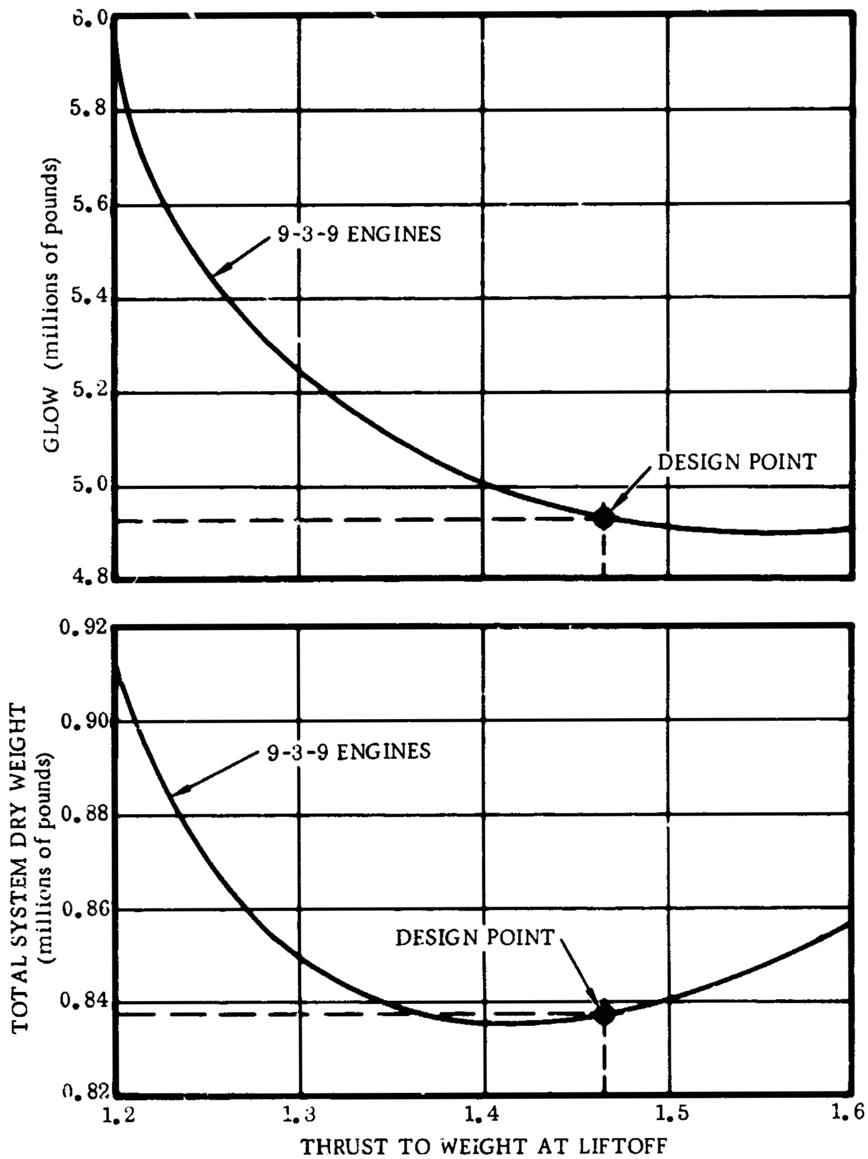


Figure 10-38. FR-4 System Weights Vs Thrust to Weight at Liftoff

SECTION 11

DEVELOPMENT PROGRAM

11.1 DEVELOPMENT CONSIDERATI 3

The primary function of the development program is to build, test and demonstrate the capability of a selected space shuttle vehicle design to satisfactorily perform all of a number of various select missions and to attain this capability within a reasonable targeted time span. The two vehicle concepts pursued are the FR-4, a three-element version, and the FR-3, a two-element version, as described in Sections 3 and 4 herein. The FR-3 and FR-4 mission and vehicle requirements are discussed in Section 2.

In general, the development programs for the FR-3 and FR-4 space shuttle concepts are very much alike. Particularly so, since both programs are constrained by an initial operational target date of mid-1976 and since there is about the same degree of difference between the booster and orbiter elements of either vehicle configuration. The orbiter elements of both vehicle concepts have essentially the same structure, general configuration, and subsystems with the FR-3 orbiter being only slightly smaller than the FR-4 orbiter. Table 11-1 compares a few of the basic physical characteristics of the orbiter elements so that a clearer understanding of the differences in development requirements may be attained.

The booster elements display the greater differences, with the FR-3 booster outsizing its FR-4 counterpart. Even the aerodynamic configurations vary considerably in the nose, crew, and jet-engine compartment areas. The FR-3 booster nose is more blunt, and the crew area is on top of the jet engine compartment rather than in front like the FR-4 vehicle. The basic structures of the two boosters differ inasmuch as the FR-3 body consists primarily of two separate integral tank structures, whereas the FR-4 is a single tank with an intermediate bulkhead. The 33-foot diameter tank of the FR-3 is some 10 feet greater than the FR-4. The cross-section of the FR-3 vehicle (not including the seven-foot landing gear or the vertical tail heights) is 41-feet across the bottom and 37-feet high, as compared to 34-feet and 29-feet, respectively, for the FR-4. The FR-3 thrust structure is more complex as it supports 15 rocket engines in lieu of the nine engines on the FR-4. Table 11-2 compares the basic characteristics of these two boosters.

Table 11-1. Orbiter Characteristics Comparison

	FR-3 Orbiter	FR-4 Orbiter
Weight, lb		
Propellant	630,000	825,500
Fly Back Fuel	2,870	3,200
Structure	213,000	246,900
Total*	925,600	1,161,100
Landing	286,600	322,400
Volume, ft^3		
Fuel	15,000	19,100
Oxidizer	7,600	10,000
Propellant	22,600	29,100
Total*	89,100	107,500
Geometry		
Length, ft	179	191
Body Wetted Area, ft^2	14,900	16,900
Body Planform Area, ft^2	4,900	5,570
Propulsion		
F/W	1.53	1.22
No. of Engines	3	3
Total Vac. Thrust, lb	1,414,800	1,414,800

*Totals are not summations of the subelements shown, but include other items as well.

For the total vehicle or multielement configuration for launch, the FR-3 concept matches its one booster element against the two required for the FR-4 vehicle concept. The differences in the vehicle launch configurations cause some differences in their respective flight trajectories. These differences, as noted in Table 11-3, reflect slightly different flight test conditions between the vehicles which are not enough to significantly alter the type or number of R&D vehicle launches between them.

Some of the program considerations that constrained establishment of the baseline development programs include the following. The foremost is the firm target date as established for the initial operational mission in mid-1976. Another is the strong emphasis on minimizing the total number of R&D vehicles because of their relatively high production costs (when compared to existing comparable launch vehicles), and maximizing the number of these R&D vehicles converted for operational use. Expendable hardware must either not exist or be held to an absolute minimum.

Table 11-2. Booster Characteristics Comparison

	FR-3 Orbiter	FR-4 Orbiter
Weight, lb		
Propellant	2,809,600	1,507,500
Flyback Fuel	46,900	30,700
Structure	469,700	294,800
Total	3,399,800	1,877,500
Landing	517,300	324,600
Volume, ft^3		
Fuel	92,900	49,800
Oxidizer	36,100	19,400
Propellant	129,000	69,200
Total	235,800	122,400
Geometry		
Length, ft	210	199
Body Wetted Area, ft^2	26,600	18,400
Body Planform Area, ft^2	8,170	6,070
Propulsion		
F/W (Veh. Launch)	1.39	1.46
F/W (Singl. Elem)	1.77	1.9
No. of Main Engines	15	9
No. of Jet Engines	4	3
Total Thrust, lb	6.0M	7.2M

Table 11-3. Trajectory Data Comparison

	FR-3 Vehicle	FR-4 Vehicle
Trajectory Data		
Max. αq, lb/ft^2	670	658
Staging Dyn Press, lb/ft^2	50	50
Staging Vel (Rel.) ft/sec	10,900	9,400
Staging Alt, ft	187,500	179,300
Staging Flt Path Angle, deg	2.2	5.8
Injection Vel. (Internal), ft/sec	25,900	25,900
Injection Alt, ft	260,000	260,000

The development approach followed is one of satisfying alternate and sometimes conflicting requirements from various sources. In meeting the imposed mission requirements, consideration must be given to the triple vehicle functional or operational requirements of a space shuttle; that is, launch vehicle, spacecraft, and aircraft. Applications of current capabilities or state-of-the-art must be utilized wherever feasible, if a timely availability is to be attained. However, for such a vehicle as a fully reusable space shuttle, new developments will be required where existing capability is not adequate. In some areas new technologies must be explored in time to implement adequate design decisions in support of an orderly planned program aimed at a specific operational target goal. These proposed support technologies are identified and described in Section 4 of Volume X, and a more detailed discussion of the proposed development programs for the FR-3 and FR-4 vehicle configurations is included in Section 2, Volume X.

11.2 PROGRAM SUMMARY

The baseline development program presented herein reflects both the FR-3 and FR-4 space shuttle configurations with exceptions noted. The development program reflects a combined Phase C/D effort, with a contract award date assumed for the beginning of the second quarter of CY 1971 and continuing some 66 months to the first operation launch on 1 October 1976. This reflects the earliest likely availability of an operational launch. A more nominal approach, where reasonable manufacturing and test activities are allowed to pace the program, would greatly reduce the risk but would likely extend the first operational launch into CY 1977, or possibly even early 1978. The current baseline is considered attainable under a degree of development risk and increased costs, especially in tooling, manufacturing, and testing.

The key milestones for the FR-3 and FR-4 development programs are reflected in Figure 11-1, and a typical total program schedule in Figure 11-2. A comprehensive wind tunnel test program and TPS material development program would be initiated prior to vehicle development go-ahead. Development of the jet (cruise) engines and the main rocket propulsion engines, it is assumed, would also be initiated prior to the start of this phase C/D go-ahead by six months or more, and engine selection would have been made prior to a preliminary design review.

Overall ILRV development time reflects a nominal state-of-the-art advance, particularly in the tooling, manufacturing, and testing areas. The state of the art achieved should be that estimated attainable by CY 1972. For specific manufacturing and tooling considerations for both the FR-3 and FR-4 vehicles, refer to Section 2, Volume X. However, Figure 11-3 is a pictorial presentation of the manufacturing sequence and flow for the typical orbiter elements; the booster elements being somewhat similar in approach but different in basic body tank structures.

Testing in support of vehicle development actually begins well in advance of the combined phase C/D program with design information type tests; i.e., wind tunnel analysis of specific configurations under varying environment and effects on critical maneuvers.

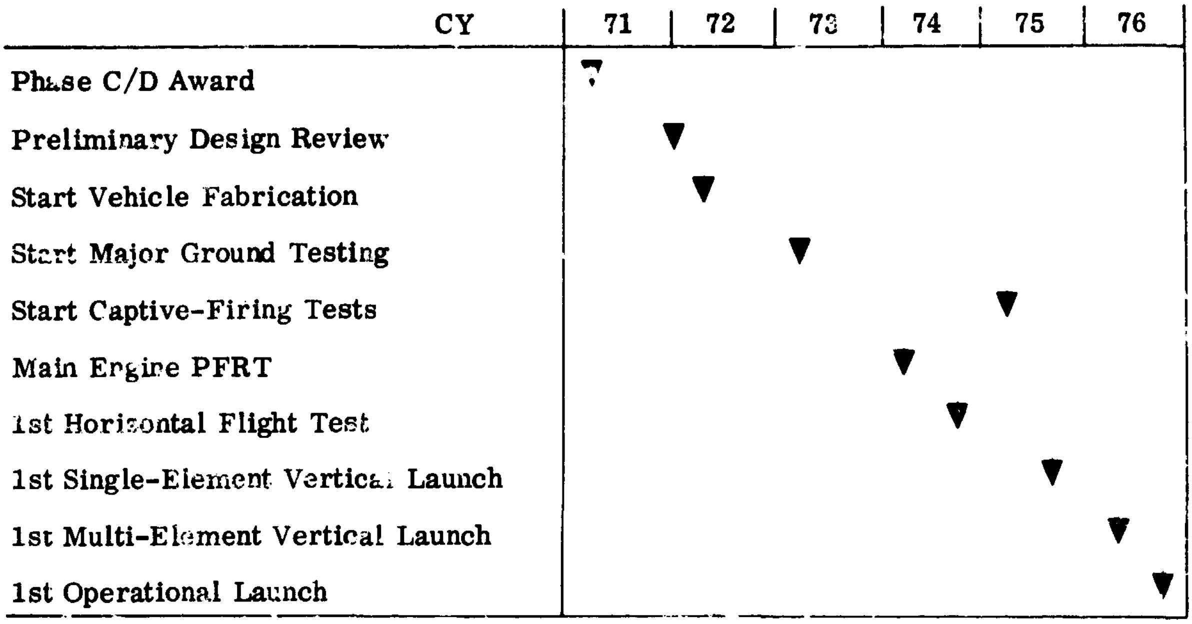


Figure 11-1. Program Summary Milestone Schedule

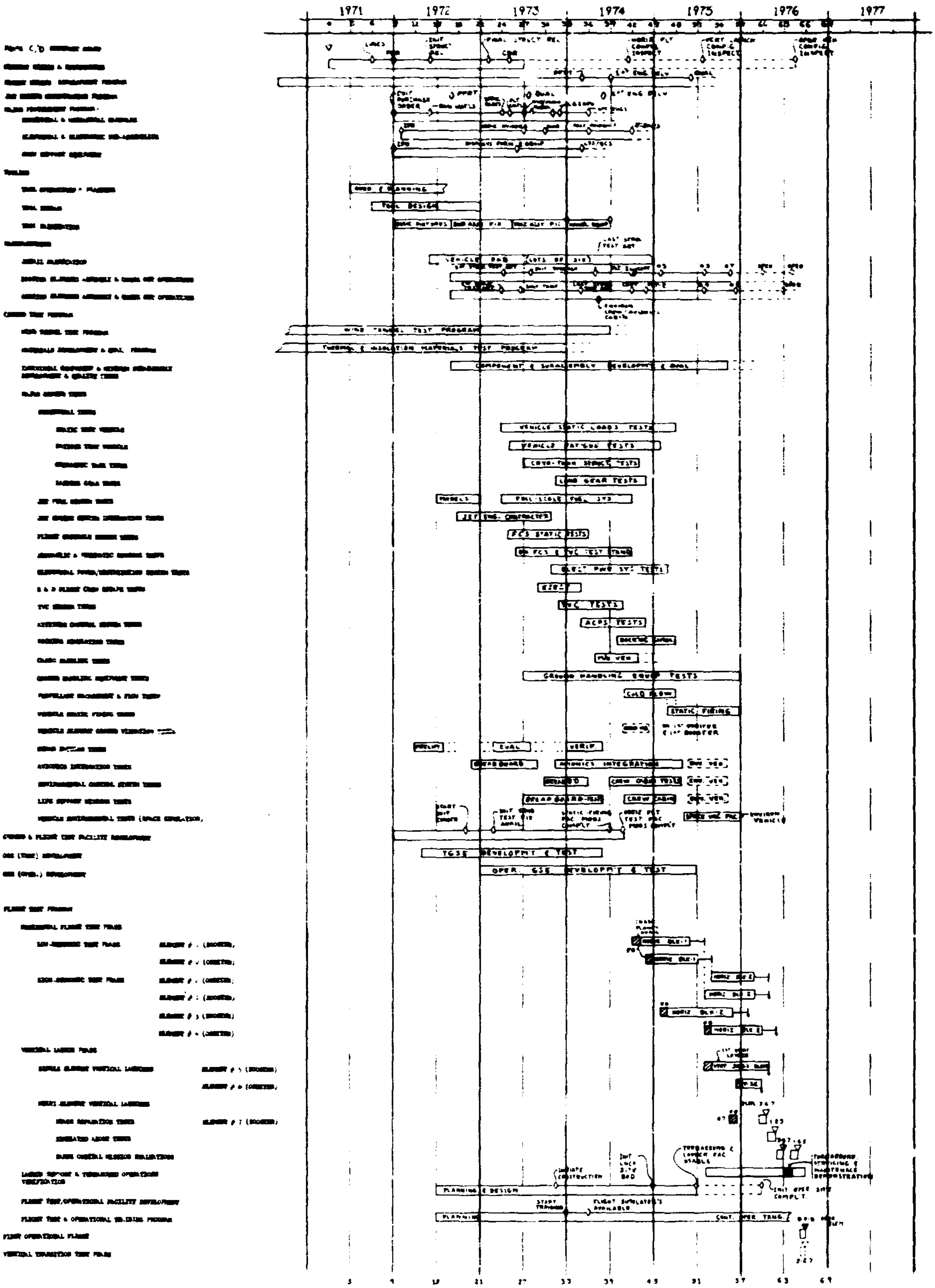
Also, advanced testing would develop special material handling techniques and applicability to the specific ILRV space missions. Individual component design support or evaluation testing and vehicle subsystem or subassembly testing would begin after the PDR of the combined C/D phase, and then only after sufficient design, tooling, and fabrication to support such tests. This later test phase, identified as the major ground tests, closely supports the horizontal and vertical flight test phases.

The combined ground and flight test programs span some 41 months. Scheduling of the ground and flight tests is aligned in support of succeeding tests which require more severe test conditions. Specific milestones must be met in the ground test program before the horizontal flight tests are begun. In turn, specific milestones must also be attained in the horizontal flights before the vertical launches may begin.

Manufacturing fabrication and assembly of the required test articles must be geared to deliver these test articles as required in the test program. The sequencing of test hardware in the subassembly areas has a direct bearing on initial flight article availabilities and must be considered in establishing the desired test article delivery requirements.

Facility and ground support equipment planning, design, construction, test, and/or checkout are scheduled for proper integration with the airborne hardware test program

11.2.1 GROUND TESTING. The major ground test program, exclusive of the wind tunnel and component and materials development and qualification programs, begins in the second quarter of CY 1973 and ends in the fourth quarter of CY 1975 (33 months). The development activities on the master program schedule concern primarily the



ACPS	ATTITUDE CONTROL PROPULSION SYSTEM	LSS	LIFE SUPPORT SYSTEM
APU	AUXILLARY POWER UNITS	M/U	MOCK-UP
CDR	CRITICAL DESIGN REVIEW (BUILD-TO-SPECS READY)	PDR	PRELIMINARY DESIGN REVIEW (START DETAIL DESIGN)
ECS	ENVIRONMENTAL CONTROL SYSTEM	PFRT	PRELIMINARY FLIGHT RATING TEST
.CS	FLIGHT CONTROL SYSTEM	PWR	POWER
FO	FIELD OPERATIONS	QUAL	QUALIFICATION FOR FLIGHT
LG	LANDING GEAR	TVC	THRUST VECTOR CONTROL

Figure 11-2. Summary Development Program Plan

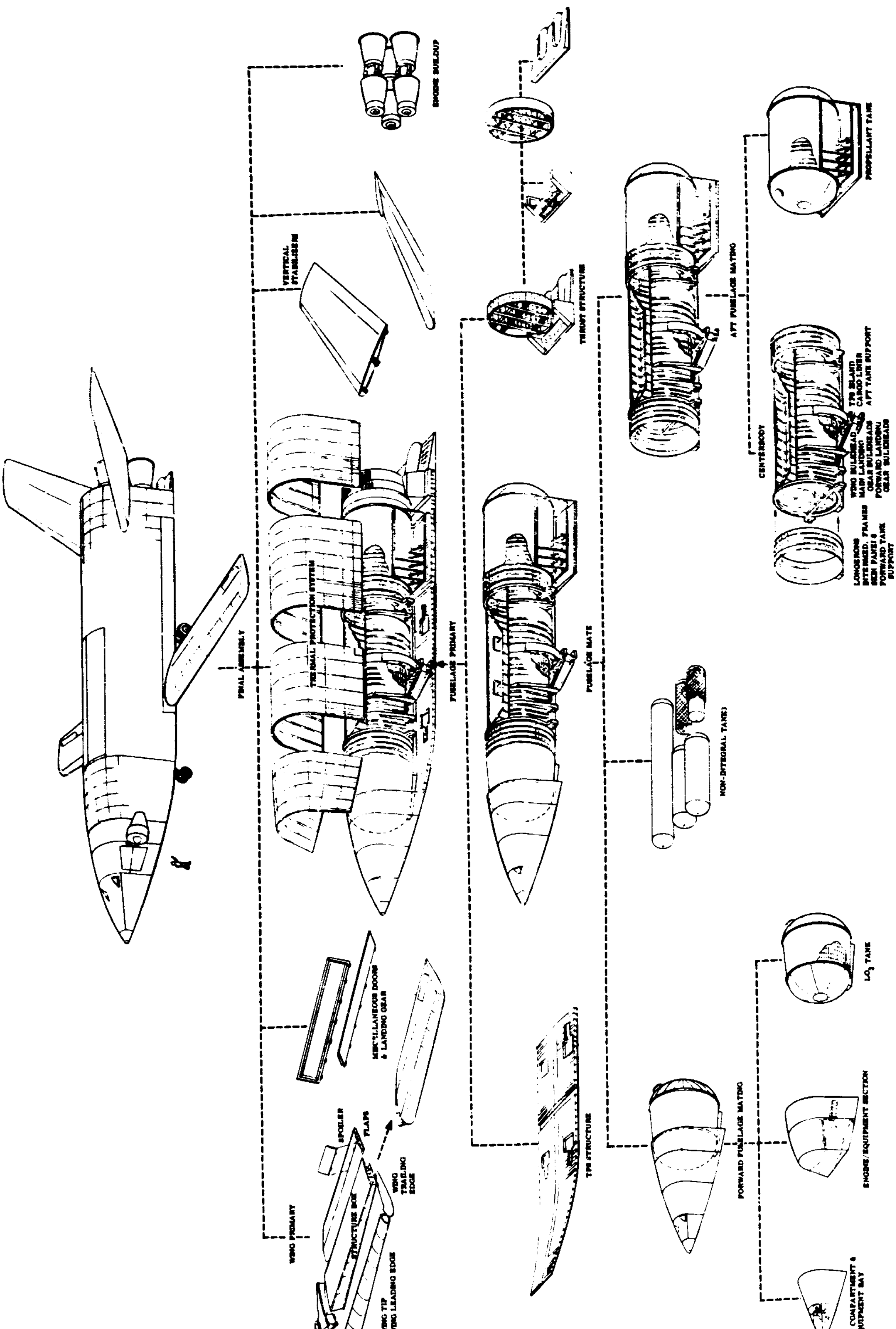


Figure 11-3. FR-4 Orbiter Element

airborne equipment development tests; testing of the ground support and handling equipment is covered only in a summary fashion to reflect appropriate time interfaces, etc. The basic ground test program is laid out in a sequential fashion that tends toward progressive support of the more severe or complex test conditions, including appropriate and timely support to the various phases of the flight test program.

The approach to establishing the ground test program for these baseline vehicles is built on two basic considerations:

a. The need to thoroughly ring out all structural and subsystem critical performance parameters insofaras possible prior to their flight phase verification or demonstration. (This is considered essential in establishing the required confidence level for the initial vertical flight tests, all of which are manned.)

b. The need to maintain the ground test program costs within reasonable budgetary constraints. (This latter goal is approached through conservation of test hardware and ground test facilities wherever feasible.)

Details of the ground test program are covered in Section 2, Volume X of this report. The major ground tests are also reflected in the program schedule of Figure 11-2 and identified with respect to the test hardware requirements in Table 11-4. This table summarizes the tests and test hardware for the orbiters only; the booster tests would be similar except for the obvious orbiter-only tests.

Due to the large size of the principal ground test articles, existing industrial or government facilities are considered as the best approach wherever the necessary modifications to these facilities to accommodate the specific space shuttle configuration are feasible. The higher risk or potential problem areas reflected in this ground test program are associated with the capability for adequately testing sufficiently large sections of thermally-protected skin surfaces or thermal protection subsystem panel assemblies at or near the temperatures expected on vehicle entry. Such tests are currently assumed limited to small sections of the body and vertical tail leading edges and test-specimens of the TD NiCr or other materials representing the booster and orbiter elements.

11.2.2 FLIGHT TESTING. The flight test program begins 42 months from go-ahead and spans some 23 months to the last development flight. The two basic flight test phases (horizontal and vertical) overlap by seven to eight months; however, the vertical launch phase does not begin until the design limit loads for horizontal flight have been adequately demonstrated. The horizontal flight tests would be conducted at an existing test site such as Edwards Air Force Base. The vertical launches would be performed at a site designated for the initial operational launches. All flights in both phases are manned, and all flight test vehicles are recoverable and reusable, as in the operational program. This overall test approach is more aligned with an aircraft approach to testing than with the launch vehicle approach, as discussed in Volume X.

Table 11-4. FR-4 Orbiter Major Ground Test Hardware Summary

Test Identification	Crew/Avionics Compartment	Jet Engine Compartment	Integral LO_2 Tank	Integral LH_2 Tank	Centerbody Structural Section	Payload Bay Propellant Tanks	Wings, Pivots, & Support Structure	Empennage	Rocket Engine Thrust Structure	Vertical Stabilizers	Major Doors & Hatches	Stage Separation Hardware	Thermal Protection Subsystem	Landing Gear	Jet Engines	Main Rocket Engines	Propellant Plumbing	Jet Fuel Tanks & Distrib. Subsystem	Hydraulic & Pneumatic Subsystem	Flight Control Subsystem	Attitude Control Propulsion Subsystem	Electrical System Hardware	Orbiter Docking Equipment	Cargo Deployment Equipment	Horizontal Flight Avionics Subsystem	Orbital Mission Avionics Subsystem	Crew Furnishings, Controls & Displays	Environmental Control Subsystem	Life Support Subsystem
Structural Static Loads Tests		●			●		● 1	●	●	●	●	● 2																	
Structural Fatigue Tests	●		● 3	● 3		●	● 1	●	●	●				●		S													
Jet Fuel Subsystem Tests																		●											
Hydraulic and Flight Control Subsystem Tests	S						S	S		S				S	S				●	●					● 2				
Thrust Vector Control Subsystem Tests									● 4							S			● 2										
Attitude Control Subsystem Tests																					●								
Docking Simulation Tests																							●						
Cargo Handling Tests																			● 2					●					
Electrical Subsystem Tests																						●							
Cold-Flow and Static-Firing Tests			●	●	●	●	S	●	●	S	● 2		● 2	S		●	●		● 2										
Human Factors Testing	S																										●		S
Avionics Integration Tests																									●	●			
Environmental Control Subsystem Tests	● 5																					● 5			S	S	● 2	●	● 2
Life Support Subsystem Tests	● 5																					●					● S	● S	●
Crew/Avionics Cabin Mission Environmental Tests	●																					●	●		●	●	●	●	●

● Test/Hardware Applicability
S Simulated Hardware Configuration
1 Only One Wing and Pivot Structure Required
2 Partial Subsystems or Hardware
3 Two Sets of Propellant Tanks
4 Uses Static or Fatigue Test Article
5 Uses Mission Environmental Test Vehicle

There are seven flight test elements (four booster and three orbiter) for the FR-4 configuration as compared to only six elements (three-booster and three-orbiter) for the FR-3 vehicle. Four, under either configuration, are required to fully satisfy the horizontal flight test phase. The first two elements delivered (a booster and an orbiter) would satisfy the basic flying requirements within a restricted, low-subsonic flight envelope. Both elements would be modified or updated as required and join the third and fourth elements in extending the flight tests into a high subsonic flight regime.

The fifth vehicle element, a modified booster in the case of the FR-4 and an orbiter in the case of the FR-3 configuration, is introduced at the start of the vertical flight test phase for single-element launches. All seven elements for the FR-4 and all six elements for the FR-3 are required to support the multi-element launches. When the four elements used in the horizontal tests complete their program, they will be modified as required and retrofitted for a vertical launch capability (initial delivery of these four elements did not include vertical launch capability) and utilized in the multi-element launch phase. The vertical flight test phase serves to verify and demonstrate the launch vehicle and spacecraft capabilities of the space shuttle configuration and its design mission compatibility. Figure 11-4 is a composite flight program schedule showing the time phasing of both the horizontal and vertical flight test phases. The principal flight test objectives and their applicability to each type test within the horizontal and vertical flight phases are summarized in Table 11-5.

The flight test articles used in the R&D program would eventually be refurbished and turned over to support the operational program. However, consideration should be given to holding back one complete vehicle for some limited period of time for extended test evaluations and potential flight problem analyses. In selecting elements for retention in the continuing flight backup test phase, consideration should be given to flight elements which were the first produced and which are heavily instrumented for limit-loads testing; these types of vehicles are the least suited for immediate operational status.

The all-manned flight test approach in this program may tend towards a reduction in total R&D costs through conservation of high-cost test vehicles, but does so with some element of associated risk. A potential problem area is associated with the transition phasing from horizontal test flights to vertical launch tests. The facets of this potential problem are concerned with adequate demonstration of the entry attitude control subsystem, the orbiter thermal protection subsystem, and the post-entry wing and engine deployment subsystem prior to their first full-requirement flight (which is manned under this baseline approach). These areas need further investigation before an optimum test solution can be devised. Possible solutions to this problem are reflected in the alternate development approaches of Section 2, Volume X.

The first two multi-element vertical launches serve to demonstrate the vehicle staging maneuver and horizontal flight-mode recovery and cruise fly-back (to launch site) for

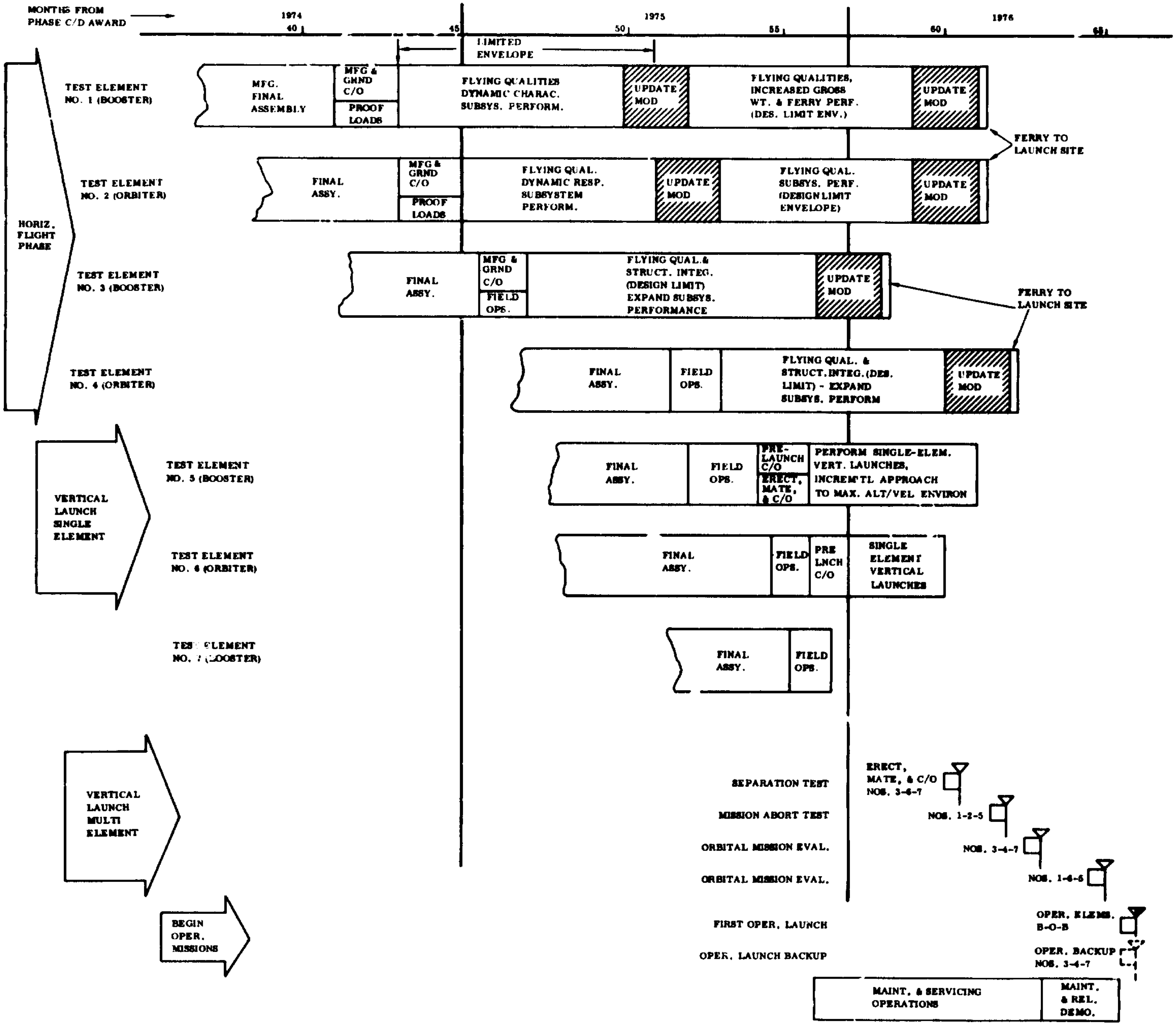


Figure 11-4. Summary Flight Test Program

Table 11-5. Flight Test Program — Summary Test Objectives

Test Objectives	Horizontal Flight Phase				Vertical Launch Phase			
	Low Subsonic		High Subsonic		Single Elem. Launch	3-Elem. Launch		
	Booster	Orbiter	Booster	Orbiter		Sep.	Abort	Orbit
Demonstrate ground handling equipment and procedures.	●	●	●	●	●	●	●	●
Verify vehicle erection and mating operations.						●	●	●
Verify vehicl /launch complex compatibility.					●	●	●	●
Verify pre-launch ground purging operation.					●	●	●	●
Verify vehicle/turnaround facilities compatibility.					●	●	●	●
Verify adequacy of cargo loading equipment.								●
Demonstrate horizontal takeoff and/or landing.	●	●	●	●	●	●	●	●
Verify airframe structural integrity.			●	●	●	●	●	●
Demons. adequacy of TPS.					●	●	●	●
Demonstrate satisfactory horizontal-flight characteristics.	●	●	●	●				
Demonstrate satisfactory hypersonic through transonic flight characteristics.					●	●	●	●
Verify satisfactory jet engine performance.	●	●	●	●	●	●	●	●
Demonstrate satisfactory performance of the rocket propulsion system.					●	●	●	●
Demonstrate adequacy of the ACPS/mission compatibility.					●	●	●	●
Demonstrate vehicle subsystem compatibility.			●	●		●	●	●
Demonstrate postentry horizontal flight configuration attainment.					●	●	●	●
Demonstrate satisfactory horizontal flight engine deployment.					●	●	●	●
Demonstrate horizontal flight cruise and ferry capability.			●	●				
Demonstrate preflight tanking and launch operations.					●	●	●	●
Demonstrate satisfactory subsystem performance.			●	●	●	●	●	●
Demonstrate satisfactory vertical flight characteristics.					●	●	●	●
Demonstrate satisfactory booster staging sequence.						●	●	●
Demonstrate adequacy of the boost phase abort maneuver.						●		
Demonstrate vehicle/mission performance capability.						●	●	●
Verify adequacy of on-orbit cargo handling.								●
Verify adequacy of the rendezvous and docking maneuvers.								●
Demonstrate vehicle postflight serviceability and maintainability.					●	●	●	●
Obtain data on vehicle components/hardware reusability.					●	●	●	●
Verify adequacy of ECS and LCS in mission environment.					●	●	●	●
Demonstrate satisfactory performance of the automatic landing subsystem.			●	●	●		●	
Demonstrate guidance and control subsystem accuracy.						●	●	●
Verify adequacy of cryogenic tank insulation.					●	●	●	●

the two booster elements. The orbiters would be required to circle the earth (once around) and return to the launch site or other designated alternate landing site. The second of the two launches would be planned to simulate a boost-phase abort condition, thereby evaluating the staging and element recovery maneuvers under alternate conditions.

The third and fourth vertical launches are aimed at extending exploration of the orbiter's flight environment and vehicle performance to operational mission simulations in orbit. The orbiter on both flights will perform the orbital transfer and target rendezvous maneuvers, docking operations as applicable, and simulated cargo/crew transfer or payload deployment and retrieval operations applicable to the orbiter element. Following the final two R&D flight test vehicle recoveries, the shuttle elements will serve during their turnaround sequence to demonstrate the booster/orbiter maintainability and serviceability, as well as demonstrating the capability of the facility to adequately support the initial operational flights and as an indication of the reusability of each shuttle element.

11.3 TEST FACILITIES

The test facilities identified as necessary to support the ground test programs for the FR-3 and FR-4 vehicle concepts are discussed in Section 2.2.7 of Volume X of this report. However, Table 11-6 summarizes the general test facility requirements and identifies the probable existing facilities capable of satisfying these requirements for both vehicle concepts. The variations in the major ground test program as caused by differences between the FR-3 and FR-4 concepts are as follows:

a. Static firing of the FR-3 booster cannot be accomplished within the MTF S-1C test stand due to the 41-foot envelope required for the fuselage. It could possibly be tested at the MSFC S-1C stand if the stand were extended vertically to accept a 186-foot long vehicle.

b. The FR-3 requires a 60 percent increase in the volume of fuel (over the FR-4) for the static firing tests in (a) above.

Two types of flight test facilities are required; those for the horizontal aircraft type tests and those for the vertical launch tests. Facilities for horizontal flight testing exist at several DOD bases within the continental United States. The recommended facility is Edwards Air Force Base, California, which has been used many times for countless varieties of experimental aircraft. In general, all necessary ground support equipment is available, the only likely exceptions being specialized tow bars. Examination of available hangar space is desirable and hangar clearances will require checking, especially in the area of empennage clearance.

Vertical launch testing will be accomplished at an operational launch facility. For this facility, either the conceptual new facility or the modified KSC Complex 39 (both are pictorially presented and discussed in Section 9 of this volume) will be used. The facility construction/modification schedule should be formulated to meet the schedule requirements of the test program (see Figure 11-2). Other than scheduled need, the test requirements imposed no constraints on facility design. In effect, partial construction completion of the launch facility will allow implementation of the early portions of the vertical launch test program.

Table 11-6. Test Facility Requirements For FR-3 and FR-4 Space Shuttle Vehicles

Test	Type of Facility Required	Location
Wind Tunnel	High and low speed tunnels	Contractor facility
	Plasma Arc tunnel	MSFC
	Hypersonic shock tunnel	
Material development	Environmental test lab	Contractor facility
Component development	Component test lab	Contractor facility
Structural static load test	Test tower and hangar	MSFC static load test facility (Contractor hangar for wing test)
Structural fatigue test		
Cryogenic cycling	Test Tower	None suitable existing
Acoustic	Cryogenic propellant	Contractor/or MSFC
	Loading equipment	
	180ds acoustic chamber	Houston MSC. acoustic test facility
Jet fuel system tests	Jet engine test facility	Contractor facility
Flight control subsystem test	Vehicle skeleton mockup	Contractor facility
Electrical subsystem test	Electrical test equipment	Contractor facility
R&D crew escape test	Compartment mockup	Contractor facility
Thrust vector control test	Test stand	Contractor MSFC
		Static test facility
Attitude control subsystem test	Hangar building	Contractor facility
Docking simulation tests	Docking simulator	MSC simulation laboratory
Cargo & Ground Handling test	GSE test facility	MSFC GSE test facility
Propellant flow & vehicle static firing	Large static firing test facility	MTF S-IC test stand*
Ground vibration tests	Hangar	Contractor facility
Human factors test	Life science support facility	Contractor facility
Avionics integration tests	Electronics lab	MSC Houston
Environmental control tests	Space vacuum chamber	MSC chamber A
Horizontal flight tests	Aircraft test facility	Edwards AFB, Calif.
Vertical launch tests	Launch complex	ETR, Florida

*FR-3 booster cannot be accommodated at MTF S-IC stand. It can be tested at MSFC S-IC stand if stand is modified.

SECTION 12

COSTS

The space shuttle cost analysis is described in detail in Volume X of this report. The following figures and tables present a summary of the FR-3 and FR-4 program costs. Program costs include the cost of development, production, and 10 years of steady state operations. Figures 12-1 and 12-2 show the development investment, and operations costs at various traffic rates. Table 12-1 compares the FR-3 and FR-4 configuration total program costs at a traffic rate of 50 launches per year. The FR-3 and FR-4 costs are compared at other traffic rates in Figure 12-3. Total program costs are quite close at the lower launch rates, but the FR-4 configuration becomes increasingly more expensive as the traffic rate increases.

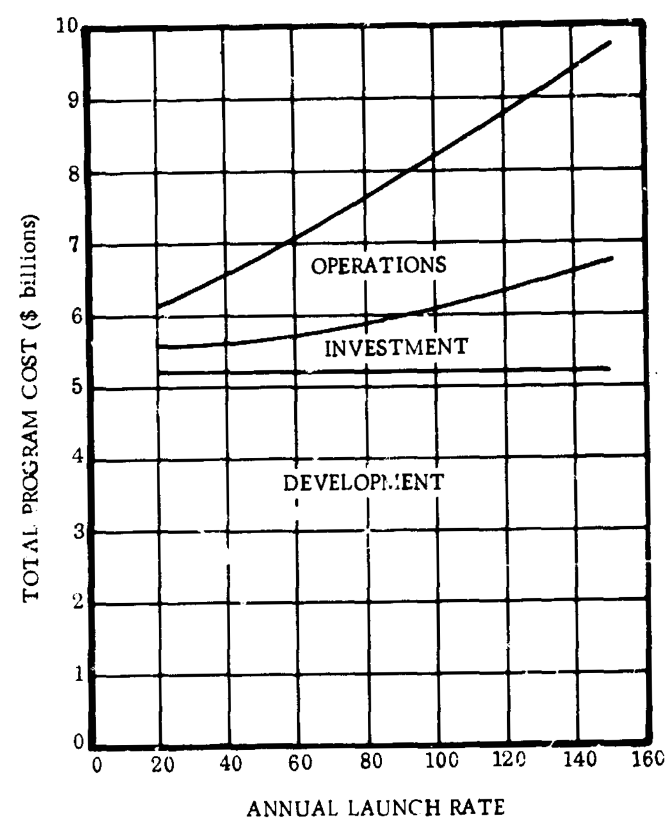


Figure 12-1. FR-3 Total Program Cost Versus Traffic Rate

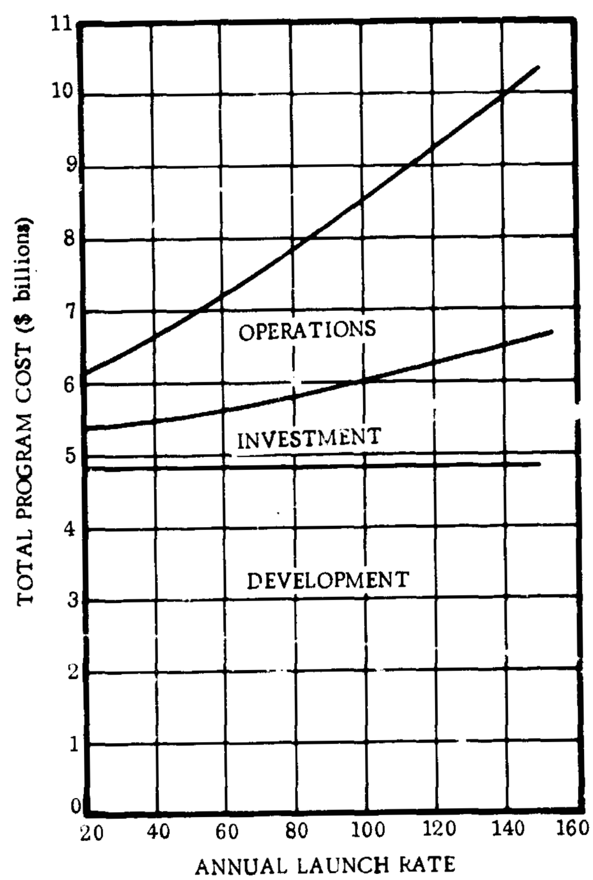


Figure 12-2. FR-4 Total Program Cost Versus Traffic Rate

Table 12-1. Total Program Cost Comparison
(50 Launches Per Year)*

	FR-3	FR-4
Development	5231	4883
Investment	485	694
Operations	1151	1387
Total Program	6867	6964

*Millions of dollars

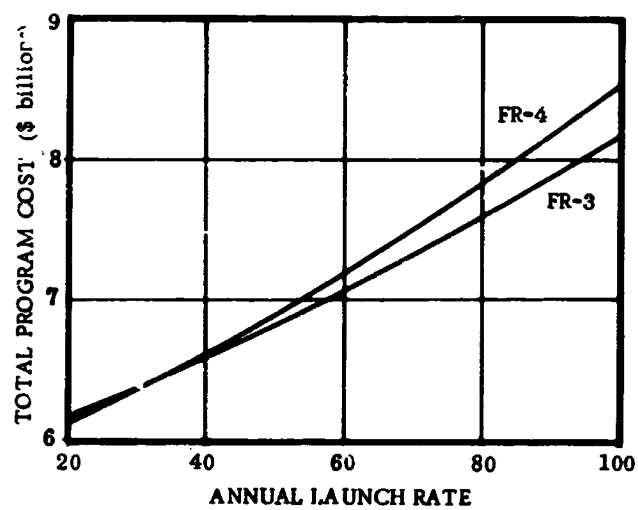


Figure 12-3. FR-3 Versus FR-4 Program Cost Comparison

A detailed comparison of the development program costs is shown in Table 12-2. Operations costs for the FR-3 and FR-4 vehicles are compared in Table 12-3 for a traffic rate of 50 launches per year. The effects of launch rate on operating costs are illustrated in Figure 12-4.

The detailed data appearing in Volume X include cost breakdown in both Convair and NASA formats, a discussion of cost methodology, and an analysis of cost sensitivities to variations in several vehicle design characteristics for the FR-3 and FR-4 configurations.

Table 12-2. Development Program Cost Comparison*

DEVELOPMENT	FR-3	FR-4
Airframe	984	942
Propulsion	557	527
Avionics	79	79
AGE	254	243
Ground Test	1267	1098
Flight Test	1384	1331
Facilities	224	248
SE&I	452	385
Total	5201	4853

*Millions of dollars

Table 12-3. Comparison of Operations Cost Per Launch (50 Launches Per Year)*

ITEM	FR-3	FR-4
Personnel	0.273	0.341
Materials		
Booster	0.805	1.121
Orbiter	0.477	0.511
Propellants & Gases	0.513	0.573
GSE Maintenance	0.108	0.102
Facilities Maintenance	0.126	0.126
Recurring Cost/Launch	2.302	2.774

*Millions of dollars

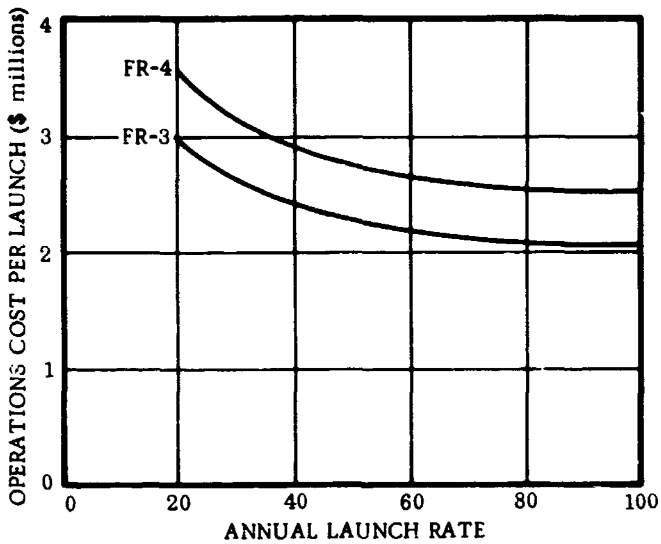


Figure 12-4. Recurring Cost Per Launch 25 to 100 Launches Per Year

SECTION 13

CONCLUSIONS

Comparison of the FR-3 and FR-4 systems shows that the FR-3 system out performs the FR-4. The FR-4 has a cost advantage below 35 launches per year, but over this number the FR-3 has lower total program costs.

The asymmetric arrangement of the FR-3 appears feasible if the thrust vector is canted as shown in the layouts. The FR-4 loses a great amount of symmetry when the vertical stack arrangement is replaced with the current arrangement necessary for payload bay access at all times.

Both systems are two-stage sequential burn concepts. The only real difference is that the booster element is made up of two equal parts in the FR-4. While the FR-3 booster is large, it is not outside current aircraft landing weight state of the art.

In assessing the overall picture, there is no apparent advantage in a three-element system when the NASA requirements are applied.

The FR-3 orbiter element should be shaped to suit the reduced cross-range requirements. This would include blunting the nose, depending on the final stipulated cross-range. Lowering the cross-range allows trimming of the orbiter element at higher angles of attack hypersonically and permits a more aft center of gravity. This is beneficial since keeping the center of gravity forward for high L/D entry is difficult. This concept should be considered in future orbiter configuring. The subsonic condition can be adjusted by placing the wing further aft in the stowed wing configurations.

While all the Phase A vehicles, boosters and orbiters, have stowable wings, it is recommended that future investigations cover fixed-type wings as alternatives, especially in the FR-3 booster element.