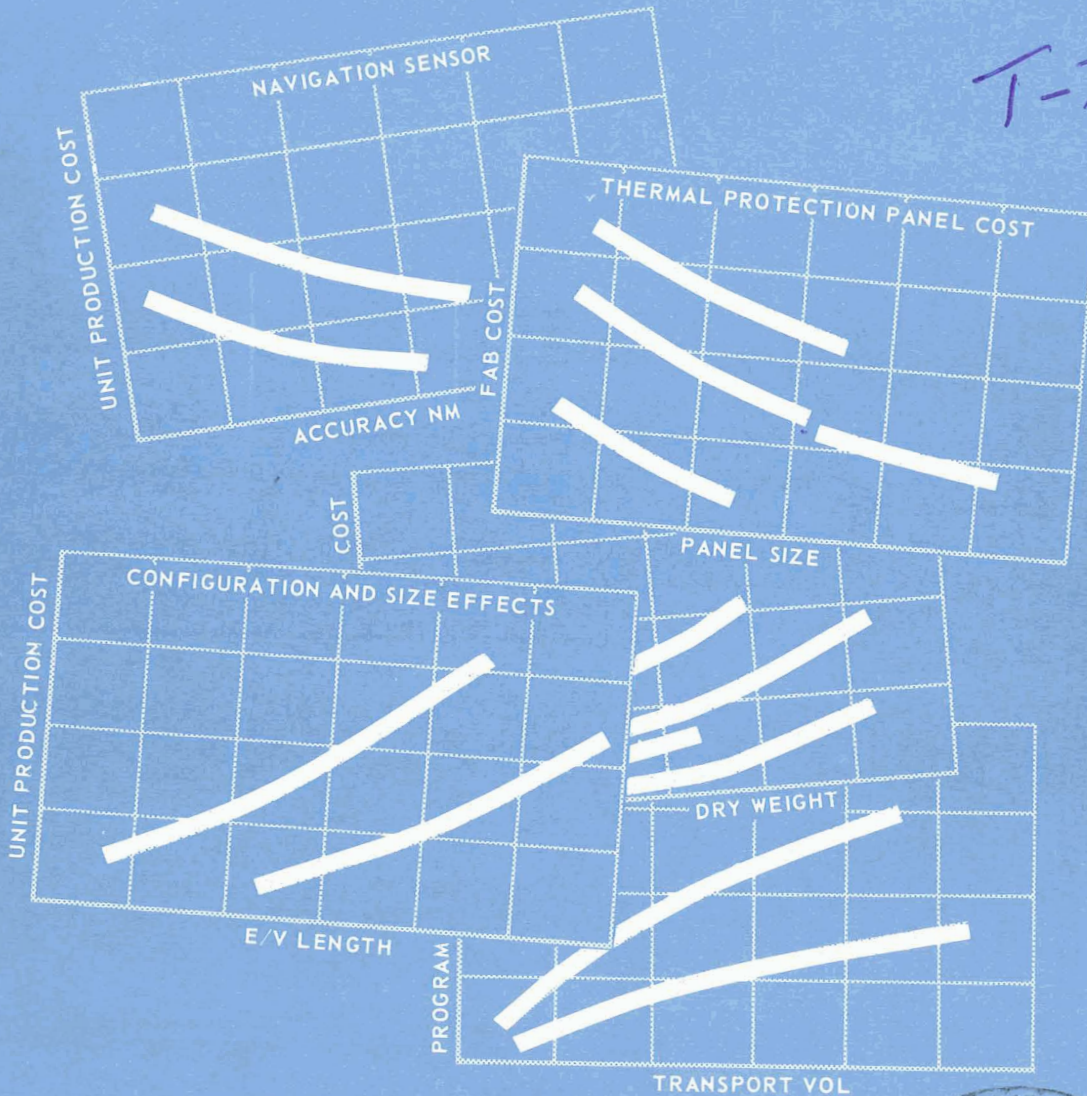


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OPTIMIZED COST/PERFORMANCE DESIGN METHODOLOGY

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VOLUME II DATA REVIEW AND ANALYSIS
BOOK 1 - DESIGN
CONTRACT NAS 2-5022



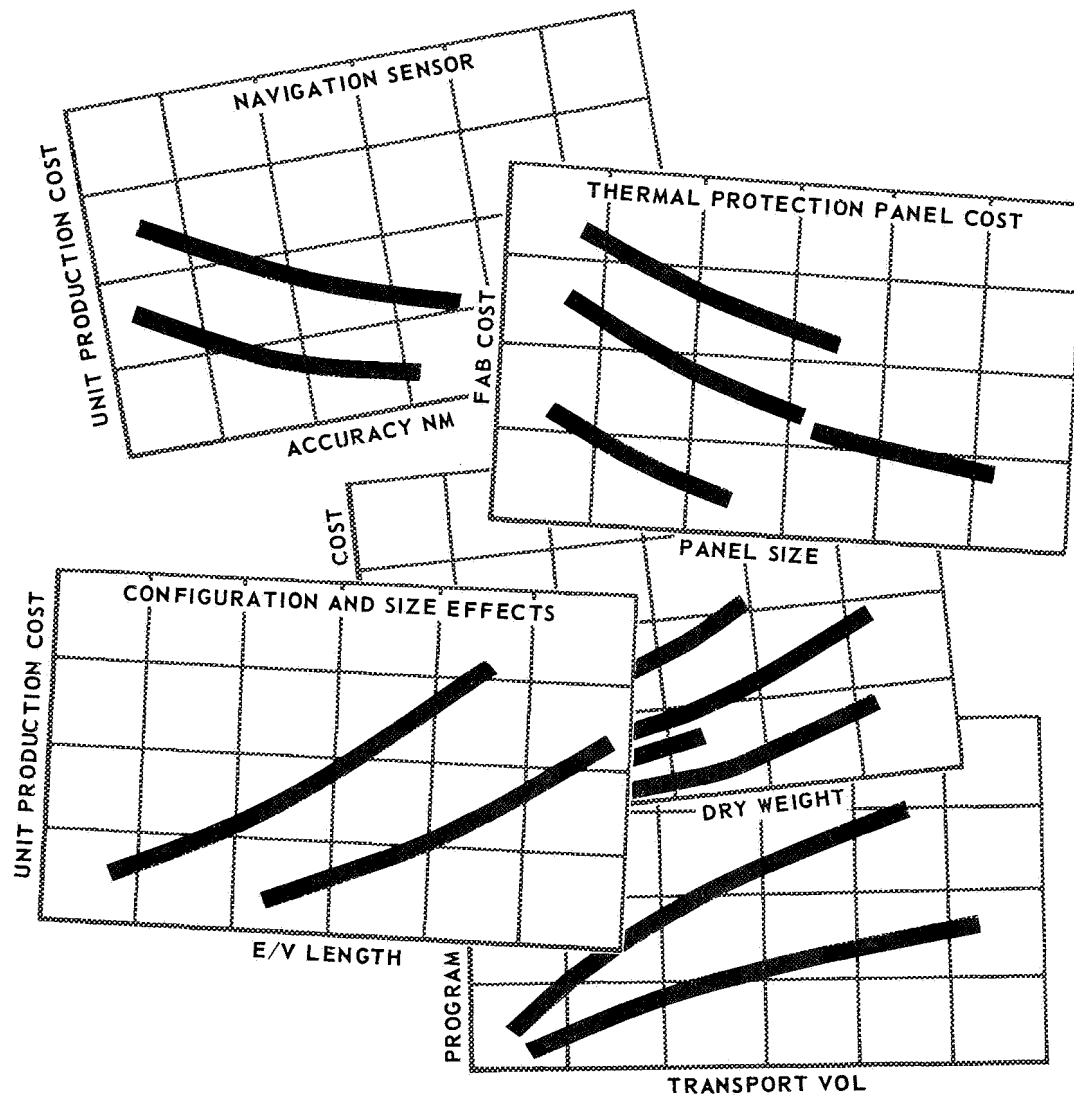
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OPTIMIZED COST/PERFORMANCE DESIGN METHODOLOGY



**VOLUME II DATA REVIEW AND ANALYSIS
BOOK 1 - DESIGN
CONTRACT NAS 2-5022**

**OPTIMIZED COST/PERFORMANCE
DESIGN METHODOLOGY**

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15 APRIL 1969

FOREWORD

This report is submitted to NASA, the Mission Analysis Division of OART, as part of the final reporting on Contract NAS 2-5022, Optimized Cost/Performance Design Methodology of Orbital Transportation Systems. This twelve month study was initiated in July 1968 and was performed in two general phases: a data review and analysis phase and a system evaluation phase. The reporting of the study is organized in three volumes but includes several books in Volumes 2 and 3. Volume 1 is a short summary of the complete study, Volume 2 covers the phase 1 data review and analysis, and Volume 3 covers the phase 2 system evaluation. The Study Manager was L. M. McKay; the major Task Leaders were P. T. Gentle, V. E. Henderson, L. E. Smith, and A. D. Trautman. The NASA Technical Monitor was C. D. Havill.

McDonnell Douglas gratefully acknowledges the support and cooperation of many companies which supplied information to the study. A list of the companies and their area of contribution is included in Appendix A.

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ABSTRACT

The broad objectives of this study were to gather historical cost and performance data, organize and analyze the data so that cost estimating relationships could be developed, and evaluate several system concepts for space logistics support.

The primary source of historical cost data was the Gemini and Saturn Programs and cost estimating relationships draw extensively on this experience. A range of reuse concepts were evaluated and optimum (least cost) concepts defined for a variety of program options. These include variations in such things as crew size, cargo capacity, program requirements, etc. for either ballistic or lifting body (M2-F2) entry vehicles.

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1. INTRODUCTION - The Optimized Cost/Performance Design Methodology study seeks to provide a method of using cost as a basic design parameter in identifying and defining more economical space transportation systems. This study was performed in six tasks as shown in Figure 1-1. Task 1 involved developing the cost data, organizing the data by categories, and developing cost estimating relationships. Tasks 2 and 3 developed the requirements and the physical and functional characteristics of the alternate spacecraft subsystems and operations. An analytical cost model was formulated in Task 4. Task 5 developed the logic, data, and methods for systematically varying the design and operational specifications of each vehicle configuration. Using the data and tools developed in the other tasks, Task 6 determined the economically optimum design and operational philosophies; sensitivities to program size, launch rate, payload size; and the problem area and technology limitations.

The work performed in Task 2 is reported here. The objectives of this task were to:

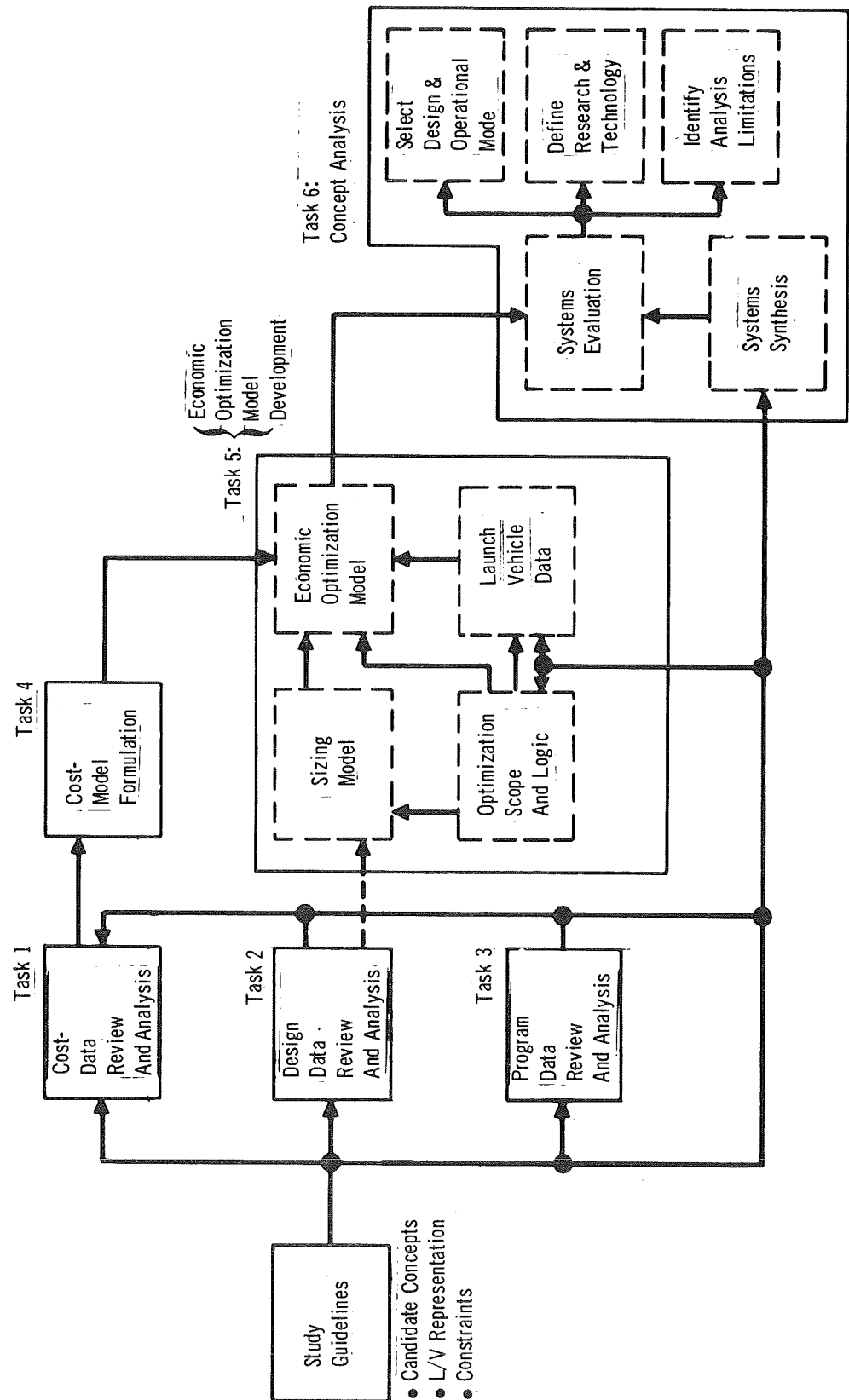
1. Define mission parameters and characteristics for a logistics mission to resupply a 12-man space station in a 300 na mi earth orbit. These mission parameters were to be based on the general parameters given in Reference 1-1.
2. Derive the functional and performance requirements for the several spacecraft subsystems. These requirements must cover a range of reuse categories ranging from completely expendable to completely reusable space vehicles.
3. Define alternate concepts for the spacecraft subsystems. These concepts are to include simple and sophisticated, current and advanced technology, and light and heavy concepts.

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Figure 1-1

STUDY TASK FLOW



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2. SUMMARY - The Task 2 effort, Design Data Review and Analysis, of the Optimized Cost/Performance Design Methodology study, has defined a range of mission parameters and characteristics for a logistics mission, derived spacecraft subsystem functional and performance requirements, and defined the physical and performance capabilities of several design approaches for each subsystem. A range of reuse concepts from fully expendable to fully recoverable space vehicles for both a ballistic and a low L/D lifting body entry vehicle was investigated. The work performed in Task 2 is reported in this book.

The progression of the reuse categories from fully expendable to fully reusable is shown in Table 2-1. These categories applied to both the ballistic (Configuration I) and the lifting body (Configuration II) configurations. For categories A and B, the ballistic vehicle had a 60 deg cone angle. A 40 deg cone angle was used for the other categories. For the lifting body configuration, the NASA M2-F2 configuration with an L/D of 1.1 was used. The boosters were treated parametrically and, for purposes of sizing a CPM and an adapter, were assumed to have a diameter equal to the base of the ballistic vehicles or to the maximum width of the lifting vehicle.

TABLE 2-1

Reuse Category Summary

Category Component	Expendable	Partially Reusable				Reusable
	A	B	C	D	E	F
Entry Vehicle	E	R	R	R	R	R
Maneuver Propulsion/ Cargo Module	E	E	R	R	R	R
Upper Stage Engines	E	E	E	R	R	R
Upper Stage Tanks	E	E	E	E	R	R
First Stage	E	E	E	E	E	R

E - Expendable

R - Reusable

Integral

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Using the general mission parameters specified in the RFP, detailed mission requirements and characteristics for each phase of a logistics resupply mission were defined. In addition to defining a nominal mission, variations were also specified. Requirements and characteristics defined included trajectories, load factors, thrust-to-weight and impulse-to-weight ratios, ΔV requirements, and length of time for mission phase.

The subsystem functional and performance requirements were then defined and quantized. Alternate concepts that could meet these requirements were synthesized for each subsystem. Five subsystems were identified as those where potentially significant variations in total system cost could be anticipated when varying the subsystem concept. These subsystems were structure (including thermal protection), propulsion, environmental control, avionics, and power. Specifications for these (excluding structure and thermal protection) were prepared and submitted to vendors for design, cost, and reliability data. The resulting data were used to verify or to up-date the spacecraft sizing program and to aid in formulating the cost estimating relationships.

From the alternate subsystem concepts, baseline subsystems were chosen for each reuse category. These are summarized in section 6, Tables 6-1 and 6-2. These baseline subsystems are a starting place for the subsystem-operational mode optimization performed in another task.

Subsystem alternates that were defined were used to synthesize total vehicle systems concepts for evaluation in a subsequent task.

Section 3 of this report contains the groundrules and assumptions that were applied in the study, then section 4, 5, and 6 contain the mission characteristics and Requirements; spacecraft characteristics; and Subsystem Requirements, Concepts, and Definitions.

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3. DESIGN GROUND RULES AND ASSUMPTIONS - The ground rules that guided the system and subsystem design effort, both those defined by NASA and by the contractor, are discussed below. Those defined by NASA in the RFP broadly defined the scope of the study.

3.1 NASA Defined Ground Rules - The mission to be performed by the spacecraft under study is for crew rotation and resupply of a 12-man space station. The following ground rules were specified at the onset to define the range of the major mission parameters.

1. The reentry spacecraft size will be varied over the range of crew sizes from two to twelve men and over the range of return cargo capacity from 200 to 2000 lb occupying from 100 to 400 cu ft.

2. Cargo space, either integral with the reentry spacecraft or in an expendable cargo module, will be provided for delivered cargo over the range of from 20,000 lb. to 200,000 lb with density variations between 5 and 10 lb cu ft.

3. Boost capability will be provided from both ETR and WTR into a 100 na mi orbit for inclinations of 50 deg, 70 deg, and polar. An orbital propulsive capability will also be provided to accelerate into a rendezvous orbit of 300 na mi altitude with a maximum plane change of 1 deg, to dock, and to initiate recovery.

4. Both land and water landings will be considered as primary landing modes.

5. The time for return from orbit will be variable over the range of from two to twenty four hours.

3.2 Contractor Defined Ground Rules and Assumptions - Additional ground rules developed during the study and a brief justification are given below:

1. Water landings will be considered for the expendable ballistic vehicles only. Requirements for refurbishment after a water landing cannot presently be estimated.

2. During orbital storage, all spacecraft subsystems will be in a quiescent state with all support requirements being supplied by the space station. Since the spacecraft is uninhabited during orbital operations,

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standby power is required. This has little effect on the space station total power requirements but does reduce the expendables required in the spacecraft.

3. The cargo will not require any support from any vehicle subsystems other than provisions for mounting and retention. Since the cargo quantity and density are mission variables, the make-up of the cargo cannot be postulated. Therefore, it is impossible to define heating, cooling, plumbing, etc., that might be required.

4. In assigning the reliability budget, it will be assumed that the probability of mission success for the spacecraft and the upper stage shall be 0.95. The goal for probability of mission success was established with NASA to be 0.95, the probability of crew survival was set at .995. Because of the variation, in design concepts (from modular to integral upper stage), it was established that these goals would in all cases be applied to the complete vehicle above the first stage of the launch vehicle. For modular configurations where a separate expendable upper stage is required, it was assigned an apportionment of 0.975. This was based on the S-IVB Criticality and Loss Reliability Analysis of the SA-501 vehicle, published 28 February 1967.

5. For the integral upper stage categories, the nominal impulsive velocity capability will be 16,000 ft/sec with a range from 15,000 ft/sec to 17,000 ft/sec investigated. Second stage velocity requirements can be varied if the first stage is a "rubber" launch vehicle. This range is based on the study reported in Reference 3-1.

6. The launch vehicle diameter will be infinitely variable with no upper limit within the range of spacecraft sizes being investigated. Because this study is oriented primarily toward spacecraft costs with only first order approximation of booster costs, the booster study will be done parametrically. Therefore, there is no reason to limit booster diameter.

7. Crew escape during launch will be provided up to the time of orbit injection, but the aerodynamic characteristics and flying quality of the escape capsule will not be investigated. For the lifting body configuration,

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the forward section of the spacecraft houses the men and equipment for an escape capsule. This section represents an aerodynamic configuration that has not yet been investigated and require extensive analyses which are beyond the scope of this study. However, the sizing of the necessary reaction control system and parachutes will be accomplished.

8. Abort trajectories will not be used to design the thermal protection system. Because the spacecraft is not reusable after abort, local hot spots or even minor failures in the thermal protection system can be tolerated during abort. Since the vehicle is designed for reuse, it can accept an increase in heating for a short time without minor failures occurring. With a 100 na mi injection altitude, the 100 deg to 200 deg increase in temperature in the laminar flow region that would occur during abort will not be great enough to cause more than minor failures in the reusable thermal protection system.

9. The nominal boost trajectory used to determine boost loads and environment will be based on Saturn V with no variation considered.

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4. MISSION CHARACTERISTICS AND REQUIREMENTS - The primary influences governing the spacecraft design are the performance capabilities and spacecraft environments during each phase of the logistics mission. In this section, a nominal set of operational procedures and mission characteristics is defined for each mission phase to form a base point for system definition. Variations from the nominal characteristics are also given. Differences in mission characteristics attributable to the variation in entry spacecraft types (Ballistic and M2-F2) are noted. Data shown were derived from in-house studies and from the studies of References 4-1 through 4-5.

4.1 Prelaunch - Typical prelaunch activities are shown in Table 4-1, based on data for Apollo/Saturn V and Gemini B/Titan IIIM. The prelaunch time to be considered for each subsystem depends upon the point in the count-down sequence where the subsystem is activated.

4.2 Launch - The launch site can be either ETR or WTR depending on the desired orbit inclination. Launch into 50 or 70 degree inclination orbits is performed from ETR while launch into a 90-degree inclination orbit is performed from WTR. The insertion orbit in all cases is a 100 na mi circular orbit. Typical launch stage characteristics and requirements are presented in Table 4-2. These have been formulated from Saturn I and Saturn V (two stage) launch vehicle characteristics. A typical launch trajectory is shown in Figure 4-1. The first stage launch vehicle characteristics are assumed constant, such that the first stage provides essentially the same performance and same trajectory characteristics regardless of payload size and shape. Only the gross launch weight varies to accommodate second-stage ignition weight.

The second stage is considerably more flexible, allowing for trade-off studies with an integral launch-re-entry vehicle concept for the logistic spacecraft. The impulsive velocity requirements for the second stage generally decrease with increasing thrust-to-weight ratio, up to T/W values of about 1.5. For this study, the first stage of launch for the integral vehicles is assumed to provide sufficient velocity increment that the nominal second stage requirements are 16,000 ft/sec. The effects of varying these requirements from 15,000 to 17,000 ft/sec were also investigated.

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Table 4-2 also defines the launch escape system requirements and significant abort trajectory characteristics. These apply for all launch vehicles being considered. The escape rocket total impulse-to-weight ratio required is shown in the table. This is determined primarily by the pad abort requirements. When the entire lifting body spacecraft is used as the escape vehicle, the requirements are reduced. This is because the gliding capabilities of the vehicle can be utilized to provide additional range. Recovery after an abort is accomplished by parachute and is discussed in Section 4.7.

4.3 Orbital Maneuvering - After booster burnout, maneuvers required to rendezvous and dock with the space station utilize the on-board propulsion system. These activities are summarized in Table 4-3. After insertion into a 100 nautical mile circular orbit, the spacecraft remains there for proper phasing. The maximum ascent time (start of launch to docking) utilized for logistic missions is 24 hours, restricting the phasing time to 20 hours maximum. If the logistic spacecraft must remain in this phasing orbit up to the maximum time, a ΔV of 20 ft/sec is required to maintain the orbit due to drag decay effects.

TABLE 4-1
MISSION REQUIREMENTS AND CHARACTERISTICS

PRELAUNCH PHASE

Operational Procedures	Mission Requirements and Characteristics	
	Nominal	Variation About Nominal
o Spacecraft Systems Power-Up	Systems Power-Up = T-7.4 hr	None
o Crew and Passengers On-board	Crew On-board = T-2.0 hr	None
o Hatches Closed	Hatches Closed = T-105 min	None
o Service Tower Removed	Tower Moved = T-100 min	None
o Internal Power Utilized	Internal Power Utilized at T-23 min	None
o Umbilical Disconnect	Disconnect and Lift-off at T-0 sec	None

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TABLE 4-2
MISSION REQUIREMENTS AND CHARACTERISTICS

LAUNCH PHASE

Operational Procedures	Mission Requirements and Characteristics	
	Nominal	Variation About Nominal
<u>First Stage Boost</u>		
o Lift-off	Orbit Inclination = 50° from ETR	70° from ETR 90° from WTR
o Roll Maneuver to Desired Launch	Number of Crew = 9 Max. Launch q = 700 lb/ft ²	2 to 12 None
o Pitch Maneuver for Gravity Turn	Max. Load Factor = 5.0 g's Burn time = 150 sec	None None
o Engine Shutdown	T/W at lift-off = 1.25	None
<u>Second Stage Boost</u>		
o Second Stage Ignition	* Second Stage ΔV = 16,700 ft/sec	* 15,000 to 17,000 ft/sec
o Vehicle Stabilization	Initial T/W = .7	Minimum of 0.5
o Jettison Escape Tower		
o Pitch Maneuver for Trajectory Shaping	Maximum Load Factor = 5 g's Insertion Altitude = 100 na mi	None None
o Engine Shutdown	Insertion Orbit = 100 na mi circ	None
<u>Launch Abort</u>		
1. Launch Escape Tower	1. Tower T/W = 10.0	None
o Malfunction Detection	Tower Impulse/ Aborted Weight	
o Activation of LES	o Ballistic Capsules = 30 sec	None
o Separation from Booster	o Escape capsules = 30 sec	None
o Jettison Tower		
o Deploy Recovery System	Overpressure Limit = 10 psi	None
o Water Landing		
2. High Altitude Abort	2. High Altitude T/W = .03	Minimum of 0.025
o Shutdown Booster	Retrograde ΔV = 200 ft/sec	None
o Separation from Booster		
o Retrograde		
o Controlled Re-entry		
o Water Landing		

* Integral spacecraft only. Saturn type launch vehicle assumed for modular spacecraft.

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TABLE 4-3

MISSION REQUIREMENTS AND CHARACTERISTICS

ORBIT MANEUVERING PHASE

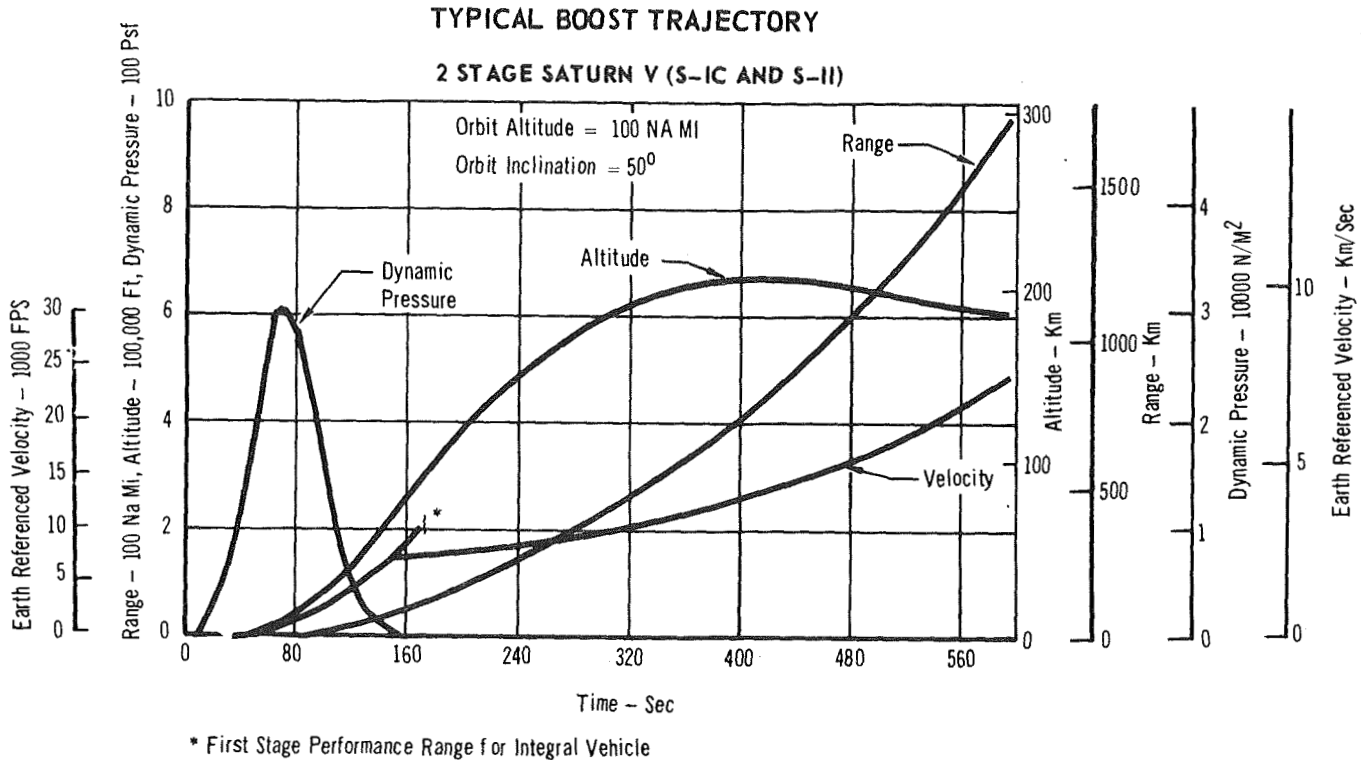
Operational Procedures	Mission Requirements and Characteristics	
	Nominal	Variation About Nominal
<u>Parking Orbit</u>		
*o Separate from upper Stage	ΔV Separation = 10 fps Minimum Time in Phasing Orbit = 30 min	None Maximum Time in Phasing Orbit - 20 hrs
o Adjust Orbit to 100 na mi Circular	ΔV for Drag Decay = 0 ft/sec	ΔV = 20 ft/sec, drag decay
o Coast in Orbit for Proper Phasing		
o Maintain 100 na mi Orbit during Phasing		
o Compute Rendezvous Maneuvers		
<u>Rendezvous</u>		
o Maneuver to Interim Elliptical Orbit	Gross Rendezvous ΔV = 775 ft/sec	ΔV Variation 640 to 775 ft/sec with Allowance for 1° Plane Change
o Adjust Orbit for Final Phasing	Space Station Orbit = 300 na mi	None
o Circularization to Terminal Rendezvous Orbit	Orbit at End of Gross Rendezvous = 285 na mi Gross Rendezvous Time = 18 hrs	None
o Verify Terminal Rendezvous Maneuvers	Terminal Rendezvous ΔV = 200 ft/sec Time to Perform Terminal Rendezvous = 25 min	None
o Perform Terminal Rendezvous	Range at End of Rendezvous = 100 ft	25 min to 70 min None
<u>Docking</u>		
o Visually Close on Space Station	Docking ΔV = 40 ft/sec Time to Dock = 30 min	None
o Dock with Space Station	T/W = .02 (fwd/aft) = 0.1 (lat/vert)	None
o Deactivate Spacecraft to Standby Status		
<u>Attitude Control</u>		
o Control Orbiting Spacecraft during all Orbital Maneuvering Phases	Angular Acceleration (min) = 1.0 deg/sec/sec Total impulse to S/C weight ratio = 0.8 sec	None None

* Unless upper stage is integral

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Figure 4-1



After achieving proper phasing, rendezvous consists of three phases: a gross rendezvous (open loop navigation), the terminal rendezvous (closed loop), and the docking (visual). The gross rendezvous is intended to transfer the spacecraft to a point close to the space station. The ΔV required to perform the gross rendezvous is shown in Figure 4-2 as a function of space station altitude. The design point selected in defining system requirements allows for a maximum one-degree plane change optimally split between the three transfer burns. This requires a ΔV capability of 775 ft/sec for gross rendezvous. At the end of the gross rendezvous, the spacecraft is typically 15 na mi below and from 50 to 100 behind the station. The thrust-to-weight ratio utilized should not be less than 0.3 in order to avoid large distortions in ΔV requirements and in orbit shape.

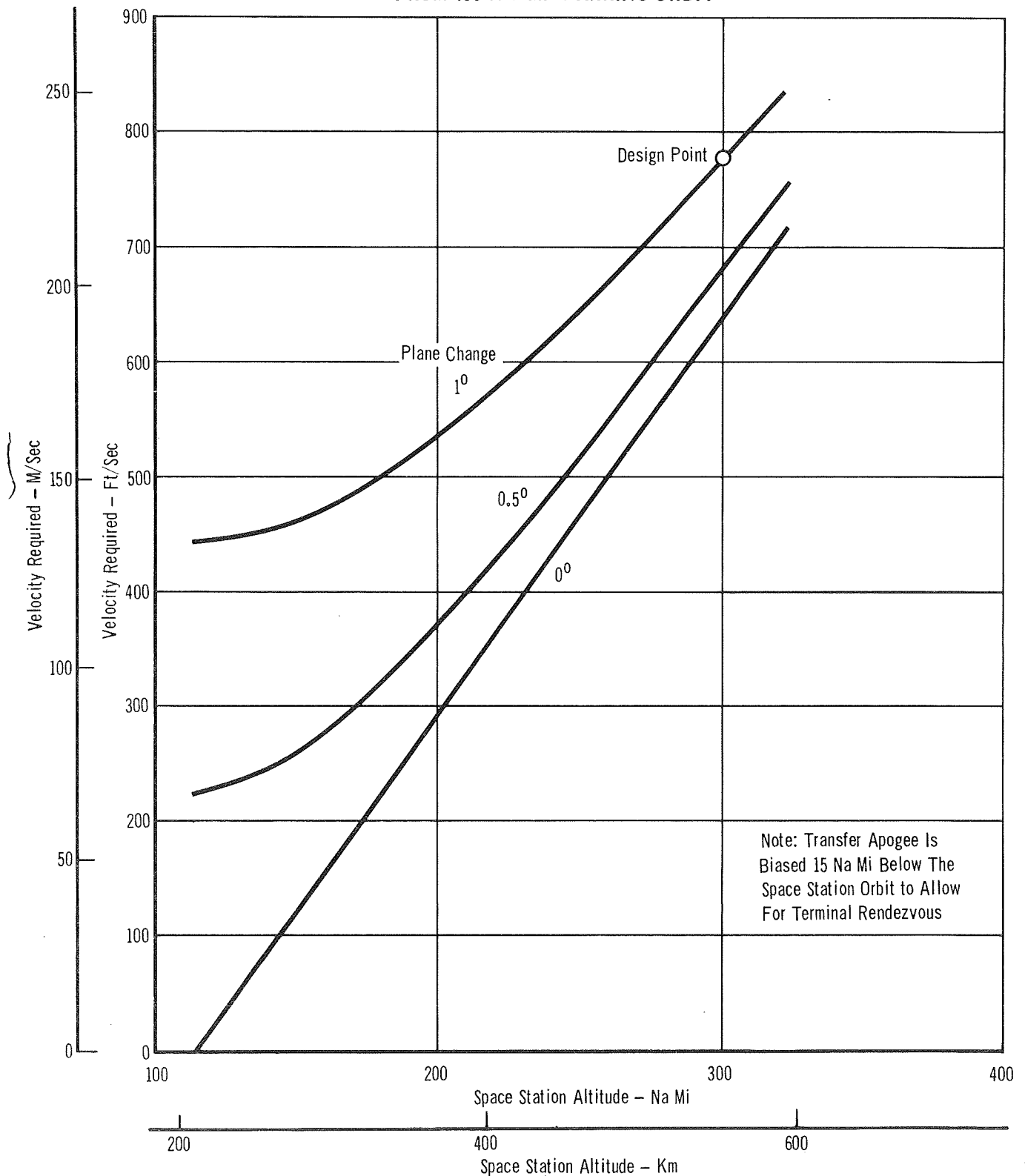
The terminal rendezvous begins after circularization at the biased altitude below and behind the space station and ends with the spacecraft about 100 feet from the station. The Gemini range-range rate terminal rendezvous technique is assumed, thereby requiring a radar, inertial platform, and digital computer. Based on extensive simulation of the rendezvous technique, including dispersions, a ΔV requirement of 200 ft/sec has been established. The time

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Figure 4-2

GROSS RENDEZVOUS MANEUVER REQUIREMENTS FROM 100 NA MI PARKING ORBIT



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required to perform this rendezvous can vary from a minimum of 25 minutes to a maximum of approximately 70 minutes. Docking is visually performed, requiring a ΔV of 40 ft/sec and approximately 30 minutes.

4.4 Orbital Storage - The main requirement for the spacecraft during the orbital storage or orbital standby period is that it be maintained by on-board or space station systems for emergency evacuation and departure from the space station. The quiescent storage period to be considered is nominally one day with a maximum duration of 90 days. Typical activities are summarized in Table 4-4.

TABLE 4-4
MISSION REQUIREMENTS AND CHARACTERISTICS

ORBITAL STORAGE PHASE

Operational Procedures	Mission Requirements and Characteristics	
	Nominal	Variation about Nominal
- o Establish Standby Condition for Critical S/C Subsystems and Provide Monitoring Signals - o Deactivate Non-Critical Spacecraft Subsystems - o Transfer Cargo to Space Station - o Install Return Cargo in Entry Module - o Re-Activate and Check-out Spacecraft Systems - o Prepare to Separate from Space Station	Storage Time Period = 1 Day	1 Day to 90 Days
	All Requirements for Power, Environmental Control, and Food Provided by Space Station	None
	Delivered Cargo of 20,000 lbs	20,000 to 200,000 lbs
	Return Cargo of 200 lbs	200 to 2,000 lbs

Note: No Space Station Orbital Decay Correction Provided

4.5 Deorbit - After the return decision has been made and all preparations and checks have been completed, the return maneuver is initiated by separating from the spacecraft, utilizing a ΔV of about 10 ft/sec. For a normal mission, the approximate return time is known far enough in advance to compute the orbital ground track during entry and the resulting crossrange and phasing requirements. These, in turn, determine the number of waiting orbits and phasing orbits needed to reach a specified landing site. Typical requirements and activities are summarized in Table 4-5.

For the M2-F2 spacecraft, the aerodynamic maneuver capability can provide a 24 hr return capability without the use of a phasing maneuver. In this case, the entry spacecraft will remain docked to the station until about one orbit prior to the daily overpass nearest the selected landing site. After separating and maneuvering clear of the station, the spacecraft is oriented to the retro attitude and, at the proper time, a retrograde impulse is applied. The retrograde ΔV required is a function of the orbit altitude as shown in Figure 4-3. For the nominal 300 na mi orbit, a ΔV of about 500 ft/sec is needed.

For the ballistic spacecraft, the retrograde ΔV required is the same as for the lifting spacecraft. However, since the crossrange requirement will usually exceed the aerodynamic maneuver capability, a phasing maneuver will be required. Separation must be accomplished a sufficient number of orbital revolutions prior to retro-fire for the required orbit track shift.

The retrograde ΔV shown in Figure 4-3 is for a retro thrust vector opposite to the velocity vector. For a ballistic vehicle having limited maneuver capability, it is advantageous to orient the retro thrust vector to minimize range dispersion sensitivity at various retro altitudes as shown in Figure 4-4.

4.5.1 Entry Crossrange Requirements - When a waiting time in orbit of 24 hours is allowed prior to entry, the crossrange maneuverability needed will be no more than half the distance between successive orbit traces. This distance depends on the orbit altitude and inclination, the landing site latitude, the number of landing sites available, and their longitudinal distribution.

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TABLE 4-5
MISSION REQUIREMENTS AND CHARACTERISTICS

DE-ORBIT PHASE

Operational Procedures	Mission Requirements and Characteristics	
	Nominal	Variation About Nominal
Update Orbital Parameters		
Verify Predicted Orbit Ground Tracks		
Select Landing Site	ZI Landing Sites	World Wide Recovery Network
Establish Time to Initiate Entry	Orbital Pass Nearest Landing Site. Up to 24 hours delay.	Within One Orbit
Determine Crossrange Required	400 NM	See Figures 4-5 to 4-7
*Determine Phasing Orbit Requirements	Up to 24 hr Phasing Time	Minimum of 2 hr Return
Select Separation Time		
Separate from Station	$\Delta V = 10$ ft/sec	None
Perform Plane Change Maneuver	Not Required	ΔV for 2 hr return Dependent on number and Latitude of Return sites, Orbit Inclination and Spacecraft L/D (See Figures 4-12 & 4-13).
Inject Into Phasing Orbit	$\Delta V = 0$ ft/sec (Lifting Body) $\Delta V = 280$ ft/sec (Ballistic) $T/W = 0.03$	See Figure 4-11
Verify Retro Time		
Orient Spacecraft at Retro Attitude	-8.0 deg. $\Delta V = 625$ ft/sec (solid) $\Delta V = 500$ ft/sec (liquid) $T/W = 0.1$	None
Fire Retro Rockets		None

* Ballistic Spacecraft Only

Figure 4-3

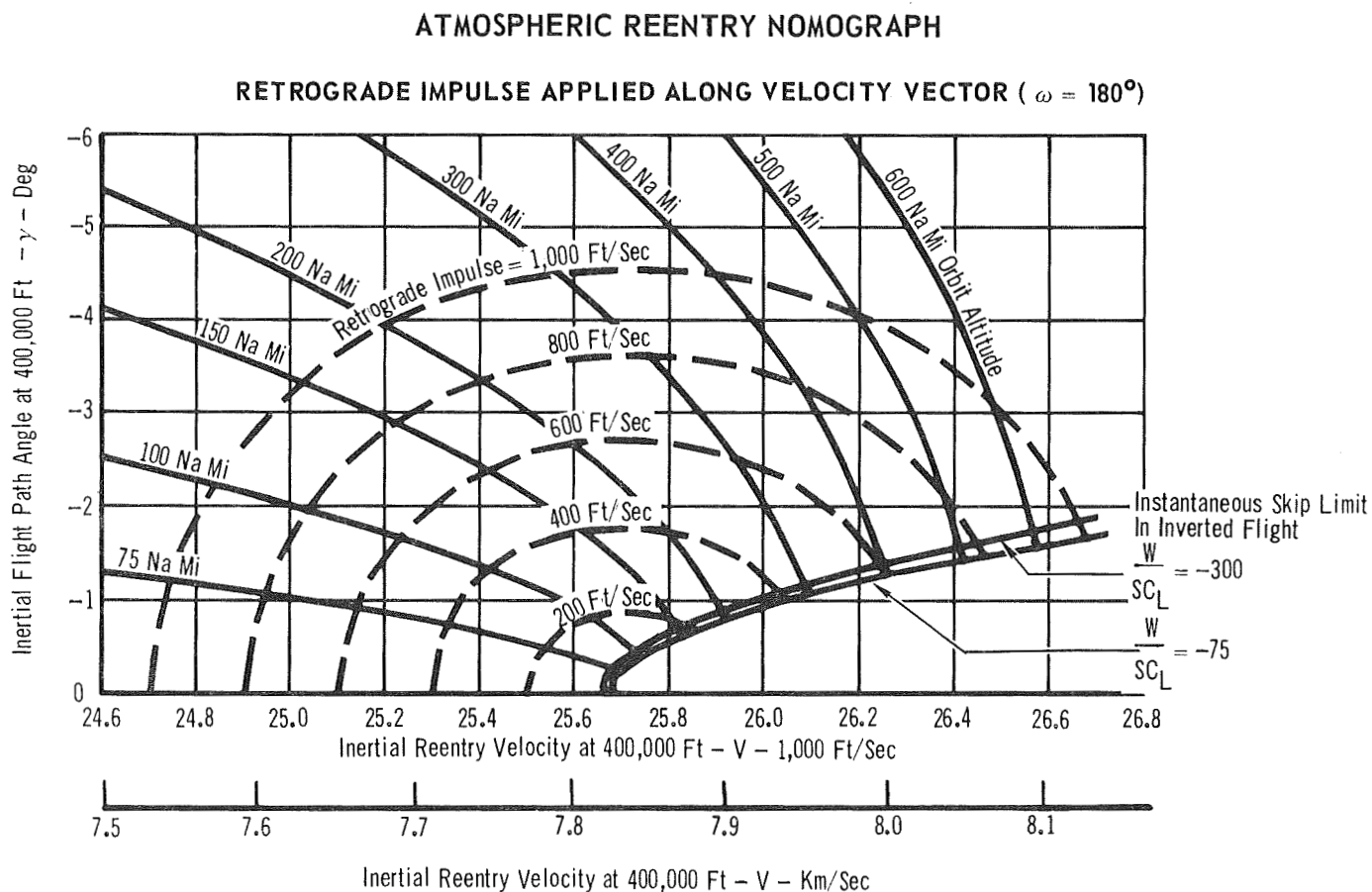
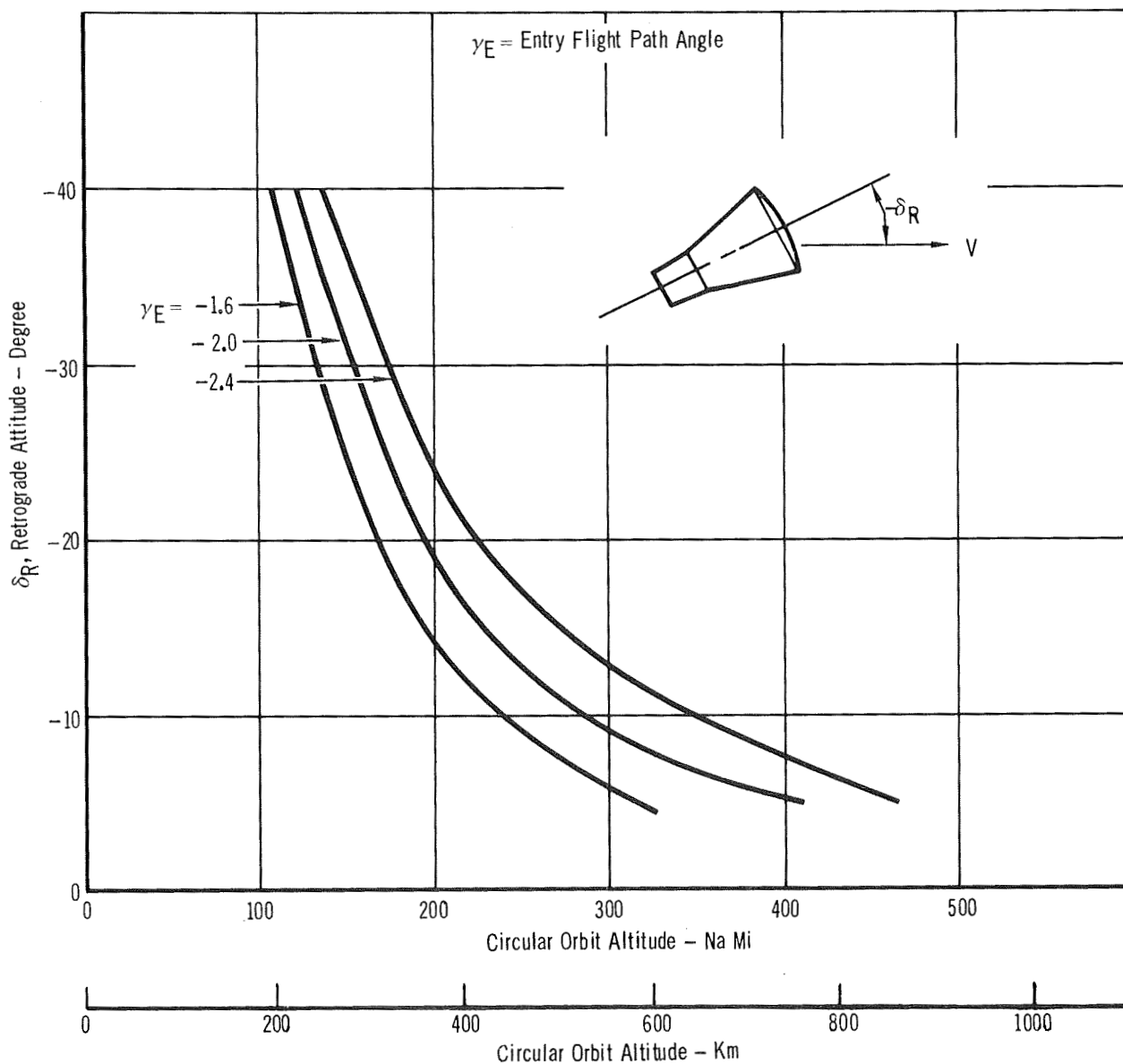


Figure 4-4

RETROGRADE ATTITUDE REQUIREMENT
FROM CIRCULAR ORBIT
FOR MINIMUM RANGE SENSITIVITY [$\partial R / \partial \delta_R = 0$]



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For emergency returns with less than 24 hours delay, the crossrange maneuver requirements increase rapidly with decreasing delay time. In Figure 4-5 the crossrange required is shown as a function of the number of available landing sites for a 24 hour return, (16 orbit delay), an immediate return, and a 3 orbit delay time. Results are shown for an orbit inclination of 50 degrees, and landing site latitudes of 30 and 50 degrees, assuming an optimum longitude distribution of landing sites. These plots illustrate the large variations in cross range requirements with the allowable delay time, number of landing sites, and landing site latitude.

Similar plots showing crossrange requirements for orbit inclinations of 70 and 90 degrees are shown in Figures 4-6 and 4-7.

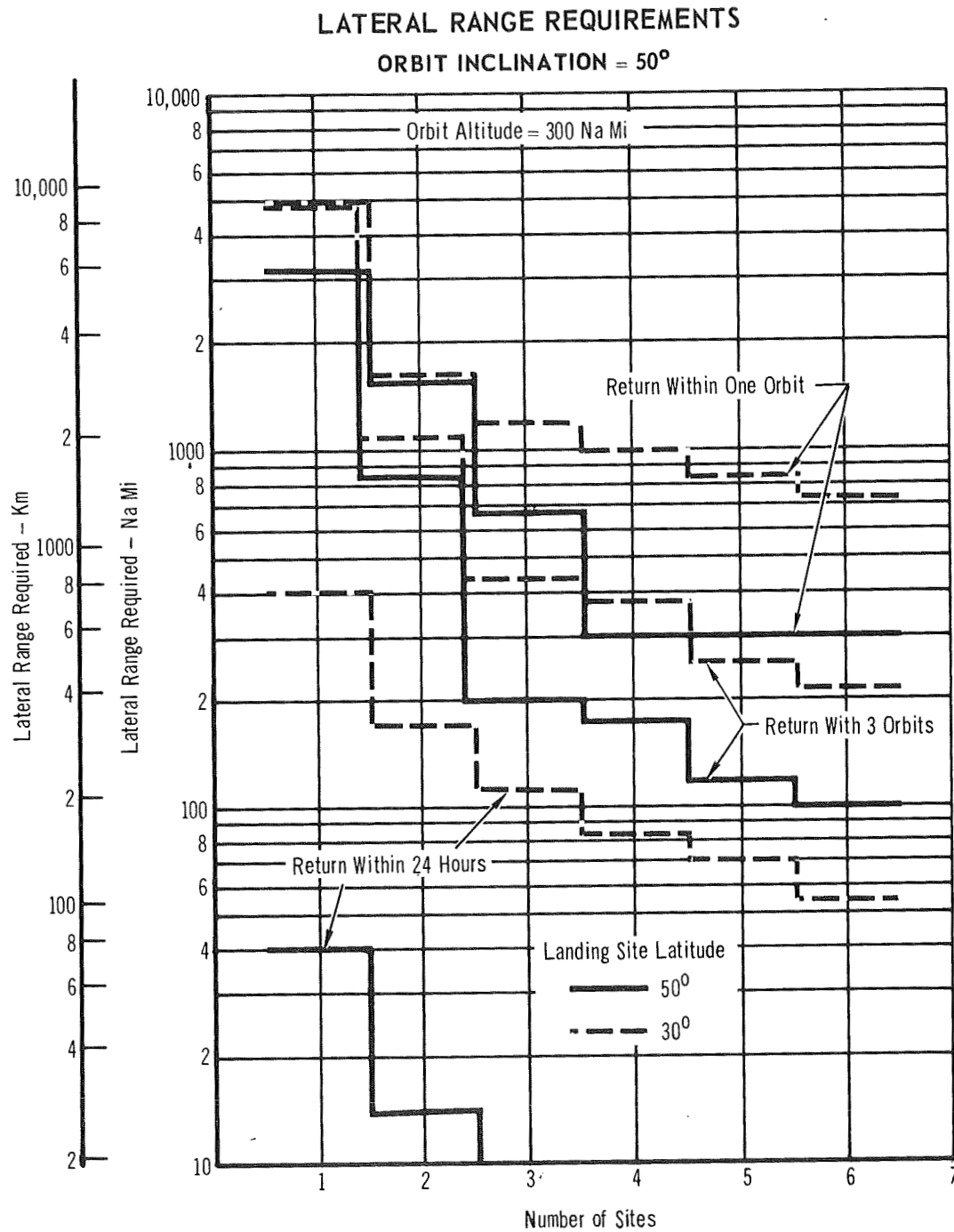
For the nominal requirement of a 24 hour return time and landing sites restricted to the continental U.S., the crossrange requirements are a function of the number and location of the selected landing sites. This variation is illustrated in Figure 4-8. This plot shows the crossrange requirements for return to a single site at either 30 deg or 50 deg latitude, or to various 3-site networks, for a range of orbit inclinations. Although the crossrange maneuver requirements can be minimized by choosing a return site latitude equal to the orbit inclination, this is not a feasible design condition since a small change in inclination would greatly increase the crossrange. For the range of orbit inclinations being considered (50 deg to 90 deg), a three-site network appears more realistic. For these, the maneuver requirements vary from about 180 na mi for sites along the northern U.S. boundaries to about 260 na mi for a three-site network in the capabilities of the Ballistic and M-2 type vehicles are also indicated. This shows that the M-2 can readily meet a 24 hr return requirement to any ZI site.

The ballistic vehicle, however, has only marginal aerodynamic maneuver capability. For this vehicle, 24 hr return to the three-site network of Edwards, Holloman, and Wendover AFB, can be assured by performing an orbit phasing maneuver as discussed below. For return to a single ZI site, the worst case occurs for a polar orbit and a 30 deg or lower landing site latitude. For this case, a 550 na mi crossrange is needed. Again this is within the aerodynamic maneuver capabilities of the M2-F2 vehicle but well beyond the range of a ballistic type spacecraft.

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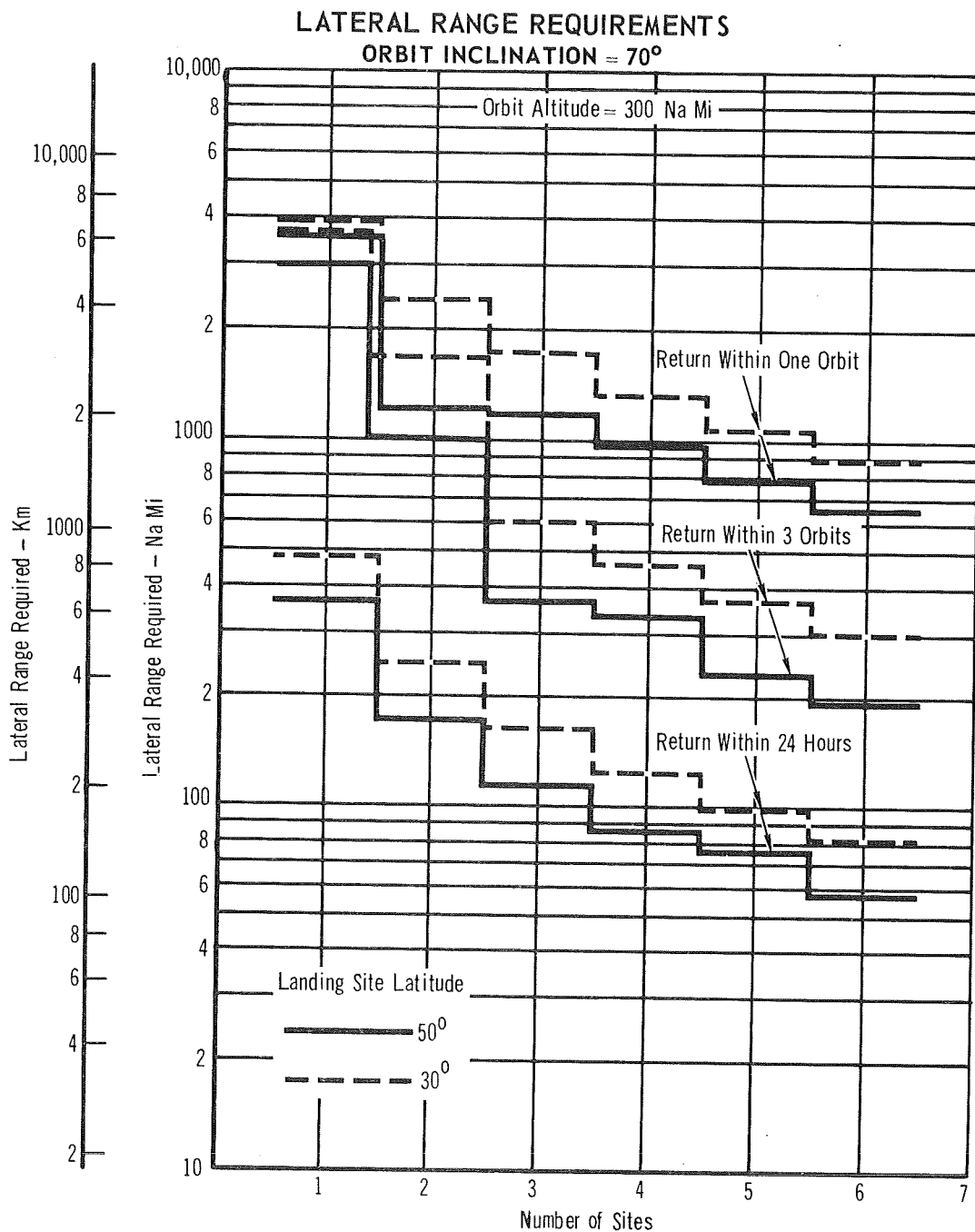
Figure 4-5



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Figure 4-6

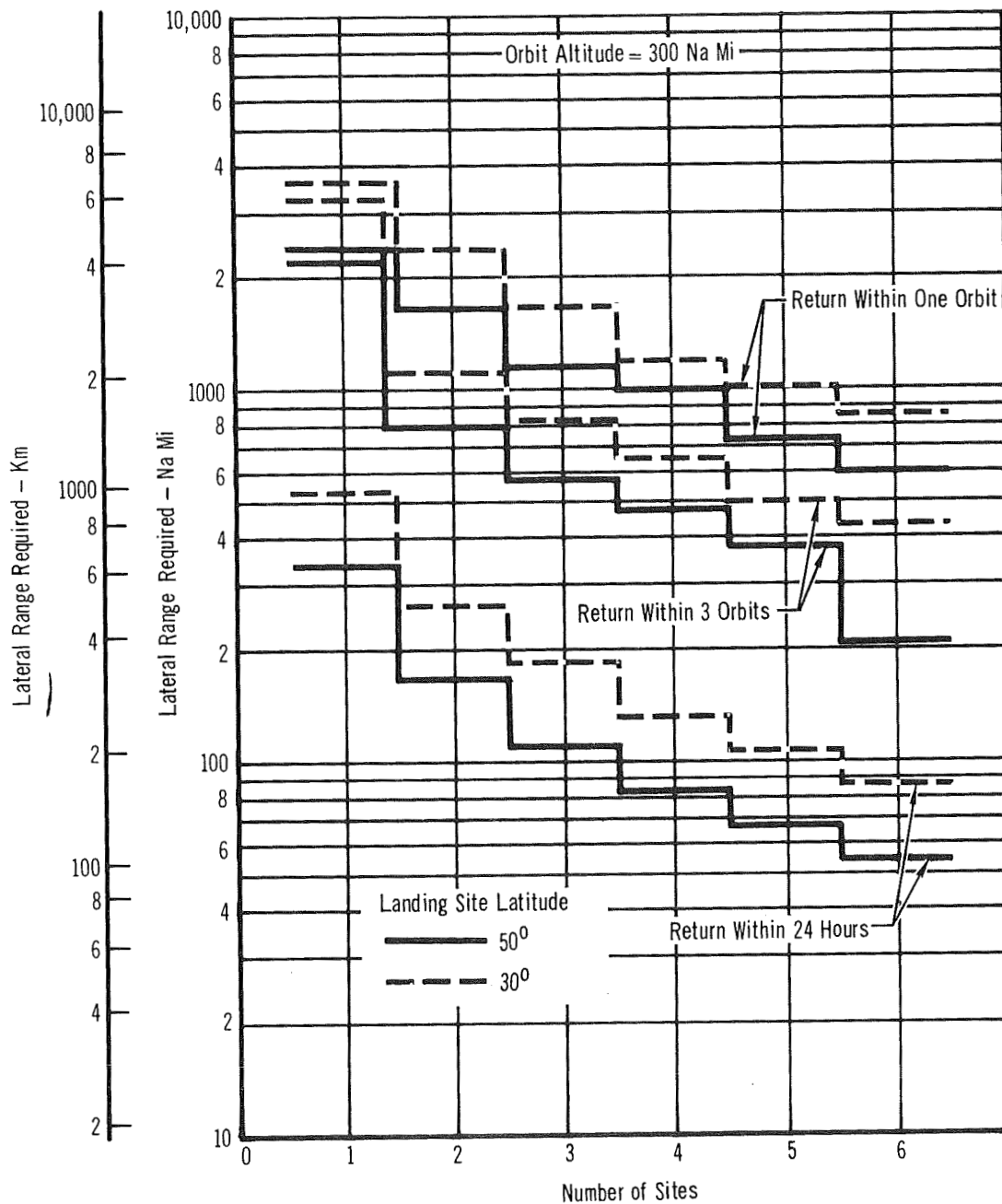


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Figure 4-7

LATERAL RANGE REQUIREMENTS ORBIT INCLINATION = 90°



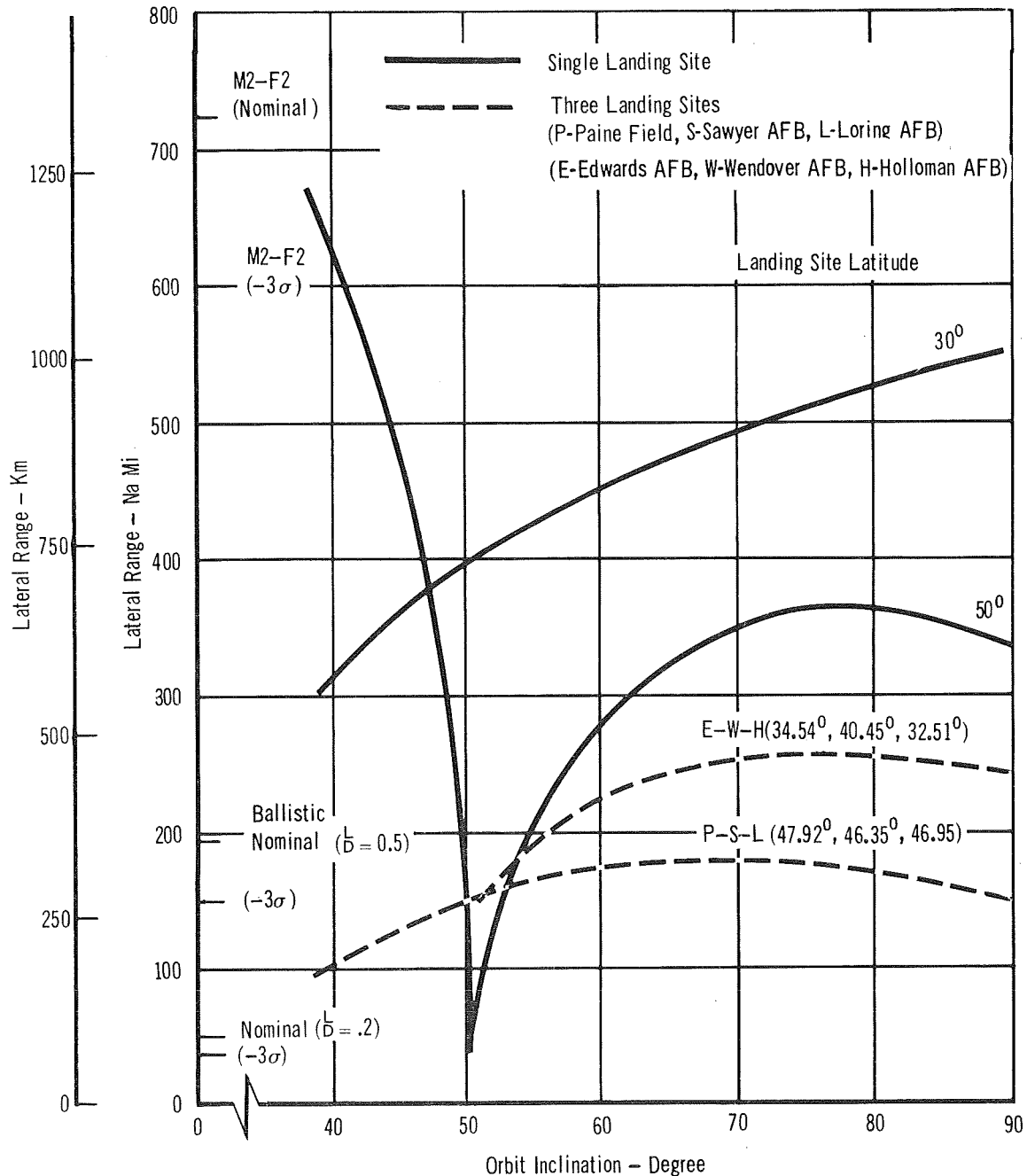
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Figure 4-8

CROSS RANGE REQUIRED FOR 24 HOUR RETURN

ORBIT ALTITUDE = 300 NA MI



Additional discussion of the recovery requirements is included in Volume II, Book 2 as part of the landing site selection in the analysis of operations and support.

4.5.2 Phasing Maneuver Requirements - When the entry crossrange required exceeds the ballistic vehicle ranging capability, a propulsive maneuver is required to reach the landing site. A plane change maneuver can supply the range required within one orbital revolution, but requires a large ΔV expenditure. If sufficient time is available (up to 24 hours), a phasing maneuver is more efficient. In performing this maneuver, the spacecraft is transferred into an elliptical orbit by either raising the apogee or lowering the perigee. This changes the period and shifts the orbital ground tracks relative to those of the circular orbit. A westerly shift is accomplished by raising the apogee and an easterly shift is accomplished by lowering the perigee. Since a perigee of 100 na mi is about the lowest feasible limit, the easterly shift attainable is not as great as the westerly.

The ΔV required to shift the orbit track a given number of degrees easterly or westerly is shown in Figure 4-9 and indicates the decreasing ΔV requirement as the allowable number of phasing orbits increases. The corresponding apogee altitude is also indicated. The data shown indicates that for short allowable phasing times, the maneuver is performed primarily by raising the apogee, since the 100 na mi perigee limit restricts the easterly shift available. Because of this limit, the most adverse landing site location is not necessarily midway between successive ground tracks.

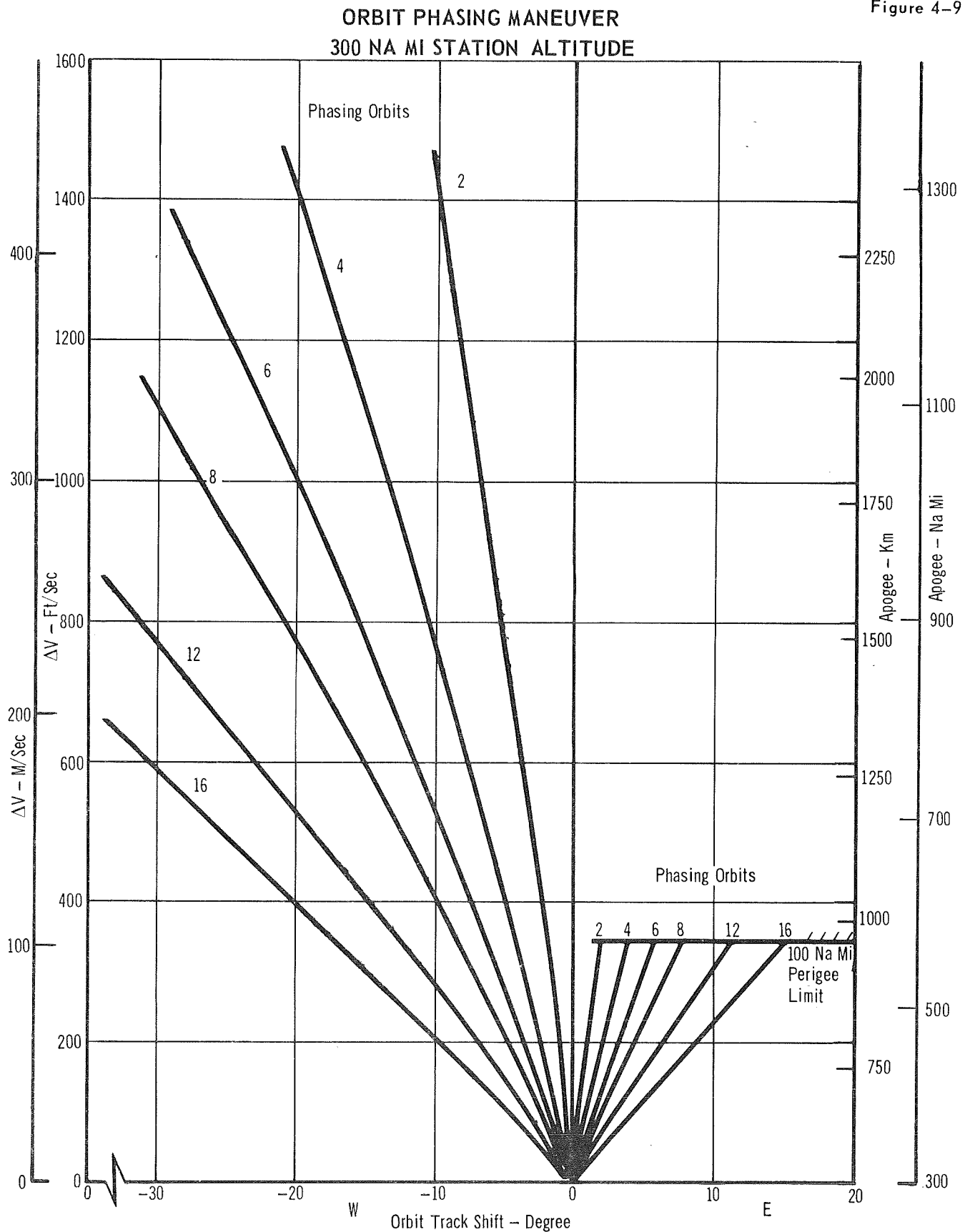
This is more clearly illustrated in Figure 4-10. For a 300 na mi orbit, the distance between successive ground tracks is 24 degrees of longitude. If the landing site were midway between them and 16 phasing orbits were allowed, the site could be reached by shifting either 12 degrees easterly or 12 degrees westerly. (ΔV required would be slightly lower for a westerly shift, however.)

As shown by the plot, the most adverse landing site (requiring the largest phasing ΔV) would be about 1/2 degree west of this point. However, if only eight phasing orbits are allowed, the easterly shift is limited to 7.7 degrees due to the 100 na mi limit on perigee altitude. Landing sites east of this point would require a westerly orbit track shift. In this case, the most adverse site would be just east of the easterly shift capability, thus

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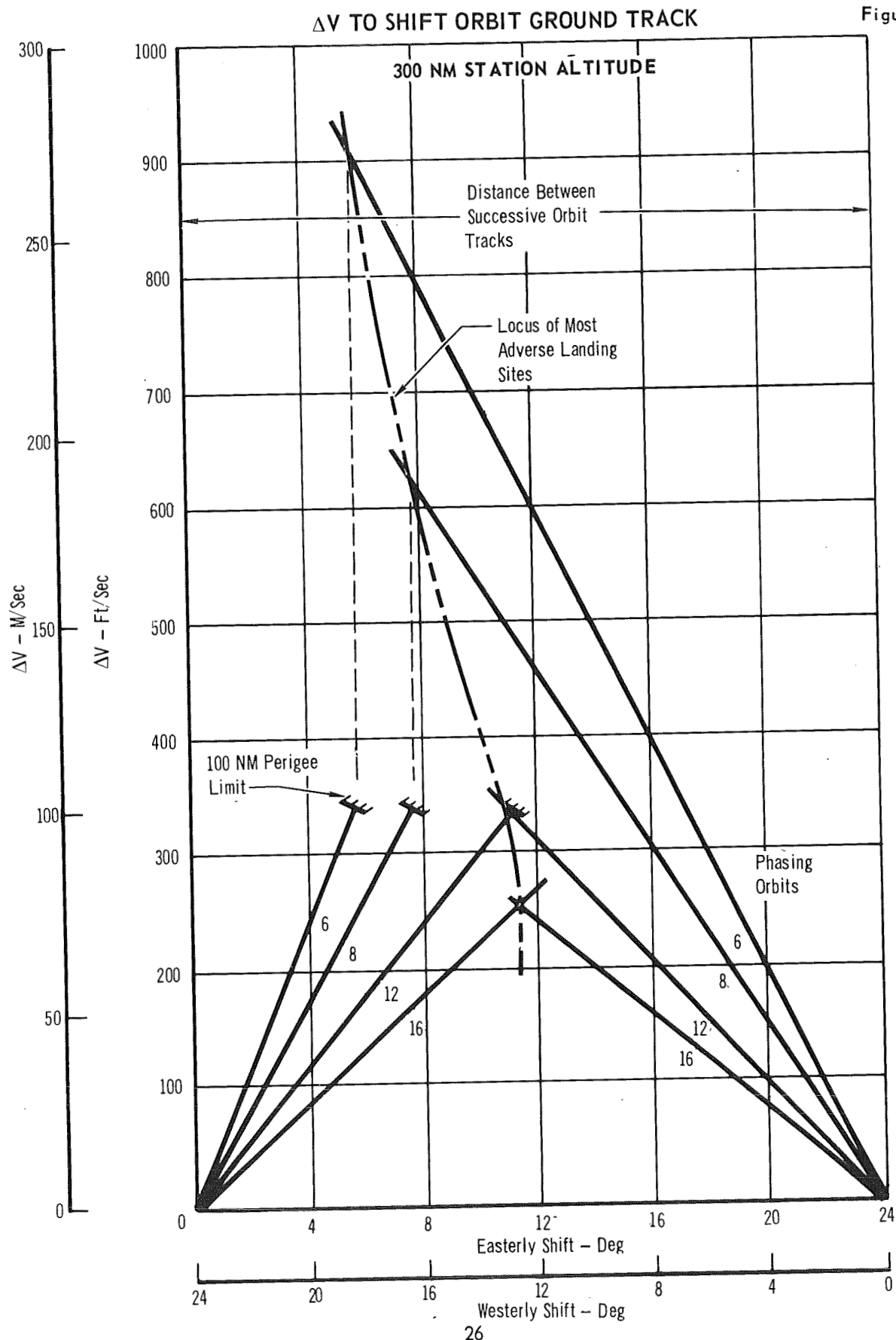
Figure 4-9



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Figure 4-10



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requiring a westerly shift of about 16.2 degrees. The variation of the more adverse landing site location with the number of allowable phasing orbits is indicated. As the allowable phasing time increases, the most adverse landing site location moves toward the center of successive ground tracks, and the ΔV required to raise the apogee approaches that required to lower perigee.

The maximum ΔV required to reach the most adverse landing site location from an initial 300 na mi orbit is shown in Figure 4-11 as a function of phasing time. For 24 hour return, the maximum ΔV required is seen to be 280 ft/sec and the maximum apogee during phasing is 480 na mi. For the three-site recovery network, this requirement could be further reduced.

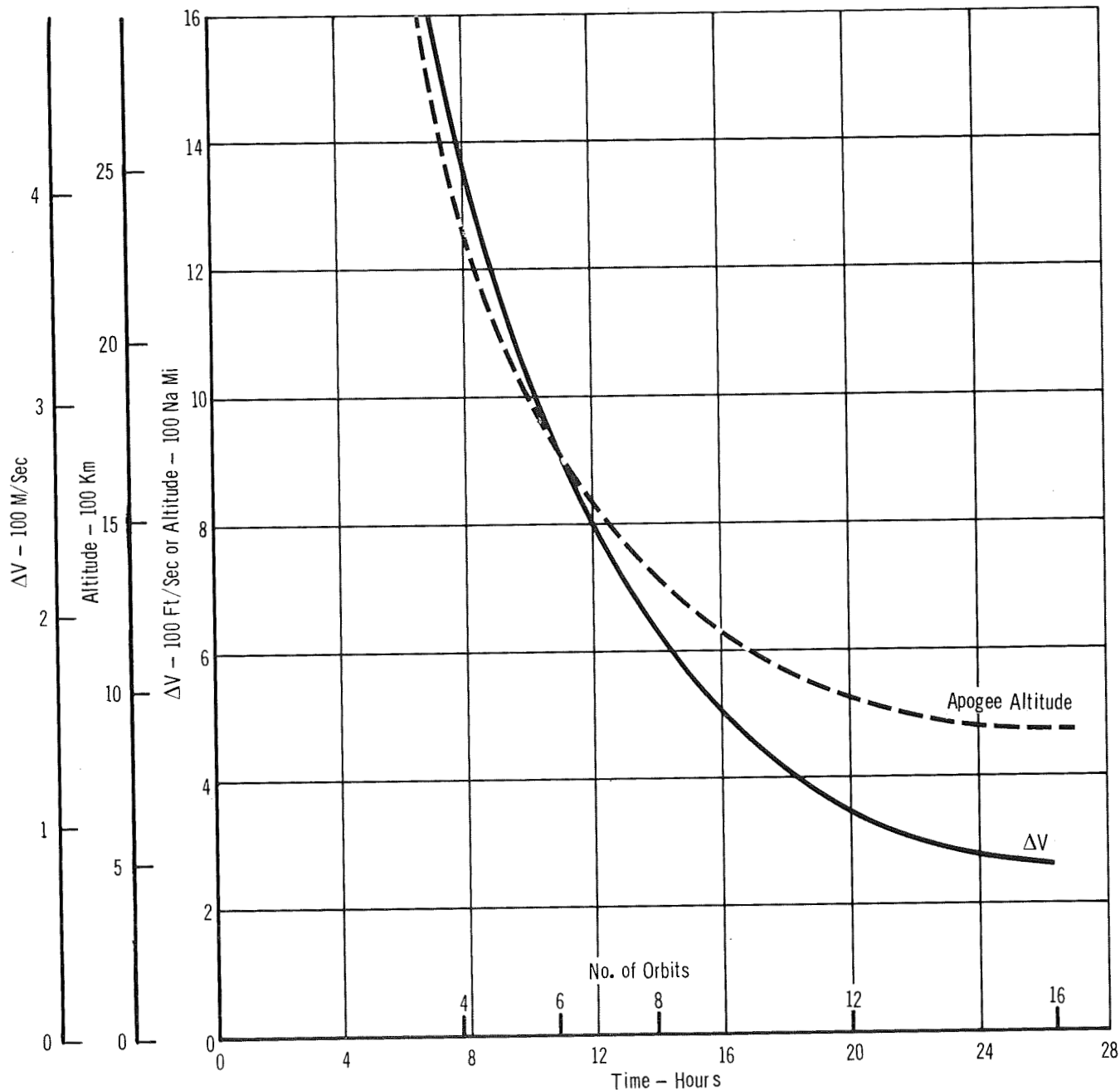
4.5.3 Emergency Return - For a normal logistics mission, an allowable return time of 24 hours is a realistic requirement that provides return capability to a single ZI landing site with relatively small ΔV expenditures. Emergency conditions, however, may necessitate entry with a much shorter waiting time. As shown in Figures 4-5 to 4-7, the crossrange requirements increase rapidly as the number of allowable waiting orbits decrease. Also, as shown above, the orbit phasing maneuver becomes more severe as the waiting time decreases and becomes unfeasible for immediate return. For quick returns, the crossrange is obtained by performing an orbital plane change. This requires large energy expenditures since each degree of plane change provides only 60 na mi of crossrange and requires about 450 ft/sec of ΔV .

For emergency return the trade-off then is between increasing ΔV requirements and increased number of landing sites. This is shown in Figures 4-12 and 4-13. Figure 4-12 shows this relationship for the ballistic vehicle as a function of both the orbit inclination and landing site latitude. Landing sites were assumed to be optimally distributed in longitude. In deriving this plot, a 30 aerodynamic maneuver capability of 150 na mi was assumed for the ballistic spacecraft and the ΔV requirements shown are for crossrange distances in excess of this capability. For the nominal fifty degree orbit inclination and four landing sites at fifty degree latitude, over 1100 ft/sec ΔV is needed for immediate return. This requirement increases rapidly for lower latitude sites or for higher inclination orbits.

Figure 4-11

ORBIT PHASING MANEUVER ΔV REQUIRED

300 NA MI STATION ALTITUDE



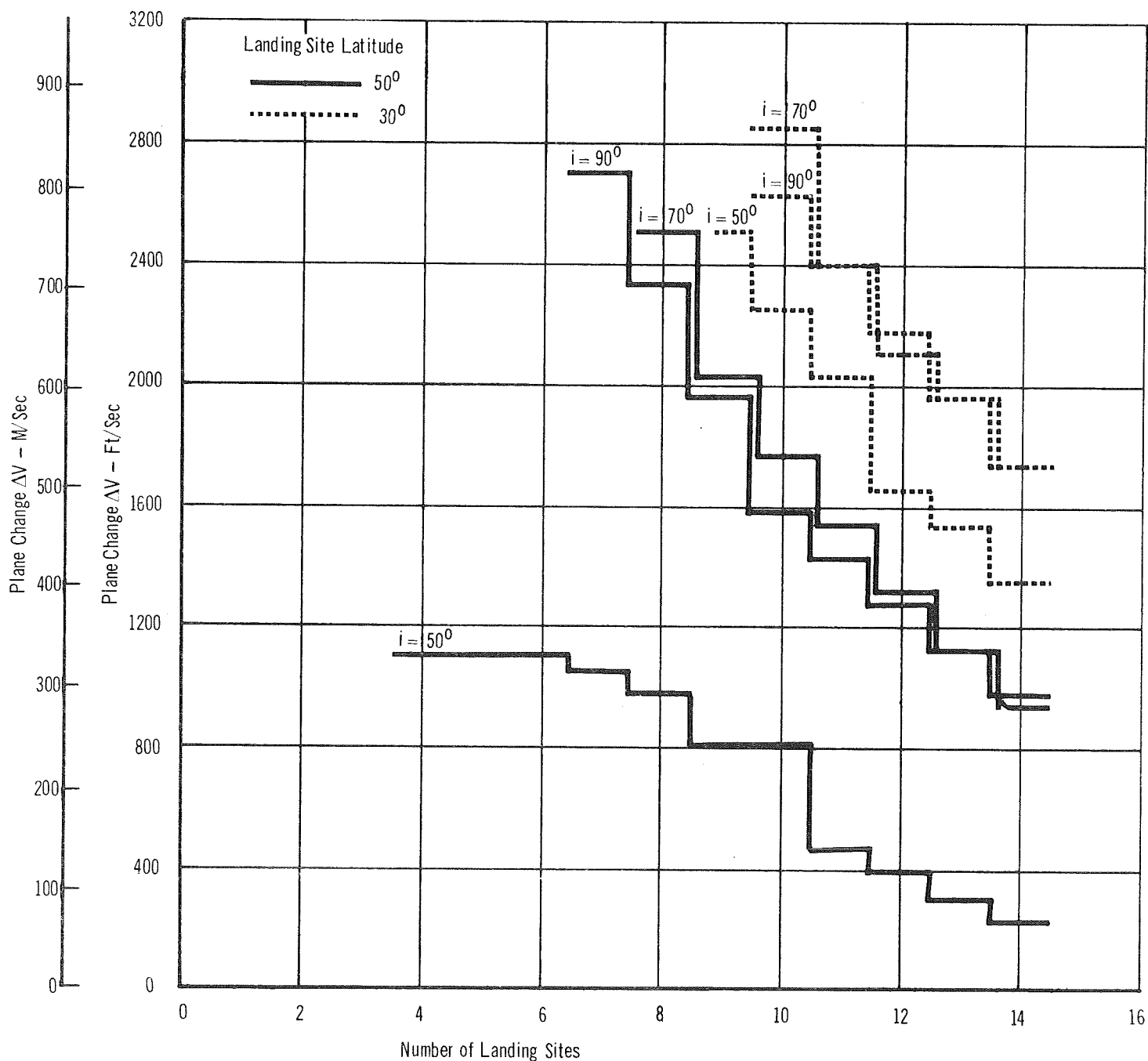
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**PLANE CHANGE ΔV REQUIREMENT
FOR 2 HOUR RETURN CAPABILITY**

Figure 4-12

Ballistic Type Spacecraft
-3 σ Lateral Range Capability = 150 Na Mi



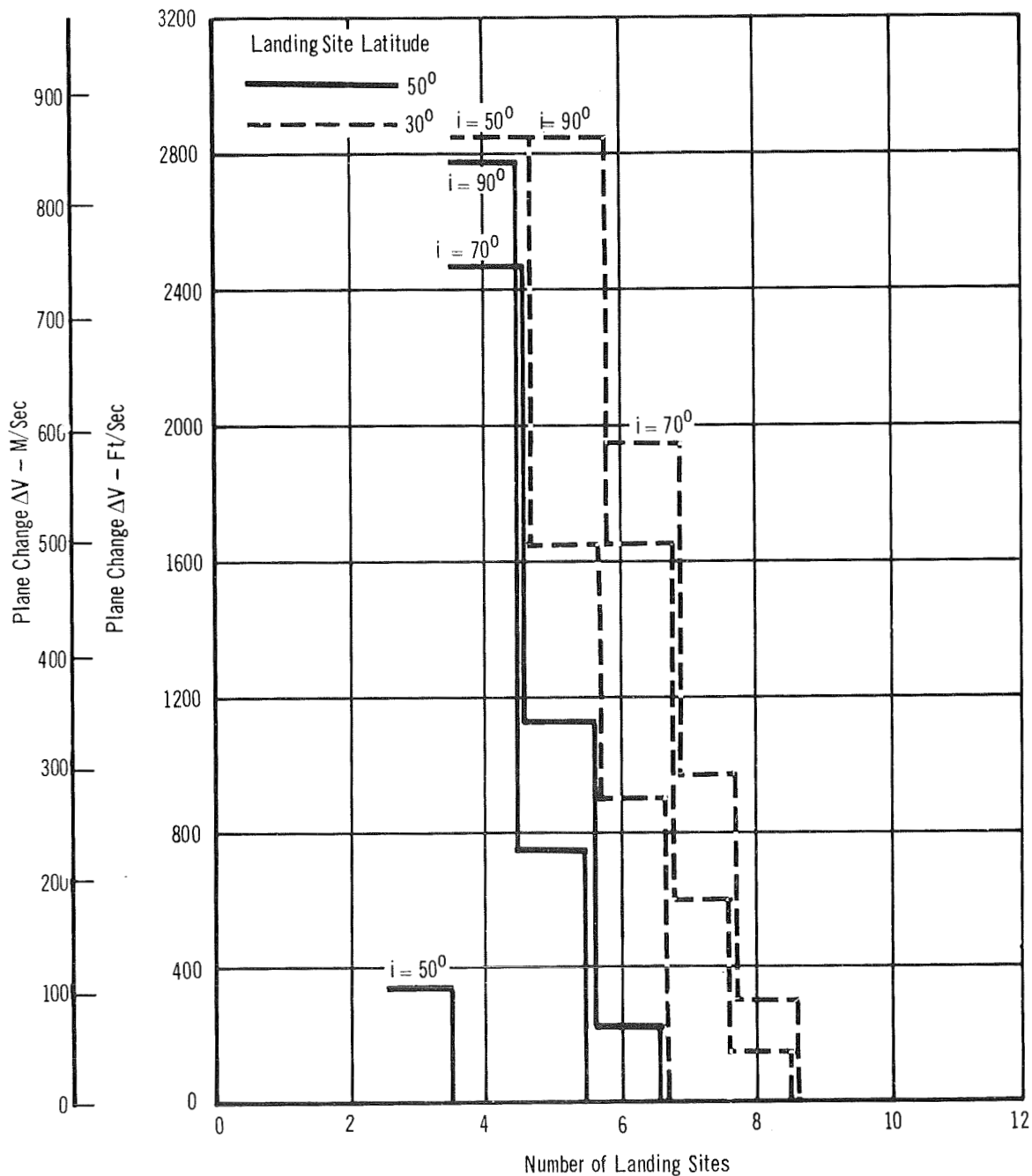
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Figure 4-13

**PLANE CHANGE ΔV REQUIREMENT
FOR 2 HOUR RETURN CAPABILITY**

M2-F2 Type Spacecraft
-3 σ Lateral Range Capability = 600 Na Mi



Similar data is shown in Figure 4-13 for the M2-F2 configuration. For this spacecraft, a -3σ aerodynamic maneuver capability of 600 na mi was assumed. The ΔV needed was for crossrange requirements in excess of this capability. Nominal and -3σ crossrange capabilities as a function of the hypersonic lift to drag ratio (L/D) are shown in Figure 4-14.

The return ΔV requirements as a function of the propulsively provided crossrange are summarized in Figure 4-15. It was assumed that a phasing maneuver is used for returns after three or more waiting orbits and a propulsive plane change for immediate returns. Lateral range shown is only that in excess of the aerodynamic maneuver capability.

4.6 Re-Entry - After completing the retrograde maneuver, the spacecraft is separated from the propulsion module and oriented to the entry attitude. Operational procedures during this phase are shown in Table 4-6.

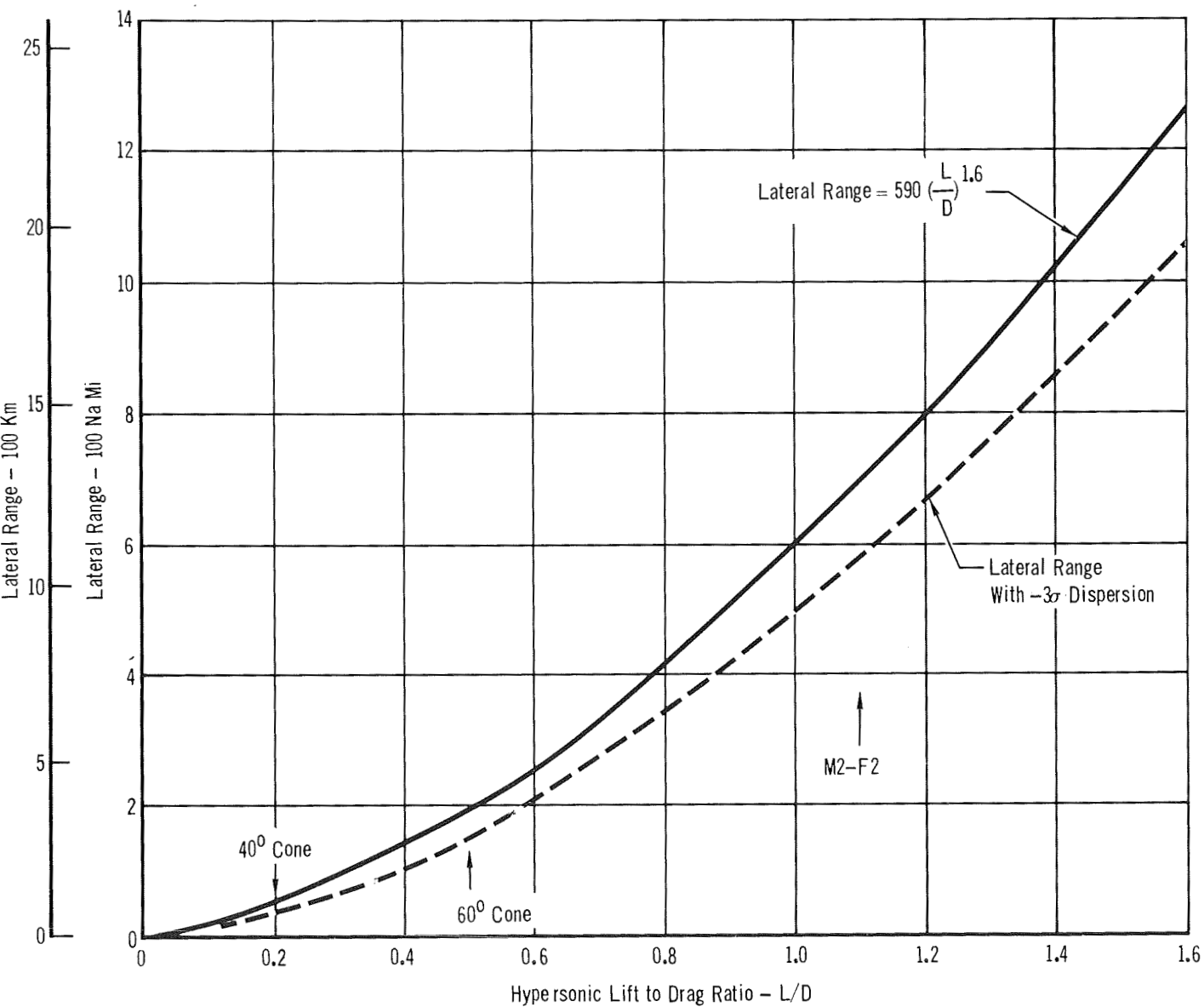
To define the design entry environment, entry trajectories have been computed for each type vehicle. Entry time histories were calculated using the point mass option of the Generalized Six-Degree-of-Freedom computer program and were run for a range of parameters bracketing the nominal entry conditions. These are summarized as follows:

Type Vehicle	Lift Conditions	Entry Angle	Bank Angle
M2-F2	Max L/D	shallow (-1.0 deg)	0
	"	steep (-2.5 deg)	45
	Max C_L	shallow (-1.0 deg)	0
	"	steep (-2.5 deg)	45
Ballistic	L/D = .5	shallow (-1.0 deg)	0
	"	steep (-2.5 deg)	0
	"	steep (-2.5 deg)	45
	L/D = .2	shallow (-1.0 deg)	0
	"	steep (-2.5)	45
	L/D = 0	shallow (-1.0 deg)	0
	"	steep (-2.5 deg)	0

The velocity-altitude histories are summarized in Figures 4-16 through 4-18 and indicate the trajectory boundaries for the given vehicle. Entries from any other conditions will be within the flight path boundaries shown, except for high altitude abort entries as discussed below. Trajectories

Figure 4-14

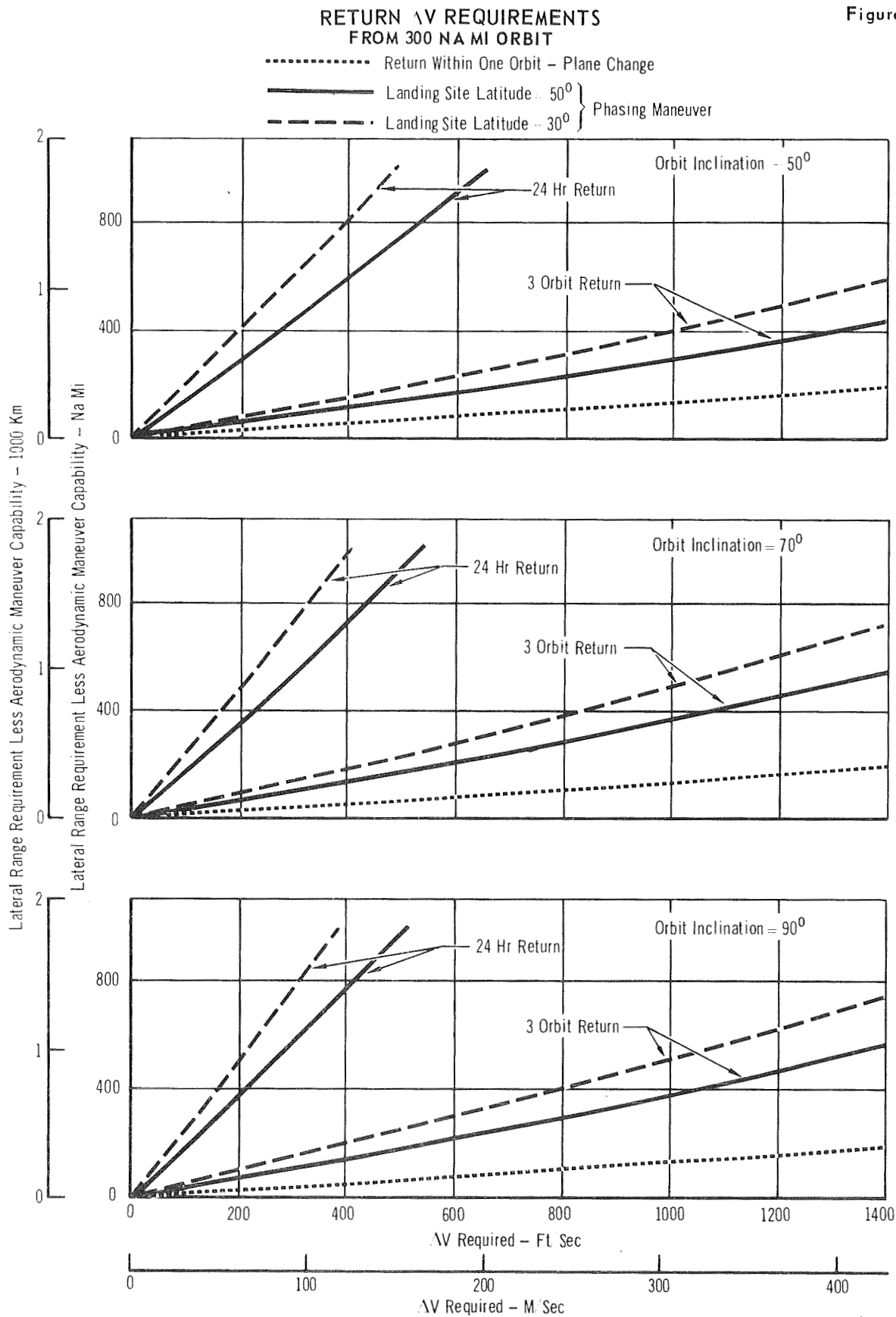
AERODYNAMIC LATERAL RANGE AS A FUNCTION OF L/D



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Figure 4-15



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TABLE 4-6
MISSION REQUIREMENTS AND CHARACTERISTICS
RE-ENTRY PHASE

Operational Procedures	Mission Requirements and Characteristics	
	Nominal	Variation about Nominal
* Orient Spacecraft for Propulsion Module Separation	Time for Re-entry See Figures 4-16 to 4-18	
* Separate from Propulsion Module	Approx 2 ft/sec	
Orient to Entry Attitude	Ballistic = 20 deg M2-F2 = 11 deg	None
± Perform Pullout Maneuver	Modulate Bank Angle	None
Perform Aerodynamic Ranging Maneuvers	Ballistic - Bank Control M2-F2 - Bank and Pitch Control	None
Attitude Control During Entry	Angular Acceleration (Min) Ballistic S/C = 5.0 deg/sec/sec M2-F2 = 2.0 deg/sec/sec Total Impulse to S/C Weight Ratio = 1.4 sec	None
± Terminal Approach Maneuvers to High Key	Selected Runway Acquisition	None

* For modular vehicle only

± Lifting Body Spacecraft only

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ENTRY TRAJECTORY BOUNDARIES M2-F2 CONFIGURATION

$W/S = 72 \text{ Lb/Ft}^2$

Figure 4-16

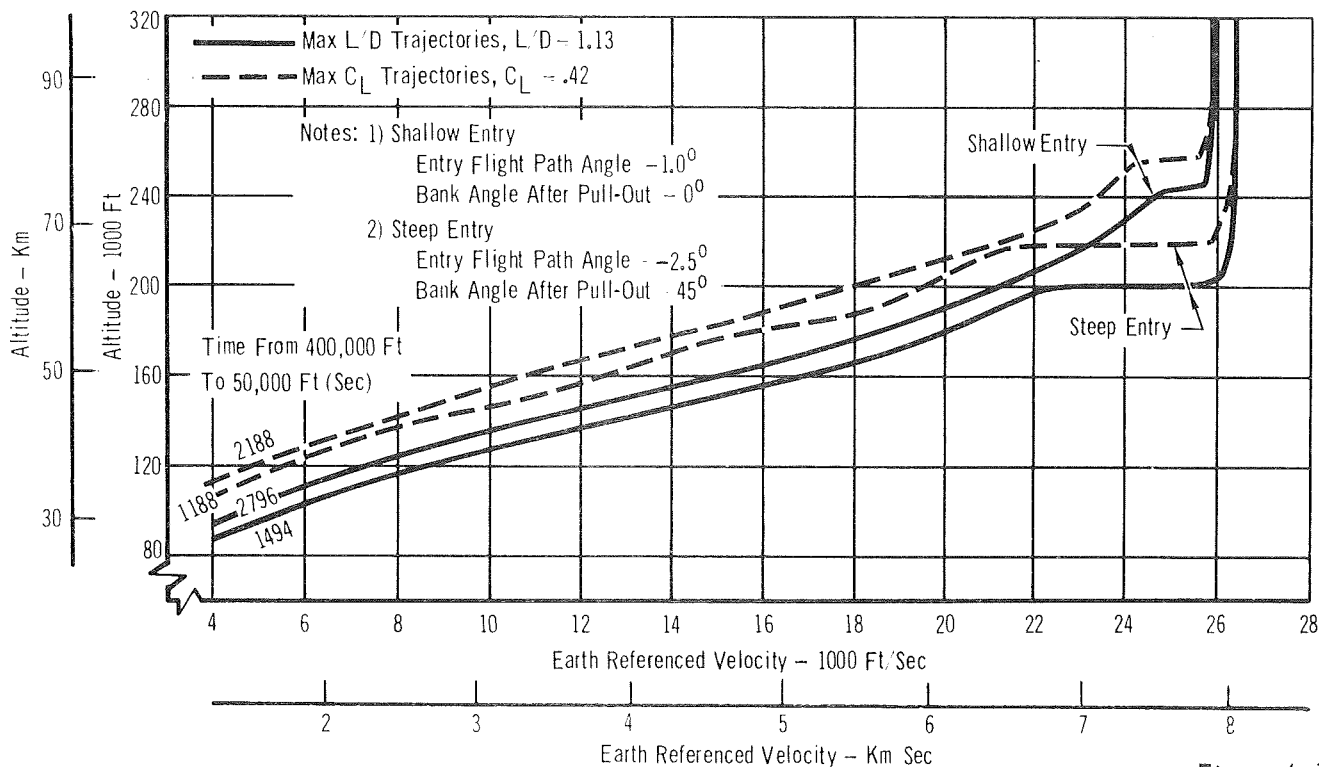


Figure 4-17

ENTRY TRAJECTORY BOUNDARIES - BALLISTIC CONFIGURATION

INCLINATION - 90°

$L/D = .5$

$W/S = 63.5 \text{ LB FT}^2$

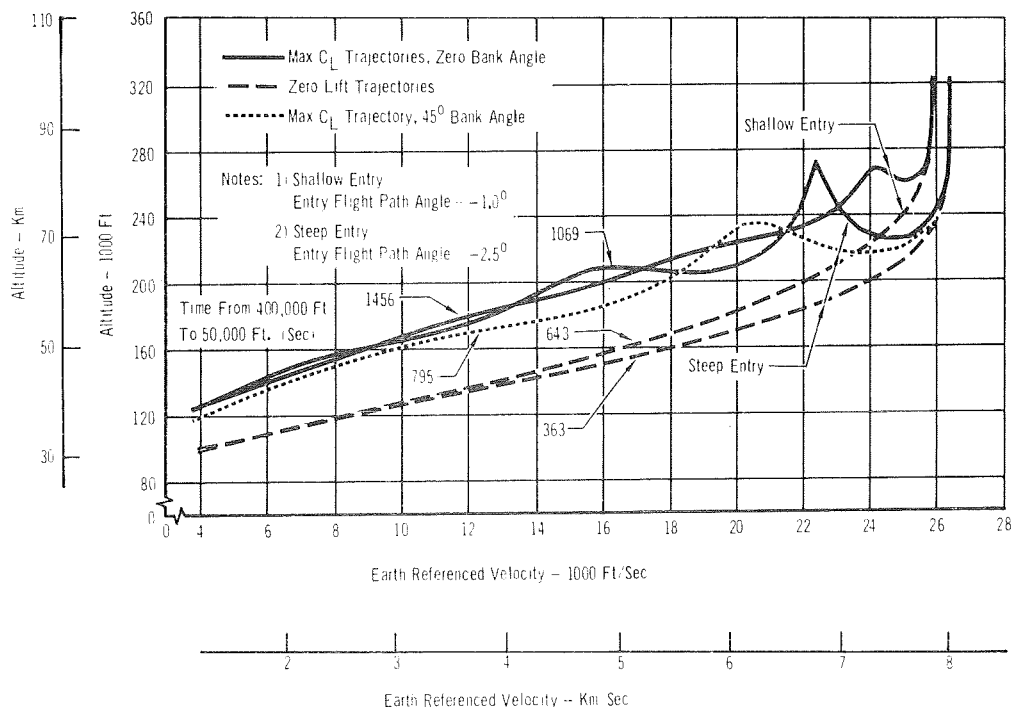
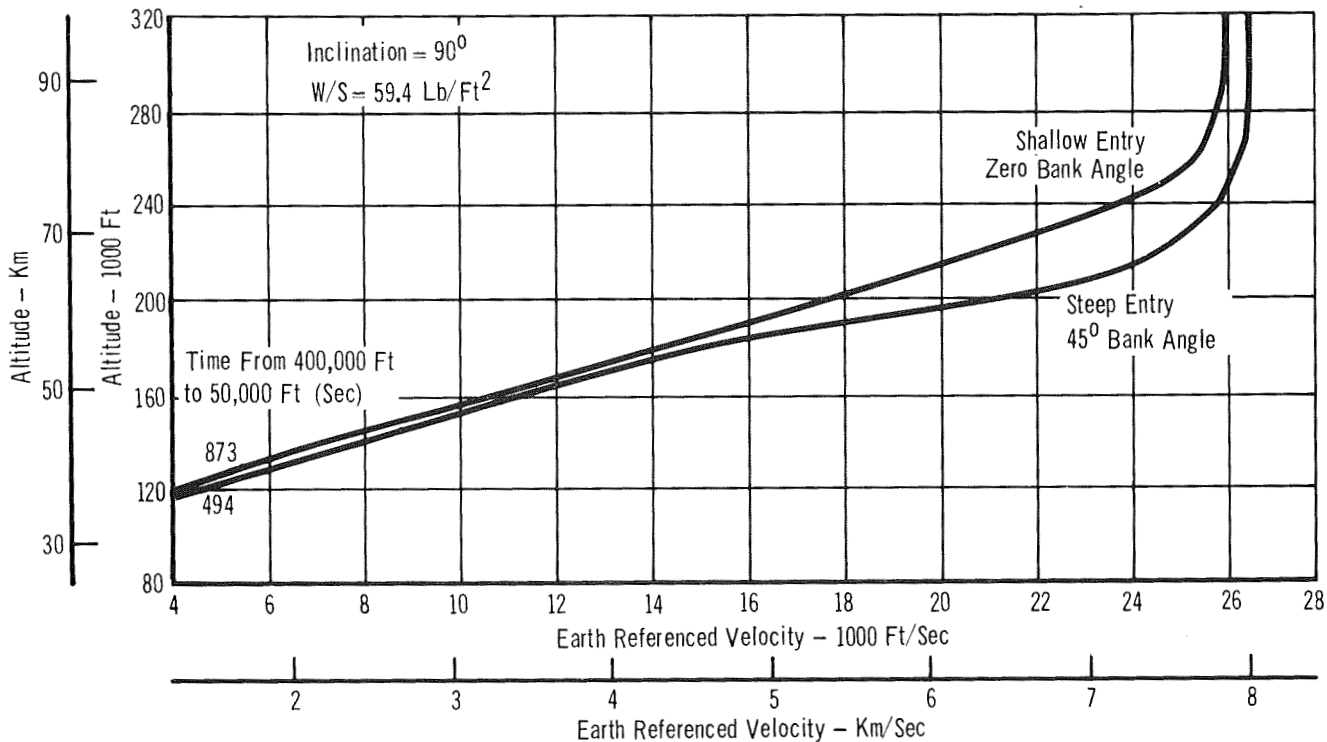


Figure 4-18

ENTRY TRAJECTORY BOUNDARIES BALLISTIC CONFIGURATION

$L/D = .2$



shown represent entries from a 300 na mi altitude polar orbit. Entries from lower inclination orbits will have less severe initial entry conditions.

Trajectories for the M2-F2 are shown in Figure 4-16. For these entries, bank modulation immediately after completion of the entry pullout was used to prevent skip. As the spacecraft decelerates, bank angle is gradually decreased until reaching the value desired for the crossrange maneuver (45 degrees for maximum crossrange). This roll modulation produces the constant altitude phase shown immediately after pullout.

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Trajectories are shown in Figure 4-17 for an $L/D = 0.5$ ballistic vehicle. For these spacecraft roll modulation at pullout is not used. This results in a skipping motion for the maximum L/D conditions. Trajectories are also shown for a zero lift entry, attained by performing a steady roll to cancel out the resultant lift vector. Both the maximum lift and zero lift entries are shown for both steep and shallow initial entry angles.

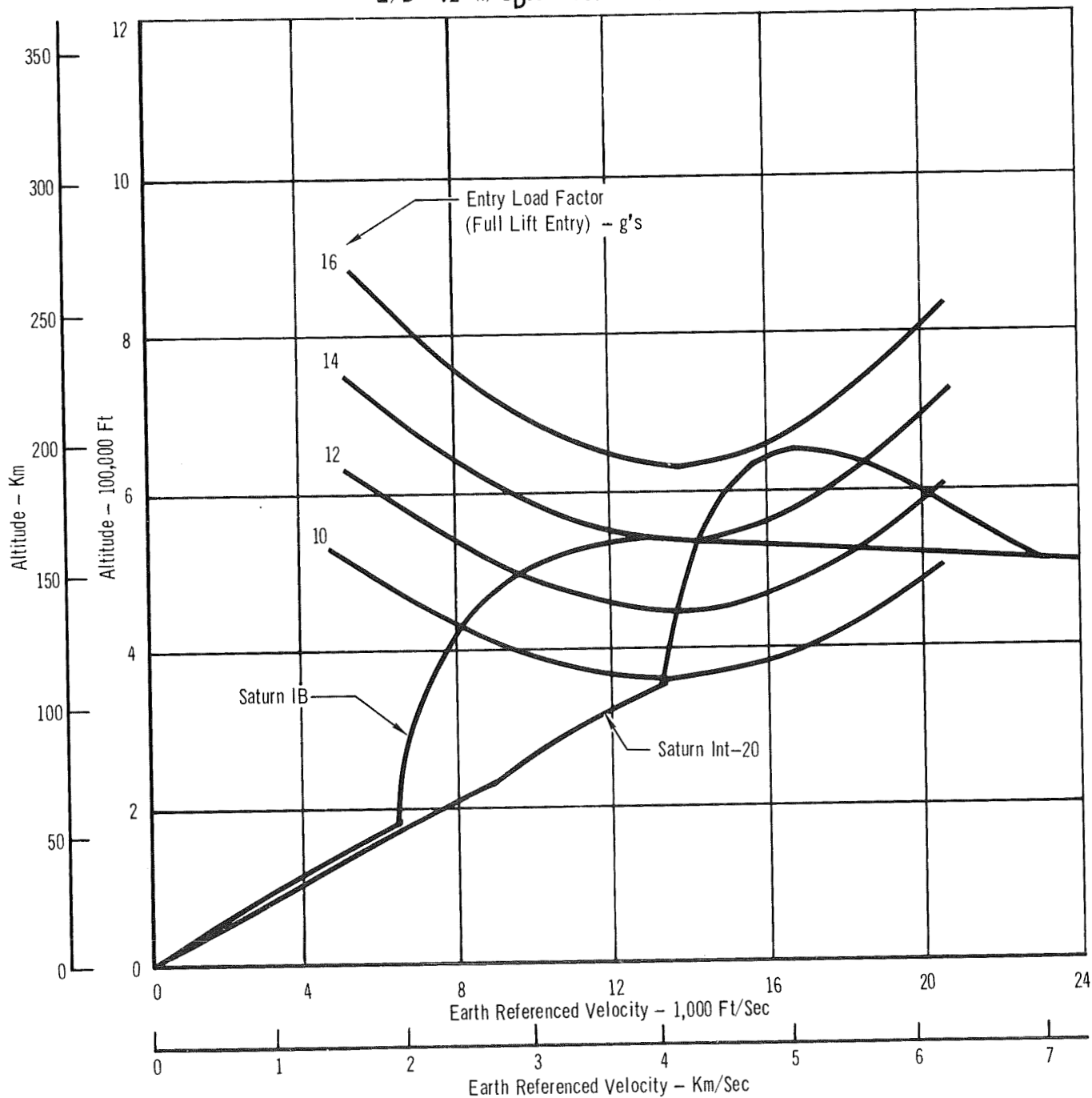
The entries of a ballistic spacecraft having an $L/D = 0.2$ are shown in Figure 4-18. Trajectories are shown only for the maximum L/D condition. Zero lift entries are similar to those for the $L/D = .5$ ballistic spacecraft.

Abort Entries - If an abort occurs during launch, the spacecraft must perform an entry from initial conditions that impose substantially higher loads and heating. The high heating environment is of relatively short duration and does not critically affect an ablatively cooled vehicle design. The maximum entry load factor, however, can greatly exceed that for a normal entry. This is shown in Figure 4-19. Contours of constant maximum load factor during entry of an $L/D = 0.2$ ballistic vehicle are shown superimposed on several representative launch trajectories. Each point on the boundaries shown represents the maximum load factor attained during an entry from an apogee at that point. In the high altitude portions of the boost trajectories, the apogee of the abort will nearly coincide with the boost trajectory. For aborts from launch trajectories injecting to a 100 na mi orbit, maximum recovery load factors of 14 to 16 g's can be imposed on a ballistic vehicle. However, for lifting vehicles the load factors are less severe. They can be further alleviated by modulating the angle-of-attack during the recovery pullout. Maximum load factors of 7 to 8 g can be attained for the M2-F2 after abort from a Saturn launch vehicle injecting to a 100 na mi parking orbit.

4.7 Landing - After completing the entry maneuvers, the reusable spacecraft must be capable of a land recovery as the primary mode. In addition, in the event of an abort during launch, a suitable landing site may not be available and provision for water landings are necessary. In this study, three general concepts of recovery are used. These are summarized below:

Figure 4-19

HIGH ALTITUDE ABORT
MAXIMUM ENTRY LOAD FACTOR
 $L/D = .2 \quad W/C_D A = 90.35 \text{ LB/FT}^2$



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TABLE 4-7
RECOVERY CONCEPTS

Spacecraft Configuration	Primary Recovery System	Abort/Back-Up System
Modular Ballistic (60 deg cone)	Sailwing	Redundant Sailwing
Integral Ballistic (40 deg cone)	3 Clustered Ringsails plus Retro-Rockets	None ⁽¹⁾
M2-F2	Runway Landing	Ringsail ⁽²⁾

Notes:

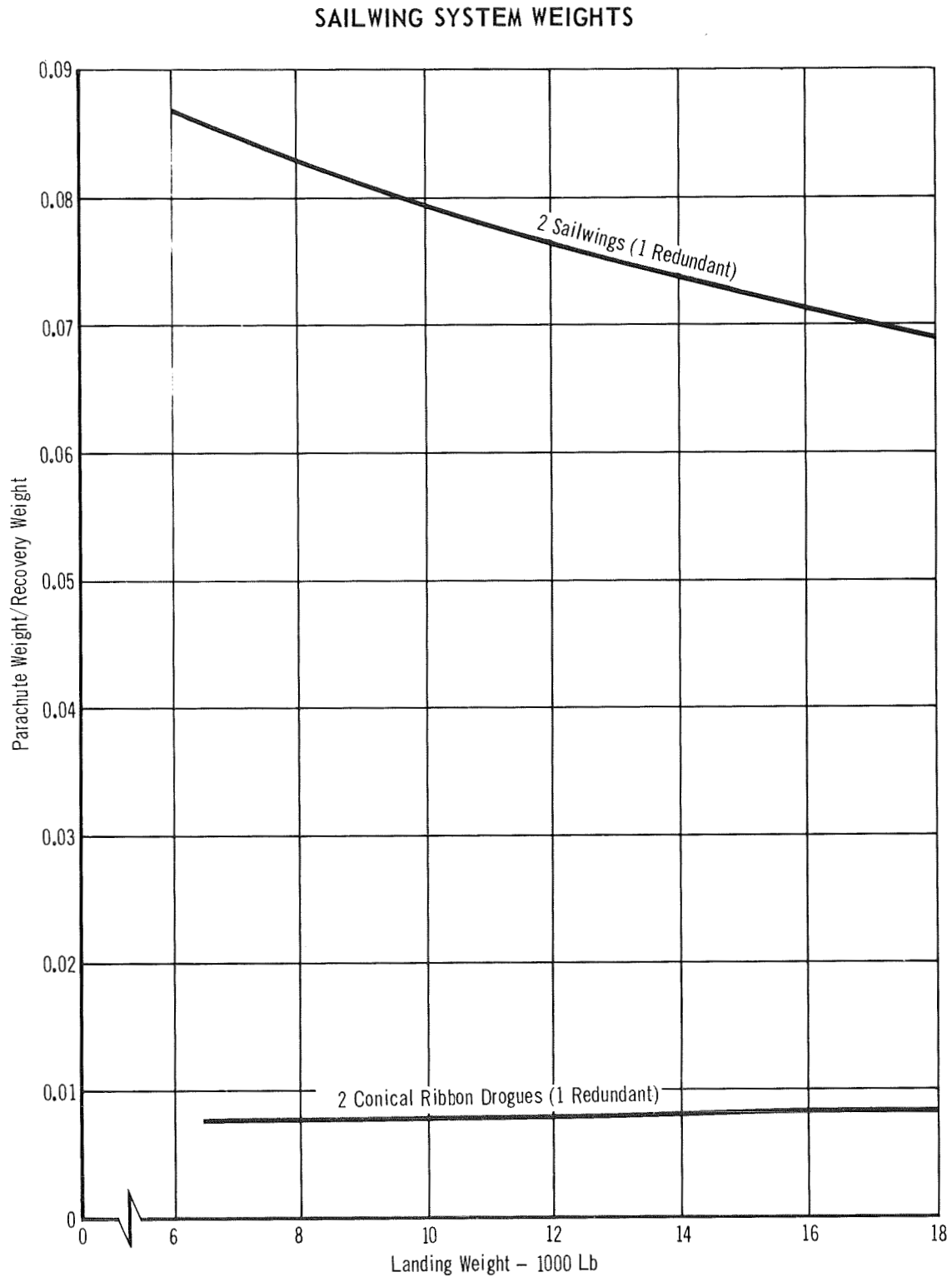
1. Back-up not needed. If one chute fails, the remaining two in cluster will satisfy emergency descent conditions.
2. Entire spacecraft recovered for modular vehicle. Escape capsule used for large integral vehicles.

4.7.1 Ballistic Spacecraft - For land landings of a ballistic spacecraft, a lifting chute provides advantages of entry dispersion correction, wind drift cancellation, and obstacle avoidance. Several such chutes are being developed for spacecraft in the 15,000 pound class. Included are the sailwing, the para-glider, and the parafoil; all having lift to drag ratios greater than 2. The sailwing was selected for the modular ballistic vehicles with an identical system as a back-up. Estimated weights are shown in Figure 4-20. The weights shown include both the primary and back-up sailwings but do not include the control systems. Figure 4-20 also presents the weights of the drogue para-chutes required to reduce the dynamic pressure to the level for sailwing deployment. This weight also includes both the primary and back-up drogues which are deployed simultaneously.

The sailwing requires several reefing stages during deployment to prevent excessive line and canopy loadings. The reefing lines and cutters are included in the weights shown.

For the large integral ballistic configurations, having landing weights to approximately 200,000 lb, use of a lifting chute becomes unfeasible. The system selected is a cluster of three ringsail chutes, similar to the Apollo system. The size of each chute in the cluster is

Figure 4-20



100 ft diameter. Using a cluster factor of 0.86, the maximum landing weight with a sea level rate of descent of 20 ft/sec is about 6700 lb. Since all of the large ballistic vehicles are heavier than this, a landing retro-rocket system is used to reduce the touchdown velocity to the design value of 20 ft/sec.

The terminal rate of descent corresponding to the selected parachute sizes are shown in Figure 4-21. The difference between the descent rate shown and 20 ft/sec represents the velocity that must be arrested with the retro-rocket system. No redundant back-up chute is provided. If one chute in the cluster fails to open, the remaining two chutes will provide a somewhat higher rate of descent as shown by the dashed curve of Figure 4-21.

With the relatively low area loadings of the large ballistic spacecraft, the dynamic pressures at main chute deployment will normally be low enough to permit direct deployment without a drogue. However, to ensure satisfactory stability and vehicle orientation at chute deployment if the reaction control system fails, a drogue system is necessary. A conical ribbon drogue parachute having a diameter about 1.25 that of the heat shield diameter is adequate. An identical chute, deployed simultaneously, provides redundancy.

4.7.2 Lifting Body Spacecraft - The M2-F2 lifting body vehicles can perform a conventional runway landing, their primary recovery mode. To abort over water or where a suitable landing site cannot be reached, a parachute recovery system is needed. With the small modular vehicles, the entire spacecraft is recovered while a separate crew escape module is utilized with the large integral vehicles. For an emergency recovery system comprised of a cluster of three ringsail parachutes designed for a vertical velocity of 30 ft/sec, the chute size required is as shown in Figure 4-22.

Flight tests of the M2-F2 vehicles have indicated a maximum L/D during approach and landing of 3.1. With this value a landing can be executed without power. The landing speed is shown in Figure 4-23 as a function of the area loading W/S. For the current range of area loadings of 32 to 80 lb/ft², the respective landing speed will vary from 157 to 249 knots.

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Figure 4-21

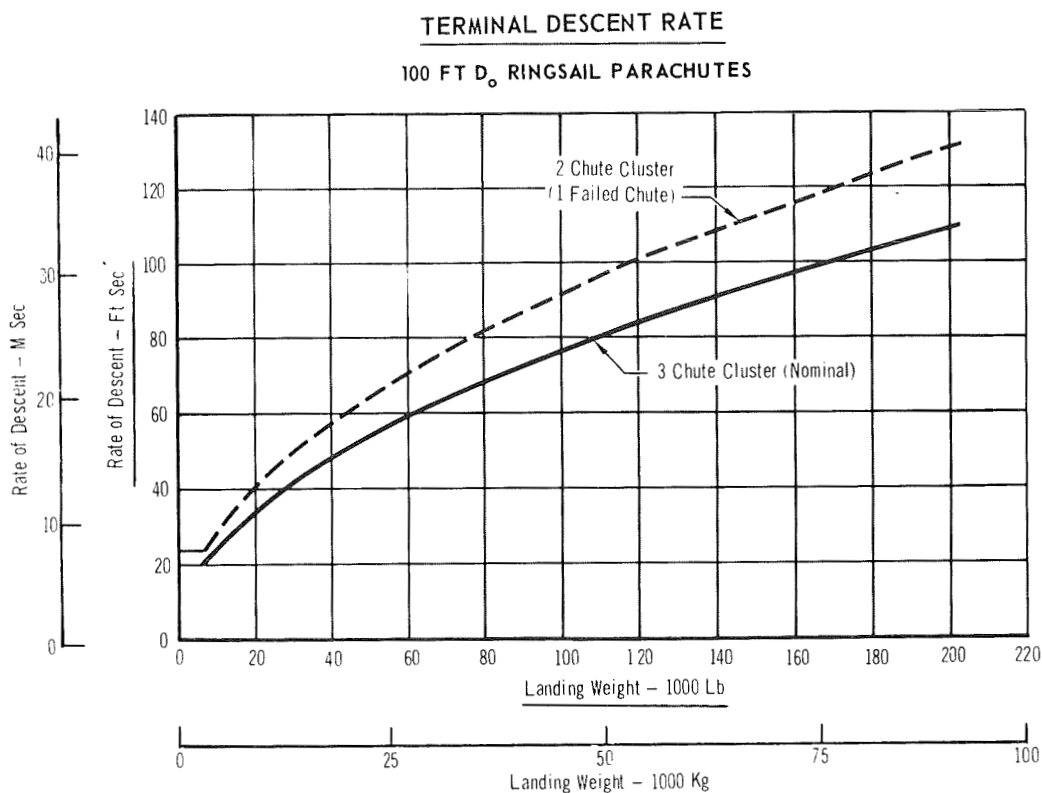


Figure 4-22

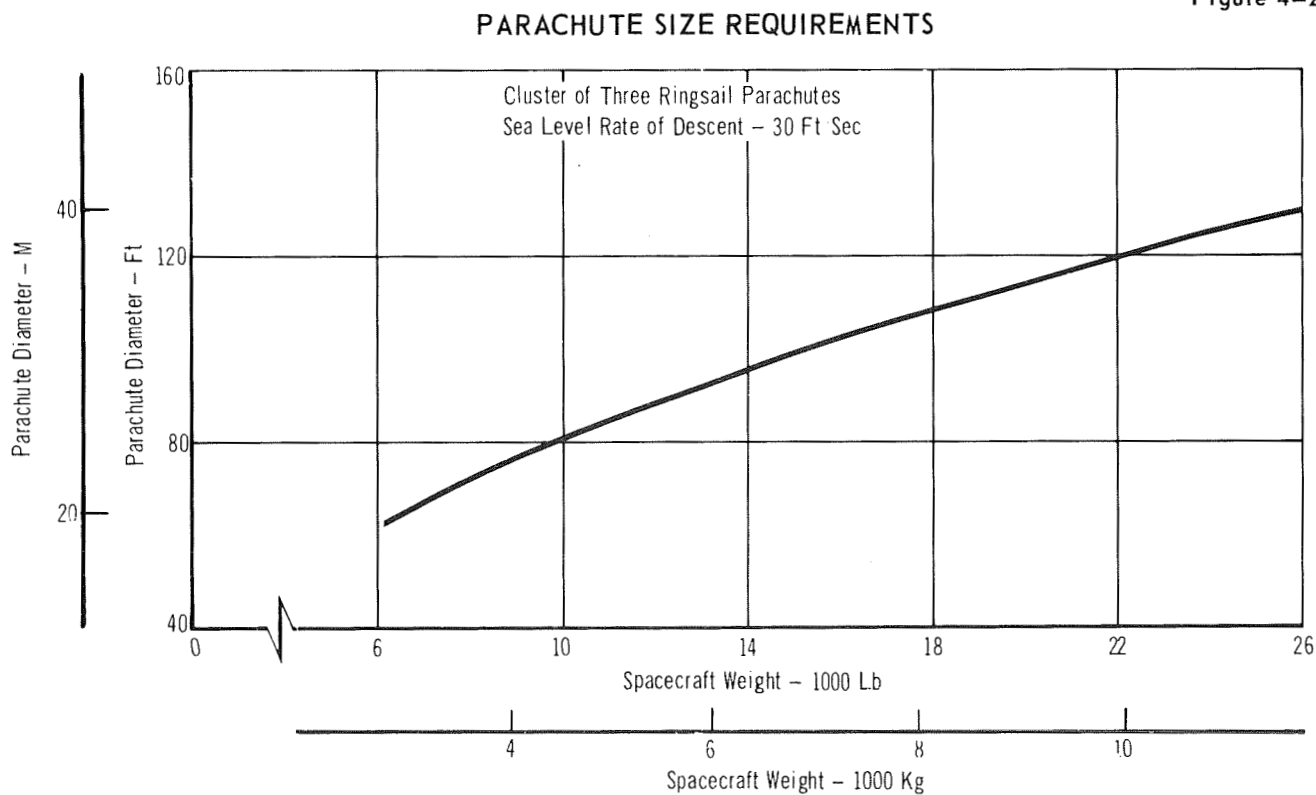
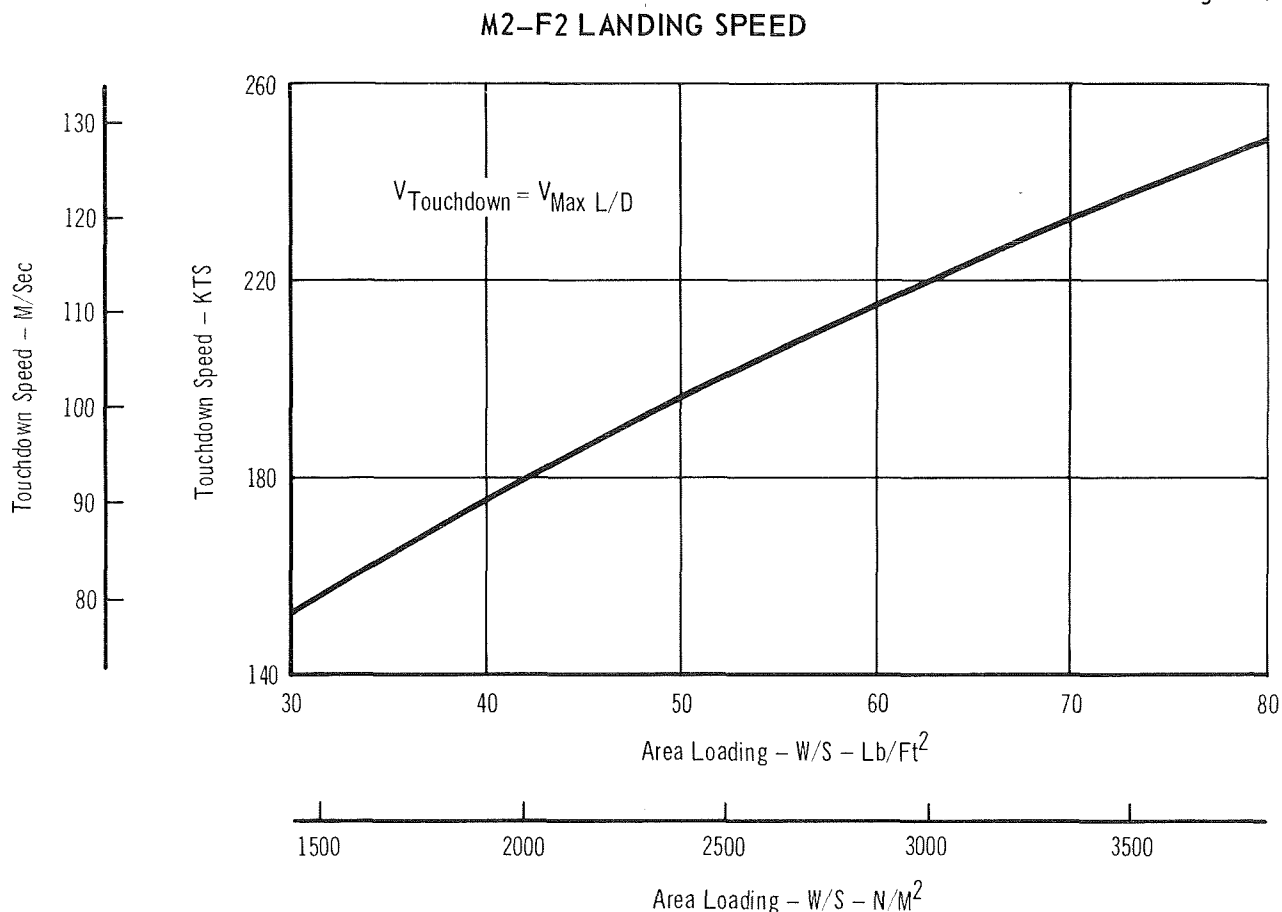


Figure 4-23



If the pilot misjudges his approach and undershoots the runway, landing assist rockets will provide "glide stretch" capability. A system having a thrust to weight ratio of .25 and a burn time of 15 seconds appears adequate and provides a range increase of slightly over 0.5 na mi, similar to that of the M2-F2 and HL-10 flight test vehicles.

Although an L/D of 3.1 is adequate for landings under favorable conditions, an L/D of 6 or greater and go-around capability is generally preferred for operational spacecraft, especially after extended periods of weightlessness. An effective L/D of 6 could be provided by installing an assist rocket having a thrust to weight ratio of about 0.16. A minimum operating time of about one minute, (from prior to flare initiation until touchdown) is needed.

Go-around capability requires more than an order of magnitude increase in propulsive capacity and is more efficiently accomplished with air breathing engines. However, these capabilities were not considered for this study.

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5. SPACECRAFT CHARACTERISTICS - The entry spacecraft of this study comprise both modular and integral reusable vehicles covering a wide range of sizes. The modular vehicles were relatively small, sized only to accommodate the crew, support equipment, and a small amount of return cargo. The integral vehicles were sized for a range of cargo weights varying from 20,000 lb to over 200,000 lb, carried internally in the entry vehicle. The spacecraft characteristics are discussed according to the entry vehicle type, the reuse category, and some special studies.

5.1 Entry Vehicle Types - Two basic types of entry configurations were considered; a ballistic capsule shape similar to Apollo, and a lifting body. Two variations of the ballistic capsule were used. For the small, modular entry vehicles, a capsule having a 60 degree cone angle at the apex was selected. The large integral vehicles utilized a 40 degree cone design because of the larger volume requirements. These vehicles can have a hypersonic L/D during entry from zero to over 0.5. This limited L/D capability provides sufficient crossrange and downrange maneuver capability to correct entry dispersions. However, it is not adequate to ensure once-a-day return capability to a selected landing site unless an orbit phasing maneuver is used. (See Section 4.5).

For the lifting body entry vehicles, the M2-F2 configuration was selected. This spacecraft has a hypersonic L/D of 1.1, sufficient to provide once-a-day return without orbit phasing.

5.1.1 Ballistic Vehicle - The ballistic vehicle achieves lift by positioning the center of gravity off the longitudinal axis. This induces an angle-of-attack and rotates the resultant force vector to produce a lift component. The magnitude of the lift force and the resulting lift-to-drag ratio varies directly with the angle-of-attack and depends on the c.g. offset attainable and the constraints imposed on the allowable angle-of-attack. An angle-of-attack greater than half the cone angle is generally undesirable since it exposes the after body to a severe heating environment. The primary parameter affecting the variation of lift coefficient and L/D with angle-of-attack is the heat-shield-radius-to-diameter ratio. (This is the ratio of heat shield radius of curvature to the projected diameter of the heat shield.)

These values for the 40 degree and 60 degree cone ballistic spacecraft are 1.45 and 1.60, respectively, compared to about 1.20 for the Apollo configuration, resulting in a flatter heat shield. The primary effect of the flatter heat shield is a higher hypersonic L/D at a given trim angle-of-attack. The flatter heat shield also results in less c.g. offset required to achieve a given trim angle-of-attack. The estimated lift and drag characteristics of the 60 degree cone and 40 degree cone vehicles are shown in Figures 5-1 and 5-2, respectively. The center of gravity offset required to trim the 60 degree cone to an L/D of 0.5 and the 40 degree cone to an L/D of 0.2 is shown in Figure 5-3 as a function of the longitudinal c.g. position. The aft c.g. boundary, based on transonic stability considerations, is also indicated.

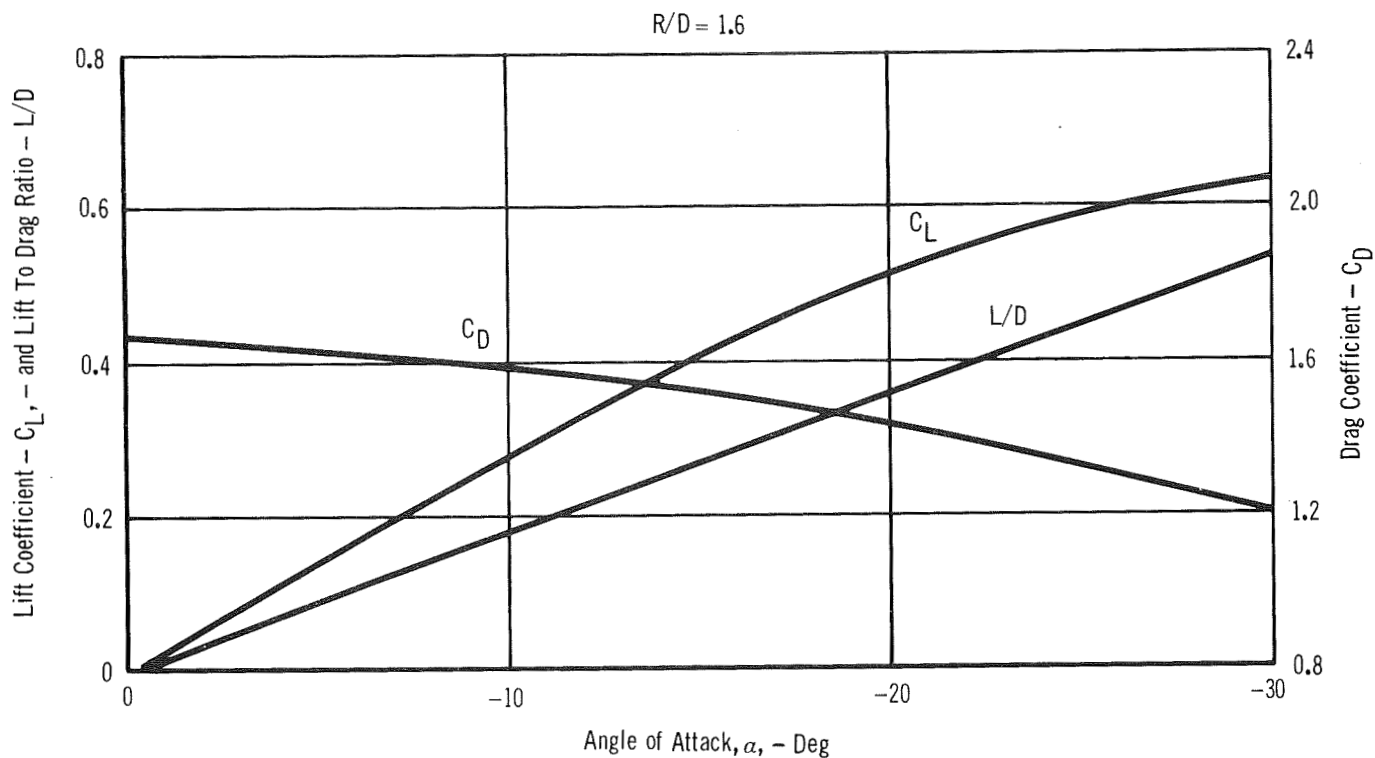
5.1.2 Lifting Body Vehicle - A lifting body entry vehicle having an L/D of about 1.0 has the advantage of once-a-day return capability to a specified landing site without the use of return phasing maneuvers. The M2-F2 configuration selected provides a maximum hypersonic L/D of about 1.1 (References 5-1 and 5-2). The estimated hypersonic trimmed lift and drag characteristics are shown in Figure 5-4 as a function of angle-of-attack. As shown, the maximum L/D is attained at an angle of 11 degrees while the maximum lift coefficient is at an angle-of-attack of 36 degrees. The vehicle attitude can be modulated between these limits and also modulated in bank angle to provide the desired entry performance.

Stability characteristics are shown in Figure 5-5. At hypersonic speeds, the vehicle is trimmed from the maximum lift attitude (36 degrees) to the maximum L/D attitude (11 degrees) by deflecting the lower flap from zero to 45 degrees. Center of gravity is at 50 percent of the body length. Rudders are deflected outboard 25 degrees for additional directional stability. As shown, the vehicle exhibits satisfactory longitudinal static stability throughout the trim angle-of-attack range.

5.2 Reuse Categories - To assess the economics of recovery, and reuse of various vehicle elements, categories covering the range from a totally expendable spacecraft and launch vehicle to a completely reusable spacecraft and launch vehicle were investigated. These categories are shown in Table 5-1. Both water and land landing were considered as primary recovery modes for the ballistic configuration in reuse category A. In category D, point designs rather

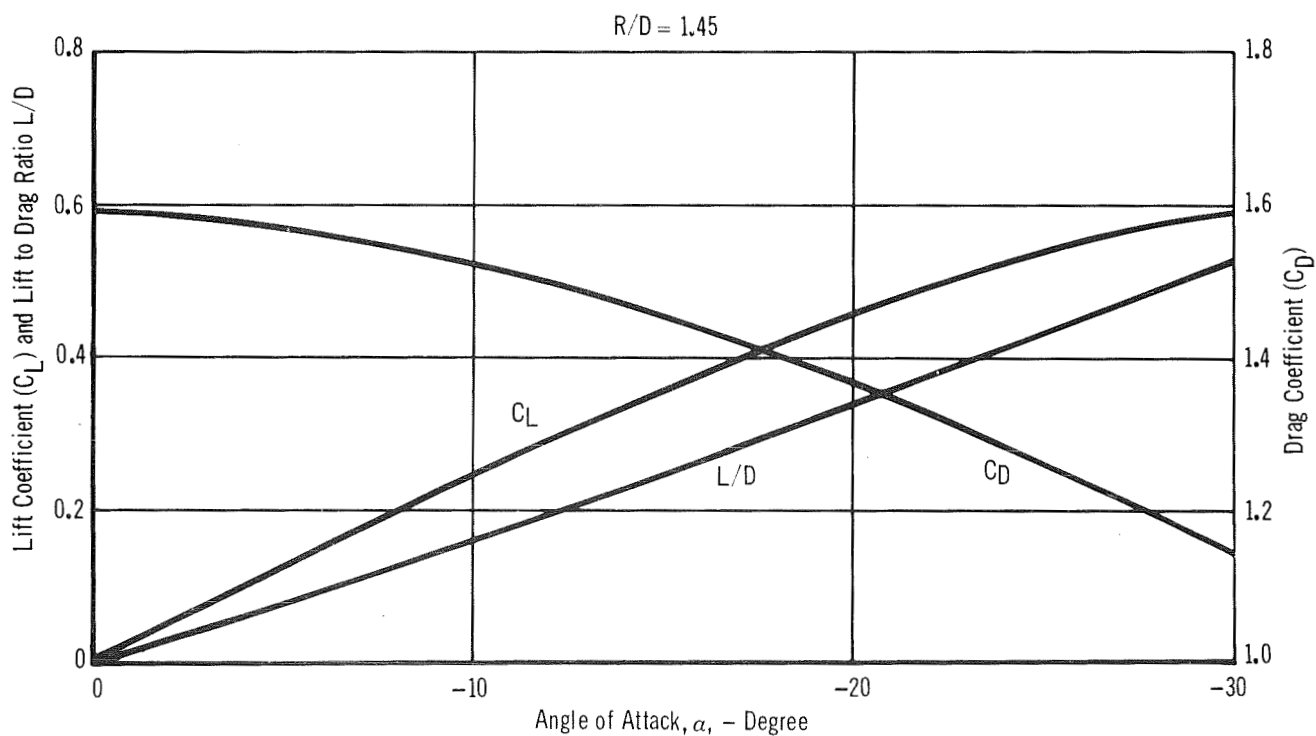
60 DEGREE CONE BALLISTIC CONFIGURATION
HYPERSONIC AERODYNAMIC CHARACTERISTICS

Figure 5-1



40 DEGREE CONE BALLISTIC CONFIGURATION
HYPERSONIC AERODYNAMIC CHARACTERISTICS

Figure 5-2



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Figure 5-3

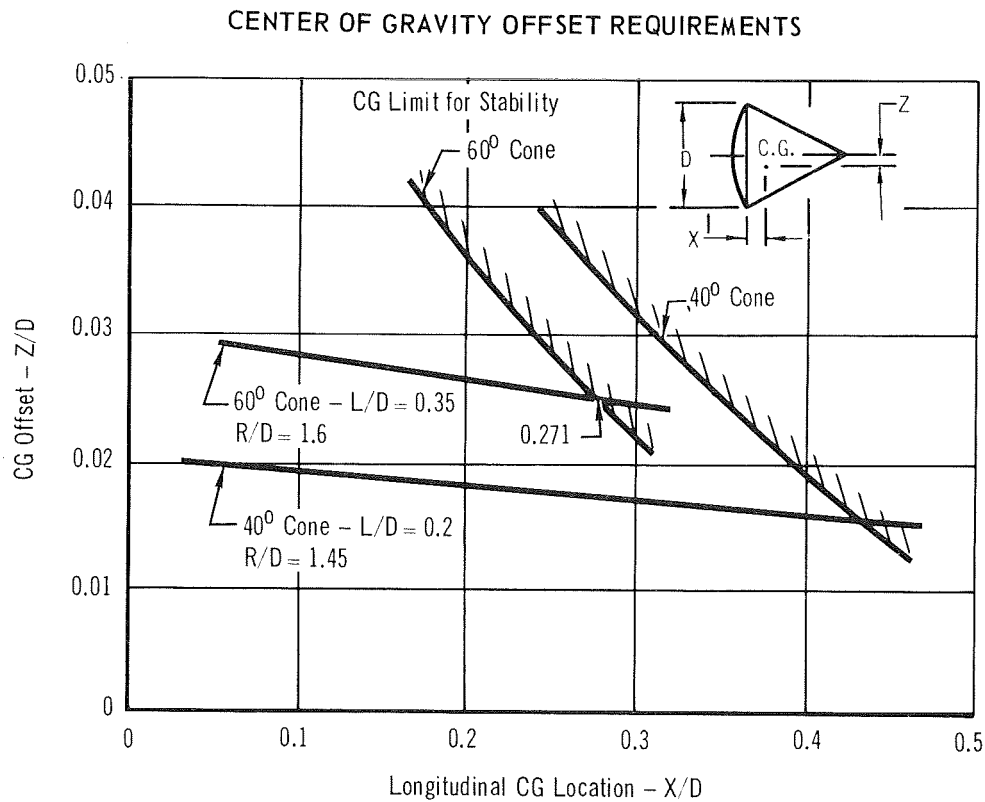


Figure 5-4

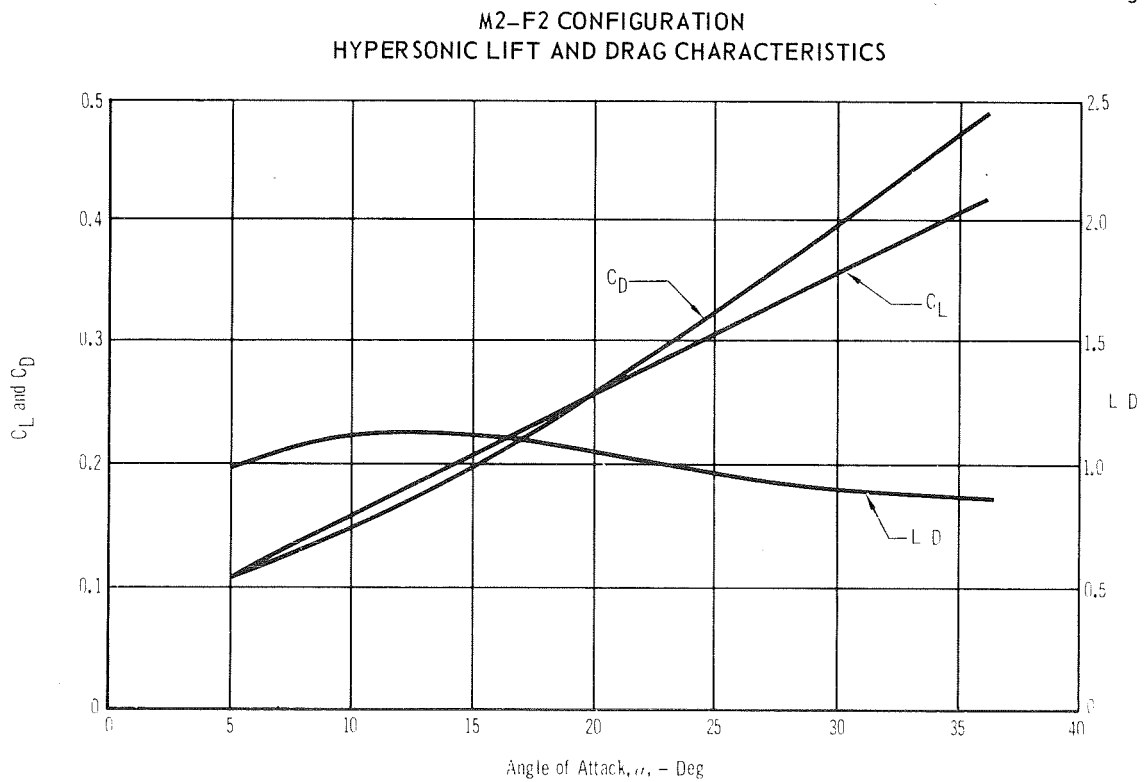


Figure 5-5

M2-F2 CONFIGURATION
HYPERSONONIC STABILITY AND CONTROL

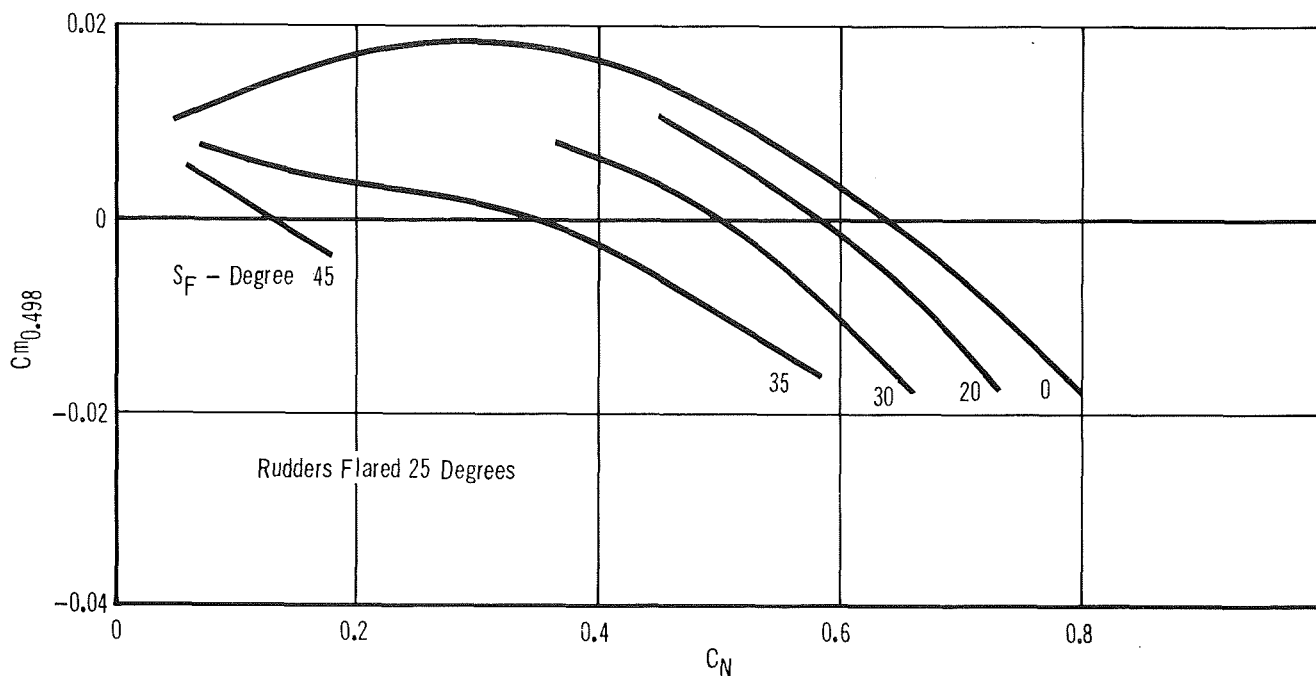


TABLE 5-1
TRANSPORTATION SYSTEMS CONFIGURATION MATRIX

Reuse Category Component	A	B	C	D	E	F
Configuration I (Ballistic)						
Entry Vehicle	E	R	R	R	R	R
Cargo/Orbit Propulsion Module	E	E	R	R	R	R
Upper Stage Engines	E	E	E	R	R	R
Upper Stage Tankage	E	E	E	E	R	R
1st Stage	E	E	E	E	E	R
Configuration II (Lifting Body M2/F2) (Same as for Configuration I)						

E - Expendable

R - Reusable

Integral

than parametric studies were made for the lifting body configuration. In category F, the spacecraft portion of the vehicle is the same as for category E in both the ballistic and lifting body configurations with the only difference between the categories being in the first stage booster.

The vehicle configurations for the reuse categories are shown in Figures 5-6 through 5-14. For each configuration, preliminary estimates of size and weights are given for the nominal mission parameters (nine-man crew, cargo density of ten lb/cu ft, and a 300-mile low inclination orbit), with the payload capability varied over the entire range of interest. For this sizing, the propellants were assumed to be liquid oxygen and liquid hydrogen for boost propulsion and storable for maneuver propulsion. These preliminary sizing estimates were used, together with the mission requirements discussed in Section 4, to determine the range of subsystem performance requirements to be investigated. These estimates will be refined as part of the Task 6 analysis by means of the detailed spacecraft sizing model.

5.2.1 Ballistic Spacecraft - The ballistic spacecraft configurations are based on 20 degree and 30 degree half cones. The baseline modular vehicles (Vehicles IA and IB) are 30 degree half cones, corresponding with the spacecraft in the Advanced Logistics Spacecraft System study (See Reference 4-1). The integral vehicles (IC through IF) use a 20 degree half cone to keep the base diameter smaller. Throughout it has been assumed that the booster diameter corresponds to the vehicle base diameter without a limit to maximum diameter. The use of the 20 degree half cone results in less squat booster configurations.

The geometric characteristics of the modular ballistic spacecraft (Vehicles IA and IB) are shown in Figure 5-6. It has been assumed that Vehicles IA and IB are identical for the land landing configuration, but with the water landing version of the IA differing from the land landing version only in the landing system and in the guidance and navigation equipment.

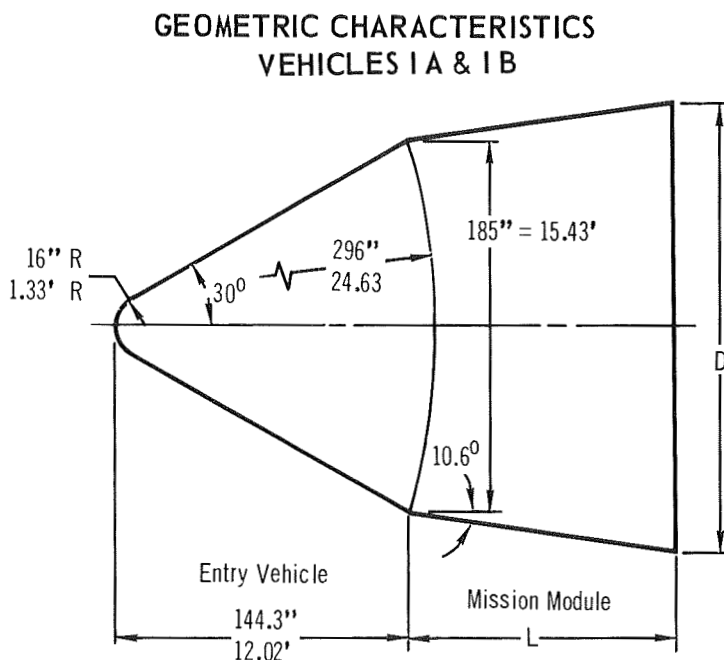
The entry vehicle contains space for the crew, the non-propulsive subsystem equipment, the reentry attitude control, and the return cargo. It also serves as the escape capsule for abort during boost. The mission module contains all the maneuvering, orbit attitude control, and retro propulsion subsystems; ascent cargo space; and the docking provisions. The pressurizable cargo area is integral with the primary structure. All boost propulsion is contained in the booster.

The geometric characteristics of the integral mission module spacecraft (Vehicle IC) is shown in Figure 5-7. In this configuration, the entry vehicle

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Figure 5-6



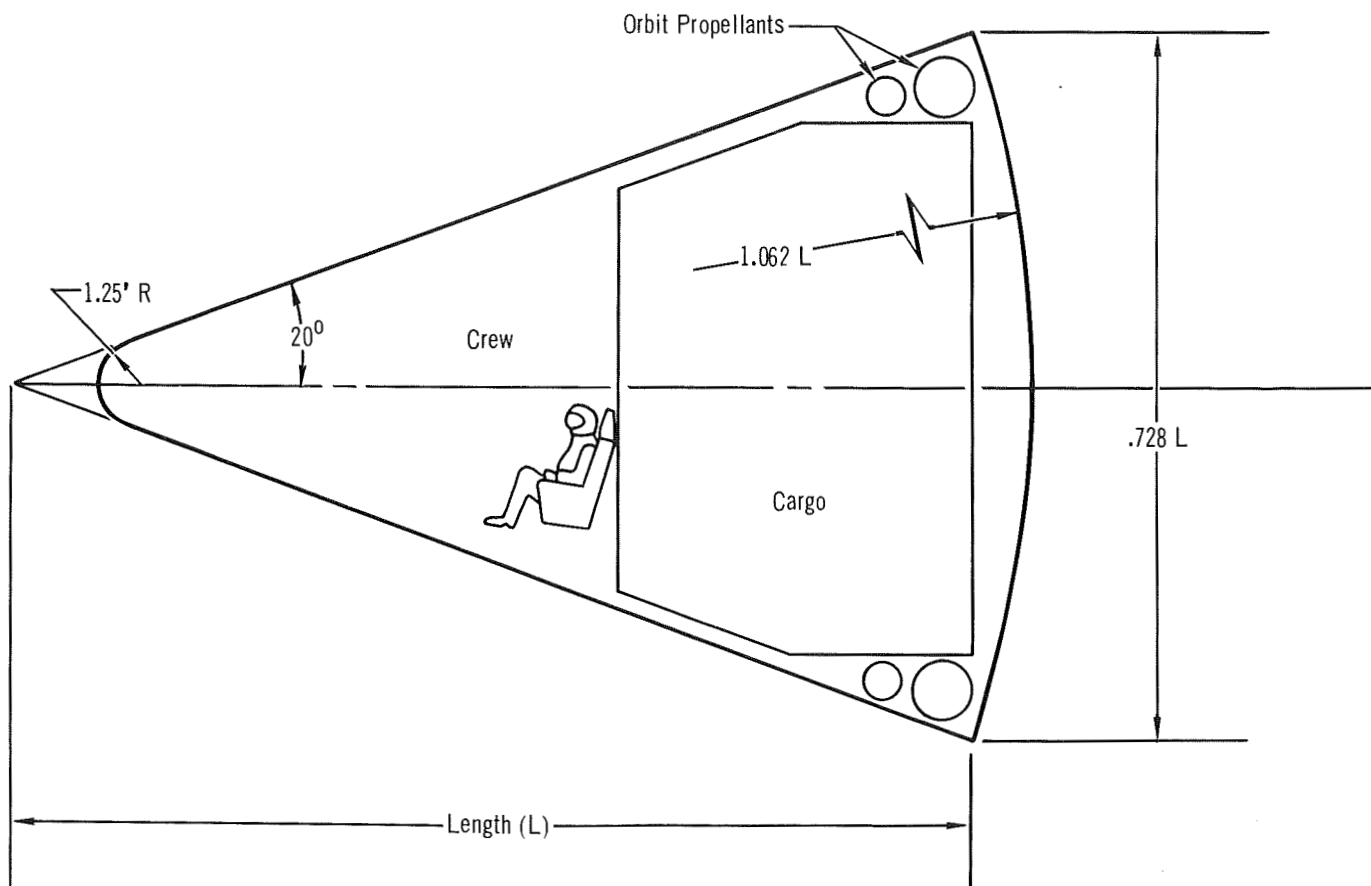
GEOMETRIC CHARACTERISTICS					
Entry Vehicle					
Heat Shield Area	191 Ft ²				
Body Wetted Area	371 Ft ²				
Wetted Area Total	562 Ft ²				
Total Volume	945 Ft ³				
Entry Weight	11,830 Lb				
Area Loading	63.5 Lb/Ft ²				
Mission Module					
	Cargo Capability at 10 Lb/Ft ³				
	20,000	50,000	100,000	200,000	
	Length (L) – Ft	11	23	36	55
	Diameter(D)– Ft	18.4	21.8	25.6	30.8
	Wetted Area – Ft ²	590	1,360	2,260	4,040
Volume – Ft ³	2,580	6,275	11,815	23,995	

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**GEOMETRIC CHARACTERISTICS
VEHICLE IC**

Figure 5-7



GEOMETRIC CHARACTERISTICS					
		Cargo Capability at 10 Lb/Ft ³			
		20,000	50,000	100,000	200,000
Length (L)	- Ft	29.16	38	47	57
Heat Shield Area	- Ft ²	362	615	940	1,380
Body Wetted Area	- Ft ²	1,030	1,760	2,700	3,960
Wetted Area - Total	- Ft ²	1,392	2,375	3,640	5,340
Total Volume	- Ft ³	3,400	7,580	14,300	25,800
Entry Weight	- Lb	18,000	25,000	36,000	56,000
Area Loading	- Lb/Ft ²	51	41	39	41

performs all the functions of both the entry vehicle and the mission module of Vehicle IA. For crew escape during launch, the portion of the spacecraft ahead of the cargo compartment separates to become the escape capsule. It contains all the systems and thermal protection required for abort from near orbital conditions.

Figure 5-8 shows the geometric characteristics of a spacecraft that integrates the booster second stage engine and related hardware into the entry vehicle (Vehicle ID). The second stage propellants are in external expendable tanks. Crew escape provision are the same as for vehicle IC.

The geometric characteristics of a vehicle that incorporates the booster second stage (Vehicles IE and IF) in the entry vehicle is shown in Figure 5-9. The entry vehicles for Vehicles IE and IF are identical, these configurations differ only in that the first stage of the IE is expendable but is reusable for the IF. For these vehicles, crew escape is the same as for Vehicle IC.

Preliminary weight estimates for the escape capsule for the ballistic spacecraft vary from 8400 lb for a two-man capsule to 12,650 lb for the twelve-man capsule. Since the escape capsule size does not vary with reuse concept, these weights apply for both modular and integral vehicles.

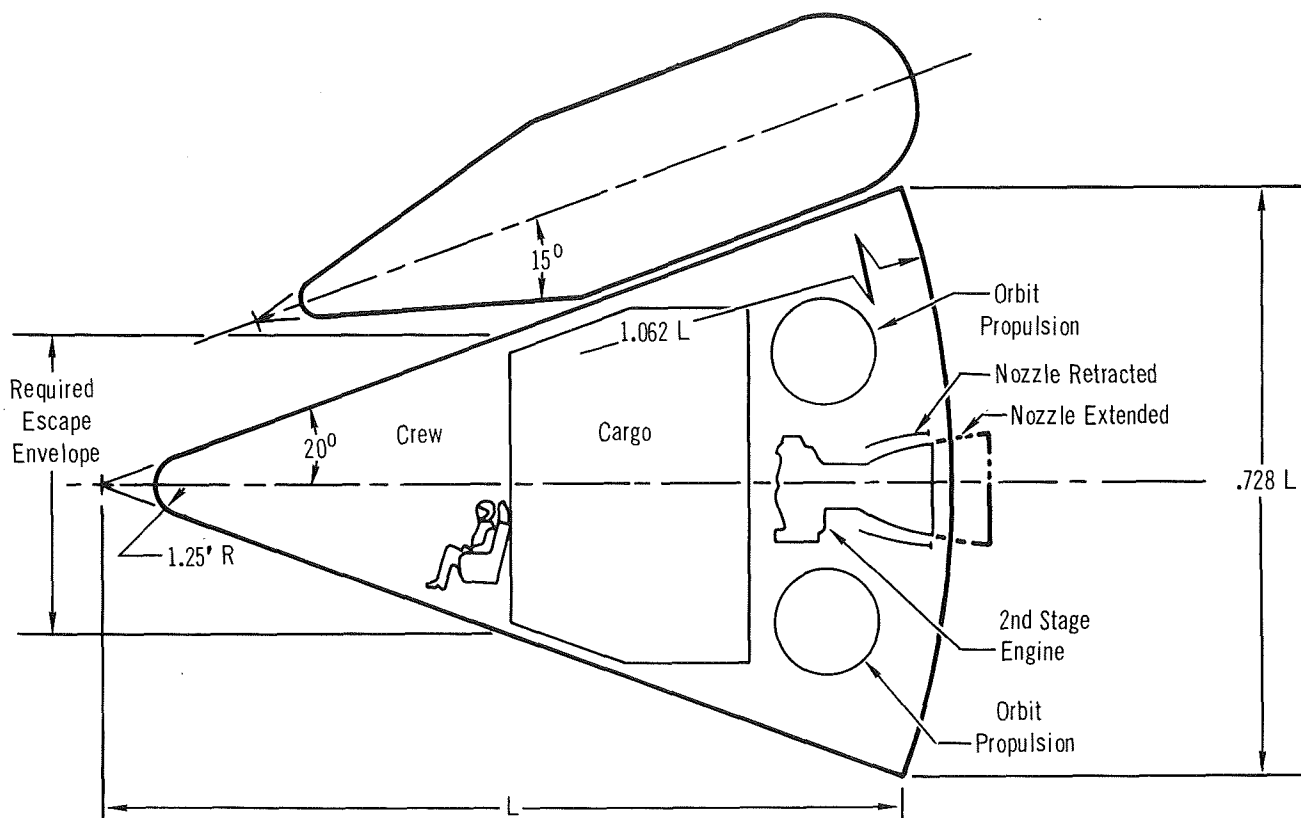
5.2.2 Lifting Body Spacecraft - The lifting body configuration used is the NASA M2-F2. The geometric characteristics for the modular configuration (Vehicles IIA and IIB) are shown in Figure 5-10. The entry vehicle contains the space for the crew, the non-propulsive subsystem equipment, reentry attitude control, and the return cargo. The mission module is similar to that for Vehicles IA and IB, except for the adapter section required for mounting the crew module.

As in Vehicles IA and IB, the entry vehicle serves as the escape module for abort during boost. The preliminary estimate of crew module weight for vehicle IIB varies from 11,850 lb for a two-man capsule to 17,846 lb for the twelve-man capsule.

Vehicles IIC, IIE and IIF are shown in Figures 5-11 and 5-12. These vehicles correspond to their ballistic entry vehicle counterparts. For abort during launch with these vehicles, the portion ahead of the cargo section is

Figure 5-8

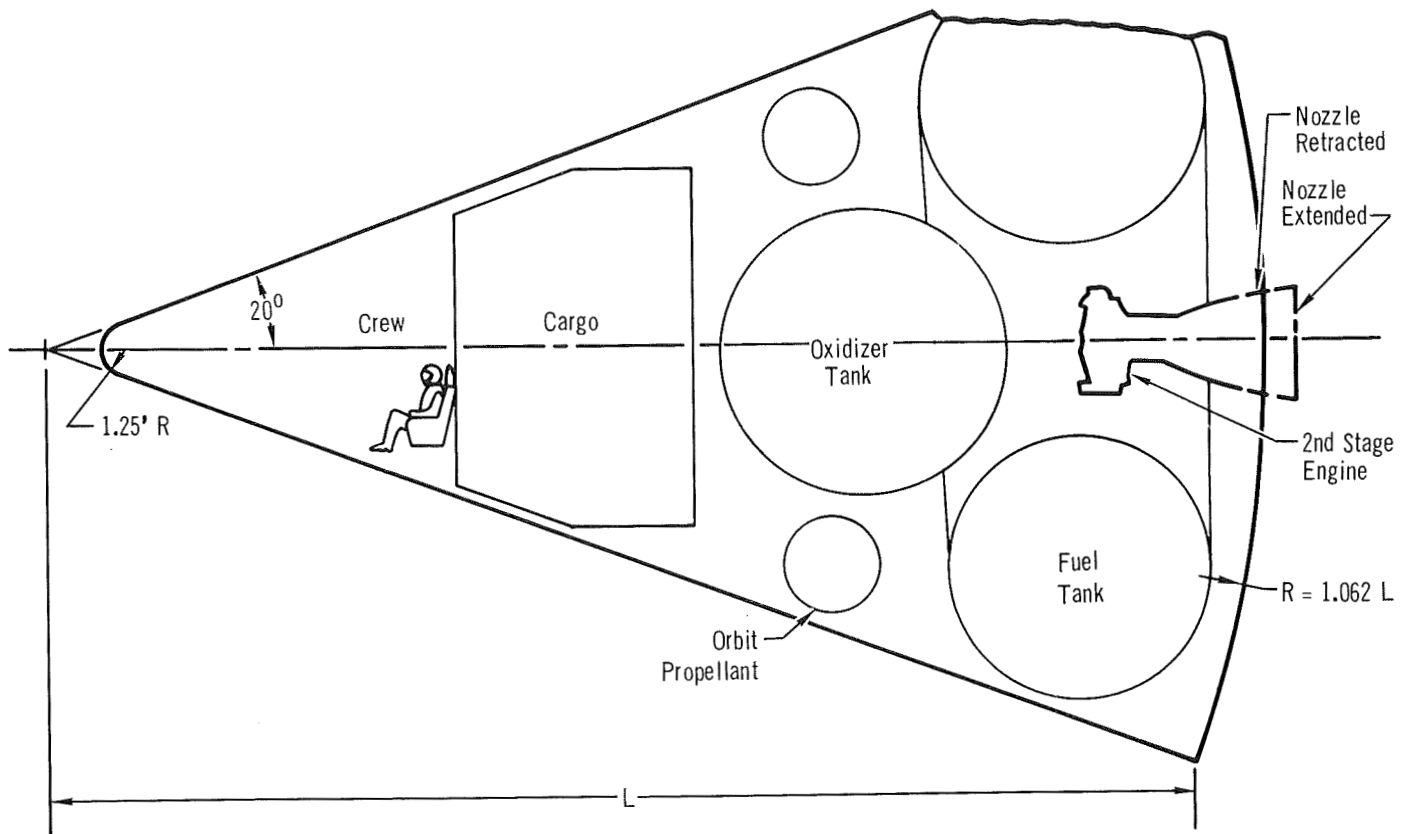
GEOMETRIC CHARACTERISTICS
VEHICLE 1D



GEOMETRIC CHARACTERISTICS					
Entry Vehicle		Cargo Capability at 10 Lb/Ft ³			
		20,000	50,000	100,000	200,000
Length (L)	- Ft	36	44	52	64
Heat Shield Area	- Ft ²	555	820	1,150	1,750
Body Wetted Area	- Ft ²	1,595	2,360	3,300	5,020
Wetted Area - Total	- Ft ²	2,150	3,180	4,450	6,770
Total Volume	- Ft ³	6,450	11,700	19,400	36,200
Entry Weight	- Lb	25,500	35,000	48,000	69,000
Area Loading	- Lb/Ft ²	47	43	43	41
Expendable Tanks					
W _p	- Lb	129,000	240,000	415,000	750,000
No. of Tanks		6	6	6	6
W _p Per Tank	- Lb	21,500	40,000	69,100	125,000
Tank Length (L _T)	- Ft	27.5	33.8	40.5	49.5
Tank Diameter (L _T /4)	- Ft	6.9	8.5	10.1	12.4

Figure 5-9

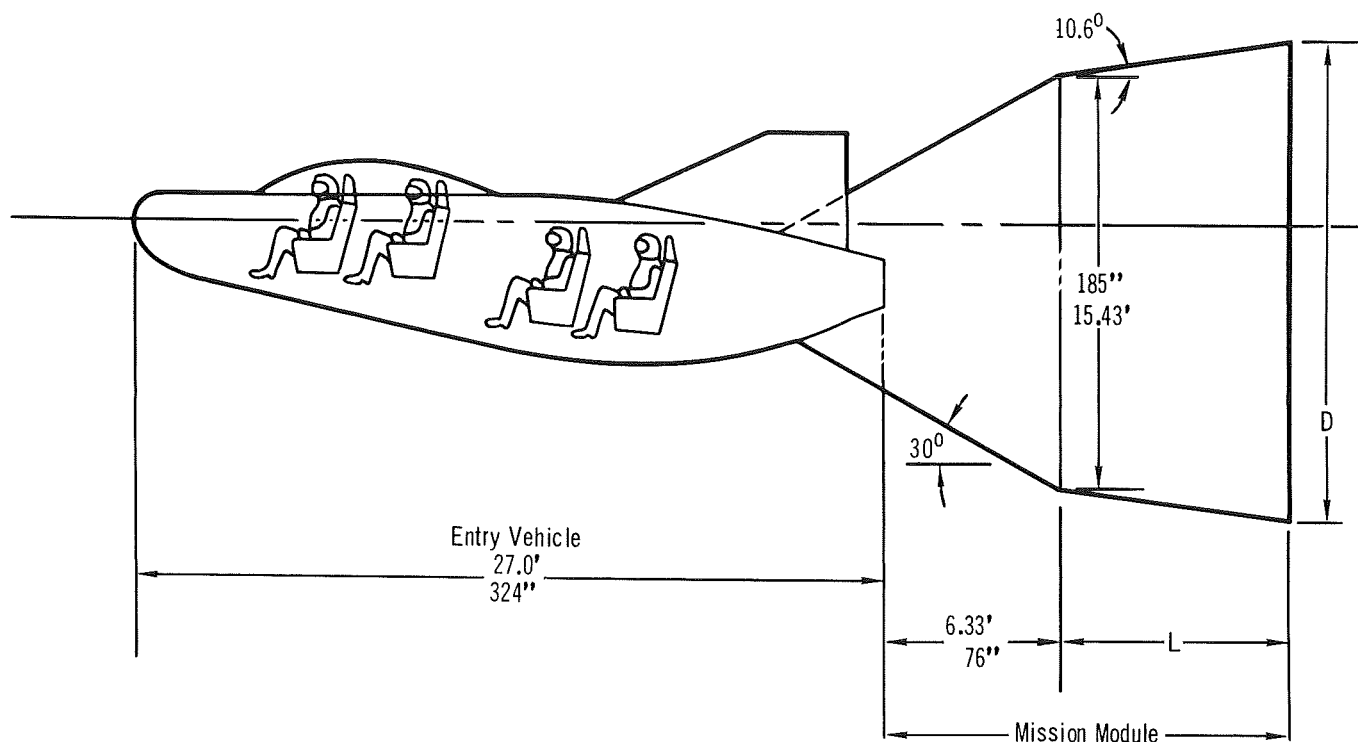
**GEOMETRIC CHARACTERISTICS
VEHICLES 1E & 1F**



GEOMETRIC CHARACTERISTICS				
	Cargo Capability at 10 Lb/Ft ³			
	20,000	50,000	100,000	200,000
Length (L) - Ft	51.5	63	75	90
Heat Shield Area - Ft ²	1,125	1,680	2,400	3,440
Body Wetted Area - Ft ²	3,230	4,850	6,870	9,900
Wetted Area - Total - Ft ²	4,335	6,530	9,270	13,340
Total Volume - Ft ³	18,800	34,500	58,200	99,500
Entry Weight - Lb	50,000	76,000	120,000	204,000
Area Loading - Lb/Ft ²	44	45	50	59

Figure 5-10

**GEOMETRIC CHARACTERISTICS
VEHICLES II A & II B**



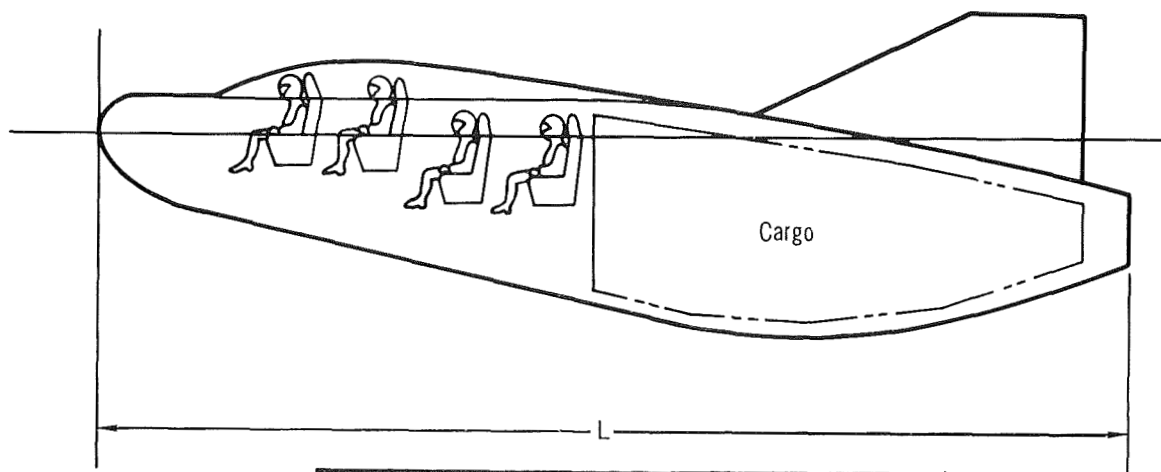
GEOMETRIC CHARACTERISTICS					
Entry Vehicle					
Planform Area – Ft ²	231				
Wetted Area –Total– Ft ²	687				
Volume – Ft ³	907				
Entry Weight – Lb	16,600				
Area Loading – Lb/Ft ²	72				
Mission Module					
	Cargo Capability at 10 Lb/Ft ³				
	20,000	50,000	100,000	200,000	
	Length (L) – Ft	8.5	21.5	34.5	54.0
	Diameter (D) – Ft	17.8	21.4	25.0	30.6
	Wetted Area – Ft ²	800	1,610	2,570	4,290
	Volume – Ft ³	2,580	6,275	11,815	23,995

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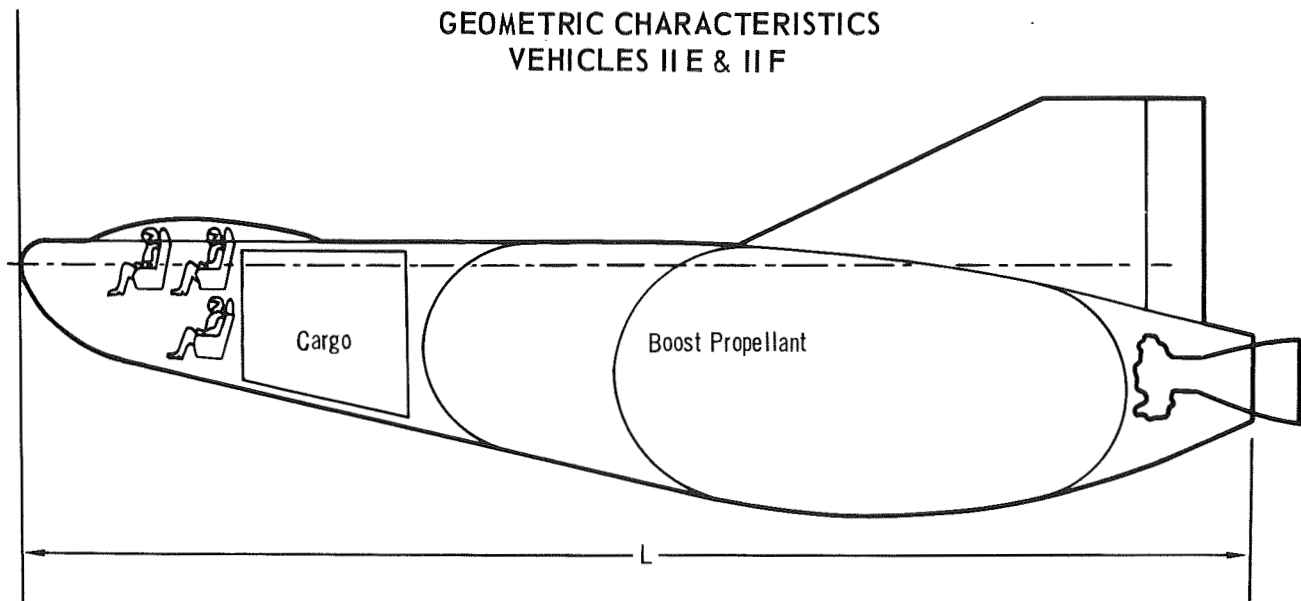
Figure 5-11

**GEOMETRIC CHARACTERISTICS
VEHICLE IIC**



GEOMETRIC CHARACTERISTICS				
	Cargo Capability at 10 Lb/Ft ³			
	20,000	50,000	100,000	200,000
Length (L) - Ft	42.0	53.5	66.5	84.0
Planform Area - Ft ²	560	915	1,400	2,250
Wetted Area - Total - Ft ²	1,670	2,720	4,150	6,700
Total Volume - Ft ³	3,420	7,100	13,600	27,000
Entry Weight - Lb	26,000	36,000	48,000	72,000
Area Loading - Lb/Ft ²	46	40	34	32

Figure 5-12



GEOMETRIC CHARACTERISTICS					
		Cargo Capability @ 10 Lb/Ft ³			
		20,000	50,000	100,000	200,000
Length (L)	- Ft	74	92	112	139
Planform Area	- Ft ²	1740	2680	3980	6120
Wetted Area- Total	- Ft ²	5160	8000	11,800	18,200
Total Volume	- Ft ³	18,700	36,000	64,700	123,200
Entry Weight	- Lb	60,000	93,000	148,000	248,000
Area Loading	- Lb/Ft ²	35	35	37	40

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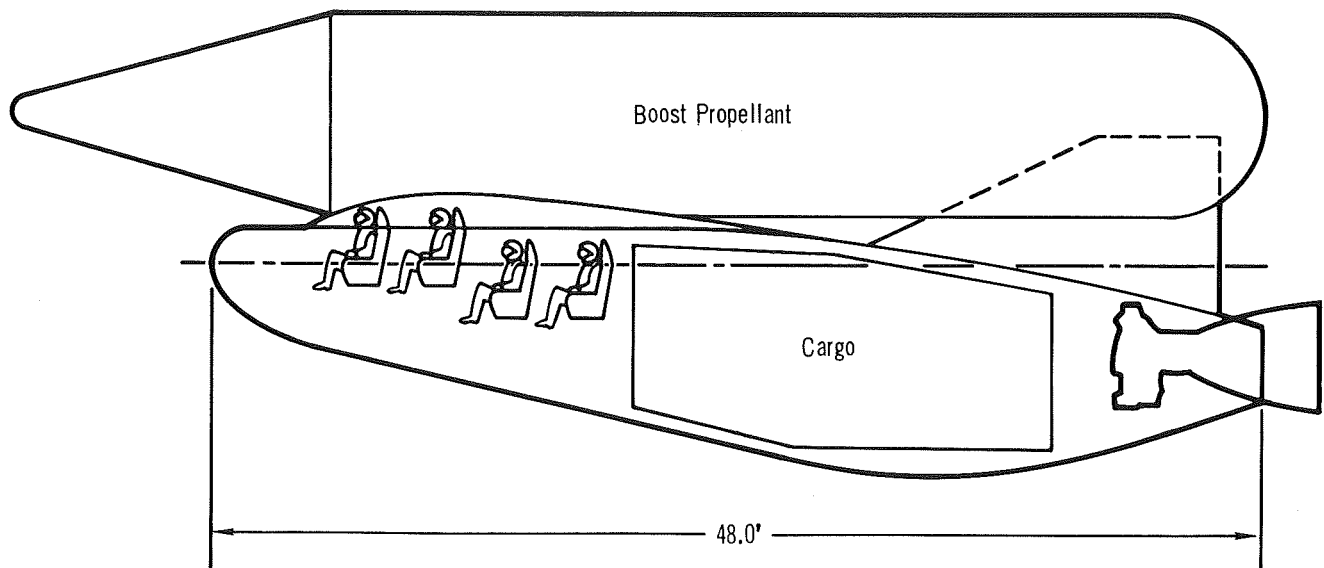
separated from the spacecraft and serves as the escape capsule. Preliminary estimates of escape capsule weight for these concepts range from 9550 lb for a two-man IIC capsule to 14,600 lb for a twelve-man IIE capsule.

5.2.3 Special Studies - During development of the spacecraft characteristics, it became apparent that the complete analysis of tip tanks on an M2-F2 vehicle would probably not be as valuable as the analysis of the other configurations.

This type M2-F2 configuration has not been defined in detail in any previous work and could not be defined in this study with sufficient depth to warrant more than a cursory evaluation. Therefore, at the direction of NASA this configuration will be evaluated only for the nominal mission (9 man, 20,000 lb of cargo, 50 deg orbit inclination), but two tank configurations will be considered as shown in Figures 5-13 and 5-14. In the first, two expendable, external cylindrical tanks carry the upper stage boost propellant. In the second, the expendable tank has a configuration similar to the M2-F2 reentry vehicle. For these two configurations, no consideration is given to the effects of c.g. travel, engine gimbaling requirements, first stage booster interface, or clearance provisions for engine gimbaling. A third point design concept that will be evaluated at NASA's request is a version of the IC vehicle in which the orbital propulsion system is integral with the entry vehicle and the cargo is contained in an expendable module.

Figure 5-13

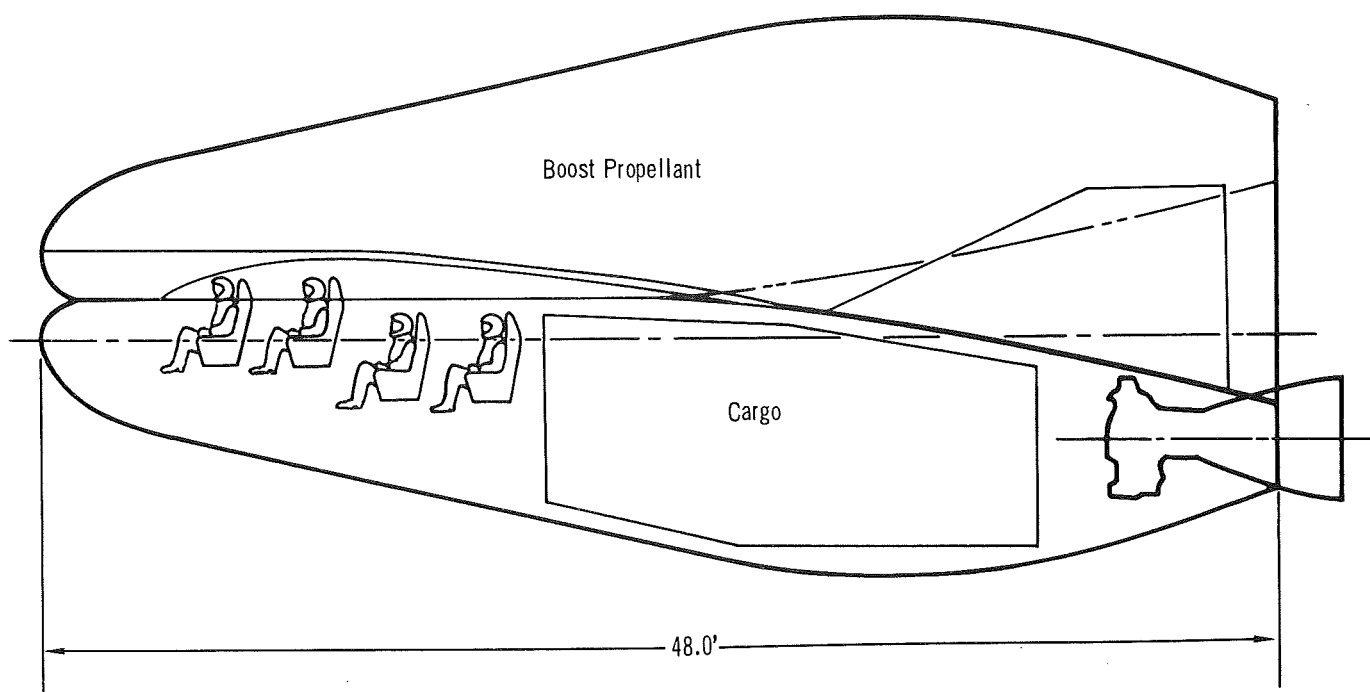
**GEOMETRIC CHARACTERISTICS
VEHICLE IID**



GEOMETRIC CHARACTERISTICS		
Cargo Capability at 10 Lb/Ft ³ - Lb		20,000
Length - Ft		48
Planform Area - Ft ²		730
Wetted Area - Total - Ft ²		2170
Total Volume - Ft ³		5050
Entry Weight - Lb		32,000
Area Loading - Lb/Ft ²		44
Expendable Tanks		
W _p - Lb		148,000
Number of Tanks		2
W _p per Tank - Lb		74,000
Tank Length - Ft		54
Tank Diameter - Ft		9.0

Figure 5-14

**GEOMETRIC CHARACTERISTICS
VEHICLE IID-I**



GEOMETRIC CHARACTERISTICS		
Cargo Capability at 10 Lb/Ft ² - Lb		20,000
Planform Area - Ft ²		730
Wetted Area - Total - Ft ²		2,170
Total Volume - Ft ³		5,050
Entry Weight - Lb		32,000
Area Loading - Lb/Ft ²		44
Volume of Expendable Tank - Ft ³		6,600
Propellant Weight - Lb		148,000

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6. SUBSYSTEM REQUIREMENTS, CONCEPTS, AND DEFINITIONS - Five subsystems were identified as high cost items, either from the standpoint of development or acquisition costs or on their effect on total spacecraft weight. These subsystems are structure (including thermal protection), propulsion, environmental control, avionics, and non-propulsive power. For each, the functional and performance requirements were identified and quantified by mission phase for each of the reuse categories and for the ranges of crew size and cargo capabilities.

Using the resulting range of performance requirements, subsystem specifications were prepared. Suppliers were invited early in the study to submit design, reliability and cost data (either parametric or point design), on hardware meeting the requirements in the subsystem specifications. Alternate subsystem concepts ranging from off-the-shelf to advanced state-of-the-art were encouraged as long as the performance requirements were met. The suppliers that submitted data are listed in Appendix A. In general, the design data substantiated data already available from previous vendor contacts.

With data from previous studies of logistics vehicles and from suppliers, various subsystem concepts which were capable of meeting the performance requirements were identified. These included current and advanced technology, low weight and low cost, and simple and sophisticated concepts. From these, baseline subsystems were chosen for each reuse category and aerodynamic configuration and are summarized in Tables 6-1 and 6-2. These baseline subsystems were chosen merely as a point of departure for the subsystems trade-off studies to be made in a later task. All the alternate subsystems will be evaluated during the trade-off studies.

In addition to the specific subsystem requirements, there are also general requirements for maintainability and reliability. The following paragraphs discuss the general requirements and then each of the five subsystem areas.

6.1 General Requirements

6.1.1 Maintainability - Reuse capability requires the ability to maintain the desired reliability and confidence levels through replacement of components that have reached the limit of their operational life and components that have

Table 6-1

BASELINE SUBSYSTEMS DESCRIPTION SUMMARY BALLISTIC VEHICLE CONFIGURATIONS

Subsystem	Reuse Categories					
	A (Water Landing) Modular-Expendable	A (Land Landing) Modular-Expendable	B - Modular-Reusable	C - Integral Cargo	D - Tip Tank	E & F - Integral
Structure	Skin-Stringer Structural Shell (Entry Vehicle and Mission Module)			Skin-Stringer Structural Shell		
Thermal Protection	Ablative heat shield - Shingles for all body surfaces where temperature is less than 3200°F. Ablative in hotter areas.					
ECS	5 psi O ₂ atmosphere from gaseous storage. Water boiler in EV. LiOH canister in EV. Primary O ₂ and H ₂ O in MM.			5 psi O ₂ atmosphere from gaseous storage. Water boiler and LiOH canister.		
Power	Batteries for ascent and return phasing in MM, batteries for reentry EV.		Fuel cells and reentry batteries in EV, Cryo RSS in MM.	Fuel Cells and Cryo RSS		
Avionics	One 4 gimbal IMU, one computer, FCS, unified S-band, and rendezvous radar in EV.	A 4 gimbal + strapdown IMU, dual computers, parawing control, FCS, unified S-band, and rendezvous radar in EV.		A 4 gimbal + strapdown IMU, dual computers, FCS, unified S-band, and rendezvous radar.	Dual 4 gimbal + strapdown IMU, dual computers, unified S-band, and rendezvous radar.	
Recovery	Multiple ringsails - no separate backup or abort. EV serves as abort escape capsule.	Dual sailing - one primary, one backup, skid landing gear. EV serves as abort escape capsule.		Multiple ringsails - no separate backup or abort; landing assist, skid gear. Separate abort escape capsule contains primary parachutes.		
Propulsion	Tower mounted solid motor for low altitude escape (jettisoned).					
1. Launch Escape	Orbit maneuver system for high altitude escape.			Multiple solid motors inside entry vehicle for high altitude escape of capsule.		
	Entry vehicle system for entry attitude control after high altitude escape.			Pressure fed, NTO/MMH system for entry attitude control of escape capsule after high altitude escape.		
2. Upper Stage Boost	—			—	Pump fed, O ₂ /H ₂ system. Multiple bell nozzle engines located in crew module. Multiple, strap on propellant tanks (jettisoned).	Pump fed O ₂ /H ₂ system in crew module. Multiple bell nozzle engines. Separate fuel and oxidizer tanks.
3. Orbit Maneuvers	Single pressure fed, NTO/MMH system for ascent, docking and attitude control; located in mission module. Single group of 16 engines (4 quads); external installation. Multiple solid motors for de-orbit; located in a de-orbit adapter.			Single pressure fed NTO/MMH system for all orbital maneuvers; located in crew module. Two propellant tank groups: one for de-orbit and one for all other maneuvers. Two engine groups: one of 16 engines (4 quads) for translation; external-retractable installation and another group of two, 8 engine "rings" (one redundant) for attitude control; buried installation.		
4. Entry Attitude Control	Two, pressure fed, NTO/MMH systems (one redundant); located in crew module.			Two, pressure fed, NTO/MMH propellant storage/feed systems (one redundant); located in crew module. Thrust-provided by orbital attitude control engines.		
5. Landing Assist				Multiple solid motors for vertical velocity attenuation.		

Table 6-2

BASELINE SUBSYSTEM DESCRIPTION SUMMARY LIFTING BODY VEHICLE CONFIGURATIONS

	Reuse Categories				
Subsystem	A – Modular – Expendable	B – Modular – Reusable	C – Integral Cargo	D – Tip Tanks	E & F – Integral
Structure	Skin-stringer structural shell (entry vehicle and mission module)		Skin stringer structural shell		
Thermal Protection	Ablative where temperature > 3200°F, shingles where < 3200°F.				
ECS	5 psi O ₂ atmosphere from gaseous storage. Water boiler in EV, LiOH canister in EV. Primary O ₂ and H ₂ O in M.M.		5 psi O ₂ atmosphere from gaseous storage. Water boiler and LiOH canister.		
Power	Batteries for ascent and return phasing in MM. Batteries for reentry in EV. Turbine for aero control power.	Fuel cells and reentry batteries in EV, cryo RSS in MM. Turbine for aerodynamic control power.	Fuel cells, cryo RSS, turbine for aerodynamic control power.		
Avionics	A 4 gimbal + strapdown IMU, dual computers, triply redundant FCS, unified S-band, and rendezvous radar in EV.			Dual 4 gimbal + strapdown IMU's, dual computers, triply redundant FCS, unified S-band, and rendezvous radar.	
Recovery	Multiple ring sails for abort, instant L/D landing assist, skid gear. EV serves as abort escape capsule.		Instant L/D landing assist, skid gear. Separate abort escape capsule has multiple ringsail chutes.		
Propulsion	Multiple solids strapped on adapter for low altitude escape (jettisoned).		Multiple solids strapped on escape capsule for low altitude escape (jettisoned).		
1. Launch Escape	Orbit maneuver system for high altitude escape.			Multiple solid motors inside entry vehicle for high altitude escape of capsule.	
	Entry vehicle system for entry attitude control after high altitude escape.		Pressure fed NTO/MMH system for entry attitude control after high altitude escape.		
	—		—		Pump fed O ₂ /H ₂ system. Multiple bell nozzle engines located in entry vehicle. Multiple strap on propellant tanks (jettisoned).
2. Upper Stage Boost	—		—		Pump fed, O ₂ /H ₂ system in crew module. Multiple bell nozzle engines. Common bulkhead propellant tank; integral with primary structure.
3. Orbit Maneuvers	Single, pressure fed, NTO/MMH system for ascent, docking and attitude control; located in mission module. Single group of 16 engines (4 quads); external installation. Multiple solid motors for de-orbit; located in a de-orbit adapter.		Single, pressure fed, NTO/MMH system for all orbital maneuvers; located in entry vehicle. Two propellant tank groups: one for de-orbit and one for all other maneuvers. Two engine groups: one of 16 engines (4 quads) for translation (external-retractable installation) and another group for two, 8 engine "rings" (one ring redundant) for attitude control		
4. Entry Attitude Control	Two, pressure fed, NTO/MMH systems (one redundant); located in entry vehicle.		Two, pressure fed, NTO/MMH propellant storage/feed systems (one redundant); located in entry vehicle. Thrust provided by orbital attitude control engines.		
5. Landing Assist	Multiple solid motors for glide range extension.				

failed or produce out of tolerance performance. Ease of maintenance places requirements on access provisions and equipment installation. For easy access, all equipment that does not need to be pressurized is located in external equipment compartments, accessible from outside the spacecraft. Equipment should be located so that it is accessible without removal of other equipment and so that removal and installation does not cause damage to cold plates, plumbing, and wiring harnesses. Ready access must be provided for filters, debris traps, CO₂ and odor absorbers, and batteries.

The amount of non-propulsive equipment does not vary greatly with vehicle or crew size nor does it differ greatly from that required for the logistics vehicle in a previous study (See Reference 4-1). This vehicle had about 70 sq ft of equipment access (exclusive of that required for the propulsion and recovery subsystems) in the external surface of the entry vehicle. Therefore, a constant 70 sq ft of access area was assumed as a baseline value for the non-propulsive, non-recovery equipment for all vehicles.

6.1.2 Reliability - The mission success goal required for the logistic vehicles is .95, including the entry vehicle and upper stage (insertion) propulsion system. For configurations where the entry vehicle and the upper stage are separate modules, the sub-goals were equally apportioned .975 for each module. The upper stage propulsion goal is achievable with state-of-the-art (S-IVB) equipment. The .975 goal for the entry vehicle is equally attainable or exceeded with the current hardware available and redundancy applications incorporated during the design phase (See Table 6-3).

The apportionment was made at the subsystem level to establish feasible subsystem reliability goals, and to estimate the magnitude of reliability achievement required at this level for the overall mission success reliability requirement. Basic to these apportionments are the following considerations and assumptions:

1. The apportioned reliabilities must be consistent with the state-of-the-art of the subsystems. Special consideration is given to those requiring more development such as guidance and navigation. These subsystems are accordingly apportioned values well within the present state-of-the-art.

TABLE 6-3

RELIABILITY APPORTIONMENT SUMMARY

LOGISTIC MISSION

Subsystems	Modular		Partial Integral		Integral with Tip Tanks		Fully Integral	
	Ballistic	Lifting Body	Ballistic	Lifting Body	Ballistic	Lifting Body	Ballistic	Lifting Body
a) Upper Stage Propulsion	.9750	.9750	.9750	.9750	.9750	.9750	.9750	.9750
b) ECS	.9975	.9975	.9975	.9975	.9975	.9975	.9975	.9975
c) Communications	.9982	.9982	.9982	.9982	.9982	.9982	.9982	.9982
d) Power (Electrical)	.9971	.9978	.9971	.9978	.9971	.9978	.9971	.9978
e) G&C, Radar (Electronics)	.9967	.9967	.9967	.9967	.9967	.9967	.9967	.9967
f) Reentry Control Propulsion	.9965	.9966	.9965	.9966	.9965	.9966	.9965	.9966
*g) Maneuver and Orbital Attitude Control	.9968	.9968	.9968	.9968	.9968	.9968	.9968	.9968
h) Primary Structure	.99934	.99942	.99934	.99942	.99934	.99942	.99934	.99942
i) Thermal Protection	.99956	.99968	.99956	.99968	.99956	.99968	.99956	.99968
j) Vertical Landing and Recovery	.9974	N/A	.9974	N/A	.9974	N/A	.9974	N/A
k) Instrumentation	.9974	.9943	.9973	.9973	.9973	.9973	.9973	.9973
l) Sequentials, Pyros, Docking, and Retro	.9986	.9986	.9986	.9986	.9986	.9986	.9986	.9986
m) Aerodynamic Control	N/A	.9985	N/A	.9985	N/A	.9985	N/A	.9985
n) Horizontal Landing and Recovery	N/A	.9979	N/A	.9979	N/A	.9979	N/A	.9979
R. Total	.9500	.9500						

* For design cases where the maneuver subsystem and the orbital attitude control subsystem are separate subsystems, the apportionment is: Maneuver Subsystem = .9987
Orbit Attitude Control = .9981

Escape system apportionments are not applicable for mission success.

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2. The apportionments are of mission success and do not directly reflect crew survival probability requirements. Those subsystems which are critical to crew safety are apportioned the highest reliabilities believed achievable.
3. The aerodynamic control subsystems of lifting body vehicles assumed some of the function of the maneuver propulsion and landing recovery subsystems of the ballistic vehicles.

6.2 Structure - Major structural tasks include definition of structural design criteria, requirements, structural concepts, and the baseline primary structure.

6.2.1 Structural Design Criteria - This section presents the structural design criteria which provide a basis for selection of the analytical model and other information pertinent to the structural design.

6.2.1.1 Design Environments - Atmospheric properties used in the trajectory analysis are defined in Reference 6-1. Wind environments are based on 95 percent probability wind data from ETR and WTR as applicable.

6.2.1.2 Design Factors - Design factors to determine ultimate loads, proof and burst pressures are summarized in Table 6-4. Factors of safety are those applied to limit loads to account for items such as uncertainties and variations in material properties, fabrication quality, and internal and external load distributions. A factor of safety of 1.40 is used for manned flight.

Pressurization factors are applied to the maximum operating pressure of all pressure vessels. Maximum operating pressure includes the effects of temperature, transient peaks, and hydrostatic pressure heads due to vehicle acceleration. Vessels are containers designed for sustained internal pressure such as crew compartments, propellant tanks, storage bottles, tubing, valves, etc. Excluded are structural items such as adapters, interstage bays, and fins not designed for sustained internal pressure. Proof pressure is the product of the maximum operating pressure and the proof factor. The component shall be capable of normal operation for its design service life after exposure to proof pressure. Burst pressure is the product of the maximum operating pressure and the burst factor.

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TABLE 6.4

DESIGN FACTORS

Factors of Safety

° Flight Conditions	1.40
° Ground Handling Conditions	1.50
Potentially Hazardous to Personnel	

Pressurization Factor	Proof	Burst
° Hydraulic Vessels	1.50	2.50
° Pneumatic Vessels	1.67	2.22
° Lines and Fittings	2.00	4.00
° Manned Cabins	1.33	2.00
° Main Propellant Tanks	1.1	1.4
° Rocket Motor Cases	1.1	1.4

Detailed evaluation of the sensitivity of structural temperatures to off-nominal operation analytical uncertainty is beyond the scope of this study. One approach which accounts for these effects is the use of structural temperature factors. This approach is qualitatively described as follows:

Temperature Factors - The temperature dispersion factor is intended to account for the effects on temperature due to deviations from the nominal trajectory. The temperature uncertainty factor is intended to account for uncertainties and variations in material thermal properties, ablative performance, and other variations not amenable to analytical estimates.

Predicated Temperature - Predicted temperature is, (a) the nominal trajectory calculated temperature (deg. R) multiplied by the temperature dispersion factor, or (b) the temperature determined from off-nominal trajectories.

Design Temperature - Design temperature is the predicted temperature (deg R) multiplied by the temperature uncertainty factor. Typical values for the temperature uncertainty factor range from 1.05 to 1.25 depending on the scope of the development test program, complexity of the shape, aerodynamic environment, etc.

In this study, the temperature dispersion factor and temperature uncertainty factor have been combined into one factor. The factor used was 1.07 for all radiative structure and 1.00 for ablative protected structure.

6.2.1.3 Strength Requirements - Spacecraft structure shall have strength to withstand limit load and predicted temperature without permanent deformation. In addition, the structure shall withstand applied ultimate load with predicted temperature or limit load with design temperature whichever is critical. Internal stress distributions shall be based on limit loads and the thermal stress associated with predicted temperature. The net stress distributions when thermal stresses are considered shall be multiplied by the factor of safety to obtain ultimate values.

6.2.1.4 Stiffness Requirement - Vehicle structural stiffness is provided to minimize structural coupling with the control system and to preclude local aeroelastic instabilities.

6.2.1.5 Mission Phase Requirements - The application of the structural design criteria to specific mission phases is described below. Table 6-5 summarizes the design load factors for the significant mission phases.

Handling - Modes of transport are air, ship, rail, or truck. Handling methods and environments during fabrication, delivery, and installation will be controlled to minimize imposing design requirements on the structure except locally at the support fittings.

Launch - Structural loading for all vehicles during the launch and ascent phase is based on Saturn V launch vehicle critical loading conditions. Rigid body loads calculated for these conditions are increased by 30 percent to account for the dynamic effects of aeroelasticity, buffeting, and control system response. The launch (lift-off) phase begins with main engine start and ends when the start and release transients are insignificant.

Ascent - The ascent phase is that period from lift-off to orbit insertion. Table 6-5 indicates the design load factors of these phases. During the ascent phase, pressurized compartments are vented to maintain an internal pressure differential of 6.0 psi limit relative to ambient; also, the unpressurized compartments are vented to maintain an internal pressure differential within the range of -0.5 psi limit (collapse) to 2.0 psi limit (burst) relative to ambient.

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Table 6-5

Design Load Factors at
Spacecraft Center of Gravity

Mission Phase		Load Factors*			Remarks
		Longitudinal	Lateral	Vertical	
Handling	Ground	± 2.0	$\pm 2.$	$\pm 2.$	HIMSD AFSCM 80-8 MIL-A-8421B
	Air	± 3.0	± 1.5	$\pm 3.$	
Launch		2.0	$\pm .65$	$\pm .65$	
Ascent	Max α_q	2.0	$\pm .3$	$\pm .3$	M = 1.3 $\alpha_q = 6546 \text{ psf-deg}$
	Max n_x	4.9	$\pm 1.$	$\pm 1.$	
Space Operations		$\pm .5$	0	0	
Retrograde		$-.5$	0	0	
Entry	Lifting	$-2.$	± 1.5	7.0	
	Ballistic	15.	± 2.5	± 2.5	
Approach	Lifting	$-.3$	± 1.5	7.0	
	Ballistic				
Landing	Lifting	.5	1.	5.	
	Ballistic				
Abort	Lifting	± 10	± 1.5	7.0	
	Ballistic	10	± 2.5	± 2.5	

* The load factors shown are the maximums in each direction and do not necessarily occur simultaneously.

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Space Operations - Vehicle structures are designed for all geophysical environments of space operations. Manned compartments are designed for a maximum expected operating pressure of 6.0 psi limit.

Entry - Vehicle structural design is based on the load factors in Table 6-5. The entry phase begins following retrograde and terminates after the transition is made from hypersonic to supersonic speed (Mach ≤ 5.0).

Approach - Aerodynamic control surfaces of M2/F2 lifting bodies experience critical loading during the approach phase (Mach 5.0 to landing). Vehicle structure and directional control surfaces are designed for a steady-state lateral load factor of ± 1.5 . Pitch control surface structures are designed for deflections required to attain a normal load factor of 7.0. Control surface rigid loads are increased by 25 percent to account for aeroelastic effects.

Landing - Normal landing for expendable ballistic and lifting body spacecrafts is on land or water. Normal landing for all reusable spacecraft is on land. A parachute system is used for back-up landing capability and as the primary landing mode for normal water landing on all spacecraft. Table 6-6 presents the parachute deployment and landing conditions. Landing gear coefficient of friction is 0.6 for skids and 0.1 for wheels.

Abort - Abort capability is required at any time during launch and ascent phases. Crew compartment or escape structure, for both ballistic and lifting bodies, is designed to sustain 8.0 psi limit overpressure simultaneously with a limit axial load factor of 10.0.

6.2.2 Structural Concepts - The structural airframe (fuselage) accounts for a major percent of the total structural weight and cost. Therefore, to assess the airframe structural weight and cost, several methods of construction and materials are investigated. Figure 6-1 is a schematic representation of the many combinations of structural and thermal protection concepts which are possible for meeting designated structural airframe requirements. Thermal protection concepts are discussed in Section 6.3. Both the ballistic and lifting body entry vehicle employ a double wall, cool airframe concept. All expendable cargo-propulsion modules designed to meet the ascent and orbit environments utilize a single wall concept. A single wall, hot airframe is considered beyond the scope of the study chiefly because it is categorized as advanced-state-of-the-art lacking sufficient cost data.

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Table 6-6

Parachute Deployment Conditions

Parachute	Maximum Pull-Off Angle (Deg)	Design Deployment Dynamic Pressure (PSF)
Drogue	90.0	120.0
Main	45.0	60.0

Landing Conditions

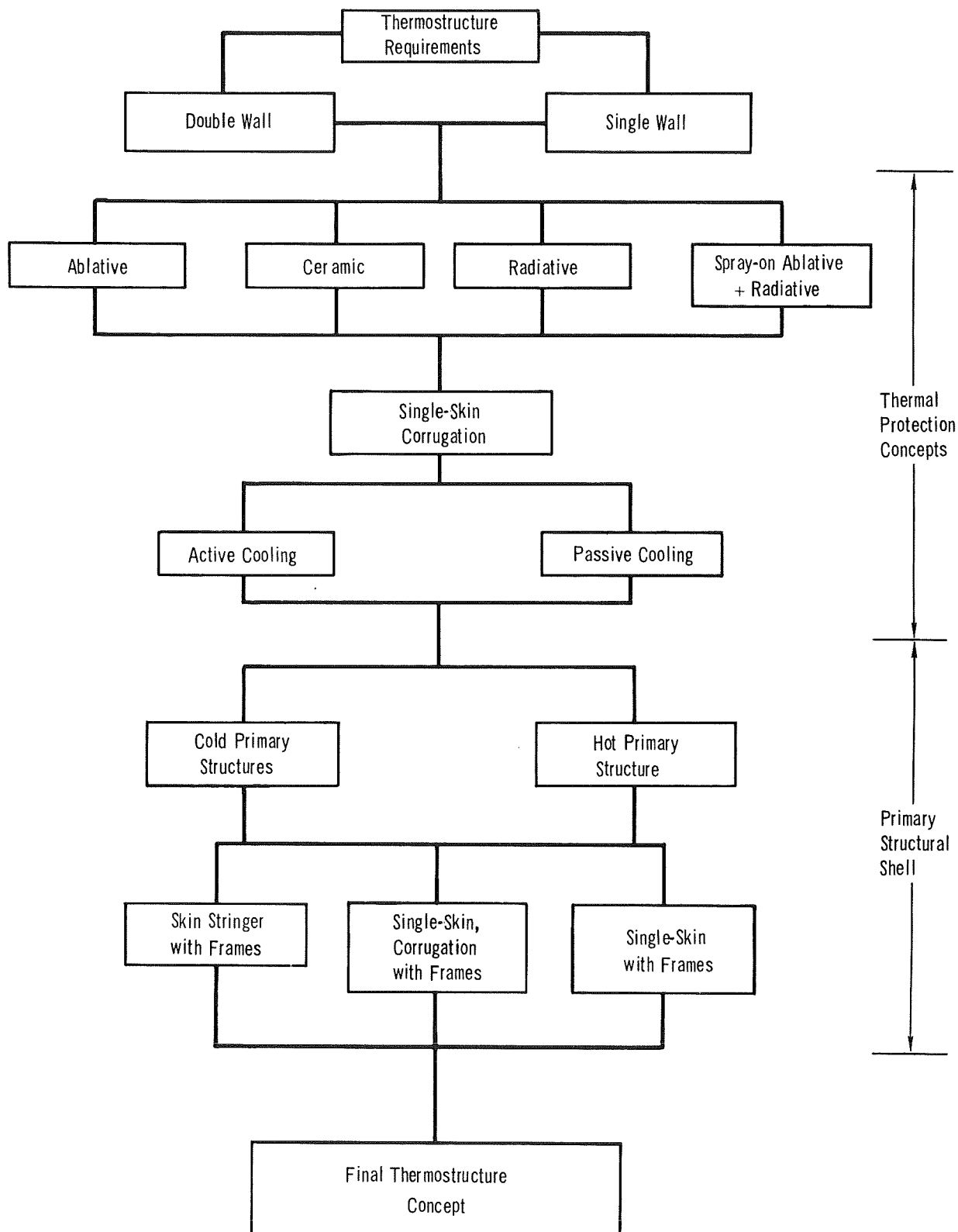
Spacecraft	Landing Surface	Maximum Sink Speed (ft/sec)	Type Landing	Remarks
Lifting Body	Land	10.0	Airplane Type Landings on Prepared Surfaces at least 10,000 ft in length	Landing Assist Propulsion will Augment S/C L/D
Lifting Body	Water (Abort only)	30.0	Parachute Land- ing in selected zones	Ringsail Para- chute.
Ballistic	Land	20.0	(Modular) Gliding Para- chute (Sailwing) in prepared landing zone (Integral) Parachute land- ing in selected zone. Landing assist rocket.	Maintain minimum L/D of about 2.1
Ballistic	Water	20.0	Parachute land- ing in selected zones	Ringsail para- chute.

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THERMOSTRUCTURE CONCEPTS

Figure 6-1



The following paragraphs discuss candidate materials and methods of construction.

Structural Materials - Evaluation and selection of candidate structural materials is based on a number of material properties and characteristics. Among these are mechanical properties, strength efficiencies, fabrication characteristics, and temperature use limits. Figures 6-2 and 6-3 summarize these properties and characteristics for several alloys within each of four material families commonly employed on aircraft, spacecraft, and missile structures.

Based on the above selection criteria, the following four alloys are considered for further evaluation:

- ° Aluminum alloy (2219-T87)
- ° Magnesium alloy (HK31A-H24)
- ° Titanium alloy (8 Al - 1 Mo - 1V)
- ° Stainless Steel (17 - 7 PH)

Figure 6-4 presents the effects of evaluated temperature on the strength and elastic modules efficiencies for the candidate materials. Mechanical properties illustrated in this figure and utilized in subsequent development of sizing models are based on minimum guaranteed values as defined in Reference 6-2.

Methods of Airframe Construction - As presented in Figure 6-1, three methods of shell construction are considered for the structural airframe: (1) Skin-stringer with frames, (2) single-skin, square corrugations with frames, and (3) single-skin with frame. This last concept is essentially a monocoque shell which is stiffened by adding circumferential frames. Figure 6-5 shows the geometry used for the three methods of shell construction. In no way, do these dimensions represent an optimum structural arrangement; however, they do represent practical sizes from a fabrication standpoint.

A parametric analysis was performed on the three methods of construction to evaluate structural weight efficiencies. The criterion utilized, for evaluating structural efficiencies, compared the structural shell unit weights (i.e., pounds per square foot) required for various load-carrying functions. Structural airframe concepts (1) and (2) were analyzed as wide beam-columns supported between frames since they were required to carry axial loads simultaneously with lateral pressure loads. Stiffened shell skins were considered to have negligible hoop tension or compression capabilities; therefore, all lateral pressure loads were beamed to the

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**CHARACTERISTICS OF CANDIDATE MATERIALS
FOR PRESSURE VESSEL STRUCTURE**

Figure 6-2

MATERIALS MATL PROPERTIES	ALUMINUM ALLOYS			MAGNESIUM ALLOYS		TITANIUM ALLOYS		STAINLESS STEELS	
	6061-T6	7178-T6	2219-T87	AZ31B-H24	HK31A-H24	6A1-4V SINGLE ANN.	8A1-1Mo-1V SINGLE ANN.	301 FULL HARD	17-7PH TH 1050
Density (Lb./In. ³)	0.098	0.102	0.102	0.0639	0.0647	0.160	0.158	0.286	0.276
Minimum Sheet Thickness Available (In.)	0.012	0.012	0.012	0.020	0.020	0.012	0.012	0.012	0.016
Minimum Face Thickness for Sandwich Const (In.)	0.010	0.010	0.010	Not Recommended	Not Recommended	0.005	0.005	0.005	0.005
Minimum Sheet Thickness for Resistance Weld (In.)	0.025	0.025	0.025	0.025	0.025	0.008	0.008	0.008	0.008
Minimum Sheet Thickness for Fusion Weld (In.)	0.032	Not Recommended	0.032	0.032	0.032	0.008	0.008	0.008	0.008
Ultimate Tensile Strength (Lb./In. ² - 10 ³)	42	83	62	39	34	134	145	185	184
Compressive Yield Strength (Lb./In. ² - 10 ³)	35	73	50	24	20	132	143	85	178
Elastic Module (Lb./In. ² - 10 ⁶)	10.1	10.5	10.8	6.5	6.5	16.4	18	26	30
UTS/Density (In. - 10 ³)	430	815	610	610	525	838	920	648	667
CYS/Density (In. - 10 ³)	257	715	490	375	319	825	905	298	645
Modulus/ Density (In. - 10 ⁶)	103	103	106	102	101	103	114	91	109

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Figure 6-3

FABRICATION CHARACTERISTICS OF CANDIDATE MATERIALS

Candidate Materials Fabrication Characteristics			Weldability		Machinability	Formability	Heat Treatment	Fabrication Cost
			Fusion	Resistance				
Low Temp Alloys (R.T. - 600°F)	Aluminum	6061-T6						
		7178-T6						
		2219-T87						
	Magnesium	AZ31B-H24						
		HK31A-H24						
Medium Temp., Alloys (R.T. - 1200°F)	Titanium	6Al-4V						
		8Al-1Mo-1V						
	Stainless Steels	301						
		17-7 PH						
	Readily	Cold	As Received		Low			
	With Some Difficulty		Stress Relieved or Annealed		Moderate			
	With Extreme Difficulty	Hot	Solution Treat and Aged		High			

ELEVATED TEMPERATURE MATERIALS STRENGTH INDICES

FIGURE 6-4a

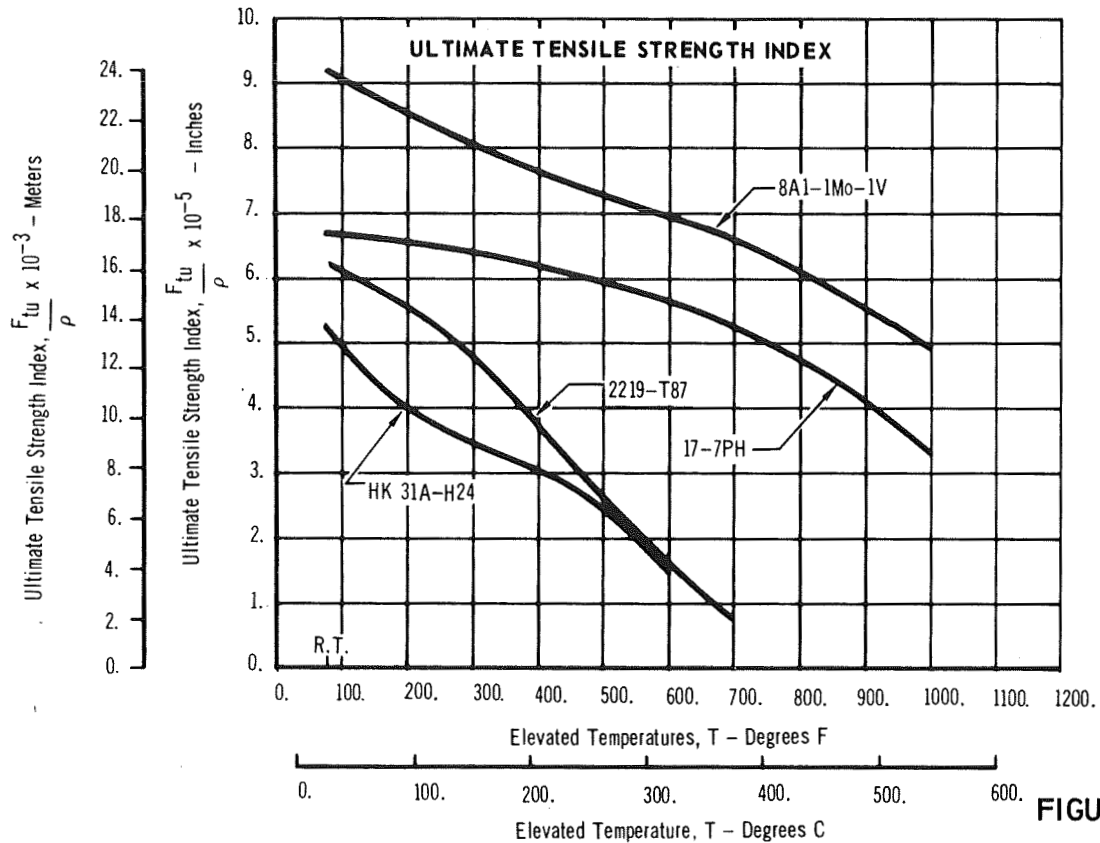


FIGURE 6-4b

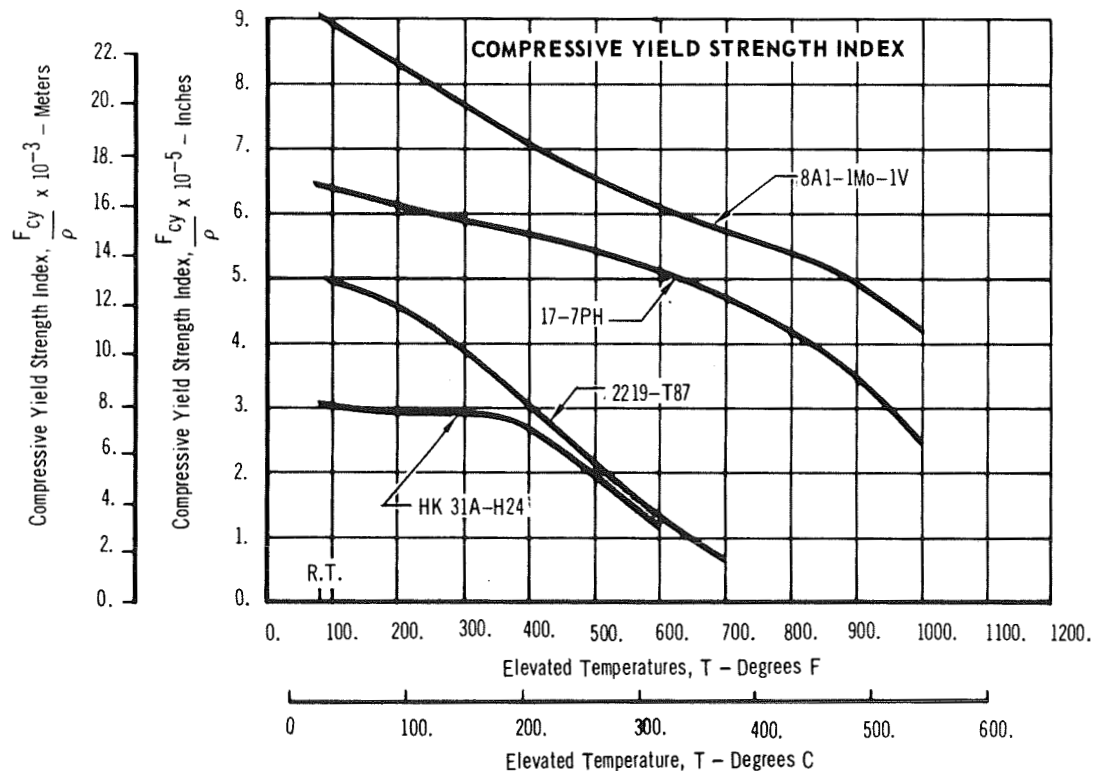


FIGURE 6-4c

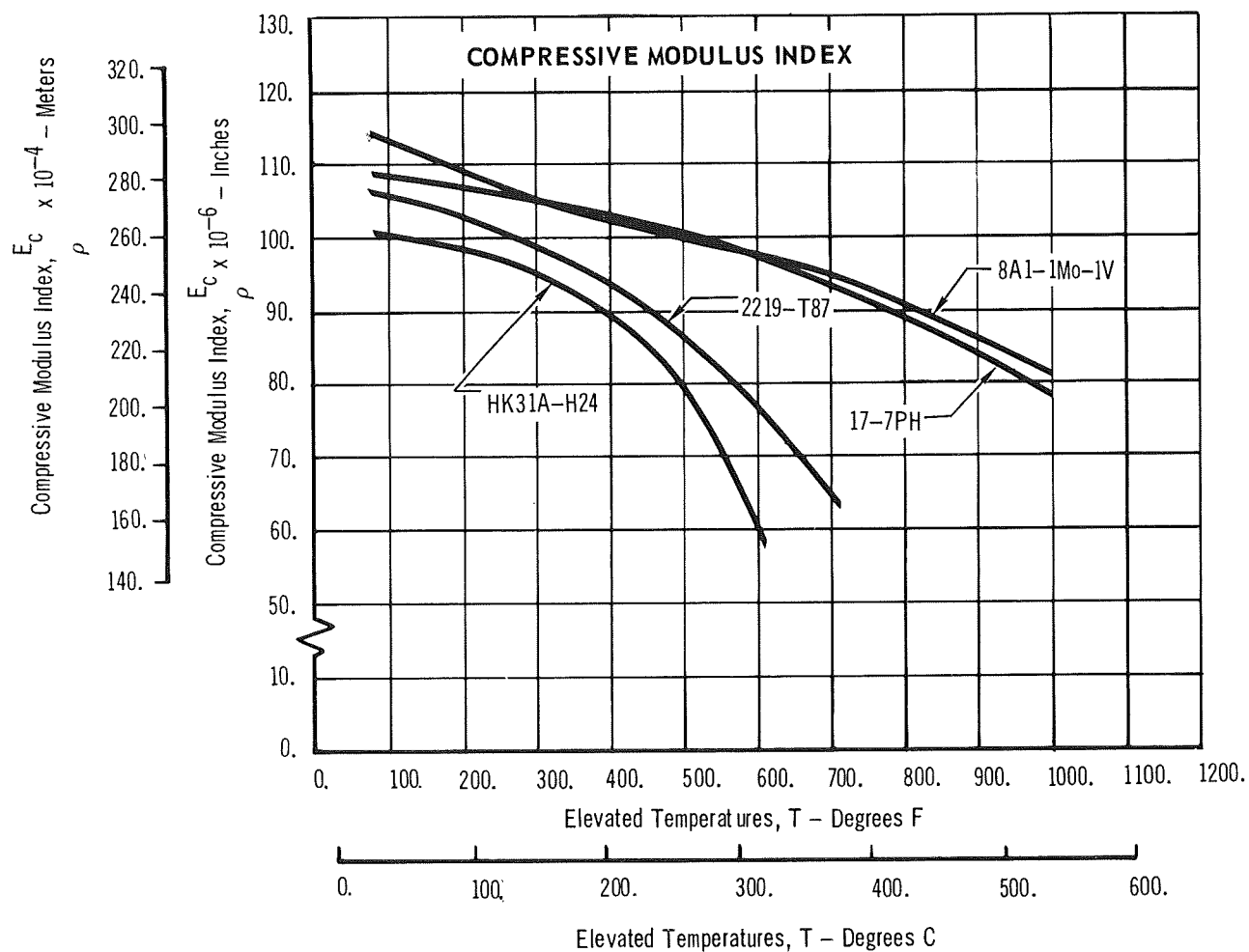
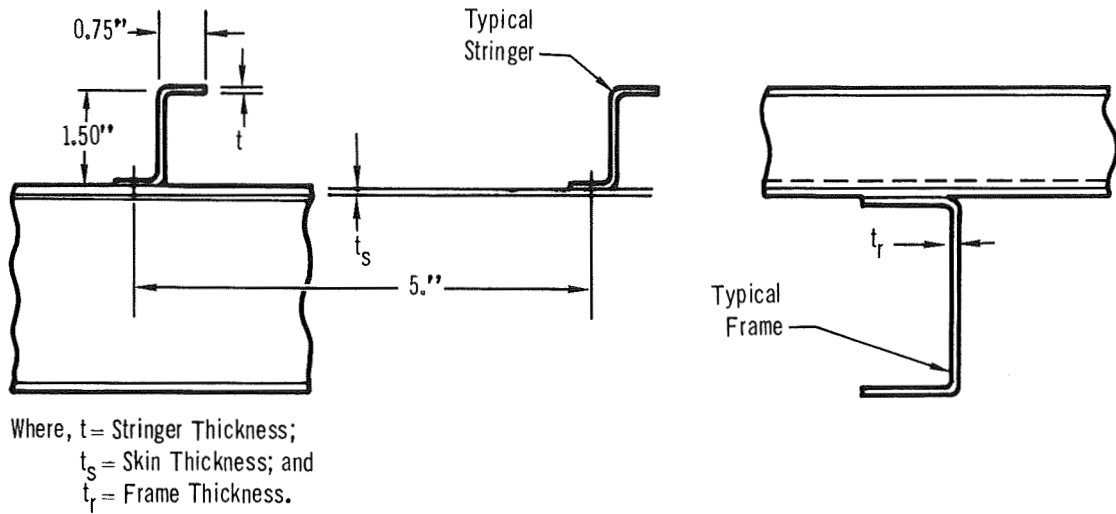


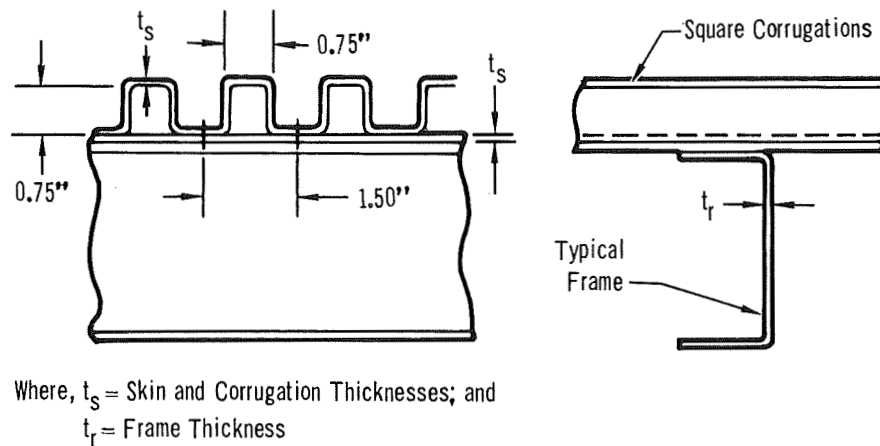
Figure 6-5

GEOMETRY OF STRUCTURAL SHELL CONCEPTS

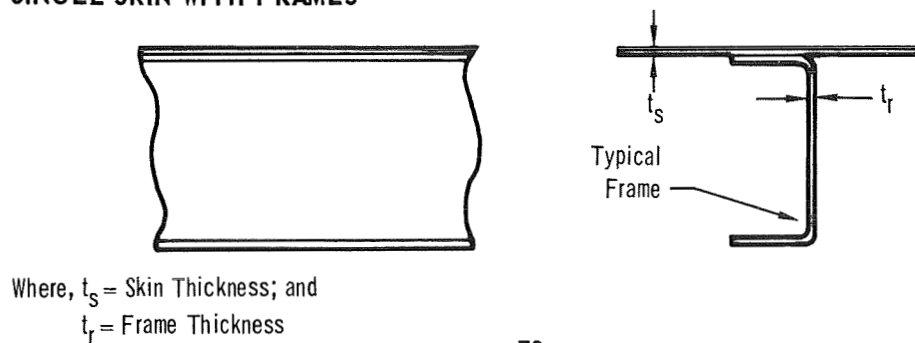
1. SHEET-STRINGER WITH FRAMES



2. SINGLE SKIN, SQUARE CORRUGATIONS WITH FRAMES



3. SINGLE SKIN WITH FRAMES



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frames. Structural shell sizing was based on instability failure analysis employing a zero margin of safety:

$$M.S. = \frac{\text{Instability Failure Load (Stress)}}{\text{Applied, Ultimate Load (Stress)}} - 1 = 0.$$

Applied, ultimate axial and bending loads on M2-F2 vehicles were determined for two critical conditions: (1) maximum airloads during ascent and (2) touchdown loads during land landings. Loads on the ballastic vehicles were determined for maximum airloads during ascent. Lateral pressure distributions associated with the ascent phase considered a maximum burst pressure of 6.0 psi limit in pressurized compartments and 2.0 psi limit in unpressurized compartments. A 1.4 factor of safety was applied to these pressures to obtain ultimate design pressures.

A third method of construction was analyzed as a frame-stiffened cylinder. Previous test results (Reference 6-3) have shown that ring-stiffened cylinders have compressive buckling coefficients approaching and exceeding the theoretical small-deflection value of 0.6 when subjected to internal pressurization.

$$\frac{F_{cr}}{E} \left(\frac{R}{t} \right) = 0.6$$

where: F_{cr} = critical buckling stress, psi,
 E = Young's modules of elasticity, psi,
 R = Radius of cylinder, inches,
 and t = Thickness of cylinder, inches.

This method was used directly for the ballistic vehicles; however, it was modified for the irregular-shaped M2-F2 vehicles. For the M2-F2 vehicles, the radius, R , is based on an assumed equivalent cylinder defined by the following expression:

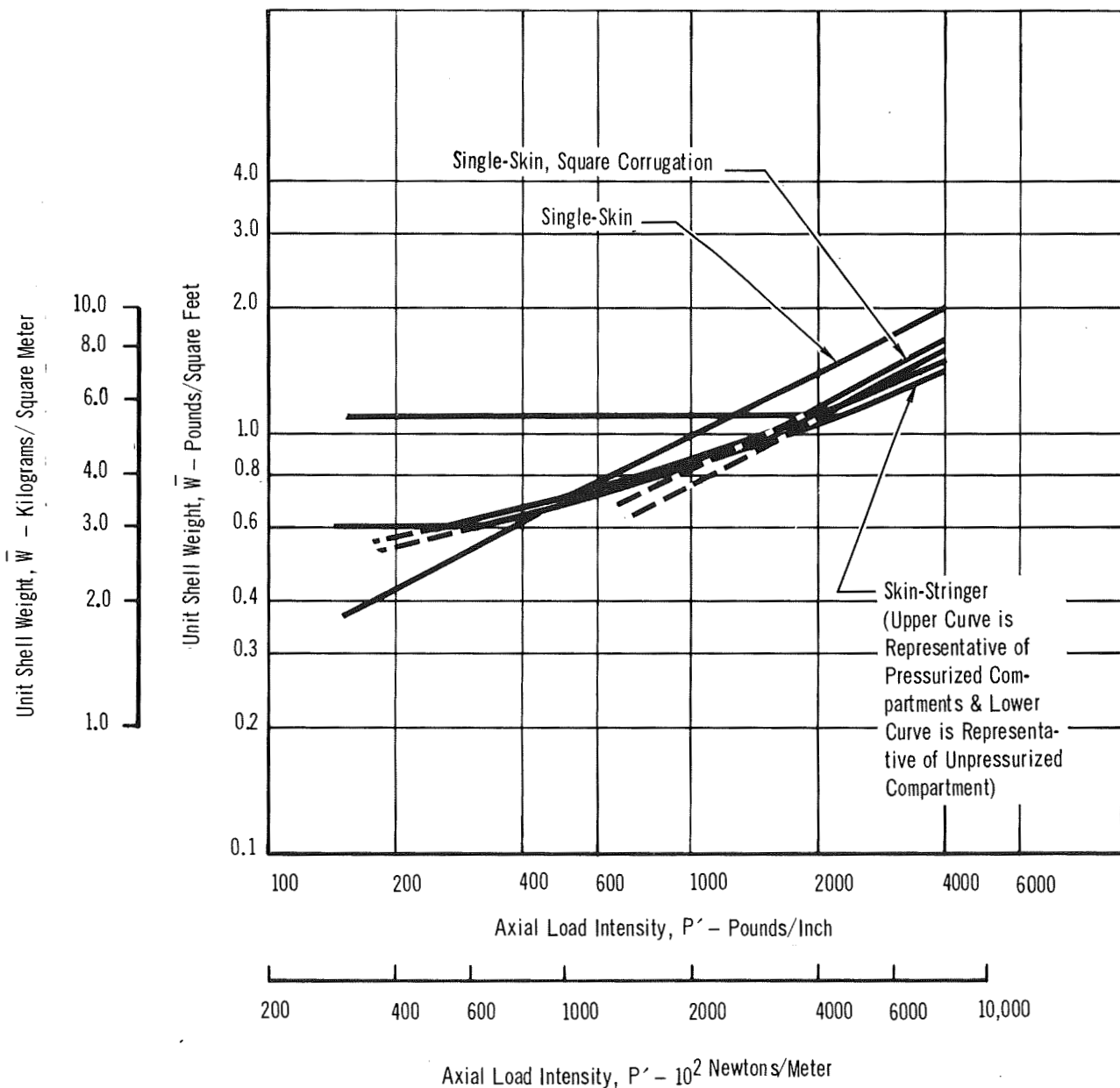
$$R = \frac{\text{Perimeter}}{2 \pi}.$$

Caution must be used when utilizing this approach since no test data exist to validate this method for the large shells considered.

Figure 6-6 summarizes the results of the parametric analysis for the aluminum alloy. For each method of shell construction, the figure shows unit shell weights for carrying a range of axial load intensities simultaneously with lateral pressure loads. The single-skin with frames concept is representative of a 60 inch diameter cylinder. Larger diameters, which are typical of larger ballistic and lifting body vehicles of this study, will result in increased unit shell weight. This

FIGURE 6-6

WEIGHT COMPARISON OF CONSTRUCTION METHODS
(ALUMINUM ALLOY 2219-T87 AT ROOM TEMPERATURE)



Note: Unit shell weights do not include the weight increments for frames.

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method of construction results in heavy structural shells but offer the least cost-per-pound of structural weight of all structural concepts investigated. However, when the structural shell is further stiffened with longitudinal stringers or corrugations a substantial weight savings is obtained which off-sets any cost advantage of the single-skin concept. All horizontal lines in the figure represent cut-offs for minimum sheet gages based on manufacturing limitations. A minimum gage of 0.025 inches is selected for aluminum and magnesium alloys and 0.010 inches is selected for titanium and stainless steel. Table 6-7 is included as a summary of the structural concepts which can be evaluated in the cost optimization program.

In the large, reusable M2-F2 vehicles containing upper stage propellant tanks and engines, a trade-off study has shown that propellant tanks integral with the structural shell and frames is more competitive from a volume and weight standpoint than a separate, supported propellant tank concept (see Section 6.4.2.2). To achieve an efficient structural arrangement for carrying primary shell loads simultaneously with the high operating pressure within the propellant tanks, the irregular-shaped frames are load relieved, with tension members placed across each frame. These tension members reduce the internal frame loads thus reducing frame cross-sectional areas. However, the advantage gained with frames is off-set by the increase in shell longitudinal stiffener size to beam the increased tank pressures. The chief advantage is the weight savings produced by the shorter vehicle length.

6.2.3 Baseline Structural Airframe - The baseline structural airframe considered for all vehicle configurations consists of a stiffened, aluminum pressure shell of thin sheet, spot welded to longitudinally-oriented stringers attached at five-inch intervals. The shell is also stabilized by circumferential aluminum frames spaced and attached at 10-inch intervals. See Figure 6-7. This sheet stringer with frames is employed in all vehicle components.

The shell structure is the primary load path for carrying aerodynamic pressures and inertia induced loads. Axial loads and bending moments from critical loading conditions induce axial load intensities around the shell circumference. These are carried by stringers and effective widths of the skin. Shear loads are reacted circumferentially by the skin. Frames stabilize the shell for internal pressure loads by redistributing beam loads from the sheet-stringers to shearing forces in the shell. They also transfer external pressure loads from the thermal protection shingles into the body shell. The baseline concept represents conventional spacecraft structural design, producing efficient airframe design at competitive cost.

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Table 6-7

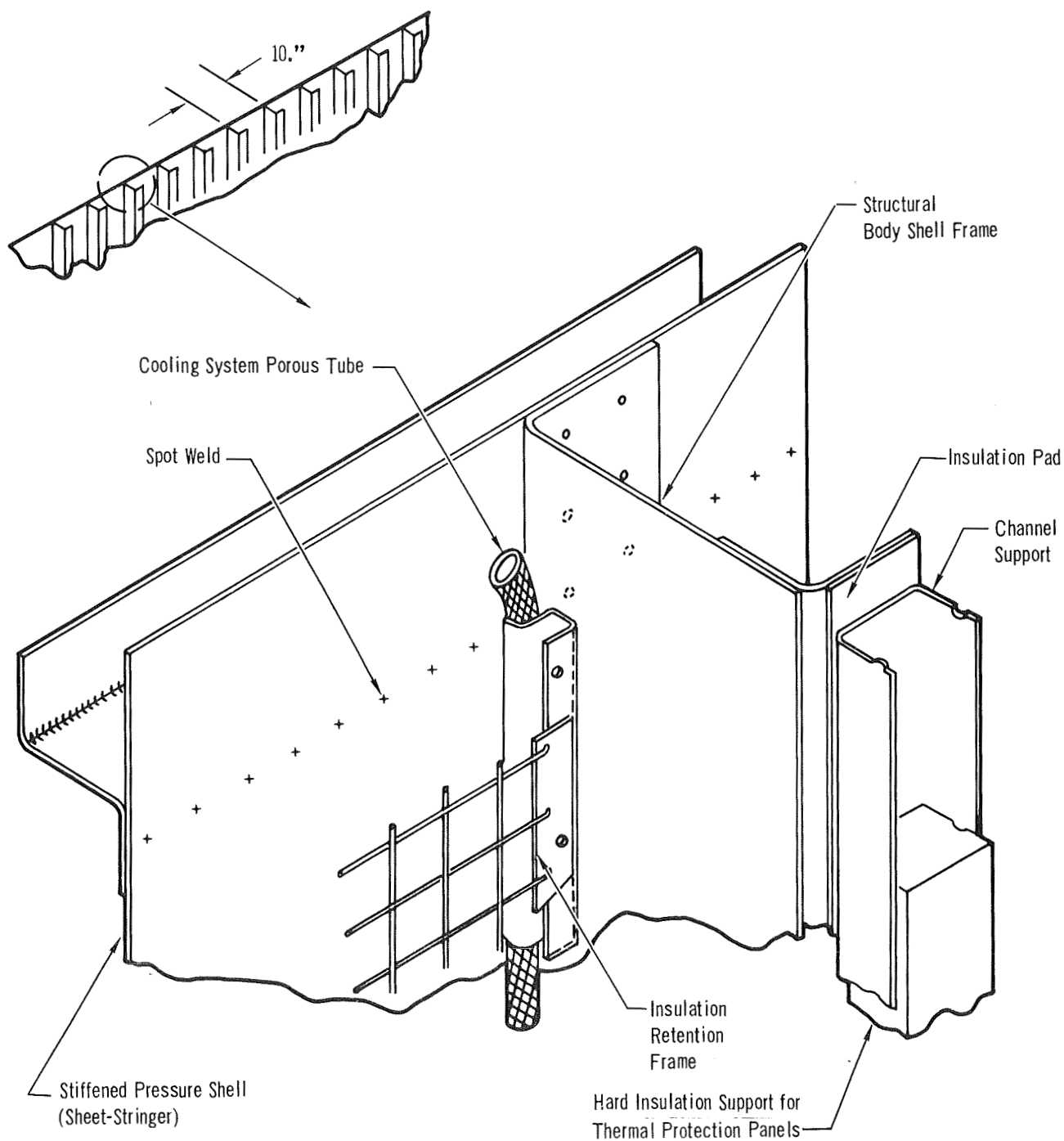
Primary Structure Concept Summary

Concept Designation	Concept Definition		Vehicle Application
	Methods of Construction	Material	
PS-1	Single-skin with frames	Aluminum	All 
PS-2*	Sheet-stringer with frames	Aluminum	
PS-3	Single-skin, corrugations with frames	Aluminum	
PS-4	Single-skin with frames	Magnesium	
PS-5	Sheet-stringer with frames	Magnesium	
PS-6	Single-skin, corrugation with frames	Magnesium	
PS-7	Single-skin with frames	Titanium	
PS-8	Sheet-stringer with frames	Titanium	
PS-9	Single-skin, corrugations with frames	Titanium	
PS-10	Single-skin with frames	Steel	
PS-11	Sheet-stringer with frames	Steel	
PS-12	Single-skin, corrugations with frames	Steel	

* Baseline Concept PS = primary structure

Figure 6-7

BASELINE AIRFRAME CONCEPT



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The structural reliability of a vehicle is difficult to assess, partly because of the basically passive nature of the structure and partly because of the usual design philosophy (i.e., select a factor of safety to be used in the design, test, and redesign should a failure occur). However, the structure has been assigned a value in the apportionment (Table 6-3); all the structural alternates are assumed to have equal reliability when designed to the same factor of safety.

However, selection of the factor of safety determines the amount of qualification testing required and also influences member sizing. This relationship introduces a trade-study in which increased hardware costs resulting from a higher factor of safety can be compared to a corresponding cost reduction in qualification testing.

Sizing of structural members for the baseline airframe for all three methods of construction uses a factor of safety of 1.4. This factor of safety is representative of the factor specified in current criteria documents for manned spacecraft. Structural capability is demonstrated by qualification testing in which test results are compared to critical design conditions specified by the structural design criteria.

Two additional approaches can be taken in manned spacecraft design to reduce qualification testing costs. These are: 1. no qualification testing, and 2. limit load testing only of the first flight article. The former permits elimination of all costs associated with structural qualification testing and the latter eliminates the cost associated with a structural static test article. These approaches to reduce qualification test costs are the accepted practice used by the McDonnell Douglas Aircraft Company. Flight testing of prototype aircraft is begun upon completion of limit load testing. Flight loads are restricted to 75% of limit load tests. (Equivalent to 50% of design ultimate strength.) Using this approach of limit load testing and no static test article is equivalent to using a design factor of safety of $\frac{1.4}{.75} = 1.85$. When no qualification testing is

specified, an additional 50% safety factor is applied. This results in a design factor of safety of $1.85 (1.5) = 2.7$. For this study a design factor of safety of 3.0 was conservatively used. Table 6-8 shows the increased factor of safety required for these approaches and the attendant increase in structural weight.

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Table 6-8

Safety Factor vs Structural Testing

Design Factor of Safety	Required Testing	Weight Factor
1.4	Total qualification program	1.0
1.85	Limit load test. No static test article.	1.15
3.0	No qualification testing required	1.60

6.2.4 Primary Structure Sizing Model - The sizing model for the primary structure sizes both the innerbody shell and rings. The shell can be sized for three distinct methods of construction; both the shell and the rings can be sized for four materials. In the integral vehicles, the methods of construction and materials can be varied between the compartments for the crew and for the cargo and propulsion. These items are sized within the program for various sections of the spacecraft, based upon the critical loading conditions for that section. Two loading conditions, maximum air loads during launch and touchdown landing, are checked. Both maximum air load condition (with inertia relief) and landing loads are a function of the mass distribution of the vehicle (i.e., location and weight of cargo, propulsion systems, crew etc.). Interactions are handled by an iterative process.

In addition, incremental weight penalties are added to the structure weight to account for hatches, doors, windows, floors, shelves, walls, bulkheads, control surfaces, and fuel tanks. These items are analyzed by using semi-empirical methods correlated with existing hardware.

6.3 Thermal Protection System - The thermal protection system maintains acceptable spacecraft temperatures to prevent structural failures and to prevent overheating of crew and equipment. Items that are primarily heat protection are difficult to separate from those that are primarily load carrying structure. For this study, the entry vehicle thermal protection is defined as the outer surface (metal panel or ablator), active wall cooling system and the insulation. The other structure such as rings, stringers, and inner load carrying walls are defined as structure and discussed in Section 6.2. The thermal protection system of the mission module is defined as the insulation and ablator needed to maintain temperature control. The space radiator on the

mission module, defined as part of the Environmental Control and Life Support system, is discussed in Section 6.5.

6.3.1 Thermal Protection Requirements - Thermal protection needs depend on mission and spacecraft shape. This study has assumed one mission - logistic supply of a space station in a near earth orbit as discussed in Section 4. The spacecraft is defined to consist of the entry vehicle and the mission module. Three basic entry vehicle shapes, as discussed in Section 5.1, are considered - a 40 deg cone ballistic, a 60 deg cone ballistic and a 30 deg half cone (M2-F2 lifting body). Both the entry vehicle and mission module must be thermally protected during all phases of a mission - i.e., launch (includes a normal launch and an abort launch), orbit, and entry.

Consistent with the ground rules, only systems which have a major impact on cost are included. Experience gained from the Gemini program and Reference 4-1 and Reference 4-3 studies has shown (a) the thermal protection system of the entry vehicle is a major cost item and is designed by entry mission phase, (b) the mission module structure is designed by boost loads, and the cost of the mission module thermal protection system is a small portion of the total mission module cost. These are discussed below by mission phase.

Launch - The heating environment during a nominal launch is less severe than during entry because high altitudes are reached before the vehicle is accelerated to high speeds. The mission module is designed by maximum launch loads that occur early in the flight when temperatures are low (about 200 deg F). Thus, for a nominal launch, the thermal environment has only a secondary or localized effect on the design.

A launch abort can occur due to an exploding booster either on the pad or at altitude. Studies conducted by MDAC for NASA MSC indicate that, due to parachute temperature limitations and entry vehicle overpressure requirements, the emergency escape system must be designed to avoid the fireball. Thus, an exploding booster has no effect on the thermal protection system.

High altitude abort can cause heating rates that are more severe than during entry. However, the severity of the abort thermal environment can be minimized by tailoring the boost trajectory. After a high altitude abort, the entry vehicle is not reusable. Thus, the entry vehicle which is designed for reuse has an inherent structural margin that can be used during a high altitude abort where even localized failures are acceptable, as long as the

crew is safely recovered. Therefore, the possible small effects of high altitude abort on the thermal protection system are not considered.

Orbit - Orbital temperature control is primarily accomplished by the Environmental Control System which collects, distributes, and rejects the spacecraft heat load. If the spacecraft is "shut-down" and stored at the station before returning to Earth, thermal control is more difficult; and can affect the spacecraft construction because the heat leak through the wall must be minimized. This can be done by using more or better insulation and special coatings (paints) on various surfaces. Both techniques are used to control temperatures during entry. Using these techniques for orbit and entry thermal control differs primarily by degree. If the orbital requirement were eliminated, the heat protection system would not be changed very much. The small incremental costs that result from orbital conditions are neglected in this study.

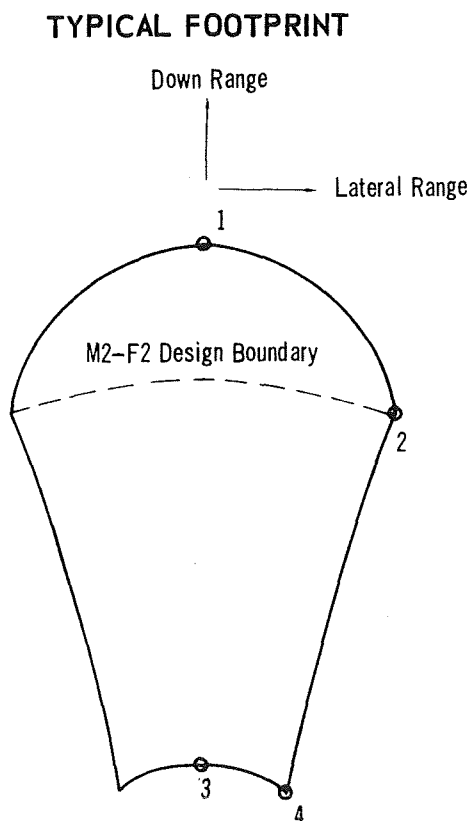
Entry - The thermal control requirements during entry strongly influence entry vehicle design. The maximum temperatures determine the type of outer surface material. The thermal protection system design must also consider the large thermal gradients that exist in the structure because of non-uniform heating rates and the need to keep the "inside" at fairly low temperatures.

An important factor influencing the entry thermal protection system is the spacecraft flight path angle when entering the atmosphere (γ_E). This angle is primarily a function of the orbit altitude and the de-orbit velocity increment used (i.e., retrograde system). The range of conditions being investigated results in a design range of from -1 deg to -2.5 deg for γ_E assuming an initial entry velocity of 25,900 ft/sec. This range is more than adequate for orbital altitude of up to 300 na mi. Another important factor is the entry mode (i.e., how the spacecraft is flown in terms of angle-of-attack and bank angle).

The thermal protection system will be designed for a wide range of entries that can best be explained by referring to a typical footprint as shown in Figure 6-8. The highest total heat load is produced by a maximum downrange entry (point number 1 in Figure 6-8). The maximum downrange entry starts at a shallow γ_E near the skip boundary and the spacecraft is flown at an angle-of-attack that results in maximum L/D with no bank angle. However, because this entry produces more range than is useful for the M2-F2 shape, it is an unreal-

istic design condition for the M2-F2. Thus, the M2-F2 shape will be designed for the heat load resulting from a maximum L/D entry with a 45 deg bank angle (point number 2). This entry produces about the maximum possible lateral range, and yet still produces sufficient downrange capability. The other design conditions are the maximum heating rates that usually occur during minimum range entries - point number 3 which is a zero lift entry, and point number 4, which is a maximum lift entry with a 45 deg bank angle. The ballistic shape will be designed for points number 1 and 3. The zero lift flight mode for the ballistic shape assumes an angle-of-attack that corresponds to maximum L/D conditions and

Figure 6-8



uses bank angle modulation to achieve a zero lift entry. The M2-F2 shape will be designed for points number 2 and 4.

6.3.2 Thermal Protection Concepts - Of all the possible heat protection concepts available, there are only a few that are significant in this study - ablative and radiative shingle/insulation concepts. Many heat protection concepts apply only to small areas of the vehicle or are beyond current state-of-the-art.

For example, transpiration cooling (i.e., injection of a fluid or gas into the vehicle boundary layer during entry) is now limited to use on small nose tips. Experience indicates that application of a transpiration system to large surface areas would require a significant development program.

Major problems with this heat protection concept are:

- a. Obtaining a known, predictable flow rate through the porous media
- b. Determining downstream effects of the injected substance
- c. Preventing clogging of the porous media

All of these become more severe as surface area increases. Also, as the area increases, it is more difficult to supply the injected substance to the backside of the porous media at the rate needed.

The magnitude of the development program needed to apply transpiration to large areas is unpredictable because of the various unknowns. Thus, transpiration cooling systems used in the near future will probably be limited to small surface areas and are of little significance for this study. Similar arguments apply to heat protection concepts such as:

- a. Ceramic Tiles - Thermal shock problems
- b. Metallic Ablators - Shape change and downstream effects

The most important heat protection concepts for this study are radiative shingle/insulation systems and non-metallic ablators. The radiative shingle/insulation system usually consists of thin metal panels that are attached to a structural shell. Between these panels and the structural shell are connecting structure and insulation blankets. These are arranged to give the required strength and the minimum heat flow. Since the heat flow through the wall is minimized, the radiative shingles become hot and radiate most of the heat entry load to space. The limiting factor is the maximum allowable temperature of the

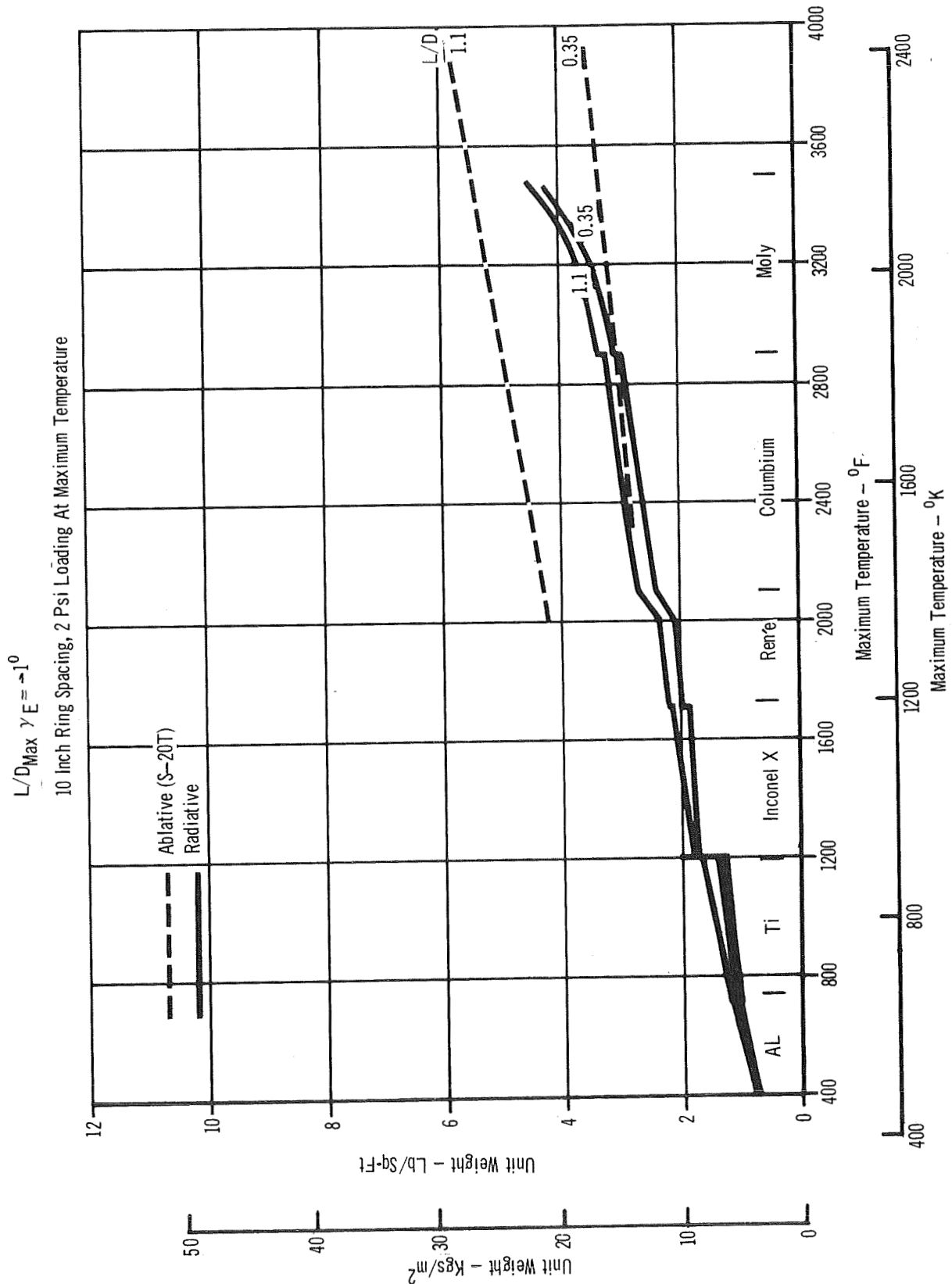
panel. This temperature is a function of material, and the number of reuses allowed. The ablative system consists of an ablator bonded to a fiberglass or metal structure that carries much of the structural load. For the heating environment expected for a logistic mission, the radiative shingle system is designed primarily by the maximum heating rate; the ablative system is designed by the total heat load.

Choices between these two systems have historically been based on a weight comparison. Usually, the radiative shingle concept is lightest until maximum temperatures expected get near the maximum allowable temperature of the selected metal. The typical weight comparison of the two concepts is shown in Figure 6-9. This chart is based on work done in Reference 4-3 study, with no change in the assumed state-of-the art, and includes shingles or ablator plus insulation and backup structure. The shingle weights assume a single skin, opened-faced, corrugated sandwich construction. This construction is discussed in more detail in Section 6.2. Only one ablator is shown - a McDonnell Douglas developed low-density ablator, the S-20T. The ablator weights assume a bondline temperature of 800 deg F maximum. These weights are based on maximum L/D flight with no bank angle and depend on vehicle shape, as transition from laminar to turbulent flow is included. No reuse capability is assumed.

There are many ablators which could be used, with the selection ranging from the flight qualified Gemini and Apollo materials to new nonqualified products. Recent advances in technology have shown the potential advantages of using a new, relatively low-density ablator for protection from thermal environments expected for a logistic mission. The S-20T mentioned above is the latest in a long series of such materials being developed by McDonnell. This material is representative of the many low-density ablators now being developed throughout the aerospace industry. It was compared with flight-qualified ablators for possible use. Ablator performance was compared using an empirical heating parameter which has been successfully used to correlate test data: $(\text{total heat})^{1/8} \times (\text{heating time})^{3/8}$, Reference 6-5. The weight advantage of the low density ablator is indicated in Figure 6-10. A comparison of material formulations and manufacturing techniques for the McDonnell Douglas "S" series of ablators indicates little manufacturing differences.

Figure 6-9

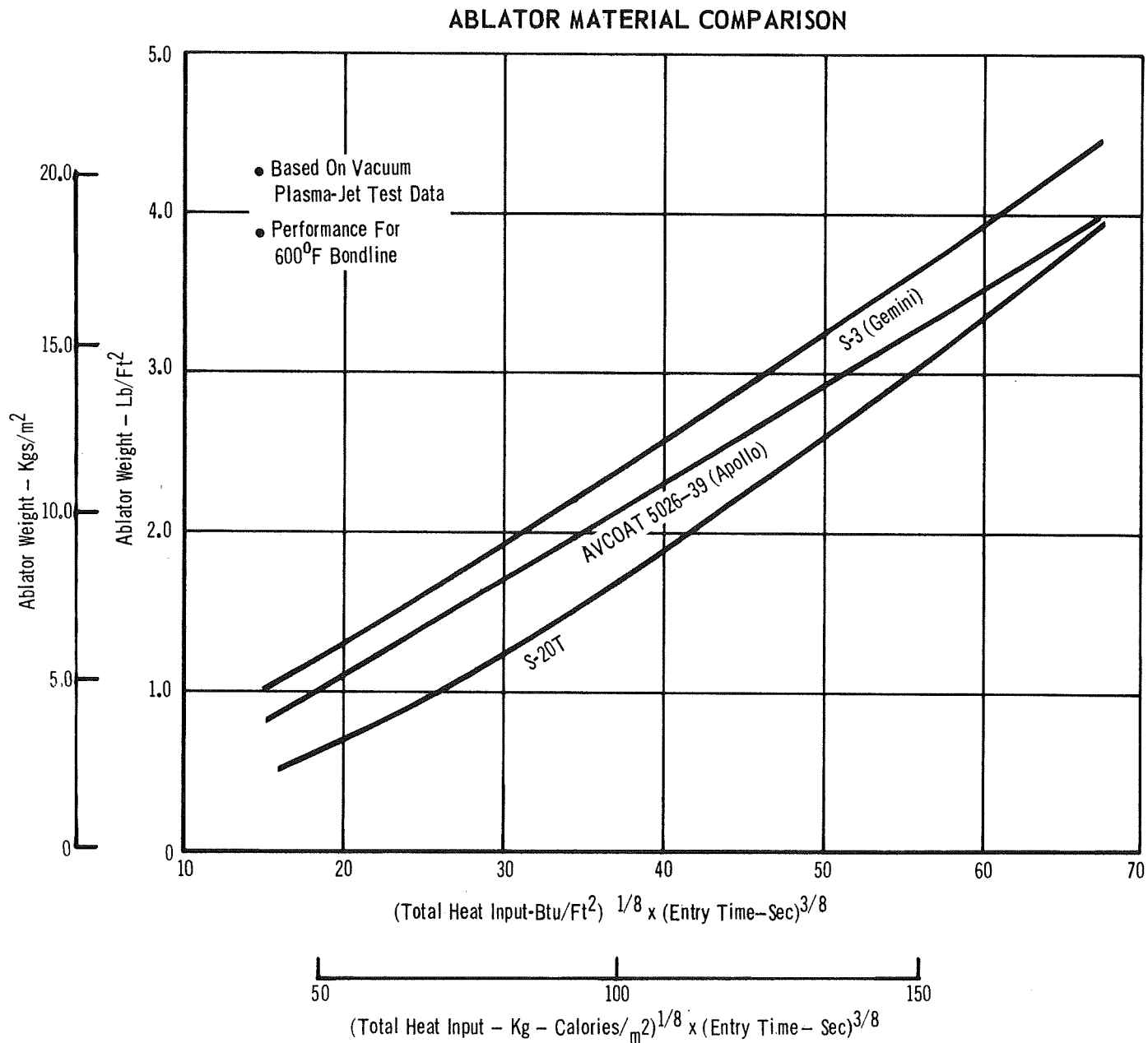
TYPICAL WEIGHTS OF HEAT PROTECTION SYSTEMS



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Figure 6-10



The major factor in cost is the curing method. Many of the "S" series materials require an autoclave to cure at the proper pressure and temperature. However, because S-20T will cure at room conditions, it is the cheapest of the "S" series materials. The performance of the "S" materials is severely reduced if significant surface recession occurs. However, the thermal environment expected for logistic spacecraft will not be severe enough to cause much surface recession. Since it is both the lightest and cheapest "S" series ablator, only the S-20T ablator will be considered in this study.

The thermal heat protection system not only limits the structural shell temperature to a value that is consistent with strength requirements, it also must maintain thermal control of crew and equipment. The cabin can be protected by passive means, such as insulation, or actively by a "coolant" loop. In the passive technique insulation is used to provide the temperature drop between the outside of the spacecraft and the cabin wall. But passive techniques are heavier and need more volume than does a combination of passive and active systems. Because the all-active approach is quite heavy and is impractical for this logistic mission, only passive and passive/active concepts are considered. A relative comparison of active and passive systems is shown in Figure 6-11. There are many "active" concepts that could be considered. All approaches intercept the heat before it reaches the inside of the cabin and dissipate it by a material phase change (usually boiling). Three basic methods are compared in Table 6-9. The open-loop porous tube concept is chosen for this study, as it is the lightest approach that is still refurbishable. This concept has been demonstrated by ground tests at McDonnell Douglas but has not yet been verified in a complete "system" test.

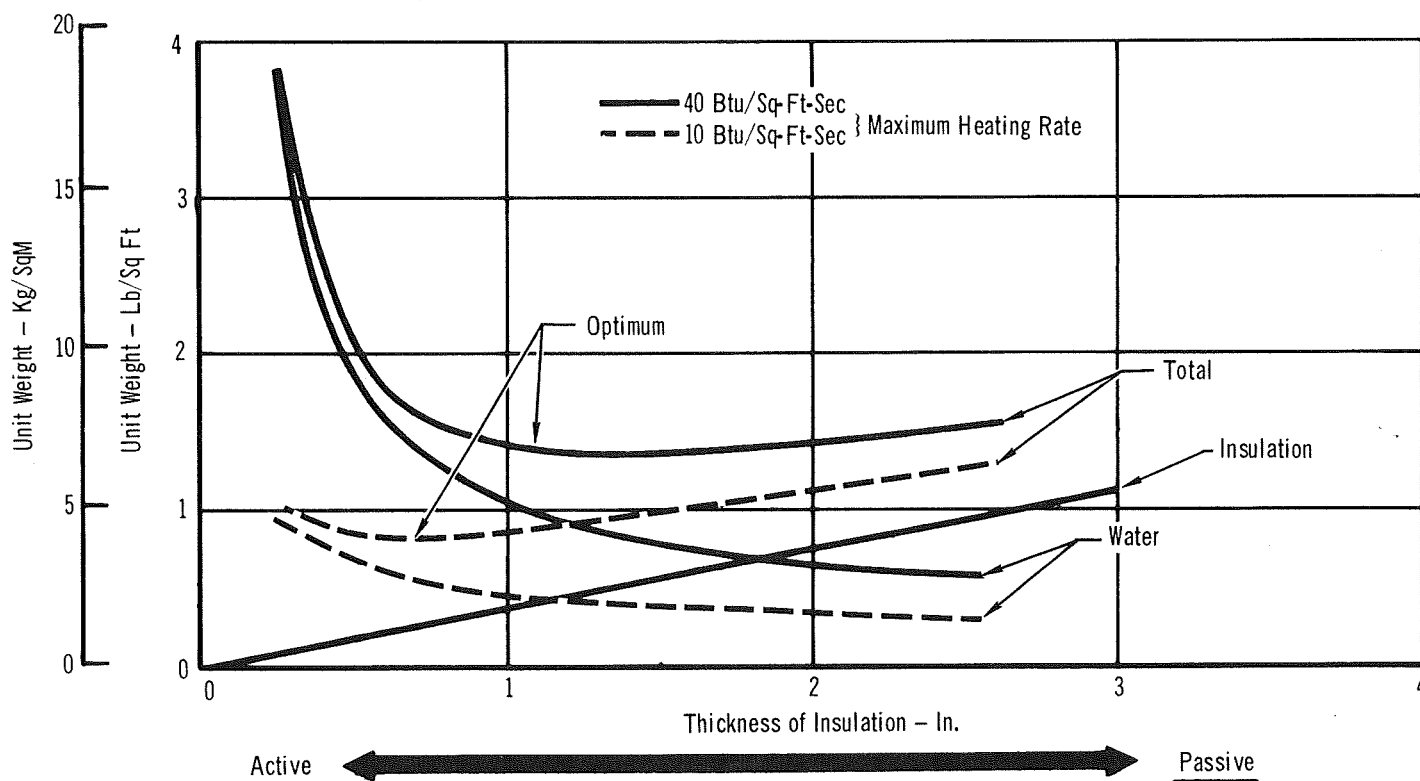
The major trade-off to be considered for thermal protection systems is whether to use ablators, radiative shingles, or some of each. The critical parameter is the maximum allowable shingle temperature. Above these temperatures, ablators will be used for each alternate (allowable temperature) case. The use of a passive inner shell cooling system instead of a combination passive/active system will also be considered. The alternates are named in Table 6-10. Insulation densities, ablator bondline temperatures, and structural temperatures will not be varied in this study.

Figure 6-11

INSULATION - WATER WEIGHT TRADE-OFF

$L/D = 1.1$ SPACECRAFT

Dyna-Quartz Insulation (4.5 Lb/Cu-Ft)



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Table 6-9

COMPARISON OF INTERNAL STRUCTURAL COOLING SYSTEMS

Type of System Feature	Water Blanket	Closed Loop	Open Loop
1. Operation	- o Entire surface covered with blankets - o Heat absorbed as water evaporates - o Self regulation	- o Heat conducted to tubes of closed loop - o Coolant transports heat to evaporator - o Pump required	- o Heat conducted to porous tubes - o Heat absorbed as water sublimates or evaporates in tubes - o Self regulation
2. Relative Weight	.8	1.6	1.0
3. Orbital Storage	- o Freezing may be problem - o Water loss in space - o Micrometeoroid puncture	- o Potential temperature problem, or - o must be filled prior to descent	- o No temperature problem - o Relatively simple to fill tubes
4. Development Status	- o Numerous ground tests - o Similar to Apollo and Gemini evaporators	- o Limited ground testing - o Similar to Gemini and Apollo coolant loops	- o Little ground testing - o Similar to Saturn and LEM sublimators
5. Refurbishment Requirements	- o Remove external heat protection - o Replace with new blankets	- o Check system continuity - o Refill with expendable coolant	- o May require heat protection removal for inspection - o Refill with water - o Extreme cleanliness required
6. Outstanding Features	- o Performance - o Weight - o Development	- o Performance - o Development - o Refurbishment	- o Performance - o Weight - o Orbital storage

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Table 6-10
Thermal Protection System Concepts Summary

Concept No.	Maximum Allowable Temperature -°F	Inner Shell Cooling	Reuse	Vehicle Application
TPS-1	400	Passive	Yes	All
TPS-2	1200	Passive	Yes	All
TPS-3	1600	Passive	Yes	All
TPS-4	2400	Passive	Yes	All
TPS-5*	3100	Passive	Yes	All
TPS-6	3500	Passive	No	All
TPS-7	400	Passive/Active	Yes	All
TPS-8	1200	Passive/Active	Yes	All
TPS-9	1600	Passive/Active	Yes	All
TPS-10	2400	Passive/Active	Yes	All
TPS-11 **	3100	Passive/Active	Yes	All
TPS-12	3500	Passive/Active	No	All

* Baseline Concept for ballistic vehicles

** Baseline Concept for Lifting Body Vehicles

6.3.3 Baseline Thermal Protection System - There are two basic thermal protection baselines - one for the ballistic shapes and one for the M2-F2 lifting body. The baseline definition includes the heating environment/distribution about the vehicle and the materials used to protect the vehicle. The heating environment expected for a logistic mission was discussed in Section 6.3.1 in terms of mission parameters. However, spacecraft shape also affects the heating environment.

Heating rate distributions about the ballistic shapes are based on test data from the Gemini and Apollo programs. The heating rate distribution assumed on the ballistic blunt heatshield is shown in Figure 6-12 and 6-13. Both figures assume a heatshield radius of curvature to base diameter ration of 1.6 - the same ratio as Gemini. The heating distribution on the conical section is shown in Figures 6-14 and 6-15. These data assume laminar flow on all surfaces. Gemini flight data indicated transition from laminar to turbulent flow did occur on the Gemini conical section. However, the resulting turbulent heating was less severe than that predicted using the laminar wind tunnel data of Figures

6-12 through 6-15. Preliminary Apollo flight data also indicate turbulent flow that was no hotter than that predicted from test data. Because the prediction of transition on ballistic shapes is extremely uncertain, and since the resulting effect on ballistic thermal protection may not be of major importance, ballistic shapes are assumed to experience only laminar flow.

The heating rate distribution about most of the M2-F2 lifting body is based on sharp cone heat transfer theory. This approach is conservative as the test data from Reference 6-6 indicates that blunt body theory more nearly correlates the data. The sharp cone predictions are based on work done by Hankey and Neumann (Reference 6-7). The leeward surface heating was predicted from an empirical correlation of ASSET wind tunnel and flight data for separated flow. The stagnation point heating is estimated from the Detra, Kemp, and Riddle empirical equation. Heating rates on the fins and flaps were based on McDonnell Douglas test data that agrees with the data in Reference 6-6. The actual equations are discussed in connection with the sizing model that is part of the overall optimization model description (Volume III).

The structural arrangement on which the thermal protection system is based is shown in Figure 6-16. The concept as shown assumed a combination passive/active cooling system for the inner structural shell. The straight passive system is similar but does not have the cooling water passages. Several important features of the system are:

- a. Hard Insulation - The hard, load carrying insulation is Min K 1301. Density = 20 lb/ft^3
- b. Fibrous Insulation - The non-load carrying insulation is Refrasil A-100. Density = 4 lb/ft^3
- c. 800 deg F Bondline - The ablator is sized to limit the bondline to a maximum temperature of 800 deg F. This is the maximum allowable temperature for current state-of-the-art adhesives. Increasing this temperature will reduce ablator weights but also will increase structural weights. Thus, the slight decrease in total weight that results is not considered worth the expense of developing a new adhesive.

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- d. 200 deg. F Inner Shell - The cooling system (passive or passive/active) for the inner body structural shell will limit the maximum shell temperature to 200 deg F and to 2400 deg F for the fixed fin primary structure. This is a localized maximum value and the area average maximum will be lower.

Both baseline thermal protection systems have a maximum allowable temperature of 3100 deg F.

In determining the reliability of the thermal protection system all materials are assumed to be of equal reliability, and can function within the temperature regions of the spacecraft. The system reliability is then dependent upon system complexity, re: number of working components (such as pumps and regulators in an active system) and the number of joints, fasteners, and fitting included in the design. It is estimated that the minimum design reliability (i.e., the most complex system) exceeds the maximum reliability requirement by at least one order of magnitude.

Radiative Protection

Cooling System Concept	Estimated Reliability
Passive	.999999
Passive/Active	.999985

6.3.4 Thermal Protection Sizing Model - The sizing model can estimate either a radiative or an ablative system backed up by insulation and water. In the higher temperature regime, a choice of molybdenum or columbium for one or five reuses is available. The inputs are generated by a simplified entry profile for the vehicle, based upon the initial entry angle, vehicle L/D, and W/S. The routine then computes velocity, density, and altitude for the time increments along the trajectory. The temperature routine uses Hankey-Neumann modified methods and a correlated Eckerts Reference Enthalpy method. This routine computes heating rates, total heat, and temperature for up to 27 points on the vehicle. The thermal protection routine converts this input to weight.

Figure 6-12

CONVECTIVE HEATING ON BLUNT HEAT SHIELD
(ANGLE OF ATTACK EFFECTS)

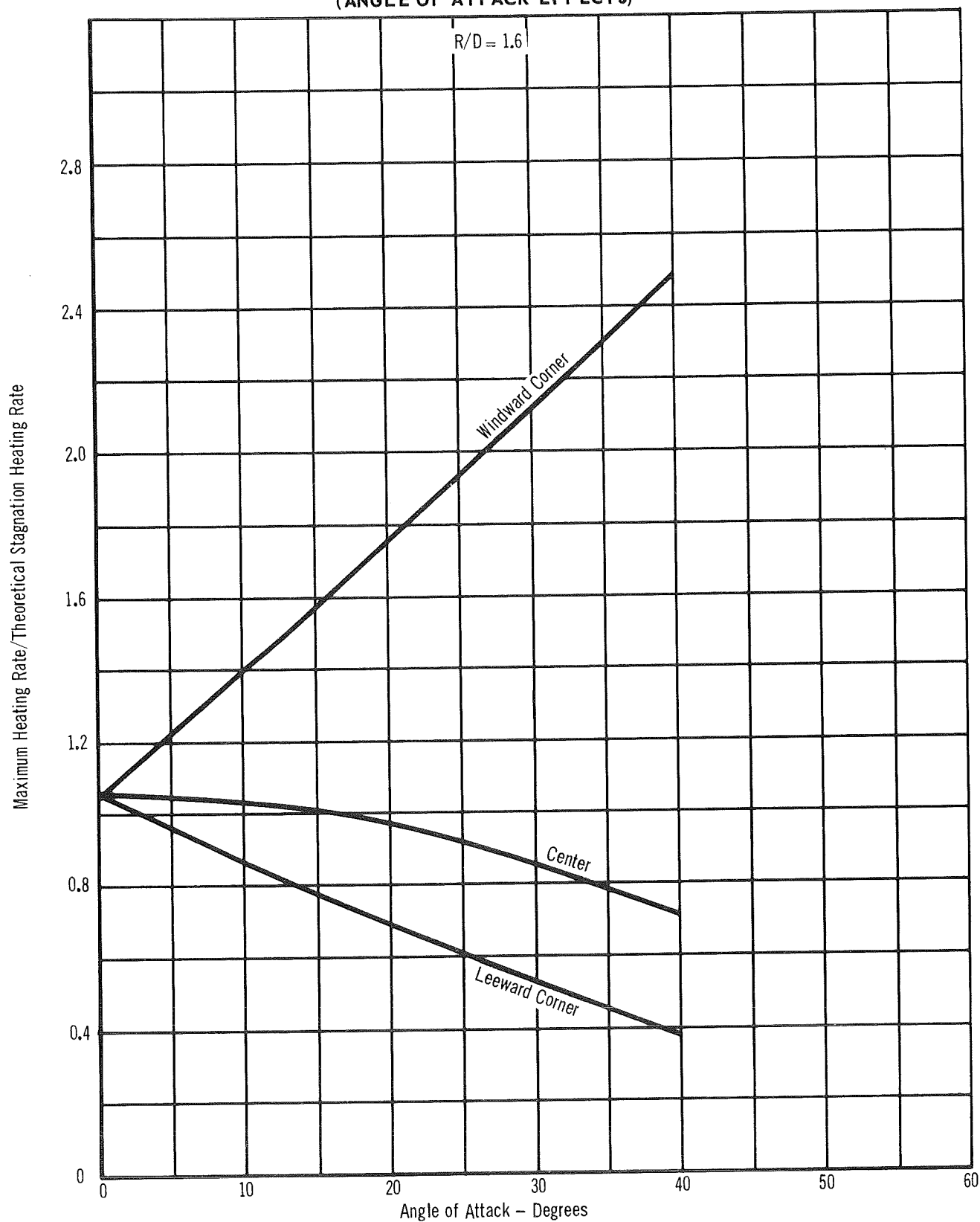


Figure 6-13

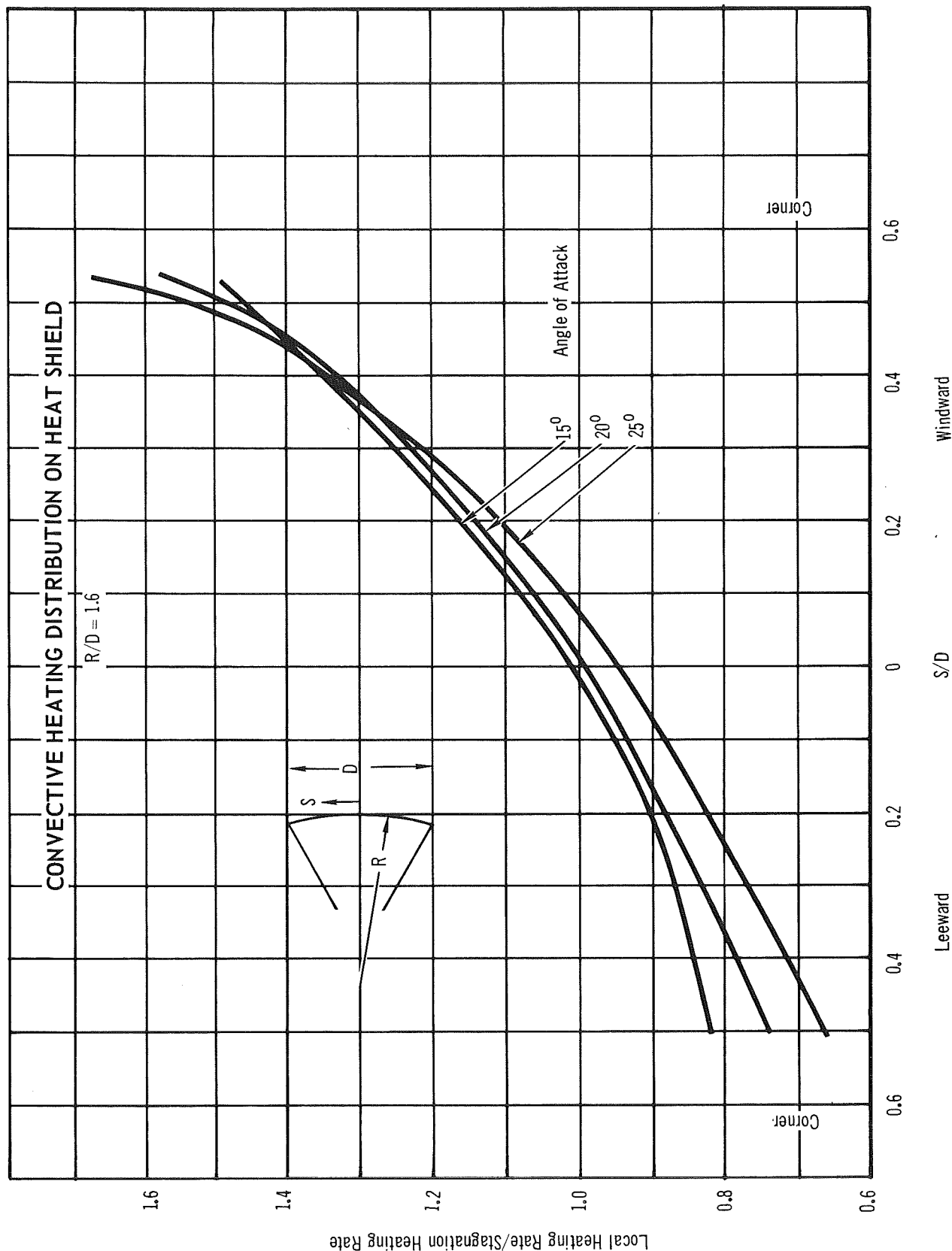


Figure 6-14

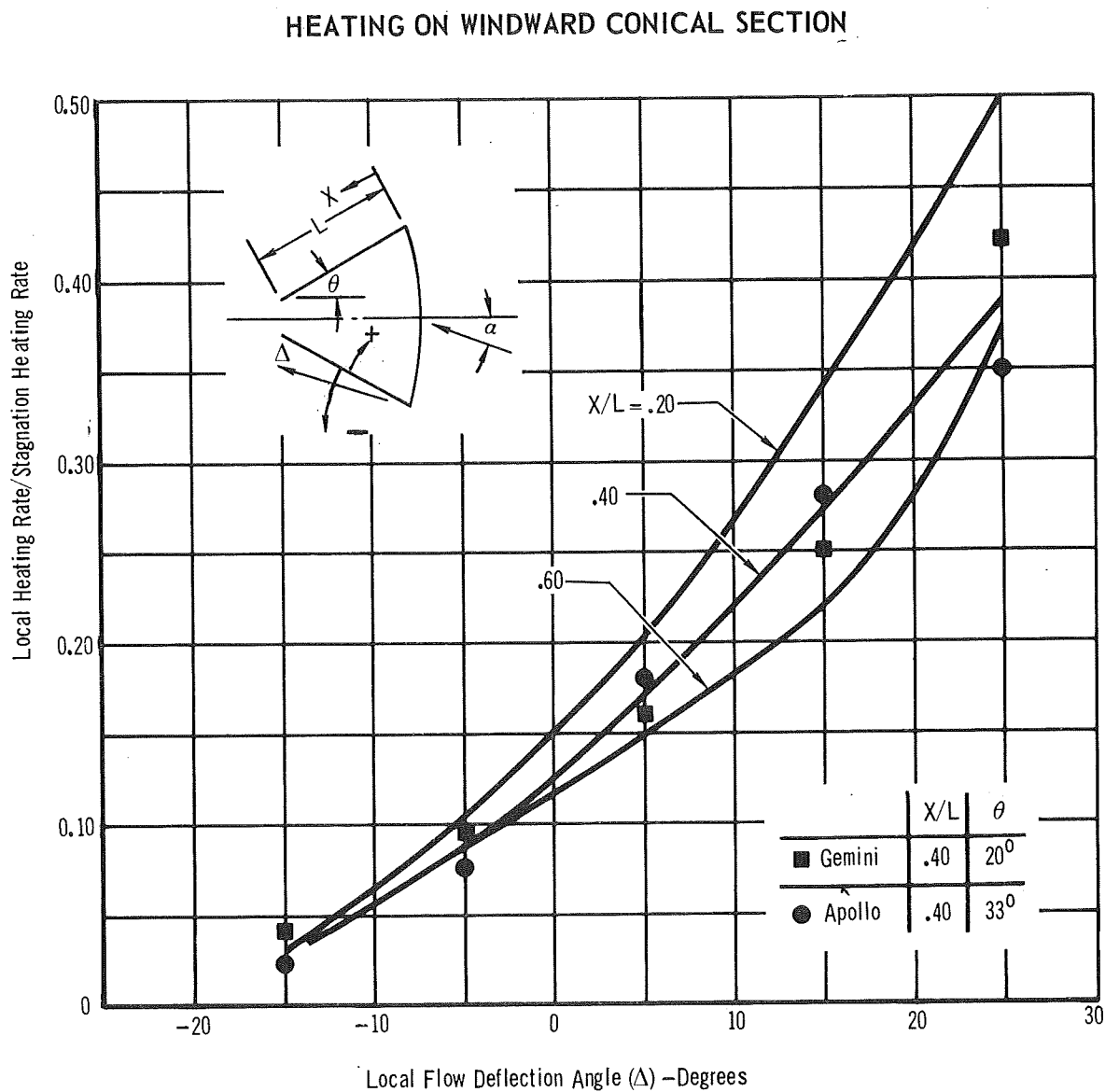
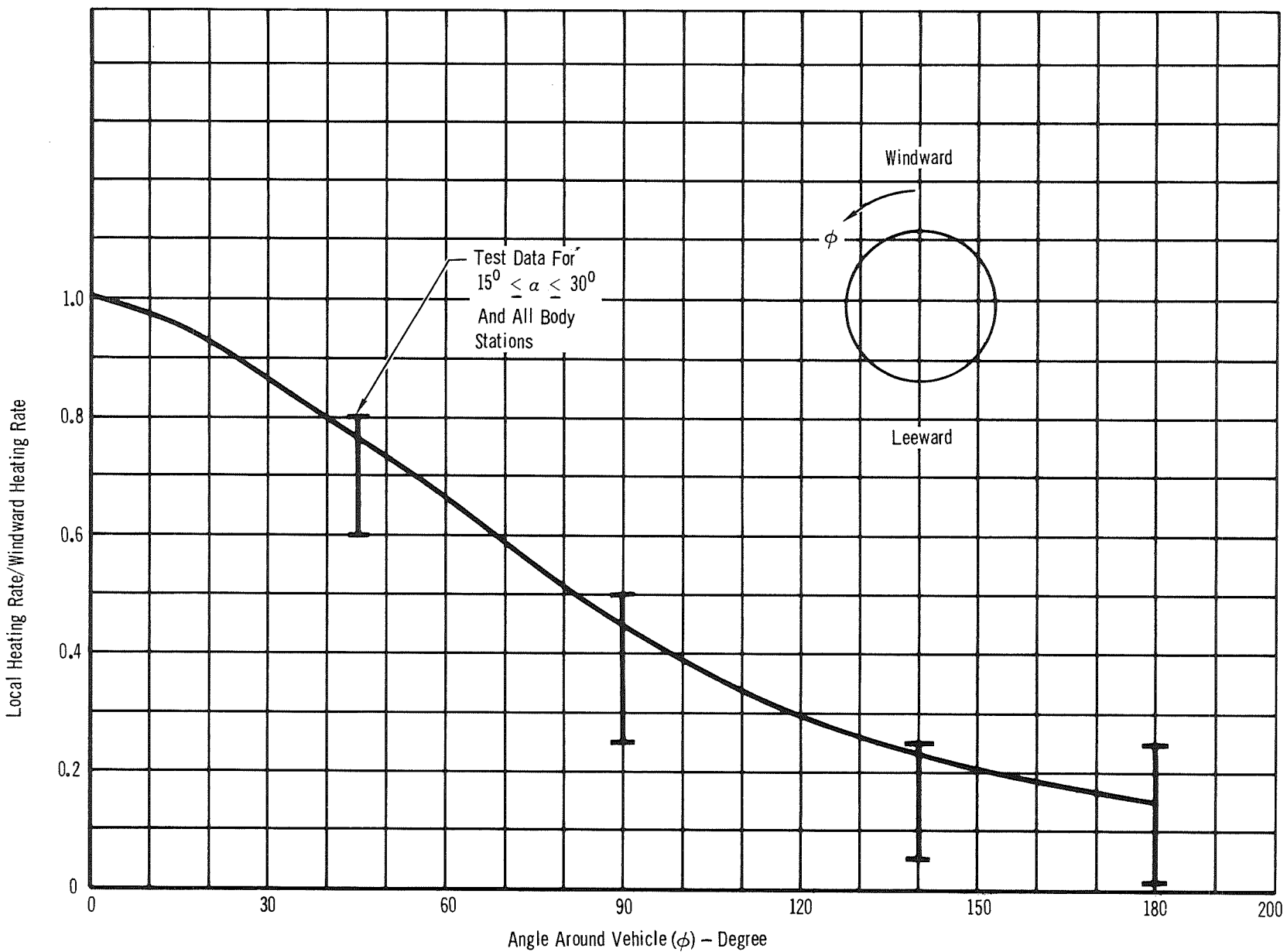


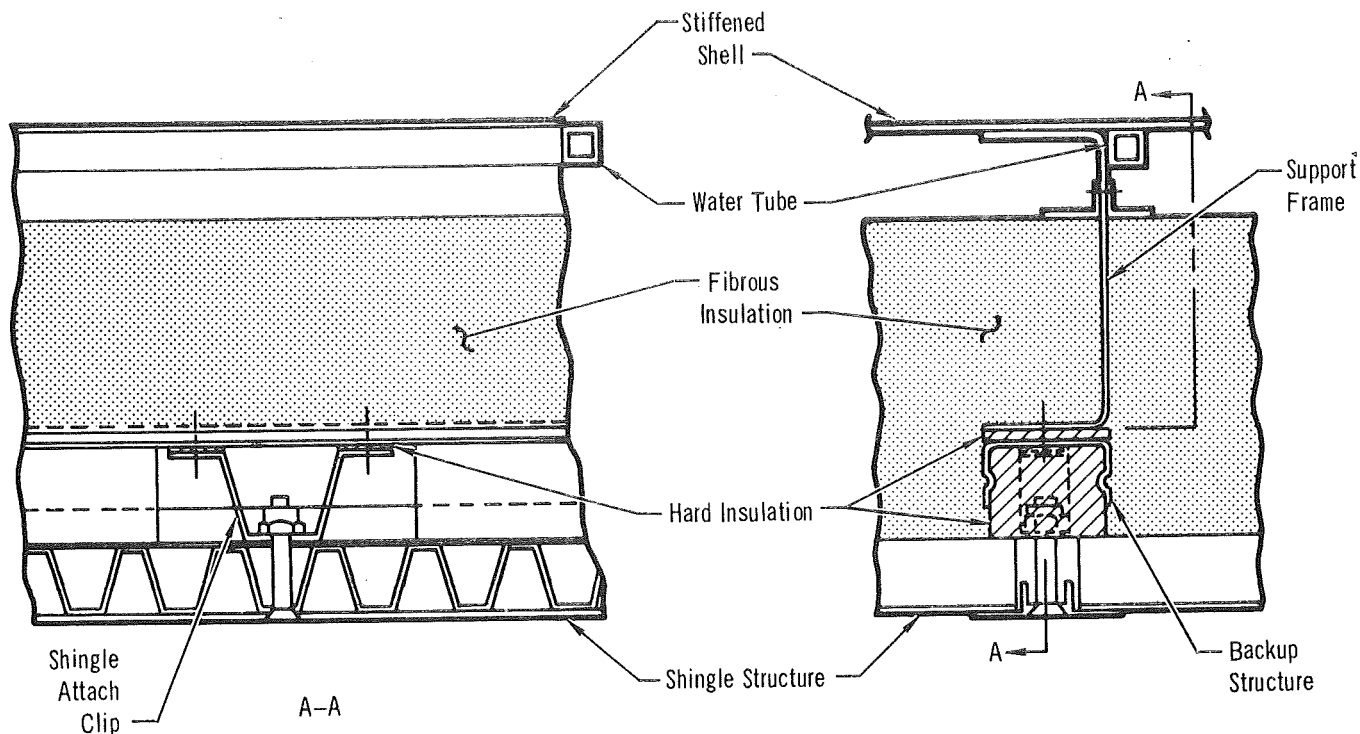
Figure 6-15



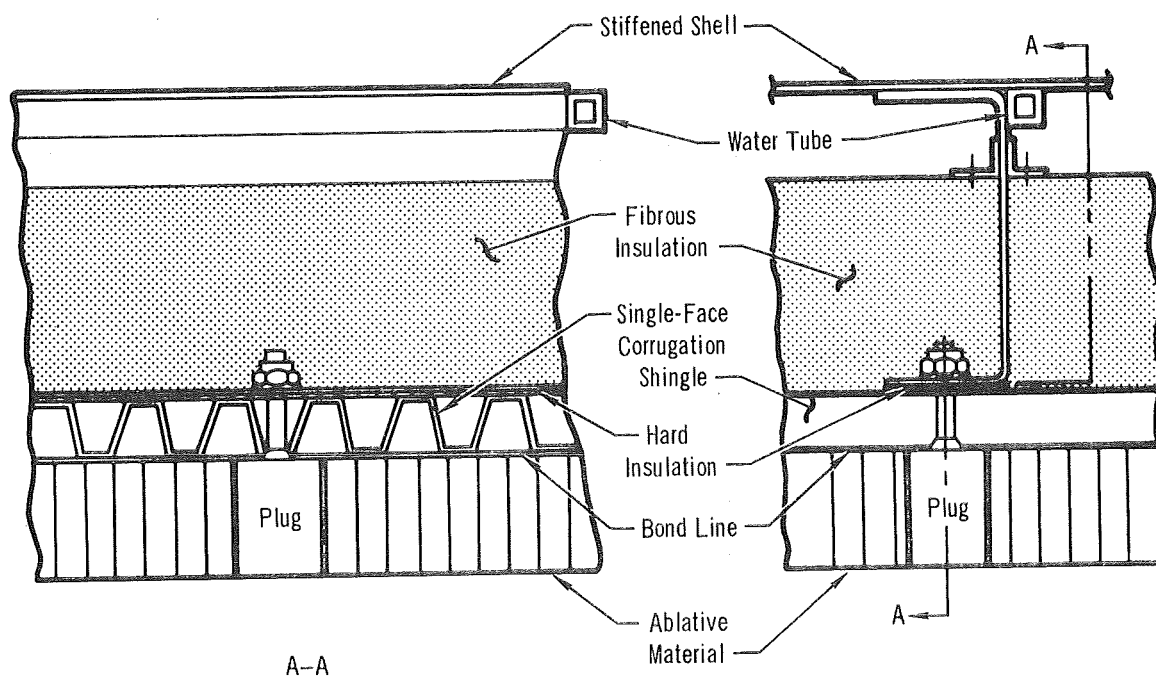
STRUCTURAL - HEAT PROTECTION SYSTEMS

Figure 6-16

RADIATIVE HEAT PROTECTION AND SUPPORT STRUCTURE



ABLATIVE HEAT PROTECTION AND SUPPORT STRUCTURE



6.4 Spacecraft Propulsion - Accomplishing the many different maneuvers, of a manned logistics mission requires a significant amount of propulsion capability on board the spacecraft. As discussed in Section 4., the following types of maneuvers must be performed.

1. Launch Escape (Low and high altitude)
2. Upper Stage Boost (Vehicle configurations D, E, and F only)
3. Orbital Maneuvers (Ascent, docking, attitude control, phasing, and de-orbit)
4. Entry Attitude Control
5. Landing Assist

Due to the consideration of number and size, propulsion systems exert a major influence on spacecraft cost and weight. The large number of potential design concepts associated with each propulsion system is also an important factor. Considering 12 spacecraft configurations and a cargo range from 20,000 to 200,000 pounds results in many possible design concepts, each having different weight and cost characteristics. To avoid evaluation of an excessive number of alternate concepts in Task 6, a limited number of typical designs must be established. Therefore, a preliminary evaluation is performed for each of the above maneuvers to establish the specific performance requirements (thrust level, total impulse); to identify several alternate system concepts (propellant type, system arrangement, major component design characteristics); and to describe a baseline propulsion system. The data are also used as the basis for checkout and modification, as required, of the sizing relationships used in the spacecraft sizing model and to identify system components for which vendor cost data is needed to support development of the cost model.

The specific system performance requirements (thrust level and total impulse) are predicated on preliminary estimates of vehicle size and shape and the general level of performance required by each maneuver. The geometry and mass properties for each vehicle configuration are described in Section 5.2. The maneuver performance requirements of thrust to weight (F/W), velocity increment (ΔV), total impulse to weight (I_T/W), and angular acceleration (α) are discussed in Section 4. These requirements are summarized in Table 6-11.

The thrust level and total impulse requirements are summarized in Tables 6-12 and 6-13; the indicated performance range reflects the effect of cargo quantity variation. A summary of the recommended baseline system

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configurations in shown in Tables 6-1 and 6-2.

A discussion of the performance, alternate concepts, and baseline system description for each of the above maneuvers follows. A brief description of the propulsion sizing model and the results of a propulsion reliability assessment are also included.

TABLE 6-11

MINIMUM PERFORMANCE REQUIREMENTS - PROPULSION

FUNCTION	THRUST LEVEL PARAMETER		TOTAL IMPULSE PARAMETER	
	F/W	α (°/Sec ²)	ΔV (Ft/Sec)	I_T/W (Lb-Sec/Lb)
LAUNCH LOW ALTITUDE	10	-	-	30
ESCAPE HIGH ALTITUDE	.03	-	200	25(a) -
UPPER STAGE ^(b) BOOST	.7	-	16,000	-
ORBIT ASCENT	.03	-	850	-
MANEUVERS DOCKING (Fwd/Aft)	.02	-	40	-
ATTITUDE CONTROL	-	1.0	-	.8
PHASING	.03	-	280 ^(f)	-
DE-ORBIT	.10	-	500	-
ENTRY ATTITUDE CONTROL	-	5.0(e) 2.0(d)	-	1.4
LANDING ASSIST	3.0(c) .25(d)	-	-	4.2 to 9.1(c) 3.75(d)

(a) Vehicle II/A,B only

(b) Vehicles I/D,E,F and II/D,E,F only

(c) Vehicles I/D,E,F only

(d) Vehicles II only

(e) Vehicles I only

(f) Vehicle I, Single Return Site only

TABLE 6-12

SUMMARY - PROPULSION SYSTEM THRUST LEVEL REQUIREMENTS

Thrust at Minimum Cargo/Thrust at Maximum Cargo (Thousands of Pounds)

VEHICLE MANEUVER	I A/B	I C	I D	I E/F	II A/B	II C	II D	II E/F
<u>Launch Escape</u>								
Low Altitude	195	195	195	195	202	202	202	202
High Altitude	1.1/7.7	1.3/8.4	.354	.354	1.4/7.9	1.5/8.9	.498	.498
<u>Upper Stage Boost</u>	NA	NA	130/770	175/1100	NA	NA	150/860	210/1130
<u>Orbit Maneuvers</u>								
Ascent	1.1/7.7	1.3/8.4	1.5/9.2	2.4/13.6	1.4/7.9	1.5/8.9	1.8/10.3	2.8/15.2
Docking (Fwd/Aft)	.6/4.7	.85/5.0	.9/5.6	1.4/8.3	.85/4.8	.9/5.5	1.1/6.2	1.7/9.3
Attitude Control	*	*	*	*	*	*	*	*
Phase	.35	.55/1.8	.8/2.2	1.6/6.6	NA	NA	NA	NA
Deorbit	1.2	1.7/5.8	2.5/7.2	5.2/21.5	1.5	2.8/7.7	3.4/10.0	6.3/26.0
<u>Entry</u>								
Attitude Control	*	*	*	*	*	*	*	*
<u>Landing Assist</u>	NA	55/170	75/210	150/610	4	6.6/18	8/24	15/61.5

* Attitude Control thrust levels determined in Task 6.
Order of magnitude range estimated at 100 to 5000 Lbs.

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TABLE 6-13
SUMMARY - PROPULSION SYSTEM TOTAL IMPULSE REQUIREMENTS

Impulse at Minimum Cargo/Impulse at Maximum Cargo (Thousands of Lb-Sec)

VEHICLE MANEUVER	I A/B	I C	I D	I E/F	II A/B	II C	II D	II E/F
Launch Escape								
Low Altitude	583	583	583	583	505	606	606	606
High Altitude	70.8	70.8	70.8	70.8	99.5	99.5	99.5	99.5
Upper Stage								
Boost	NA	NA	56,900/ 335,000	76,700/ 445,000	NA	NA	65,700/ 378,000	495,000
Orbit Maneuvers								
Ascent	900/ 6,500	1,100/ 7,000	1,300/ 7,700	2,000/ 11,500	1,200/ 6,600	1,300/ 7,600	1,500/ 8,600	2,200/ 2,800
Docking	40/285	50/315	59/340	88/510	56/300	57/335	68/385	107/570
Attitude Control	28/205	36/225	40/245	70/365	38/210	40/238	47/272	75/405
Phase	80/100	150/525	225/625	450/1,900	NA	NA	NA	NA
Deorbit	170	270/880	400/1,100	780,3,250	225	410/1,150	500/1,510	940/3,430
Entry Attitude								
Control	16.5	25/79	35/96	70/285	23.2	36/100	45/133	79/315
Landing Assist	NA	50/275	75/325	225/1825	60	100/270	120/360	220/720

6.4.1 Launch Escape Propulsion - A launch escape system is required on all vehicle configurations to provide safe crew removal if a launch vehicle failure occurs. This capability must be provided during the entire launch trajectory. Specific system performance requirements, a discussion of alternate design concepts, and a description of the baseline system follow.

6.4.1.1 Launch Escape Performance Requirements - The launch escape propulsion function consists of two separate maneuvers: low altitude escape and high altitude escape. The low altitude maneuver is applicable to that part of the launch trajectory beginning when the vehicle is on the pad, and continuing through the flight until the dynamic pressure has reduced to a low value (about 1.0 psf). High altitude escape applies to the remainder of the launch trajectory (through initial orbit insertion). As performance requirements for these two maneuvers differ significantly, they are considered separately.

Low Altitude Escape - The propulsion performance requirements for this maneuver are characterized by high thrust levels, relative to the total impulse required, and, therefore, short burn times (~ 2 seconds). Thrust level requirements are set by the necessity to "outrun" the overpressure wave generated by a sea level explosion of the launch vehicle and by the amount of force needed to separate from the launch vehicle at maximum dynamic pressure. The total impulse requirements are based on an off-the-pad escape with the magnitudes being set by the altitude needed for parachute deployment and the separation distance necessary to escape the fire ball. A fast response time is also needed (~ 100 milliseconds) because the time interval between detection of an impending launch vehicle failure and its actual occurrence is extremely short.

Thrust level and total impulse requirements for low altitude escape are shown in Figure 6-17 and 6-18 as a function of the total escape weight. As discussed in Section 4.3, when the entire lifting body spacecraft is used as the escape vehicle, the total impulse requirements are reduced since the gliding capability of the vehicle can be used for additional range. The total escape weight consists of the crew module escape weight plus the weight of the launch escape propulsion systems. For modular vehicles, the crew module escape weight is identical to that of the entry vehicle. For integral vehicles, however, the crew module escape weight corresponds to that of an escape capsule (Reference

Section 5.2).

High Altitude Escape - The propulsion performance requirements for high altitude escape are much less than for the low altitude maneuver. The ambient pressure is very low, substantially reducing the propagation of the fire ball and overpressure wave. The thrust level required is that needed to separate and to maneuver away from a tumbling launch vehicle. The total impulse required is that necessary for de-orbit from 100 na mi since the escape maneuver could occur near the end of the launch trajectory and result in inadvertant orbit insertion.

The thrust level and total impulse required for high altitude escape are shown in Figures 6-19 and 6-20. The discussion of excape weight composition, presented in Section 6.2.1.1, is also applicable.

**LOW ALTITUDE ESCAPE
THRUST LEVEL REQUIREMENTS**

Figure 6-17

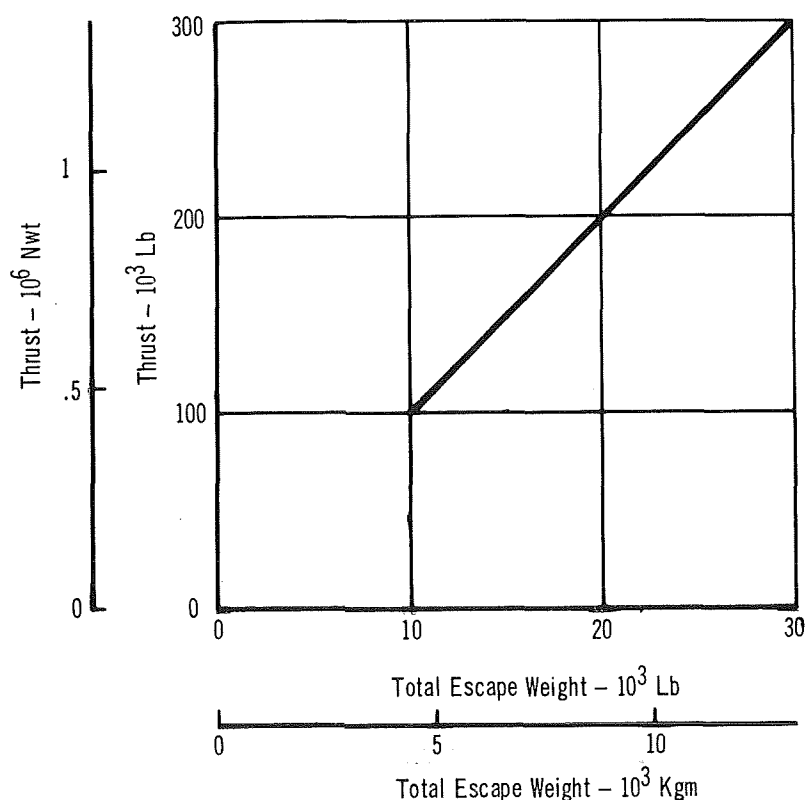


Figure 6-18

**LOW ALTITUDE ESCAPE
TOTAL IMPULSE REQUIREMENTS**

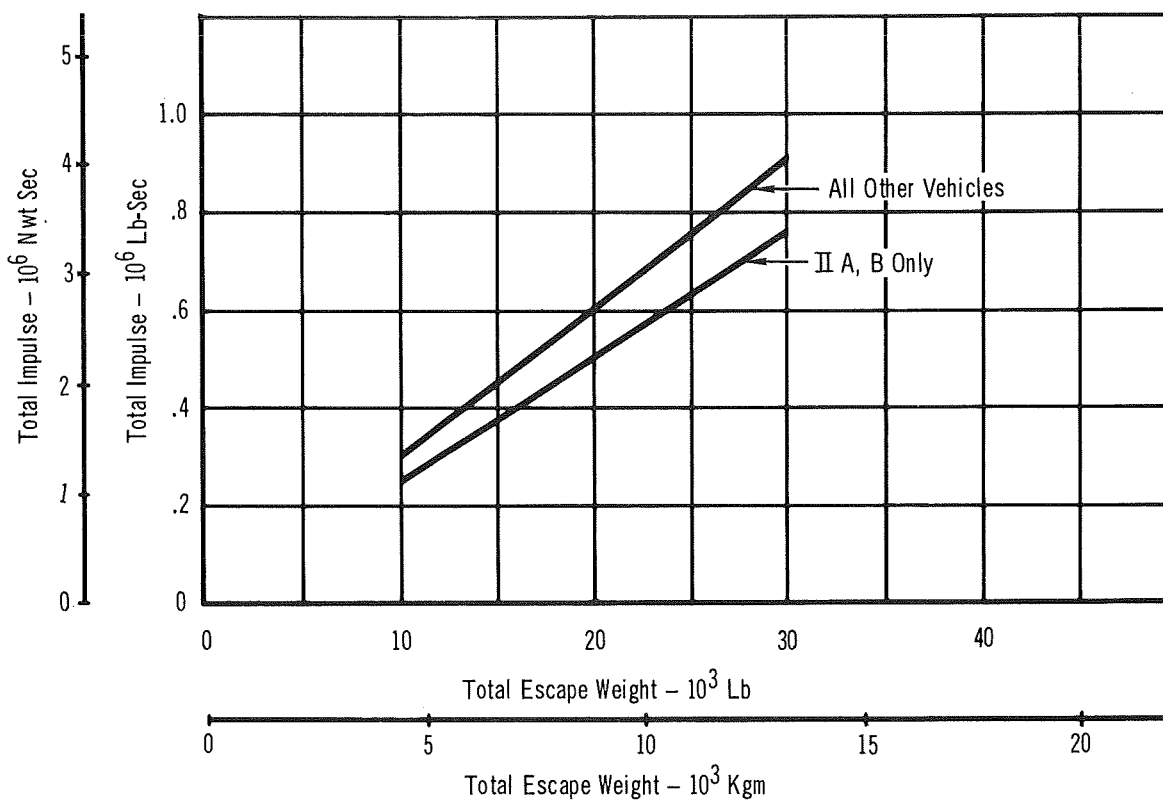
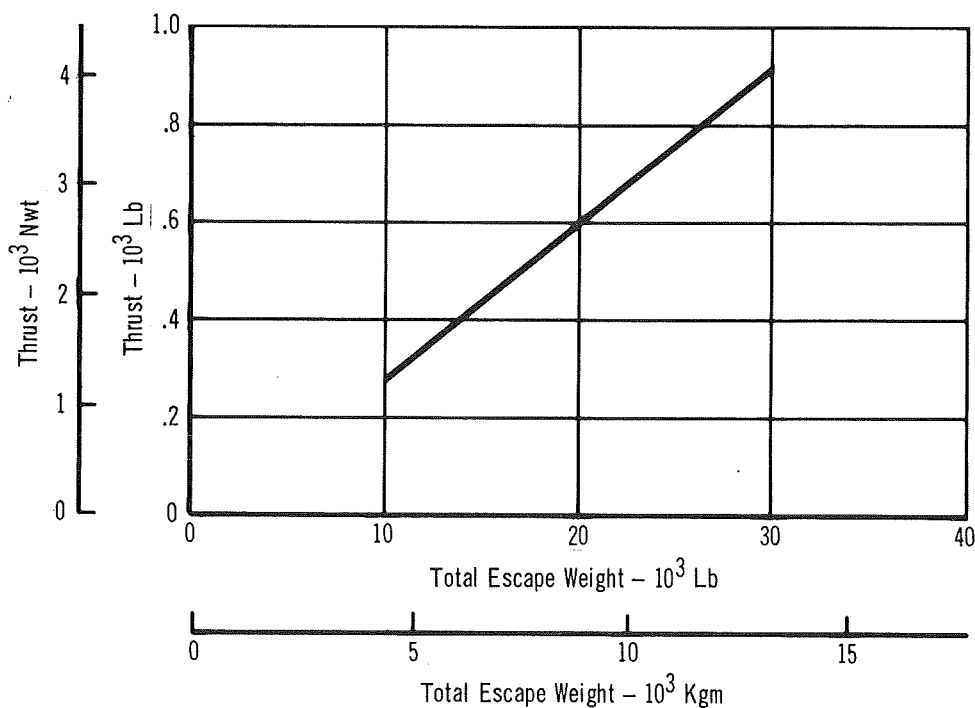


Figure 6-19

**HIGH ALTITUDE ESCAPE
THRUST LEVEL REQUIREMENTS**

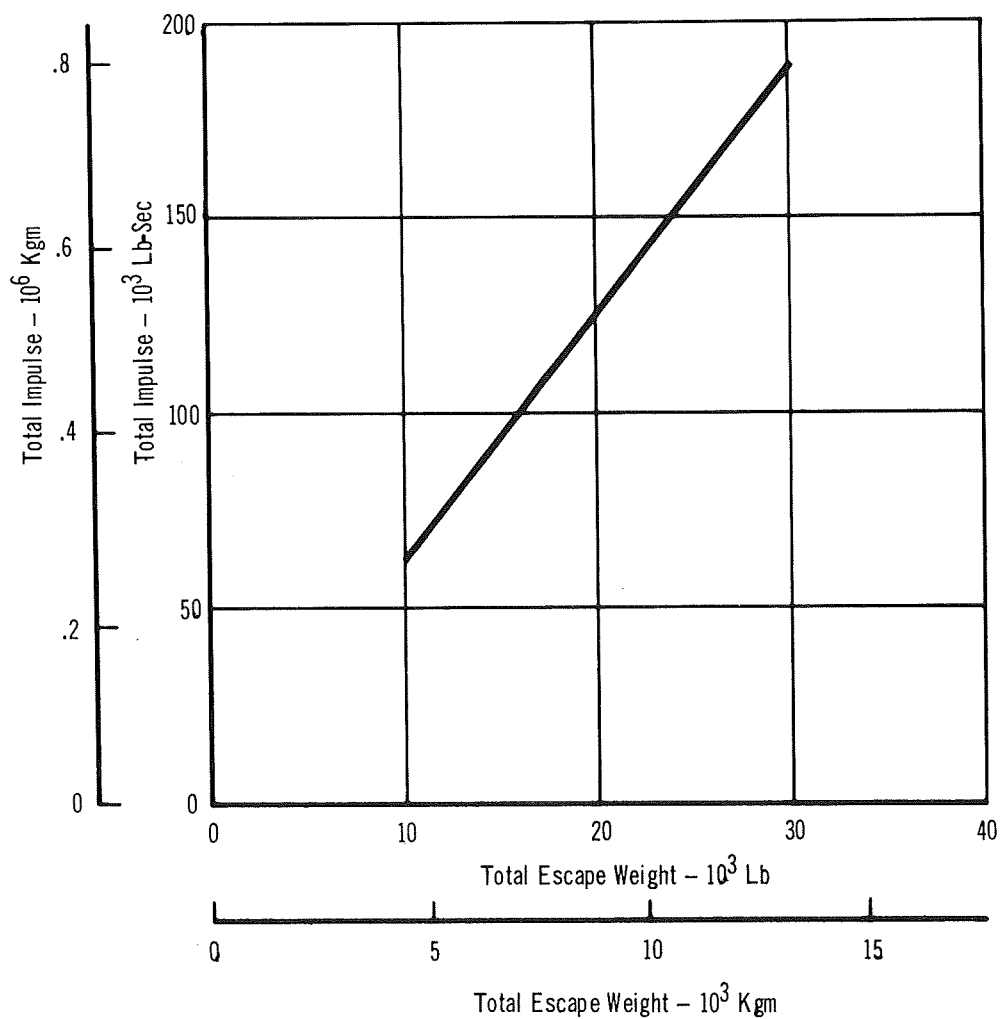


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Figure 6-20

**HIGH ALTITUDE ESCAPE
TOTAL IMPULSE REQUIREMENTS**



A requirement for attitude control is also associated with the high altitude escape maneuver. This results from the necessity to perform an entry maneuver following escape from the launch vehicle. Thus, a propulsion system is needed to orient the escape vehicle prior to entering the sensible atmosphere and for angular control of the vehicle during atmospheric flight. Propulsion performance requirements for this maneuver are the same as those for the normal entry maneuver (Reference Section 6.2.4). These requirements are significant only for integral vehicles where an escape capsule is required. For modular vehicles, the normal entry altitude control system is used.

6.4.1.2 Systems Concepts - Launch Escape - The propulsion systems concepts for launch escape are established by considering propellant type and systems arrangement (systems location and number). These design variations and the resulting effect of the systems concept are discussed below.

Propellant Type - Two general classes of propellants are considered for use in the launch escape systems: Solids and liquids. Hybrids could also be included from the standpoint of feasibility, however they offer no significant performance advantage for this application and do not have the development background of either solids or liquids.

The major factor influencing the propellant evaluation is the type of systems performance required (response time, thrust level, total impulse). Since the low and high altitude escape maneuvers each require a different level of performance capability, they are discussed separately.

The low altitude maneuver requires a very fast response system with a high thrust level and short burn time. Solid motor systems meet these requirements and have been used extensively in the past for this application, i.e., Mercury, Gemini B, Apollo. High thrust liquid systems, of the Saturn and Titan launch vehicle type, have relatively slow response and are therefore considered unacceptable for this function. Liquid propellant missile systems, such as Lance and Bullpup, exhibit response and performance characteristics that are comparable to those of solids. These systems, however, are expected to be more costly than solids. They are more complex (more components) and have been developed only for unmanned applications. Hence, the propulsion concepts for low altitude escape are limited to those using solid propellants.

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The high altitude escape function requires a relatively low thrust and total impulse system and imposes no response time constraints. Therefore, either a solid or liquid propellant system could be used. For those spacecraft configurations which have a suitable propulsion system on board for other applications it will also be used for this function. However, for vehicle configurations where a separate, high altitude escape system is required, solid propellant motors will be used since they are essentially equal in weight and less costly than a comparable liquid system.

Systems Arrangement - As indicated in Section 6.4.1.1, the capability to perform both a low and a high altitude escape maneuver is necessary. The low altitude maneuvers require a separate escape system because of the unique performance capability needed; no other spacecraft system has the combination of thrust level, total impulse, and response time necessary. The high altitude maneuver, however, requires a much lower level of system performance. The orbit maneuver systems (Section 6.4.3) could be used to perform this maneuver, providing that the orbital system is not a part of the launch vehicle upper stage. This is the approach used for reuse categories A, B, and C. The integral boost vehicles (categories D, E and F), however must have a separate high altitude system since the orbital system is essentially a part of the upper stage. Because of this difference in systems requirements, the system arrangements for the two groups of vehicles are considered separately.

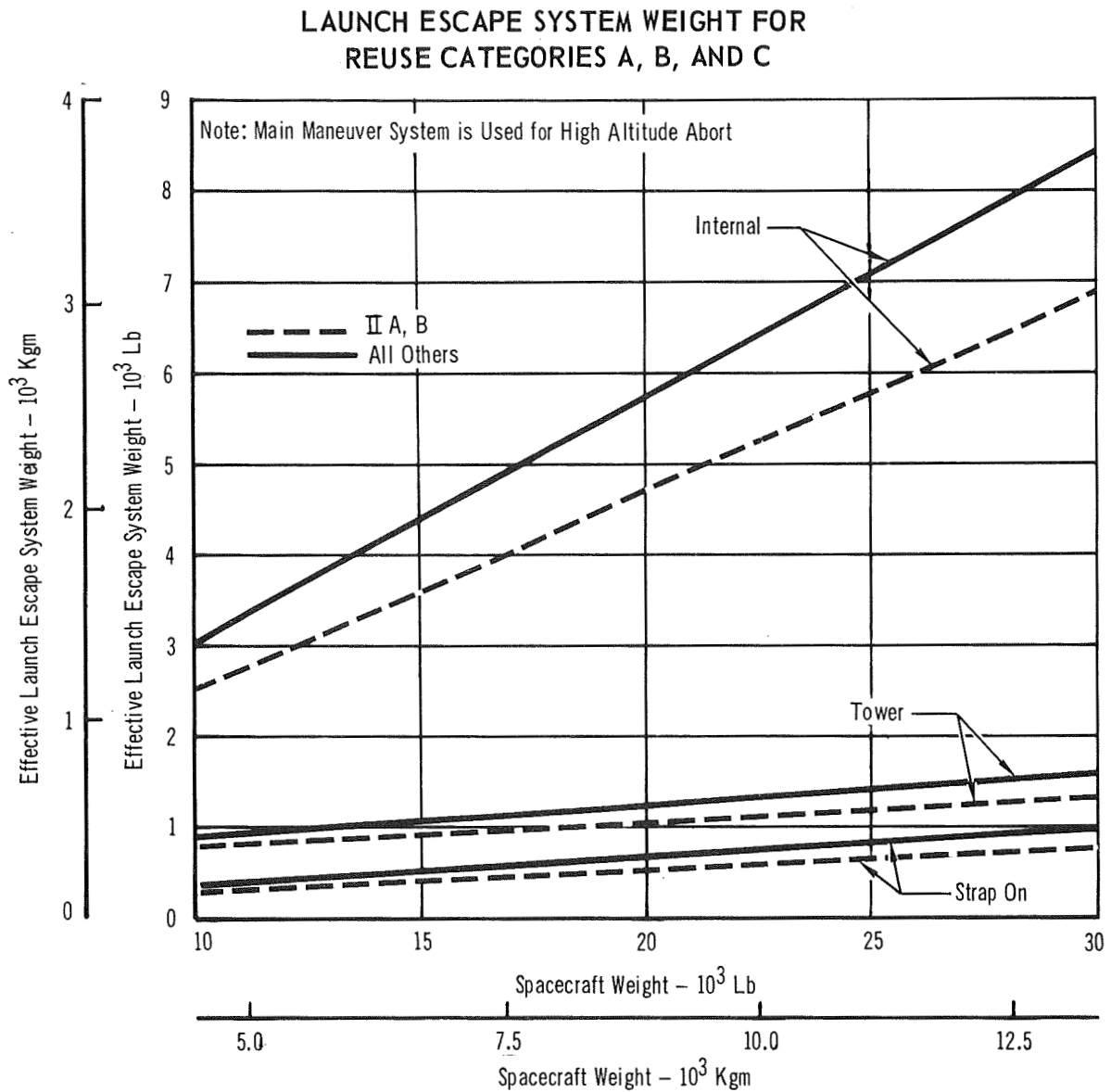
Reuse Categories A, B and C - Three low altitude system installation concepts are considered: (a) a single, tower mounted motor, (b) multiple strap-on motors, and (c) multiple motors located inside the entry vehicle. The tower and strap-on systems are located external to the spacecraft and can be jettisoned. An effective payload weight penalty of 12.5 percent of the actual systems weight is used as being typical for the launch trajectories shown in Section 4. The tower concept has a greater development background but, because of the tower structure, is heavier than the strap-on concept. The internal system exhibits a relatively high weight penalty since it is not jettisonable, but has the advantage of being retained and is, therefore, reusable.

A weight comparison of these concepts is presented in Figure 6-21 as a function of spacecraft (excluding propulsion) weight. As shown, the internal installation is more than 1500 pounds heavier than either of the other concepts.

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Figure 6-21



Preliminary investigations indicate that this weight differential is excessive, even in light of the potential cost savings through reuse of the internal configuration. Thus, only the tower and strap-on concepts are retained for more detailed evaluation in Task 6.

Reuse Categories D, E, and F - The tower, strap-on, and internal low altitude installation concepts discussed above are also applicable to this group of vehicles. In addition, the use of a single vs a dual system concept for the low and high altitude maneuvers is considered. The single system concept involves retaining the low altitude system for use at high altitude. With a dual system, the low altitude system is jettisoned while the high altitude system is retained and reused.

A weight comparison of these concepts is presented in Figure 6-22. The weight difference between the single and dual system concepts is mainly attributable to the small effective weight penalty of the jettisonable, low altitude system. Because of the large weight difference between concepts, only the dual systems are considered further.

Concept Summary - The systems arrangement alternatives result in a total of four launch escape system concepts, two for reuse categories A, B, and C and two for categories D, E, and F. These concepts are summarized in Table 6-14. As noted, the baseline system for ballistic vehicles is a tower while for the M2-F2 shape, strap-ons are used. The tower is selected due to its long history of usage with a ballistic shaped vehicle; the choice of strap-ons for the M2-F2 is based solely on weight considerations.

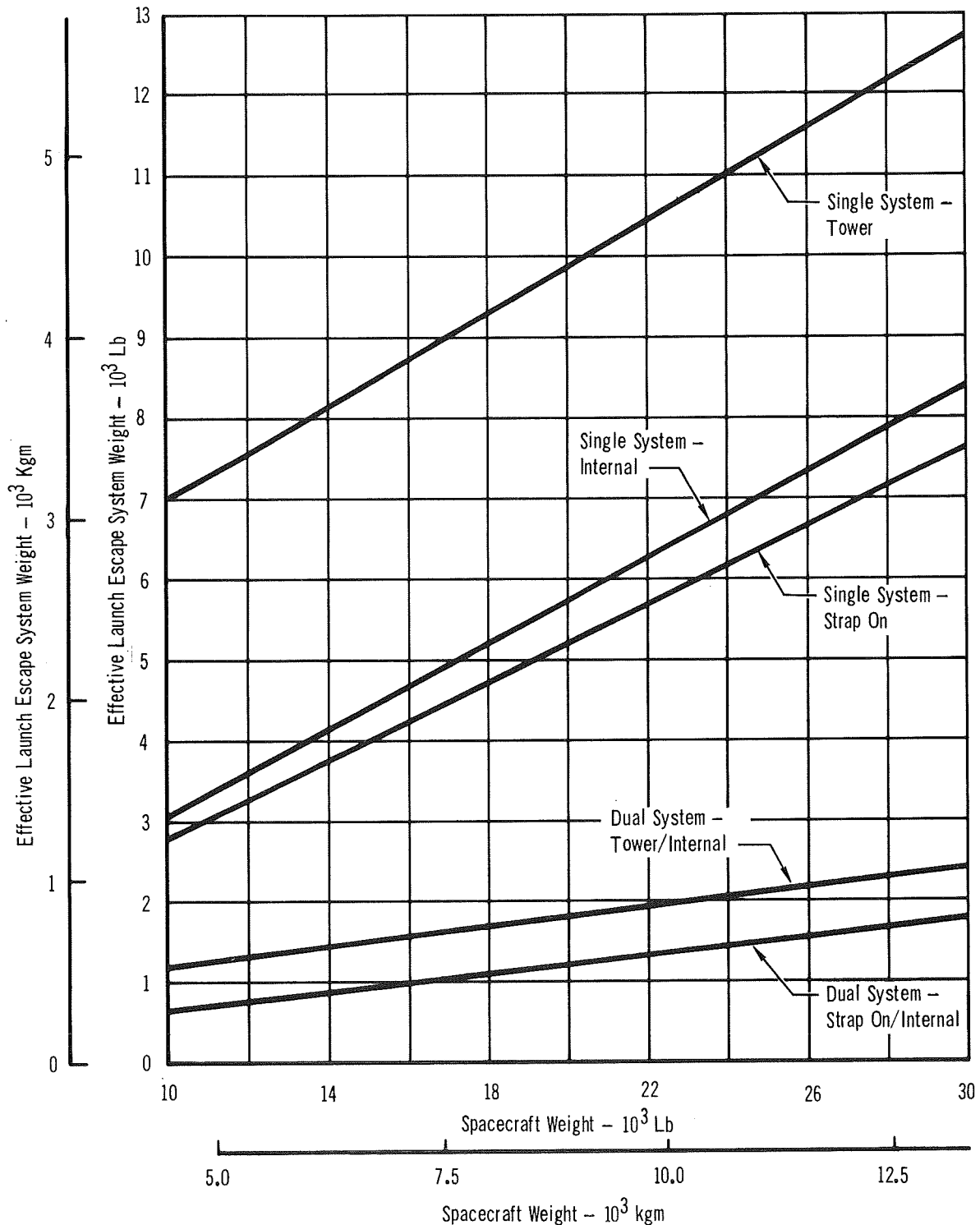
6.4.1.3 Baseline Descriptions - The baseline configurations for the launch escape propulsion system are shown in Figure 6-23.

All ballistic vehicles utilize a jettisonable tower mounted solid motor for low altitude escape. The systems design is similar to that of the Apollo tower, including the use of a pitch and jettison motor. Low altitude escape of the M2-F2 vehicles is accomplished by a group of jettisonable, strap-on solid motors. For modular vehicles, these motors are installed on the forward surface of the mission module; for integral vehicles, they are attached to the escape capsule.

High altitude escape, of all vehicles that do not incorporate an upper stage boost system, is accomplished by the orbit maneuver system. Attitude

Figure 6-22

LAUNCH ESCAPE SYSTEM WEIGHT FOR
REUSE CATEGORIES D, E, AND F



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control during the subsequent entry maneuver is provided by the vehicle's entry control system. For all integral boost vehicles, a group of internally mounted solid motors is used for high altitude escape of the crew capsule. A storable bipropellant system is located in the escape capsule for necessary attitude control. Both the internal solid motors and the escape capsule attitude control systems are retained and reused.

TABLE 6-14

Concept Summary - Launch Escape

Concept No.	Low Altitude System	High Altitude System	Vehicle Application
LE-1*	Tower	Orbit maneuver System	I/A, B, C
LE-2**	Strap-on	Orbit maneuver System	II/A, B, C
LE-3*	Tower	Internal	I/D, E, F
LE-4**	Strap-on	Internal	II/D, E, F

* Baseline - Ballistic Vehicle (Type I)

** Baseline - M2/F2 Vehicle (Type II)

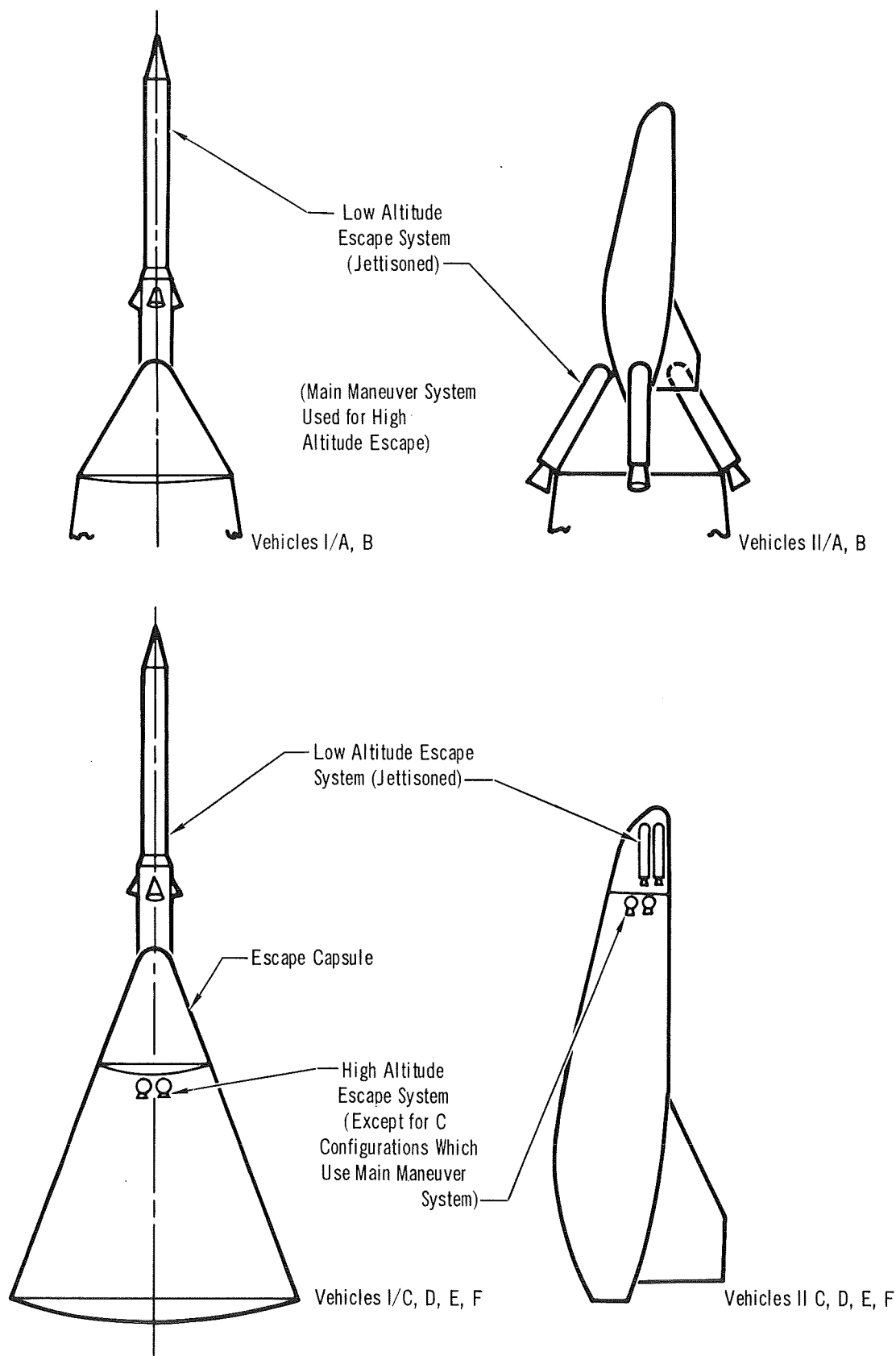
6.4.2 Upper Stage Boost Propulsion - Defining reuse categories

D, E, and F includes the requirement to provide the final boost velocity increment (ΔV) with an integral spacecraft propulsion system. Category D uses external propellant tanks that are jettisoned after initial orbit insertion. Engine and feed subsystems are located inside the entry vehicle. In categories E and F, the entire boost propulsion system is installed within the entry vehicle. The system performance requirements are presented below, followed by a discussion of alternate systems concepts and a description of the baseline design.

6.4.2.1 Upper Stage Boost Propulsion Requirements - The upper stage boost maneuver is the same as that performed by the second stage of a two stage launch vehicle. As such, this maneuver is characterized by the single burn application of a large velocity increment (ΔV) to a high vehicle weight. Therefore, the total impulse and thrust requirements are significantly larger

BASELINE CONFIGURATIONS – LAUNCH ESCAPE SYSTEM

Figure 6-23



than those for any other maneuver under consideration. However, the required thrust levels are low relative to the large amount of total impulse needed, resulting in a long burn time (~ 450 seconds). In addition, a gimbal capability is needed to provide thrust vector control.

The thrust level and total impulse requirements for upper stage boost are shown in Figures 6-24 and 6-25. These requirements are predicated on a system having a propellant mass fraction (propellant weight to total system weight ratio) of 0.90 and the use of O_2/H_2 propellants. However, the large ΔV necessary makes these thrust level and total impulse requirements very sensitive to both mass fraction and specific impulse. As discussed in the following paragraphs, propellant combinations covering a wide range of specific impulse and bulk density are considered. Further, to completely establish the mass fraction of an integral vehicle requires a more detailed spacecraft sizing analysis than was possible at this point in the study. Therefore, although these performance requirements must be considered preliminary, they are sufficiently accurate to gain an insight into the relative size of the upper stage boost systems and to use as a basis for concept selection. A detailed sizing analysis is included in the Task 6 evaluation.

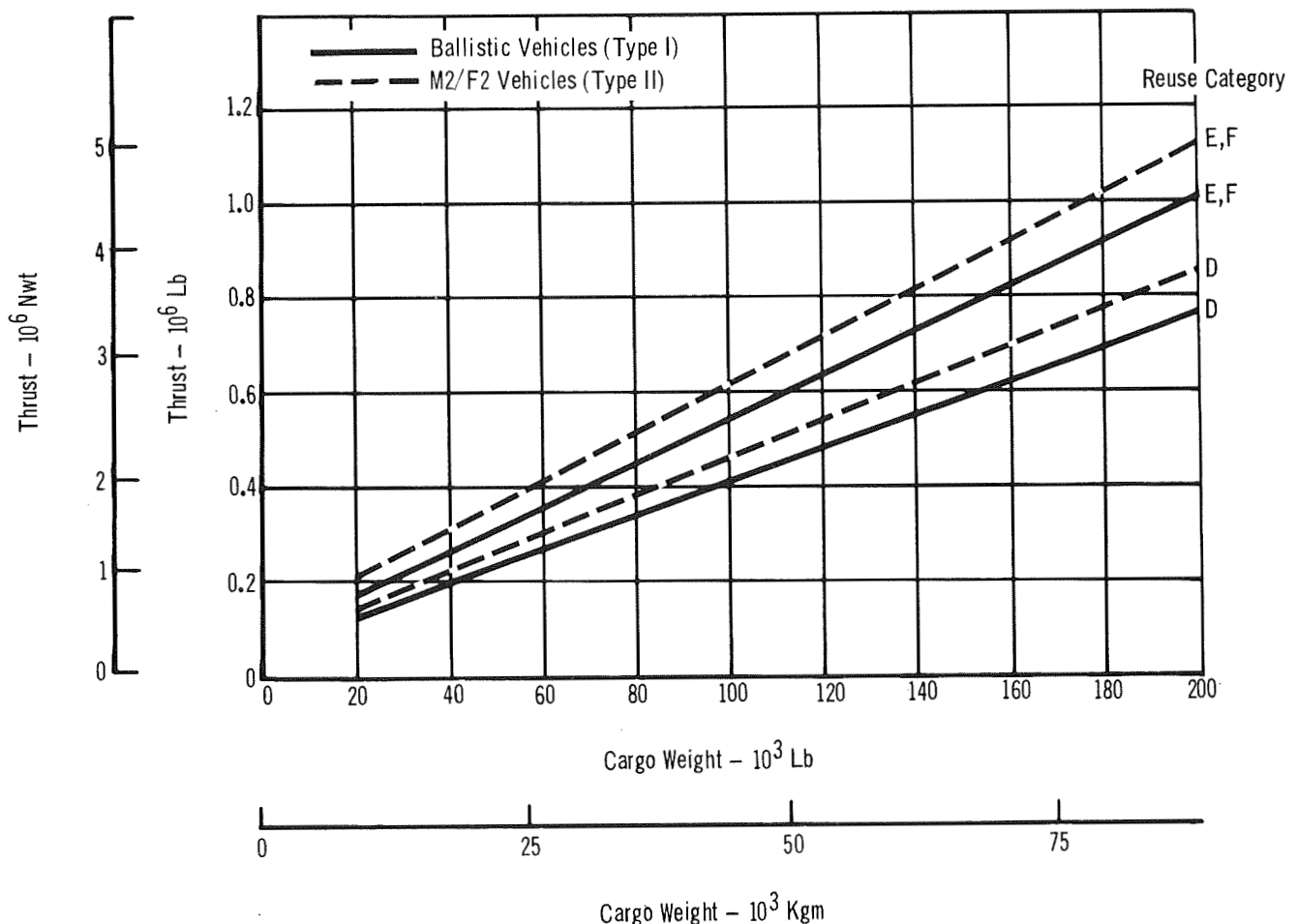
6.4.2.2 System Concepts - Upper Stage Boost - The design concept for the upper stage boost system is dictated by the system's performance, reuse and installation requirements. A very large (total impulse and thrust level), reusable system is necessary with at least the engines and feed subsystems located inside the entry module. As a result, only pump-fed systems utilizing high chamber pressure (1500 to 3000 psia) regenerative or transpiration cooled engines are considered. A pump-fed system is selected to prevent excessive pressurization and tankage subsystems weight penalties. Pump-fed systems also have a considerable background of usage for the boost function while development of pressure fed designs have been limited to much lower thrust and total impulse applications. High chamber pressure engines are considered necessary to minimize engine compartment volume and the size and weight of the entry module. The length and diameter of large bell nozzle configurations are shown on Figure 6-26 and is based on a fixed specific impulse for an O_2/H_2 engine with a chamber pressure of 3000 psia. The aerospike engine configuration was also considered but cost data was not received early enough for incorporation into the cost estimating relationships. The engine cooling techniques are chosen to provide the

extended life capability needed for reuse.

There are, however, additional design options that must be evaluated to fully identify the possible boost system concepts. The most influential factor is the choice of propellant combination. Detailed design variations involving

**UPPER STAGE BOOST
THRUST LEVEL REQUIREMENTS**

Figure 6-24

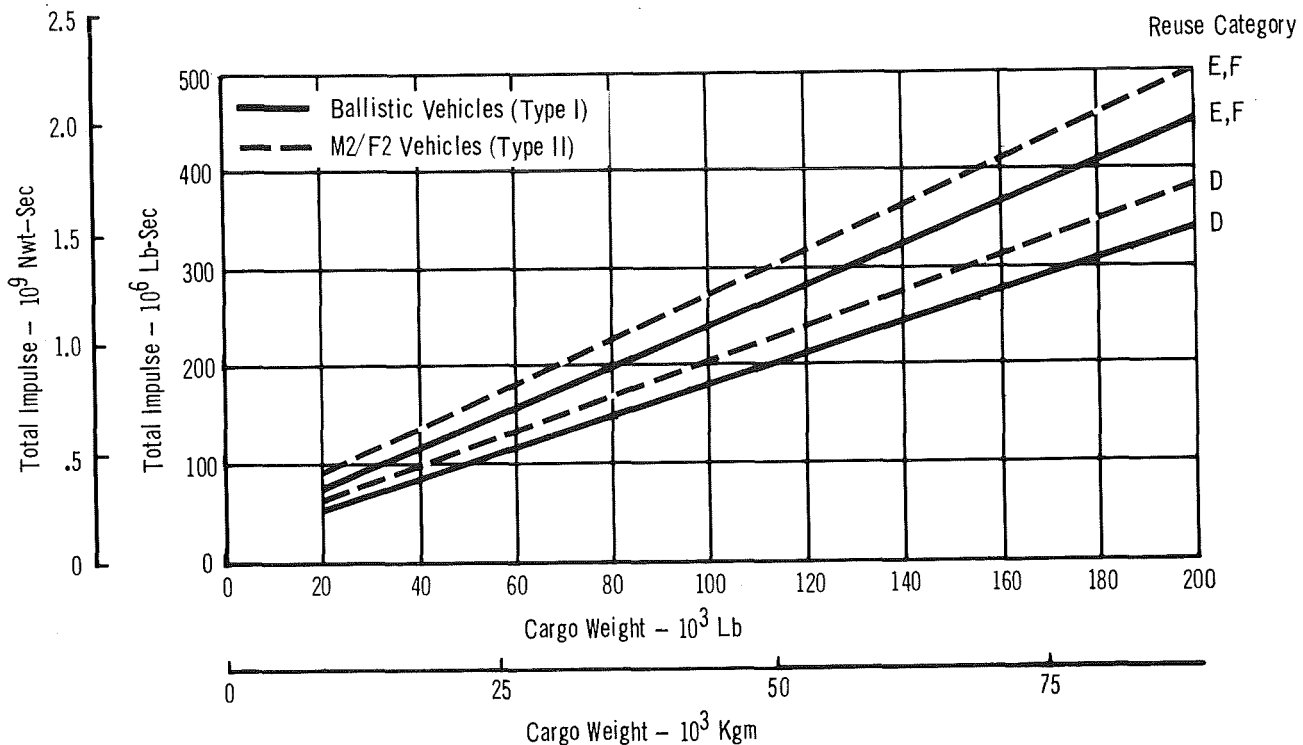


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UPPER STAGE BOOST TOTAL IMPULSE REQUIREMENTS

Figure 6-25



engine nozzle design and propellant tankage configurations are also considered. These design options and the resulting alternate system concepts are discussed below.

Propellants - The following four propellant combinations are considered;

- Oxygen/Hydrogen; (O_2/H_2)
- Fluorine/Hydrogen; (F_2/H_2)
- Fluorine (87%), Oxygen (13%)/Methane; (FLOX/ CH_4)
- Nitrogen Tetroxide/50% Hyrazine, 50% unsymmetrical dimethylhydrazine; (NTO/A-50)

These combinations were selected to cover a broad spectrum of development background, specific impulse and bulk density characteristics. A comparison of these parameters is shown in Table 6-15. Bulk density is considered an important characteristic particularly for the fully integral vehicles (Categories E and F). With these vehicles, the volume required for upper stage boost propellant is a major factor in determining entry module size. Thus, propellant density and specific impulse greatly influence the design and weight of the entry module structural systems. The strong interaction between the type of propellant used and the overall size of the vehicle requires a

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detailed vehicle sizing analysis to adequately compare these design concepts. Thus, all four propellant combinations are retained for further evaluation.

TABLE 6-15

Boost Propellant Characteristics

Propellant	Specific Impulse Seconds ($\epsilon = 100$)	Bulk Density lbs/ft ³	Relative Development Required
O ₂ /H ₂	456	22.2	Low
F ₂ /H ₂	468	40	High
FLOX/CH ₄	398	61.8	High
NTO/A-50	324	75.5	Low

Detail Design - The large propellant quantities required for upper stage boost also necessitate consideration of the propellant tankages configuration. Two designs are considered: the common bulkhead tankage integral with the vehicle primary structure (inner body structure used as tank side walls) and the separate fuel and oxidizer tanks supported from the vehicle primary structure.

Preliminary analysis indicates that for the M2-F2 vehicles, the integral tank design exhibits a volumetric efficiency about 20 percent greater than the separate tank configuration which results in a shorter and therefore lighter weight vehicle. An additional weight advantage results from eliminating the

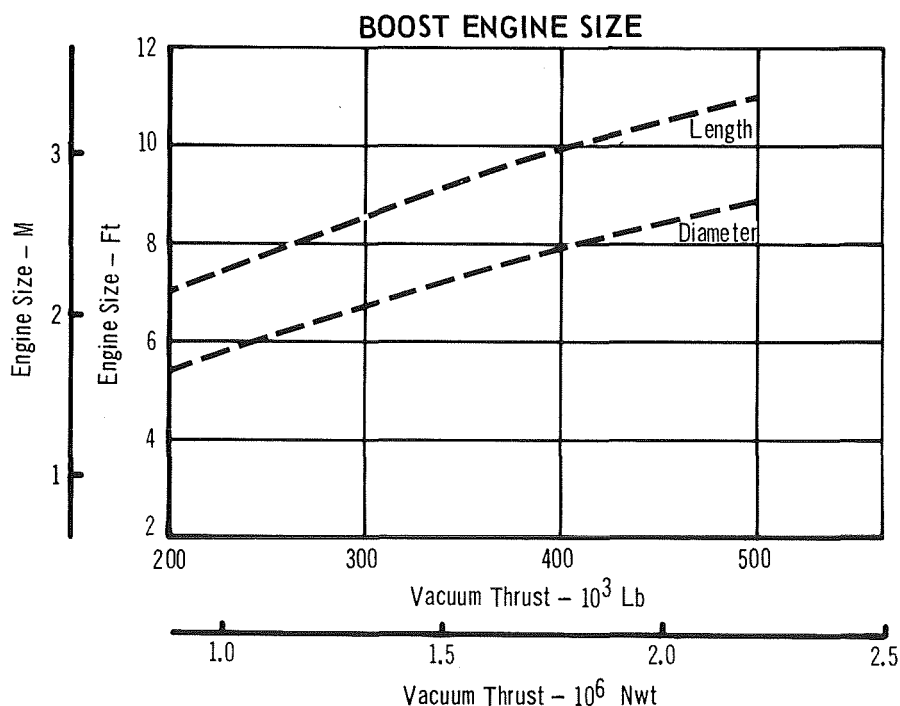


Figure 6-26

need for separate tank support structure. However, the M2-F2 cross section is not uniform (one side is essentially flat) and, therefore, with the integral tanks, tension straps are needed to reduce the internal bending loads within the structural frame (Reference Section 6.2.2). This constitutes a weight penalty relative to a separate, uniform shaped tank, but this weight penalty is small compared to the weight advantage from a shorter vehicle length and from elimination of the tank support structure. Therefore, a common bulkhead integral tank configuration will be used for the upper stage boost system on M2-F2 vehicles. With the ballistic shape, however, the vehicle diameter is very large at the aft body station where the boost propellant is located. If an integral tank configuration were used, the bulkheads would be essentially flat and located very close together relative to their diameter. Such a tank design is not considered practical; thus, for the ballistic vehicle, separate propellant tanks are used in the upper stage boost systems.

Concept Summary - The design variations of propellant type and engine nozzle design result in a total of five boost system concepts, a summary of which is presented in Table 6-16. As noted, the O_2/H_2 concept is selected as the baseline system. This concept typifies system having low development cost and risk in combination with relative high performance. A more detailed description of this baseline system is presented in Section 6.4.2.3.

6.4.2.3 Baseline Description - The baseline upper stage boost system configurations are shown in Figure 6-27. All vehicles incorporate a pump fed, O_2/H_2 propellant system having multiple, bell nozzle engines.

Vehicle configurations I/D and II/D utilize external, common bulkhead propellant tanks that are jettisoned after initial orbit insertion. For the fully integral vehicles, configurations E and F, the propellant tanks are located inside the crew module. The ballistic vehicles utilize separate tanks: a spherical oxidizer tank and a toroidal fuel tank. For the M2-F2 vehicle, propellant is contained in common bulkhead tanks that are integral with the inner body structure.

As noted in Figure 6-27, all ballistic vehicles require a hatch in the heat shield to allow boost engine operation. The hatch is open during launch but must be closed before the de-orbit maneuver is performed for a safe entry maneuver.

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Table 6-16
Concept Summary - Upper Stage Boost Propulsion System

Concept No.	Propellant	Engine Design	Vehicle Applications
US-1*	O_2/H_2	Bell	
US-2	F_2/H_2	Bell	
US-3	Flox/ CH_4	Bell	
US-4	NTO/A-50	Bell	

* Baseline System Concept

6.4.3 Orbital Maneuver Propulsion - Propulsion systems are required to perform all orbital maneuvers for rendezvous and docking with the space station and for descending from orbit to land at a predetermined site. Specifically, this group of maneuvers consists of:

- a. Ascent
- b. Docking
- c. Attitude Control
- d. Return Phasing
- e. De-orbit

The systems performance requirements, alternate design concepts, and baseline configurations are discussed below.

6.4.3.1 Orbital Maneuver Performance Requirements - Each of the specific orbital maneuvers requires a different level of system performance capability. Hence, these maneuvers and their propulsion system requirements are discussed separately.

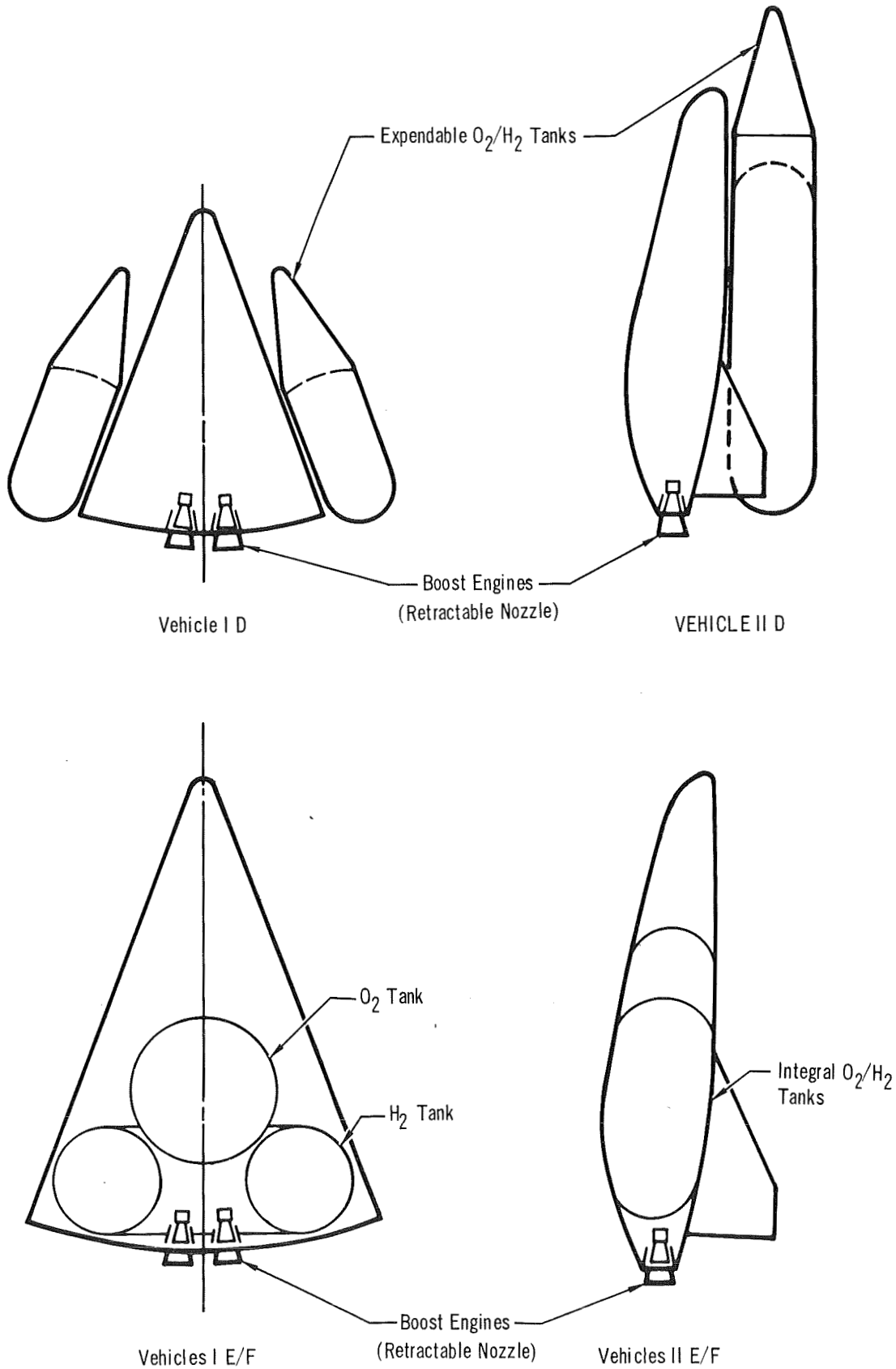
Ascent - The ascent maneuver is performed to translate the spacecraft from its position at initial orbit insertion to the proximity of the space station. Ascent is, therefore, a series of separate maneuvers, consisting of (a) a launch vehicle separation maneuver, (b) a Hohmann transfer from the initial injection

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BASELINE CONFIGURATIONS – UPPER STAGE BOOST SYSTEM

Figure 6-27



altitude to the altitude of the space station, and (c) a terminal rendezvous maneuver to close within a few hundred yards of the space station. At least five starts are required, ranging in burn time from about 10 seconds to about 300 seconds. The Hohmann transfer is the major maneuver, however, requiring about 80 percent of the total ascent ΔV .

The thrust level and total impulse requirements for ascent are shown in Figures 6-28 and 6-29. The indicated total impulse values were determined using a specific impulse of 300 seconds. Although propellants having much higher specific impulse performance than this will be considered for alternate design concepts in Section 6.4.3.2, the total impulse requirement will change very little. This is due to the low ΔV requirements of the ascent maneuver.

Docking - The docking maneuver is performed to translate the vehicle such that it makes contact with the docking port on the space station. This maneuver is characterized by the need for translational thrust in all axes (forward, aft, up, down, left, and right). A large number of systems starts are also required, approaching a pulsing mode of operation. Pulse widths, however, are long enough (0.5 to 2 seconds) to base performance considerations on steady-state operation.

Thrust level and total impulse requirements for the docking maneuver are presented in Figures 6-30 and 6-31 as a function of cargo weight. The total impulse requirements are based on a specific impulse of 300 seconds. The thrust level requirements are the total thrust needed in both the forward and aft directions. The lateral and vertical thrust level requirements are one-half of that needed for the forward or aft directions.

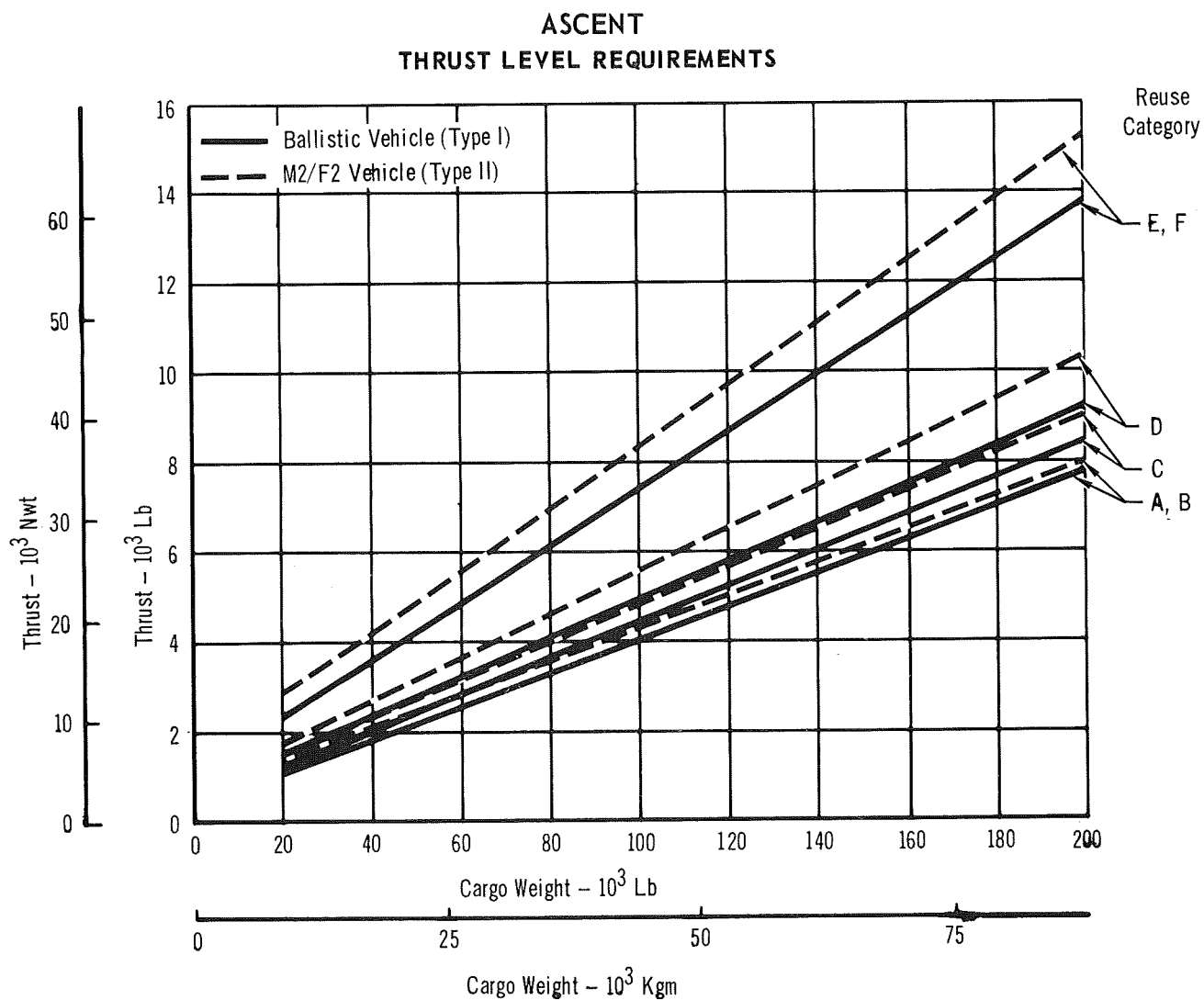
Attitude Control - Angular control of the vehicle attitude is required during the orbital phase of the mission to (a) damp out any disturbance torques that might occur, particularly those induced by the translation maneuvers, and (b) to periodically re-orient the vehicle for such purposes as antenna or thrust vector pointing and visual or optical sightings. The maneuver is characterized by the need for multiaxis thrust (pitch, yaw, and roll) and a pulsing mode.

The total impulse required for attitude control is presented in Figure 6-32 as a function of cargo weight. Specific thrust level magnitudes will be calculated in Task 6, using the sizing model to define the required inertia

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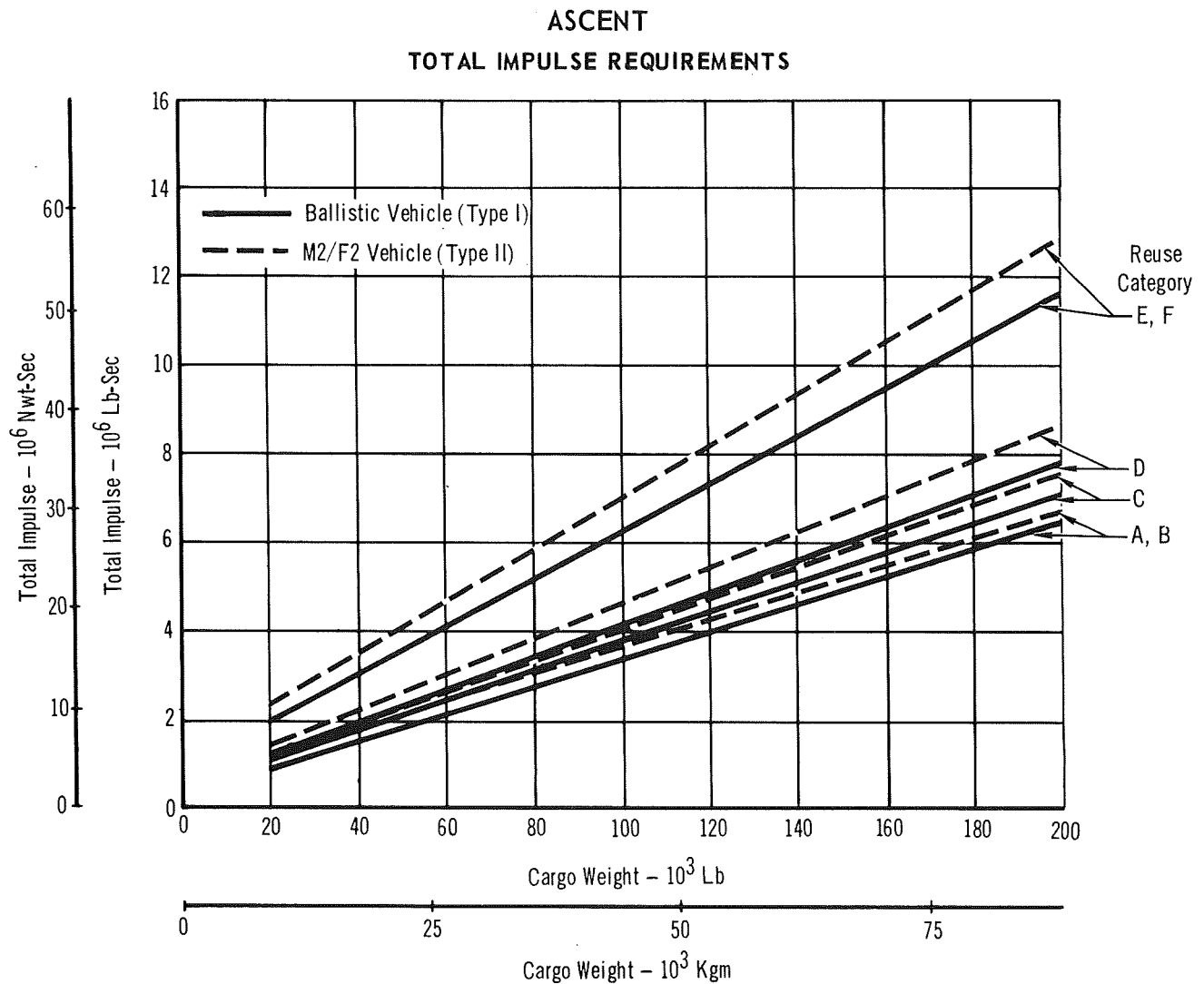
Figure 6-28



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Figure 6-29



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Figure 6-30

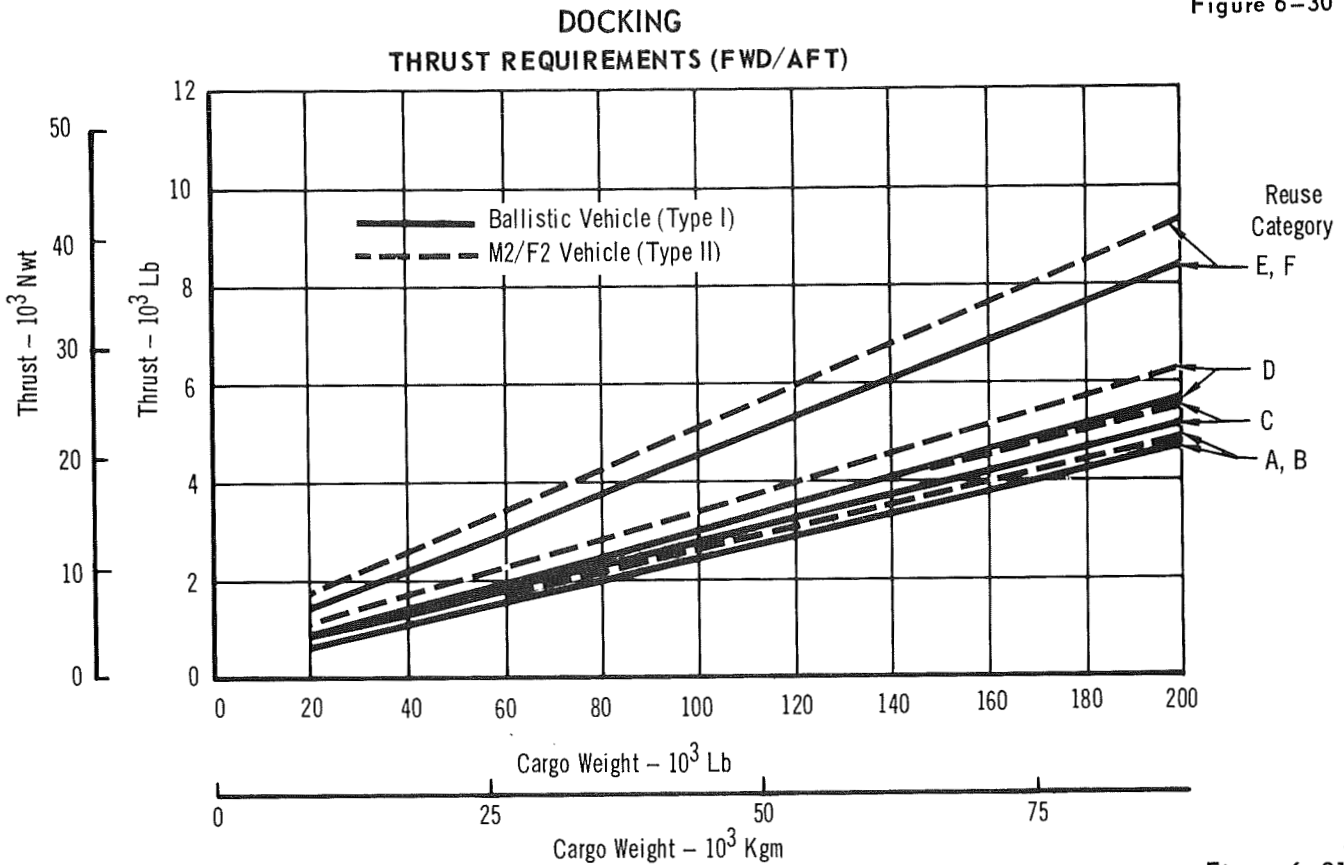
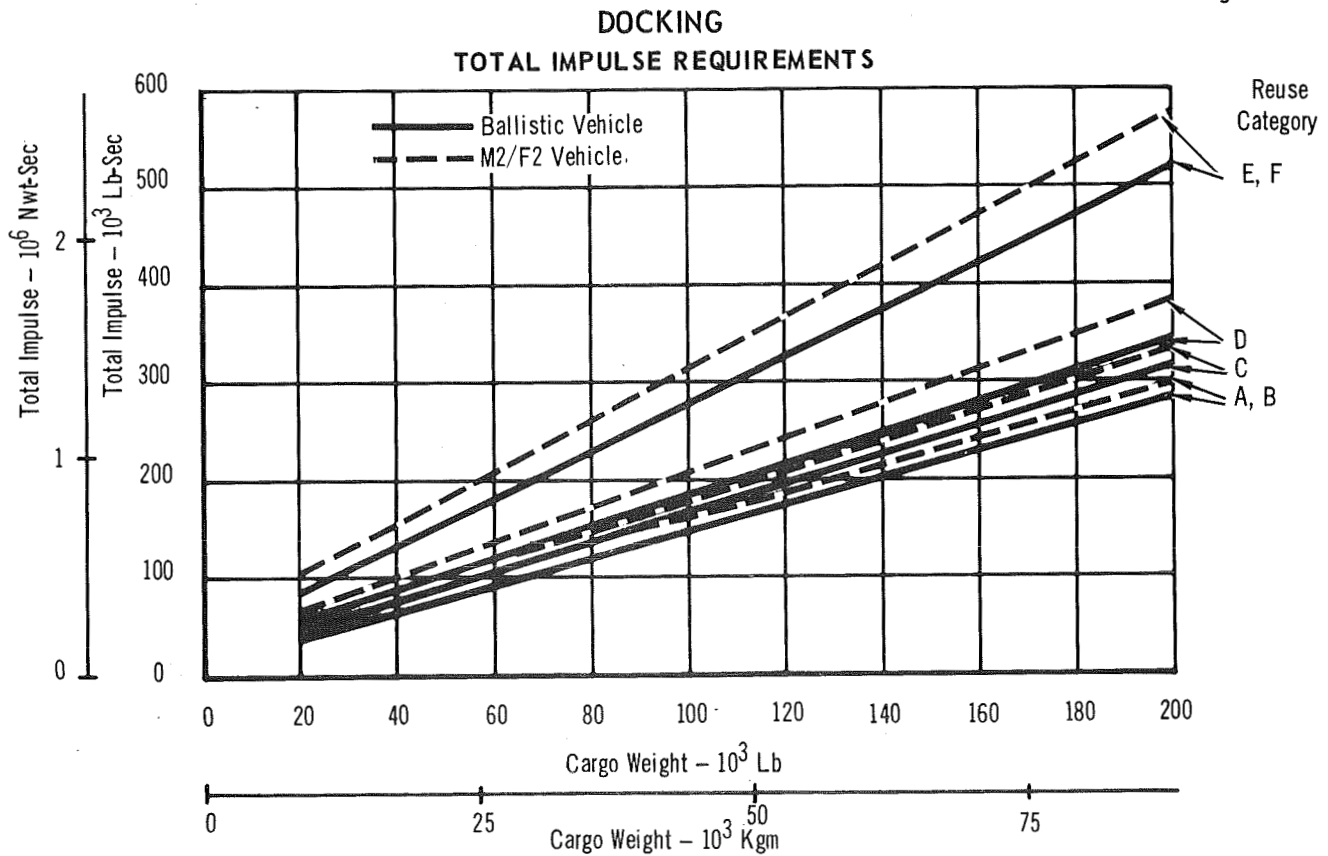


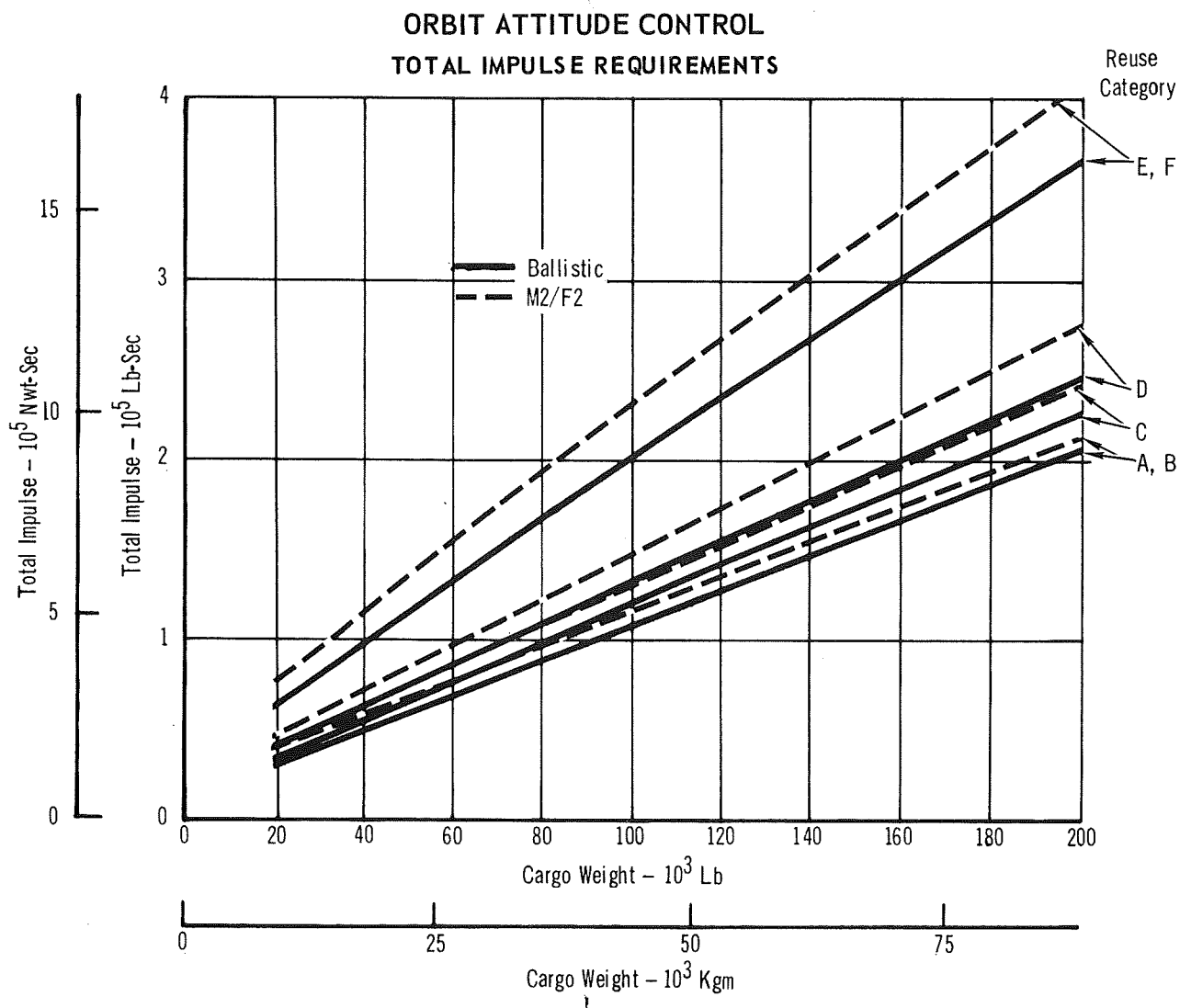
Figure 6-31



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Figure 6-32



characteristics of the vehicles. Preliminary estimates, however, indicate that for a given vehicle configuration, increasing the cargo weight from 20,000 to 200,000 pounds will cause the thrust per axis to increase by a factor of from 10 to 12. This estimation assumes that the angular accelerations, the engine moment arm (as a percent of length), and the ratio of radius of gyration to vehicle length all remain constant. It is also estimated that the pitch or yaw thrust level per direction will range from a few hundred pounds for the smaller vehicles to on the order of five thousand pounds for the larger vehicles.

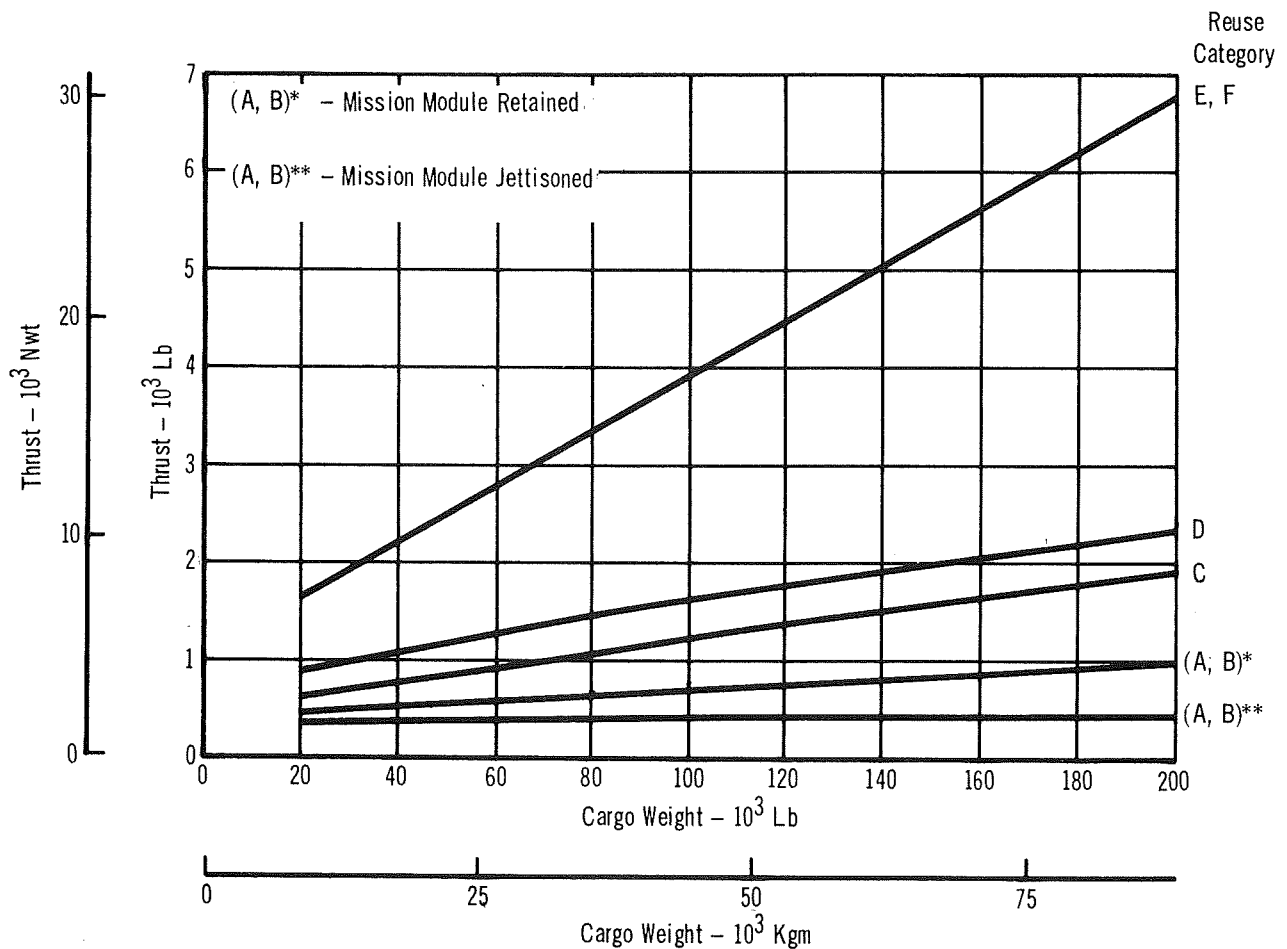
Return Phasing - Return phasing is an orbit period change maneuver, performed to alter the vehicle ground track such that it passes near or through the desired landing site. This maneuver is used to allow completion of the return phasing of the mission within a specified time (nominally 24 hours). Essentially, it is a propulsive augmentation to the vehicle's aerodynamic maneuver capability. Ordinarily, only the ballistic vehicles require such a maneuver; the M2-F2 vehicles have a sufficiently high hypersonic L/D to reach the landing site without phasing. Only for the special case of a very fast return (two hours) as discussed in Section 4.5, does the M2-F2 require phasing. The performance requirements for this special case are not considered here. Return phasing is a single burn maneuver of about 300 seconds.

The more detailed analysis of landing site number/location, discussed in Section 5.0 of Book 2, considers a baseline operational mode in which a phasing maneuver is not required. The potential phasing requirements are included here to make provisions in the optimization model that will allow a tradeoff between phasing and the recovery site characteristics to be made in Task 6.

Thrust level and total impulse requirements for the ballistic vehicle return phasing maneuver are presented in Figures 6-33 and 6-34. Impulse requirements are based on a specific impulse of 300 seconds. Two versions of the modular configuration are shown, one in which the empty mission module weight is included and another where the empty module is jettisoned before phasing. The former version corresponds to the case where the phasing propulsion system is located in the mission module. The latter case reflects phasing systems installation in a special adapter section between the mission module and entry vehicle.

Figure 6-33

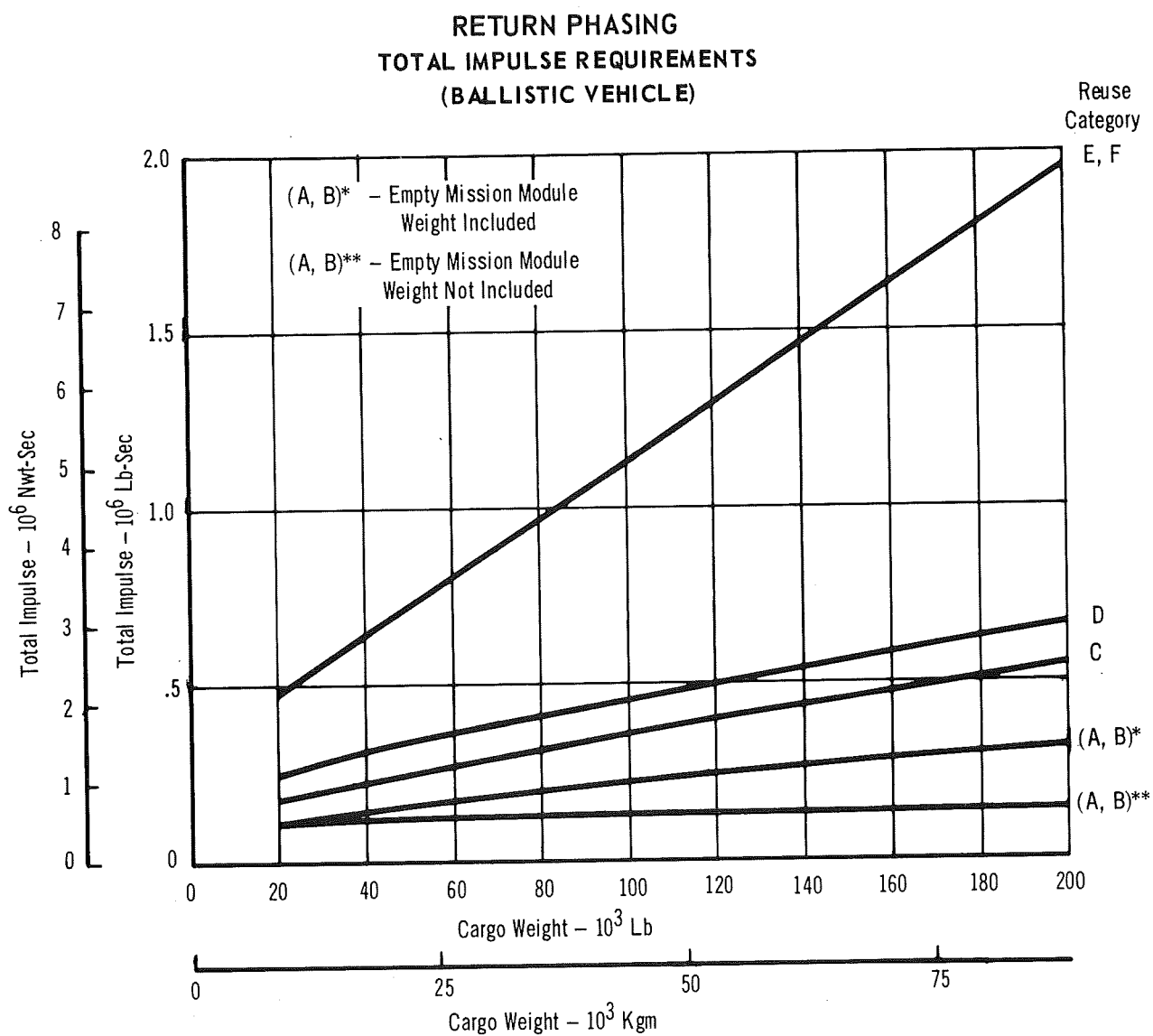
RETURN PHASING
THRUST LEVEL REQUIREMENTS
(BALLISTIC VEHICLE)



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Figure 6-34



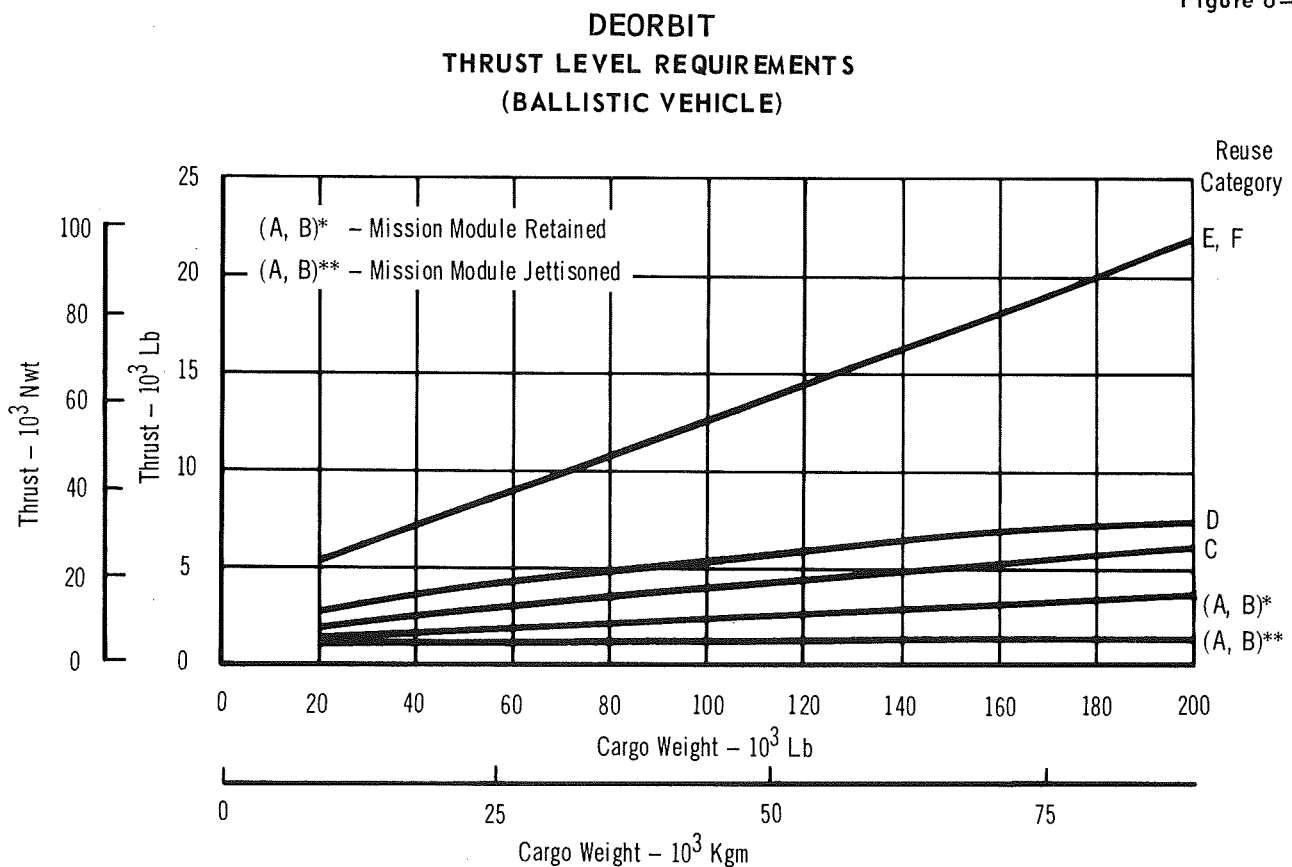
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De-orbit - The de-orbit maneuver is performed to modify the vehicle orbit such that it enters the atmosphere at a predetermined flight path angle and velocity. The thrust-to-weight ratio requirement is the estimated minimum needed to ensure acceptable touchdown point dispersions. In an emergency, a safe entry can be performed with less than one-half the nominal thrust requirement. De-orbit is a single burn maneuver of about 200 seconds.

De-orbit thrust and total impulse requirements are shown in Figures 6-35 and 6-36 for the ballistic vehicles and in Figures 6-37 and 6-38 for the M2-F2 vehicles. The two modular configurations shown for each vehicle reflect the location of the deorbit system, either in the mission module or in a special adapter between the mission modules and the entry vehicle. In the later case, the mission module would be jettisoned before de-orbit.

Figure 6-35

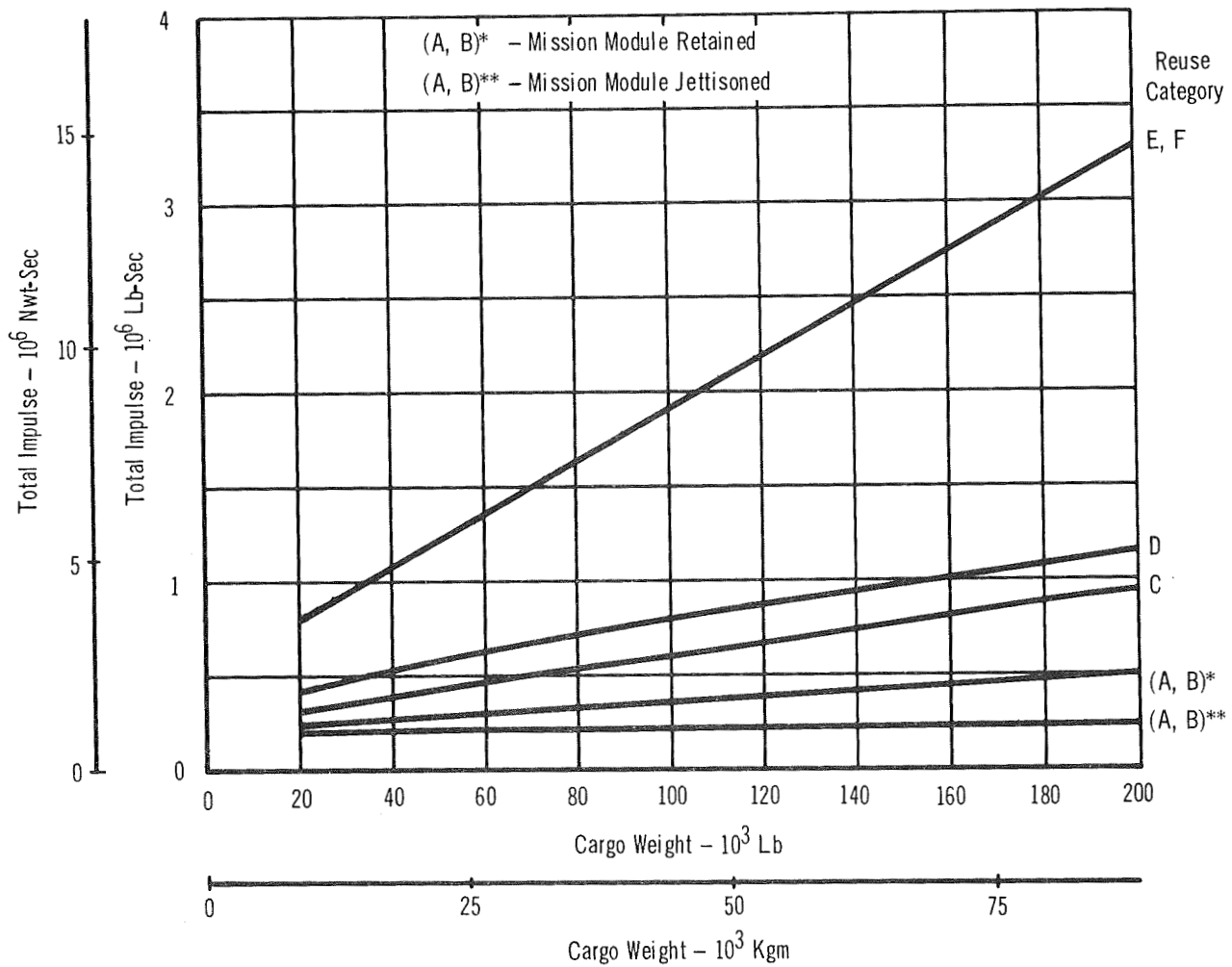


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Figure 6-36

**DEORBIT
TOTAL IMPULSE REQUIREMENTS
(BALLISTIC VEHICLE)**

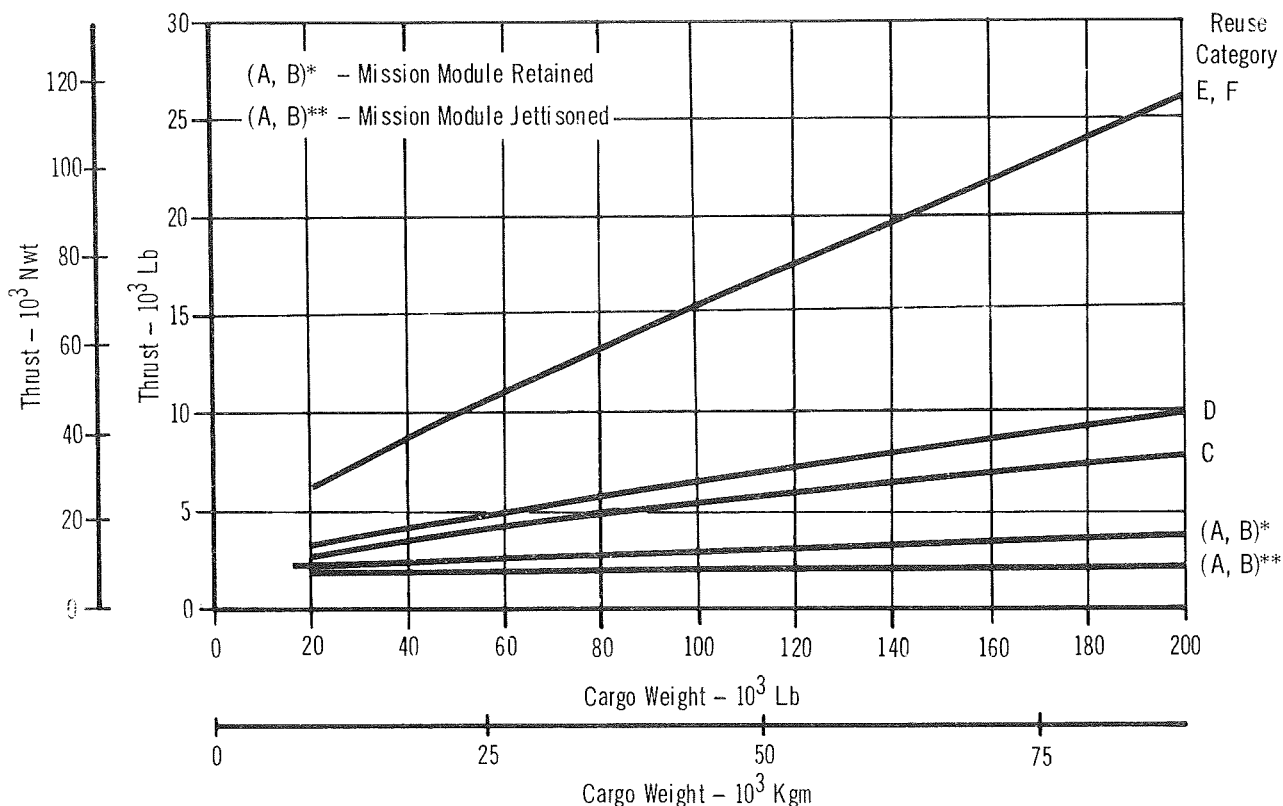


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DEORBIT THRUST LEVEL REQUIREMENTS (M2/F2 VEHICLE)

Figure 6-37



6.4.1.3 Systems Concepts - Orbital Maneuver - As just discussed, the primary propulsive maneuvers during the orbital phase of the mission are: ascent, docking, attitude control, return phasing, and de-orbit. Major factors considered in the selection of systems concepts to perform these maneuvers are: (a) propellant type, (b) number of independent systems, and (c) systems design characteristics (engine arrangement and cooling method, type of feed subsystems, etc.). These factors and their influence in selecting systems concepts are discussed below.

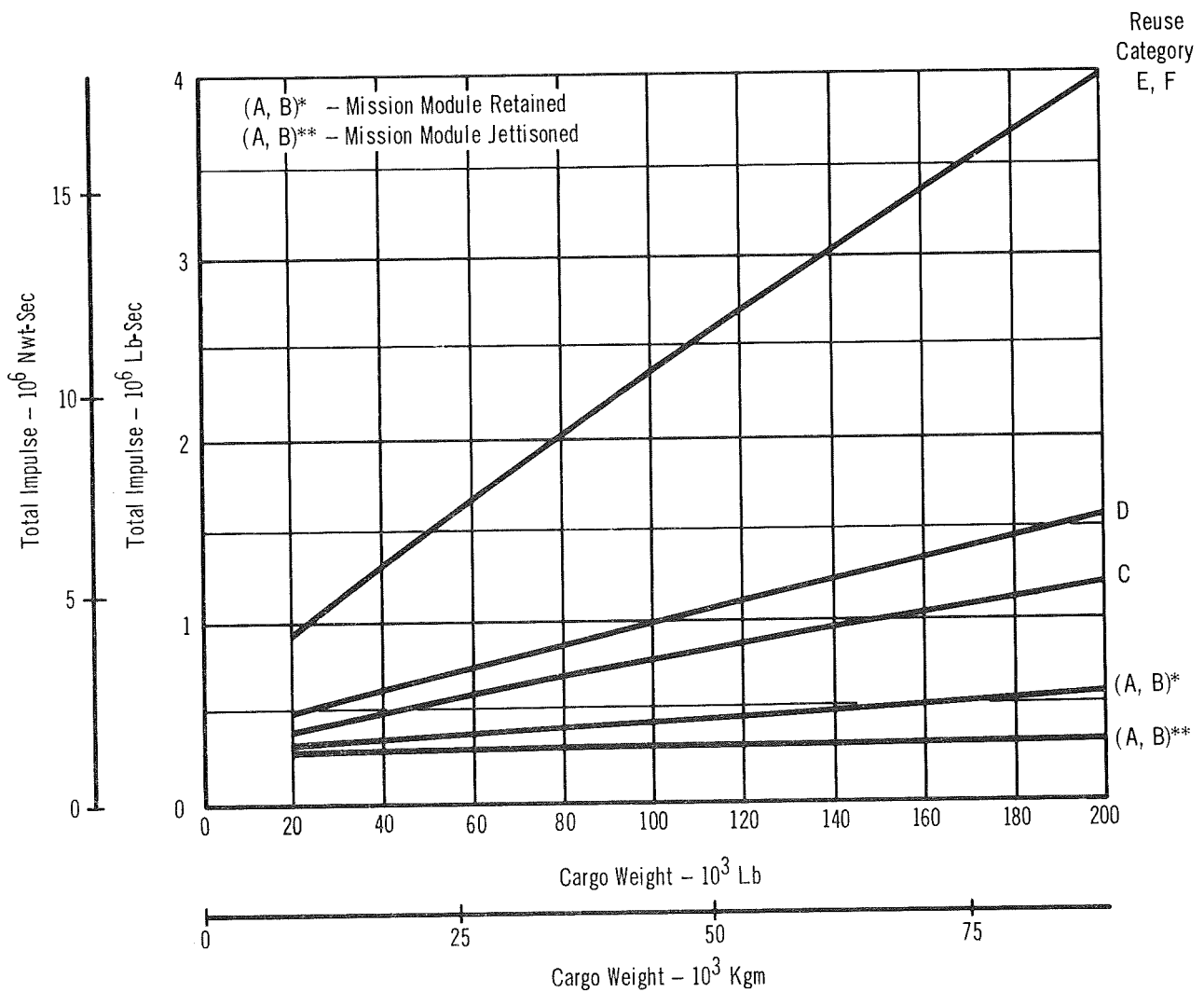
Propellants - Both solid propellants and liquid bipropellants (storable and cryogenic) are considered for use in the orbit maneuver systems. Hybrids have not been so considered because of the large amount of development anticipated relative to any weight or operational advantage that might result. Neither are monopropellants considered; the resultant higher system weights due to their lower specific impulse, relative to bipropellants, is expected to offset any cost advantage that might result from reducing the number of systems components.

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Figure 6-38

DEORBIT TOTAL IMPULSE REQUIREMENTS (M2/F2 VEHICLE)



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The first step in propellant selection is to determine what propellant is applicable for the given maneuvers. The main determining factor is the type of systems operation (pulsing, restartable, or single burn). Table 6-17 presents the propellant and maneuver combinations that are considered. Solid propellants are considered only for single start maneuvers. Although restartable or pulsing solid motors are technically feasible, the development cost and risk is considered excessive for any possible performance or recurring cost advantages. Liquid cryogenic propellants are applicable for both single and multistart maneuvers typical of ascent, phasing, and de-orbit. They are not considered however, for pulsing, attitude control applications. The small potential weight savings due to low systems total impulse requirements is not expected to offset the cost of developing positive expulsion tanks and pulsing engines for use with cryogenic propellants.

Table 6-17
Propellant Applicability - Orbital Maneuvers

Maneuvers	Type Operations	Propellant
Ascent	Restartable	Liquid - Storable - Cryogenic
Docking and Attitude Control	Pulsing	Liquid - Storables
Phasing and De-orbit	Single Burn	Liquid - Storables - Cryogenic Solid

Storable liquid propellants are considered for all maneuvers in that the associated hardware has been developed and is currently in use for such applications.

The specific combinations considered for each general type of propellant are:

- Solids - Ammonium Perchlorate/Polybutadiene/Aluminum (AP/PB/AL)
- Storables - Nitrogen Tetroxide/Monomethylhydrazine (NTO/MMH)

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- c. Cryogenics - Oxygen/Hydrogen (O_2/H_2)
 - Fluorine/Hydrogen (F_2/H_2)
 - Fluorine (87%), Oxygen (13%)/Methane (Flox/ CH_4)

The solid and storable liquid combinations are selected as having development and performance characteristics typical of their respective propellant classes. However, a number of cryogenic combinations, covering a wide range of development requirements and performance capabilities, can be used. To further examine these combinations, a propellant weight comparison is presented in Figure 6-39. The storable propellant combination is shown only for reference. As indicated, the propellant quantities are small relative to the total vehicle weight. Thus, propellant volume is not a critical parameter nor is specific impulse of overriding importance. The high bulk density of Flox/ CH_4 , therefore, offers little advantage while its lower Isp causes a slightly higher propellant weight relative to the other cryogenic combinations. Neither is the slight weight advantage of F_2/H_2 , over O_2/H_2 , considered enough to offset the added development cost and risk of fluorine propellants. For these reasons, O_2/H_2 is selected as the typical cryogenic propellant for the applicable orbit maneuvers.

System Arrangements- A variety of orbital maneuvers is required, some involving alternate types of propellant. Therefore, the use of more than one propulsion system must be considered. Based on the examination of numerous propellant and system combinations, four typical arrangements of from one to three separate systems are selected for additional evaluation in Task 6. These system arrangements concepts are summarized in Table 6-18. Concept 1 requires the fewest number of components (single system) while concept 2 represents the minimum weight, all liquid system (cryogenic propellants). Concepts 3 and 4 are the same as Concepts 1 and 2, respectively, except that solid motors are utilized for de-orbit.

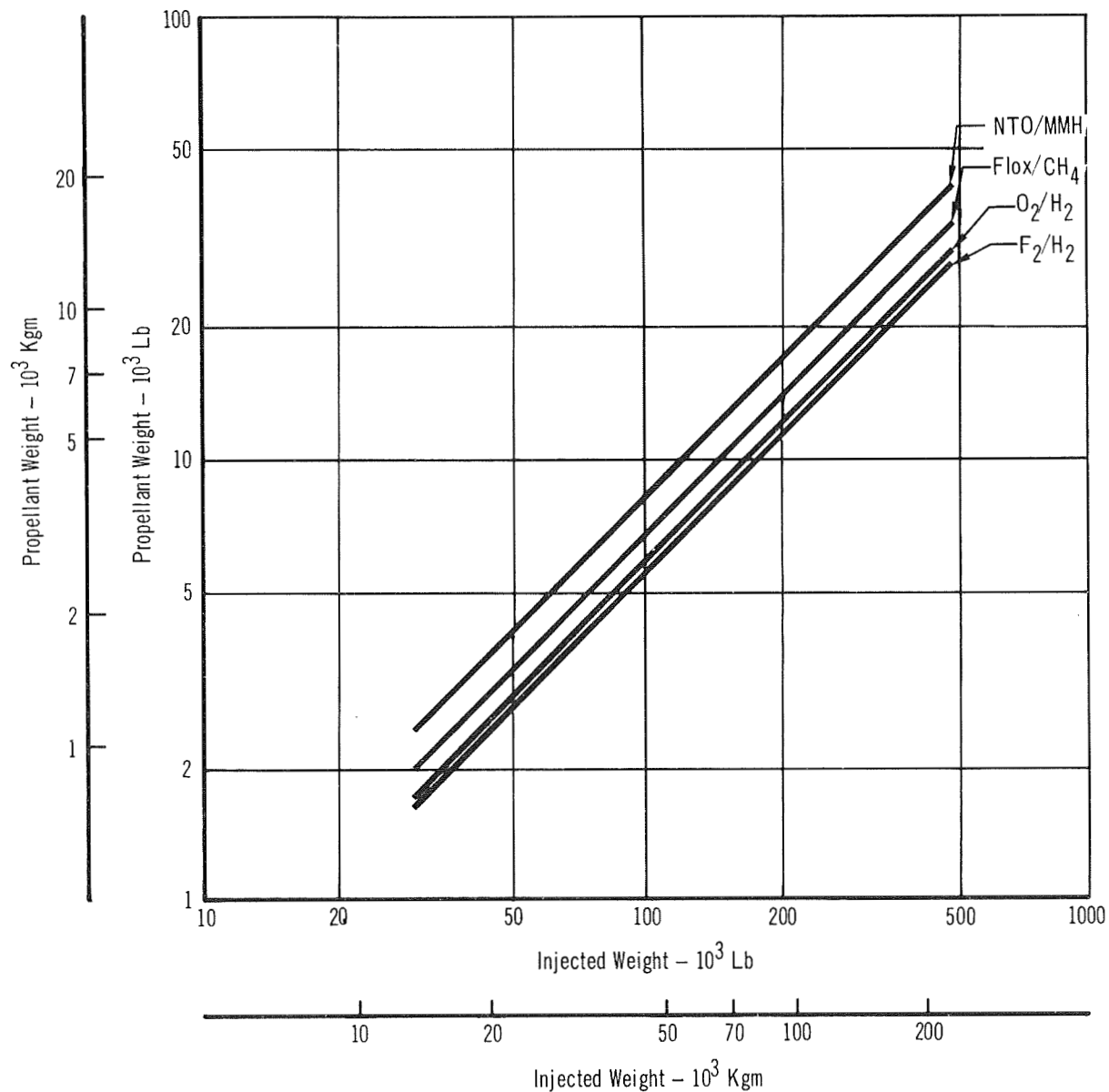
A much larger number of arrangement concepts could be considered; additional concepts can receive limited evaluation, depending on the results of the Task 6 analysis. However, the selected concepts are considered sufficiently varied to accomplish the purpose of this study without requiring an excessive amount of analysis or computer time.

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Figure 6-39

**PROPELLANT WEIGHT COMPARISON
(ASCENT MANEUVER)**



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Table 6-18
System Arrangements - Orbital Maneuvers

Arrangement Concept	System	Maneuver	Propellant
1	A	All	NTO/MMH
2	A B	Docking/AC Ascent/Phasing/Deorbit	NTO/MMH O ₂ /H ₂
3	A B	Ascent/Docking/Phasing/AC Deorbit	NTO/MMH Solid
4	A B C	Docking/AC Ascent/Phasing Deorbit	NTO/MMH O ₂ /H ₂ Solid

AC= Attitude Control

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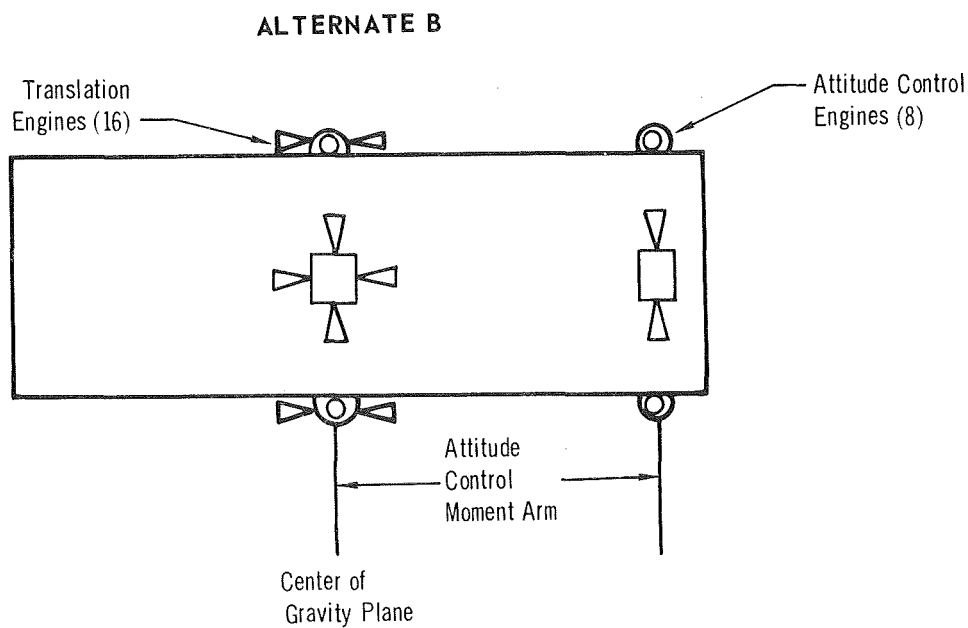
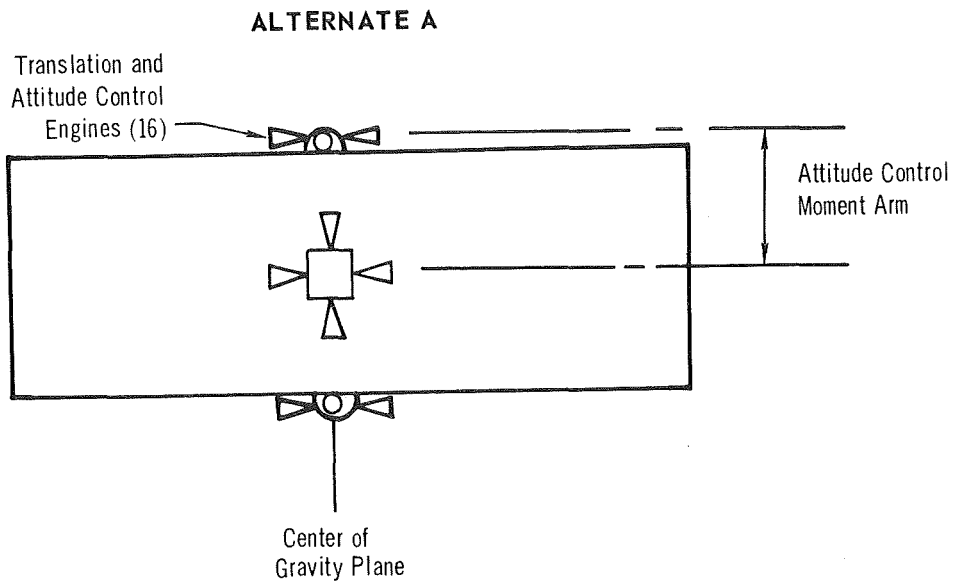
Detailed Design - Preliminary evaluation indicates that once the propellant type and system arrangements are established, the major design consideration is the choice of engine configuration to be used for the docking and attitude control functions.

The alternate engine configurations, considered for modular vehicles, are shown in Figure 6-40. The single engine group (Alternate A) uses the same engines for both translation maneuvers and attitude control. The dual engine group (Alternate B) employs a separate set of engines for each function. The main differences between these configurations is the number of engines required and the length of the moment arm available for attitude control. Alternate A requires fewer engines but has a shorter attitude control moment arm than Alternate B. For small cargo quantity both alternatives require the same maximum thrust level per engine. Therefore, since Alternate A has the fewest number engines, it is the preferred concept. At large cargo quantities, however, Alternate A requires much higher engine thrust levels than Alternate B to provide the necessary attitude control. This is a result of the difference in length of the available moment arm and the fact the vehicle inertia increases more rapidly than vehicle weight as cargo is added. In this case, a trade-off exists between using a relatively few, high thrust engines (Alternate A) versus a lower number of lower thrust engines (Alternate B). Hence, for modular vehicles, both engine arrangement concepts are retained for further evaluation in Task 6. For integral vehicles, consideration is limited to a dual engine group similar to Configuration B. This allows use of the same engines for both orbit and entry attitude control.

Concept Summary - The propellant, systems arrangement, and engine configuration alternatives, identified above, result in four system concepts for integral vehicles and eight for modular vehicles. These concepts are summarized in Table 6-19. As indicated, a single, NTO/MMH system is selected for the baseline integral vehicles while for modular vehicles, a solid propellant system for de-orbit and an NTO/MMH system for the remaining maneuvers is chosen as the baseline. Use of storable propellants is based on the expectation that the lower cost of this system will more than offset the weight advantage of an O_2/H_2 system. A solid propellant de-orbit system is selected for the modular vehicles because of weight. The motors can be installed as a separate adapter (Gemini fashion), thus eliminating the need to de-orbit the empty mission module. These baseline systems are further described below.

Figure 6-40

ALTERNATE ENGINE CONFIGURATIONS – ORBITAL MANEUVERS



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Table 6-19
Concept Summary - Orbital Maneuver

Concept No.	System - Function	Propellant	Engine Groups	Vehicle Application
OM-1	A - All functions	NTO/MMH	Single	I/A, B; II A/B
OM-2*	A - All functions	NTO/MMH	Dual	All
OM-3	A - Docking/AC B - Ascent/Phasing/Deorbit	NTO/MMH O ₂ /H ₂	Single Single	I/A, B II/A, B
OM-4	A - Docking/AC B - Ascent/Phasing/Deorbit	NTO/MMH O ₂ /H ₂	Dual single	All
OM-5**	A - Ascent/Docking/Phasing/AC B - Deorbit	NTO/MMH Solid	Single Single	I/A, B II/A, B
OM-6	A - Ascent/Docking/Phasing/AC B - Deorbit	NTO/MMH Solid	Dual Single	All
OM-7	A - Docking/AC B - Ascent/Phasing C - Deorbit	NTO/MMH O ₂ /H ₂ Solid	Single Single Single	I/A, B II/A, B
OM-8	A - Docking/AC B - Ascent/Phasing C - Deorbit	NTO/MMH O ₂ /H ₂ Solid	Dual Single Single	All

AC = Attitude Control

* Baseline concept selection - Integral vehicles (C, D, E, F)

** Baseline concept selection - Modular vehicles (A, B)

6.4.3.3 Baseline Description - The orbital maneuver system baseline configurations are shown in Figure 6-41. A typical systems schematic is presented in Figure 6-42. The modular vehicles incorporate a solid propellant deorbit system that is located in a special adapter between the crew and mission modules. The mission module is jettisoned just before the deorbit maneuver. The remainder of the orbital maneuvers, including attitude control, are performed by a pressure-fed, NTO/MMH propellant system in the mission module. Thrust is provided by four external groups of engines or "quad packs" installed in the plane of the vehicle center of gravity.

Integral vehicles use a single, pressure-fed NTO/MMH propellant system located in the crew module for all orbital maneuvers. Two sets of engines are provided, one for translation maneuvers and another for attitude control. The translation engine arrangement is the same as that used for the modular vehicles, except that the quad packs are retracted inside the crew module prior to entry. This is necessary to protect the engines so that they can be reused. The attitude control engines are located at the vehicle aft end and are used during entry, as well as for orbital operations. Two rings of engines are installed, either one of which can perform both the orbit and entry control functions. The second ring is used as a back-up for the entry maneuver.

For integral vehicles, the propellant storage subsystems consist of two separate groups of tanks. One group is used to store the deorbit propellant, while the other contains the propellant needed for all other orbit maneuvers. The deorbit propellant is isolated to ensure its availability for this critical maneuver.

6.4.4 Entry Attitude Control - An entry attitude control propulsion system is required to control the spacecraft attitude during the entire entry phase of the mission. This includes the time period immediately following deorbit and the subsequent atmospheric flight portion of the descent trajectory. Performance requirements and design concepts for this system are discussed here.

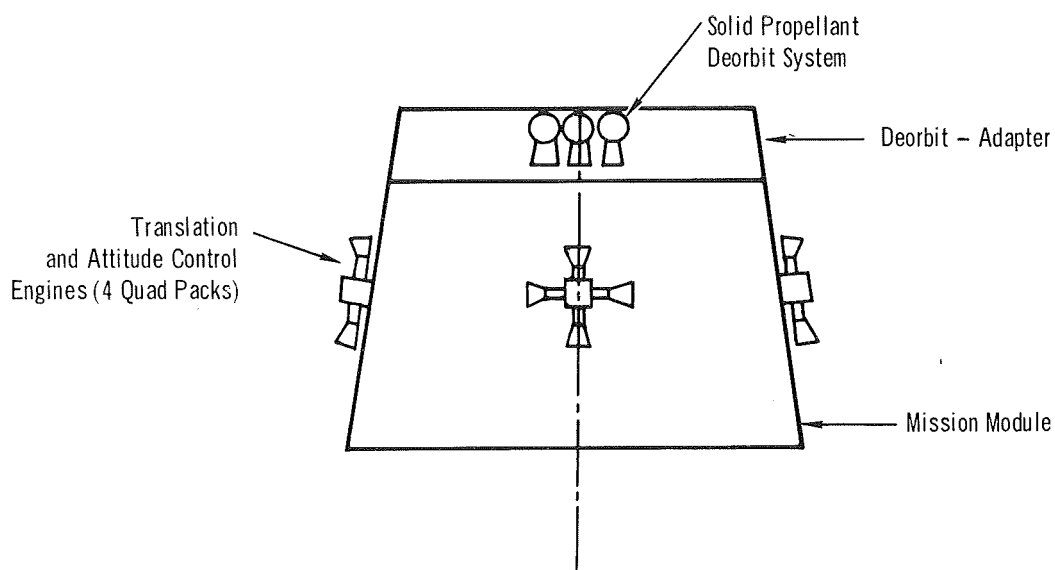
6.4.4.1 Performance Requirements - Entry attitude control is required (a) to acquire the appropriate spacecraft orientation before entering the sensible atmosphere and (b) to provide the attitude control necessary to fly the desired entry trajectory. These maneuvers are characterized by the need for a system with multi-axis thrust (pitch, yaw, and roll) and a pulsing mode of operations.

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Figure 6-41

BASELINE CONFIGURATIONS – ORBITAL MANEUVER SYSTEM



Vehicles I/A, B and II/A, B

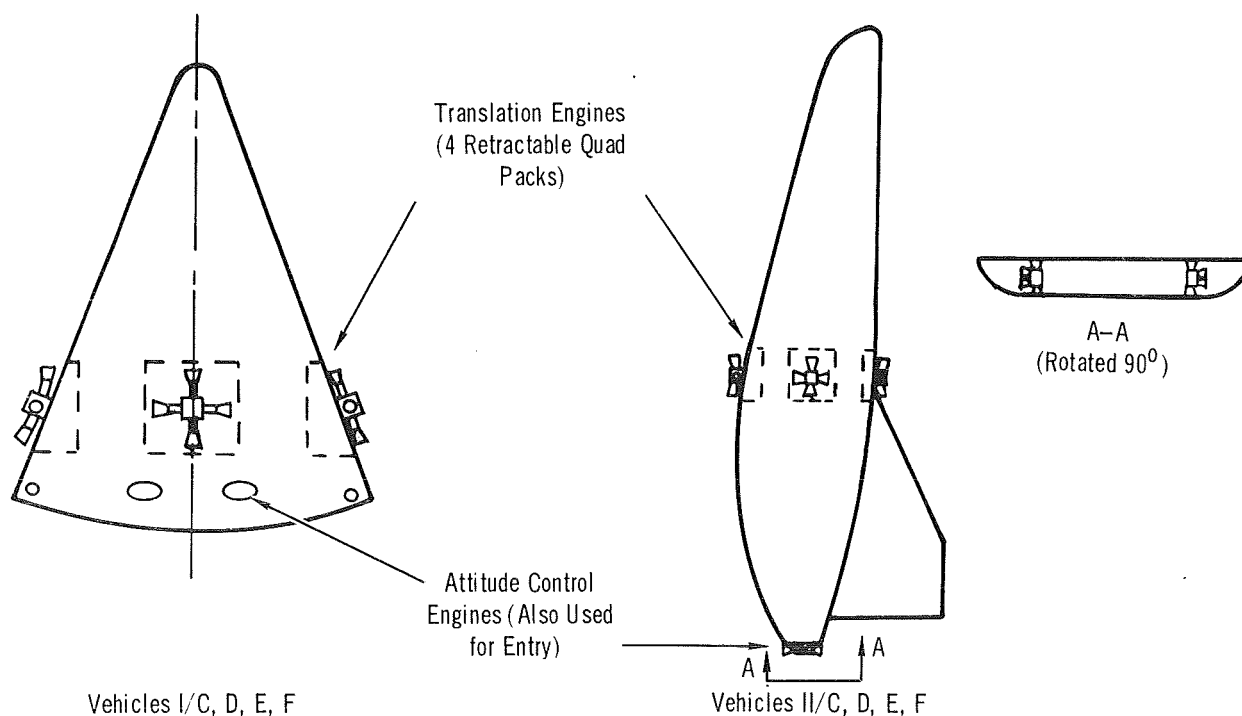
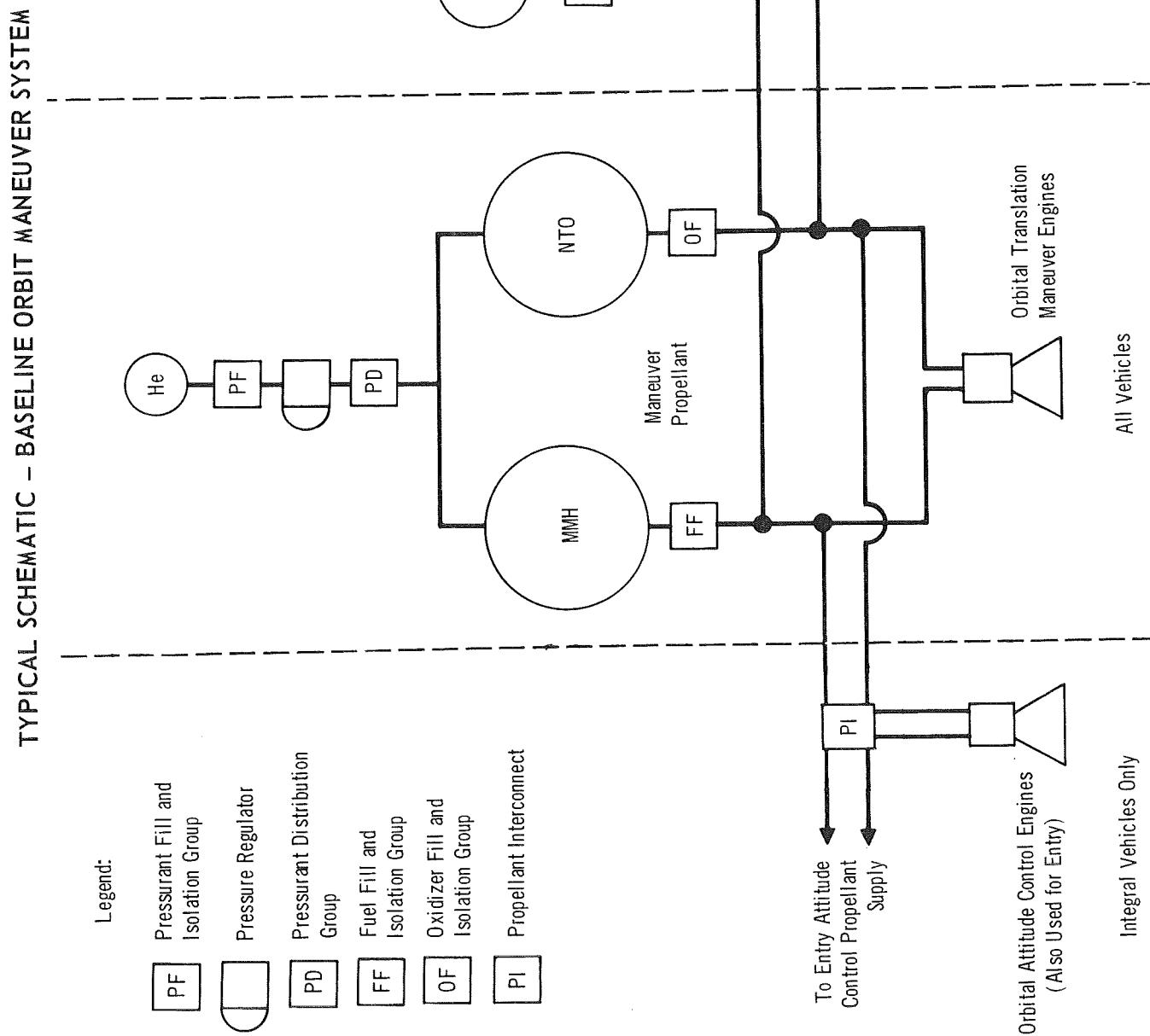


Figure 6-42



The total impulse required for entry attitude control is shown in Figure 6-43. Specific thrust level determination will be made in Task 6 using the computerized sizing model to obtain the necessary vehicle inertia data. It is estimated, however, that these thrust levels will range from less than 100 pounds to about 5000 pounds, depending on cargo weight and vehicle configuration.

6.4.4.2 System Concept - Only one design concept is considered for entry attitude control: a pressure fed, NTO/MMH system similar to those developed for Gemini and Apollo. Alternate propellant or feed subsystem designs were not investigated. The potential weight or cost savings in such a low total impulse system were considered too small to warrant such an evaluation in this study.

The system location and engine arrangement is influenced by the choice of orbit maneuver system (Section 6.4.3) for the vehicle types. On integral vehicles, the orbit attitude control engines are also used for entry attitude control. For modular vehicles, a separate engine group for entry attitude control is installed.

Additional system design characteristics are presented in following baseline system descriptions.

6.4.4.3 Baseline Systems Descriptions - The baseline system configuration for entry attitude control is shown in Figure 6-44. Two, pressure fed, NTO/MMH propellant systems are installed; one of which is redundant. A complete backup system is provided because this maneuver is critical for crew safety. The modular vehicle entry systems are completely independent of all other spacecraft propulsion systems. On integral vehicles, although the entry propellant is stored in independent tanks, the thrust required is provided by the orbit attitude control engines.

6.4.5 Landing Assist - The terminal portion of the landing maneuvers is critical for touchdown point accuracy and impact velocities for the M2-F2 and ballistic vehicles, respectively. Thus, a landing assist propulsion system is considered for both vehicle geometries. System performance requirements and design concepts are discussed in the subsequent paragraphs and followed by a description of the baseline configuration.

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Figure 6-43

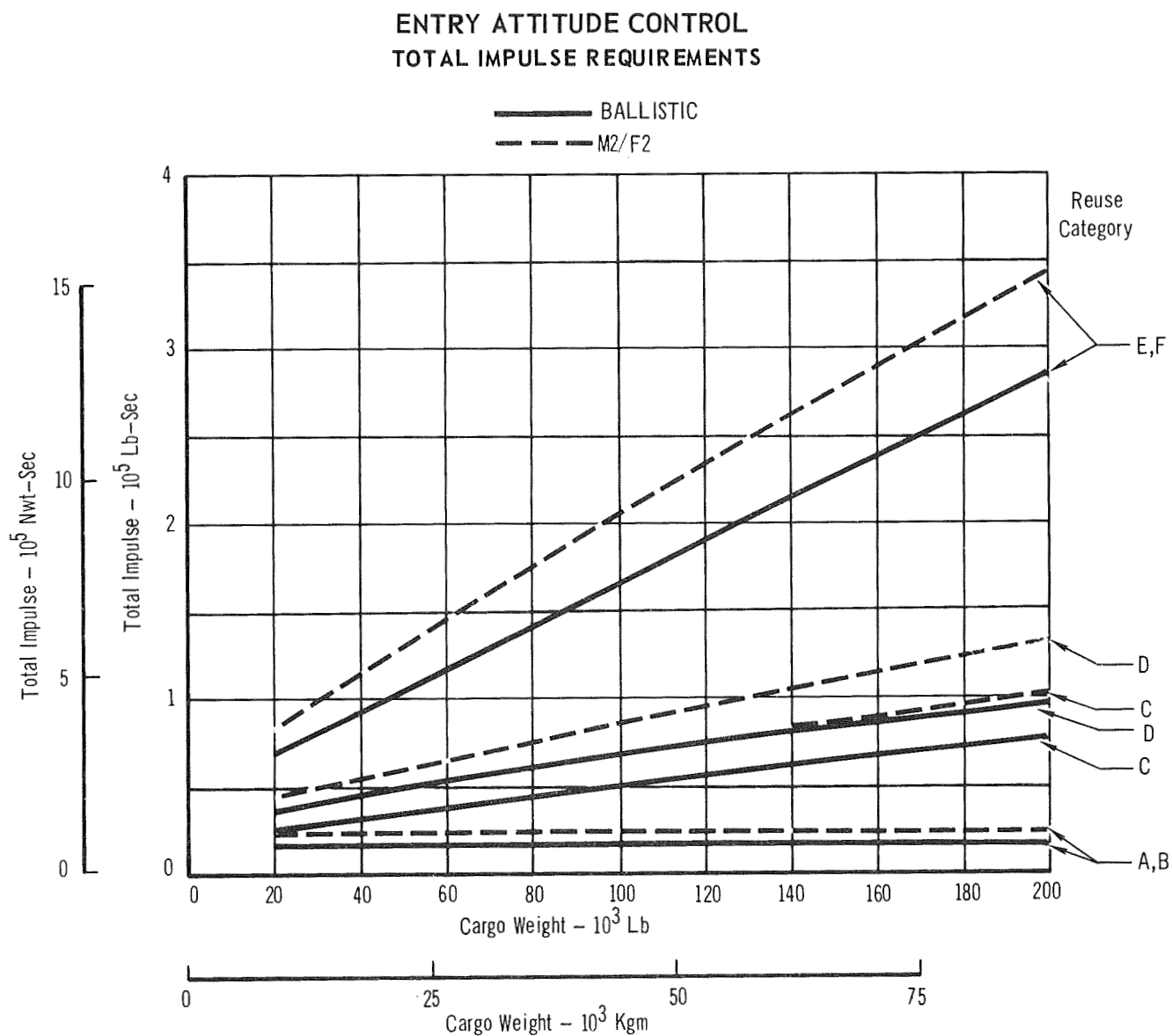
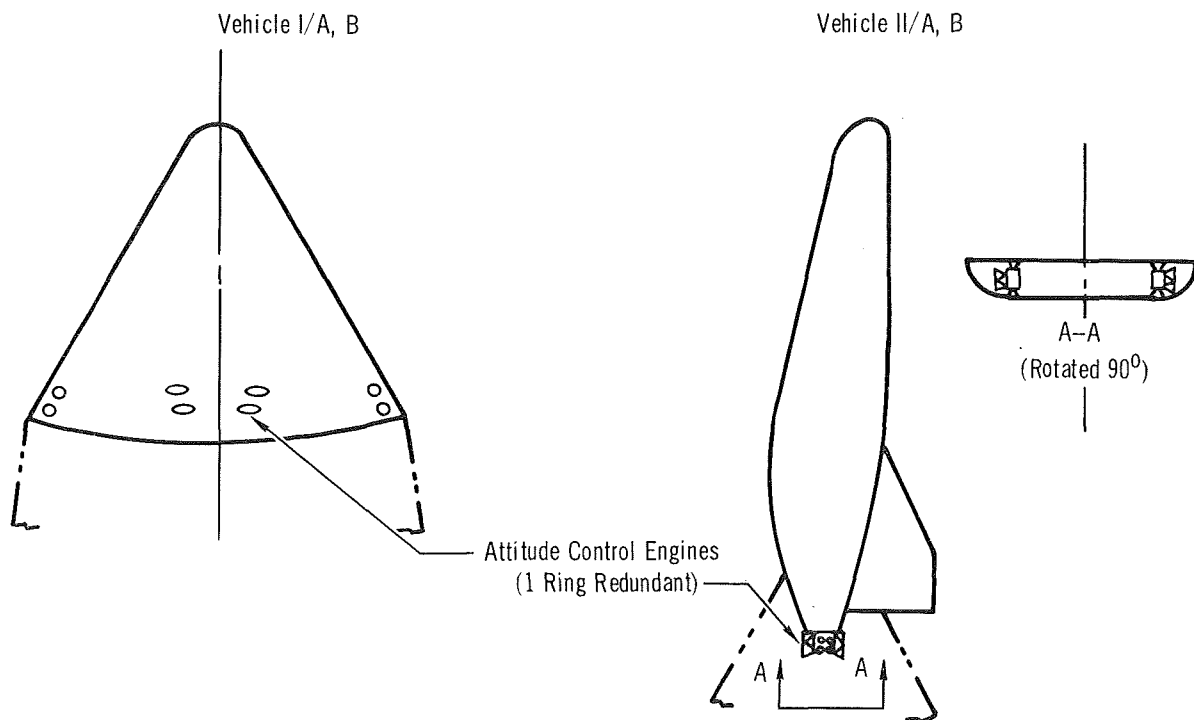


Figure 6-44

BASELINE CONFIGURATIONS – ENTRY ATTITUDE CONTROL



Note: Same configuration for integral vehicles except orbit attitude control engines used

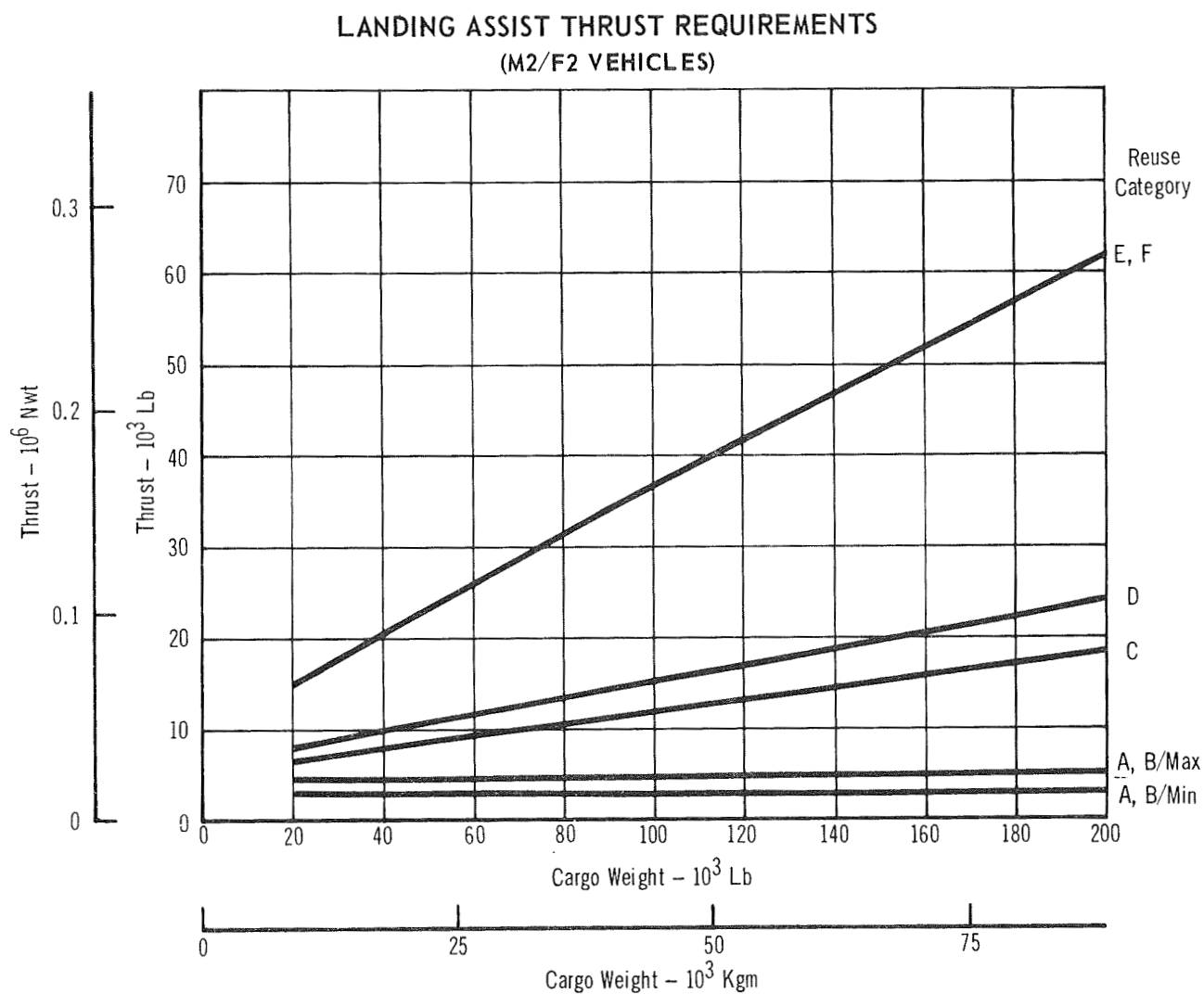
6.4.5.1 Performance Requirements – Landing assist for the M2-F2 is a glide range extension maneuver in which longitudinal thrust is provided, after the final flare, to augment the vehicles's subsonic lift to drag ratio (L/D). Although not theoretically madatory, this maneuver capability reduces the accuracy requirements for both the guidance system and the pilot's judgment. Glide range extension is a single burn maneuver requiring relatively low thrust levels and long burn times. Specific thrust and total impulse requirements are shown in Figures 6-45 and 6-46.

Landing assist of the ballistic vehicle is a vertical velocity attenuation maneuver, accomplished just before touchdown. Thrust is applied to reduce the vertical velocity at impact from that of parachute terminal velocity down to an acceptably low (20 ft/sec) value. Only the integral ballistic vehicles (categories C, D, E, and F) require such a maneuver. As discussed in Section 4.7, their landed weights are so high that to achieve acceptable impact

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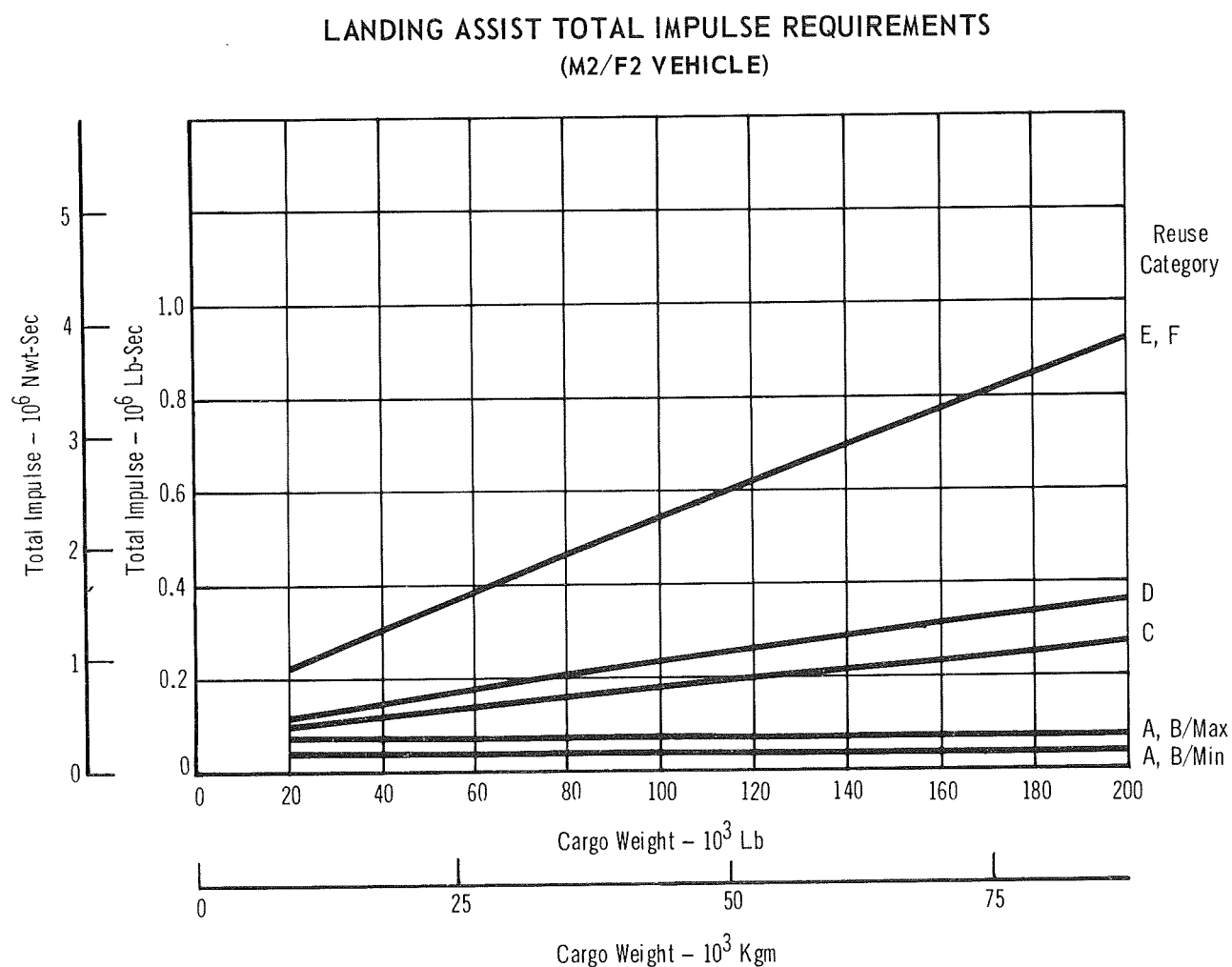
Figure 6-45



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Figure 6-46



velocities with parachutes alone is not considered feasible. Vertical velocity attenuation is a single burn maneuver requiring high thrust levels and a very short (~ 1 sec) burn time. The thrust and total impulse requirement for this maneuver are shown in Figures 6-47 and 6-48.

6.4.5.2 Systems Concepts - The system design concept for glide range extension (M2-F2 vehicles) is assumed to be a group of solid propellant motors. Alternate design concepts are not evaluated. The potential weight or cost savings in such a low total impulse system are considered too small to significantly improve vehicle weight or cost. Similarly, solid motors are used for vertical velocity attenuation of category C ballistic vehicles.

Figure 6-47

BALLISTIC LANDING ASSIST THRUST LEVEL REQUIREMENTS

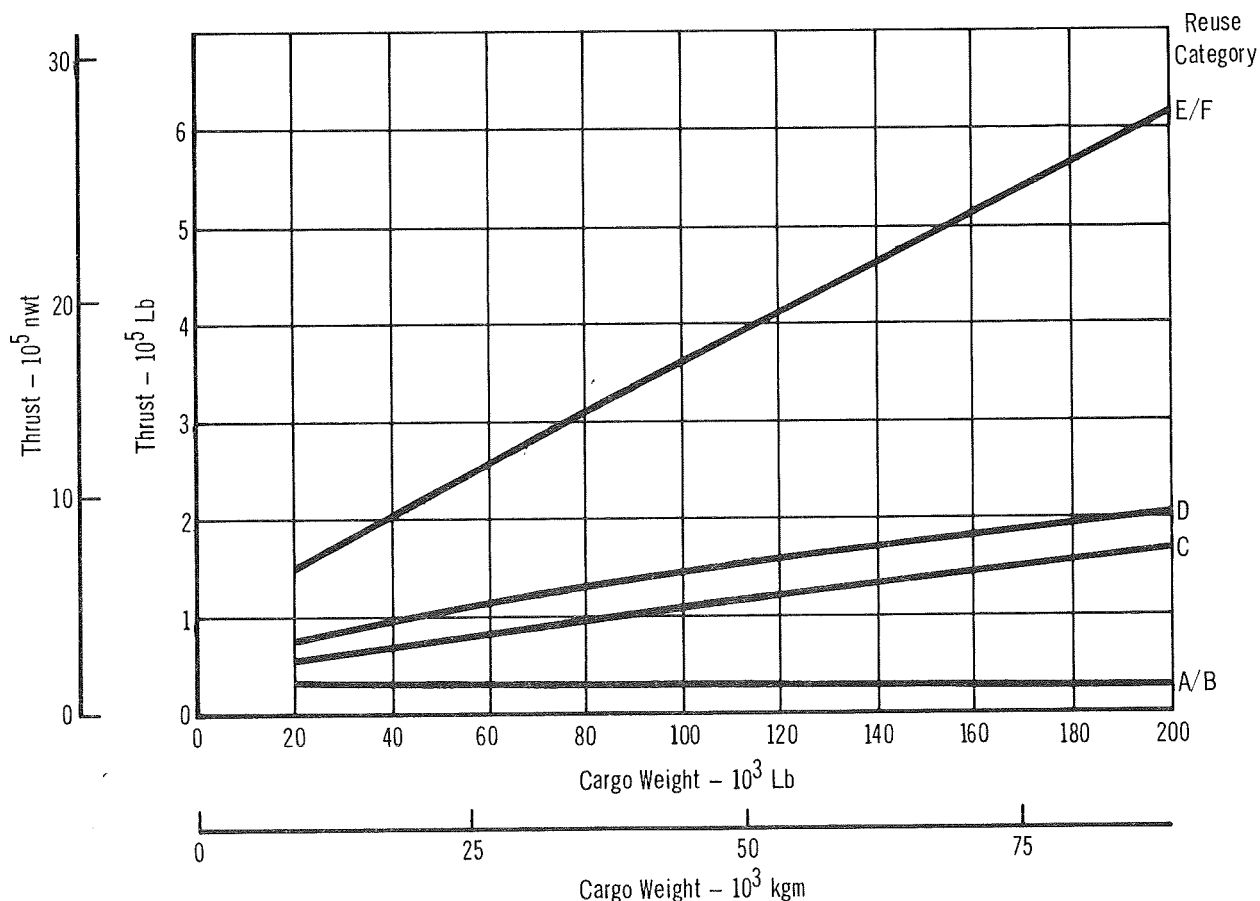
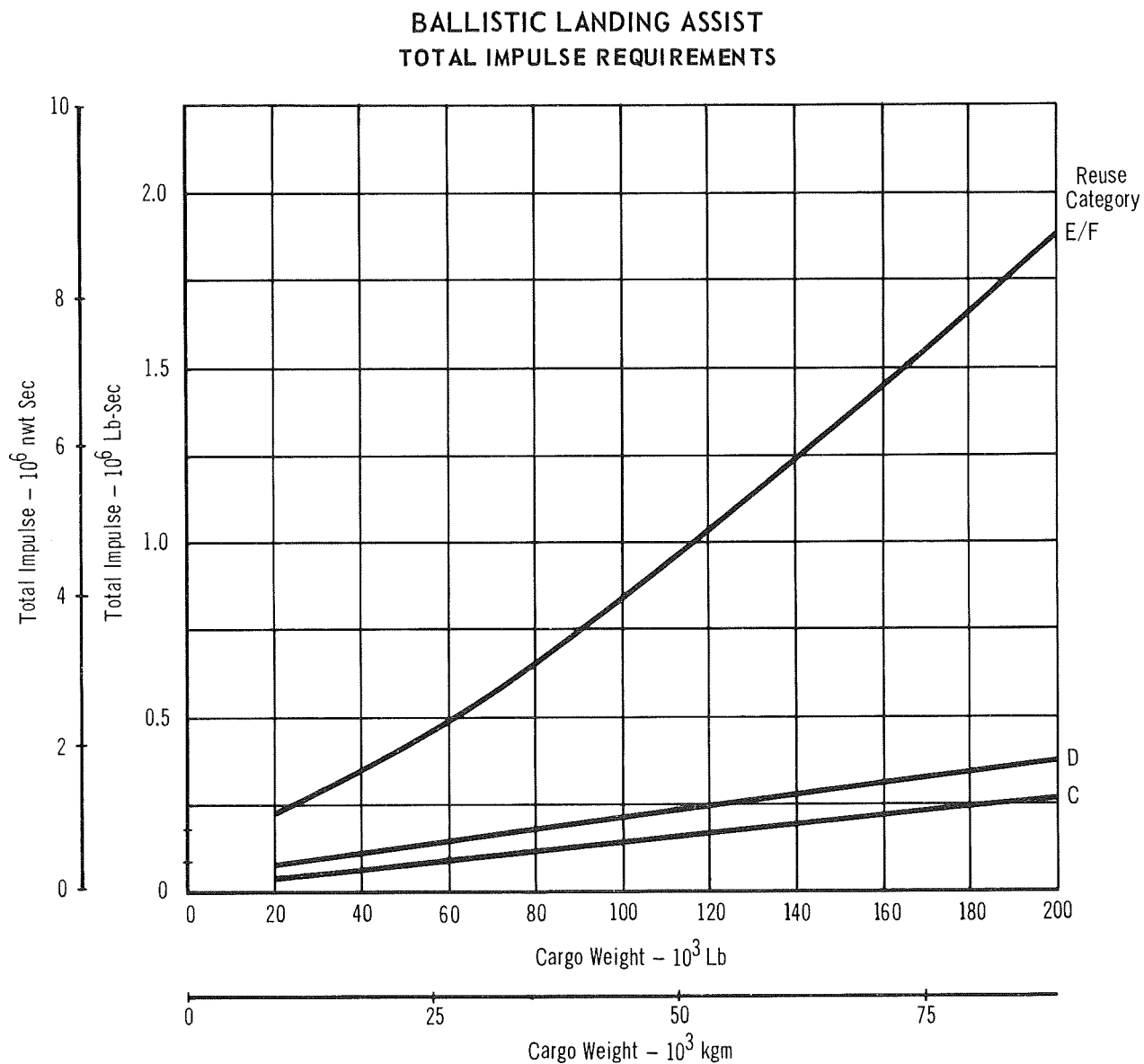


Figure 6-48



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However, for integral boost versions of the ballistic vehicle (category D, E, and F), use of the upper stage boost engines was investigated for landing assist as an alternate to the solid motor concept. These engines provide sufficient thrust for the landing maneuver. As a result, prohibitive engine weight penalty, normally associated with a high thrust liquid propellant system, is eliminated. Additional tankage is needed, however, as the quantity of propellant required for landing is too small to allow use of the large boost system tanks.

A major difficulty involved in using the boost engines results from their relatively slow response characteristics. As it is desirable to accomplish this deceleration maneuver in a short time period to minimize gravity losses, burn times are usually less than one second duration. Typically, the boost engines require about three seconds to attain full thrust and another two seconds for shutdown. Although an acceptable sequence of operation could be established using these response times, the longer time duration would increase total impulse requirements. Additional development testing, both on an engine component and total spacecraft level, is also required because of the increased importance of transient performance and the complex operational mode. Further, landing development tests, at the module level, would require another ship set of boost engines. The additional development and test hardware cost is estimated to exceed that of the total solid motor development program. Hence, while use of the boost engines appears technically possible, little if any weight or cost advantage is anticipated. For this reason, the less complicated method of solid propellant motors is used for landing assist of the integral boost ballistic vehicles.

6.4.5.3 Baseline System Description - The baseline landing assist system configurations are shown in Figure 6-49.

Glide range extension of the M2-F2 is accomplished by multiple, solid motors located inside the crew module. The motors are ripple fired to provide a degree of flexibility in the duration of thrust and, therefore, in the amount of glide range extension that is produced.

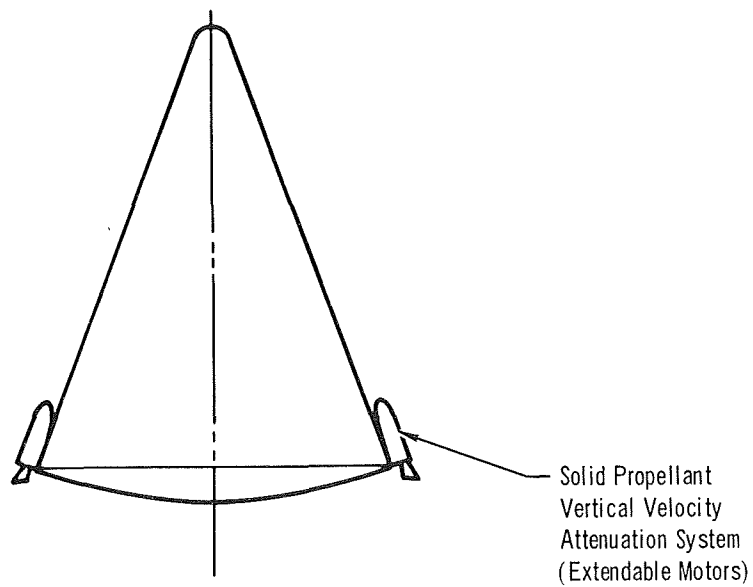
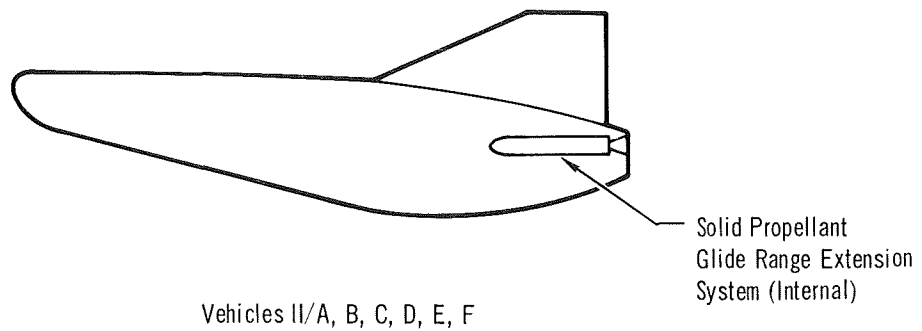
Vertical velocity attenuation of the integral ballistic vehicles is performed by salvo fire of multiple solid propellant motors located around the periphery of the heat shield. As noted in Figure 6-49, extendible motors are used to eliminate the need for blowing out ports in the heat shield.

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Figure 6-49

BASELINE CONFIGURATIONS – LANDING ASSIST



6.4.6 Propulsion System Sizing Model - The spacecraft sizing model includes a subprogram which can separately size each of the propulsion system. General design characteristics and performance requirements for each propulsion system, are specified as input data. The system characteristics include such items as type of propellant, specific impulse, and number of engines. The general performance requirements are those summarized in Table 6-11, i.e., thrust to weight ratio, velocity increment (ΔV), angular acceleration, etc. These input data and the overall vehicle weight that is calculated in a separate mass properties subprogram are then used to establish the specific size of each propulsion system. Major size characteristics that are determined include propellant weight, system inert weight, thrust level per engine, and propellant tank volumes. An iterative calculation procedure is used because of the strong interaction between overall spacecraft weight and the required propulsion system size. The output data are used as inputs to the cost model as well as for a system description printout. A more detailed discussion of the propulsion subprogram is provided in Volume III.

6.4.7 Propulsion Reliability - The nominal reliability requirements, for each on board propulsion system, are shown in Table 6-3. To ensure that such requirements are realistic, an estimate of the attainable reliability is made for each alternate propulsion systems concept. These estimates are based on component reliability data received from propulsion hardware contractors and on component failure rate information that has been compiled by McDonnell. These data are shown in Tables 6-20 and 6-21. The resulting reliability estimates for each propulsion system are summarized in Table 6-22. The reliability range, shown for each system, reflects the variation due to alternate design concepts. Comparing these estimates with the requirement of Table 6-3 indicates that the attainable system reliability is always greater than that required. Therefore, the nominal propulsion system reliability requirement are considered to be realistic and can be met by the concepts to be evaluated further in Task 6.

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Table 6-20
Component Reliability Estimates

Component	Reliability
Attitude control engine	.9982/engine
Main orbital maneuver engine	.999 to .9995 (.9989/engine start)
Upper Stage Boost Engine	.9975
Propellant tanks	.9999 nominal

Table 6-21
Component Failure Rates

Component	(Failures x 10 ⁶ or Ps)	Time or Cycle
Pressurant Supply Leakage (N ₂)	2.02	per hour
Fuel Supply Leakage	2.51	per hour
Oxidizer Supply Leakage	5.02	per hour
Pressure Regulator	2.14	per hour
Solenoid Valve	.99989	per cycle

Table 6-22
Propulsion System Reliability Estimates

System	Estimated Reliability Range
Orbit Attitude Control and Docking	.998 to .9999
Main Orbital Maneuver	.9985 to .9999
Deorbit (solid)	.99989 to .999996
Launch Escape	.9988 to .99999
Upper Stage Boost	.975 to .9989
Entry Attitude Control	.996 to .9995
Glide Range Extension	.999 to .99995
Vertical Velocity Attenuation	.9989 to .99999

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6.5 Environmental Control and Life Support (ECLS) Subsystem - The ECLS provides a controlled temperature environment for the crew and equipment and supplies necessary life support expendables. Certain other functions, such as CO₂ partial pressure control, are also performed to provide an acceptable atmospheric environment. The ECLS requirements, some concepts for accomplishing the requirements, and baseline system definitions are discussed here.

6.5.1 ECLS Requirements - The general ECLS requirements for various mission phases are listed in Table 6-23 below. The requirements are nearly the same for all vehicles, except for the cabin repressurization gas and the additional cooling required by the aerodynamic control-hydraulic system on lifting vehicles. These requirements are a function of vehicle size.

Table 6-23
ECLS Mission Requirements

Mission Phase	ECLS Requirements
Prelaunch	Same as launch except cooling provided by AGE
Launch (400 sec)	Crew and equipment cooling Crew - 635 Btu/hr-man (61K to 366K Btu) Equip. - 4000 to 10,020 Btu/hr. (192K to 480K Btu)
Orbit Maneuver (2 - 24 hr)	Expendables Supply O ₂ - 2 lb/man-day (8 to 48 lb) H ₂ O - 6 lb/man-day (24 to 144 lb)
Deorbit Operations (2 - 24 hr)	Atmosphere Control CO ₂ - Remove 2.25 lb/man-day (9 to 54 lb)
Entry (60 min)	Cabin Repressurization: one/flight Leakage: 5 lb O ₂ /day (10 lb)
Entry (30 min)	Crew Cooling: 0.088 lb O ₂ /man-min (5 to 32 lb)
Landing (20 min)	Equipment Cooling: 4000 to 10,020 Btu/hr (1330 to 3340 Btu) Aero Control System Cooling: (Lifting vehicle only) 71,000 to 4,920,000 Btu/hr 12,000 to 253,000 Btu
Post Landing	None

6.5.2 Environmental Control/Life Support (ECLS) Subsystem Concepts - The ECLS subsystems developed for Mercury, Gemini, and Apollo manned flight are very similar in design and concept. Extensive and detailed optimization were performed in developing the ECLS subsystems for each of the above vehicles, and they have general characteristics which are similar. (See Table 6-24).

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Table 6-24
Mercury, Gemini, and Apollo ECLS Subsystem
Characteristics

Spacecraft	Atmosphere	CO ₂ Control	Heat Rejection	Mission Capability
Mercury	O ₂ @ 5 psia	LiOH absorber	Water Boiler	2 man-day
Gemini	O ₂ @ 5 psia	LiOH absorber	Water Boiler/ Radiator	28 man-day
Apollo CSM	O ₂ @ 5 psia	LiOH absorber	Water Boiler/ Radiator	42 man-day
Apollo LM	O ₂ @ 5 psia	LiOH absorber	Water Sublimator	4 man-day

Several possible subsystem variations are discussed below. However, of the alternate subsystem approaches discussed, only the radiator/boiler vs all boiler system for heat rejection and the cryogenic vs high pressure gas storage trades are expected to have a significant effect upon ECLS subsystem cost.

6.5.2.1 Atmosphere Supply and Gas Processing - Pure oxygen atmosphere vs a two-gas atmosphere, method of storage, and total cabin atmosphere pressure are discussed.

Pure oxygen atmosphere vs a two-gas atmosphere - A two-gas atmosphere tends to reduce the fire hazard of pure oxygen atmospheres. This advantage, however, is most pronounced during ground operation at sea level pressure where ground servicing equipment can provide a mixed gas atmosphere. Further, some flammable items provided for convenience or for comfort in long duration flights can be eliminated in a short duration flight logistics vehicle. There are no physiological reasons requiring diluent gas in a logistics vehicle. Crew habitation periods are short; pure oxygen has been used in orbital flights

of up to 14 days with no adverse affects. Therefore, a pure oxygen cabin atmosphere was selected for this study.

Method of gas storage - Here a meaningful trade can be made between cryogenic and high pressure tanks. Cryogenic tanks cost more, weigh less, and take less volume than high pressure tanks but have a storage life limited by the quality of their insulation. High pressure tanks can store gas for indefinite periods, they are also less complicated and more reliable. Thus, both methods will be studied.

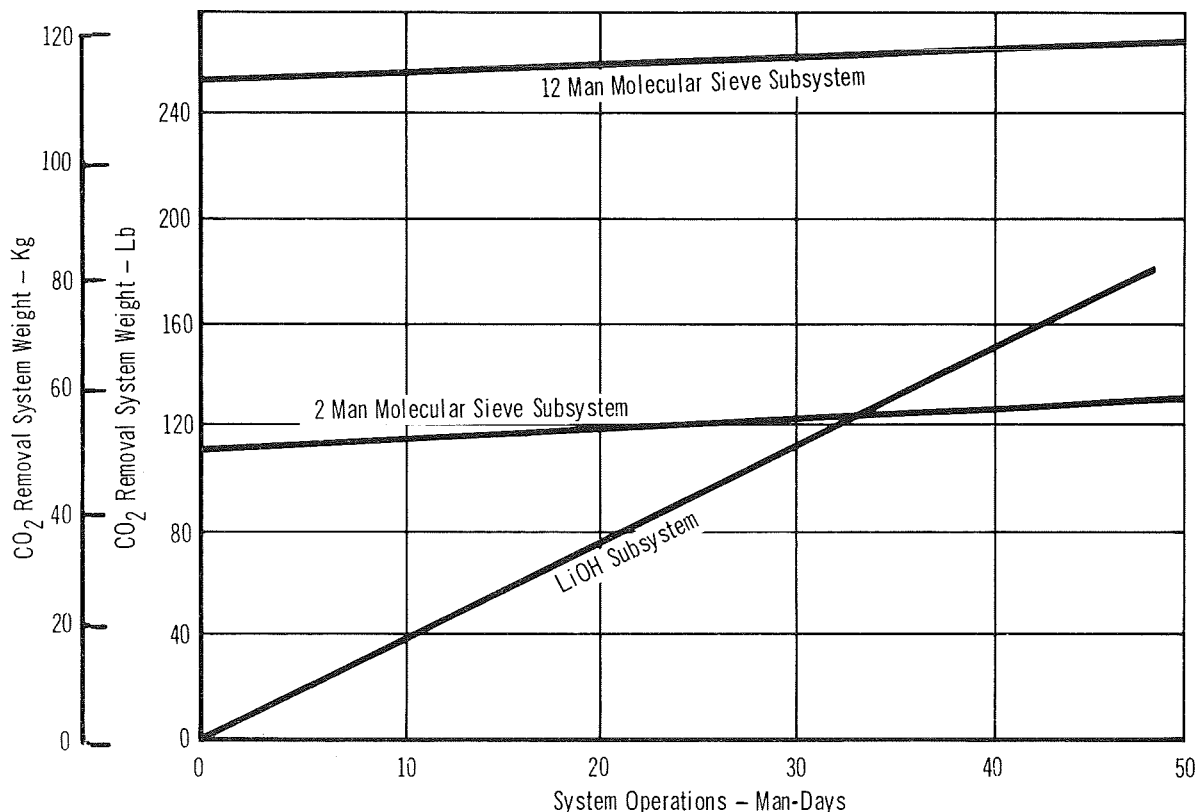
Total cabin atmosphere pressure - The cabin pressure could vary between 5 and 14.7 psia, affecting the ECLS design only slightly, (for example, fans would have to be optimized for the particular pressure chosen). This would not significantly affect either ECLS subsystem weight or cost. Thus a cabin pressure of 5 psi was selected for ECLS subsystem design in all vehicles.

6.5.2.2 CO₂ Control - A considerable mission capability advantage for long missions would occur if a molecular sieve were used rather than lithium hydroxide for CO₂ removal. For example, Figure 6-50 shows that the molecular sieve is far superior for even a two-man system once the mission duration exceeds 16 days. However, for the logistic mission duration and range of crew sizes under investigation, the LiOH system weighs less and is also less complex so that the first cost of a LiOH system should be smaller. Re-use of either concept would require comparable replacement and checkout. The LiOH must be replaced and the molecular sieve material would probably be replaced to avoid extensive tests of its performance after each flight. Since there is no apparent advantage for molecular sieves in logistics flights, LiOH was selected for CO₂ absorption.

6.5.2.3 Spacecraft Cooling - The Reference 4-1 study indicates a logistics vehicle weight saving of more than 300 lb. by using a space radiator with a water boiler, rather than boilers only. Although usually lighter, the radiator-boiler method of heat rejection can result in greater cost. Extensive analysis and testing are required for the radiator to ensure survival through launch heating (temperatures of about 300 deg F) and orbital storage temperatures as low as -160 deg F. However, if the heat dissipation or mission duration requirements increase, the weight advantage of the radiator-boiler system increases proportionately. Therefore, both techniques for heat rejection will be studied. The radiator approach is not feasible on integral vehicles because atmospheric entry

CO₂ REMOVAL SYSTEMS

Figure 6-50



heating would overheat the coolant and damage the radiator unless a thermal protective cover is provided.

6.5.3 Baseline ECLS Subsystems - A summary of the subsystem concepts is given in Table 6-25 which also shows to which vehicle configuration each is applicable. As indicated, there are two baseline concepts, one for the modular vehicles and one for the integral vehicles. The modular vehicle baseline is described first and the differences which are required for integral vehicles follow. These baseline subsystems as described represent best first estimates of minimum cost configurations. The subsystem make-up will be varied systematically in Task 6 to determine the subsystem corresponding with the minimum total program cost. Table 6-26 shows the reliability estimates for the major components of the ECLS subsystem based on a 48-hour mission.

6.5.3.1 Baseline 1: Modular Vehicles - The Environmental Control and Life Support subsystem (ECLS) consists of four subsystems which provide breathing oxygen, atmosphere control, metabolic and equipment waste heat removal, and water supply and waste disposal. These are: the Gas Supply and Control Subsystem, the Gas Processing Subsystem, the Heat Transport Circuit Subsystem,

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Table 6-25

ECLS SUBSYSTEM CONCEPT SUMMARY

Concept No.	Subsystem Definition	Vehicle Application
EC1*	5 psi O ₂ atmosphere - gaseous storage; Water boiler and LiOH in EV; Primary O ₂ , H ₂ O and radiator in MM	IA, IB, IIA, IIB
EC2*	5 psi O ₂ atmosphere - gaseous storage; Water boiler and LiOH	IC, ID, IE, IF IIC, IID, IIE, IIF
EC3	5 psi O ₂ atmosphere - cryo storage; Water boiler and LiOH	IC, ID, IE, IF IIC, IID, IIE, IIF
EC4	5 psi O ₂ atmosphere - gaseous storage; Water boiler and LiOH in EV; Primary O ₂ and H ₂ O in MM	IA, IB, IIA, IIB
EC5	5 psi O ₂ atmosphere - cryo storage; Water boiler and LiOH in EV; Primary O ₂ and H ₂ O in MM	IA, IB, IIA, IIB
EC6	5 psi O ₂ atmosphere - cryo storage; Water boiler, LiOH and primary O ₂ in EV; H ₂ O in MM	IB, IIB
EC7	5 psi O ₂ atmosphere - cryo storage; Water boiler and LiOH in EV; Primary O ₂ , H ₂ O and radiator in MM	IA, IB, IIA, IIB
EC8	5 psi O ₂ atmosphere - cryo storage; Water boiler, LiOH and primary O ₂ in EV; Radiator and H ₂ O in MM	IB, IIB

* Baseline subsystem

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Table 6-26

ECLS COMPONENT RELIABILITY ESTIMATES

<u>Component</u>	<u>Estimated Reliability</u>
Supercritical O ₂ tank	.999899
O ₂ Heat exchanger	> .99999
Gaseous leakage	.999424
Pressure regulation	.999897
Suit demand pressure regulator	.999376
Compressor and power supply	.993838
Helium tank and leakage	.998844
Solenoid Valves (shutoff)	P _S = .99999/cycle

and the Water and Waste Management Subsystem. These subsystems are shown schematically in Figures 6-51 through 6-52.

Gas Supply and Control Subsystem - Oxygen is supplied for pressurization, leakage makeup, and metabolic consumption. The subsystem is shown schematically in Figure 6-51. The primary oxygen supply is stored in spherical high pressure gas tanks. Primary oxygen regulators reduce the primary oxygen supply pressure to 150 psig before delivery to the demand, cabin pressure, and other low pressure regulators. The secondary oxygen supply is also stored in spherical high pressure gas tanks. Secondary oxygen regulators reduce the pressure to 75 psia for supply to either the secondary rate valve or to the Gas Processing Subsystem. The higher primary oxygen supply pressure (150 psi) prevents inadvertent usage of the secondary oxygen by preventing flow through the secondary oxygen regulator which only opens when the primary oxygen pressure drops below 75 psi. Both sets of tanks are sized based on an operating pressure of 5000 psig. This pressure level was chosen to minimize volume, yet to remain near minimum weight point.

Gas Processing Subsystem - The gas processing subsystem consists of two separate suit ventilation circuits, one for the pilots and a second for the other

Figure 6-51

BASELINE 1 AND 2 GAS SUPPLY AND CONTROL SUBSYSTEM SCHEMATIC

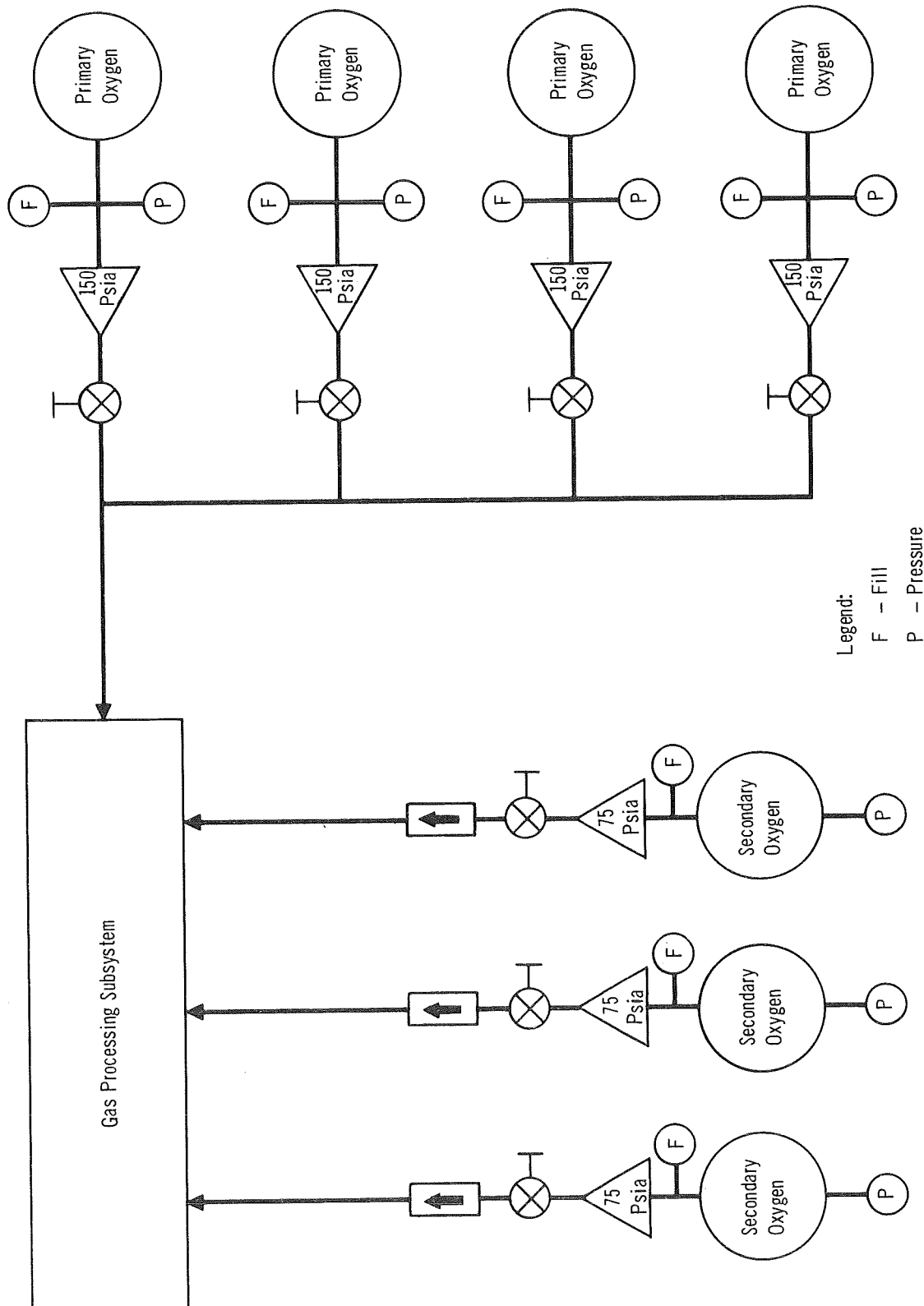
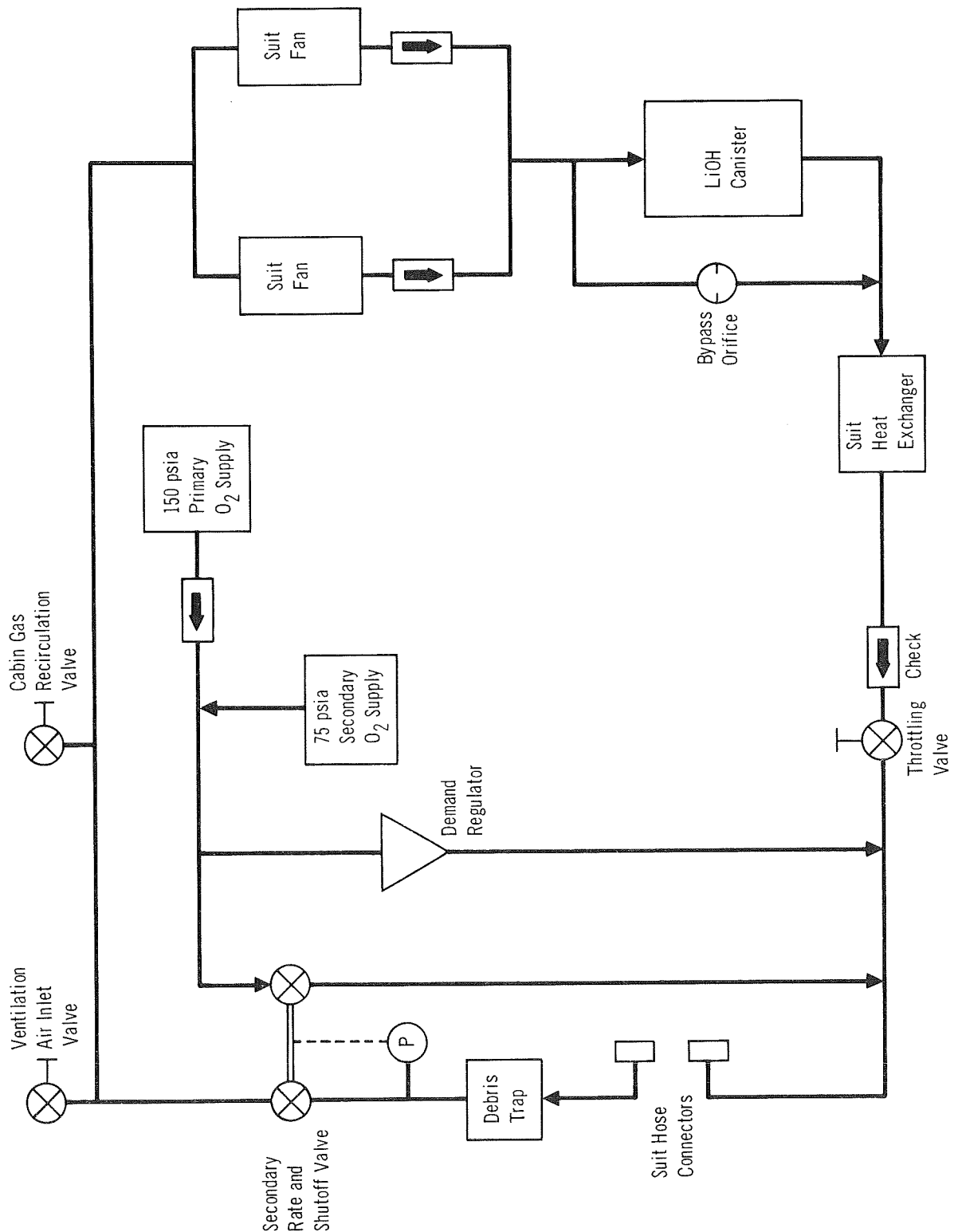


Figure 6-52

BASELINE 1 AND 2 GAS PROCESSING SUBSYSTEM



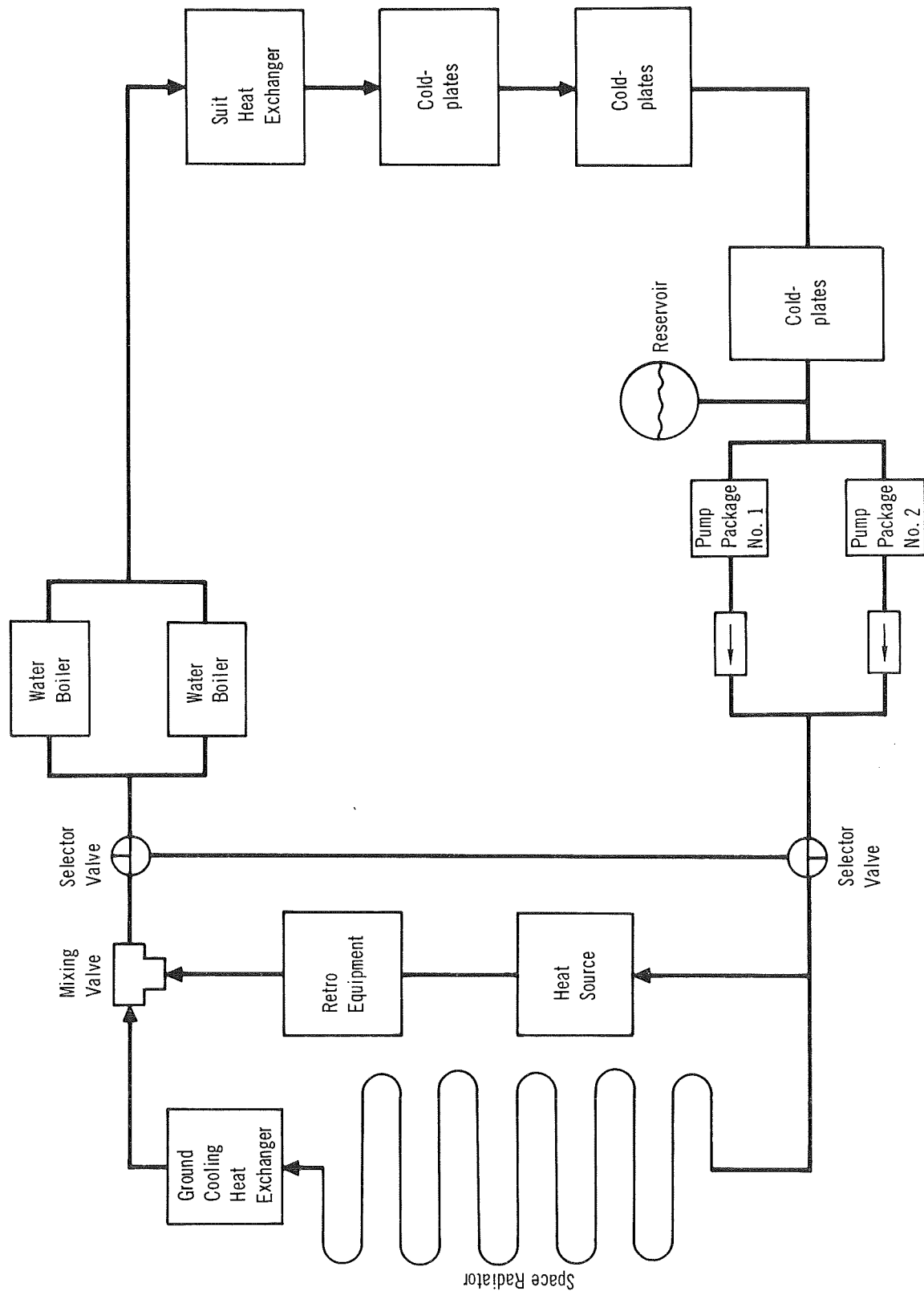
crewmembers. Figure 6-52 shows one ventilation circuit schematically; for clarity the suit ducting to only one man is shown since the ducting for the other men is identical. In operation, redundant fans provide ventilation flow. The suit gases flow from the fans through an LiOH canister for CO₂ absorption and a parallel canister bypass duct. This arrangement allows a lower flowrate through the CO₂ absorber with retention of high CO₂ absorption effectiveness. Downstream of the CO₂ absorber, the suit gases flow through heat exchangers which cool the suit ventilation oxygen and condense and remove moisture. The oxygen then flows through check and shut-off valves to the pressure suit. Leaving the suit, the oxygen flows through debris traps to the secondary rate shutoff valve. This valve, in conjunction with the check valve upstream of the suit allows isolation of the man from the remainder of the circuit in an emergency. (In addition to closing the suit outlet, the valve opens the secondary oxygen supply valve to permit a flow of oxygen to purge CO₂ from the suit). After leaving the secondary rate shutoff valve the ventilation gases return to the compressor inlet.

Heat Transport Circuit Subsystem - Operation of the subsystem, shown in Figure 6-53 is as follows: a heat transfer fluid is circulated through two circuits by redundant pumps in each loop. The cold plates and heat exchangers each have dual, redundant passages for the separate loops. Only one circuit is shown, however, in Figure 6-53. The flow leaving the pumps flows through a selector valve to the radiator and through a parallel branch in the mission module. The warm fluid flowing through the mission module bypass is mixed with the cold fluid leaving the radiator to provide temperature control. The radiator bypass branch provides waste heat to keep various mission module tanks at acceptable temperatures. The controlled temperature fluid leaving the mixing valve then flows through the suit heat exchangers, various equipment coldplates, and then flows to the pump inlet. Just prior to discarding the mission module, flow is shunted from the pump outlet to the water boiler inlet by the selector valves. This will allow active cooling down to an altitude of about 100,000 ft. Heat sink temperature control will then suffice for all but the aerodynamic control-hydraulic system until landing. Cooling for the hydraulic system is discussed separately below.

Aerodynamic Control Hydraulic System Cooling - Lifting entry vehicles will have high power hydraulic systems (Section 6.7) to move aerodynamic control surfaces. Hydraulic system cooling is performed by redundant water boilers

Figure 6-53

BASELINE 1 HEAT TRANSPORT CIRCUIT SUBSYSTEMS



installed in the hydraulic system upstream of the pumps. The hydraulic fluid flows through the boiler heat exchanger where it is cooled by the evaporation of water. Water is supplied to the boiler from water tanks separate from the drinking water supply. The quantity of water required varies with vehicle size.

Water and Waste Management Subsystem - Water for drinking and for system cooling is stored in a pressurized bladder-type tank shown in Figure 6-54. Water is supplied on demand to the crew at a regulated pressure of 5 psi above cabin pressure. Boiler water is supplied only when the solenoid supply valve is manually opened. This valve is intended to prevent leakage from the boiler during orbital storage. The boilers provide waste heat rejection after launch while the radiator is cooling down and during re-entry after the mission module and radiator have been discarded. During the intervening flight, it is used for peak load sharing. Urine will be collected in urine receivers and dumped overboard as in the Gemini and Apollo systems.

6.5.3.2 Baseline 2: Integral Vehicles - The integral vehicle baseline ECLS concept is identical with the modular except that the integral vehicle does not use a radiator for heat dissipation and requires additional water for cooling. A revised Heat Transport circuit schematic is given in Figure 6-55 which indicates the circuit changes. The additional water needed will also require either a larger tank or multiple tanks.

6.5.4 Environmental Control System Sizing Model - The sizing routine computes an atmospheric gas supply system based upon the number of men and the volume required for the men, a heat transport loop based upon peak and total heat loads, and a water management system based upon crew and equipment requirements. The mass properties routine computes volume requirements for the system and accounts for its weight and center of gravity in the balance portion of the routine.

6.6 AVIONICS SUBSYSTEM - The avionics subsystem consists of guidance and control, telecommunication, and on-board checkout. Each of these is discussed separately.

6.6.1 Subsystem Requirements - Avionics subsystem requirements are summarized for each mission phase in Table 6-27. Basic requirements are independent from the type of entry vehicle and are only slightly affected by the modular concept.

Figure 6-54

BASELINE 1 AND 2 WATER AND WASTE MANAGEMENT SUBSYSTEM

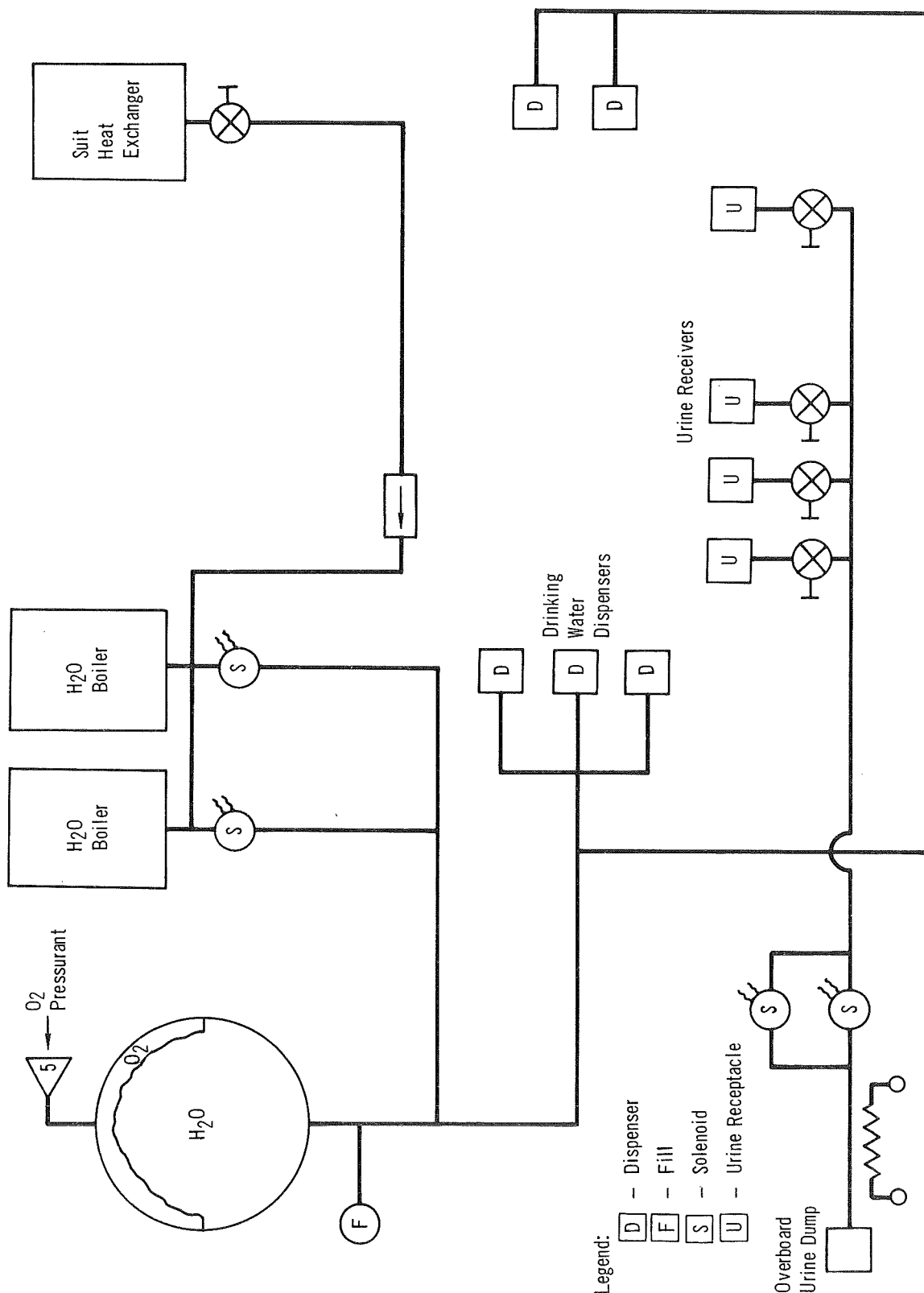
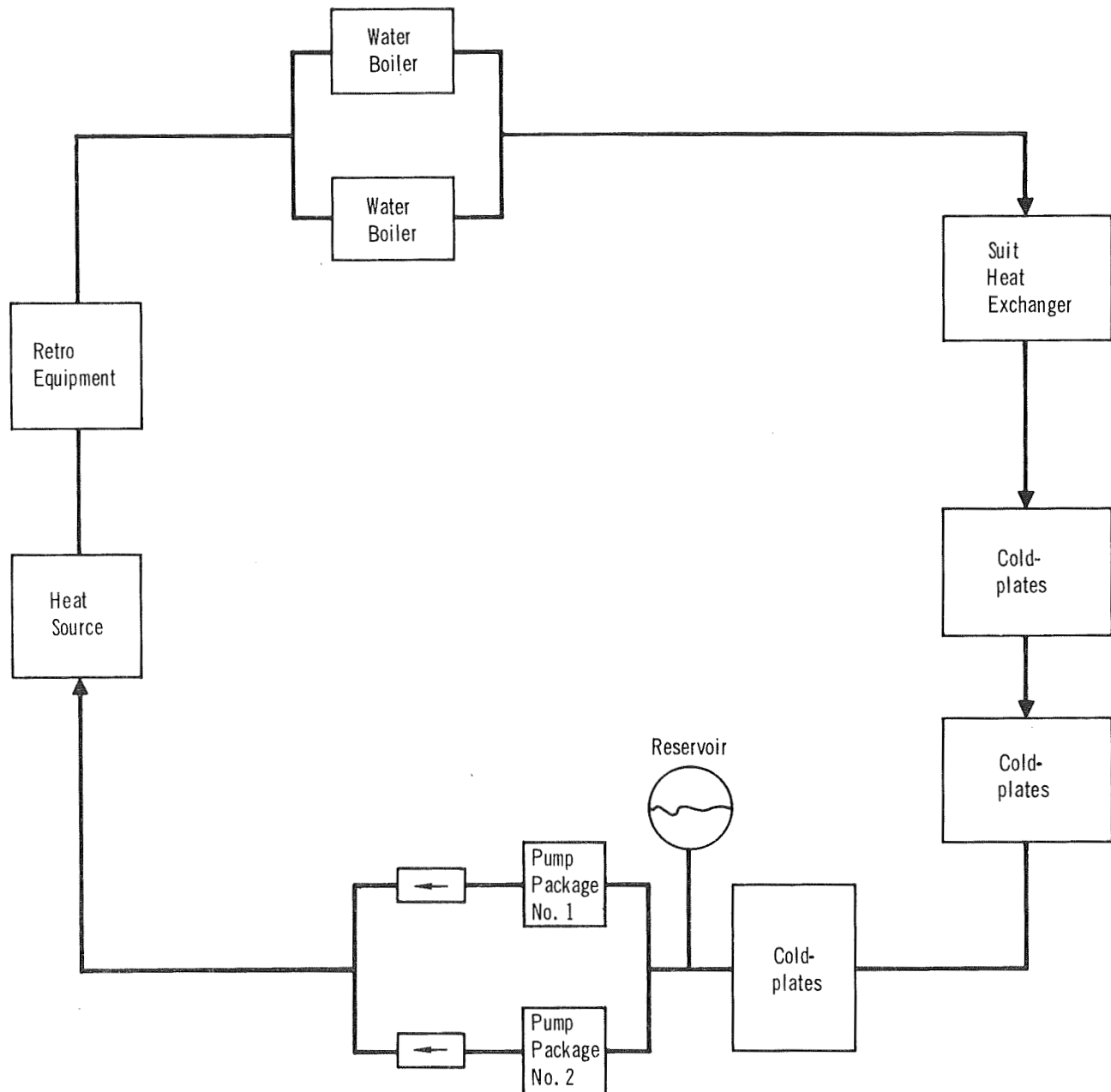


Figure 6-55

BASELINE 2 HEAT TRANSPORT CIRCUIT SUBSYSTEM



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TABLE 6-27

AVIONICS SYSTEM REQUIREMENTS

MISSION PHASE	FUNCTIONAL REQUIREMENTS	PERFORMANCE REQUIREMENTS
Prelaunch	Voice and Telemetry, Onboard	
Launch	Voice and Telemetry Back-up Guidance ± 20 ft/sec insertion accuracy	Accuracy requirement increases to ± 10 ft/sec for vehicles with integral upper stage engine
Launch Abort	Voice and Telemetry Abort Stabilization	
Orbit Maneuver	Voice and Telemetry Determine Target Location Monitor Rendezvous Maneuvers	Range - $\pm 0.1\%$ or ± 75 ft Direction - ± 8.5 mrad Thrust - $\pm 0.3\%$ Attitude - ± 0.3 deg
Orbital Storage	Onboard Checkout	
De-orbit Operations	Voice and Telemetry Monitor Retrothrust	Thrust - $\pm 0.3\%$ Attitude - ± 0.3 deg
Entry	Voice and Telemetry Re-entry Navigation	± 4 na mi @ 50K ft Accuracy requirement relaxes to ± 12 na mi for water recovery
	Rate Stabilization	
Post Landing	Recovery Beacon and Long Range Voice	

The reliability requirements for a land landing are more stringent than for a water recovery, requiring the addition of a back-up guidance system (inertial reference and computer) to the G&C system. When an upper stage engine is included as an integral part of the entry vehicle, the G&C system must further expand over that required for a land landing. A third inertial reference must be added for voting redundancy and the flight control system must be expanded.

6.6.2 Guidance and Control (G&C) - Two concepts, Gemini and advanced, were considered for the Guidance and Control system. The four baseline definitions considered differed only in their degree of redundancy.

6.6.2.1 Concepts - The Gemini system, the Apollo system, and an advanced system were initially considered as three separate concepts for the G&C system. However, the only significant difference between the Gemini and Apollo systems is the method of in-orbit alignment. The Apollo inertial alignment subsystem is an optical subsystem. Although it could be used, it is considered too bulky, expensive, and complex for the needs of a logistic vehicle. In contrast, a horizon sensor was used with the Gemini system for automatic local vertical alignment of the inertial platform. Although physically different, the Gemini and Apollo inertial, computer, and flight control subsystems are virtually the same in concept. Circuit and construction techniques are virtually the same in the two systems. Physical differences are the result of different functional requirements and mode definitions. Consequently, only two G&C concepts are considered: Gemini and advanced. The term "Gemini concept" is used for convenience only. It should not be construed to mean exclusive use of Gemini hardware. Mission requirements for the logistics vehicle dictate the basic definition of the G&C. Circuit and construction techniques will cause the advanced system to be smaller, with a higher reliability and with an anticipated longer life and reuse capability.

In addition, it is assumed that strap-down platforms will replace gimballed platforms in the advanced concept. The strap-down platform offers many advantages in comparison with a gimballed platform, but its use in a primary inertial reference capacity should be approached with caution. It is known that strap-down platforms exhibit performance degradation due to effects of vehicle body motion and vibration. Since accurate descriptions of these

environments are lacking, it is difficult to predict the extent of performance degradation. Also, its performance as a navigation aid has not yet been demonstrated during actual spacecraft flights.

Table 6-28 summarizes the concepts for the Guidance and Control subsystem and indicates the vehicle applications.

6.6.2.2 G&C Baseline Definitions - There are four G&C baseline definitions (identified as GC-1, -2, -3 and -4); they differ in the degree of redundancy which is employed in the subsystems. Each baseline system consists of an inertial reference subsystem, a computer subsystem, and a flight control subsystem. The rendezvous radar has been assigned to the telecommunication system to permit a more equitable comparison of the telecommunication concepts. Redundancy requirements of the four systems are influenced by mission requirements which are related, to a certain degree, to vehicle configuration. Figure 6-56 is a block diagram which shows the overall differences between systems. Physical characteristics of the four systems are presented in Table 6-29.

System GC-1 is the smallest and is intended for a water recoverable vehicle (vehicle configuration I-A). It contains an Inertial Measurement Unit (IMU), a horizon sensor, a computer, a flight control system, a time reference, an Auxiliary Tape Memory (ATM), and displays and controls. The IMU consists of a four-gimballed platform, its associated electronics, and a platform mount. A gimballed platform is selected in preference to a strap-down platform because of the latter's performance uncertainty. A four-gimballed platform is selected in preference to a three-gimballed unit to eliminate attitude control restrictions. The Gemini program demonstrated that a four-gimballed platform is sufficiently accurate for the guidance system's accuracy requirements for insertion and re-entry.

The computer is a general-purpose digital computer with flexible memory similar to the Gemini because it is flight-proven, and has sufficient versatility, size, and capability to meet the mission requirements of a logistics vehicle.

The flight control system is of the Gemini type, consisting of an analog computer, two sets of rate gyros (for reliability), and control switching electronics.

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TABLE 6-28

GUIDANCE AND CONTROL SUBSYSTEM CONCEPT SUMMARY

<u>CONCEPT NO.</u>	<u>CONCEPT DEFINITION</u>	<u>VEHICLE APPLICATION</u>
GC-1*	Single IMU, Computer, FCS	IA (Water Landing)
GC-2*	Gimballed and Strap-down IMU Dual Computers, Single FCS	IC
GC-3*	Gimballed and Strap-down IMU Dual Computers, Redundant FCS	IA (Land Landing), IB, IIA, IIB and IIC
GC-4* 	2 Gimballed and 1 Strap-down IMU, Dual Computers, Redundant FCS	ID, IE, IF, IID, IIE, IIF
GC-5 	Advanced Concept of GC-1	IA (Water Landing)
GC-6 	Advanced Concept of GC-2	IC
GC-7 	Advanced Concept of GC-3	IA (Land Landing), IB, IIA, IIB and IIC
GC-8  	Advanced Concept of GC-4	ID, IE, IF, IID, IIE, IIF

* Baseline Concept

- 1 These concepts are also applicable for vehicles IA, IB, IC, IIA, IIB, and IIC for integrated booster-spacecraft guidance systems.
- 2 Advanced concepts employ advanced circuit techniques and replace gimballed platforms with strap-down platforms.

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TABLE 6-29
Physical Characteristics of G & C Systems

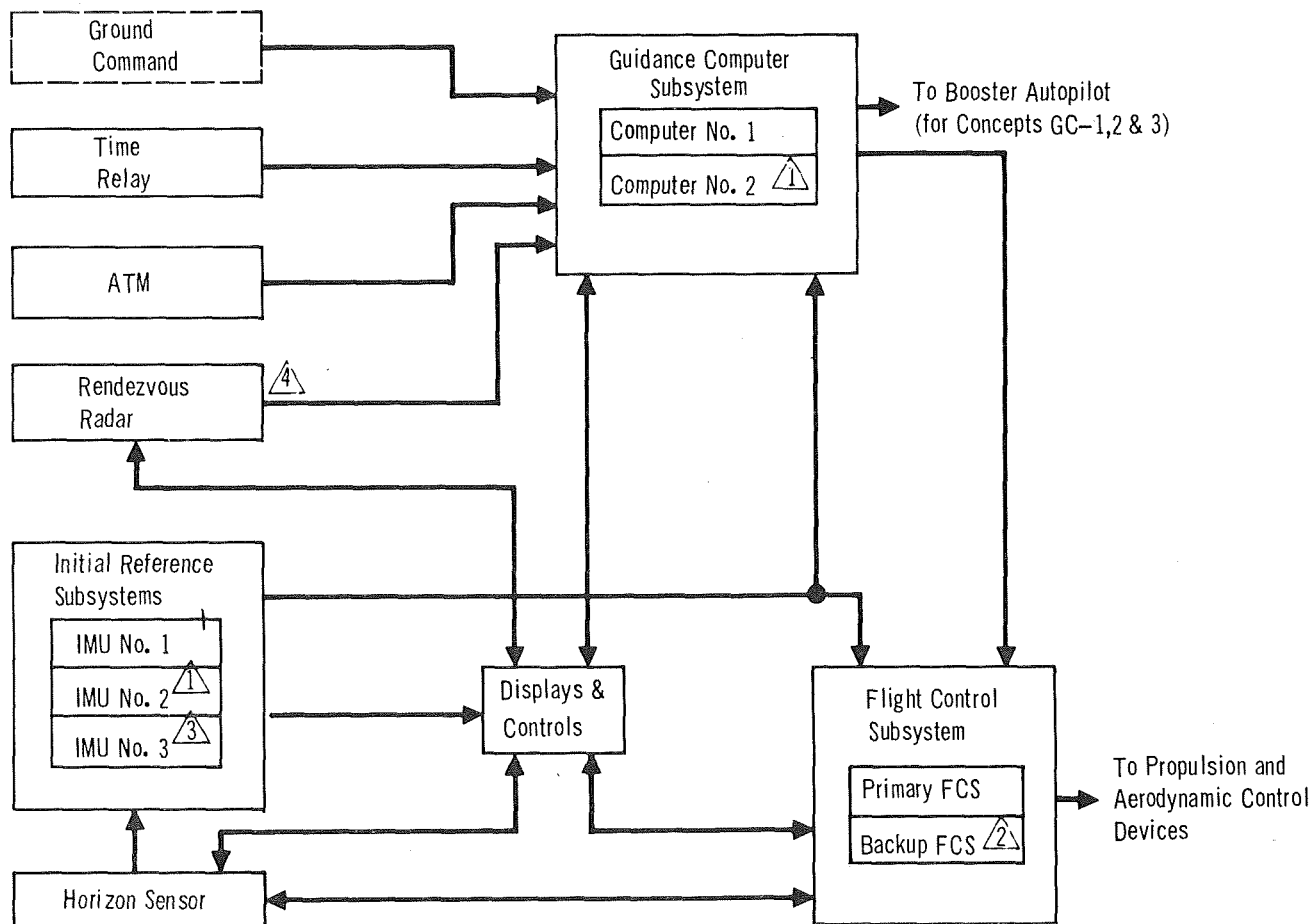
EQUIPMENT	GC - 1			GC - 2			GC - 3			GC - 4		
	Vol. ft ²	Wt. lbs	Pk Pwr watts	Vol. ft ³	Wt. lbs	Pk Pwr watts	Vol. ft ³	Wt. lbs	Pk Pwr watts	Vol. ft ³	Wt. lbs	Pk Pwr watts
1. Inertial Measurement Unit	1.9	135	280	1.9	135	280	1.9	135	280	3.8	270	560
2. Guidance Computer	1.5	60	90	3.0	120	180	3.0	120	180	3.0	120	180
3. Flight Control System	1.2	40	50	1.2	40	50	3.2	120	100	3.2	120	100
4. Displays and Controls	1.0	50	30	1.0	50	30	1.0	50	30	1.0	50	30
5. Horizon Sensor	.3	25	15	.3	25	15	.3	25	15	.3	25	15
6. Time Reference	.2	10	10	.2	10	10	.2	10	10	.2	10	10
7. Auxiliary Tape Memory	.4	25	30	.4	25	30	.4	25	30	.4	25	30
8. Strap-Down IMU	-	-	-	.9	60	180	.9	60	180	.9	60	180
9. Landing Displays and Controls	-	-	-	-	-	-	.5	20	10	.5	20	10
TOTALS	6.5	345	505	8.9	465	775	11.4	565	835	13.3	700	1115

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Figure 5-56

GUIDANCE AND CONTROL BLOCK DIAGRAM



Notes:

- △1. This equipment added for GC-2 concept and is also common to concepts GC-3 and GC-4.
- △2. This equipment added for GC-3 concept and is also common to concept GC-4.
- △3. This equipment added for concept GC-4.
- △4. The rendezvous radar is considered as a part of the telecommunication system

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The displays and controls consist of two attitude and rate displays, a manual input-readout unit to the computer, and the controls for the G&C operation and selection. For modular concepts that use an aft docking mode, the displays and controls for docking are duplicated in the docking station in the mission module.

The horizon sensor is a thermopile edge tracker which can align the inertial reference to the accuracy required. A four-head system is selected in preference to a three-head system for increased reliability.

The time reference is an electronic timer similar to the one used on Gemini.

The auxiliary tape memory providing an enhanced program storage and reload capability, was successfully used on Gemini. Added primarily for reliability considerations, it also serves to increase the effective size and capability of the digital computer.

Baseline system GC-2 is the same as GC-1 with an added back-up guidance system to increase the probability of meeting the land landing accuracy requirement. The back-up guidance system is needed on all vehicles intended for a land landing, it consists of a computer and an IMU. The computer will be the same as the primary computer, but the IMU will be a strap-down platform. The strapped-down configuration is selected because of its definite advantages in size, weight, mechanical simplicity, and cost. The performance uncertainty of the strapped-down mechanization is not a serious deficiency in a back-up capacity.

Baseline system GC-3 is for vehicles which are steerable during the terminal landing phase. GC-3 is the same as GC-2 with the addition of a fully redundant flight control system for landing safety. Some additional displays and controls will also be required.

Baseline system GC-4 is for all vehicle configurations with an integral upper-stage engine. GC-4 is the same as GC-3 with the addition of a third IMU for complete ascent guidance capability. This IMU will be the same as the primary IMU. That is, it will have a four-gimballed platform. A gimballed platform is preferred for back-up ascent guidance because of the insertion accuracy requirement. Although a three-gimballed unit would be satisfactory, a four-gimballed platform was selected for commonality.

It has been conservatively estimated that electromechanical equipment, such as gyros and displays, can be used, and refurbished as necessary, for four times. For predominately electronic equipment, it is estimated that this equipment can be used 10 times. An acceptance test would be required to determine the degree of refurbishment that would be required.

Table 6-30 shows the reliability estimates for the four baseline G&C concepts.

6.6.2.3 Alternatives - The primary G&C alternative assumes extensive use of advanced circuit techniques (i.e., integrated circuits) and the replacement of gimballed platforms with strap-down platforms. The use of the advanced technique does not affect the basic G&C system definition. Therefore, there is an advanced equivalent to each of the four baseline systems which were discussed in Section 6.6.2.2. These systems, identified as GC-5, -6, -7, and -8, are the advanced counterparts of GC-1, -2, -3, and -4, respectively. Refer to Table 6-31 for a comparison of the physical characteristics of all eight systems.

Advance circuit techniques are rapidly becoming state-of-the-art and will cause an appreciable size and weight savings in units which are predominately electronic. A potential cost savings can also be realized with advanced circuit techniques, but only on relatively large production quantities.

Caution should be used in considering strap-down platforms as a replacement for gimballed platforms. The advantages of a strap-down platform are many and significant and might cause its major drawback (its performance uncertainty) to be overlooked. When it does become a suitable substitute, it will offer a significant savings in size, weight, power, and cost. It should also provide a longer life with a lower cost for maintenance and repair. It should be noted that the development and procurement cost variations for these alternate systems cannot be readily estimated. They are included here to show the expected design variations; the approach used in this cost analysis is explained in Vol. II, Book 3.

A second alternative which can be considered for future cost studies involves integration of the booster guidance and control system with that of the spacecraft. It cannot be evaluated as part of this study however because

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TABLE 6-30

RELIABILITY OF BASELINE G & C CONCEPTS

Equipment	GC-1 Vert./Water Lndg.	GC-2 Vert./Land Lndg.	GC-3 Horiz.-Land	GC-4 Horiz.-Land
1. IMU	.9813	(R) .99932	(R) .99932	(R ₂) .99999
2. Computer	.9965	(R) .99998	(R) .99998	.99998
3. Flt. Control Syst.	.9999	.9980	(R) .999	(R) .9998
4. Displays & Controls	.9999	.9953	.9998	.9998
5. Horizon Sensor	.9998	.9999	.9999	.9999
6. Time Reference	.9907	.9992	.9999	.9999
7. Aux. Tape Memory				
8. Strap Down IMU	N/A	.9832	.9832	.9832
9. Landing Displays and Controls	.9936	.9950	.9999	.9999

(R) Redundant paths

(R₂) Double redundancy

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TABLE 6-31
Comparison of all G & C System Concepts

	Vol Ft ³	Wt. Lbs	Pk.Pwr. Watts	
GC-1	6.5	345	505	I-A (Water recovery)
GC-2	8.9	465	775	I-C
GC-3	11.4	565	835	I-A (Land), I-B, II-A, II-B & II-C
GC-4*	13.3	700	1115	I-D, I-E, I-F, II-D, II-E, & II-F
GC-5	4.2	205	410	I-A (Water recovery)
GC-6	6.4	305	710	I-C
GC-7	8.1	370	780	I-A (Land) I-B, II-A, II-B & II-C
GC-8*	8.8	410	930	I-D, I-E, I-F, II-D, II-E & II-F

* GC-4 and GC-8 are also applicable for vehicular concepts
I and II -A, -B and -C for integrated booster-spacecraft
guidance systems.

launch vehicle costs are being treated parametrically and do not identify costs at the subsystem level.

6.6.3 Telecommunications (Telecom) - Four concepts for the telecommunication system were considered. The baseline system selected is a Unified S-Band with separate rendezvous radar using state-of-the-art circuit techniques.

6.6.3.1 Concepts - The telecommunications system provides the intelligence link between the ground and the spacecraft and between the space station and the spacecraft. It provides for: ground tracking of the spacecraft; two-way voice communications to the ground and to the space station; ground to spacecraft data or command link; spacecraft to ground data link; recovery aids; and range, range rate, and body referenced target angle data.

Four concepts were considered. The first concept involves the use of separate subsystems as was done on Gemini. In the second, all subsystems with the exception of the rendezvous radar are unified into an integrated (or Unified S-Band) system as was done on Apollo. The third concept is natural extension of the second and includes rendezvous ranging as an integral part of the Unified S-Band system. A fourth concept assumes the use of advanced circuit techniques (integrated circuits).

Advantages of the "separate systems" concept are its small size, weight, and simplicity. Its disadvantages include limited flexibility and limited growth capability.

Using Apollo equipment as a basis for comparison, the Unified S-Band with separate radar concept offers flexibility and growth capability, reduces the number of separate systems, and offers a higher reliability. It is also larger, heavier, and has a greater circuit complexity than the "separate systems" concept. These disadvantages are related in part to the increased use of redundancy in the Apollo equipment.

The Unified S-Band system, with integral rendezvous ranging, further reduces the number of separate systems, however, its overall circuit complexity is inherently the greatest.

The advanced integrated circuit concept can be applied to each of the three previous concepts. Although it offers significant reductions in size and weight, its use would also require developing of new equipment. Equipment

which employs advanced circuit techniques will have a longer life and a longer reuse capability.

6.6.3.2 Baseline Definition - The telecommunication system is unaffected by vehicle configuration; only one baseline system is required. The selected baseline system is the Unified S-Band with separate rendezvous radar using state-of-the-art circuit techniques. A block diagram of this system is presented in Figure 6-57; its physical characteristics are presented in Table 6-32. This concept was selected because it uses flight-proven hardware and existing ground equipment. It also offers more than adequate reliability.

The selected radar is an interferometer such as that which was used on Gemini. It will operate with a co-operative target (transponder) to increase its range and accuracy. This technique was selected to minimize mechanical installation and environmental problems and to eliminate reaction torque problems which are inherent with a gimbaled parabolic antenna.

The reliability estimate for the baseline concept is given in Table 6-33.

6.6.3.3 Alternatives - Alternate telecommunication systems are based upon those concepts which were mentioned in Section 6.6.3.1. The alternate systems concepts are the "separate systems," the Unified S-Band with integral rendezvous ranging, and the advanced. Although the integrated circuit concept can be applied to all three, it will only be applied to the two Unified S-Band systems.

Table 6-34 summarizes the five telecommunication systems which are possible with the concepts presented. Table 6-35 presents the gross physical characteristics of these systems. The baseline system (TC-2) is larger than the "separate systems" concept (TC-1). This is the result of comparing Apollo equipment with Gemini equipment. The Apollo equipment is designed for a higher output power and employs more internal redundancy than Gemini equipment.

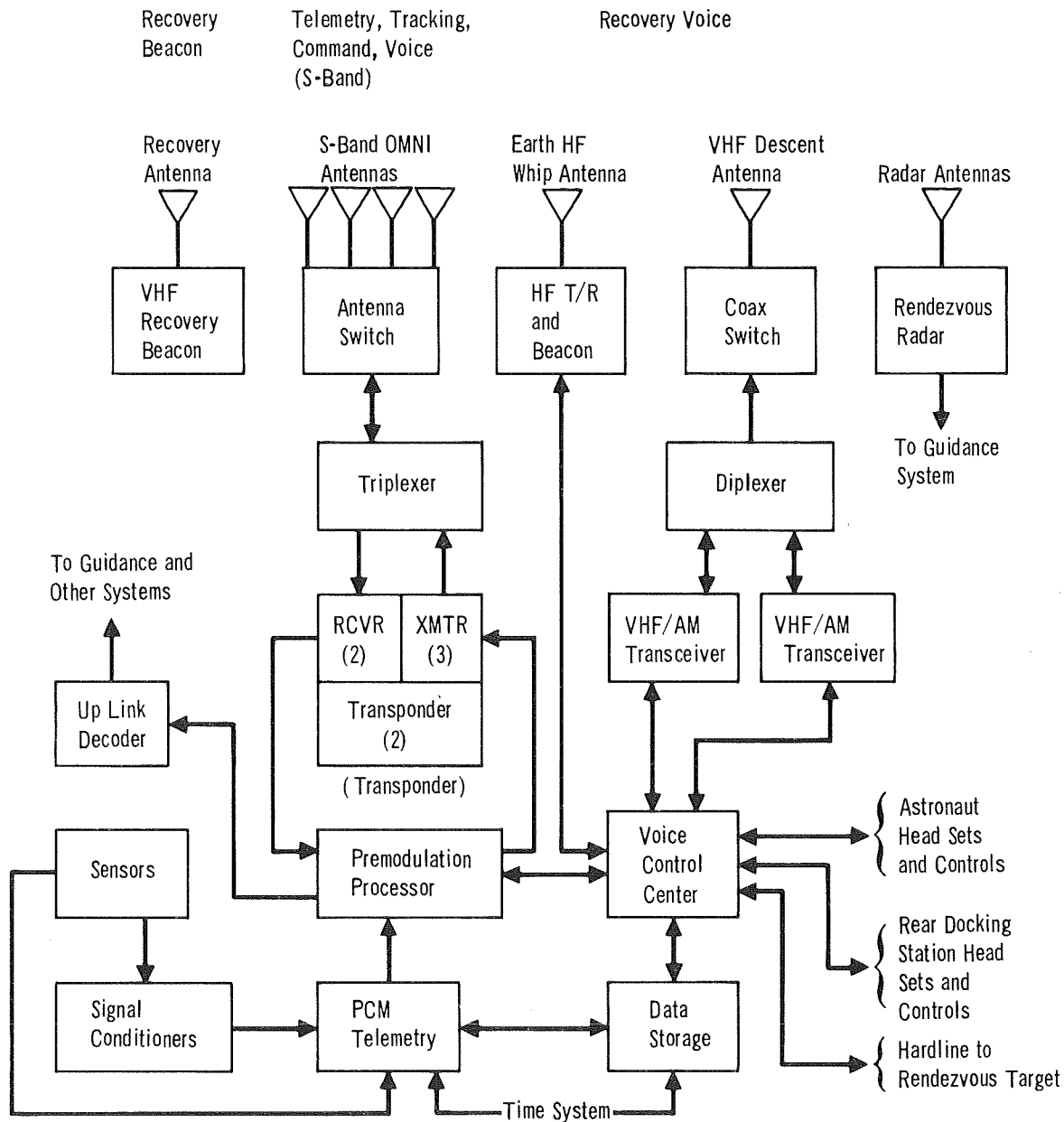
Two other variations should be mentioned although the design and cost variations cannot be evaluated now. One concept concerns the telemetry subsystem. A Stored Program Data Processor (SPDP) can be substituted for the PCM programmer and the multiplexers. The SPDP consists of a Digital Control and Combiner Unit (DCCU), a Flexible Format Generator (FFG), and up to 16

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Figure 6-57

BASELINE TELECOMMUNICATION SYSTEM BLOCK DIAGRAM (UNIFIED S-BAND WITH SEPARATE RADAR)



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TABLE 6-32

Physical Characteristics of the
Baseline Telecommunications System

Equipment	Size (ft ³)	Weight (lb)	Power (watts)
Telemetry Equipment			
PCM Programmer	.77	42	21
Signal Conditioners (2)	.5	25	40
Data Storage Equipment	.72	40	40
Special Sensors and Conditioners	2.5	40	40
Voice Equipment			
VHF/AM Transceivers (2)	.20	13	40
VHF Antenna System	.40	31	2
Audio Control Equipment	.14	8	20
Aft Voice Equipment	.05	3	5
Recovery Equipment			
VHF Recovery Beacon System	.07	6	10
HF Recovery System	.32	11	50
Rendezvous Radar			
Radar System	1.75	75	80
Unified S-Band Equipment			
Power Amplifier	.5	32	90
Triplexer	.05	2	2
Premodulation Processor	.17	15	13
S-Band Transponder	.69	32	26
S-Band Antenna System	.2	13	15
Digital Command System			
Command Decoder	.30	15	7
Totals	9.33	413	501

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TABLE 6-33

Telecommunications Reliability Estimate

<u>Component</u>	<u>P_s EST.</u>
Telemetry Equipment	.9999!
Voice Equipment	.9998
Recovery Equipment	.9999
Command Equipment	.9999
Ground Tracking Equipment	.9999
Unified S-Band Equipment	.9999
Radar	.9999

TABLE 6-34

Definition of Telecommunication Systems

System Concept No.	System Definition	Application (Vehicle Configuration)
TC-1	Separate Systems (Gemini Type Concept)	All
TC-2*	Unified S-Band System (Apollo Type Concept)	All
TC-3	Unified S-Band System with Integral Rendezvous Ranging Capability	All
TC-4	Same as TC-2 but with the use of Advanced Circuit Techniques	All
TC-5	Same as TC-3 but with the use of Advanced Circuit Techniques	All

* Baseline Concept

TABLE 6-35

Comparison of All Telecommunication Systems

System No.	Vol. Ft	Wt. Lbs	Pk. Pwr. Watts	Application (Vehicle Configuration)
TC-1	7.52	207	380	All
TC-2	9.27	413	450	All
TC-3	8.42	390	350	All
TC-4	7.60	336	400	All
TC-5	6.92	329	325	All

Remote Acquisition Units (RAU). It offers significant savings in weight and volume but its prime advantages are its versatility and expandability. Channel sample rates, number of channels sampled, and bit rates can be easily changed by either the crew or by ground command. The SPDP is also potentially capable of eliminating hundreds of pounds of wiring through its concept of remote multiplexing.

Another alternate concept, which can be applied to future cost studies, pertains to vehicles with the modular upper stage. The launch vehicle also has a telecommunication system with many similar functions and characteristics of that of the spacecraft. It is feasible to integrate the two telecom systems with an appreciable savings in the net weight and cost. However, as with the launch vehicle guidance system, this alternative will not be evaluated in this study.

6.6.4 On-board Checkout System - The primary purpose of the On-board Checkout System (OCS) will be to verify the adequacy of all spacecraft systems to work properly prior to separation from the space station. If a system malfunction occurs, the OCS should isolate the problem. It can also aid in pre-launch and post recovery tests. The size of the OCS will be largely determined by the level of fault isolation required from the system. Another factor in the system's size will be the degree of automation required.

Figure 6-58 is a block diagram of an OCS; physical characteristics are presented in Table 6-36. This system can monitor approximately 500 test points and can generate approximately 100 commands. An OCS of this size should be able to automatically isolate malfunctions to the black box level, provided that the crew performs switching functions (selection of redundant units, actuation of attitude hand controller, etc).

The selected size of the OCS (500 test points and 100 commands) has been influenced by the Gemini telemetry system and by prelaunch test facilities. Simply monitoring selected operating parameters (as is done with a data acquisition system) can provide a high level of system assurance and fault isolation. Therefore, the number of parameters to be monitored would be of the same order of magnitude as was monitored by the Gemini telemetry system which was about 350. Allowing for growth, additional subsystem complexity, etc., it is estimated that 500 test points will be sufficient for the OCS. During Gemini

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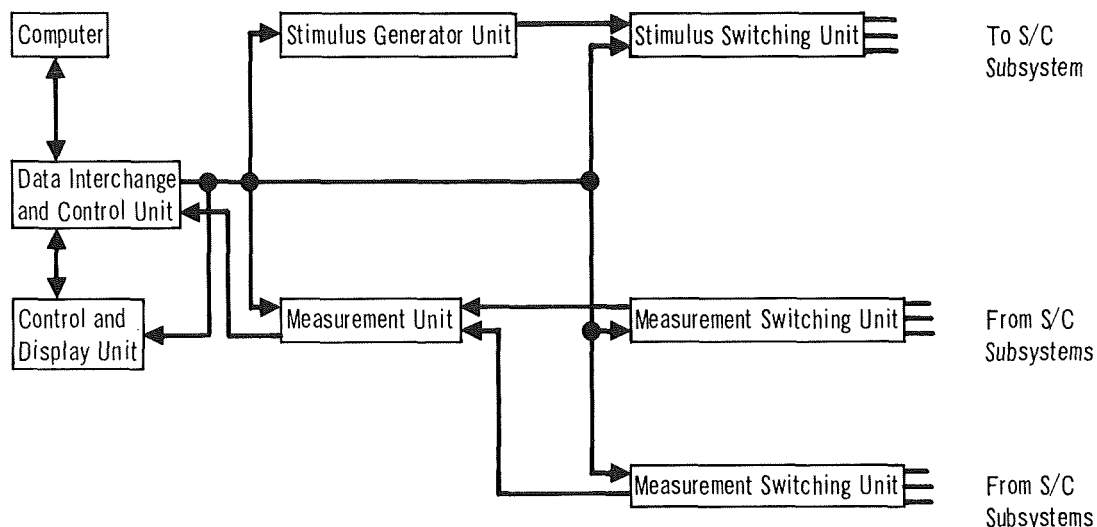
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TABLE 6-36
PHYSICAL CHARACTERISTICS OF ON-BOARD CHECKOUT SYSTEM

UNIT	NO. OF UNITS	SIZE (FT ³)	WEIGHT (LB)	POWER (WATTS)
Digital Computer	1	1.6	55	120
Data Interchange and Control Unit	1	.25	10	20
Control & Display Unit	1	.35	32	48
Measurement Unit	1	.25	8	48
Measurement Switching Unit	2	.5	21	80
Stimulus Generator	1	.35	13	105
Stimulus Switching Unit	1	.25	10	62
TOTALS		3.55	149	483

Figure 6-58

OCS BLOCK DIAGRAM



prelaunch operations, the ground complex had access to approximately 1500 Gemini test points, 20 percent of which were for signals and commands from the ground to the spacecraft. The estimate of 100 commands is determined by assuming that the ratio of commands to measurements will be the same for the OCS as for the prelaunch operation.

No alternative concepts or systems have been considered. Since the concept is relatively new to spacecraft applications, no meaningful trade-offs can be made. The system would also be unaffected by vehicle configuration.

6.6.5 Avionics Sizing Model - In general, the size and weight of the avionics for a logistics mission depend upon the concepts of spacecraft reuse and subsystem design. They are not influenced by such variables as crew size, cargo capacity, mission duration, and orbit parameters. Thus, the sizing model does not estimate avionics systems weights. These weights are estimated external to the program and are used as inputs. Based on subsystem weight, the model then estimates the weights of the mounting structure and the interconnecting cabling using empirical equations. The avionics subsystem can be distributed as desired between an entry vehicle and a mission module.

6.7 Power - Electrical and hydraulic power requirements, various concepts, and reliability are discussed in this section.

6.7.1 Requirements - The power supply requirements consist of electrical loads for the ballistic logistics vehicles, and of both electrical and hydraulic loads for the lifting body logistics vehicles. The hydraulic loads are only present during the landing phase of the lifting body mission; however, electrical loads are present during all phases of both ballistic and lifting body missions.

Electrical - The electrical power supplies must provide energy to load equipment connected to two types of distribution buses: main buses and pyrotechnic buses. The pyrotechnic buses are redundant and isolated from the main buses. Pyrotechnic bus loads include tubing cutters, structure separation shaped charges, and wire bundle guillotines. The buses will be powered from separate batteries. The main bus loads include all other subsystems which require electrical power, including environmental control, guidance and navigation, stabilization and control, and telecommunications.

The power to be provided for all loads is nominally 28 VDC. Equipment or subsystems requiring ac power or dc power of a different voltage level are provided such through the use of self-contained power converters. The level of power required for the family of logistics vehicles is based on review of previous studies, see References 4-1 and 4-2, as well as the Mercury, Gemini, and Apollo spacecraft.

Average power requirements for each mission phase and reuse category are shown in Table 6-37. Figure 6-59 shows the range of average electrical power for the logistic vehicle configurations. The variation in load results from crew size variation from two to twelve, the additional guidance and navigation equipment for recoverable vehicles, and the control requirements for the sailing recovery system in configurations IA(land landing) and IB.

Hydraulic - Hydraulic power is provided to move the aerodynamic control surfaces on the lifting body spacecraft during re-entry and during landing flight through the earth's atmosphere. Table 6-37 shows the peak and steady-state hydraulic horse power requirements, based on hinge moment derived in the study reported in Reference 6-8.

TABLE 6-37
POWER SUBSYSTEM PERFORMANCE REQUIREMENTS

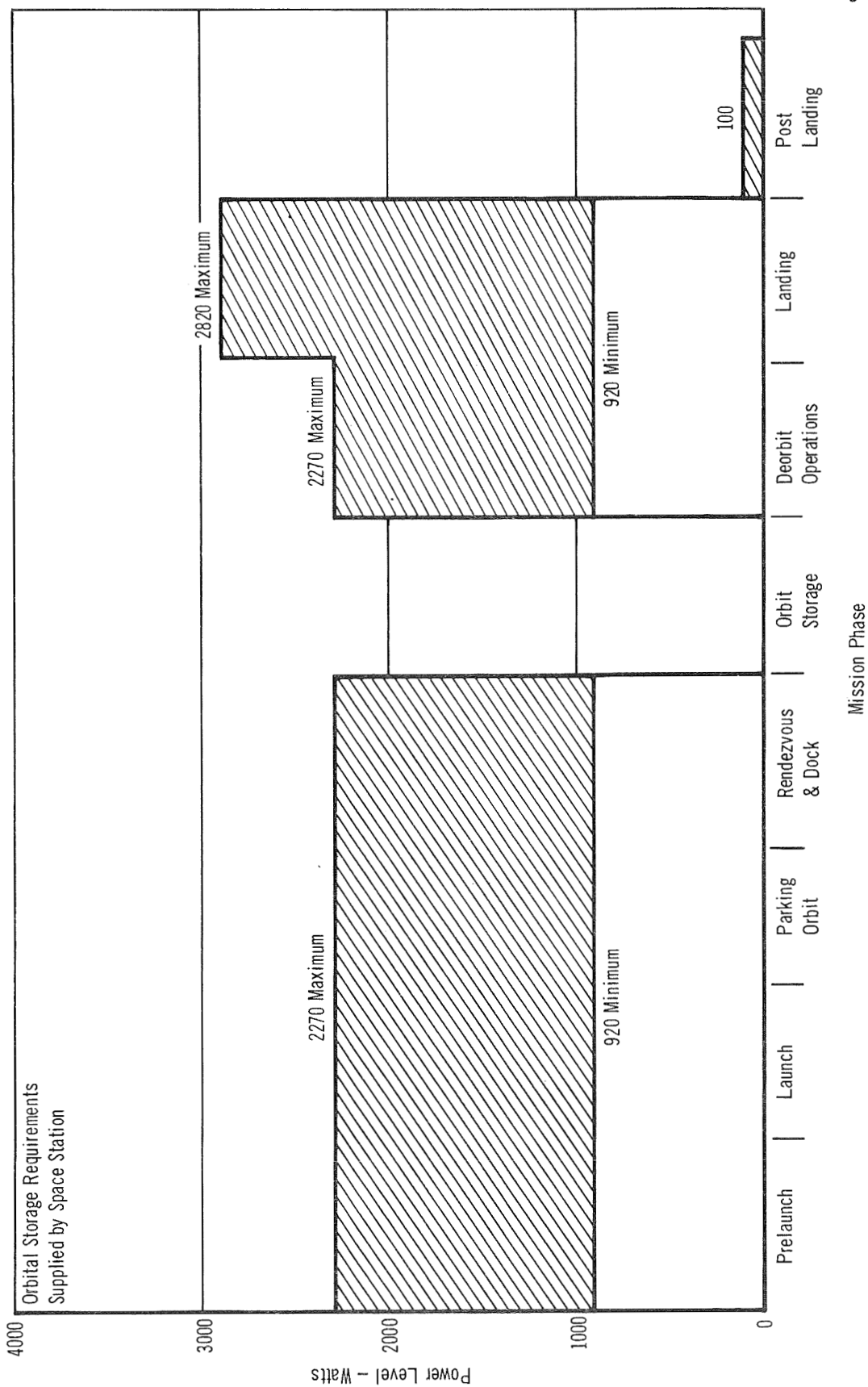
Mission Phase	Functional Requirements	Reuse Concept							
		IA (Water Landing)	IA (Land Landing)	IB	IC	ID, IE, IF	IIA & IIB	IIC	IID, IIE, IIF
Electrical Power Requirements									
Prelaunch (After Transfer to Internal Power) Launch (400 Sec) Orbit Maneuver (2-24 Hr) Deorbit Operations (2-24 Hr) Entry (90 Min)	Supply Electrical Power to All Spacecraft Subsystems	Main Bus Avg 750 Watts Plus 85 W/Man (920 to 1770 W) Pyro Bus 2.0 Kw Pulses	Main Bus Avg 1.0 Kw Plus 85 W/Man (1170 to 2020 W) Pyro Bus 2.0 Kw Pulses	⇒	⇒	Main Bus Avg 1.25 Kw Plus 85 W/Man (1420 to 2270 W) Pyro Bus 2.0 Kw Pulses	Main Bus Avg 1.0 Kw Plus 85 W/Man (1170 to 2270 W) Pyro Bus 2.0 Kw Pulses	⇒	Main Bus Avg 1.25 Kw Plus 85 W/Man (1420 to 2270 W) Pyro Bus 2.0 Kw Pulses
Orbital Storage (1-90 Days)	Spacecraft Electrical Power Shutdown. Stand-by Power Supplied by Space Station	-	-	-	-	-	-	-	-
Landing (20 Min)	Supply Electrical Power to All Spacecraft Subsystems	Main Bus Avg 750 Watts Plus 85 W/Man (920 to 1770 W) Pyro Bus 2.0 Kw Peaks	Main Bus Avg 1.8 Kw Plus 85 W/Man (1970 to 3000 W) Pyro Bus 2.0 Kw Peaks	⇒	Main Bus Avg 1.0 Kw Plus 85 W/Man (1170 to 2020 W) Pyro Bus 2.0 Kw Peaks	Main Bus Avg 1.25 Kw Plus 85 W/Man (1420 to 2270 W) Pyro Bus 2.0 Kw Peaks	Main Bus Avg 1.0 Kw Plus 85 W/Man (1170 to 2270 W) Pyro Bus 2.0 Kw Peaks	⇒	Main Bus Avg 1.25 Kw Plus 85 W/Man (1420 to 2270 W) Pyro Bus 2.0 Kw Pulses
Post Landing (2-24 Hr)	Provide Electrical Power for Recovery Aids, Lights, Ventilating Fans	Main Bus Avg 100 W	⇒	⇒	⇒	⇒	⇒	⇒	⇒
Hydraulic Power Requirements									
Entry (30 Min) Landing	Provide Power to Move Aerodynamic Control Surfaces	-	-	-	-	-	Peak - 14 Hp Steady State - 5 Hp	Peak - 50 to 290 Hp Steady State - 8 to 32 Hp	IID Peak - 80 to 560 Hp Steady State - 10 to 60 Hp IIE & IIF Peak - 290 to 1940 Hp Steady State - 32 to 200 Hp

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Figure 6-59

ELECTRICAL LOAD PROFILE



6.7.2 Concepts - The types of electrical power sources applicable to Earth orbital logistic missions have been extensively studied in past programs, including References 4-1, 4-2, and 6-9. Both solar and nuclear power sources were not considered for the logistics missions, primarily due to the short operating periods, and to deployment and shielding problems. The advantages and disadvantages of the chemical power sources considered for the electrical load requirements are shown in Table 6-38. The battery data used has been derived from information obtained previously. Current performance and cost data have been obtained from selected fuel cell and chemical dynamic power source suppliers, and combined with information on-hand for updating. The chemical dynamic power source propellant consumption rates verified previous decisions not to consider such equipment for only electrical requirements. Silver-zinc batteries and fuel cells can readily supply the electrical power requirements shown in Section 6.7. The trade-off is between the high reliability, low volume, and low cost features of batteries, and the low weight, reusable features of fuel cells.

Power systems for aerodynamic control surface actuation, required for lifting body logistics vehicles, have been studied in Reference 4-2, 6-9, and 6-10. Again, solar and nuclear power sources were not considered. Higher power requirements of aero-control surface actuation, and short operating periods within the Earth's atmosphere, eliminated fuel cells from the chemical power sources. Current performance and cost data have been obtained to up-date chemical dynamic power source characteristics. Battery-motor-pump units and storable propellant turbine and pump units were selected as candidates. Batteries were attractive for low cost and simplicity but resulted in an excessive weight penalty as vehicle sizes increased.

Table 6-39 summarizes the concepts for the power subsystem and indicates the vehicle applications.

6.7.3 Baseline Subsystem Description - The power subsystem provides prime power. Two prime sources supply all the spacecraft power requirements. An electrical power source, either batteries or fuel cells, supplies the electrical requirements in both the ballistic and lifting body spacecraft. A hydraulic source provides power for the aerodynamic control surface actuation.

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Table 6-38
Electrical Power Source Comparison

Electrical Power Source	Advantages	Disadvantages
Silver-Zinc Batteries	- o High Reliability - o Low Volume - o Low Cost - o Low Development Risk - o Extensive Flight Experience - o Minimum Controls and Monitors	- o High Weight - o Short Life - o Not Reusable
Fuel Cells	- o Low Weight - o Reusable, Potentially - o Low Development Cost on Existing Systems - o Proven Concept Feasibility Through Apollo and Gemini Flight Experience - o Low reactant consumption Rate - o Good Growth Potential - o Start-Stop Capability Allows Dormant Orbital Storage	- o High Cost - o Large Volume - o Complex System - o Auxiliary Reactant Controls and Monitors Required - o Active Cooling Required
Hot Gas Turbine/Generator	- o Low Fixed Weight and Volume for high power output - o Reusable - o Operates on Propellant which is storable	- o High Cost - o High Propellant Consumption Rate - o Extensive Development Required - o Auxiliary Controls and Monitors Required - o Possible Flight Path Perturbations From Exhaust Gases

Table 6-39
Power Subsystem Concept Summary

Concept No.	Ascent and Phasing	Re-entry	Vehicle Application
EP-1*	Batteries in MM	Batteries in EV	IA, IIA
EP-2*	Fuel Cells and Batteries in EV, Reactant Supply in MM	Batteries in EV	IB, IIB
EP-3*	Fuel Cells, Batteries and Reactant Supply in EV	Batteries in EV	IC thru IF IIC thru IIF
EP-4	Batteries in EV	Batteries in EV	IA (Land Landing only) IB thru IF IIA thru IIF
EP-5	Fuel Cells and Reactant in MM	Batteries in EV	IA, IB IIA, IIB
Hydraulic Power			
HP-1	Batteries		IIA, IIB
HP-2*	Turbine		IIA thru IIF

* Baseline Concept

6.7.3.1 Electrical Power - There are basically two baseline electrical power subsystem configurations, one for the expendable entry vehicles and one for the reusable entry vehicles. An expendable system is shown in Figure 6-60. This concept is an all-battery concept using manually activated silver-zinc batteries. Most of the batteries, those used for main bus loads during ascent and return phasing, are located in the mission module. The squib batteries and those used for main bus loads during re-entry and post landing are located in the entry vehicle. Two batteries provide isolated power sources for each redundant squib bus which power the pyrotechnic devices. The main bus load requirements are supplied by multiple batteries. For this study, the energy required from the squib batteries is assumed to be a constant 400 watt hours. All of the batteries are mounted on coldplates through which spacecraft coolant is circulated.

The reusable electrical power subsystem is shown in Figure 6-61. In this concept, fuel cells are used to supply average main bus power with batteries supplying peak power loads and main bus power during re-entry and landing. The batteries are the same types as those used in the expendable system. All are installed in the entry vehicle. This concept applies to both the ballistic and to the lifting body spacecraft. The two fuel cell modules are installed in the re-entry module. Each module contains 30 cells in series, resulting in a nominal output voltage of 28 VDC. Each module can provide the logistics vehicle average power requirements, thereby providing 100 percent redundancy through the use of two modules. Both fuel cell modules are operated in parallel during active mission phases up to re-entry, at which time both are shutdown. Temperature is controlled through the use of spacecraft circulating coolant. The potable product water, generated during fuel cell operation, can be either collected or vented. Overboard venting reduces the fuel cell heat load by 30 percent. The hydrogen and oxygen reactants are stored cryogenically from tanks installed in either the mission module or in the entry vehicle.

The fuel cell modules can be shut-down and re-started by only using a small amount of power to open valves and to operate power relays during start-up. The fuel cell modules can provide rated power output for more than 2500 hours, without major overhaul.

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Figure 6-60

ELECTRICAL POWER SYSTEM (VEHICLE I A)

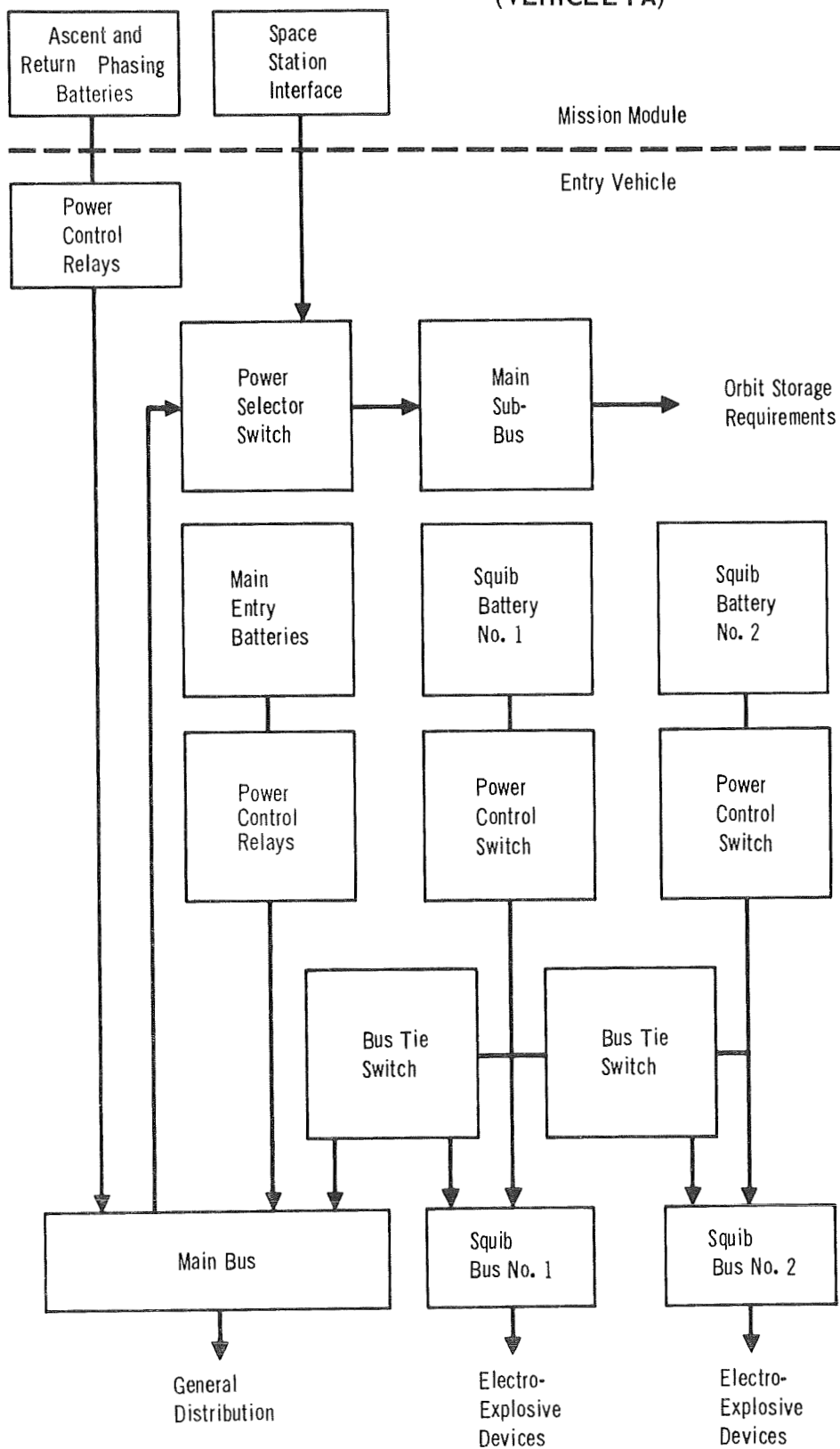
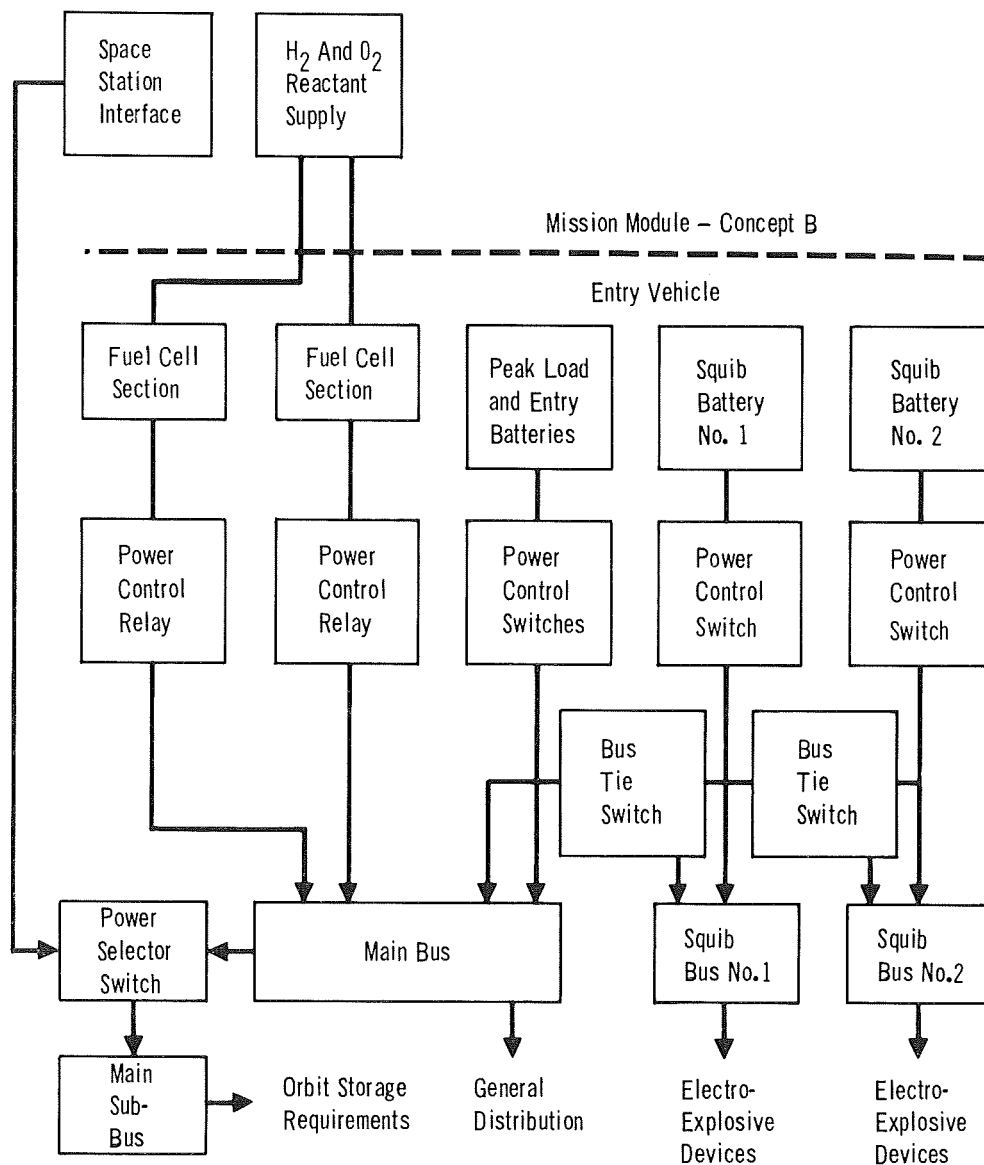


Figure 6-61

**ELECTRICAL POWER SYSTEM
(VEHICLES 1B THRU 1F, 11B THRU 11F)**



For the logistics vehicles operation periods, the capability of nominally 50 missions is indicated. The battery selected, for extended activated life, gives the inherent capability to provide many discharge-charge cycles. However, battery replacement before each flight has been selected because of the relatively low cost factor and the elimination of refurbishment and checkout operation which result. Calendar time periods must be considered, with regards to O-ring replacement, exposure to ground testing, etc., in evaluating any refurbishment and reuse factors.

6.7.3.2 Hydraulic Power - Hydraulic power is used to operate the aerodynamic control surfaces of the lifting body spacecraft for maneuvering in atmospheric flight. The two independent hydraulic systems, except for actuators, are powered by independent auxiliary power supplies. Each system can provide all the aero control power requirements.

Each auxiliary power supply consists of a turbine-gear box unit, propellant tanks, pressurization system, and the valves and plumbing for system servicing and operation. A schematic is shown in Figure 6-62. The axial flow turbine is driven by the hot gases generated by the decomposition of monopropellant hydrazine. Helium is used as the propellant tank pressurant gas. The turbine-gear box unit is capable of 50 hot operations of 18 minutes before overhaul.

Each hydraulic system is conventional and consists of a positive displacement, variable delivery pump and accumulator, a boot-strap reservoir, servo actuators, and valves and plumbing. The servo actuators are conventional dual linear types controlled by quadruple redundant servo valves. Nominal system operating pressure is 3000 psig. Figure 6-63 shows the schematic of the hydraulic system.

6.7.3.3 Reliability - The reliability of the prime power supplies was estimated and is shown in Table 6-40. These estimates are for bus power only and do not reflect the reliability gains that are inherent when switchover between buses is provided.

Figure 6-62

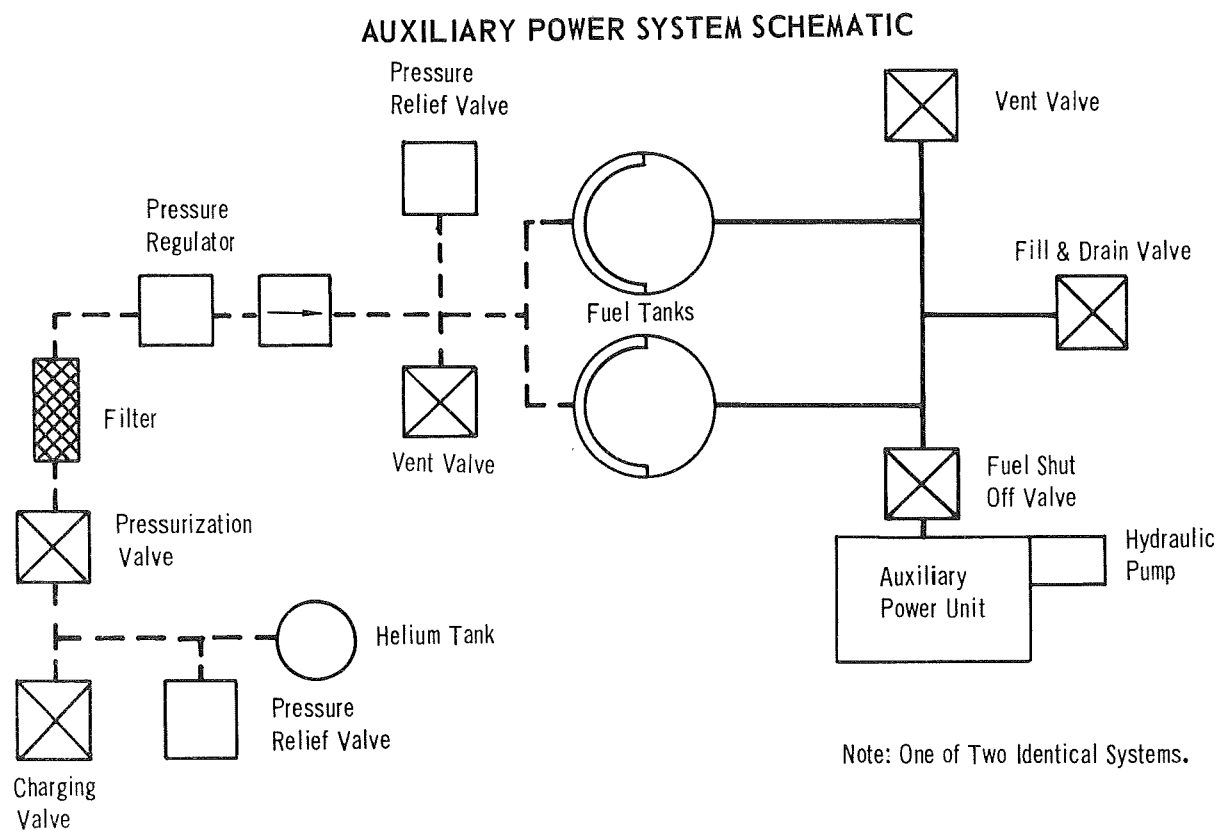
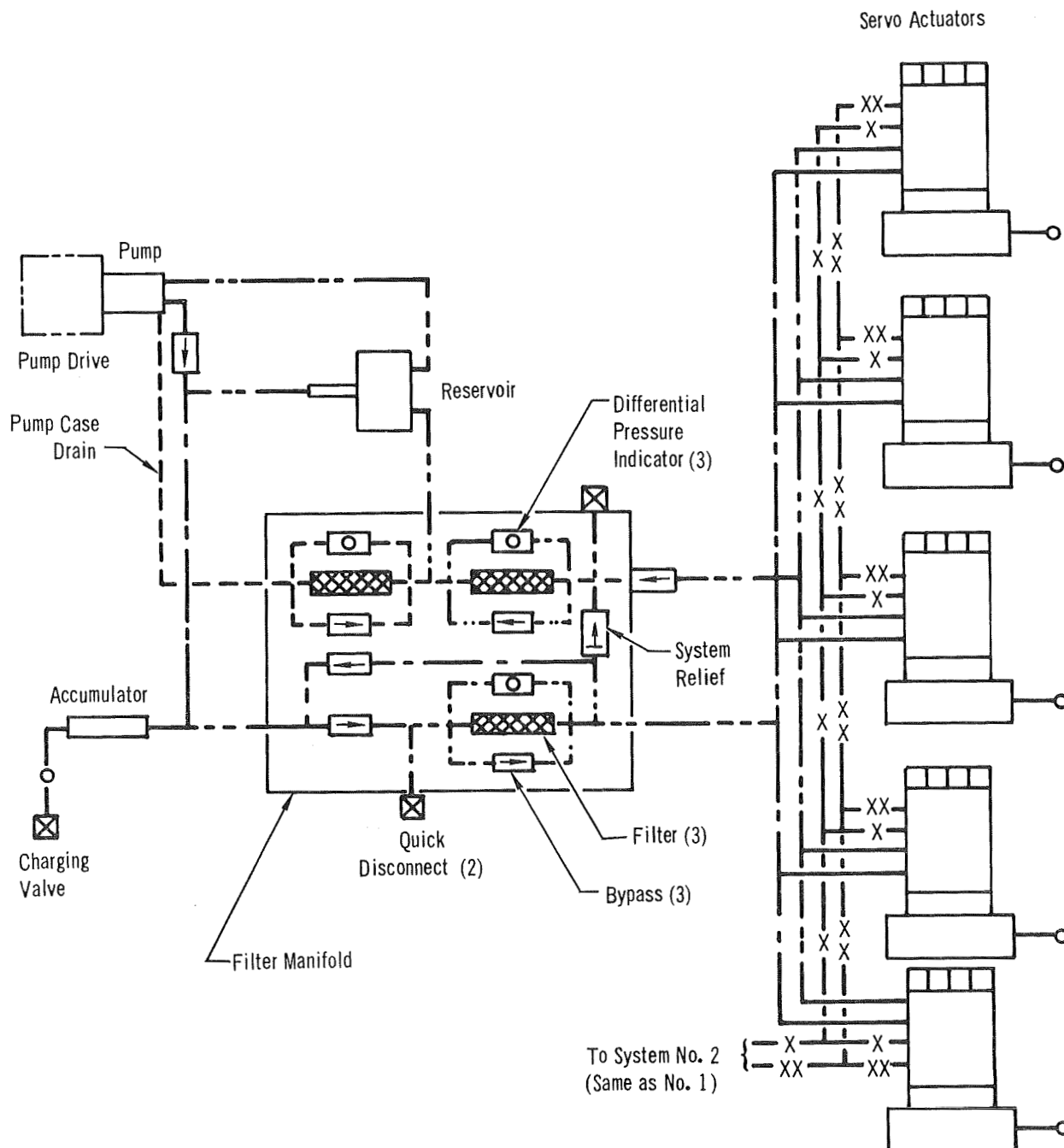


Figure 6-63

HYDRAULIC SYSTEM SCHEMATIC – SYSTEM NO. 1



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6.7.4 Power System Sizing Model - The power system sizing model sizes fuel cell systems or batteries according to requirements. These requirements vary with the crew size and the subsystem power. In the modular vehicles, the batteries and the fuel cell system can be located in either the mission module or entry vehicle or split between them. For the lifting body vehicles, the model sizes a hydraulic system for control surface actuation and either a monopropellant turbine-pump or a battery-motor-pump for power. The sizing of this aero control power system is based on vehicle size.

Table 6-40

Reliability of Prime Power Supplies

Prime Power Supply	Redundancy	Reliability
Main Batteries	3 of 4 required	.99985
Squib Batteries	1 of 2 required	.99997
Fuel Cells	1 of 2 required	.999804
Turbine	1 of 2 required	.99999

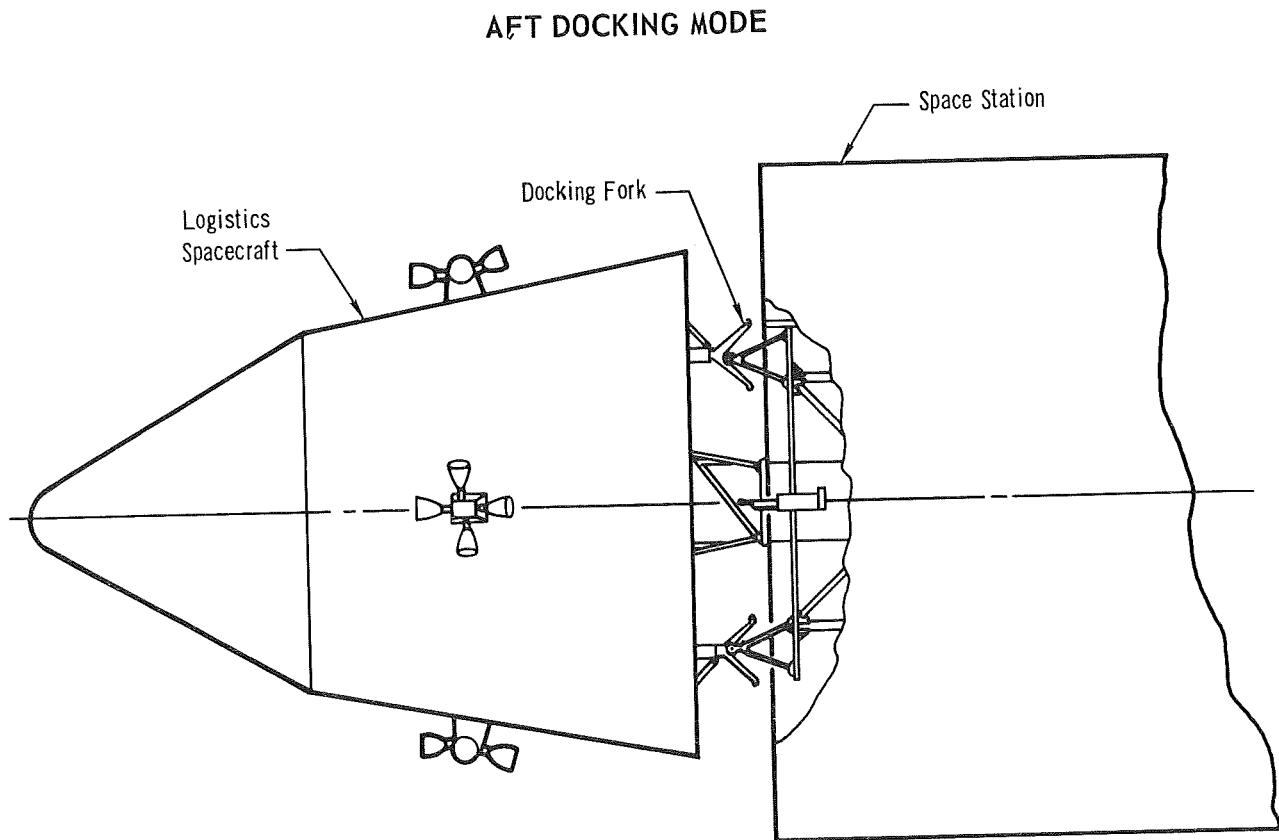
6.8 Miscellaneous Subsystems - The subsystems concepts defined in this section are the docking system, the recovery system, and aerodynamic control surfaces.

6.8.1 Docking System - Two docking modes are assumed: aft docking for configurations with a separate cargo module and side docking for those configurations having the cargo integral with the entry vehicle. The aft docking mode uses the universal ring segment fork docking mechanism shown in Figure 6-64. This mechanism consists of two ring segments and two v-shaped forks with latching mechanisms. The ring segments and forks are mounted on the spacecraft 90 deg apart, alternating ring segment and fork. An identical system is mounted on the space station. Docking starts with the maneuvering of one ring segment into a fork. A small cable triggers the latch on the fork, captivating

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Figure 6-64



the ring segment. The spacecraft then rotates the other ring segments into captive positions and the hooks lock on, rigidly centering the spacecraft with the docking port on the space stations.

For the large ballistic vehicles where cargo transfer through aft end is unattractive because of the location of the propulsion system and for the lifting vehicles which are not readily adaptable to aft docking, the side docking mode is used. In this mode, retractable probes adjacent to the pilot's window are used. The probes are maneuvered into the space station receptacles and form a hinge line for pivoting. Vehicle attitude control thrusters are used to pivot the vehicle to a side moored position to connect the transfer tunnel. This concept is shown in Figure 6-65.

For the aft docking mode, a crew station must be provided in the mission module. This crew station contains personnel restraints, maneuvering and attitude hand controllers, attitude display group, environmental controls and intercom outlets and controls.

6.8.2 Recovery - The different reuse categories, aerodynamic configurations, and the range of vehicle sizes all influence the recovery subsystem configuration. As a result, there are several recovery subsystem configurations as shown in Table 6-41. The recovery system includes: (a) the parachute system to recover the ballistic vehicles and the abort escape capsule in both the ballistic and lifting vehicle, (b) the impact attenuation system, and (c) the landing assist rockets. The first two are discussed below. The landing assist rockets are discussed in Section 6.4.5.

6.8.2.1 Ballistic Spacecraft Systems - For the ballistic spacecraft, the recovery system uses ringsail chutes for the water landing expendable spacecraft and, also, because of their size, for the large integral spacecraft. Lifting canopies are used for the modular spacecraft. All of the land landing spacecraft touch down with the heat shield down on a skid gear that extends through the heat shield. Landing assist rockets are used to reduce the touch-down velocity to 20 ft/sec from the descent velocity when using a cluster of three 100-ft diameter ringsail parachutes. In all the ballistic vehicles, the crew section is separated from the remainder of the spacecraft for abort. Since the parachute system is located in this portion of the spacecraft, a

Figure 6-65

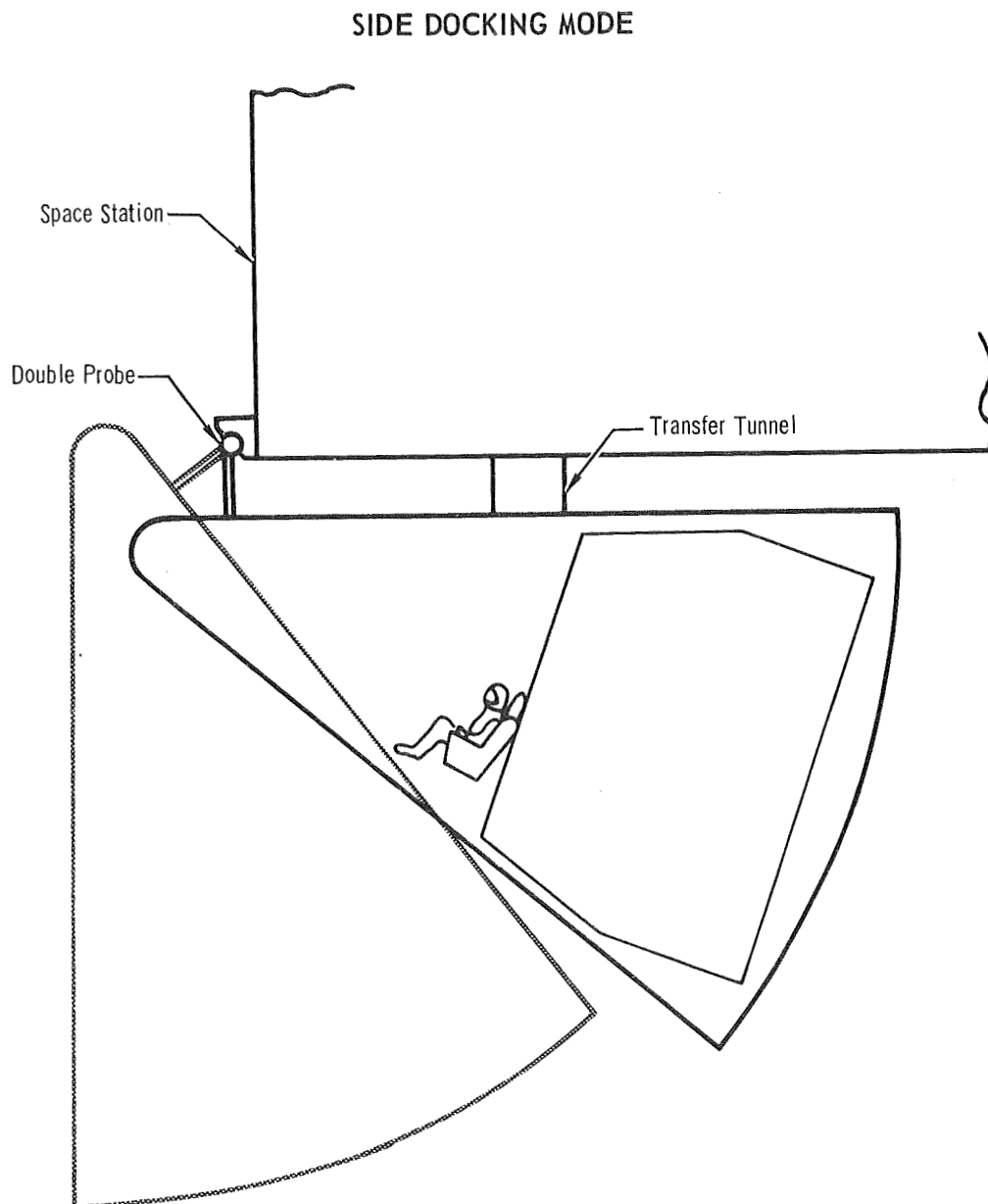


TABLE 6-41
RECOVERY SYSTEM CONFIGURATIONS

CONFIGURATION	PRIMARY RECOVERY MODE	PRIMARY PARACHUTE SYSTEM	LANDING ASSIST ROCKETS	ABORT/BACKUP PARACHUTE SYSTEM	IMPACT ATTENUATION SYSTEM
IA	Water	x (1)		x (5)	
	Land	x (2)		x (6)	x
IB	Land	x (2)		x (6)	x
IC, ID, IE, IF	Land	x (1)	x	x (5)	x
IIA, IIB	Land		x	x (4)	x
IIC, IID, IIE, IIF	Land		x	x (3)	x

- (1) Require aerodynamic decelerator only.
- (2) Required lifting capability to reach selected sites.
- (3) Systems sized by escape capsule only.
- (4) Systems sized by entire entry spacecraft.
- (5) These configurations require multiple canopies for normal operation.
Backup systems are not required, since canopy failures can occur without serious degradation in performance. For abort, primary system canopies can be used.
- (6) A system duplicating the primary system will be used for backup.

separate abort parachute system is not required.

The parachute system consists of: (a) a drogue parachute for initial deceleration to reduce the dynamic pressure to a safe level for deployment of the main canopy and for spacecraft stabilization and orientation prior to main canopy deployment, (b) a main canopy, either ballistic or lifting, for further deceleration, (c) a drogue mortar for the initial deployment of the drogue chute; and (d) the risers, lines, and releases. The drogue is used as the extraction chute for the main canopy.

For the vehicles using the lifting canopy (IA-land landing and IB) the Barish sailwing which has an L/D of about 2.3 was selected. In these vehicles, the primary system consists of two drogues and mortars, either of which can decelerate the spacecraft and extract the main canopy; a main canopy; and a sailwing control unit. A back-up system, consisting of a duplicate sailwing, control unit, and drogue is provided. A typical installation is shown in Figure 6-66.

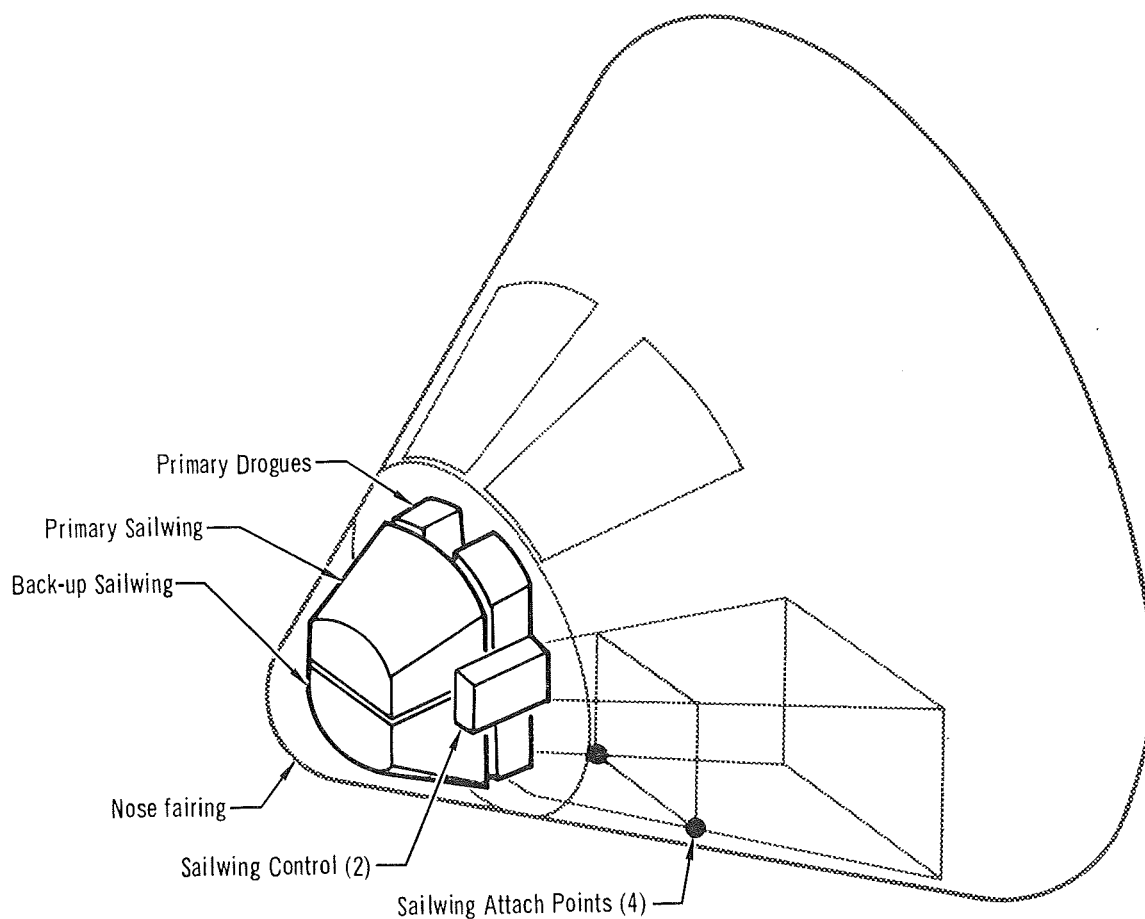
For vehicles using ballistic parachutes (IA-water landing, IC, ID, IE, IF), the main canopy is the ringsail such as used on Gemini and Apollo. In these, the parachute system consists of two drogues, each with a separate mortar and each capable of decelerating the spacecraft to the desired dynamic pressure, and three main chutes, each with its pilot chute and pilot chute mortar. No separate back-up system is required since only two of the main chutes need to function for a safe recovery.

The impact attenuation system for the land landing spacecraft consists of three skids which extend through the heatshield similar to that shown for Vehicle IB in Figure 6-67. To permit extension of the gear, the heat shield contains three plugs which are pushed out by the skids as the gear extends. Each of the three gears is a fixed air-oil oleo shock strut that is fully compressed and unpressurized when retracted. It is held by a tension strap which is broken when the strut is pressurized. A circular skid is gimbal mounted to the lower end of each shock strut piston.

For the water landing version of Concept IA, no impact attenuation system is required. This spacecraft will land on one corner of the heatshield in the manner of Gemini.

Figure 6-66

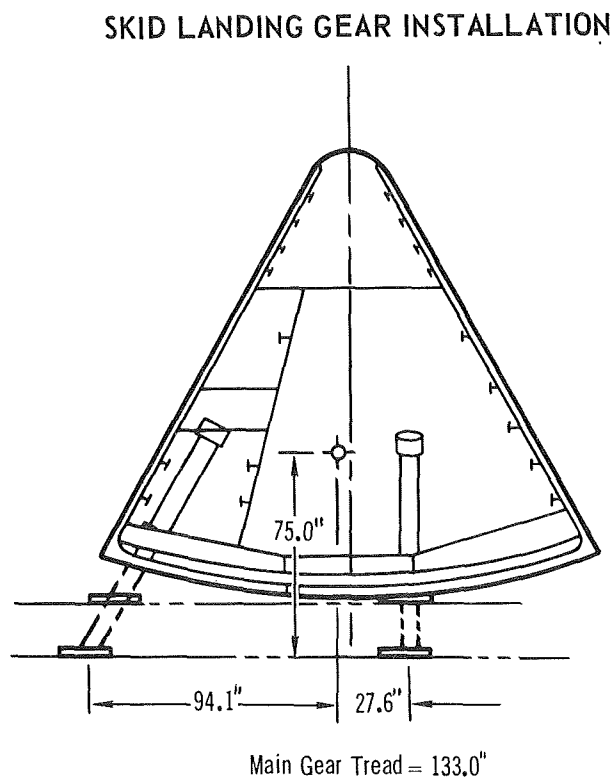
LIFTING PARACHUTE SYSTEM INSTALLATION



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Figure 6-67



6.8.2.2 Lifting Spacecraft Systems - Since the lifting spacecraft configuration normal recovery mode is a horizontal landing, the only requirement for parachute systems is to recover the abort escape capsule. For abort escape in Vehicles IIA and IIB, the entire entry vehicle serves as the escape capsule. For the other vehicles, the portion ahead of the cargo section is separated. For all vehicles, the escape capsule contains a ringsail chute system such as described previously for ballistic spacecraft.

The impact attenuation system is a typical tricycle landing gear with skids on the main gear, which is located aft, and a castering nose wheel in the front. This gear is stowed within the entry vehicle and extends through doors in the thermal protection system.

6.8.3 Aerodynamic Control Surfaces - Control of the ballistic vehicle requires no external control surfaces. The M2/F2 vehicle, however, requires aerodynamic surfaces for control. Typical structural concepts and materials employed for the M2/F2 control surface are discussed here. Material selection for the surfaces is dictated by the most severe heating and loading conditions experienced by any surface during entry or the high dynamic pressure abort situation.

Figure 6-68 illustrates the lower flap structural design. Ablative or radiative heat protection panels are mounted on the external side of a super alloy substructure. The figure shows an ablative heat protection panel. The internal sheet closure panel is also made from super alloy. The closure structure inside the wall, from which the panel rotates, and aft panel members are fabricated from super alloys due to the temperature developed by wash effects around the edges of the panel while extended. The lower flap is extended approximately 45 degrees from the mold line during some phases of entry for a spacecraft trim at 11-degree angle-of-attack.

The upper flaps are in the closed position during entry, down to approximately Mach 4. Therefore, the substructure, heat shield panels, and inside closure sheet are made from titanium. Figure 6-69 shows the recommended design.

The fin construction is shown in Figure 6-70. The substructure is constructed from a super alloy material. The heat protection panels on the inside and outside, including the rudder, are built with ablative or radiative panels. The figure shows a radiative panel approach.

Figure 6-68

LOWER CONTROL SURFACES

STRUCTURAL CONCEPT

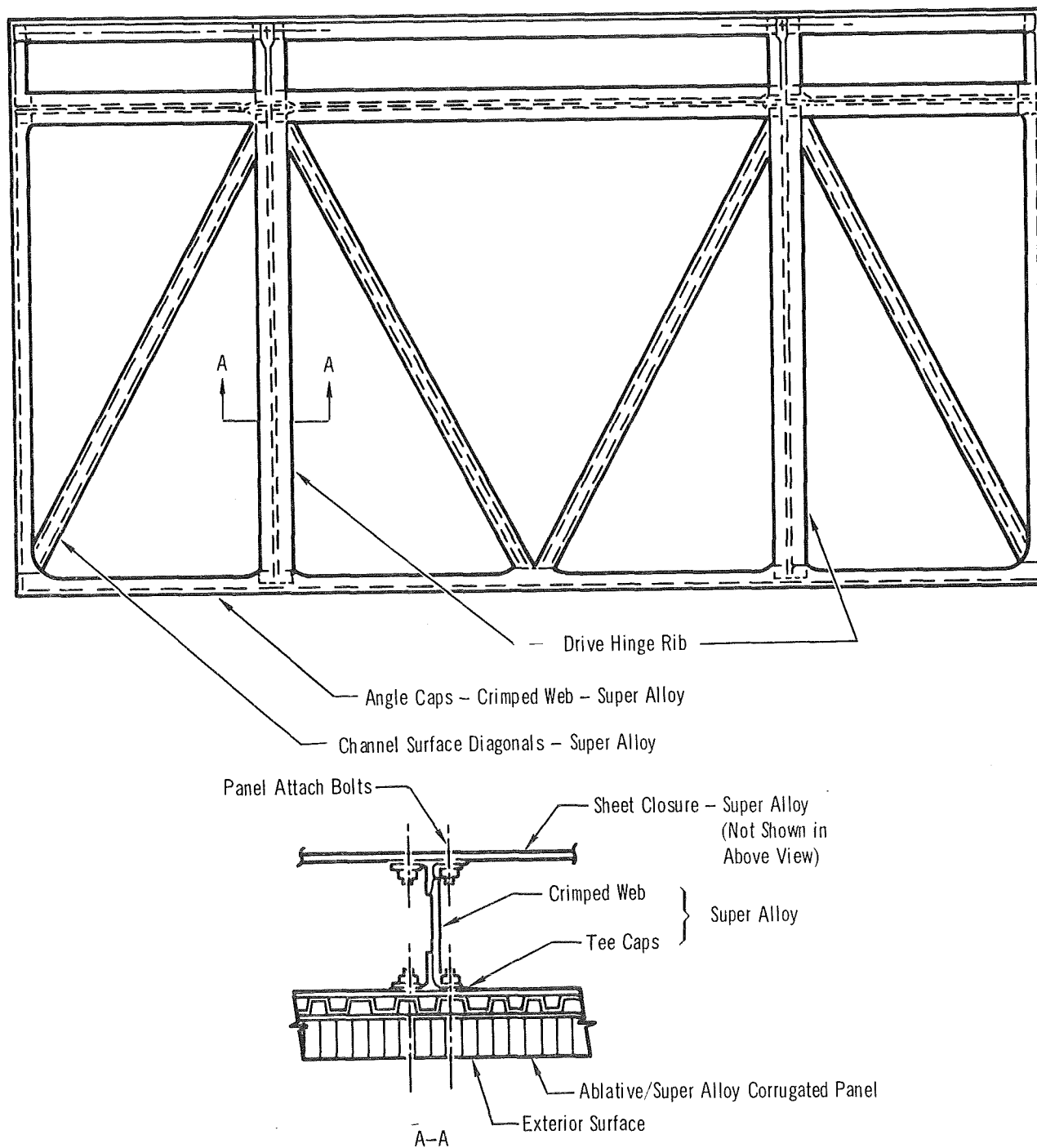


Figure 6-69

UPPER CONTROL SURFACE STRUCTURAL APPROACH

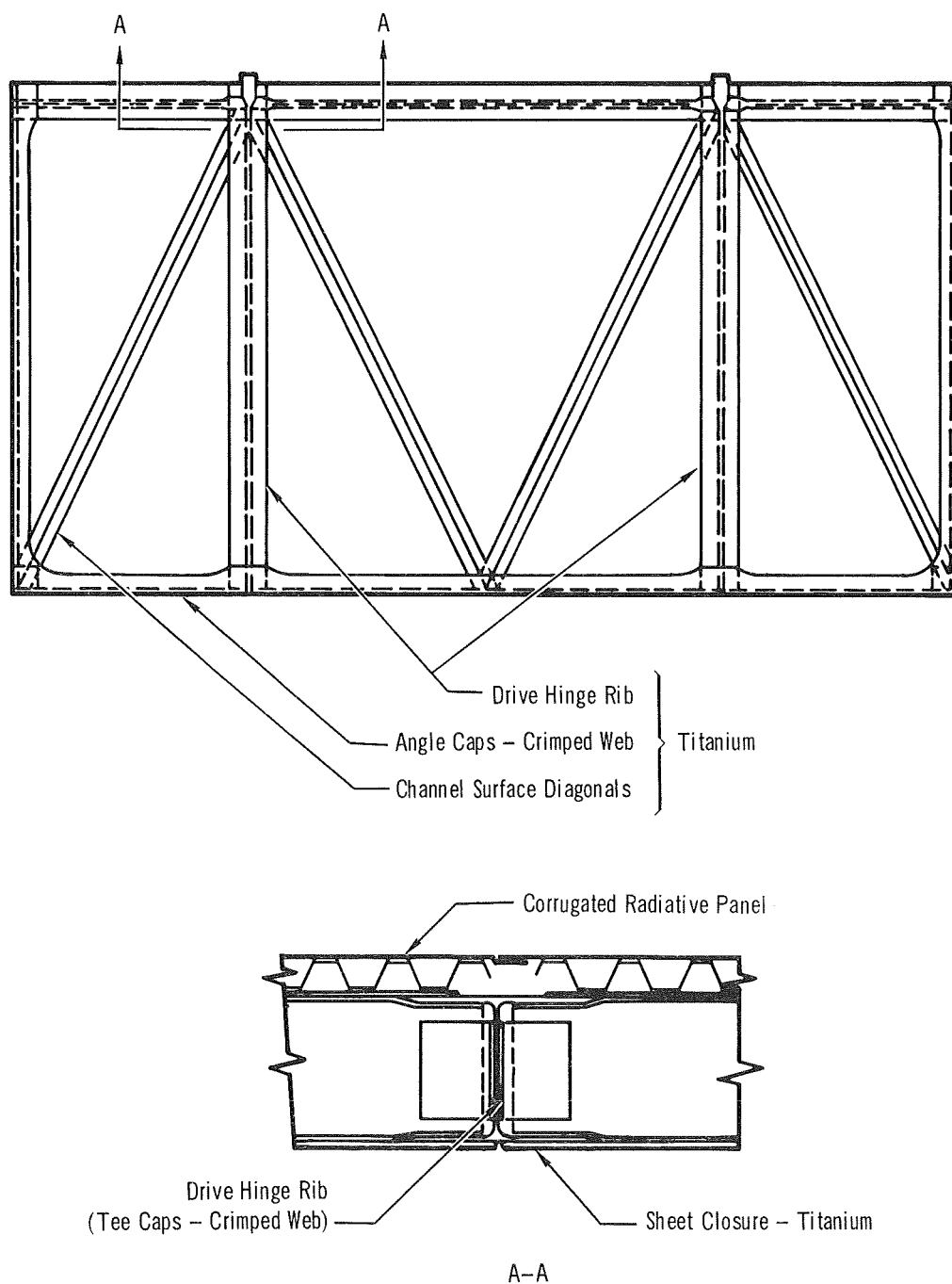
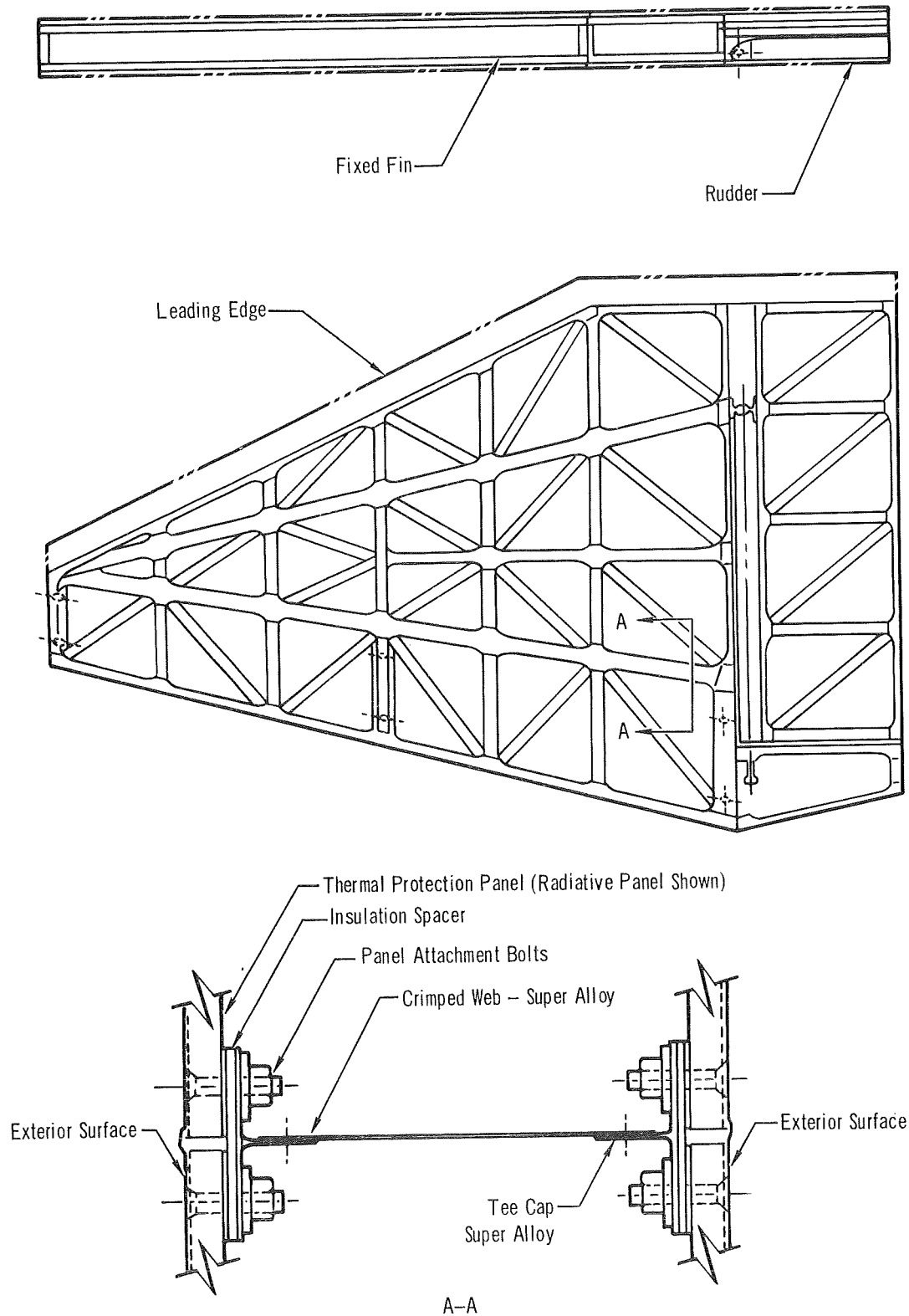


Figure 6-70

FIN AND RUDDER STRUCTURAL CONCEPT



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REFERENCES

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- 4-2 "Multipurpose Reusable Spacecraft Preliminary Design Effort", McDonnell Report No- F749, dated 30 October 1967
- 4-3 "Mission Requirements of Lifting Systems - Engineering Aspects", McDonnell Report No. B831, dated 27 August 1965
- 4-4 "Mission Requirements of Lifting Systems - Operational Aspects", Boeing Report No. D2-82531-3, dated August 1965
- 4-5 Aerospace Corp. Ltr 67-2310 C211, dated 12 September 1967, "Multiple Site Return Probability"
- 4-6 "A Lifting Re-entry Horizontal-landing Type Logistic Spacecraft", Boeing Report D2-22921, dated 3 February 1964
- 4-7 "Research Studies and Analysis to Define Manned Lifting Entry Flight Environment", NASA CR-639, dated November 1966
- 5-1 "A Lifting Re-entry Horizontal-Landing Type Logistic Spacecraft", Boeing Report D2-22921, dated 3 February 1964
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- 6-1 "The U.S. Standard Atmosphere", published by NASA, USAF, and the U.S. Weather Bureau, December 1962
- 6-2 MIL Handbook 5A, February 8, 1966, "Metallic Materials and Elements for Aerospace Vehicle Structures"
- 6-3 "Bending and Comparison Tests of Pressurized Ring-Stiffened Cylinders", NASA TN D-360, dated April 1969

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- 6-4 "Factor of Safety versus Vehicle Weight", USAF (RTD) TDR-63-4048, dated 1963.
- 6-5 AIAA Paper 68-757, "A Semi-Empirical Method of Correlating the Thermal Performance of Charring Ablative Materials", S. A. Mezines, January 1968
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- 6-7 WADC-TR 59-610, "Design Procedures for Computing Aerodynamic Heating at Hypersonic Speeds", W. L. Hankey, R. D. Neumann, June 1960
- 6-8 "Minimum Manned Lifting Body Vehicle", MAC Report E145
- 6-9 Multipurpose Reusable Spacecraft Preliminary Design Effort (Category A)
USAF SAMSO TR67-4
- 6-10 Lifting Reentry Test Vehicle Preliminary Design for FDL-7MC and FDL-5MA Configurations, USAF FDL-TR-68-97

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Appendix A Vendors

Table A-1 lists the subsystem and component suppliers that responded to requests for design, cost, and reliability data for use in this study. These suppliers, at no cost to the study, provided one or more types of requested data for the type of subsystem listed by their name.

TABLE A-1

Suppliers of Design, Cost, and Reliability Data

<u>Supplier</u>	<u>Subsystem</u>
Aerojet-General	Propulsion
Airesearch	Power Supply
Allis-Chalmers	Power Supply
Barnes Engineering	Avionics
Bendix Corporation	Environment Control
Collins Radio Company	Avionics
Hamilton Standard	Environment Control
Honeywell, Inc.	Avionics
IBM	Avionics
Leach, Inc.	Avionics
Marquardt	Propulsion
Motorola	Avionics
Pratt and Whitney Aircraft	Power Supply
Pratt and Whitney Aircraft	Propulsion
Rocketdyne	Propulsion
Spacecraft, Inc.	Avionics
Sundstrand Aviation	Power Supply
TRW, Inc.	Avionics
Westinghouse	Avionics

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Appendix B

Saturn V/S-IVB Stage Description

Analysis of the Saturn S-IVB stage cost data required definition and analysis of the design data as well. The following description provides the required technical data on the S-IVB stage.

The S-IVB stage is the third stage of the Saturn V launch vehicle and is approximately 59 feet high and 22 feet in diameter. It is powered by a single Rocketdyne J-2 engine that burns liquid oxygen and liquid hydrogen; useable propellant is 230,000 lbs. After second stage separation, the J-2 engine on the S-IVB stage ignites and propels the payload to the desired altitude. The S-IVB as presently designed has a 4-1/2 hour orbital coast plus a 2 hour translunar coast capability.

Inboard profile and cut away drawings of the stage are shown in Figures B-1 and B-2 respectively. Four subsystem areas are described in more detail in the following text. A summary of the system characteristics is indicated in Table B-1.

STRUCTURE - The forward and aft skirts, the aft interstage and the thrust structure are all made of aluminum using skin and stringer construction. The propellant tank is a milled, internally stiffened cylindrical section with internal insulation, welded gore forward and aft domes, and a common bulkhead of fiberglass honeycomb core bonded to aluminum skins.

PROPULSION - Two solid-propellant ullage rockets are fired at stage separation, just prior to ignition of the J-2 engine, to assure that the propellant is settled in the bottom of the tanks during the start phase. These are Thiokol TX-280 rockets with a thrust of 3400 lbs. each. The main engine is a single Rocketdyne J-2 engine that burns liquid oxygen and liquid hydrogen to provide a thrust of 205,000 lb at an engine mixture ratio of 5/1. The expansion ratio of this engine is 27.5:1. During flight, the main engine is hydraulically gimballed in pitch and yaw to provide thrust vector control in response to commands from the instrument unit. Maximum gimbal angle is 7°. Powered roll control is provided by fixed position hypergolic engines located in two auxiliary propulsion system (APS) modules mounted 180° apart on the aft skirt. Three axis (roll, pitch and yaw)

attitude control during coast is also provided by the APS. The APS modules also contain two small hypergolic engines for ullage control during the flight. These are fired during the first shutdown of the J-2 engine to control the position of the propellants and again, prior to the second J-2 start, to position the propellants during chilldown and restart of the main engine.

The attitude control system is a pulse-modulated on-off system. The system is based on the minimum impulse capability of the 150-lb thrust engine, which has a minimum impulse bit capability of 7.5 lb-sec with an electrical input pulse width of approximately 65 milliseconds. The attitude control system is designed to operate with an attitude dead zone of ± 1 degree in all axes. The undisturbed limit cycle rates are approximately 0.2 deg/sec in roll and 0.001 deg/sec in pitch and yaw (0.003 deg/sec during translunar coast.)

The auxiliary propulsion system is a completely self-contained modular propulsion sub-system. The electrical power requirements are 28 volts at a maximum of 26.5 amp for operating valves and switches.

Each module contains one 72-lb thrust and three 150-lb thrust ablatively-cooled liquid bi-propellant hypergolic engines, a positive expulsion (Teflon bladder) propellant feed system for zero gravity operations and a helium pressurization system. Each contains 119.4 lb of MMH (Monomethylhydrazine) fuel and 192.6 lb of N_2O_4 (nitrogen tetroxide) oxidizer. The nominal oxidizer to fuel mixture ratio is 1.65/1.

ENVIRONMENTAL CONTROL - The forward skirt thermal control systems for cooling electronic equipment differ from those used in the aft skirt. The equipment mounted on the forward skirt is conditioned actively and that mounted aft is conditioned passively.

In the active system, electronic components are mounted on 16 thermal conditioned panels (cold plates) which transfer heat to a coolant (60% menthanol and 40% water) flowing through the panel. The coolant enters the cold plates at 60°F maximum and leaves at 70°F maximum. A coolant flow rate of approximately 0.5 gpm per panel is used. The total allowable heat load per panel is 500 watts.

No cooling is available from the end of the prelaunch phase until approximately 130 sec after lift-off. A prelaunch purge gas system utilizing air and gaseous nitrogen, provides the forward skirt area with a warming medium. It operates only up to the time of launch and provides no thermal control after that time. This system protects the electronic components and reduces oxygen present to 4% by volume. The total flow rate in the forward area is about 275 lb/min. The purge gas surrounding the components located in the I.U. and S-IVB forward skirt will be at a temperature of 35°F to 75°F.

In the passive system, electronic components are mounted in the aft skirt area on 18 fiberglass panels. No fluid thermo-conditioning system is used. Temperature is controlled through the proper surface finish of each electronic package and by providing conduction paths and insulation. Appropriate coatings are added when a special heating or cooling problem is revealed by calculation or test.

A separate prelaunch purge gas system maintains the equipment mounted on the aft skirt at a temperature of 20°F to 70°F during prelaunch procedures. Dry air at a flow of 300 lb per minute to the S-IVB stage is provided from a ground source. Gaseous nitrogen purge of approximately the same flow rate is initiated about 30 minutes before LH₂ loading. During flight, heat is radiated to space and to local sinks such as the LOX tank.

ELECTRICAL POWER - The S-IVB has four independent electrical systems with 56- and 28-volt silver-oxide primary batteries. Forward system #1 (350 ampere hours, 28 vdc) supplies power to the data acquisition system which produces low-level, high-frequency signals that must be isolated from other systems. Forward system #2 (15 ampere hours, 28 vdc) supplies power to systems which cannot tolerate switching transients or high frequency interference, such as the propellant utilization system and inverter-converter. Both batteries for the forward systems are mounted in the forward skirt.

Aft system #1 (270 ampere hours, 28 vdc) supplies power to valves, heaters and relays in the main propulsion engine, pressurization system, stage sequencer, that must be isolated from other systems. Aft system #2 (70 ampere hours, 56 vdc) supplies power for an auxiliary hydraulic pump, LOX chilldown inverter and LH₂ chilldown inverter. Both batteries for aft

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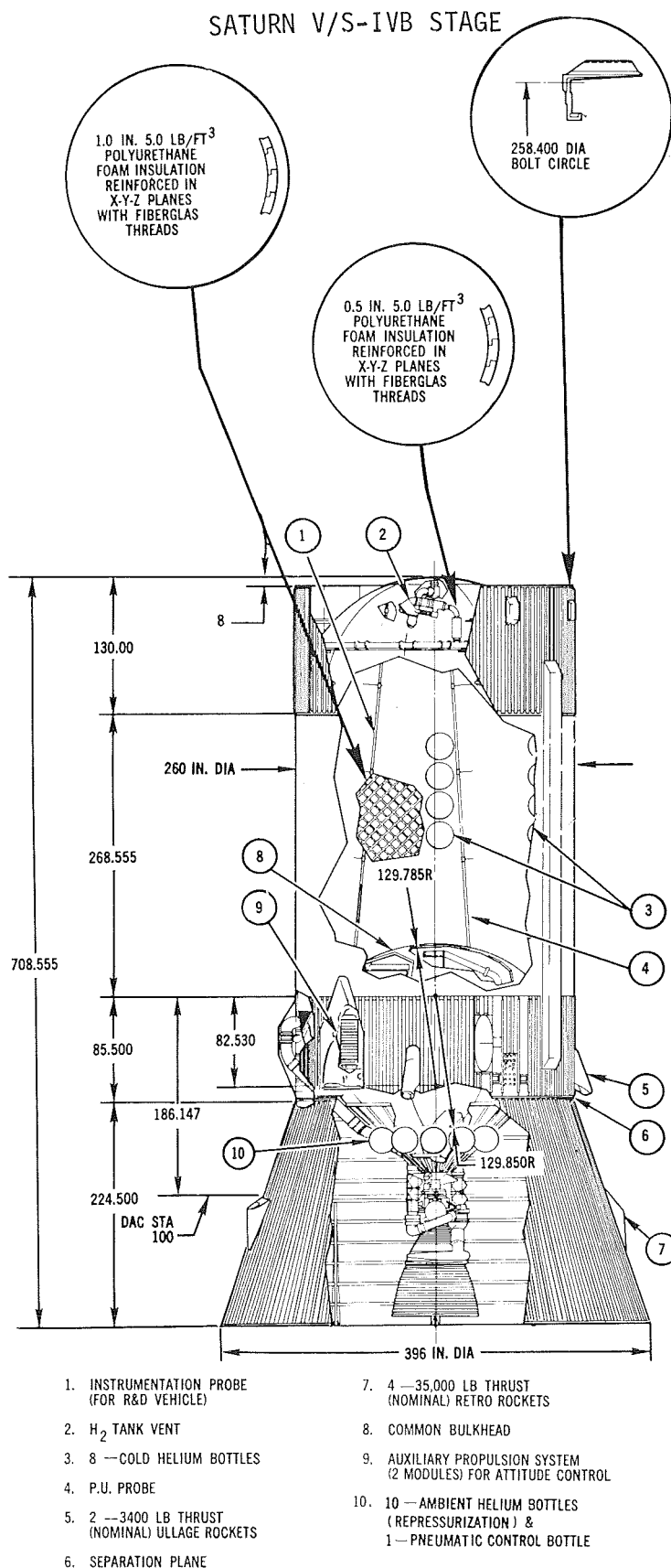
system #1 and #2 are located in the aft skirt. Both the aft and the forward systems are wired through distribution boxes located in their respective areas.

The batteries are sized to handle the stage load requirements for 6-1/2 hours.

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Figure B-1



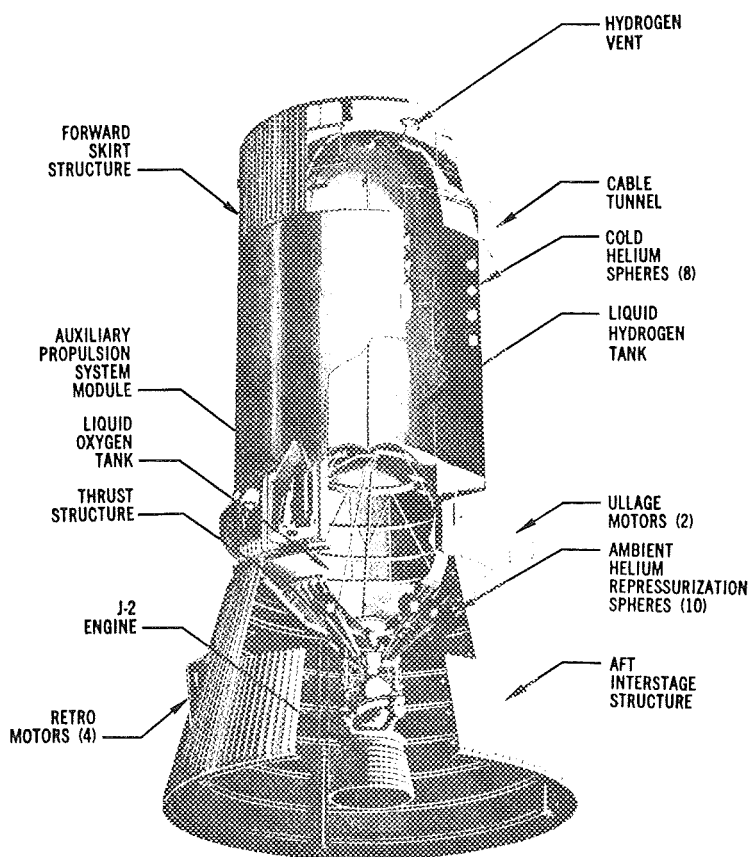
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Figure B-2

SATURN V/S-IVB STAGE

CUT A WAY



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Table B-1

SATURN S-IVB/S-V STAGE DESCRIPTION

I. STRUCTURES

Overall Diameter: 260 inches

Overall Length: 708.6 inches

Usable Main Propellant Weight: 230,000 lbs.

Reliability: 0.9998

Weight: 19,192 lb.

A. Forward Skirt -

Material: 7075-T6 Aluminum

Construction: Skin, Ring and Stringer

Length: 130 inches

Weight: 1515 lb.

B. Propellant Tank Assembly

Material: 2014-T6 Aluminum

Construction: Milled, internally stiffened cylindrical section;
welded domes, common bulkhead,

Length: 268 inches

Weight: 8938 lb.

Volume: LOX tank - 2,828 ft³, LH₂ tank -10,426 ft³

C. Thrust Structure

Material: 7075-T6 Aluminum

Construction: Skin and stringer

Weight: 767 lb.

D. Aft Skirt

Material: 7075-T6 Aluminum

Construction: Skin and stringer

Length: 97 inches

Weight: 1957 lb.

E. Aft Interstage

Material: 7075-T6 Aluminum

Construction: Skin and stringer

Length: 213 inches

Weight: 5700 lb.

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- F. Fairing and Misc. Structure
Weight: 304 lb.

II. MAIN PROPULSION SYSTEM

Weight: 7543 Lbs.

Reliability: 0.9806

A. J-2 Engine (GFE)

Reliability: 0.9862

Weight: 3507 lb. (including accessories)

Vacuum Thrust: 205,000 lb.

Expansion Ratio: 27.5:1

Mixture Ratio: 5:1

Maximum Gimbal Angle: 7°

- B. Oxidizer Fill Drain and Vent System (Plumbing, Disconnects, Valves, and Pressure Switches)

- C. Fuel Fill, Drain and Vent System (Plumbing, Disconnects, Valves, and Pressure Switches)

Maximum Fill Rate: 3000 GPM

Weight: 710 lb.

- D. Pressurization System (10 Ambient Helium Spheres, Valves, Filters, Pressure Switches, etc.)

Weight: 2042 lb. (Dry)

- E. Propellant Settling (2 Solid Propellant Ullage Rocket Motors with EBW Initiators, Firing Units and Jettisoning System)

Thiokol TX-280 SRM's

Thrust: 3,390 lb. (each)

Reliability: 0.9997 (Total)

Weight: 192 lb. (Total)

- F. Retrorockets (4 Solid Propellant Retrorockets, EBW Initiators and Firing Units)

Thiokol TE-29-18 SRM's

Thrust: 35,700 lb. (each)

Weight: 500 lb. (Total)

Reliability: 0.9987 (Total)

- G. Chillydown and Purge

Weight: 300 lb.

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III. AUXILIARY PROPULSION SYSTEM (APS) - 2 Modules

Weight: 888 lb. (Total - Dry)

Reliability: 0.9956 (Total)

- A. Attitude Control Engines (3 APCO, 150 lb. thrust)
- B. Ullage Engine (1 Rocketdyne, 72 lb. Thrust Gemini "OAMS" Engine)
- C. Propellant Control System (Helium Positive Expulsion)
- D. Propellant Tanks (312 lb. of N_2O_4 /MMH Propellant Capacity) - Titanium

IV. PROPELLANT UTILIZATION SYSTEM (Fuel and Low Capacitance Mass Sensors, Electronics and Inverter/Converter)

Weight: 195 lb.

Reliability: 0.9951

V. POWER

- A. Electrical System (Control Switching Network, Sequencer, Silver-Zinc Batteries, Static Inverter, Distribution System Grounding and EBW Firing Systems)
- B. Weight: 981 lb.
Reliability: 0.9916
- B. Environmental Control System (Local Electrical Heaters and Cold Plates for Electrical/Electronic Equipment Conditioning)
Weight: 514 lb.
Reliability: 0.9999

VI. AVIONICS

- A. Flight Control (Main Engine and APS Control Computers are Located in LV)
 - 1. Hydraulic and Pneumatic System (Main Engine Thrust Vector Steering - Hydraulic Pumps, Accumulator Reservoir Assembly, and Hydraulic Actuator Assemblies)
Weight: 467 lb.
Reliability: 0.9966
 - 2. Roll and Coast Control (Supplied by APS)
 - 3. Control System Electronics
Weight: 136 lb.
- B. Telemetry System (Instrumentation, Signal Conditioning and Data Transmission System - FM/FM, PAM/FM/FM, PCM/FM, and SS/FM)
Weight: 2503 lb.
Data Channels: 690 Continuous and Sampled

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VII. SEPARATION SYSTEM (MDF Explosive Harness Assembly, Detonator Block,
EBW Detonator and Firing Unit)

Weight: 274 lb.

Reliability: 1.0

VIII. RANGE SAFETY SYSTEM (Antenna, Range Safety Receivers, Controllers, Firing
Unit, EBW Detonator, S & A Device, Primacord Harness Assembly, Linear
Shaped Charge)

Weight: 81 lbs.

Reliability: 0.9999

TOTALS:

Stage Dry Weight: 32,764 lb. (including interstage)

Stage Gross Weight: 271,600

Stage Reliability: 0.959898

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Appendix C

Gemini Spacecraft Systems Description

Analysis of the Gemini Spacecraft cost data required definition and analysis of the design data as well. The following description provides the required technical data on the Gemini.

1. GENERAL - The Gemini Spacecraft was designed to provide life support for two orbiting astronauts during launch, orbit, entry and landing. Missions with orbit stays times of up to 14 days were accomplished.

The Gemini Spacecraft is a conical structure 18.82 feet long and ten feet in diameter at its base. It is composed of two major assemblies, an entry vehicle and a mission module. The general arrangement is shown in Figures Cl-1 and Cl-2. Spacecraft geometry is shown in Figure Cl-3.

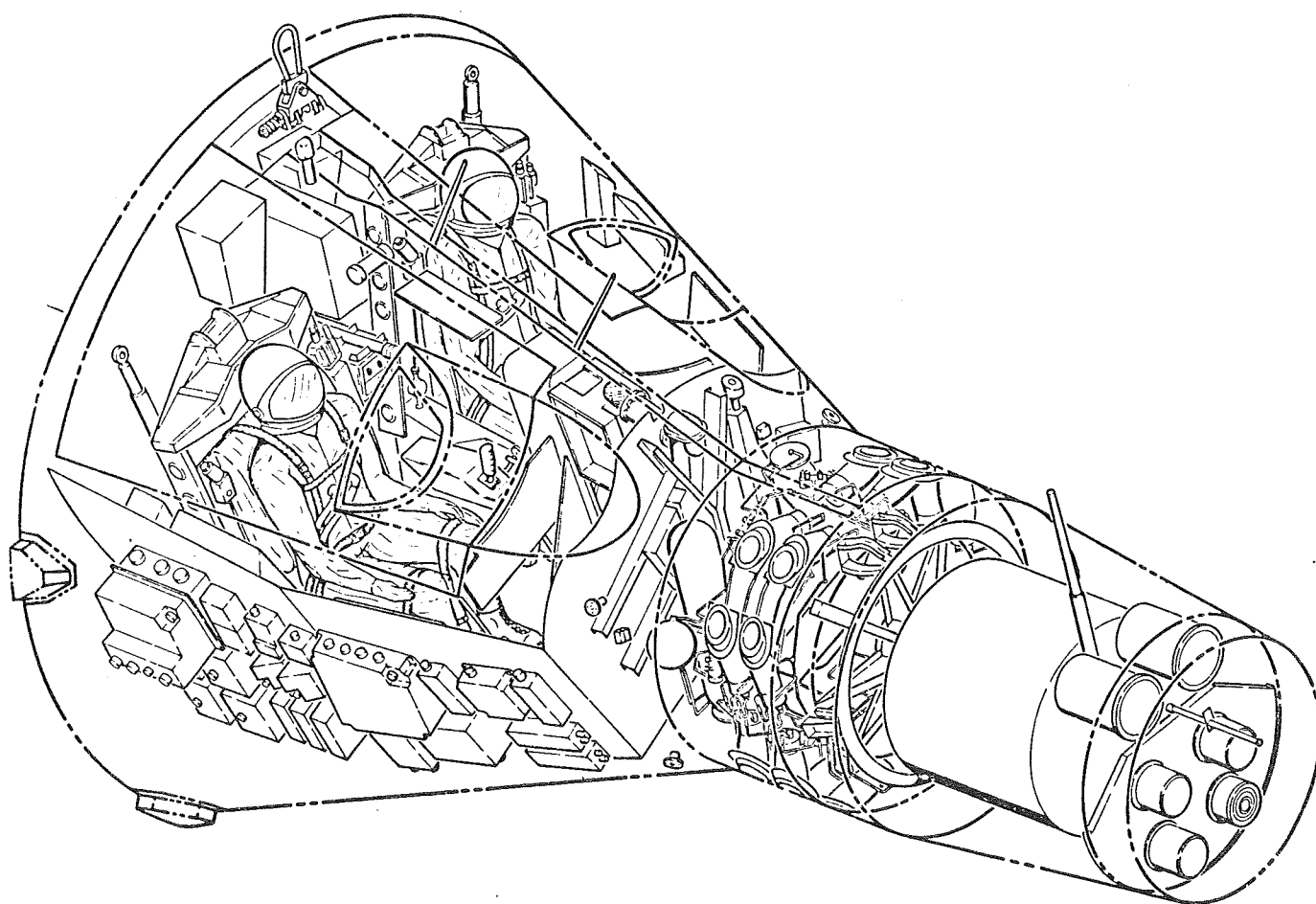
Spacecraft weights varied, depending on the experiments, mission objectives, and mission stay time of the particular spacecraft. The weights of Spacecrafts 7 and 14, which accomplished 14 and 3 day missions respectively, are presented in Table Cl-1.

The reliability apportionments for a typical Gemini Spacecraft are presented in Table Cl-2.

The following sections are organized by major subsystem and cover structure, power sources, communications and instrumentation, guidance and control, propulsion, environmental control system, crew station, and escape, landing and recovery.

Figure C1-1

ENTRY VEHICLE INTERIOR ARRANGEMENT (TYPICAL)

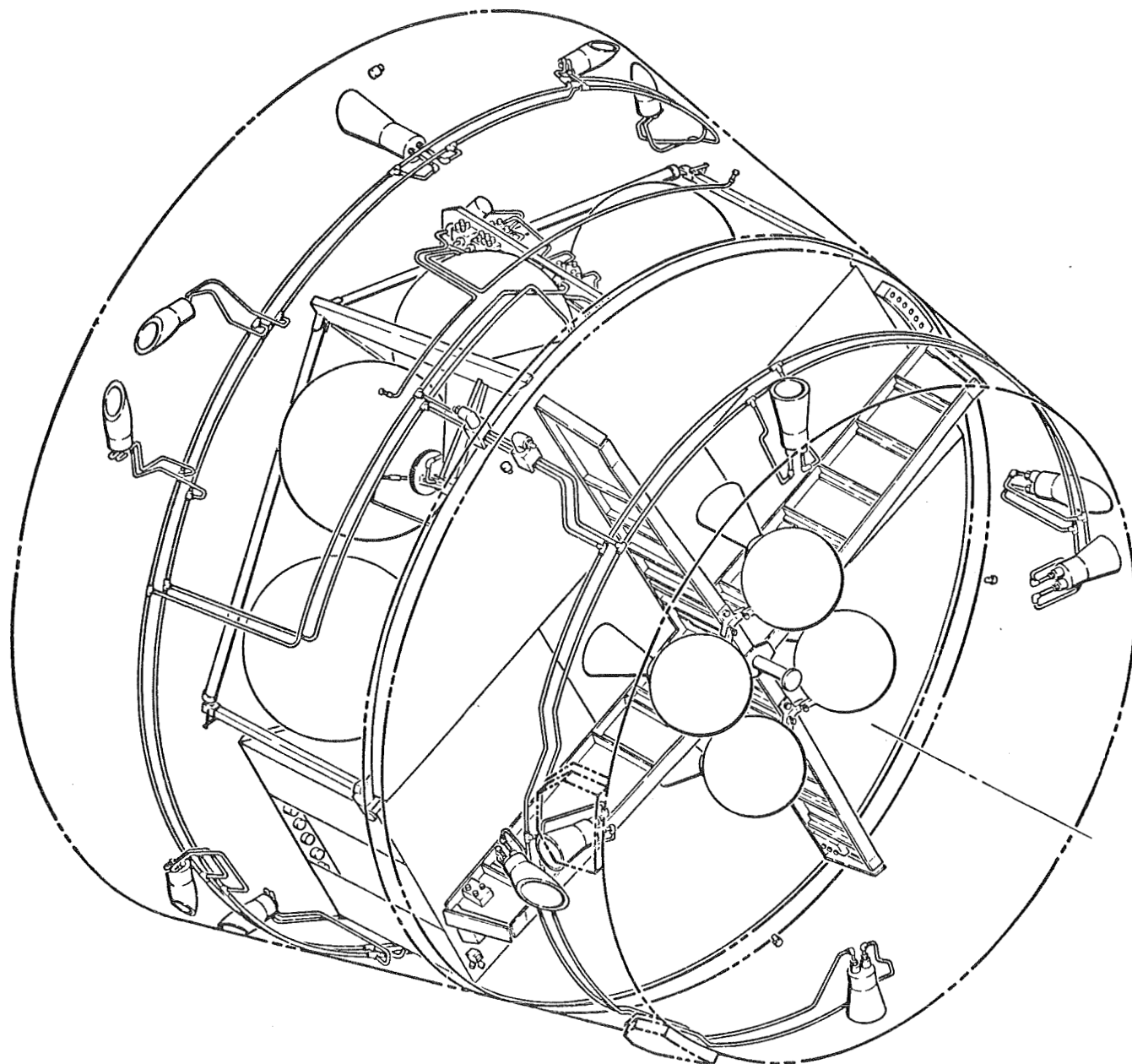


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Figure C1-2

MISSION MODULE INTERIOR
ARRANGEMENT (TYPICAL)

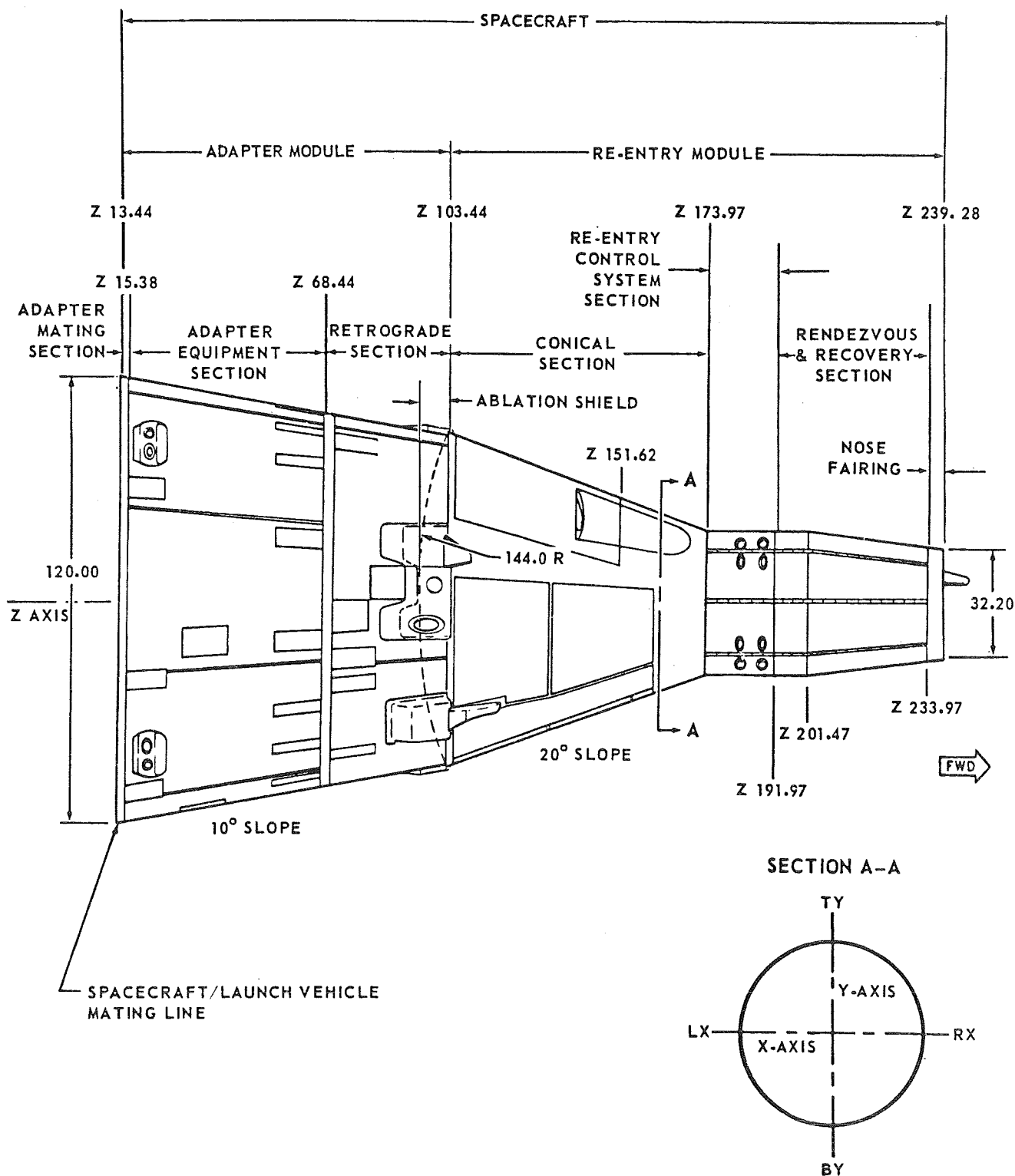


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Figure C1-3

SPACECRAFT GEOMETRY



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Table C1-1

GEMINI WEIGHT SUMMARY

SUBSYSTEMS	3 DAY S/C 12			14 DAY S/C 7		
	CM	M.M.	TOTAL	CM	M.M.	TOTAL
Structure	1063	416	1479	1080	408	1488
Ind. Env. Prot	733	15	748	732	11	743
Docking	33		33	8		8
Landing	211		211	211		211
Retro	7	366	373	7	366	373
Entry Att. Cont.	218		218	218		218
Orbit Att. Cont.		1439	1439		777	777
Prime Power	266	489	755	264	563	827
Guid & Nav.	393	33	426	319		319
Inst. & Tele.	125	82	207	128	81	209
Comm.	134	60	194	134	62	196
ECS.	396	298	694	416	665	1081
Pers & Pers.Prov.	944		944	952		952
Crew Sta. Cont	122		122	124		124
Experiments	166	330	496	118	177	295
Ballast	71		71	89		89
Wt. Adj.	5	30	35	-5	18	13
Total	4887	3558	8445	4795	3128	7923

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Table C1-2

GEMINI RELIABILITY APPORTIONMENTS

(GROSS MISSION SUCCESS)
48 HR ORBIT

Gemini 12
Estimates

1.0	1. Structures
.9965	2. Power Sources
.9999	3. Communications & Tracking a. digital command system
.9998	4. Instrumentation
.9964	5. Guidance and Control
.9845	6. Propulsion
.9923	7. ECS
N/A	8. Crew Station
.9943	9. Landing, & Recovery
.9241	10. ESCAPE (both seats) (Back-up to S/C landing system)

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2. STRUCTURE - The spacecraft as previously noted, is composed of two sections, the entry vehicle and a mission module. These sections are separated before deorbit and only the entry vehicle, which houses the astronauts, must be designed to withstand the entry environment.

2.1 ENTRY MODULE - The entry module is an all metal, stressed skin, semi-monocoque structural body. Stringers and longerons spaced circumferentially around the shell, aid in carrying bending and axial loads and help stiffen the shells. The entry module is constructed in three sections and then structurally spliced. These three primary sections are the cabin, the re-entry control system (RCS) section, and the rendezvous and recovery (R&R) section. A nose fairing is fitted to the forward end and an ablative heat shield is attached to the aft end of the cabin section. The entry module structure showing these major subsections is presented in Figure C2-1.

2.1.1 CABIN SECTION - The cabin or conical section consists of a pressurized inner body, personnel access hatches, windows, equipment bays, insulation, shingles, and the ablation shield.

The primary load paths consist of stringers to carry the bending and axial loads and the inner mold line skin, side panels, and floor to carry the shear and local pressures. This inner body structure employs titanium as the material of construction. Welding is the primary method of attachment to reduce the possibility of leakage. The pressure vessel is designed to withstand an ultimate burst pressure of 12 psi and an ultimate collapse pressure of three psi. The collapse pressure requirements are imposed to provide flotation capability after a water landing.

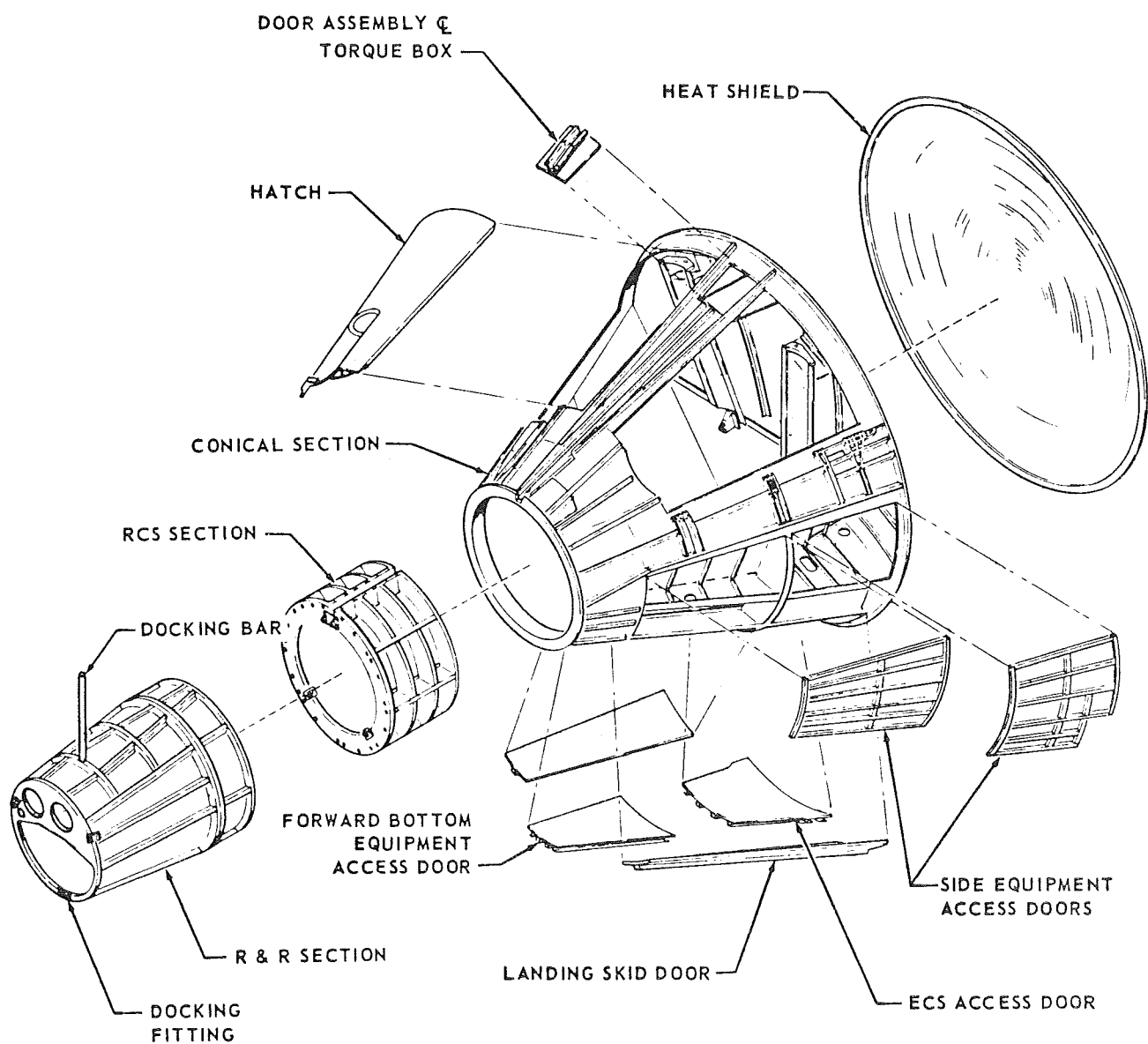
Outer mold line panels are designed carry local air loads only and do not contribute to the primary load carry capability. These radiative heat protection panels are constructed of super alloys. Rene' 41 and L-605 are the materials most commonly used.

Insulation materials used to impede heat transfer to the inner body are Min-K, fiberglass and Thermoflex RF.

Two structural hatches are located symmetrically on the top side of the pressure vessel to allow ingress and egress. The hatches may be manually opened by means of a handle; in an emergency pyrotechnic charges

Figure C2-1

RE-ENTRY MODULE STRUCTURE



are used. Each hatch incorporates a window which contains 3 panes of glass separated by air spaces.

Three major equipment bays are located outside the cabin pressure vessel. Most system components are located in these bays. Mold line doors allow quick access to these compartments.

2.1.2 RE-ENTRY CONTROL SYSTEM SECTION - The RCS section extends from the Conical Section splice at Z173.97 to the R&R Section separation joint at Z191.97. This section houses the Reaction Control and also is the primary support structure for loads induced by deployment of the recovery parachute.

The basic structure is of titanium sheet with longitudinal stiffeners. There are four external titanium rings. The mold line panels are Beryllium shingles that vary in thickness from 0.28 to 0.09 inches. These shingles are attached to the structure by Beryllium retainers that allow thermal expansion.

Between the mold line shingles and the primary structure thermoflex RF series blanket insulation is installed.

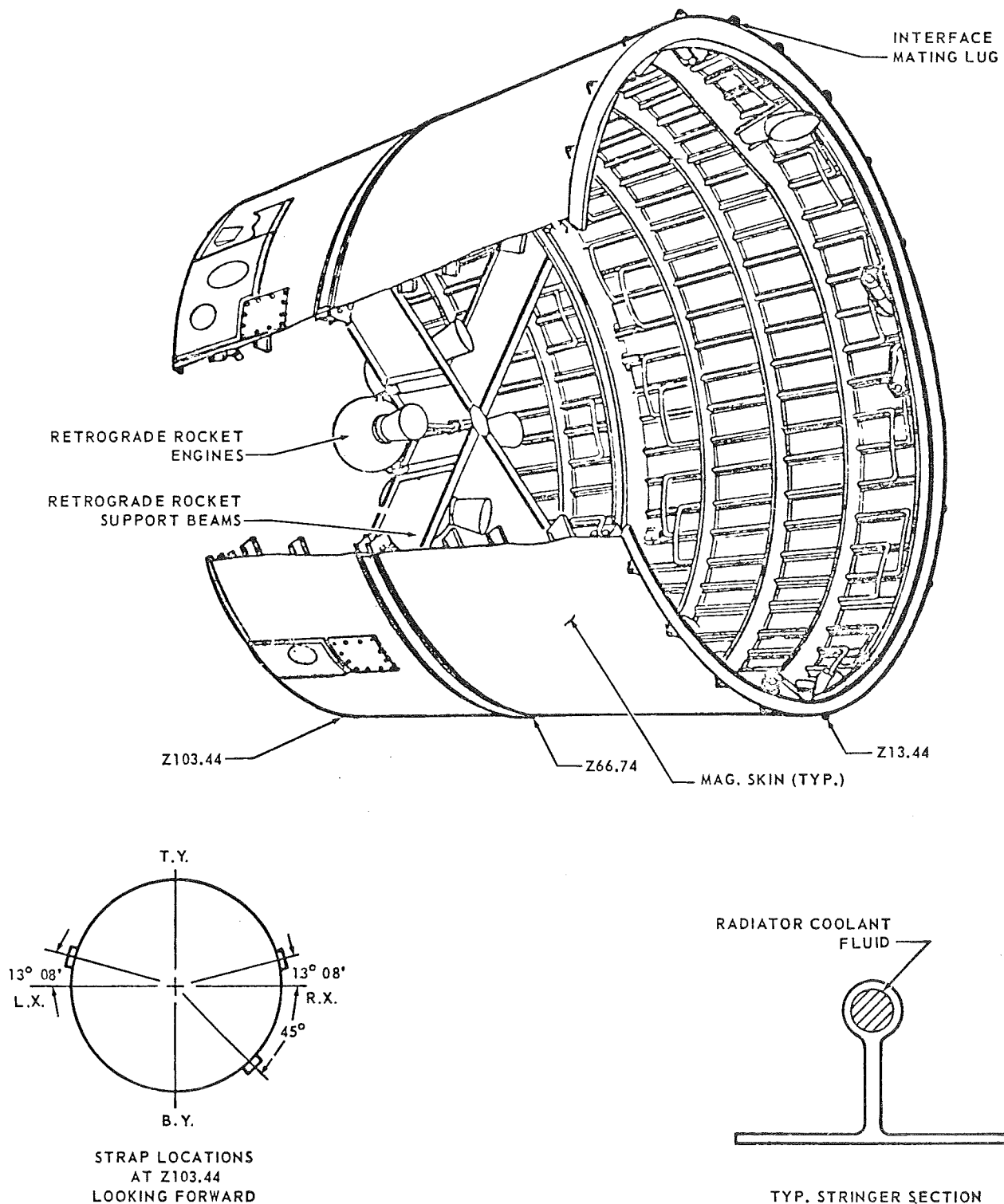
2.1.3 RENDEZVOUS AND RECOVERY SECTION - The R&R Section extends from the separation joint at Z191.97 to the nose. The structural concepts, thermal protection techniques, and materials are the same as those employed in the RCS section.

2.2 MISSION MODULE - The Mission Module is a truncated cone 90 inches long that contains the systems and equipment needed for long duration orbital flights. It also serves as an adapter between the entry spacecraft and the launch vehicle and as a radiator for the spacecraft coolant system. For functional purposes the module is divided into two sections, the retrograde section and the adapter equipment section. The mission module structure is shown in Figure C2-2.

The retrograde and adapter equipment sections are both of semi-monocoque type construction. The stringers are of extruded magnesium, and are welded to rolled magnesium sheets. The inboard flange of the extrusions is a hollow bulb that interconnects between stringers. A coolant fluid is passed through these bulbs and utilizes the adapter surface as a space

Figure C2-2

MISSION MODULE STRUCTURE



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radiator. Aluminum rings are spaced at regular intervals to stabilize the stringers and to support the adapter equipment.

The retrograde section contains the retrograde rocket motors and their supporting structure. The rocket motors are supported by two perpendicular cross beams that transmit loads to the adapter shell.

The adapter equipment section houses the various equipment modules. Most of this equipment is supported by aluminum and fiberglass honeycomb panels at Z66.74. These panels plus an access door combine to form a blast shield that protects the equipment and booster from retrograde rocket engine blast if these rockets are used during launch abort.

3. POWER SOURCES - The basic electrical power in the Gemini spacecraft is supplied by fuel cells for the ascent and orbital phases and batteries for the entry, landing and post-landing phases. Power supply is through a multiple bus at 22 to 30 VDC.

3.1 FUEL CELL SYSTEM - The fuel cell system is made up of two fuel cells and a reactant supply system (RSS). The Fuel Cell System for Gemini Spacecraft 7 is shown in Figure C3-1. A schematic of the system is shown in Figure C3-2.

3.1.1 FUEL CELLS - The two sections have a combined output of 2.0 KW. Each section consists of three stacks in parallel, electrically. Each section is cylindrical, with a length of approximately 24" and a diameter of 12".

3.1.2 REACTANT SUPPLY SYSTEM - The fuel cell reactants, oxygen and hydrogen, are stored in double wall, vacuum insulated, spherical containers. Several tank sizes were used depending on the mission duration. The reactants are stored at supercritical pressures and cryogenic temperatures.

Reactant	Pressure PSIA	Temperature °F
Oxygen	14.7	-297
Hydrogen	14.7	-423

3.2 BATTERIES - Power during entry was supplied by silver-oxide zinc batteries. Four 45 amp/hour and three 15 amp/hour squib batteries provide the necessary power. All batteries were activated and sealed at sea level pressure.

RSS/FUEL CELL MODULE (S/C 7)

Figure C3-1

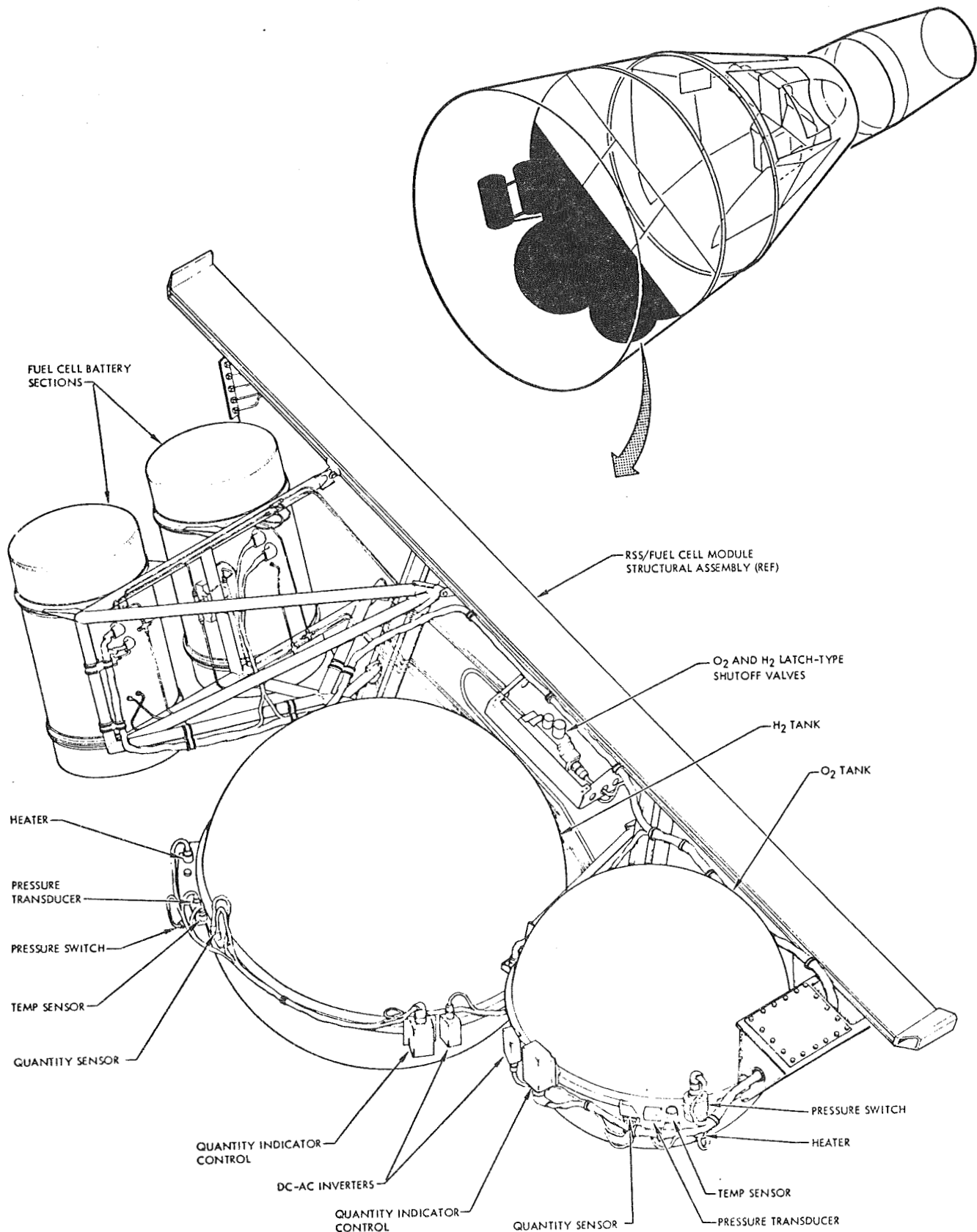
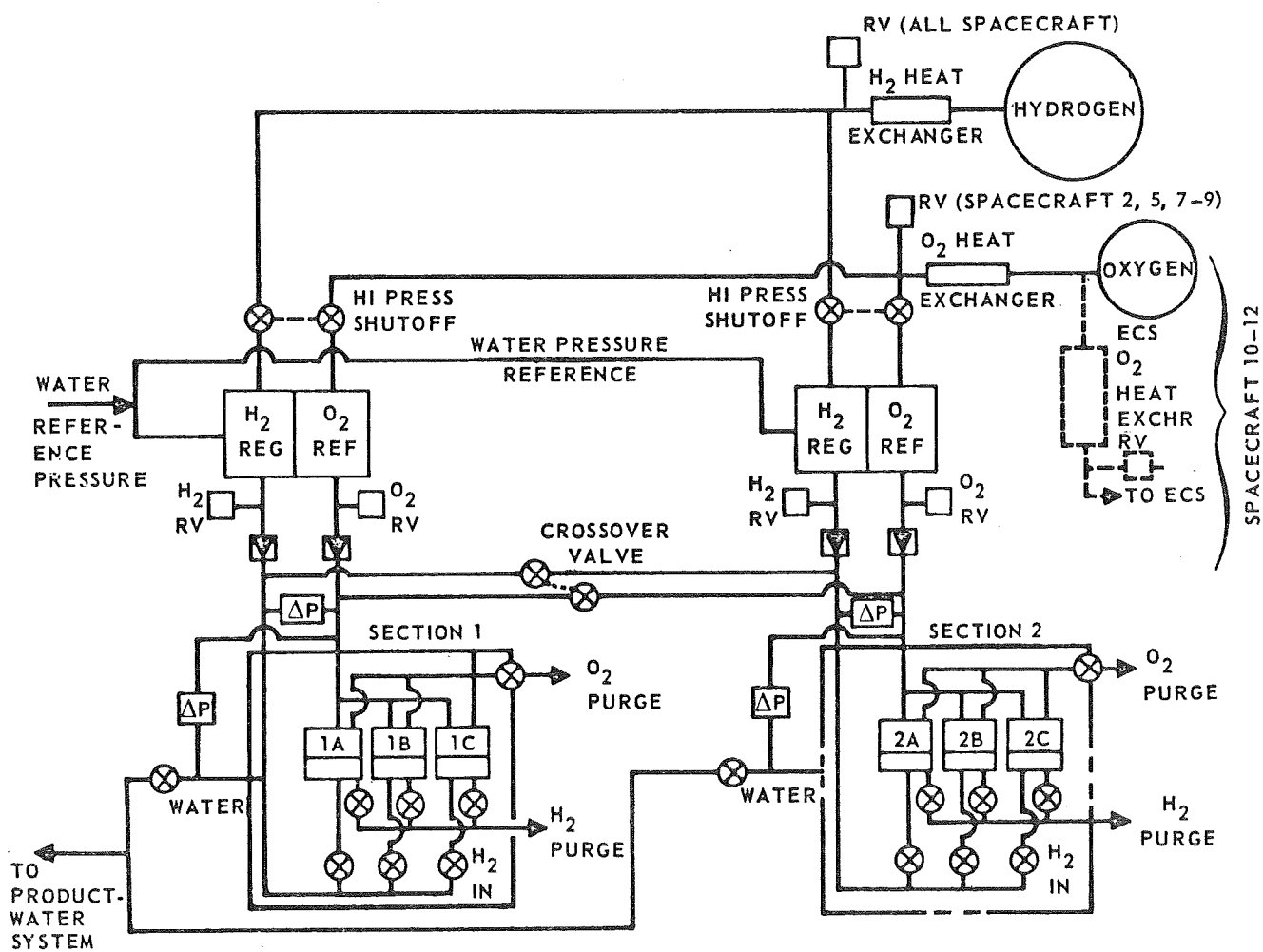


Figure C3-2

FUEL CELL RSS SCHEMATIC



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4. COMMUNICATIONS AND INSTRUMENTATION - The Gemini telecommunications system provides the intelligence link between the ground and the Gemini spacecraft. It provides the capabilities for (a) radar tracking of the spacecraft, (b) two-way voice communications between the ground and the spacecraft and from astronaut to astronaut, (c) ground to spacecraft data or command link, (d) spacecraft to ground data link and (e) post-landing recovery aids.

The equipment used to provide the capability of communications and instrumentation is shown in the block diagrams, Figures C4-1 and C4-2. A description of the equipment for these two subsystems follows:

4.1 COMMUNICATION SUBSYSTEM EQUIPMENT - The communications equipment includes all RF equipment plus the audio portion of the voice, tracking, command and recovery equipment. The communications system provides for the RF link between the spacecraft and the ground stations.

The voice control center provides processing, switching, and routing of audio signals transmitted within the vehicle and between the vehicle and ground. It is a single unit containing microphone amplifiers, headset amplifiers, squelch controls, volume controls, voice level operated switches, and transmission mode (VHF or HF) select switches.

The VHF voice transceivers provide the normal RF link for voice transmission between the vehicle and the ground stations. The HF voice transceiver provides a backup voice link while in orbit and an over-the-horizon RF voice link for emergency landings. Both VHF and HF transceivers contain fixed tuned AM transmitters, superheterodyne receivers, power supply, modulator, audio amplifiers and squelch circuitry.

The C-Band radar beacon (transponder) enhances the ground tracking signals and provides positive spacecraft (or target) identification for the ground tracking network. The ground tracking signals are received on-board, shifted in frequency, by a preset amount, amplified and retransmitted to the ground network. Both position and velocity of the spacecraft is determined at the ground network. The C-Band beacon consists of a superheterodyne receiver, decoder, transmitter, power supply and regulator, modulator and circulator.

Figure C4-1

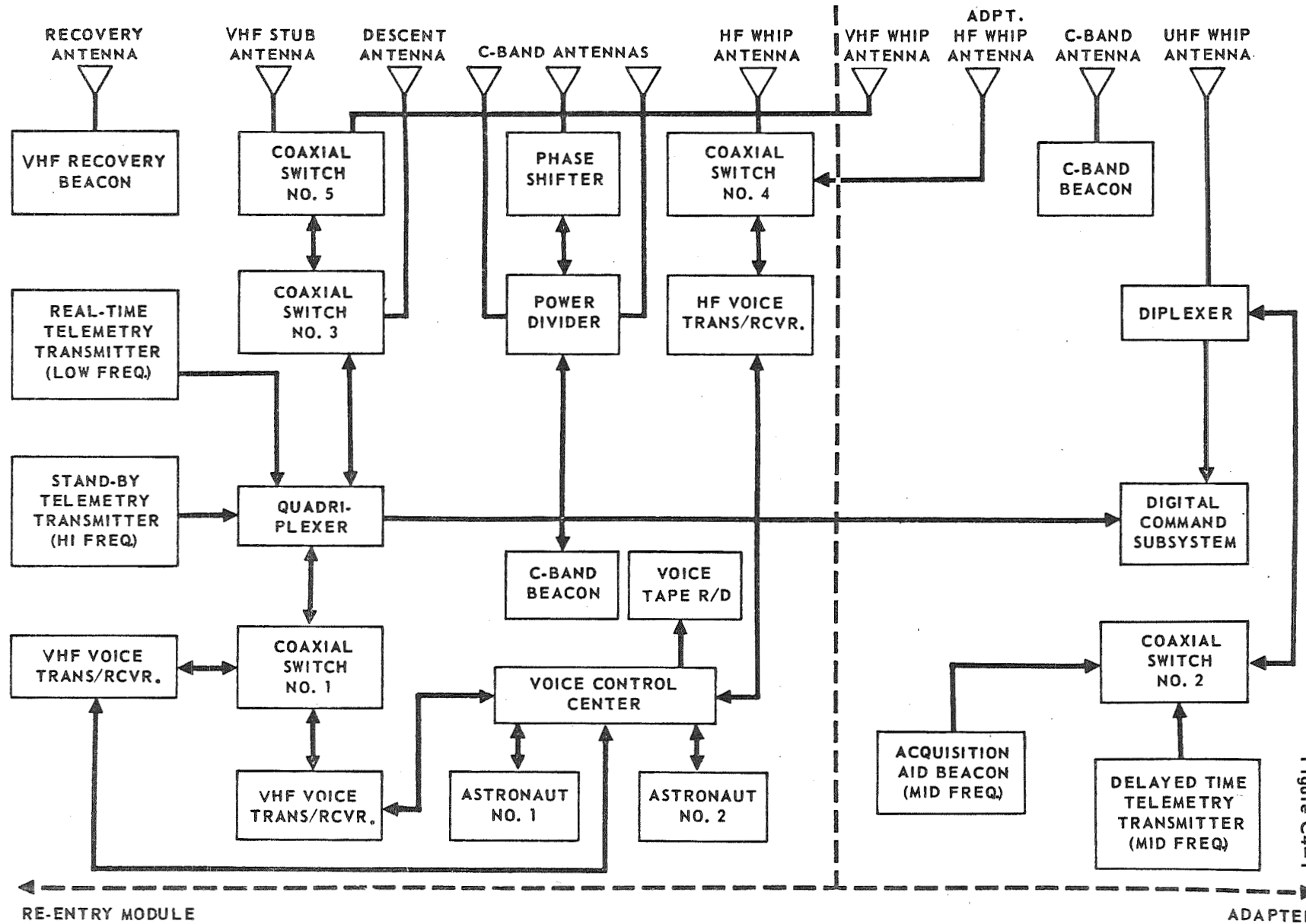
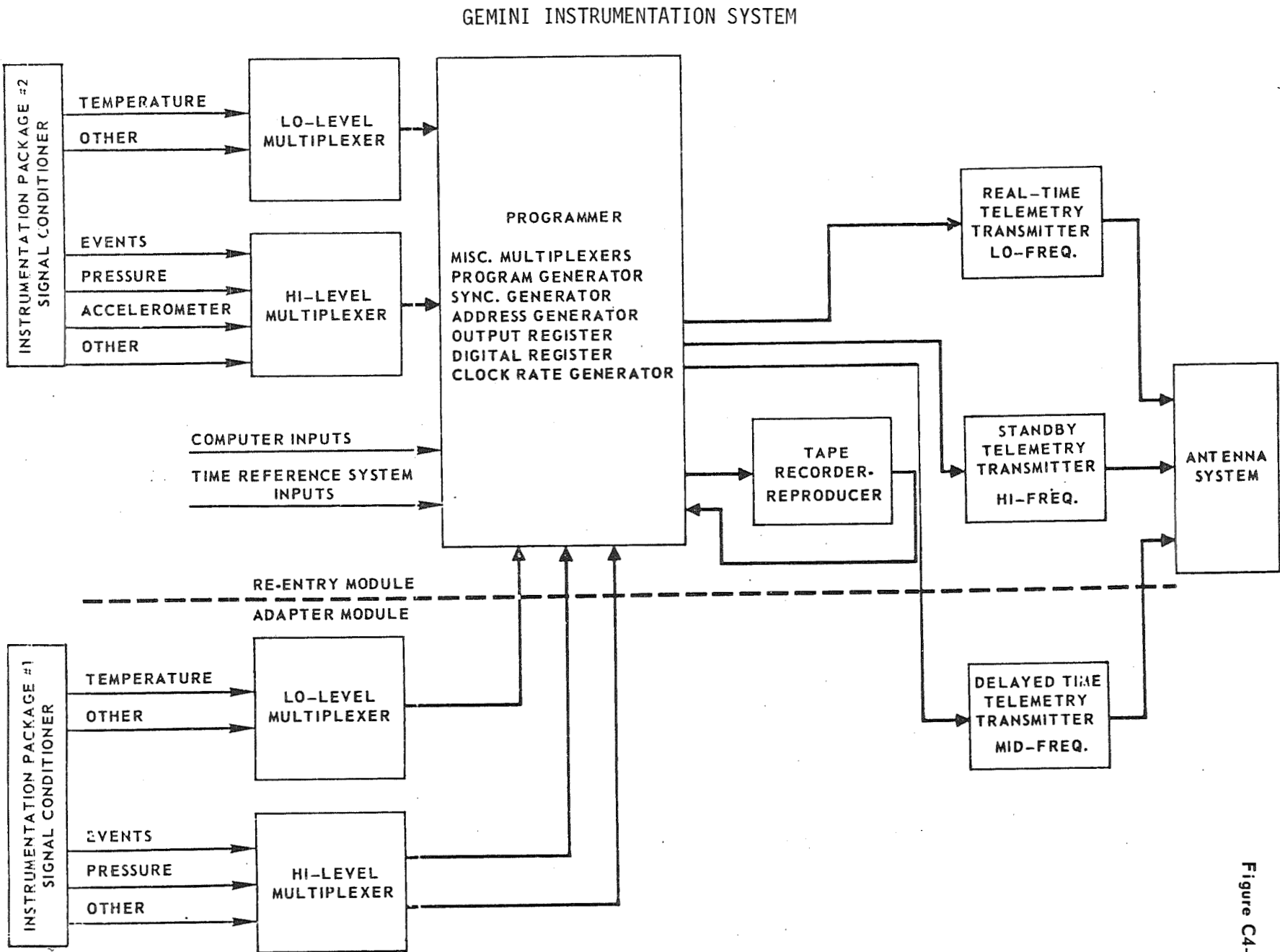


Figure C4-2



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The digital command subsystem (DCS) receives and decodes the uplink or command signals transmitted from the ground stations. The decoded signals provide commands for control of spacecraft systems and provide navigation updates to the spacecraft guidance computer. The digital command subsystem consists of RF receiving equipment, a detector, a decoder, and a buffer storage unit. The receiver equipment contains two identical UHF superheterodyne, dual conversion receivers which are operated actively redundant. The detector extracts the system synchronization and data from the audio output of the receivers. The decoder recognizes sub-bit coding (five sub-bits per bit for redundancy) in data received from the detector and processes the 30 bit message. Data is sent to other systems from buffer storage, upon a request from that system, or upon activation of relays (32 max.) located within the DCS.

The VHF recovery beacon serves as a spacecraft direction finding aid for recovery craft during the Earth descent and post landing mission phase. The beacon consists of a crystal controlled transmitter, power amplifier, and power supply.

4.2 INSTRUMENTATION SUBSYSTEM EQUIPMENT - The Instrumentation subsystem provides the means for monitoring the performance and operation of the spacecraft through the entire mission. The subsystem modifies the performance and operation signals, when necessary, to usable levels, records certain data for delayed transmission, provides signals for real-time transmission to the ground, and monitors certain physiological characteristics of the astronauts. The instrumentation system is composed of six subsystems which are (a) sensors, (b) signal conditioners, (c) multiplexers/encoder, (d) tape recorder/reproducer, (e) physiological data acquisition system and (f) telemetry transmission system.

A variety of sensors including pressure, temperature, acceleration, vibration, and CO₂ partial pressures are monitored by the multiplexers/encoder. Also, the sensor outputs are displayed on cabin indicators when direct astronaut readout is desired. Signal conditioning when required is provided by two signal conditioning packages. The multiplexer/encoder equipment samples and converts the various analog and digital signals into a serial pulse code modulated (PCM) signal which is supplied to the real

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time telemetry transmitter and tape recorder/reproducer. The multiplexer/encoder equipment is composed of four multiplexers (five max.) and a PCM programmer. The programmer provides timing signals for the multiplexers, and performs 8 bit PCM encoding plus interleaving of serial digital data from the computer and time reference system into the 51.2K bit PCM bit stream.

The tape recorder and reproducer provides means of storing data during periods when the spacecraft is beyond range of the ground stations. The capacity of the tape recorder permits 4.5 hours of recording. The recorded parameters are transmitted to Earth via the delayed time telemetry transmitters and used to verify spacecraft conditions when the spacecraft is not within range of a ground station.

The physiological data acquisition subsystem monitors the physiological condition of the flight crew. It consists of sensors, signal conditioners, a biomedical power supply, and two special tape recorders. Electrocardiograms, heart sounds, blood pressure, respiration rate and volume, and oral temperatures are measured. The resultant data is applied to the multiplexer/encoder subsystem.

5. GUIDANCE AND CONTROL -

5.1 SYSTEM FUNCTIONAL DESCRIPTION - The Guidance, Navigation, and Control (G, N and C) system of the NASA Gemini spacecraft is responsible for trajectory management and spacecraft orientation during rendezvous and entry. In addition, the guidance and navigation equipment serves as a back-up to the Titan II launch vehicle guidance system during ascent by providing inertially derived steering signals to the launch vehicle control system.

5.2 SYSTEM EQUIPMENT (Ref. Fig. C5-1)

5.2.1 The G, N, and C system consists of the following main sub-system:

- Inertial Guidance System
 - Inertial Measurement Unit (IMU) - platform, electronics, and power supply
 - Digital Computer
- Flight Control System
 - Attitude Control Electronics (ACE)
 - Power Inverter
 - Rate Gyro Packages (2)
 - Orbital Attitude and Maneuvering Electronics (OAME)
 - Pilot Controls - attitude land controller, maneuver hand controller (2)
- Sensors
 - Horizon Sensors (2)
 - Rendezvous Radar
- Time Reference System
 - Electronic Timer
 - Event Timer
 - Mechanical Clock
- Instruments
 - Manual Data Insert and Read-out Units (MDW, MDRU)
 - Incremental Velocity Indicator (IVI)
 - Attitude Display Group (ADG)
 - Range and Range Rate Indicator

[illegible]

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5.2.2 Salient features of the basic units of the G, N, and C system are as follows:

Inertial Measurement Unit

- Platform - 3 single-degree-of-freedom floated gyros and 3 partially floated pendulous accelerometers mounted in orthogonal triodes on a stable element. This element is suspended in a 4 gimbal mounting using gimbal synchors for attitude display and phase shift resolves for digital conversion within the computer. The gimbals are driven by direct drive pancake type torque motors.
- Electronics - 3600 pps pulse rebalanced accelerometer loops, gimbal control electronics, and attitude and acceleration loop malfunction detection ascent.
- Power Supply - 3 sections, separate pulse width regulations.
 - IMV DC section
 - Computer DC section
 - 26v AC, 400 cps sections (2) - for reference voltage and for backup to the ACME power inverter.

Digital Guidance Computer - This unit is a serial, binary, general purpose computer with an electrically alterable memory. The computer interfaces with the IMV, radar, digital command system (DCS), telemetry system, flight control system, attitude display instruments, electronic timer, launch vehicle autopilot, MDIV and IVI, and with the Auxiliary Life Memory unit (ATM) for in-flight re-programming.

- Memory capacity - 4096 words (39 bits/word)
- Data work - 26 bits
- Instruction words - 13 bits
- Add time - 140 microseconds
- Multiply time - 420 microseconds
- Divide time - 840 microseconds
- Discretes - 40 inputs, 30 outputs
- Incremental channels - 3 inputs, 3 outputs

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- Digital data - 5 inputs, 4 outputs
- Analog channels - 3 inputs, 6 outputs

Flight Control System - accepts signals representing attitude errors and rate commands from various sources (hand controller, computer, IMU, and horizon sensor), combines the signals with attitude rate signals from the rate gyros, and converts these signals into on-off logic commands to the Re-entry Control System and Orbit Maneuver System and Orbit Maneuver System solenoids.

- Max. command rates
 - pitch and yaw - 10 deg./sec.
 - roll - 15 deg./sec.
- Attitude deadband - ± 1.1 to ± 5 degrees.
- Rate deadband - ± 0.2 to ± 4.0 deg./sec.

Horizon Sensor - operates in the 8 to 18 micron IR band; scans the horizon in azimuth to produce pitch and roll attitude error signals that indicate the attitude of the spacecraft relative to local vertical, to within $\pm 0.3^\circ$ (3.0).

- Instrument accuracy 0.1 deg.
- Scale factor $\pm 10\%$ from -14 to $+14$ deg.
- Attitude range -20 to $+20$ degrees

Rendezvous Radar - operates in the L-band with a cooperative transponder in the target vehicle. Provides digital range and low-sight angle data to the computer, and analog range and range rate to the crew display unit.

- Digital range 500 ft. to 250 NM
- Range rate -100 to $+500$ ft./sec.
- Angle ± 25 degrees
- PRF 250 PPS

5.3 Mission usage of G, N, and C system:

- Ascent - The system provides backup steering signals to the launch vehicle from lift-off through second stage cut-off. Insertion velocity accuracy is approximately ± 30 ft./sec. in plane and ± 130 ft./sec. out of plane, but the latter error is corrected to ± 30 ft./sec. by

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radio update based on ground tracking during launch.

- Rendezvous - During the catch-up and gross rendezvous phase of the mission (i.e. until the trailing displacement is less than 250 nm.), spacecraft maneuvering is predicted on ground-based guidance up-dates. When the intervehicular range becomes less than 250 nm the radar is used to provide relative velocity, range, and any data to the guidance computer. The computer then used this data, together with platform gimbal angle data, to compute the velocity changes required for rendezvous within one orbit. Vernier connections are computed and applied during this orbit, with final rendezvous maneuver, based on visual observations of the target and range and range rate information from the radar. Docking is completed without further aid of guidance equipment.
- Entry - The inertial guidance system is used during retrograde and entry to measure and compensate for thrust errors at retrograde and compensate for thrust errors at retrograde and dynamic errors (ex., i.g. and atmospheric uncertainties) during entry. The precision of the various guidance and navigation system components, is largely predicted on the requirement for minimal measurement uncertainties during this phase of the mission. The goal of ± 7 to 10 nm (30 at recovery system deployment was met on those spacecraft flights in which the fully automatic steering capability of the G, N, and C system was used during entry.

6. PROPULSION SYSTEMS - A Spacecraft Propulsion System is installed in the Gemini to provide the reaction forces necessary for translation maneuvers and control of spacecraft attitude. The Spacecraft Propulsion System is comprised of three separate and independent propulsion systems.

- (a) A liquid propellant Orbital Attitude and Maneuver System (OAMS)
- (b) A liquid propellant Re-Entry Control System (RCS)
- (c) A solid propellant Retrograde System

A more detailed description of these propulsion systems is given in the following paragraphs.

6.1 ORBIT ATTITUDE AND MANEUVERING SYSTEMS (OAMS) - The OAMS controls the spacecraft attitude and provides translation maneuver capability from the time of launch vehicle separation until the initiation of the retrograde phase of the mission. This combination of attitude control (3 axis) and translation maneuvering (6 directions) provides the spacecraft with an orbital rendezvous and docking capability.

The OAMS is a helium pressure fed system utilizing nitrogen tetroxide (N_2O_4) and monomethylhydrazine (MMH) propellants which are stored in positive expulsion tanks. Thrust is provided by 16 ablatively cooled engines buried at various points about the adapter. Major system components are shown schematically in Figure C6-1; an installation isometric is presented in Figure C6-2. Summary descriptions of the components are presented in Table C6-1.

6.2 RE-ENTRY ATTITUDE CONTROL SYSTEM (RCS) - The RCS controls the spacecraft attitude during the retrograde maneuver and during the subsequent re-entry phase of the mission. This function is critical with regard to crew safety; hence two separate but identical systems (or rings) are provided, either one of which is capable of providing the necessary control torques.

The RCS is a nitrogen pressure fed system utilizing nitrogen tetroxide (N_2O_4) and monomethyl hydrazine (MMH) propellants which are stored in positive expulsion tanks. In each system, thrust is produced by 8, ablatively cooled engines buried at various points about the forward end of

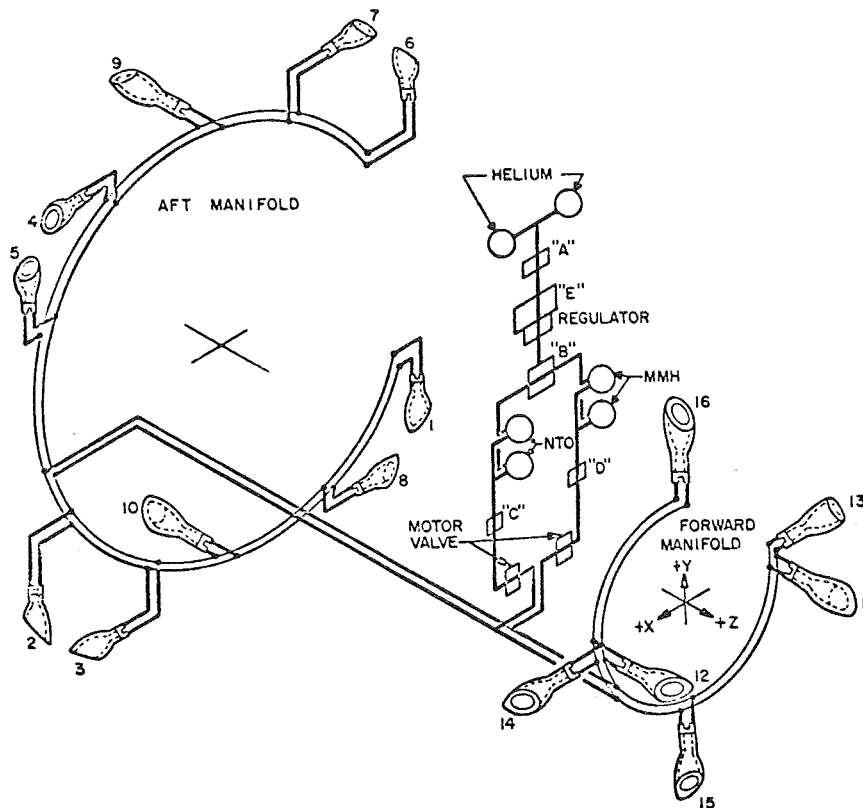
the entry module. Major system components are shown schematically in Figure C6-3; an installation isometric is given in Figure C6-4. Summary descriptions of the major components are presented in Table C6-2.

6.3 RETROGRADE PROPULSION SYSTEMS - The retrograde system provides the incremental velocity change needed to de-orbit the spacecraft. This system is also used to remove the crew module from the launch vehicle in the event of an abort during the final portion of the launch trajectory. (Low altitude escape capability is provided by ejection seats.)

The retrograde systems is comprised of four solid propellant motors, any three of which are sufficient to accomplish the retrograde maneuver. The motors are series or ripple fired in 5.5 second intervals; each motor delivers an average thrust of 2584 lbs. Each motor has dual igniters and consists of a propellant grain which is contained as a spherical titanium case, a graphite throat insert and an ablative nozzle. A motor cut-away is shown in Figure C6-5, and an installation isometric in Figure C6-6.

Figure C6-1

SCHEMATIC - GEMINI ORBITAL ATTITUDE AND MANEUVERING SYSTEM



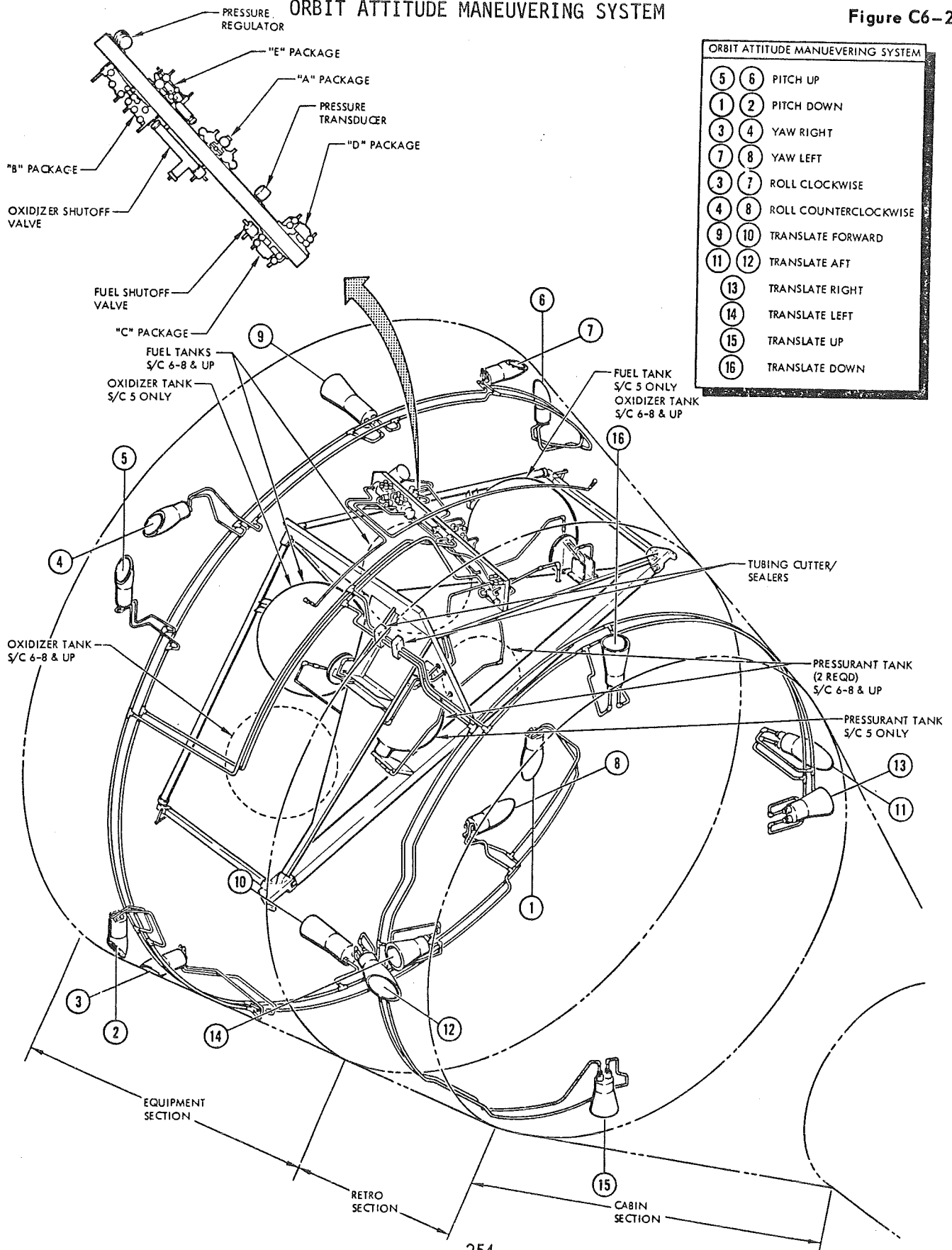
23 lb thrust engines (1-8)
85 lb thrust engines (11, 12)
94.5 lb Thrust engines (9, 10, 13-16)

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ORBIT ATTITUDE MANEUVERING SYSTEM

Figure C6-2



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Table C6-1

SUMMARY DESCRIPTION - OAMS COMPONENTS

1. Pressurant Group

Pressurant Tank - Helium storage; a 3000 psia titanium sphere having a 16.2 inch O.D. and an internal volume of 1696 cubic inches.

"A" Package* - Pressurant fuel and isolation; consists of a normally closed pyrotechnic valve, charging and ground test connections, filter and source pressure transducer.

"B" Package* - Pressurant relief and reverse flow control; consists of burst diaphragm/relief valves, check valves, regulated pressure transducers, filter and ground test connections.

"E" Package - Backup pressure regulation; consists of a normally open and a normally closed pyrotechnic valve, manual by-pass valve, pressure switch, a normally closed, two-way solenoid valve and filter.

Pressure Regulator* - Pressure regulation; single stage unit with redundant bellows, contains flow limiter valve.

2. Propellant Group

Oxidizer Tank - Oxidizer storage/expulsion; consists of a spherical titanium shell, teflon bladder and perforated expulsion tube. There are two tank sizes, one 20.12 inch O.D. with a 4130 cubic inch propellant volume, the other 21.96 inch O.D. with a 5355 cubic inch volume. Operating pressure, 295 psia.

Fuel Tank - Fuel storage/expulsion; same as oxidizer tanks.

"C" Package* - Oxidizer isolation and fill; consists of normally closed pyrotechnic valve, fill and ground test connections and filter.

"D" Package* - Fuel isolation and fill; same as "C" package.

Motor Valve - Backup propellant isolation; and electric motor driven, poppet valve. The motor drives a jackscrow, through a three stage, planetary gear train, which in turn translates the poppet.

* This component also used in re-entry control system.

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Table C6-1 (Continued)

3. Engine Group

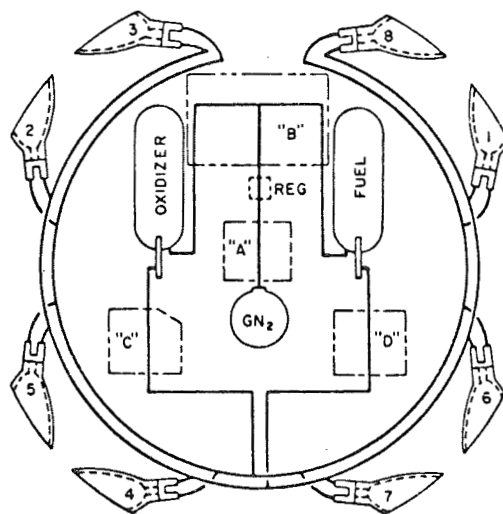
The engine group is composed of eight, 23-lbs thrust engines, two 79-lb thrust engines and six, 94-lb thrust engines. The 23-lb thrust units are used for attitude control; the 79-and 94-lb thrust units for translation. The engines are all of the same basic design, consisting of our ablative chamber/nozzle, a graphite chamber liner, a tungsten throat insert, a stainless steel injection and two, fast response solenoid valves - one in the fuel line and one in the oxidizer line.

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Figure C6-3

SCHEMATIC - GEMINI RE-ENTRY CONTROL SYSTEM (RCS)

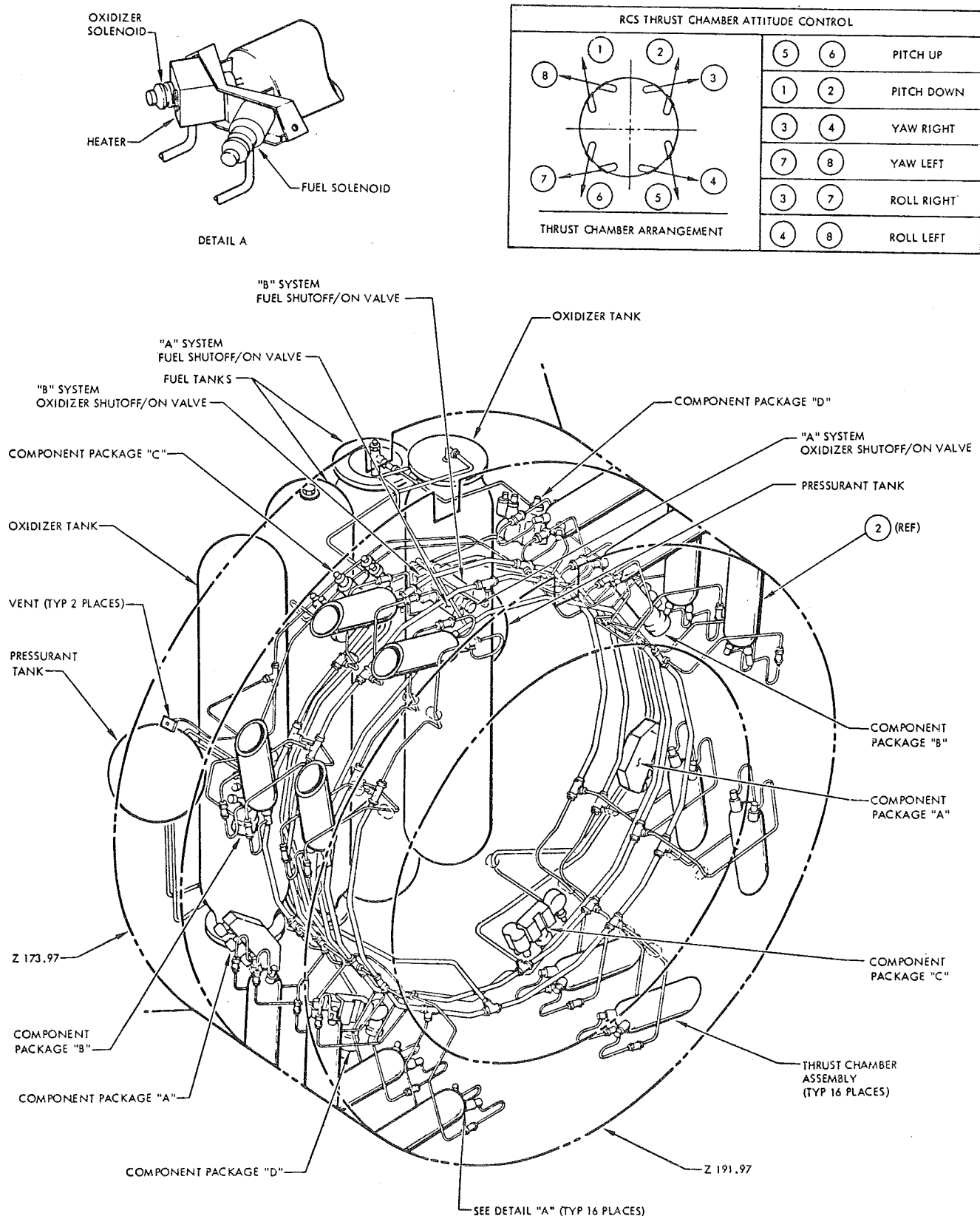


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RE-ENTRY CONTROL "A" AND "B" SYSTEMS

Figure C6-4



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Table C6-2

SUMMARY DESCRIPTION - RCS COMPONENTS

1. Pressurant Group

Pressurant Tank - Nitrogen storage; a 3000 psia titanium sphere having a 7.25 inch O.D. and an internal volume of 185 cubic inches.

"A" Package - Same as OAMS component.

"B" Package - Same as OAMS component.

Pressure Regulator - Same as OAMS component.

2. Propellant Group

Oxidizer Tank - N_2O_4 storage/expulsion; consists of a cylindrical, titanium shell, teflon bladder and performed expulsion tube. The tank is 5.1 inches in diameter, 25.4 inches long and has a propellant volume of 439 cubic inches.

Fuel Tank - MMH storage/expulsion; same basic design as oxidizer tank. This tank is 5.1 inches in diameter, 30.9 inches long and has a propellant volume of 546 cubic inches.

"C" Package - Same as OAMS component.

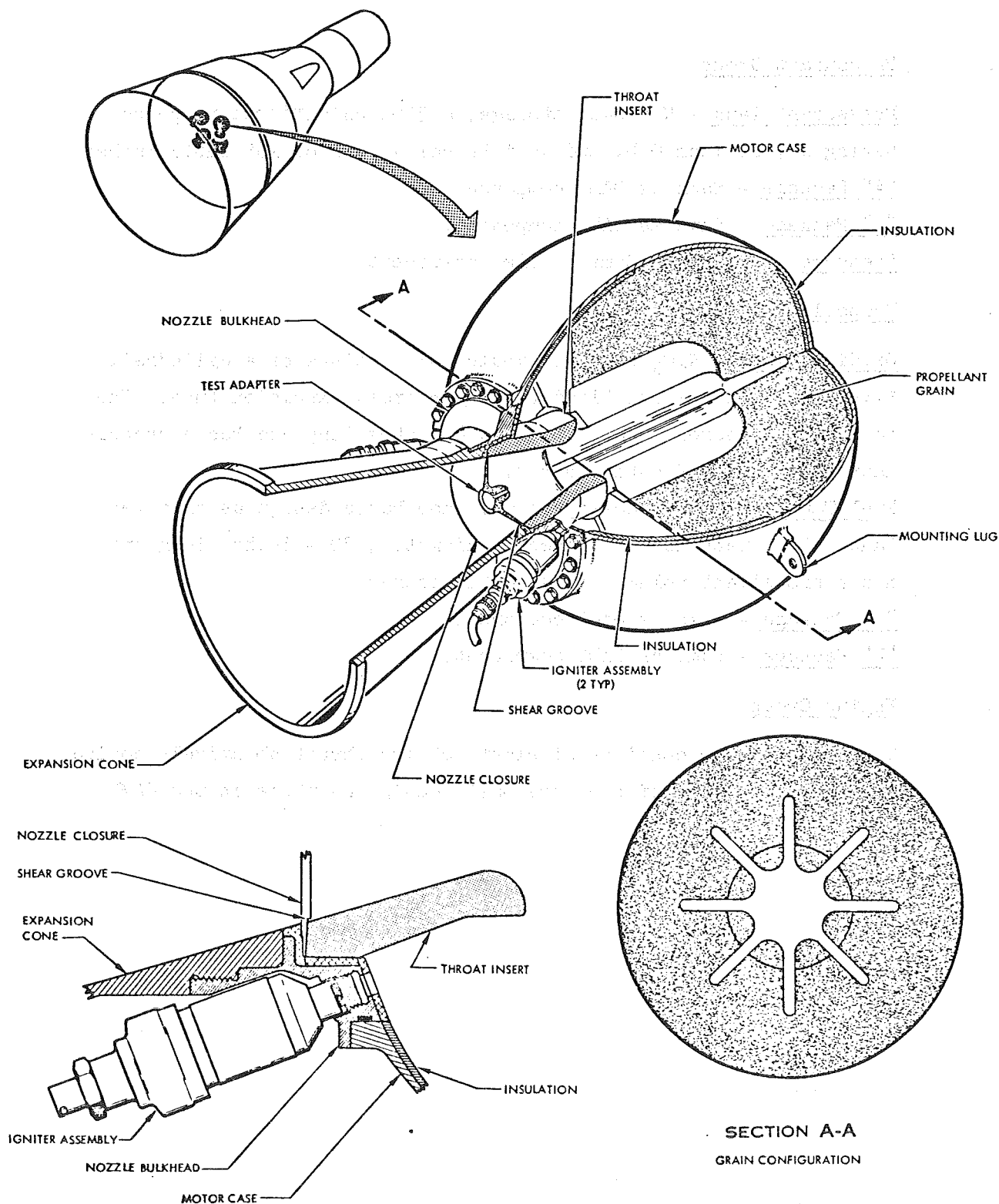
"D" Package - Same as OAMS component.

3. Engine Group

The engine group consists of eight, 23 lbs thrust ablatively cooled engines. They are of the same basic design as those in the OAMS.

Figure C6-5

RETROGRADE ROCKET MOTOR ASSEMBLY

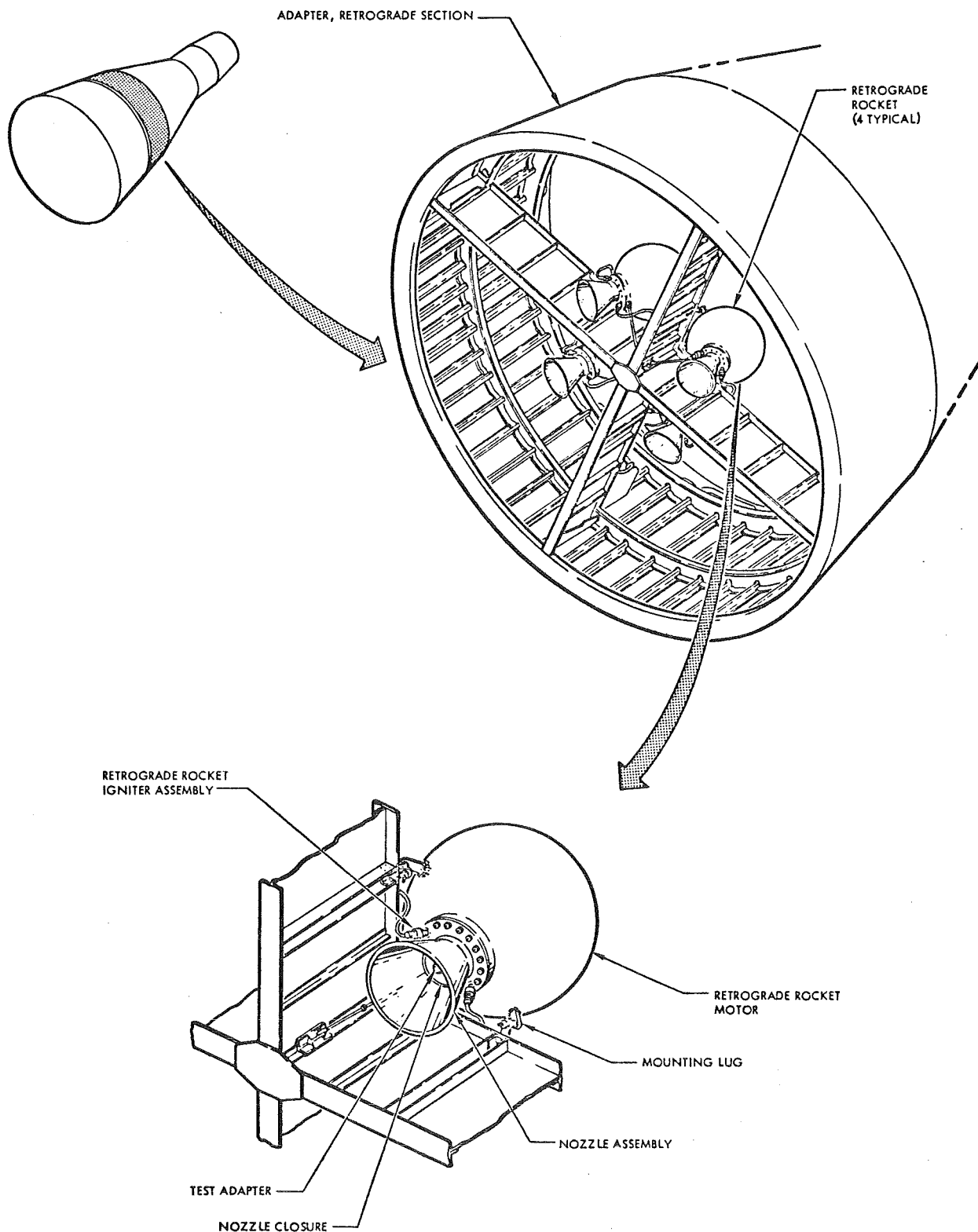


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RETROGRADE ROCKET SYSTEM

Figure C6-6



7. ENVIRONMENTAL CONTROL SYSTEM - The Environmental Control System (ECS), reference Figure C7-1, provides controlled environments for the astronauts, electronic equipment, and regulated temperature for the fuel cells through-out all phases of the mission. General design criteria are given in Table C7-1.

The astronauts are provided with redundant atmospheres by having a closed pressure suit circuit within the pressurized cabin. Both the suit and cabin circuits provide temperature, pressure, circulation, and purge type composition control.

In the event that it becomes necessary for an astronaut to eject from the capsule, below 45,000 ft. altitude, his egress system purges his suit of carbon dioxide, provides pressurization, provides breathing oxygen, and isolates him from an environment which could be toxic, and provides buoyancy for water landings.

The electronic equipment is maintained within the required temperature limits by thermal communications with the coldplates on which they are mounted. Each coldplate is cooled by some portion of the coolant which is circulated for all temperature control on the vehicle.

The temperature level of the gradient across the fuel cells is maintained within the required limits by the automatic temperature regulator at the inlet to the fuel cell and by the mass flow rate or number of pumps operating.

The Environmental Control System may be separated into five systems or loops which operate somewhat independent of each other. They are: oxygen supply system, cabin loop, suit loop, water management system and coolant system.

A block diagram of the ECS is shown in Figure C7-2.

7.1 OXYGEN SUPPLY SYSTEM - There are three oxygen systems: primary, secondary, and egress.

Primary Oxygen - Reference: Figure C7-3. This system stores and dispenses oxygen for breathing, and suit and cabin pressurization. It provides oxygen during the period beginning two hours prior to launch and

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ending with jettison of the adapter section at retrograde.

The primary oxygen supply is stored at supercritical pressure in a cryogenic spherical container in the adapter section. This container is filled with liquid oxygen at atmospheric pressure.

Secondary Oxygen - Reference: Figure C7-3. The secondary oxygen system performs the same functions as the primary oxygen system when the pressure in the primary system falls below 75 $\pm$ 10 psi. It also assumes the duties of the primary system at retrograde, when the primary container is jettisoned with the equipment adapter.

The secondary gaseous oxygen supply is stored in cylindrical tanks located in the re-entry module. Each tank contains 6.5 pounds of usable oxygen pressurized to 5,000 psig maximum at 70°F. This oxygen supply is then regulated before it enters the suit or cabin.

Egress Oxygen - This system provides breathing and suit pressurization oxygen for each pilot below 45,000 feet altitude when ejection procedures are initiated.

The egress oxygen supply is stored in a tank located in each seat-mounted egress kit. Each tank contains 0.31 pound of usable pressurized oxygen.

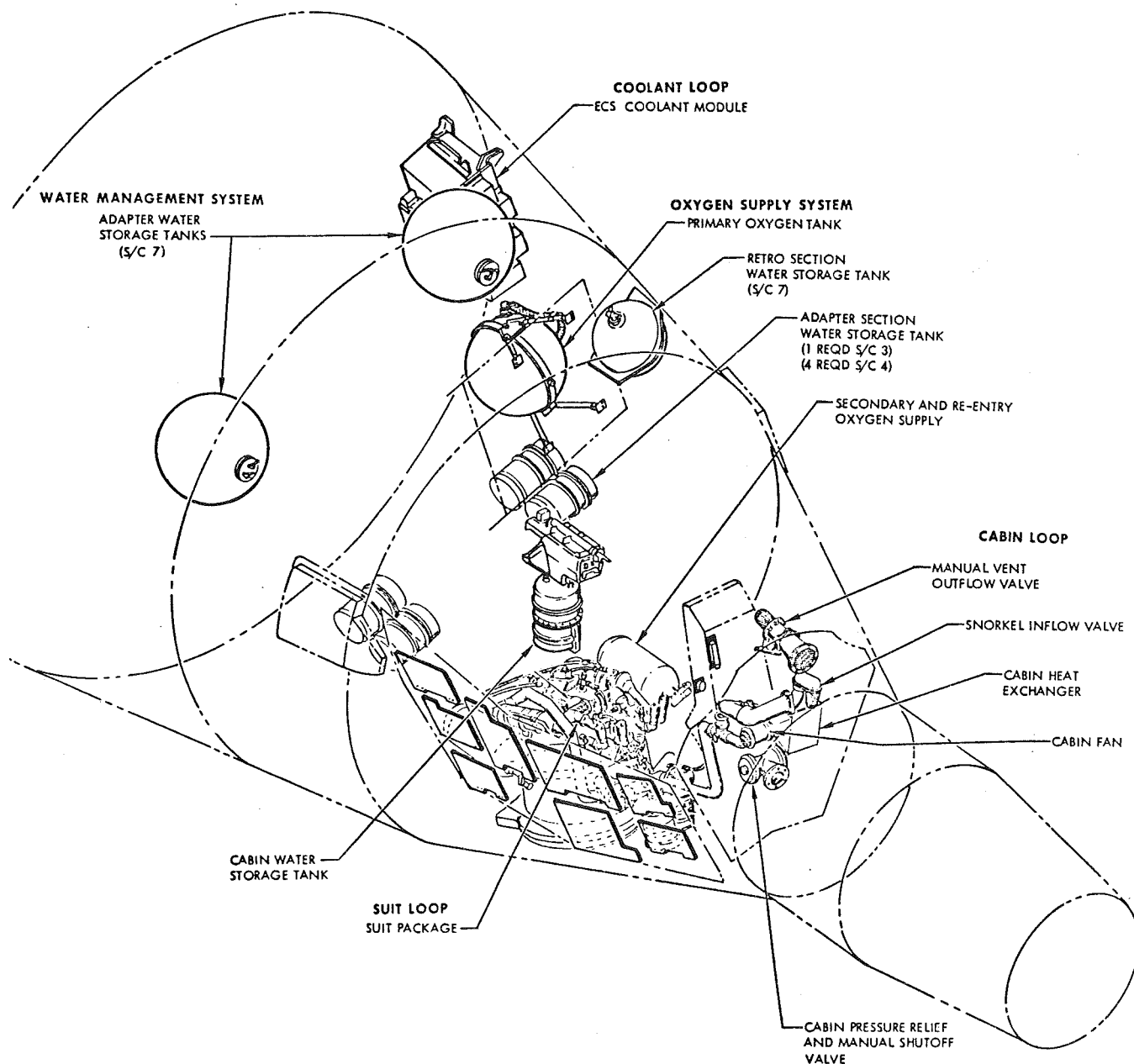
Cabin Loop - Cooling of the cabin gas is accomplished by a cabin heat exchanger and fan. The cooling medium is a fluid that circulates through the heat exchanger. Normal cabin leakage allows the cabin pressure to decay to a nominal value of 5.1 psia and the cabin pressure control valve maintains this value automatically. In order to obtain maximum utilization of oxygen, it first passes through the suit loop before it is dumped into the cabin through the suit pressure relief valves.

Suit Loop - The pilots are provided with redundant atmospheres by having a closed pressure suit circuit within the pressurized cabin. This suit circuit provides for cooling, pressurization, purification and water removal.

The suit loop is a closed system with two pressure suits operating in parallel. Circulation of oxygen through the suit is provided by a

Figure C7-1

ENVIRONMENTAL CONTROL SYSTEM



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Table C7-1

ENVIRONMENTAL CONTROL SYSTEM DESIGN CRITERIA
GEMINI SPACECRAFT

METABOLIC

Heat Production Rate	500 BTU/Man/Hr.
O ₂ Consumption Rate	2.0 Lbs./Man/Day
CO ₂ Production Rate	2.5 Lbs./Man/Day
Heat Production Rate	309 BTU/Man/Hr.
Drinking Water - Post Landing - Cabin	14 Lbs.
- Emergency - Survival Kits	9 Lbs. approx.
Drinking Water Required	7.5 Lbs./Man/Day
Water Stored in the Adapter - 14-Day	210 Lbs.
Water Production	
Perspiration and Respiration	4 Lbs./Man/Day
Urine	3 Lbs./Man/Day
Feces	0.5 Lb./Man/Day
Latent Cooling	
Pre-Launch	291 BTU/Man/Hr.
Launch	291 BTU/Man/Hr.
Orbit	Avg. 350-380 BTU/Man/Hr.
Re-entry	480 BTU/Man/Hr.
Post Landing	540 BTU/Man/Hr.

LEAKAGE

Cabin (measured at sea level with N ₂ and cabin pressure of 19.7 PSIA)	500 cc/min.
Suit (measured at sea level with N ₂ and suit pressure of 19.7 PSIA)	300 cc/min.

CABIN EQUIPMENT HEAT INPUT

683 BTU/Hr.

SUPERCRITICAL TANKAGE - ECS

14-Day Mission - O ₂	106 Lbs. (Residual 8.95 Lbs.)
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Table C7-1 (Continued)

WATER PRODUCTION - NON-DRINKING

14-Day Mission - Water Separator	112 Lbs.
- Fuel Cells	189 Lbs. Max.

ATMOSPHERIC PURIFICATION

Heat Production Rate	400 BTU/Man/Hr.
Maximum Allowable CO ₂ Partial Pressure	8 mm Hg.
Heat Generated/Lb. CO ₂ Absorbed	1286 BTU
LIOH Required/Lb. CO ₂ Absorbed	1.08 Lb.
LIOH Required - 14-Day Mission	71.0 Lbs.

Figure C7-2

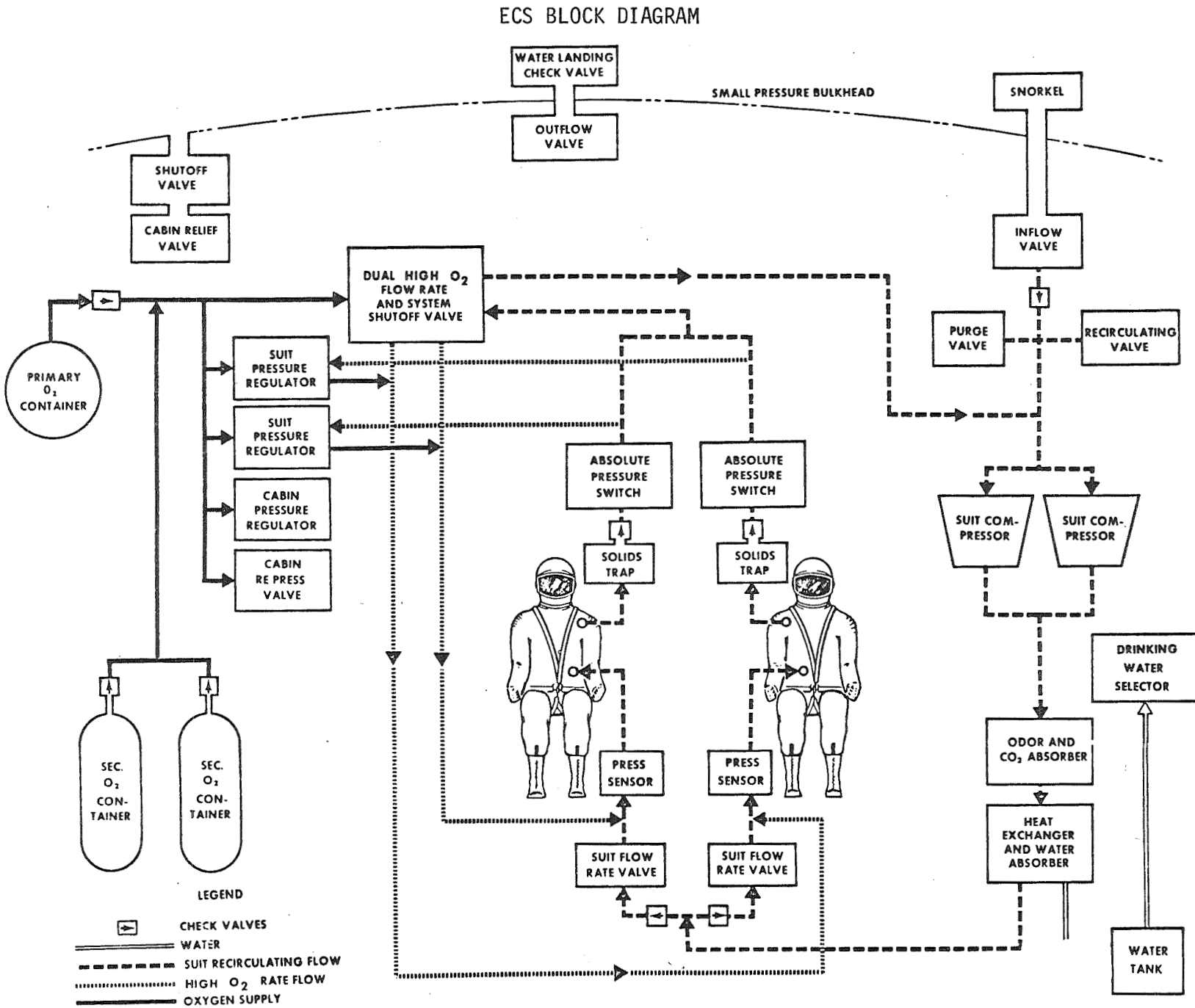
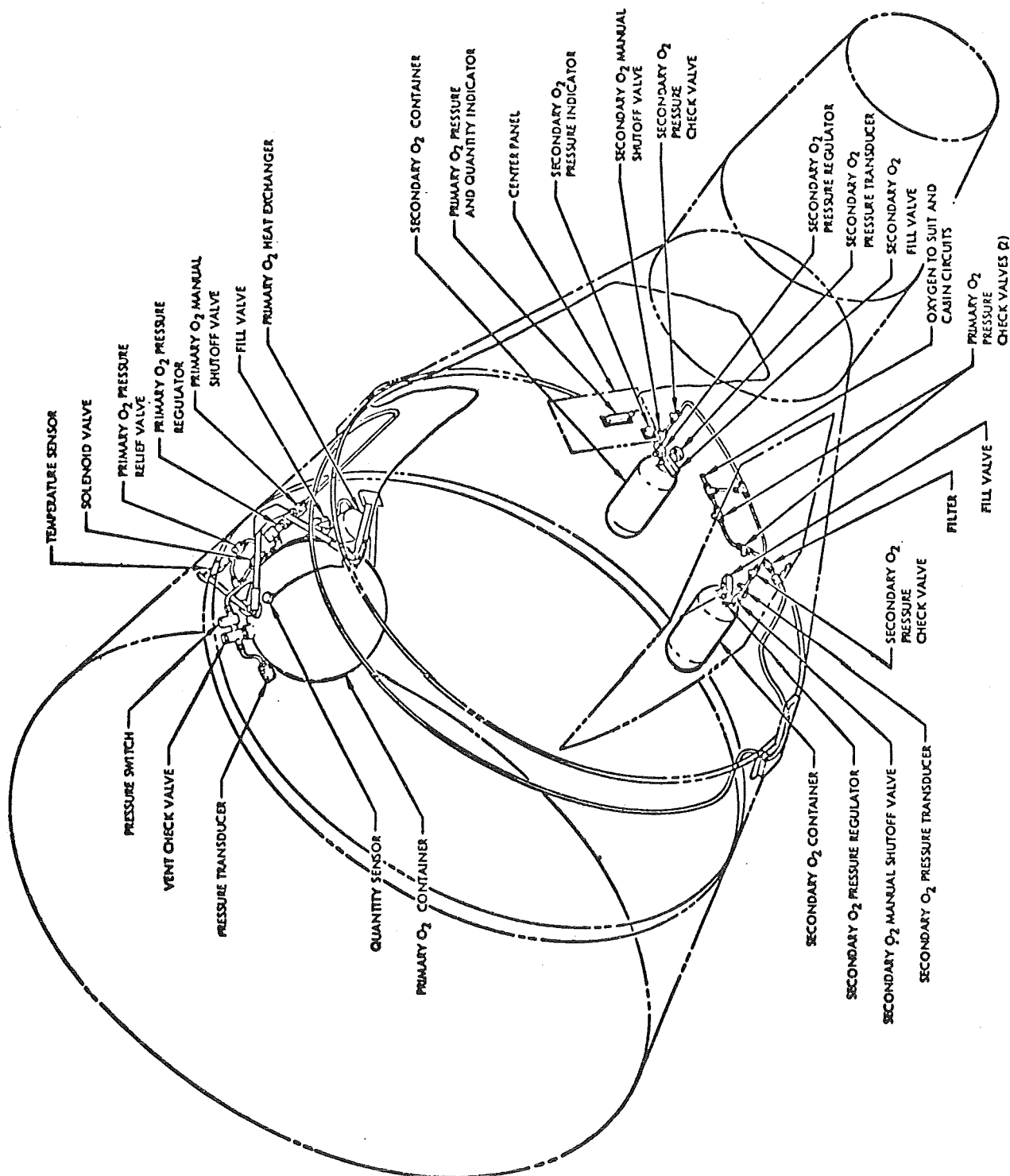


Figure C7-3

PRIMARY AND SECONDARY OXYGEN SYSTEM



centrifugal compressor. Carbon dioxide and odors are removed by an absorber bed containing lithium hydroxide and activated charcoal. The gases are cooled in a heat exchanger by a liquid coolant, MSC-198, at a point below the dew point temperature. Water condensing within the heat exchanger is dumped overboard or routed to the water evaporator. The reconditioned oxygen is mixed with fresh makeup oxygen.

The suit circuit has two modes of operation; the normal recirculation mode is discussed above and the high rate mode which shuts off the recirculation system and dumps oxygen directly into the suit. This mode of operation is normally used during the re-entry phase of the mission.

7.2 WATER MANAGEMENT SYSTEM - The functions of the water management system are to store and dispense drinking water and to collect and route unwanted water to the evaporator or dump overboard.

Drinking water is stored in three tanks in the adapter. Each tank contains a bladder and is pressurized to supply water to the transparent tank in the re-entry module. Two storage tanks store both drinking water and fuel cell by-product water. Both of these tanks use a combination of oxygen and fuel cell water as the pressurant. The third tank is pressurized with oxygen and stores only drinking water.

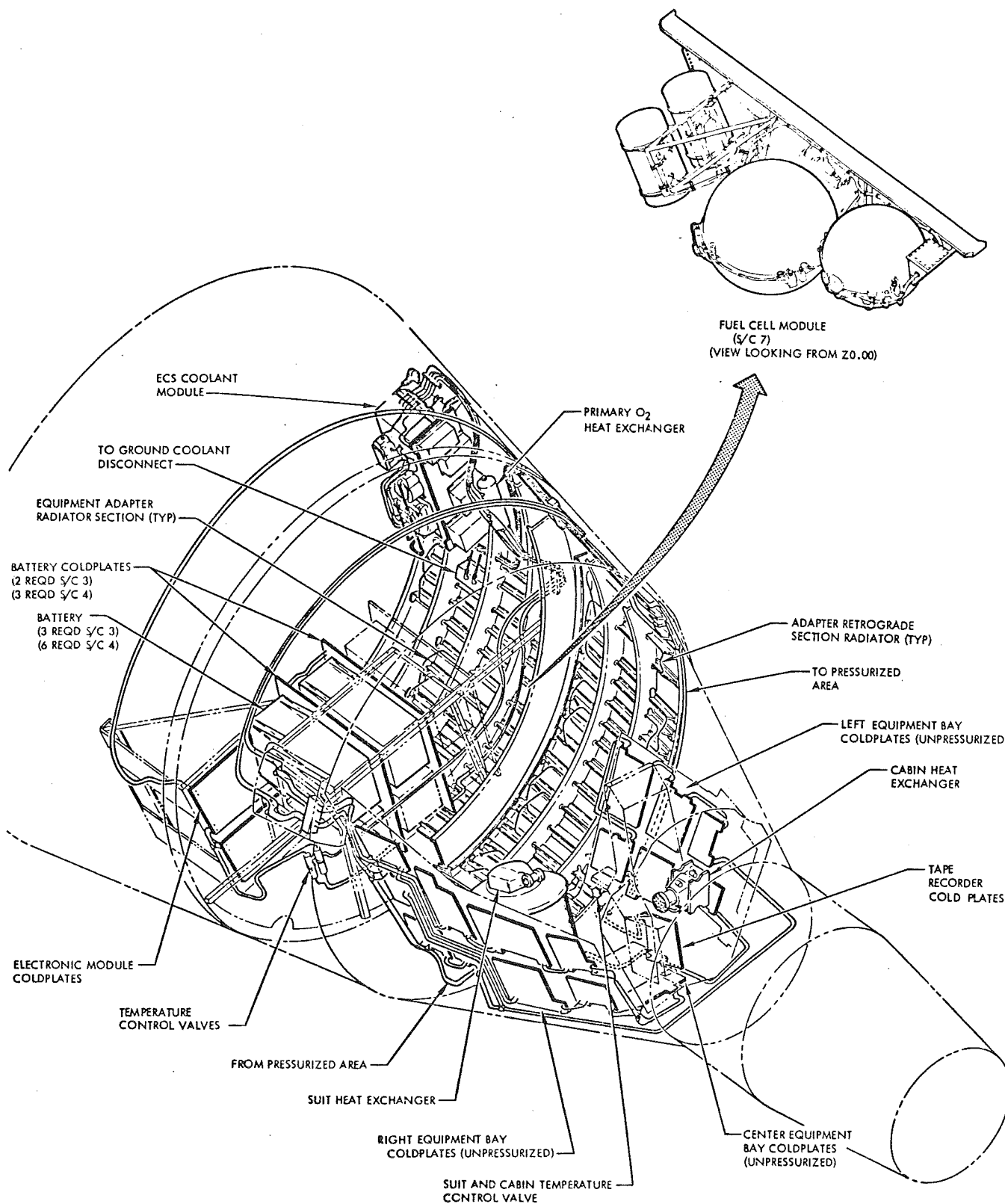
Urine and condensated water from the suit circuit heat exchanger is absorbed by the wick in the water boiler or dumped overboard.

The urine disposal equipment is government furnished equipment (GFE). It includes the urine line, bellows assembly, quick disconnect coupling, and uriceptacle.

7.3 COOLING SYSTEM - Reference: Figure C7-4. The spacecraft cooling system consists of two identical temperature control circuits functioning independently of each other to provide the cooling requirements of the spacecraft. Each cooling circuit consists of a pump package, thermostatic and directional control valves, various type heat exchangers, radiators, filters, and the necessary plumbing required to provide a closed circuit. The cooling system may be operated in either the primary and/or secondary circuit, and is capable of carrying maximum heat loads in either circuit.

SPACECRAFT COOLANT SYSTEM

Figure C7-4



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During orbital flight, Monsanto MCS-198 coolant is supplied throughout the cooling system and thermostatic control valves regulate the coolant temperature. Temperature sensors, located in the system provide the necessary telemetering to ground stations.

Cooling is provided throughout the mission up to pre-retrograde firing. At this time the coolant pump packages are jettisoned with the adapter equipment section, terminating spacecraft cooling. The primary circuit operates continuously providing the required cooling during low temperature loads. The secondary circuit is used, in conjunction with the primary circuit, during phases of high temperature loads such as launch and pre-retrograde. Under normal heat loads, the #1 pump in the primary circuit provides the required cooling. Under peak heat loads, the #1 pump in the secondary circuit is used with the primary circuit #1 pump to provide maximum cooling. In the event of a #1 pump failure in either circuit, the #2 pump in that circuit is used. In the event of both pumps failing in one circuit, both pumps of the remaining circuit can be used to provide the required cooling.

During pre-launch an external supply of MSC-198 coolant is circulated through the spacecraft ground cooling heat exchanger. The #1 pumps of the primary and secondary cooling circuits are activated by an external power source. The spacecraft radiator is bypassed in the circuit and flow of coolant is directed through the ground cooling heat exchanger. Shortly before launch, the external cooling and electrical power are disconnected.

During launch, the launch cooling heat exchanger goes into operation when the coolant temperature exceeds 46^{+4}_{-2} °F. At this time the vernatherm valve opens pressurizing a bellows which unseats the poppet valve exposing the water in the heat exchanger to reduced pressure as altitude increases. When the spacecraft altitude reaches 100,000 feet, the water in the heat exchanger boils absorbing heat from the coolant. Thus, the absorbed heat is expelled overboard in the form of steam.

After orbiting for approximately 30 minutes, allowing the radiator to cool after being subjected to launch heating, the coolant flow is directed through the space radiators. This bypasses the ground cooling heat

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exchanger. The evaporator selector valve is positioned so that only flow to the suit and cabin heat exchangers passes through the evaporator.

Prior to adapter jettisoning, the #1 coolant pumps for both the primary and secondary circuits are activated. The suit, cabin and equipment bays are cooled to as low a temperature as possible, before the adapter equipment section is jettisoned.

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8. CREW STATION - The equipment within the cabin is arranged to permit the Command Pilot, seated to the left, and the Pilot, seated to the right, to operate the controls and observe displays and instruments. The cabin equipment arrangement is shown in Figure C8-1.

8.1 CREW SEATING - The crew is seated in rocket catapult ejection seats. Crew seating provisions include seats, restraint mechanisms, ejection devices, seat-man separator, survival gear, and egress kit assembly. Escape provisions will be discussed in greater detail in Section 9.

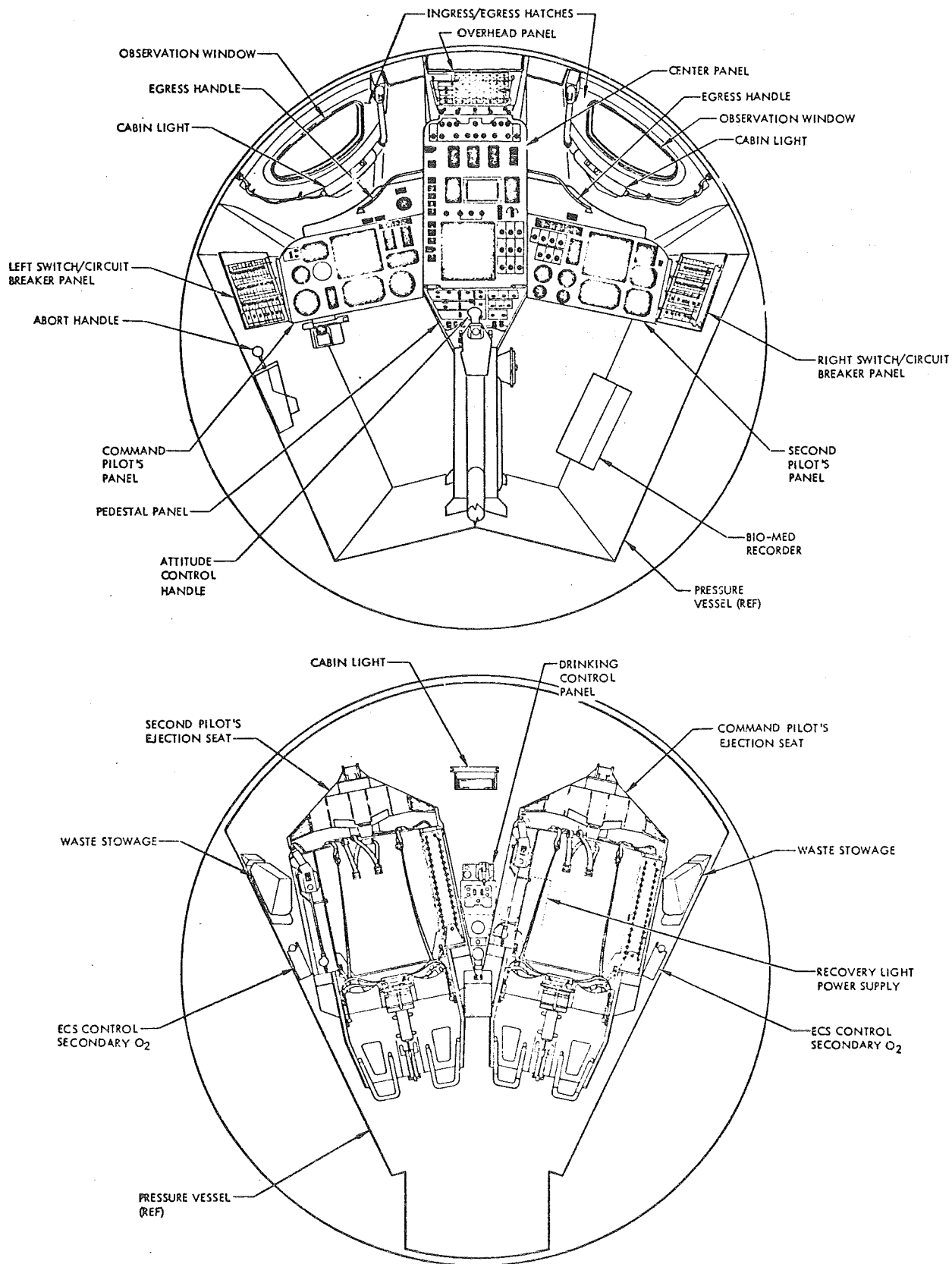
8.2 CONTROLS AND DISPLAY - The controls and displays are located on three panels, one centered between the astronauts, one in front of the Command Pilot, and one in front of the Pilot (Reference Figure C8-1). The attitude control handle is located between the astronauts.

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CABIN EQUIPMENT (TYPICAL)

Figure C8-1



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9. ESCAPE, LANDING AND RECOVERY

9.1 ESCAPE - The primary escape system in the Gemini Spacecraft is the ejection seats. After the Spacecraft has exceeded the operational limits of the ejection seats abort is accomplished by firing the retrorockets to separate the spacecraft from the launch vehicle and mission module. A normal landing sequence then returns the spacecraft to the Earth's surface.

The ejection seats are canted outboard 12 degrees and 8 degrees 20 minutes forward to assure separation and provide required elevation for off-the-pad abort. A rocket catapult separates the seat and astronaut from the spacecraft. Maximum acceleration of the astronaut is approximately 17 g's. The ejection seat is operable to 70,000 feet altitude, speeds up to Mach 2.86, and at a dynamic pressure up to 820 psf.

9.2 LANDING AND RECOVERY - A three parachute system, in series, stabilizes and retards the entry vehicle velocity. The three parachutes are a drogue, pilot, and main parachute.

The drogue is an 8.3 feet D_0 conical ribbon parachute. The pilot parachute is an 18.3 feet D_0 ringsail configuration.

The main decelerator is an 84.2 D_0 ringsail parachute. This parachute allows a 31.6 fps rate of descent for a module weight of 4,400 pounds.

9.3 RECOVERY - Flotation of the entry vehicle is achieved by displacing water with the cabin vessel. To improve the flotation attitude styrofoam was installed in several of the equipment bays.