INTEGRAL LAUNCH AND REENTRY VEHICLE SYSTEM

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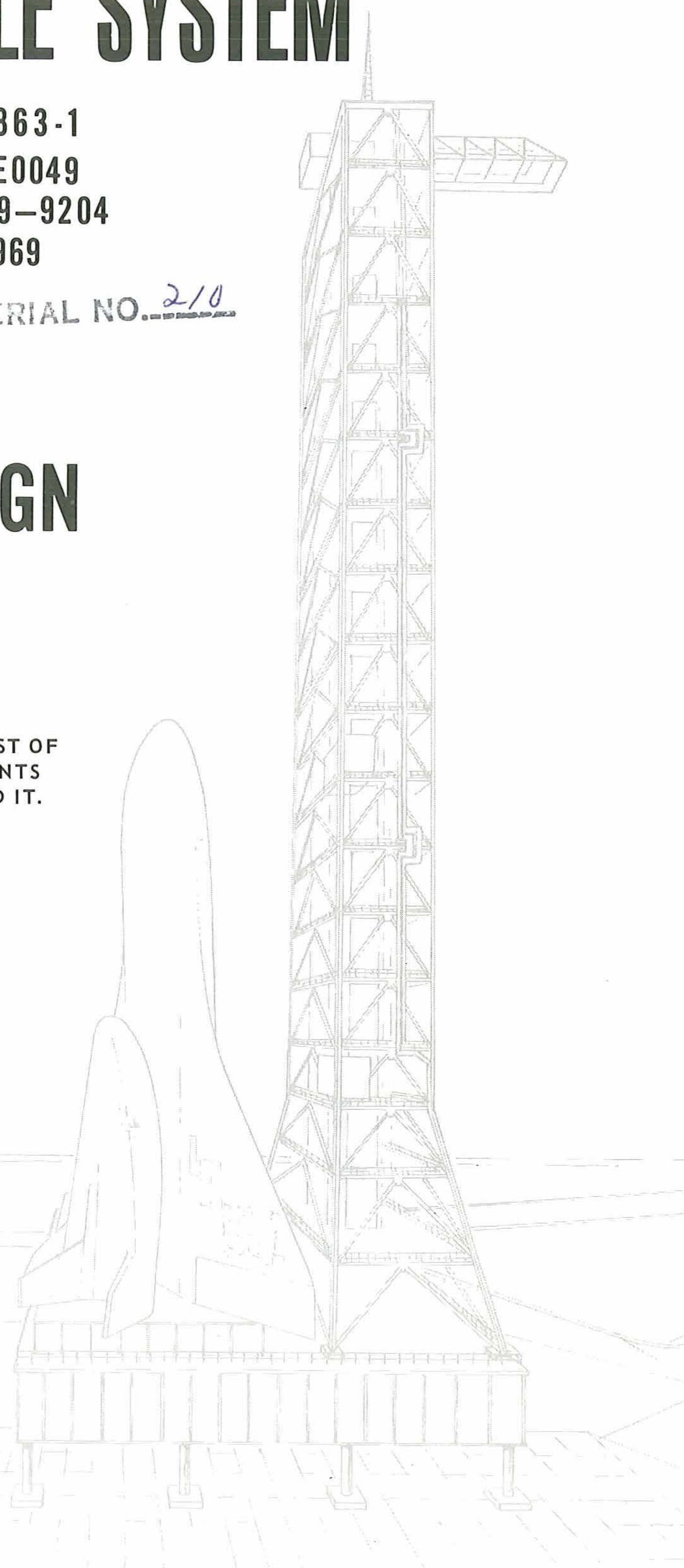
VOLUME I

SERIAL NO. 210

BOOK 1

CONFIGURATION DESIGN AND SUBSYSTEMS



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MCDONNELL DOUGLAS
CORPORATION

FOREWORD

This volume of McDonnell Douglas Astronautics Company Report Number MDC E0049 constitutes a portion of the final report for the "Integral Launch and Reentry Vehicle Systems Study". The study was conducted by the MDAC for the NASA-Langley Research Center under Contract NAS9-9204.

The final report consists of the following:

Executive Summary
Vol. I - Design, Configuration and Subsystems
Vol. II - Performance, Aerodynamics, Mission and Operations
Vol. III - Plans, Costs, Schedules, Technologies
Vol. IV - One and a Half Stage

McDonnell Douglas Astronautics Company gratefully acknowledges the cooperation of the companies which provided technical assistance during this study. They are:

Pratt & Whitney Aircraft Division, United Aircraft Corporation
Rocketdyne Division, North American Rockwell Corporation

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ABSTRACT

This study emphasized a two stage to orbit reusable spacecraft system for use in transporting cargo and passengers to and from a near earth orbital space station. A single conceptual "point" design was treated in detail and several alternate systems, corresponding to alternate payloads (size and weight), were examined based on parametric excursions from the "point" design. The overall design goal was to configure the carrier and orbiter vehicles to minimize operational and program recurring costs. This goal was achieved through high system reliability, vehicle recoverability,and rapid ground turnaround capability made possible through modular replaceable component design and use of an integrated onboard self test and check-out system. Launch and land landing of both stages at the ETR launch site was a study groundrule as was the nominal 25,000 lb payload delivered to and returned from orbit and packaged in a 15 ft. diameter by 30 ft. long cylindrical canister. The resulting system has a gross lift-off weight of 3.4 million pounds.

The Orbiter is a 107 ft. HL-10 configuration, modified slightly in the base area to accommodate the two boost engines. The launch propellant tanks are integral with the primary body structure to maximize volume available for propellant.

The Carrier is a 195 ft. clipped delta configuration with ten launch engines identical to those of the orbiter. A dual lobed cylindrical launch propellant tank forms the primary body structure. A 15% thick delta wing is incorporated which contains the landing gear, airbreathing engines and propellant.

A broad range of weight, cost and performance sensitivity data were generated for the baseline and alternate system designs. Pertinent development and resource requirements were identified, development and operational schedules were prepared and corresponding recurring and non-recurring cost data were estimated. Program plans were outlined for the design, manufacture and testing of the Orbiter and Carrier vehicles and for the pursuit of critical technologies pacing vehicle development.

Stage and a half and reusable systems employing expendable launch vehicles were considered initially, but, these efforts were subsequently terminated prior to completion. The expendable launch vehicle data are reported separately. The stage and a half effort employed a version of the McDonnell Douglas Model 176 with four drop tanks.

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1.0 INTRODUCTION - DESIGN CONFIGURATION AND SUBSYSTEMS

This volume is separated into two books. Book I contains the configuration analysis, subsystems exclusive of propulsion system, weights and reliability analyses, and a definition of the baseline designs. Book II contains the propulsion subsystems. These are treated separately due to the classified nature of much of the data presented, and because a major portion of the study was directed to the launch propulsion aspects of the configuration.

Two special emphasis areas are included in this book: "Re-entry Heating and Thermal Protection System Analysis" and "Integrated Avionics".

2.0 GUIDELINES AND CRITERIA

The general guidelines, criteria and ground rules applied to the design of the two stage reusable spacecraft are listed below. Additional specific ground rules are included in their applicable sections.

- The HL-10 shape defined by the NASA-LRC will be used for the orbiter. Modifications to vehicle lines will be with NASA-LRC approval. Separate upper and lower elevons will be used.
- The carrier will be developed and defined by MDAC during the study. The baseline shape will be selected with NASA-LRC approval.
- Baseline payload for point design and comparison purposes will be 25,000 lb. contained in a 15 ft. diameter, 30 ft. long container. Alternate payloads are defined as:
  - 50,000 lb. in a 15 ft. dia., 60 ft. long envelope
  - 25,000 lb. in a 15 ft. dia., 60 ft. long envelope
  - 10,000 lb. in a 15 ft. dia., 3,000 cu. ft. envelope
  - 50,000 lb. in a 22 ft. dia., 60 ft. long envelope
- The configuration will have the capability of exchanging payload weight and volume for up to 10 passengers.
- The configuration will have the capability of carrying the specified cargo to orbit, and returning the cargo.
- Boost propellants will be LOX/LH_2
- High chamber pressure bell nozzle engines will be used for boost as a baseline.
- Boost engines will be the same size for both stages if possible.
- Capability of boost engine burn on both stages at launch is preferred.
- Alternate boost engine configurations will be investigated and recommended if applicable.
- Both stages will have air breathing engine powered landing with once around go-around capability.
- Landing will be at a prepared horizontal runway.
- Ferry capability will be investigated.
- Both stages will have a 2 man crew.

- Crew will operate in a shirtsleeve environment.
- EVA will not be required for orbital transfer of crew and cargo.
- Longitundinal acceleration loads will be 3g maximum with passenger, 4g maximum for crew only.
- Mission will be completed with one engine out in either stage.
- Propellant will not be transferred between stages (desirability of propellant transfer will be investigated).

3.0 CONFIGURATION ANALYSIS

A major portion of the two stage recoverable spacecraft study was devoted to configuration development and analysis. The HL-10 shape defined by the NASA-LRC was designated as the second, or orbiting, stage. The first, or carrier, stage was developed by MDAC. The configuration analysis was an evolutionary process to define the orbiter, carrier and the launch configuration. The two stages were optimized in terms of gross launch performance, mission interfaces, and first stage configuration. The results of the study are illustrated as baseline carrier, orbiter and launch configurations.

3.1 Configuration Evolution - The evolution of the two stage reusable spacecraft concept from the inception of its study to selection of candidates, plus the rationale for their selection, is presented in this section. The study progressed from a matrix of possible configurations to a selection of two candidates and refinement of configuration details.

3.1.1 Two Stage Concepts - The configuration analysis of the two stage reusable spacecraft study was initiated with a study of many candidate launch configurations. Results of the major preliminary concepts which were investigated are shown in Figure 3-1. The HL-10 configuration, shown as the orbiter, and the carrier have a negative camber body in each concept, which was a desired feature to obtain favorable hypersonic trim characteristics as a result of positive zero-lift pitching moment coefficients. Another desired arrangement permits firing of boost engines in both stages prior to or at lift-off. Hence, the base of both stages should be nearly in-plane.

Concepts A and B show HL-10 configurations also for the carrier stage. Concept A is a twin arrangement, with both vehicles the same size, while Concept B shows a carrier larger than the orbiter. The twin concept produces a higher gross launch weight. However, some cost savings might be realized due to similarity of vehicles. Concept C depicts an end-to-end arrangement. This produces a greater overall length, and degrades the capability for using the orbiter boost engines during carrier burn. Concept D shows the orbiter nested in a V-tank carrier arrangement. The upper surface of this carrier is closed with a web between tanks to present a delta planform shape. Concepts E and F present carriers which utilize parallel launch propellant tanks to form the basic

TWO-STAGE CONCEPTS

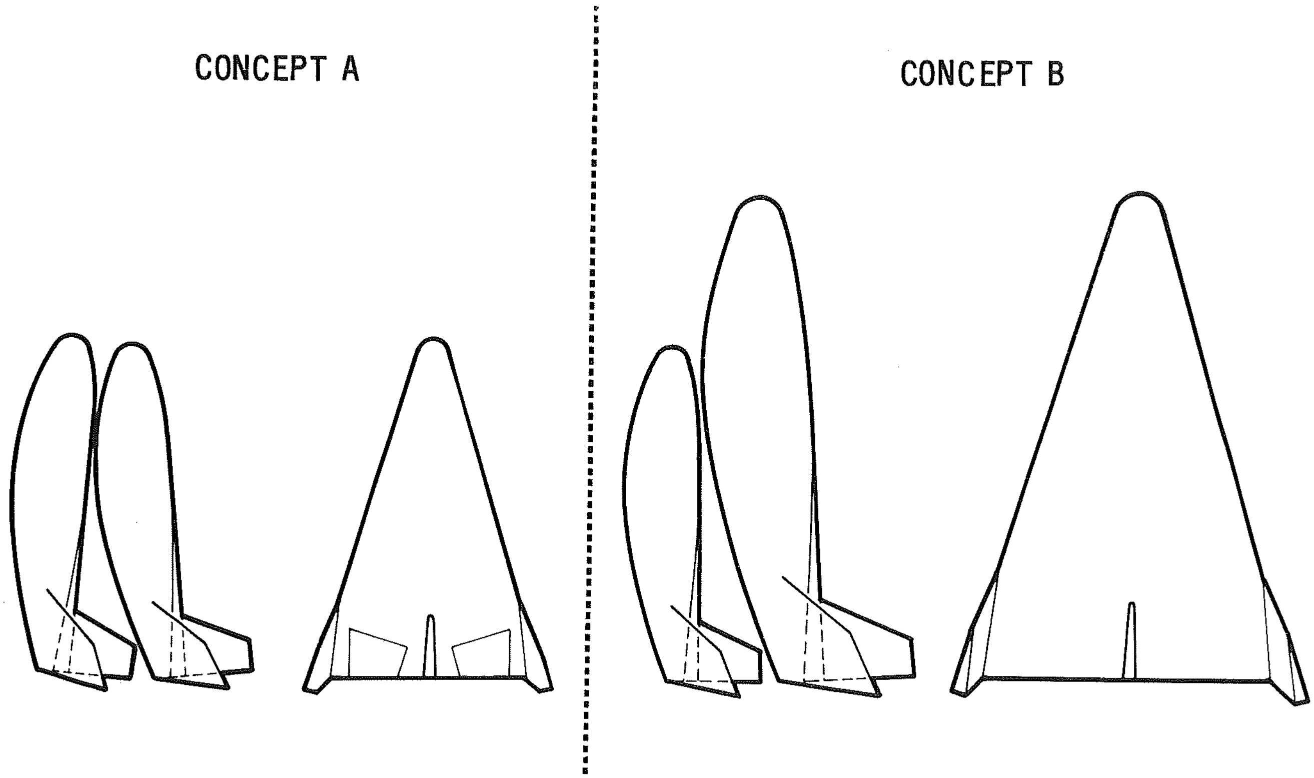

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Figure 3-1

TWO-STAGE CONCEPTS
(Continued)

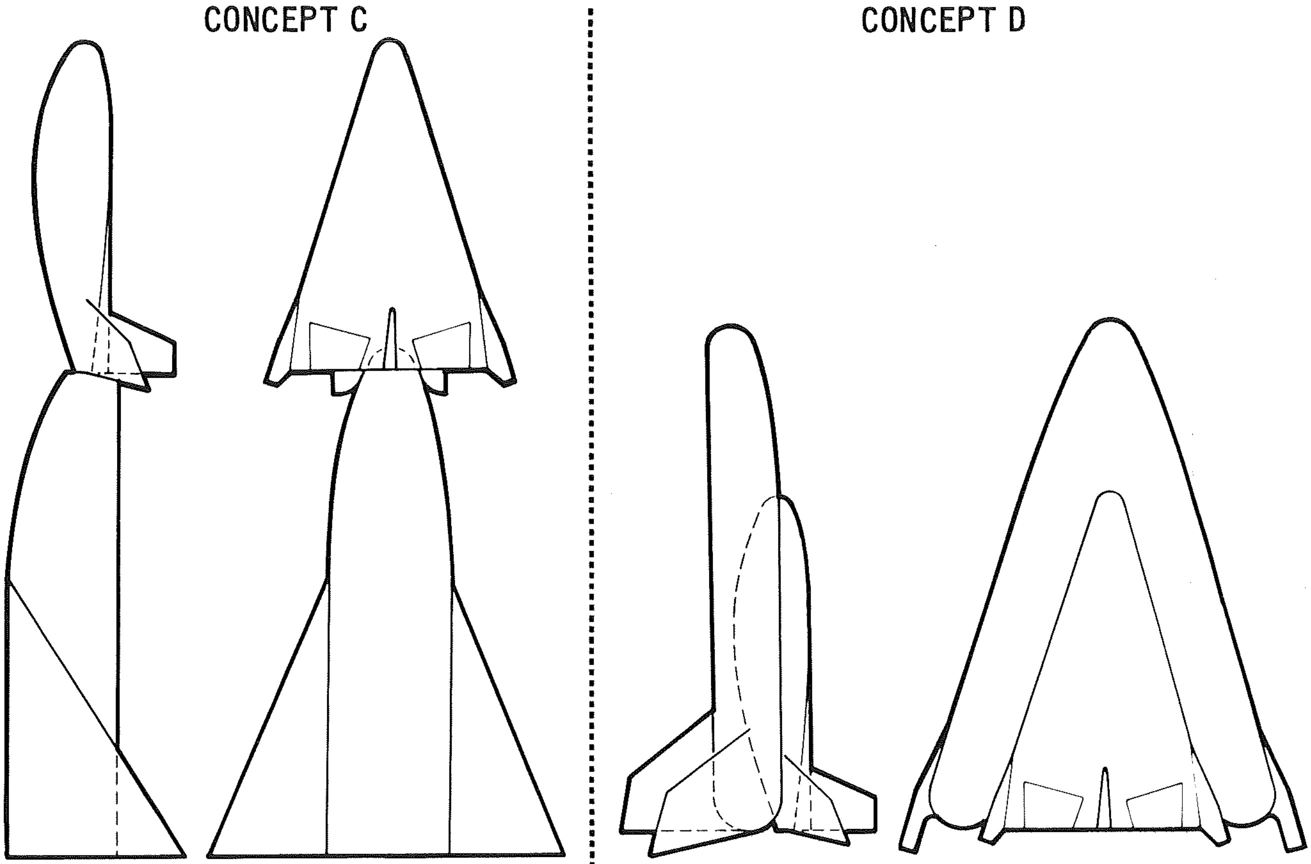

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Figure 3-1 (Con't)

TWO-STAGE CONCEPTS
(Continued)

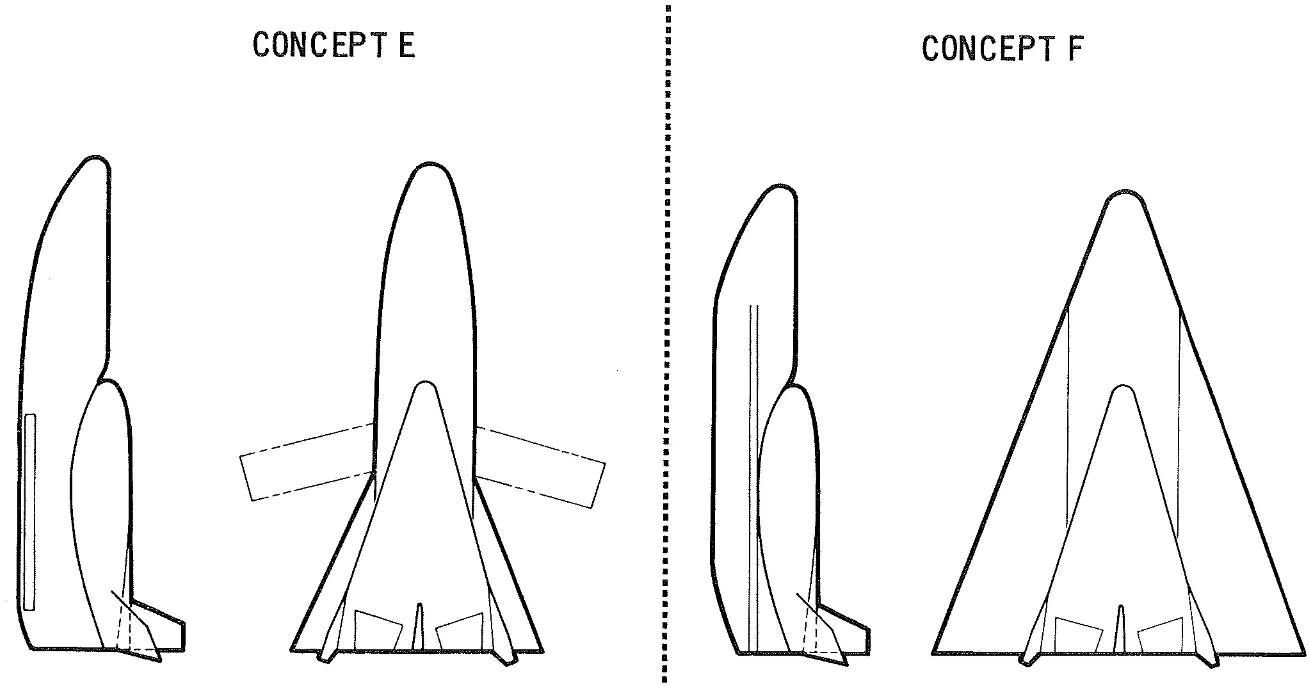

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Figure 3-1 (Con't)

TWO-STAGE CONCEPTS
(Continued)

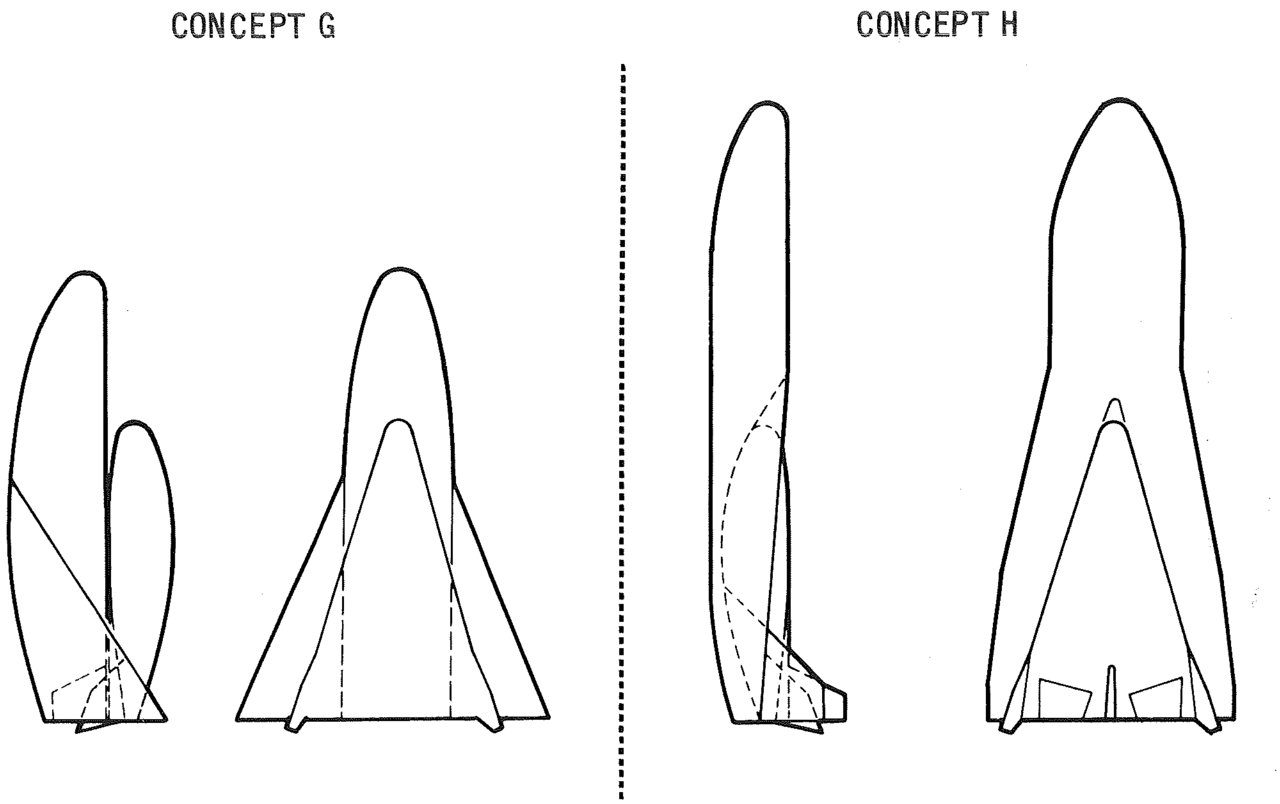

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Figure 3-1 (Con't)

TWO-STAGE CONCEPTS
(Continued)

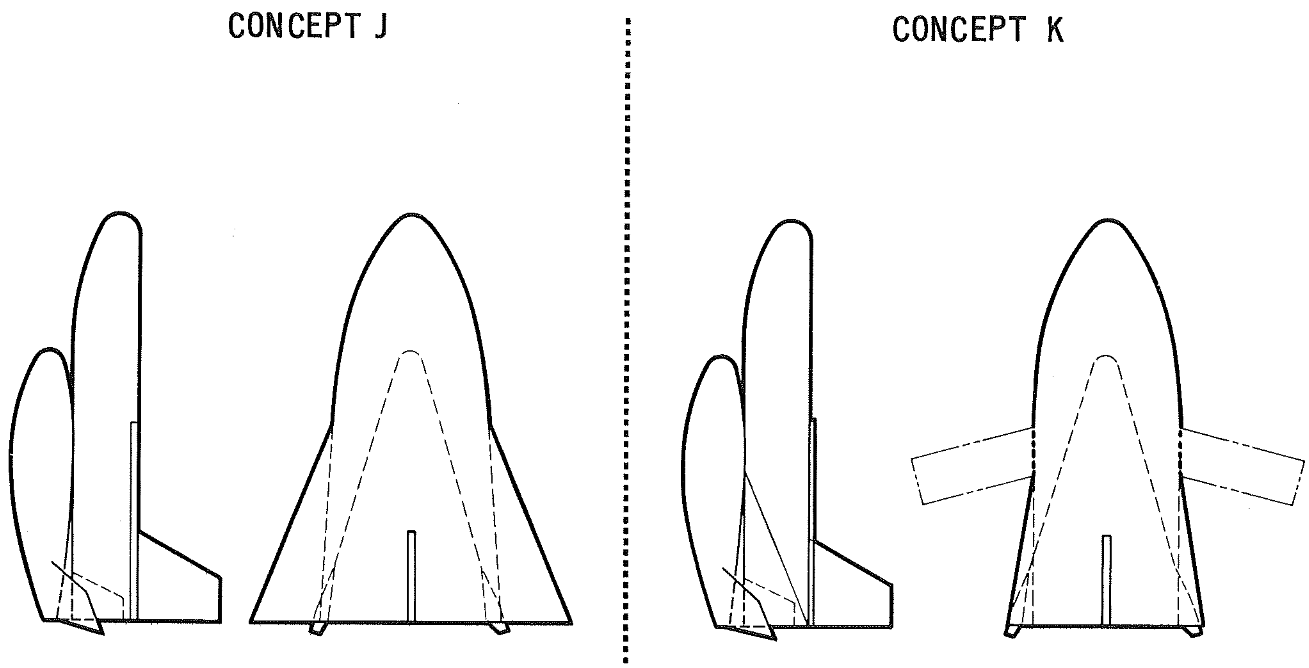

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Figure 3-1 (Con't)

TWO-STAGE CONCEPTS
(Continued)

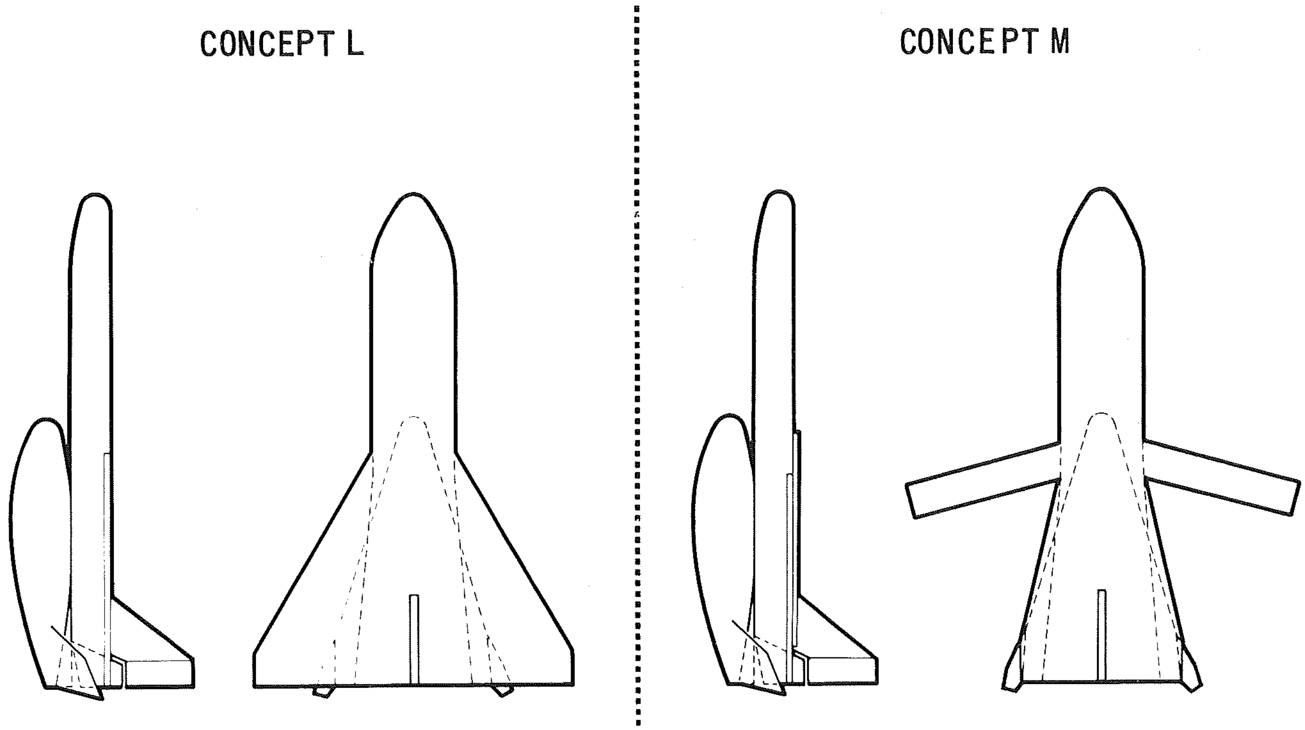

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Figure 3-1 (Con't)

body cross section. These tanks are forward of and underneath the orbiter. Concept E uses a V-tail with variable geometry wings for landing, and Concept F uses a fixed delta wing with vertical control surfaces near the wing tips (not shown in the figure). The parallel tank arrangement provided the basis for subsequent shapes, and development of the baseline configuration was a process of evolution from this point. Concept G depicts an arrangement where the upper surfaces of both stages are mated. This concept presents a separation problem requiring accurately controlled translation of vehicles to effect removal of the orbiter vertical tail from the booster. Concept H shows the orbiter nested in the carrier, with the carrier propellant located forward of the orbiter. This arrangement also results in a long launch configuration because the aft end of the booster is a structural support for the orbiter and does not contribute to efficient propellant utilization. Concepts J and K have the upper surface of the orbiter interfacing with the lower surface of the carrier. The vertical centerline fin of the orbiter is nested in a cavity on the carrier. The carrier for Concept J has a clipped delta fixed wing and a vertical centerline fin. The carrier for Concept K has a V-tail nested between the orbiter side fins, a vertical centerline fin, and variable geometry wings for landing. Both of these concepts also present a separation problem similar to Concept G. Concepts L and M employ the same basic characteristics as Concepts J and K. The carrier bodies have a smaller cross section, providing a higher fineness ratio for a given propellant volume. The tank walls in these two concepts provide the structural skin for the carrier body.

Concepts D, E, L, and M are evaluated in terms of the parameters indicated in Table 3-1. These four concepts represent the most desirable arrangements from the matrix of 12.

Table 3-1

Qualitative Concept Evaluation

Parameters	Concept D Nested, V-Tank	Concept E Nested, Lift Body	Concept L Parallel Tank, FG	Concept M Parallel Tank, VG
L/D - Subsonic	Poor	Acceptable	Good	Good
L/D - Hypersonic	Poor	Acceptable	Good	Good
Stability Margin	Poor	Acceptable	Good	Good
Flow Separation	Yes	Yes	No	No
Structural Efficiency	Poor	Poor	Good	Good
Adaptability to other orbiter	Poor	Poor	Good	Good
Stage separation	Poor	Poor	Good	Good
Growth potential	Poor	Poor	Good	Good
Orbiter-carrier mating	Good	Good	Acceptable	Acceptable

Concept D is expected to have a poor subsonic L/D as a result of a low aspect ratio planform and a poor hypersonic L/D as a result of a large leading edge radius. The stability margin is considered poor as a result of the low aspect ratio planform and the likely aft c.g. position. Because of the large leading edge radius at intermediate angles of attack, a supersonic Mach number flow separation on the lower surface is a likely phenomenon. The region between the two tanks supporting the orbiter is likely to be heavy and structurally inefficient because of the peculiar cross-sectional shape. Furthermore, the shape will limit the carrier application to the HL-10 vehicle. Thus it is not easily adaptable to other orbiter vehicles. The nested orbiter in the carrier increases separation complexity and reduces system reliability. The V-tank does not allow for increase in payload capability through carrier growth. The only point in favor of the concept is its mating with the HL-10 which could be tailored to fit the HL-10 vehicle.

Concept E will display all the characteristics attributed to Concept D. However, as a result of configurational improvements, higher subsonic and hypersonic L/D and stability margins are predicted.

With parallel tanks and fixed geometry as shown in Concept L, good low and high speed characteristics are attained. The parallel tanks result in good structural efficiency, and adaptability to other orbiter configurations. Stage separation is eased because of elimination of the nesting concept. The parallel tanks allow carrier growth if it is deemed necessary. Mating is considered acceptable, although the stage is not tailored to the HL-10 shape.

Concept M, with parallel tanks and variable geometry wing combines all the features of Concept L with improved subsonic characteristics at some increase in vehicle structural weight.

3.1.2 Candidate Carrier Configurations - Two carrier concepts were selected and defined for further investigation and refinement as potential baseline shapes. These were evolved from Concept L shown in Section 3.1.1. This basic concept was selected because of the good aerodynamic performance, structural efficiency, versatility of stage mating and sizing potential, and simplicity of design.

These shapes are shown in Figures 3-2 and 3-3. Figure 3-2 has a clipped delta wing planform shape, and Figure 3-3 has a swept wing with a separate horizontal tail. The vehicle structural skin for both concepts is formed by the launch propellant tank walls. This provides parallel sides for most of the vehicle length. The lower surface at the aft end has a boat-tail arrangement providing a negative camber. The forward body is faired from the propellant tank cross section to the nose radius on both upper and lower surfaces.

Crew compartment and spacecraft systems such as electronics and power supply are located in the section forward of the propellant tanks. Launch engines, feed and pressurization systems are in the section at the base of the vehicle. The clipped delta configuration has a thick wing which provides stowage of the cruise and landing engines during launch. These engines are deployed for cruise flight and landing. Cruise and landing engines for the wing/tail configuration are shown stowed in the forward fuselage. These are also extended for operation.

CANDIDATE – CARRIER
(Clipped Delta)

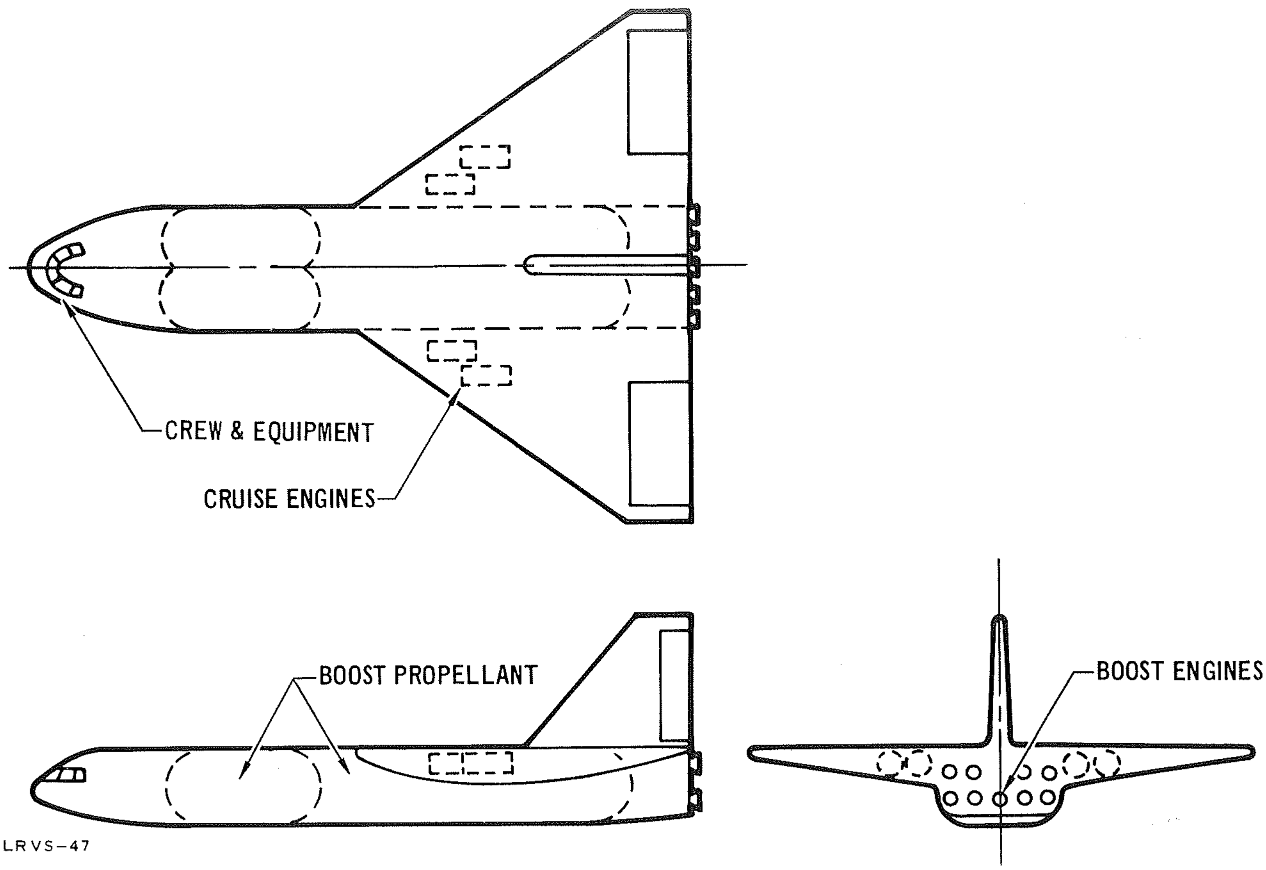

Figure 3-2

CANDIDATE – CARRIER
(Wing-Tail)

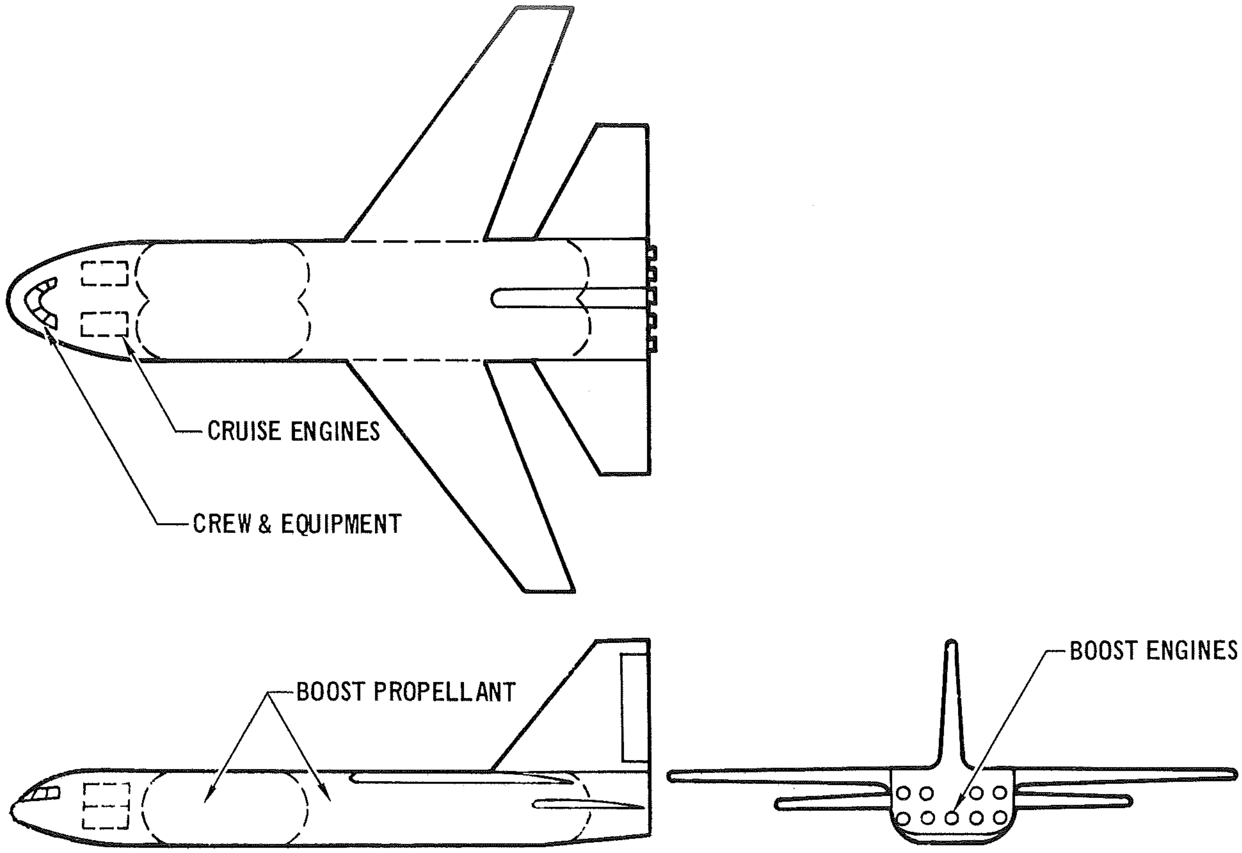

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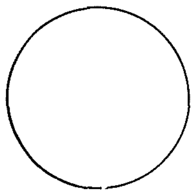
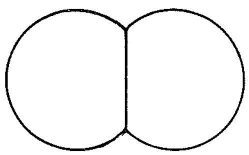
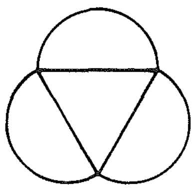
Figure 3–3

3.1.3 Carrier Development - The two candidate carrier shapes selected in Section 3.1.2 were investigated in depth for the purposes of arriving at detailed definition of a baseline vehicle. The basic vehicle body is common to both the clipped delta and wing/tail configurations. Results of analysis of the propellant tank and body fairing sections were then applicable for either configuration. Definition of the aerodynamic surfaces requirements was similar for both vehicles, although this was the major deviation between the two.

Body Cross Section - In the interest of minimizing the weight of the first stage configuration, it was decided that the propellant tanks should serve as the primary load carrying structure. Several configurations of the "tank-structural core" were considered, and from these, three basic candidates were selected as shown in Figure 3-4. These candidate tank configurations were evaluated on the basis of structural properties, cost, aero-thermo consideration and design complexity. Tank weight for the three configurations is equal. Weight per unit volume is independent of the number of lobes and reduces to a function of internal pressure, material density, and allowable stress ($\frac{Wt}{Vol} = \frac{2p\rho}{\sigma}$). For equal length tanks, the cross sectional areas are equal and the capability for carrying body axial loads is essentially equal. The cylindrical section, with a higher moment of inertia, is slightly more efficient for reacting body bending moments, and is only slightly less costly than the siamese configuration. The "siamese" design also provides relatively more lift (C_L) than the other configurations and thus a smaller W/SC_L. The design complexity of the siamese design is somewhat higher than the simpler cylinder shape for the attachment of thrust structure and the general integration of supporting and secondary structures around the primary core. Installation of equipment such as aerodynamic surfaces, control lines, propellant feed ducts, etc. may be accomplished in the "Vee" between tank lobes on the siamese tank body, whereas the circular tank requires special external body fairings to provide for this equipment.

The siamese design was selected as the baseline approach because of the lower W/SC_L and the secondary equipment installation capability. The basic body is symmetric in profile to accommodate parallel tanks. A slight forward ramp and aft boattail would impact some positive zero lift pitching moment which is desirable

CARRIER STRUCTURAL CORE EVALUATION

RELATIVE QUANTITIES / TANK* CONFIGURATIONS	STRUCTURES		AERO-THERMO	DESIGN	COST
	RELATIVE WEIGHT	RELATIVE I	RELATIVE W/SC_L	RELATIVE COM-PLEXITY	RELATIVE COST
	1	1	1.0	1.0	1
 √	1	0.6	0.61	1.1	1.1
	1	0.6	0.82	1.2	1.2

* EQUAL CROSS SECTIONAL AREAS

ILRVS-16

Figure 3-4

from trim considerations. The body nose radius was therefore raised above the position shown in Figure 3-2.

Clipped Delta Wing Characteristics - A thick wing provides useful volume for cruise engine installations, landing gears and cruise fuel. Furthermore, it yields fairly large leading edge radii and correspondingly lower leading edge temperatures. A wing with 15% thickness was selected, influenced by installation requirements for turbo-fan cruise engines.

Hypersonic directional stability was initially being provided with aft-body flare, and high wing location was then selected as the baseline approach. Subsequent configuration analysis eliminated the use of aft body flare. It then appeared that a low wing installation had considerable merit with deployable equipment installed in the wing. The landing gear struts do not have to be as long, for example, and landing load induced moments can be reduced. The selected wing then had a low position and incorporated a 15° dihedral to provide the required hypersonic directional stability at high angles of attack. It includes large leading edge radius to provide hypersonic directional stability at low angles of attack and reduced heating rates.

A negative camber wing would improve the zero lift pitching moment characteristics and was shown in Figure 3-2 for the clipped delta concept. The negative camber low wing, however, introduces shock wave heating due to the discontinuity between wing and body. The wing was then replaced with a positive camber airfoil to provide a smoother wing-to-body transition. A fairing was also incorporated to provide a large fillet radius between wing leading edge and body.

The dorsal fin sweep angle was increased to enhance subsonic and supersonic directional stability by moving the center of pressure aft and to improve launch heating characteristics. These modifications were incorporated on the candidate carrier to provide a baseline for subsequent analysis.

Wing/Tail Characteristics - The considerations that resulted in the selection of the wing for the clipped delta concept are basically applicable to the wing/tail concept. One exception was that the wing/tail concept reduces the need for positive zero-lift pitching moment characteristics in the wing. The elimination of body flare, however, necessitates negative camber or other modification to obtain the required control characteristics improvement. The flared body at the aft end might also have been used for engine and landing gear installation so a thick wing might be incorporated for these provisions.

A low wing with a 15% thick chord provides less true thickness than the clipped delta wing due to the shorter chord. Landing engines contained in the wing mold-line then have a smaller diameter, thus lower thrust per engine and more engines would be required. A positive camber low wing will maintain a smooth wing-to-body transition, and hypersonic control capability may be provided with a tail surface dihedral. These modifications were also to be incorporated to provide an alternate baseline, however, the selected baseline included only the clipped delta configuration as described in Section 3.2.1.

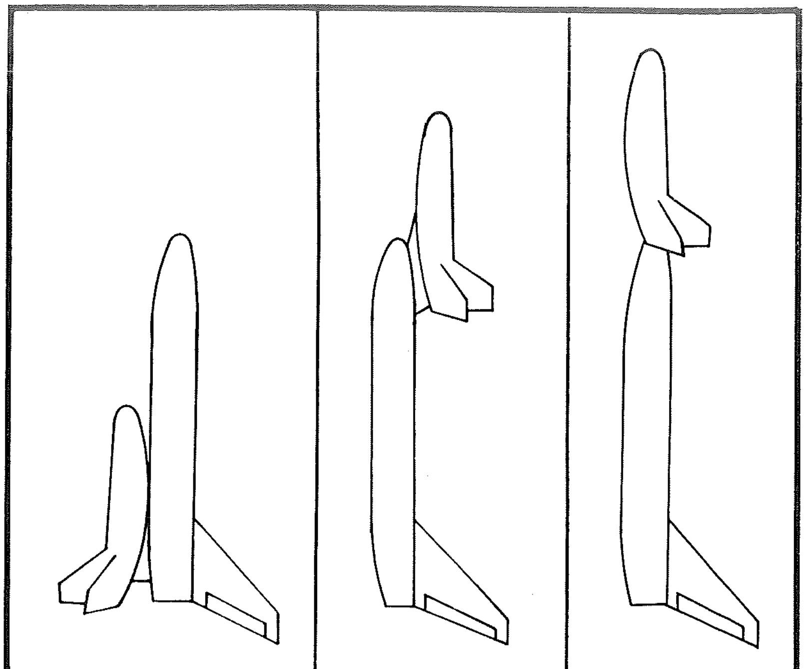
3.1.4 Candidate Launch Configurations - Various candidates for a launch configuration are depicted in Figure 3-5. The carrier includes the modifications discussed in the previous paragraphs. The nominal condition utilized during the early part of the study is represented by the first configuration (on the left). The stage mating is modified from the concept selected earlier in order to minimize the separation problems and carrier structural complexity. The aft location of the orbiter was basically required because the orbiter engines were to be operating at reduced thrust throughout the launch phase and at full thrust for checkout prior to launch. This requirement was subsequently removed, so arrangements such as the other two might be considered. The second configuration may also permit orbiter engine idle operation, however, a plume impingement study would be required to determine detrimental effects. The third configuration eliminates the capability of operating orbiter engines, with the engines near vehicle centerline.

The engine gimbal requirement of the carrier is a function of c.g. offset between the two vehicles. Total engine gimbal is referenced from the longitudinal axis of the carrier and consists of a combination of c.g. travel and launch dispersions. This requirement represents the highest penalty for the first configuration, and reduces in magnitude across the spectrum of configurations.

Separation complexity is greatest for the configuration where one vehicle lower surface translates longitudinally with respect to the other. This effect is offset, however, by the increasing over-all length as the launch configuration approaches the pure tandem arrangement. The greater lengths represent penalties in terms of ground support and servicing equipment. The lowest location for the orbiter permits a more accessible cargo bay for loading and servicing. It also represents the maximum potential for personnel accessibility. Launch pad escape from an emergency condition for instance, might be more readily accomplished with the crew and passengers located closer to the ground.

3.2 Vehicle Sizing Analysis - Vehicle sizing for both stages consisted of definition of the vehicle geometric properties and determination of the lengths required to perform specified missions. The task of defining geometric properties was initially performed on a non-dimensional basis. The factors defining vehicle

CANDIDATE LAUNCH CONFIGURATIONS



ORBITER BURN	YES	IDLE	NO
ENGINE GIMBAL REQUIRED	HIGH	MODERATE	LOW
SEPARATION COMPLEXITY	HIGH	LOW	LOW
LAUNCH LENGTH	~ 200' FT	~250 FT	300 + FT
ORBITER ACCESSIBILITY	GOOD	POOR	POOR

Figure 3-5

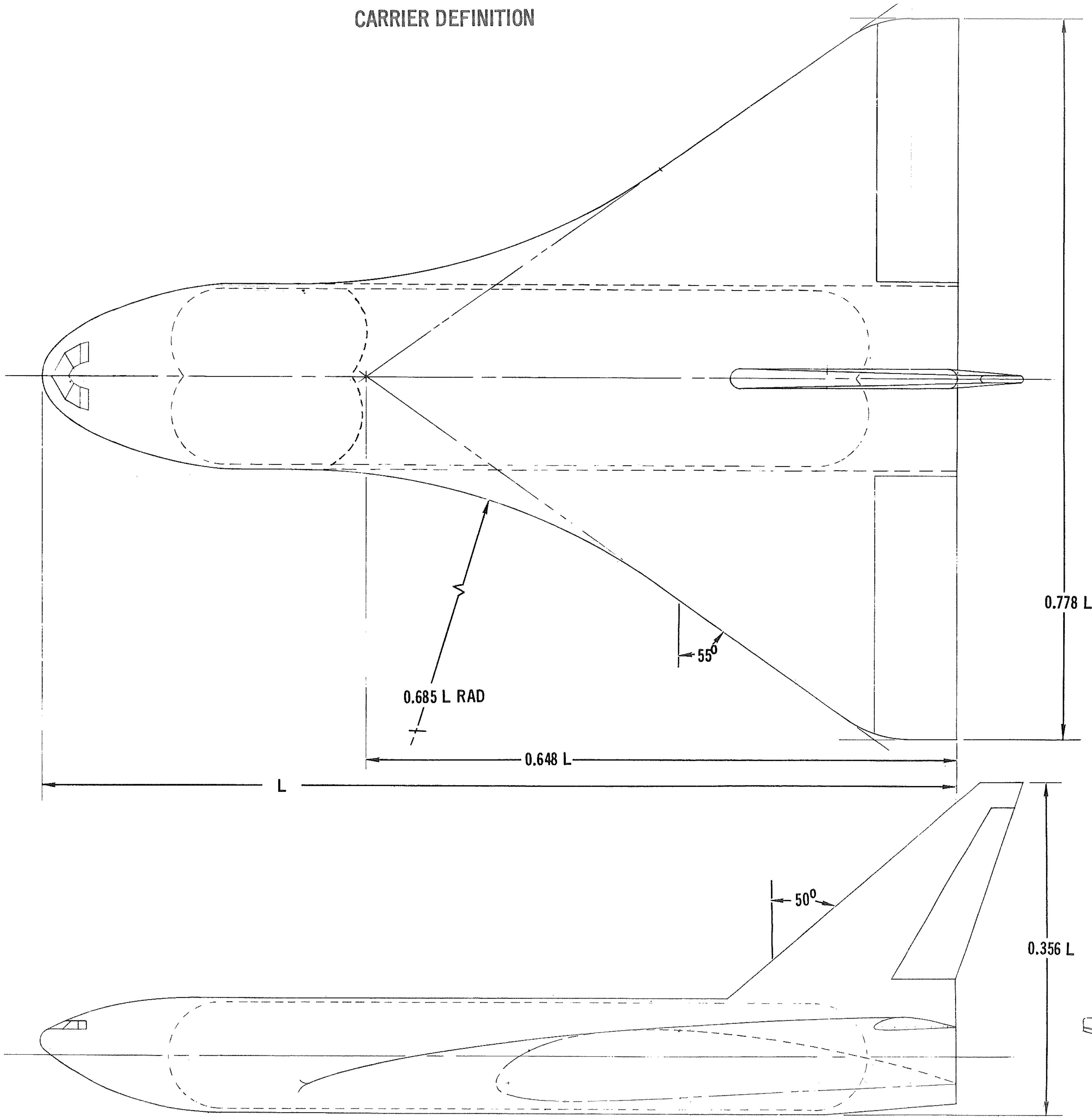
properties were then converted to values as a function of length. The capability of various vehicles for containing constrained cargo and launch propellant was determined. These values were also converted to a graphic form to provide a range of payload and performance capabilities. This analysis was conducted to provide a tool for rapid inputs to a computer program. The outputs of this program then provided an optimum vehicle size for each payload of interest, and a configuration for which the detail design analysis was conducted.

3.2.1 Carrier Sizing Analysis - The first stage vehicle selected as a baseline for the sizing analysis is illustrated in Figure 3-6. The clipped delta configuration shown was selected, with NASA-LRC concurrence. The selection of one carrier configuration at this point permitted a more detailed analysis for sizing and design definition. A continuation of both the clipped delta and wing/tail concepts would have diluted the level of effort on both vehicle stages. The two carrier concepts also are very similar, and the sizing results would be nearly the same. A major portion of the detail design definition would additionally be directly applicable to either of the first stage configurations.

The vehicle area ratios shown in Figure 3-6 were determined from a layout for an interim 205 ft. long vehicle. The scaled areas were non-dimensionalized by dividing by the square of the length to arrive at the values presented. Wetted area of the body was calculated by the method represented in Figure 3-7. The scaled perimeter at various body stations was divided by scale length to derive the P/L parameter. These values were then plotted to provide a graphic presentation of the equation for perimeter as a function of length. The area under this curve was then integrated to produce a value for total body wetted area.

Wetted areas for aerodynamic surfaces were assumed to be two times the value for projected area. This introduces a very small error (<1%) for thin sections such as the vertical tail, and an error of approximately 1% for the thicker wing section. The value for volume was determined in a manner similar to body wetted area by plotting cross-sectional area vs. length as shown in Figure 3-8 and integrating the area under the curve.

CARRIER DEFINITION



WING DATA – NACA 4415 AIRFOIL

AREA PER WING (INCL FLAP)	$0.0895 L^2$
SPAN	0.778 L
ASPECT RATIO	1.9
TAPER RATIO	0.18
THICKNESS RATIO	0.15
ELEVON AREA	$0.0254 L^2$

TAIL DATA

AREA	$0.0334 L^2$
SPAN	$0.221 L^2$
ASPECT RATIO	1.63
THICKNESS RATIO	0.0815
TAPER RATIO	0.186
RUDDER AREA	$0.0089 L^2$

WETTED AREA

BODY	$0.512 L^2$	TOTAL
WING	$0.358 L^2$	$S_W = 0.89 L^2$
TAIL	$0.0668 L^2$	

PLANFORM AREA – $0.361 L^2$

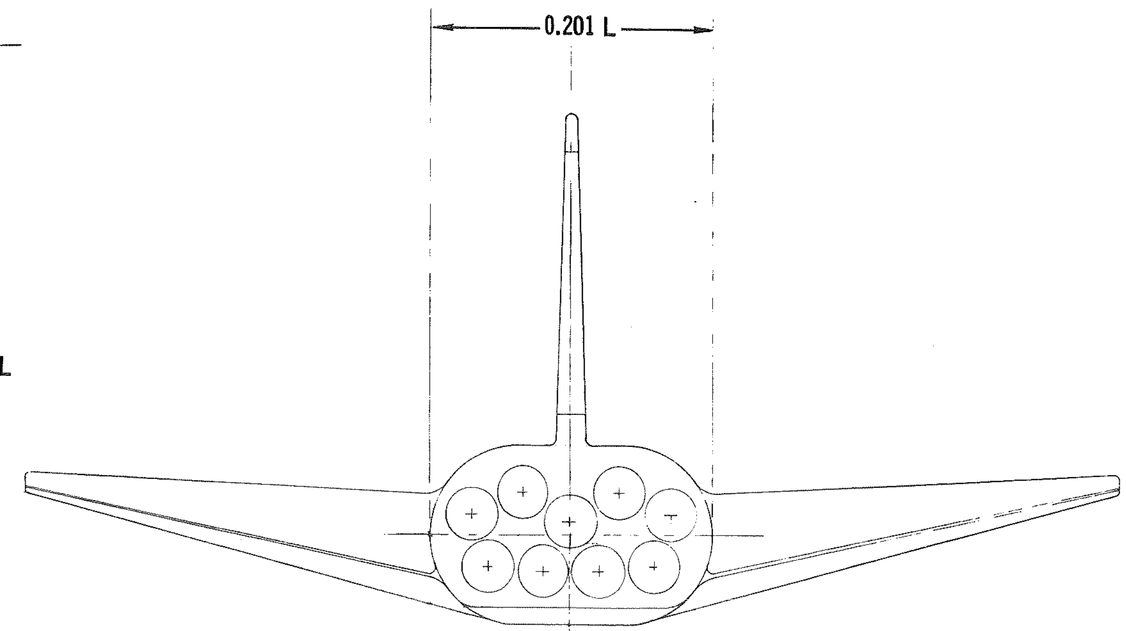


Figure 3–6

DERIVATION OF WETTED AREA (S_W) – CARRIER
(Body Only)

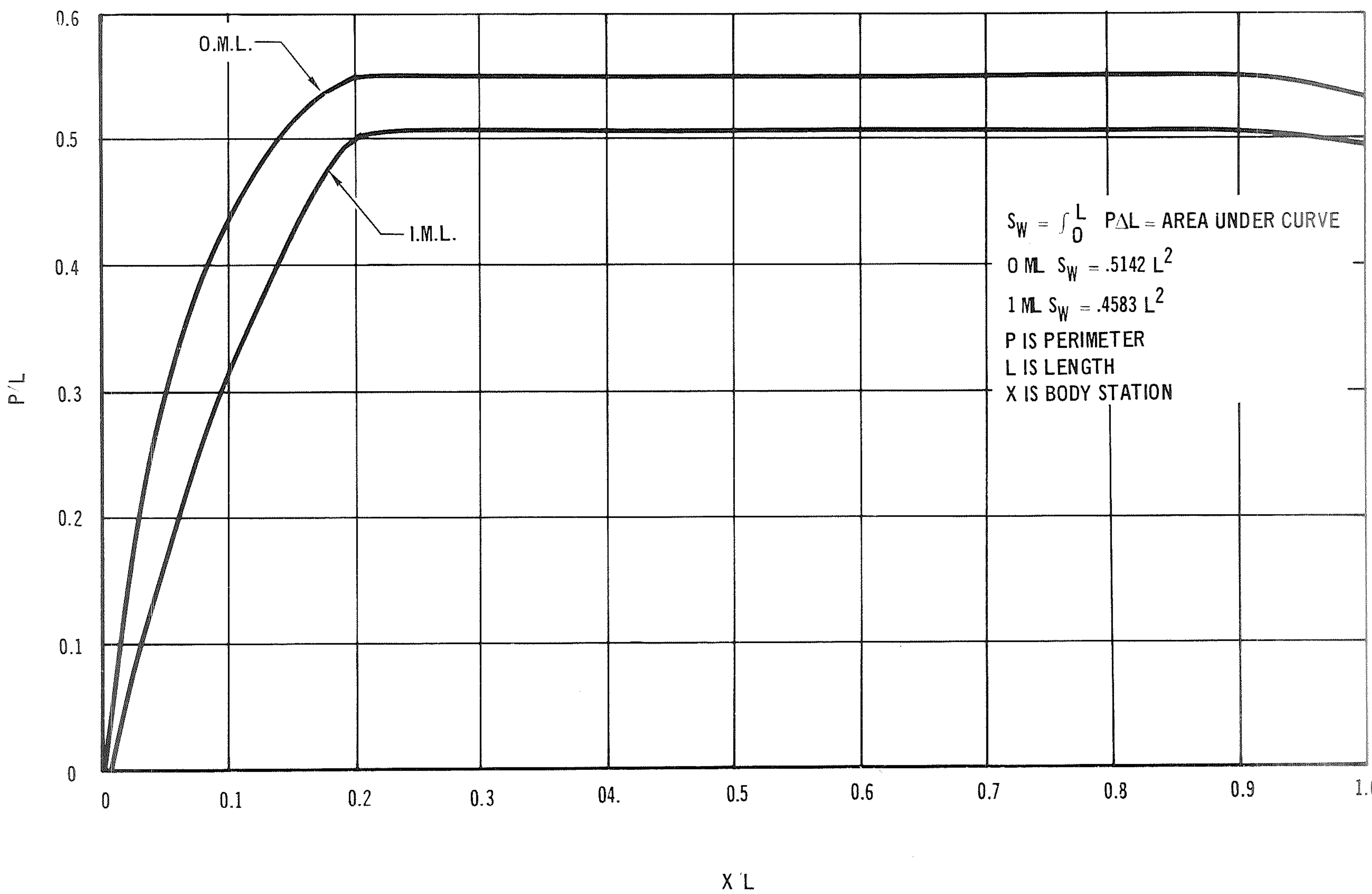


Figure 3-7

DERIVATION OF VOLUME (V) – CARRIER
(Body Only)

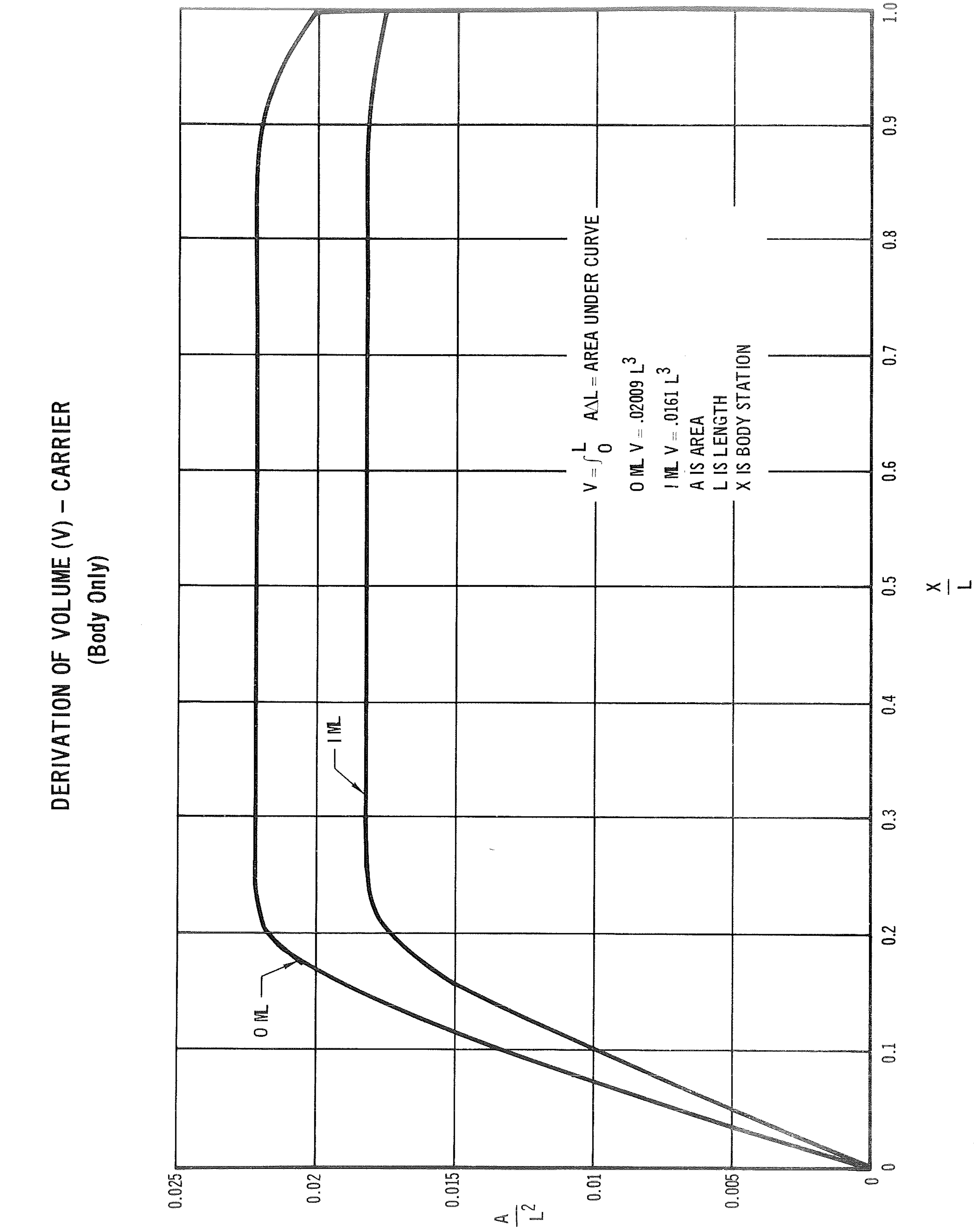


Figure 3–8

The factors determined by the analysis described above were converted to true values for vehicle length from 200 to 250 ft. These were then presented graphically as seen in Figures 3-9, 3-10 and 3-11. The total wetted area shown in Figure 3-9 includes exposed body, wing and vertical tail but excludes base area. The wetted area for body includes that area represented by the wing and fairing root chord. Planform area includes both wings and body. The values shown in Figure 3-10 represent the true projected area of each aerodynamic surface. The value shown for wing is thus for one wing only.

The volumes shown in Figure 3-11 include only the vehicle body. Launch propellant volume (concept) was determined on the basis of the tank arrangement shown in Figure 3-6. This volume therefore includes a tank only in that portion of the vehicle where body sides are parallel. The propellant volume for baseline was increased over the concept shown, as described in Section 3.3. The values for this improvement are defined as Baseline Launch Propellant.

Wing volume was also derived to determine the potential capability for equipment or propellant installation. This volume was determined, for one wing, for the interim 205 ft. carrier and is illustrated in Figure 3-12. Useful volume is limited to the fixed wing and is shown as functions of span and vehicle length. The total useful volume of both wings represents approximately 27% of body outer mold line volume.

CARRIER AREAS
(Vehicle)

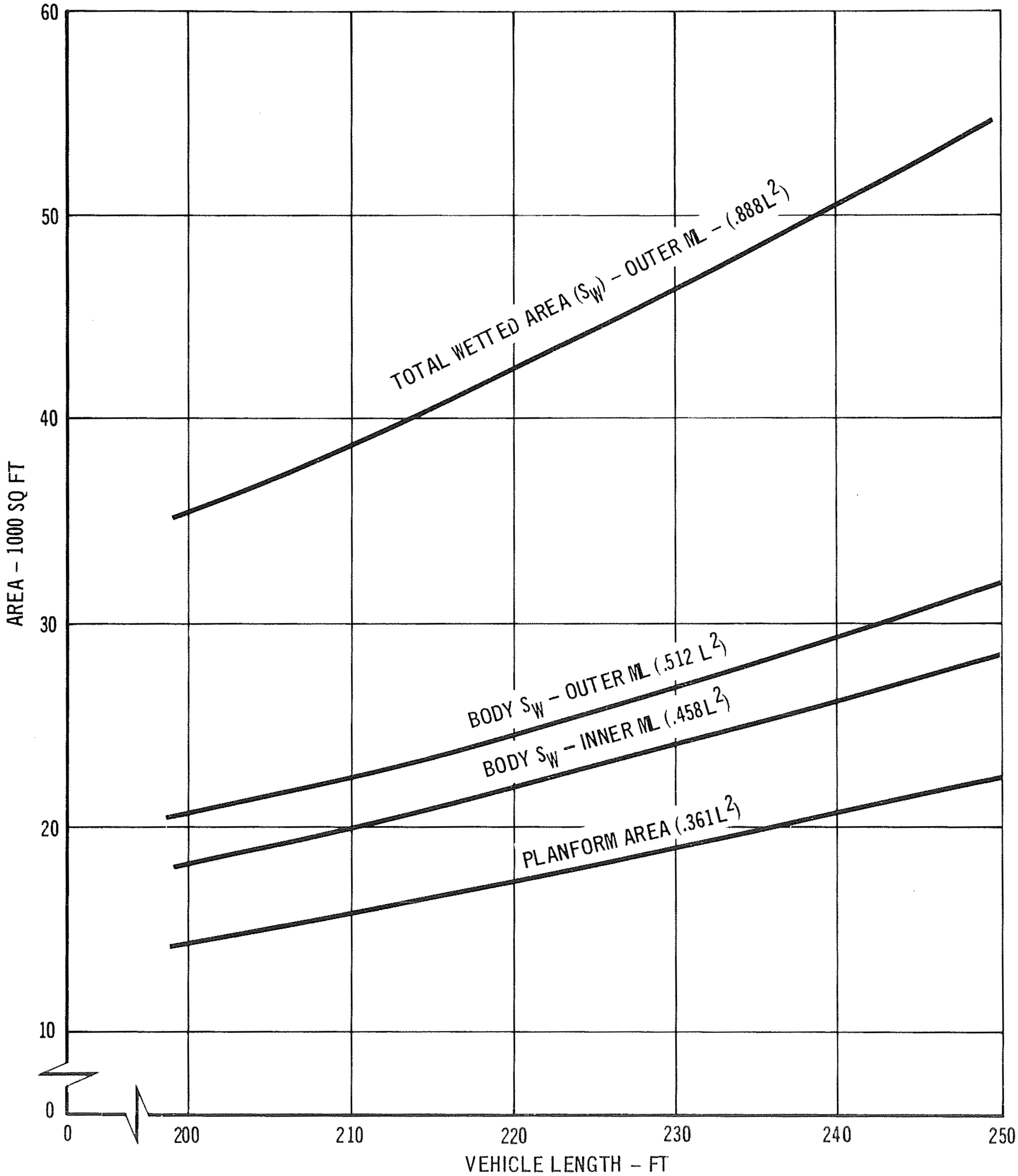


Figure 3-9

CARRIER AREAS
(Aero Surfaces)

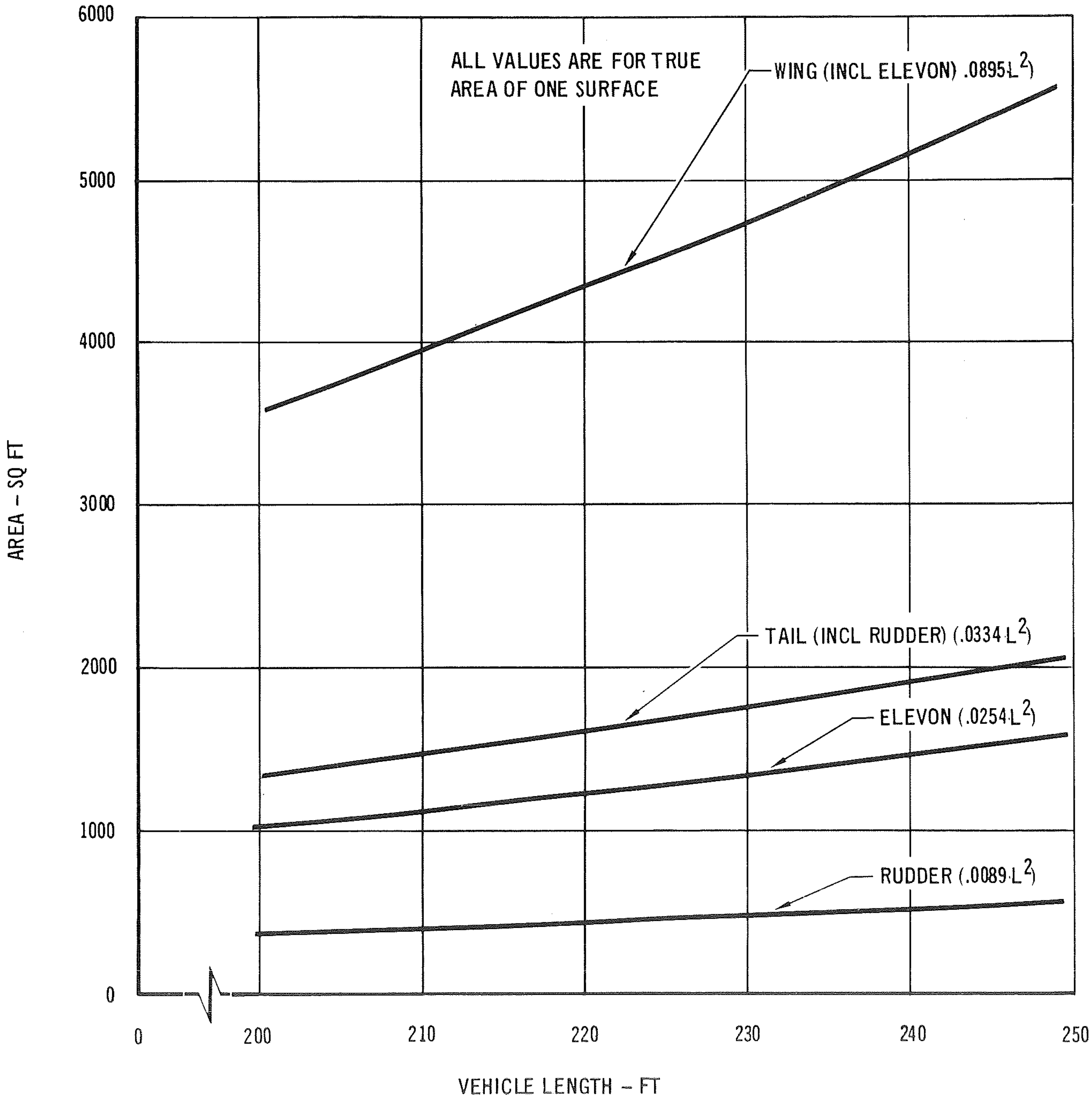


Figure 3-10

CARRIER VOLUMES

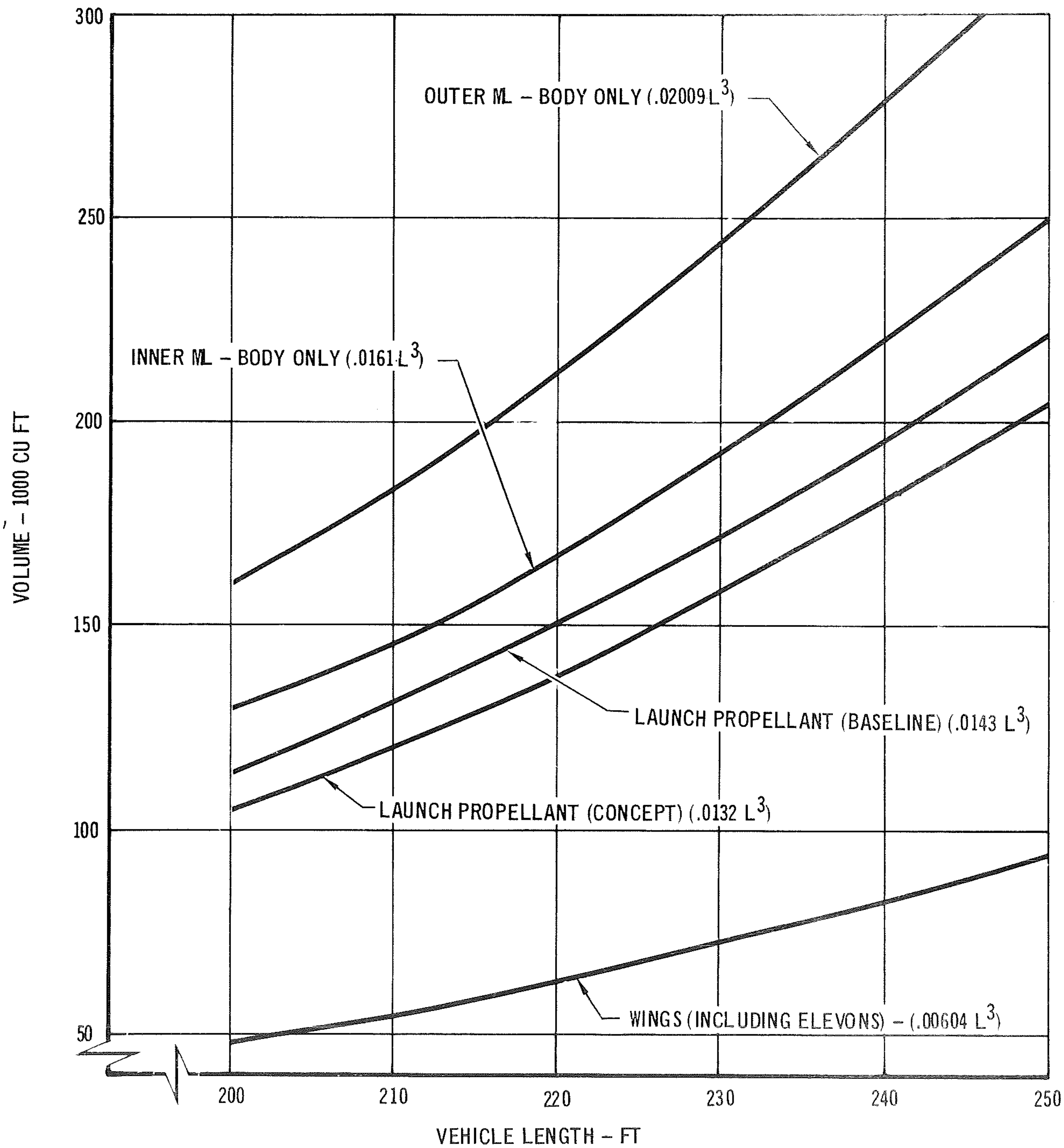
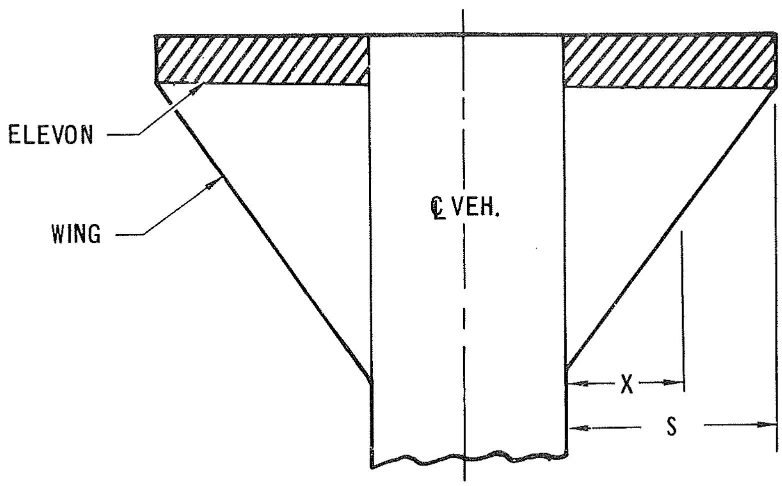


Figure 3-11

DERIVATION OF WING VOLUME – CARRIER



VALUES ARE FOR ONE WING

$V = \int_0^S A \Delta S$ = AREA UNDER CURVE

WING $V = .1130\ S^3 = .00270\ L^3$

ELEVON $V = .0137\ S^3 = .00032\ L^3$

TOTAL $V = .1267\ S^3 = .00302\ L^3$

S IS SPAN

A IS CROSS SECTIONAL AREA

X IS SPAN STATION

L IS VEHICLE LENGTH

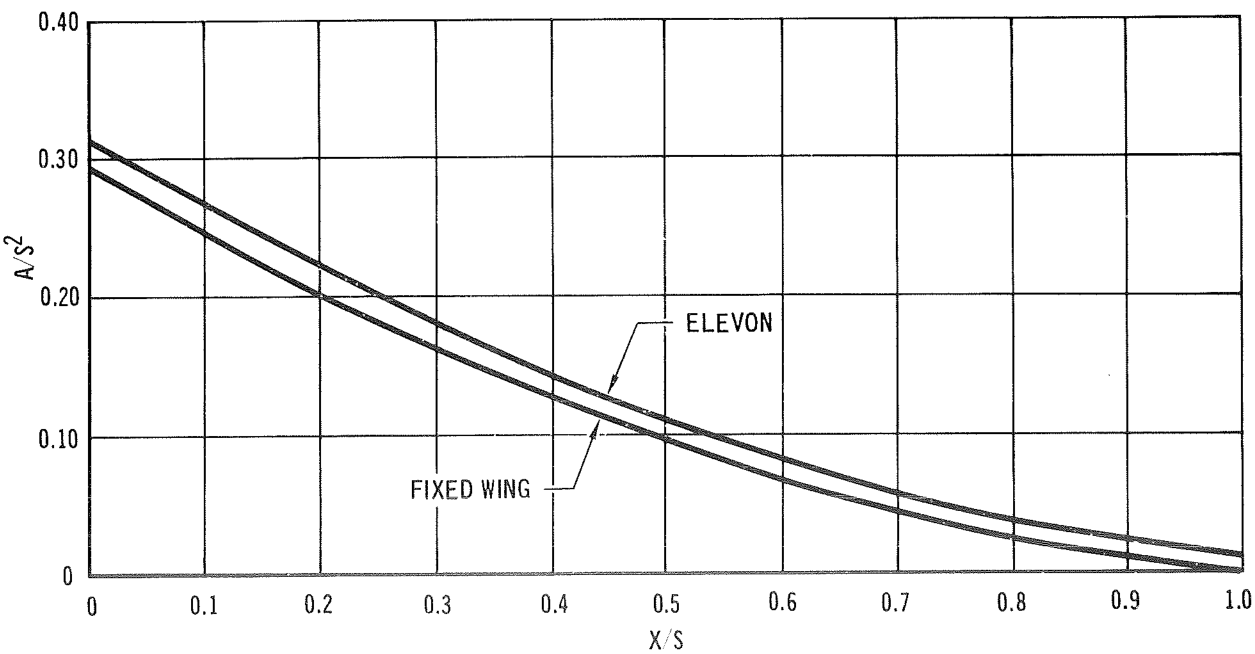


Figure 3–12

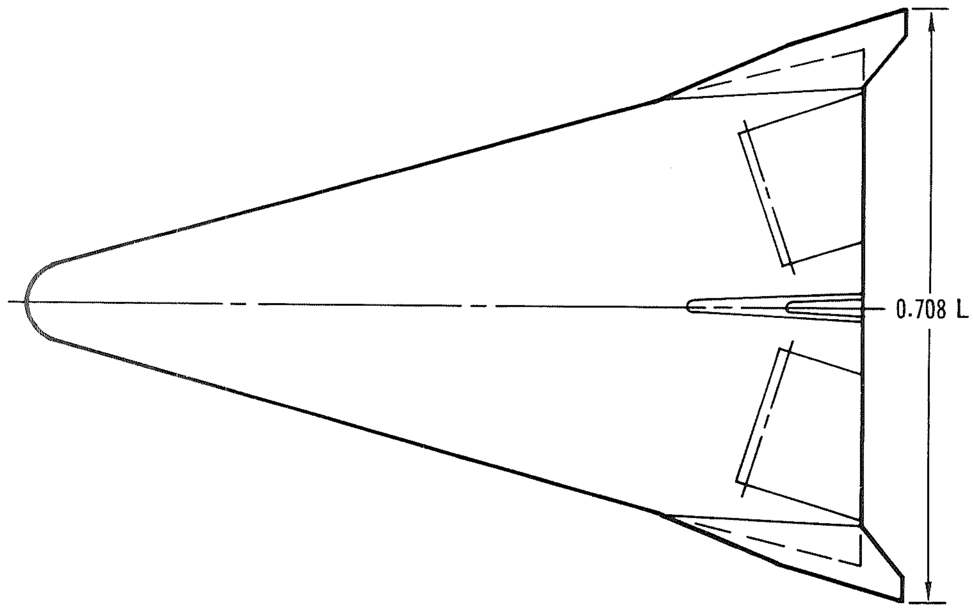
3.2.2 Orbiter Sizing - The orbiter is shown in Figure 3-13. This configuration is shown as a non-dimensional shape like the clipped delta carrier. Derivations for wetted area and volume are also similar and are presented in Figures 3-14 and 3-15 respectively. Vehicle areas and volumes were calculated for the spectrum of lengths from 100 to 150 ft. The results were plotted as shown in Figures 3-16, 3-17 and 3-18.

The body wetted areas shown in Figure 3-16 include both upper and lower elevons plus the area covered by the root chords of tip fins and centerline fin. These values do not include the wetted area of these fins. The planform area is total projected area, including that portion of the tip fins which is visible in the plan view. Areas for the aerodynamic surfaces shown in Figure 3-17 are true view projected area of each specified surface. The body volumes shown in Figure 3-18 include the total vehicle body, excluding fins. Derivation of launch propellant volume is described in the succeeding paragraphs.

A sizing model was established to provide a means for satisfying the requirements imposed by payload geometric constraints. This model is illustrated in Figure 3-19. The geometric center of the payload container was located longitudinally at the vehicle c.g. One foot was added at either end and to the diameter of the payload container. This envelope was used to define a payload bay which included provisions for structural support and deployment mechanisms plus installation and deployment clearances. The orbiter shape was scaled as required and superimposed on this envelope so that the inner mold line became coincident with the aft end of the payload bay. A check for installation clearance in the lateral direction revealed that the longitudinal constraint was the critical factor. The proper scaling of the orbiter then yielded the required minimum length vehicle.

This analysis was performed for the baseline payload (15 ft. dia. x 30 ft. long) as well as for several dispersions from the baseline. The payload definitions as well as the required orbiter lengths are listed in Table 3-2.

ORBITER DEFINITION



ORBITER AERO SURFACE AREAS	
VERTICAL CENTER FIN (INCLUDING RUDDERS)	$0.0346\ L^2$
SIDE FAN (INCLUDING FLAP)	$0.0303\ L^2$
LOWER ELEVON	$0.0203\ L^2$
UPPER ELEVON	$0.0198\ L^2$
RUDDER	$0.0092\ L^2$
TIP FIN FLAP (OUTER)	$0.0081\ L^2$
TIP FIN FLAP (INNER)	$0.0058\ L^2$
WETTED AREA	
OUTER ML (BODY ONLY)	$0.934\ L^2$
INNER ML (BODY ONLY)	$0.840\ L^2$
PLANFORM AREA	$0.365\ L^2$

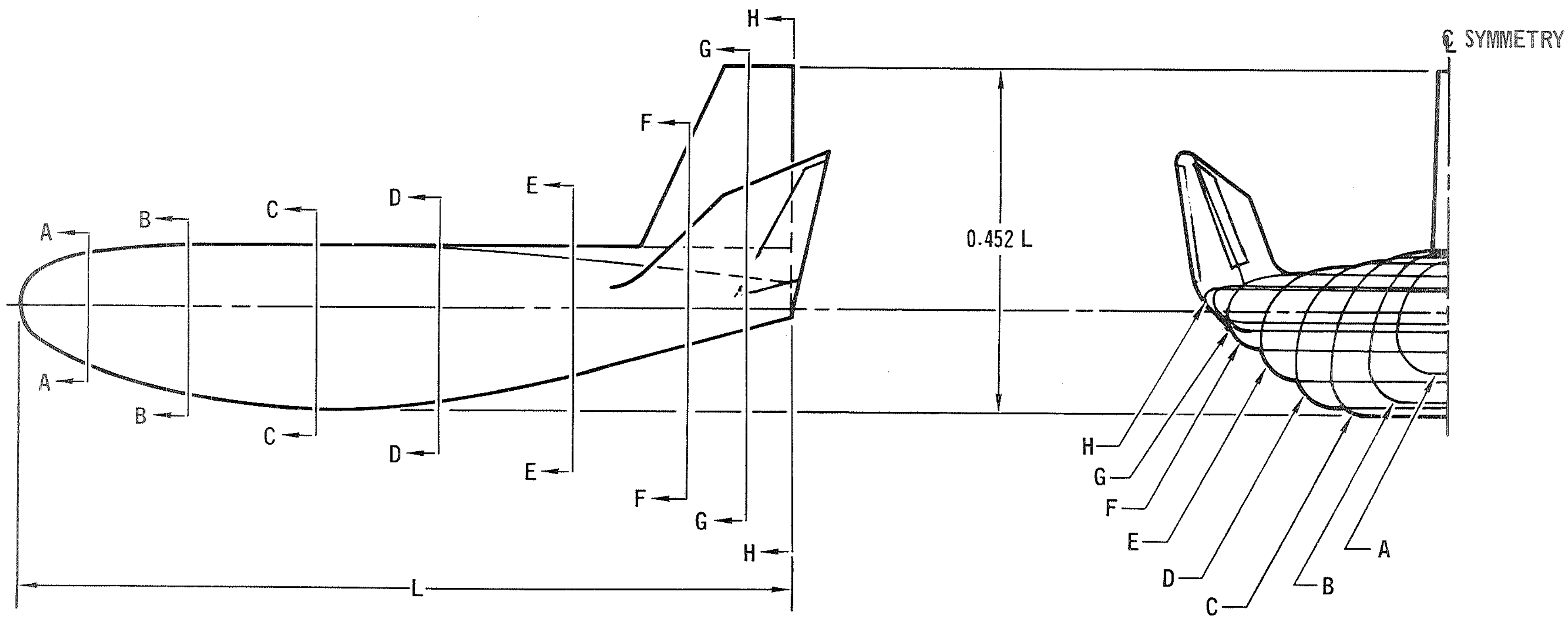


Figure 3-13

DERIVATION OF WETTED AREA (S_W) – ORBITER
(Body Only)

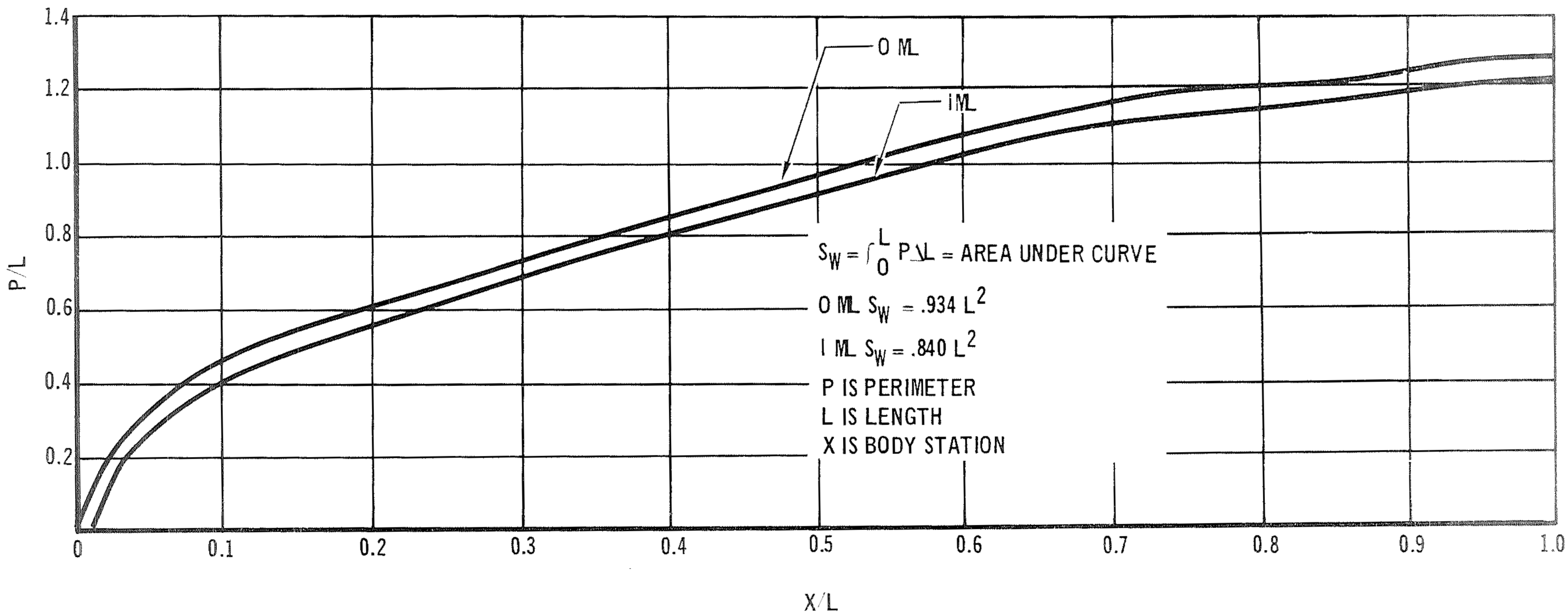


Figure 3-14

Integral Launch and Reentry Vehicle System

DERIVATION OF VOLUME (V) – ORBITER
(Body Only)

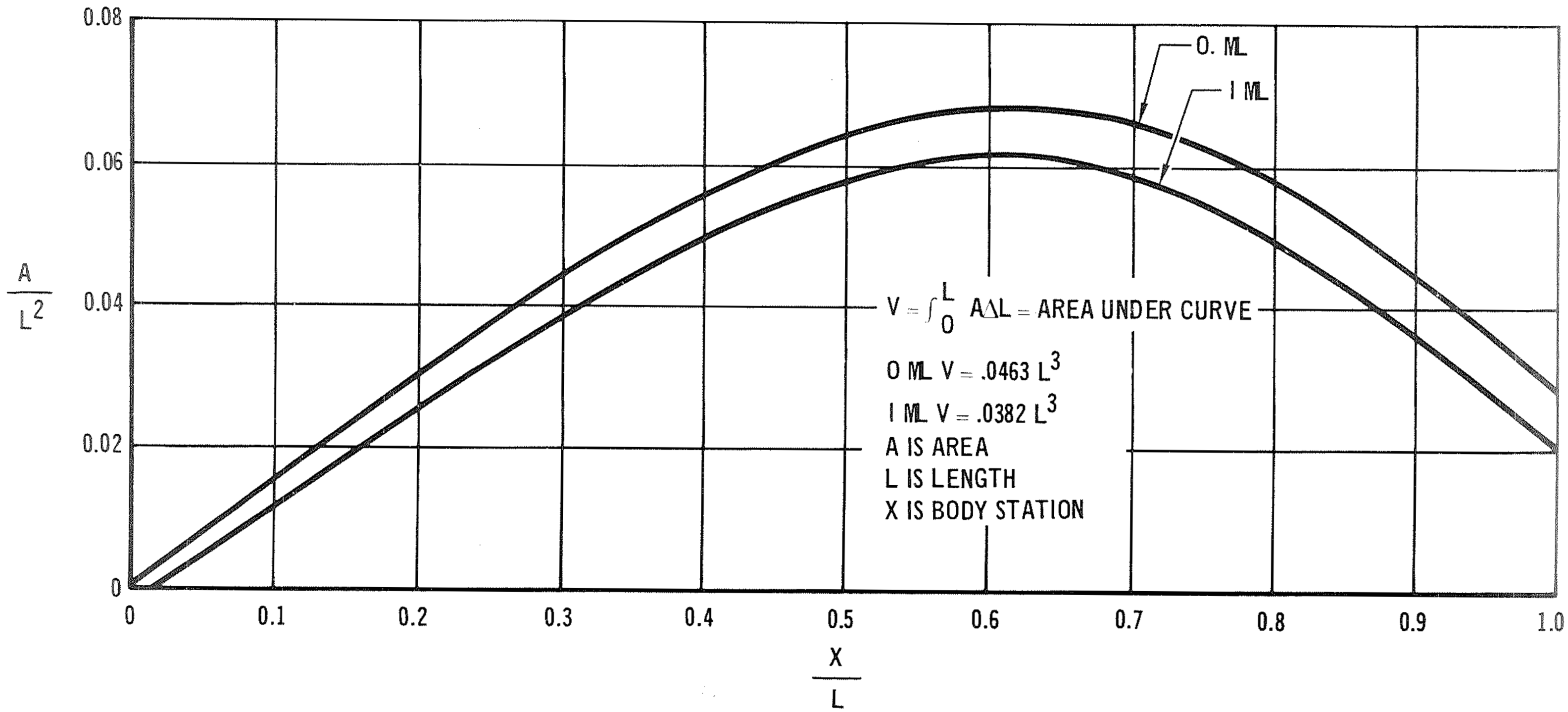


Figure 3–15

ORBITER AREAS
(Vehicle)

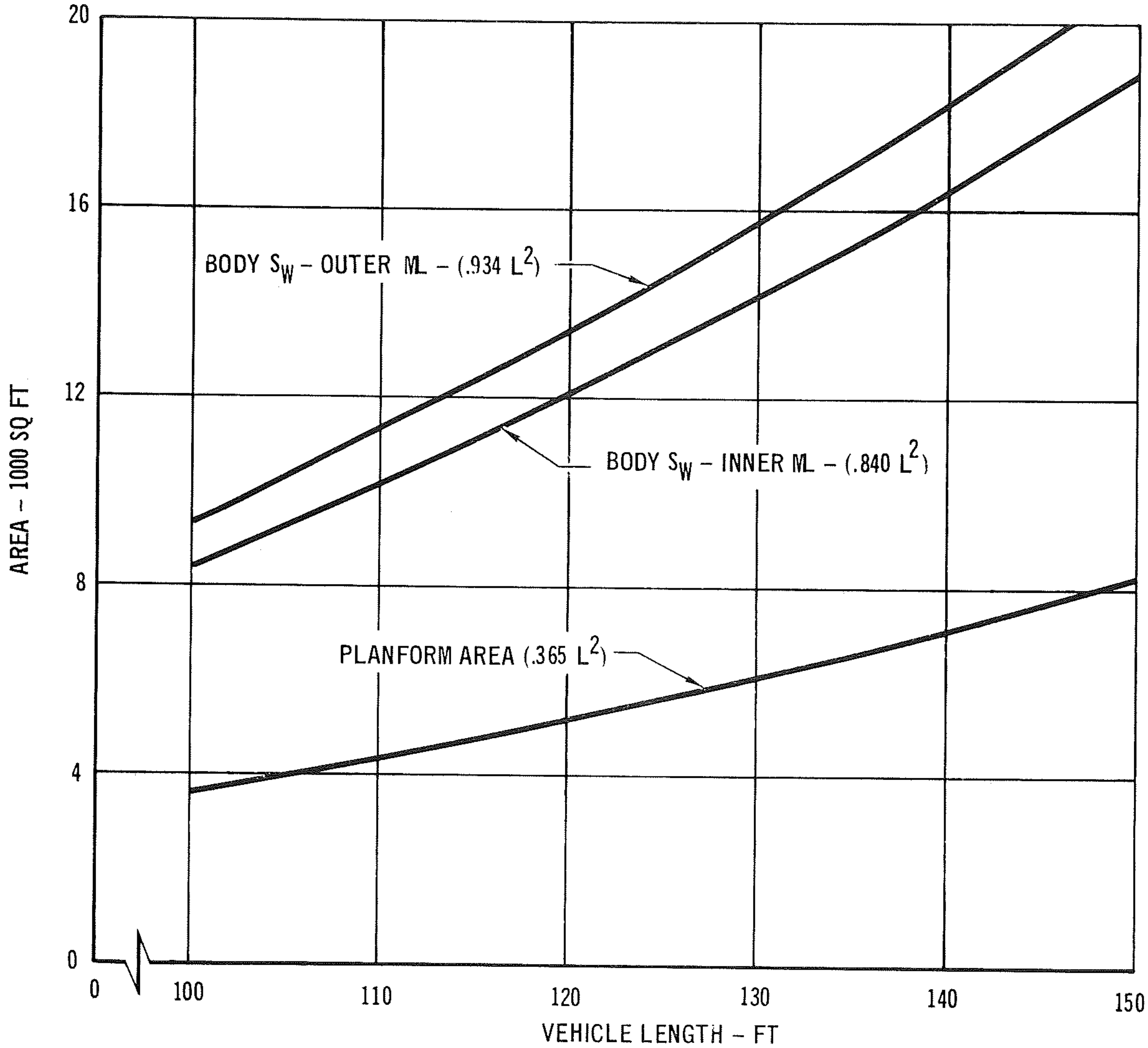


Figure 3–16

ORBITER AREAS
(Aero Surfaces)

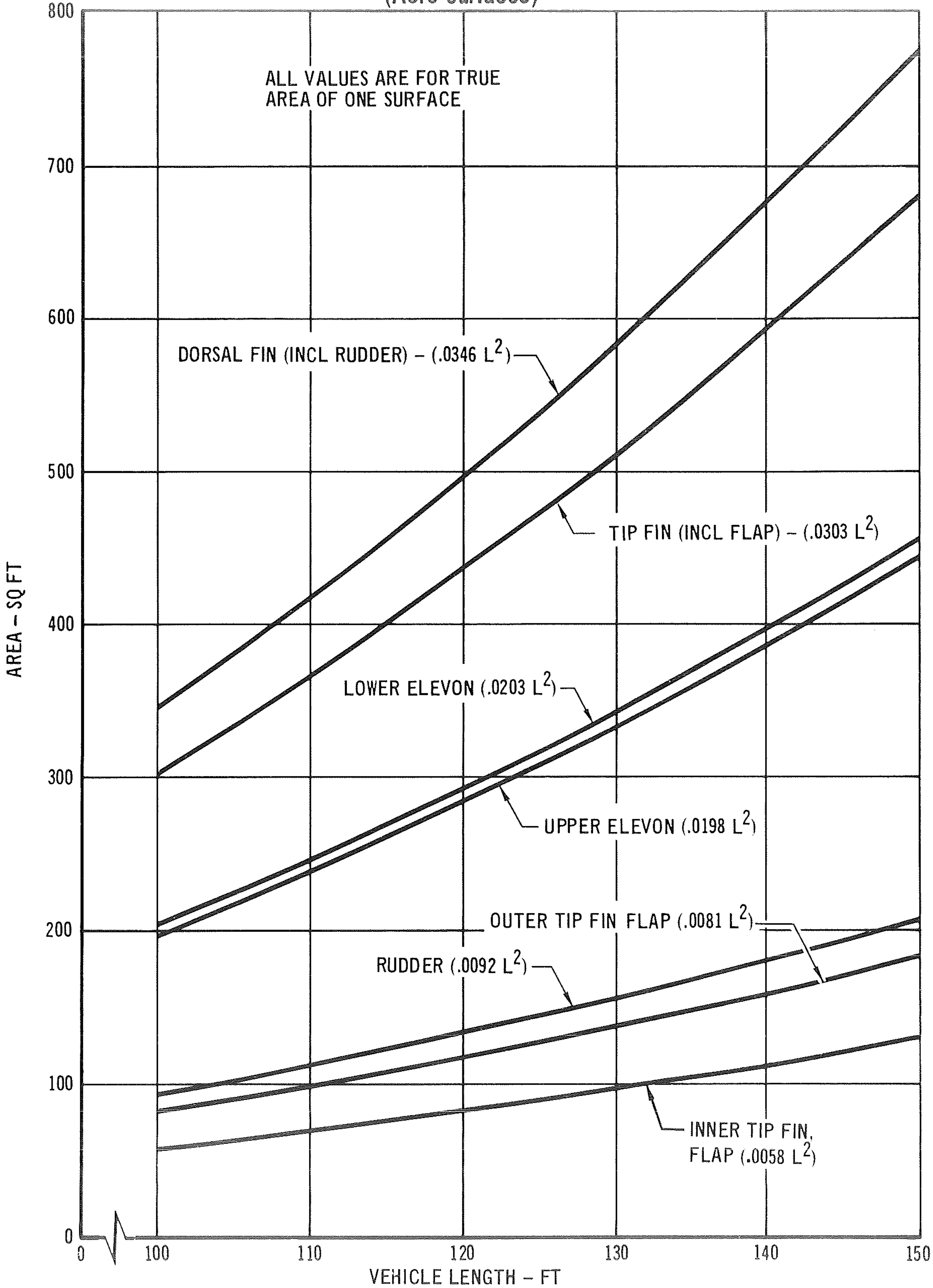


Figure 3-17

ORBITER VOLUMES
(15' Dia – 30' L Payload Container)

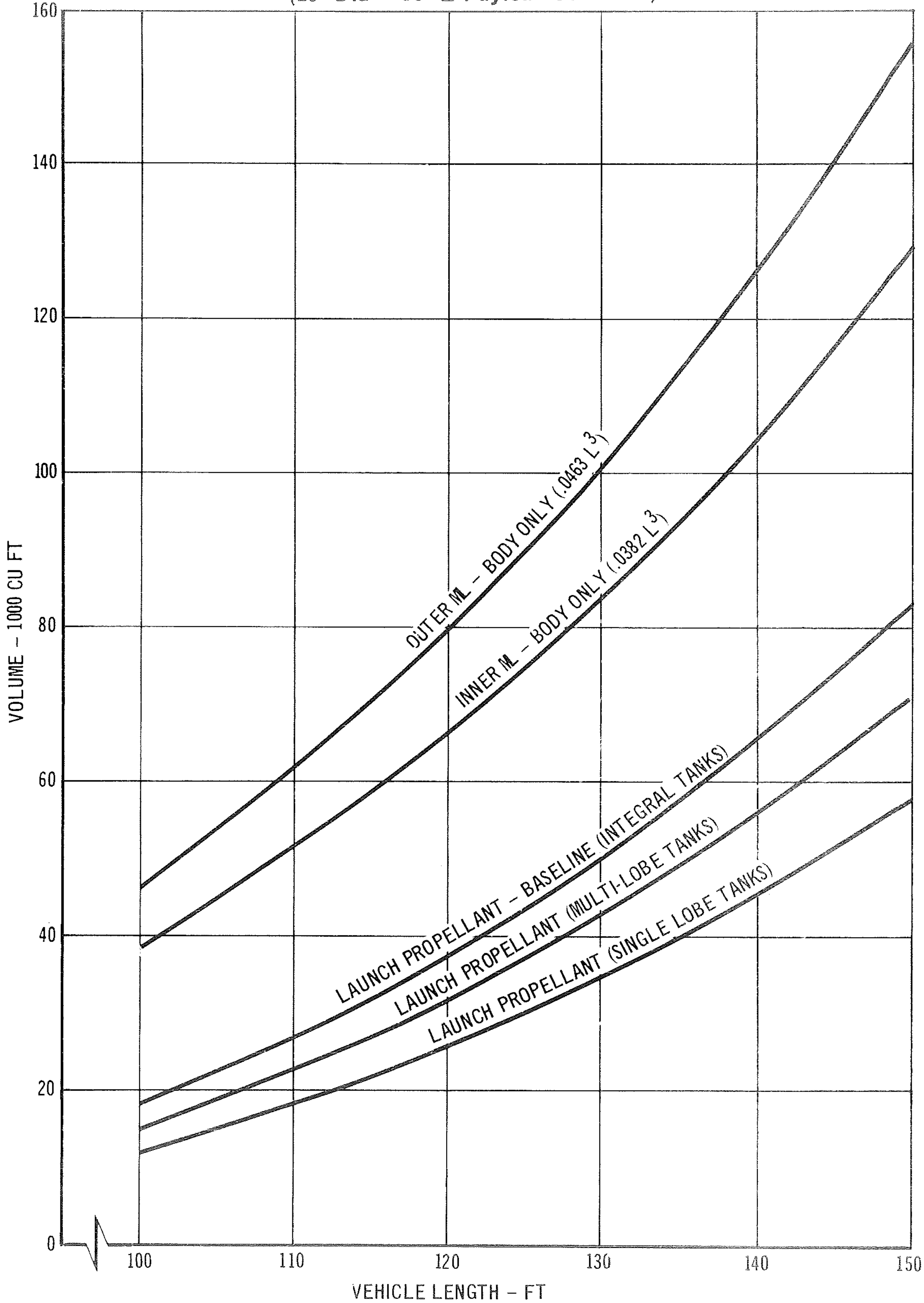


Figure 3-18

ORBITER SIZING MODEL

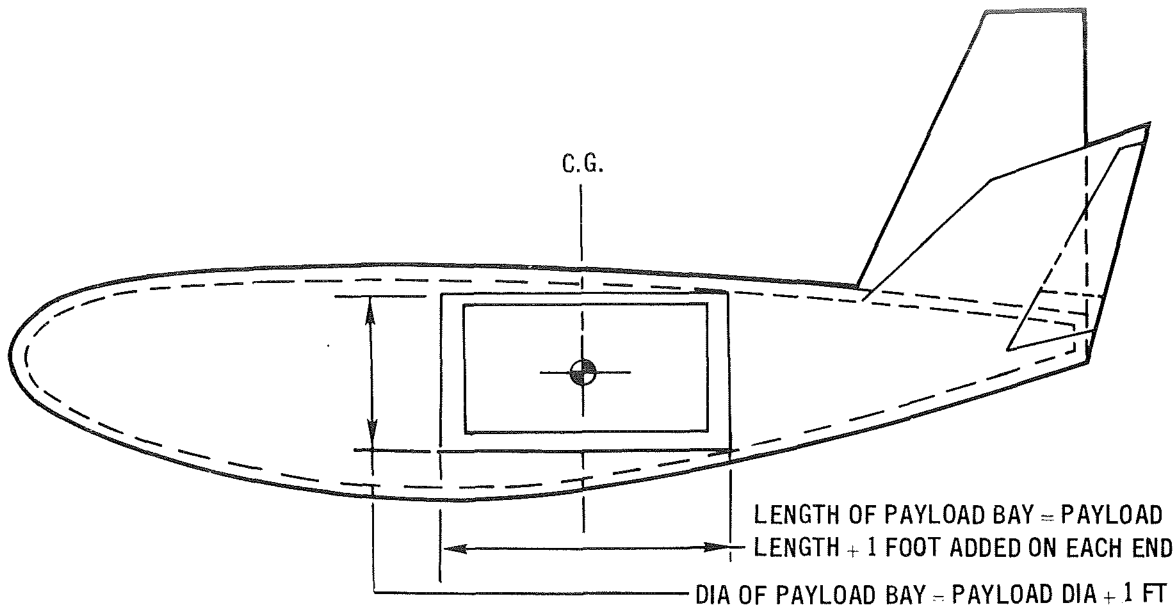


Figure 3-19

Table 3-2

REQUIRED ORBITER SIZE
(Constrained Payload)

Payload Definition		Orbiter Size
Dia. (ft.)	Length (ft.)	Length (ft.)
15	60	130
15	30	107
15	15	97
9	35	83
9	60	104
22	60	168
22	30	142

The values from Table 3-2 were then plotted with parameters of payload diameter, payload length and orbiter length. This is shown in Figure 3-20 and provides a method of determining orbiter size for any intermediate payload configurations of interest.

The volume available in the orbiter for storing boost propellant was determined for the baseline payload. A preliminary configuration layout, shown in Figure 3-21 was made for this purpose. This arrangement considered the application of propellant tanks independent of vehicle structure. This layout is intended to show only the maximum amount of pressure tank volume attainable for the 107 ft. long orbiter. The forward portion of the vehicle was reserved for crew, spacecraft systems and landing propulsion. The payload was installed as previously described and the remainder of the internal volume was allotted to boost propellants. The results of this layout determined that a total volume of approximately 17,000 cu-ft may be provided.

An analytical model was devised to permit extrapolation of this data point for a range of vehicle lengths. This model assumed a constant spacecraft volume for crew and subsystems of 1050 cu-ft. This value was extracted from previous parametric studies. Boost engine volume was assumed to approximate 1% of spacecraft volume. The spacecraft volume was derived for payload, an allowance for installation and deployment, landing gear, maneuvering propellant and miscellaneous equipment required in the aft portion of the spacecraft. The mathematical model then took the simplified form:

ORBITER MINIMUM LENGTHS
(Constrained Payload)

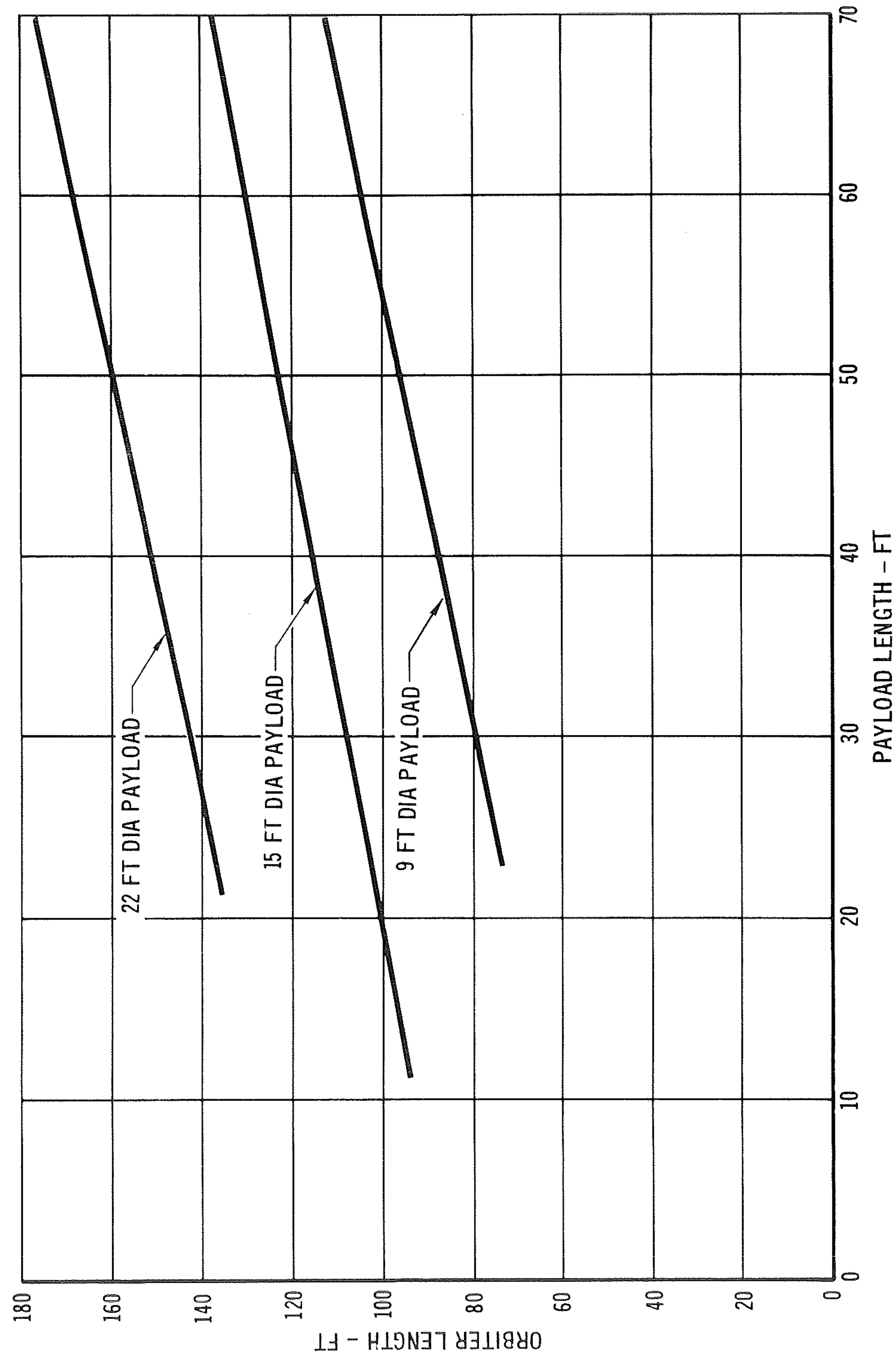


Figure 3-20

ORBITER BOOST PROPELLANT VOLUME
(Preliminary)

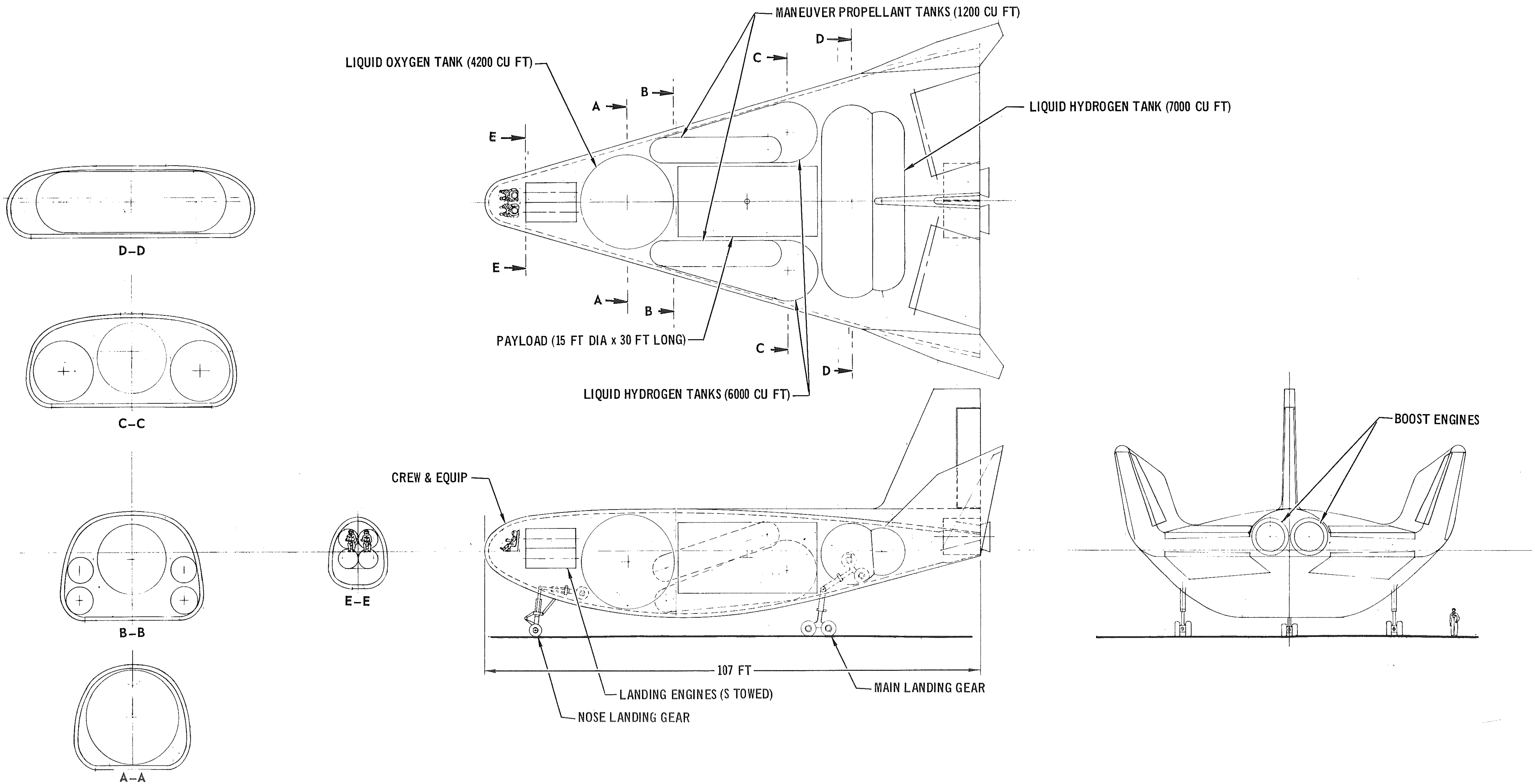


Figure 3–21

$V_{S/C} = V_K + V_E + V_C + V_P$, where;

$V_{S/C}$ = spacecraft internal volume

V_K = constant (1050 cu-ft)

V_E = spacecraft engine volume

V_C = spacecraft payload volume

V_P = spacecraft propellant volume

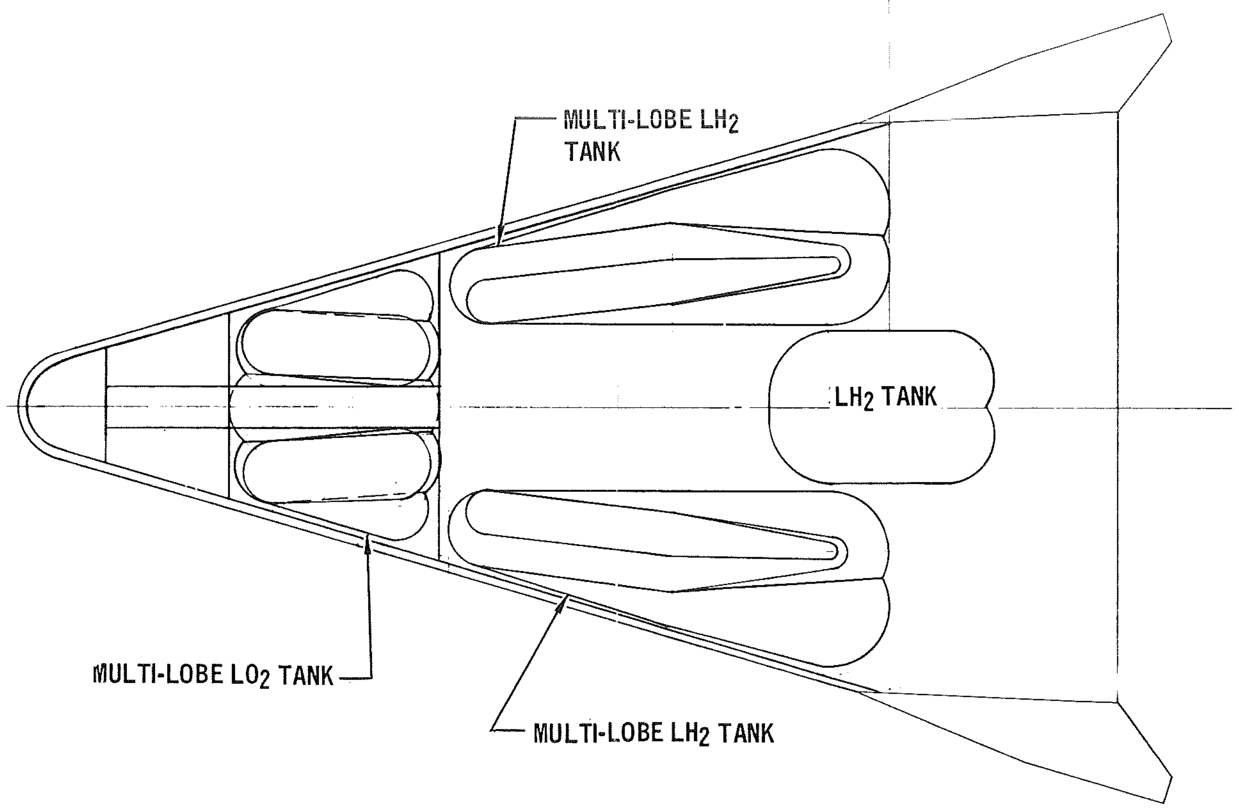
The above equation was solved for V_P. The actual propellant volume previously derived (17,000 cu-ft) was divided by V_P to define a packaging efficiency of 55%. This efficiency was applied to vehicle lengths up to 150 ft. and the results plotted on Figure 3-18 to produce the launch propellant volume curve.

The independent propellant tanks concept was utilized until mid-way through the two stage recoverable vehicle study. The effort at that time was redirected toward development of a concept to use more of the internal orbiter volume for propellant. The final result was a method of employing integral tanks, formed to the shape of the orbiter inner mold line. The attainable propellant volume, using integral tanks, was determined. This volume was provided as an input to the previously described analytical model to provide a packaging efficiency of 80%. The launch propellant for orbiters containing the baseline payload and integral tanks is also shown in Figure 3-18.

An intermediate propellant tank concept employs multi-lobe pressure vessels. A representative arrangement is shown in Figure 3-22. The tanks here retain a more optimum pressure vessel cross-section than the integral tanks, while attaining a better packaging efficiency than the independent tanks. This concept could easily use the tank walls as the vehicle primary skin, like the integral tanks. The packaging efficiency for the multi-lobe tanks is 68%, and the resulting propellant volumes are presented in Figure 3-18.

A typical cross-section through the orbiter, in the payload bay area, is shown in Figure 3-23. This section summarizes the packaging efficiencies of the three concepts and illustrates the cross-sectional area utilization. The selected concept for baseline definition employed the integral tanks.

ORBITER MULTI-LOBE BOOST PROPELLANT TANKS



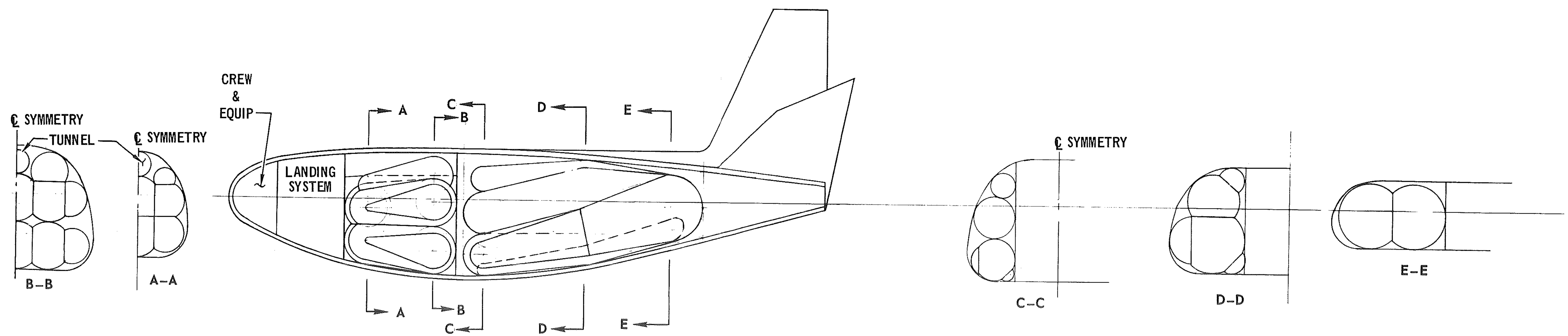
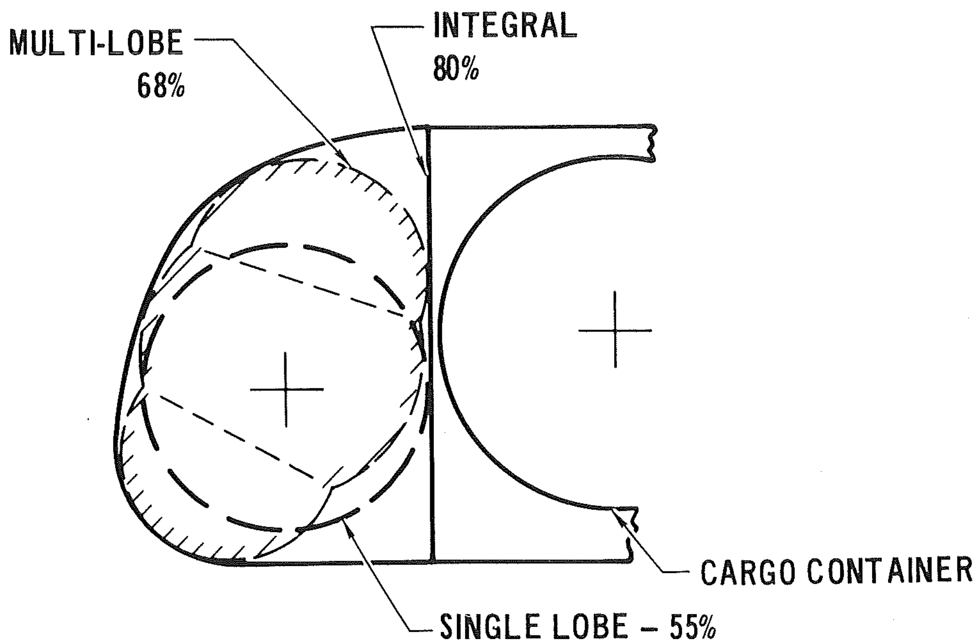


Figure 3–22

ORBITER TANK CONCEPTS
Volume Utilization Comparison



NOTE: SECTION AT 60% LENGTH

Figure 3–23

3.3 Baseline Configuration - The baseline lengths were determined from a computer program which provided weights, performance and vehicle size. The requirements for size were based on vehicle capabilities determined in the sizing analysis of Section 5. The resulting configurations and lengths were used during the remainder of the study for detail design definition.

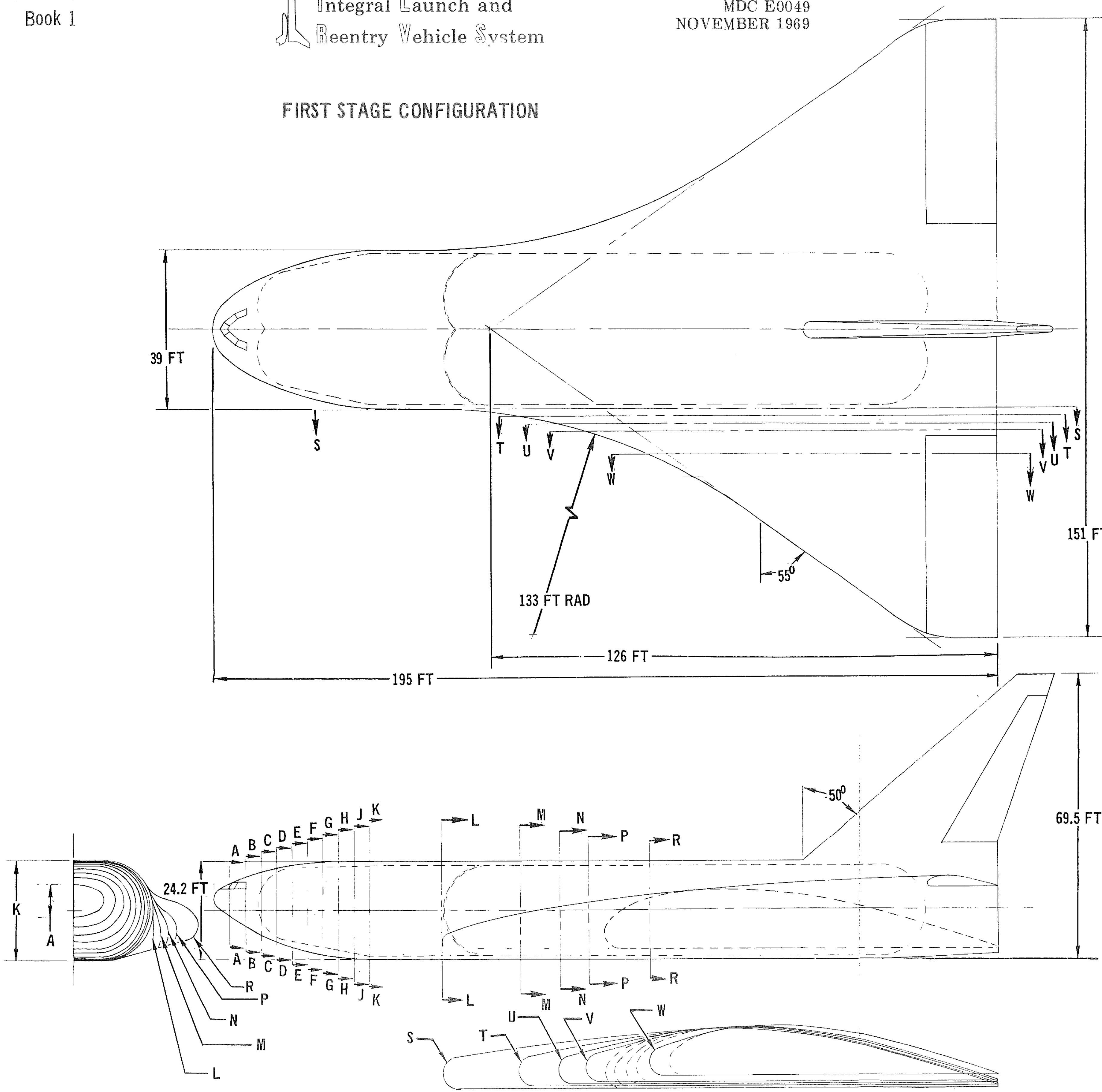
3.3.1 Carrier Configuration - The baseline 195 ft. long carrier configuration is shown in Figure 3-24. This configuration incorporates a few modifications from that shown in Figure 3-6. The bulkhead between oxygen and hydrogen tanks was reversed for propellant feed considerations as described in Section 7.1.2. The oxidizer forward dome was also extended farther forward in the vehicle. It was recognized that the maximum propellant capability reduces vehicle length, thus weight, and represents improved performance. The oxidizer tank forward end was therefore located just aft of the crew and equipment compartment, since landing engines and propellant were to be located in the wing. The vehicle external shape was not modified as the oxidizer tank extension is formed by two intersecting cones designed to fit inside moldline.

The NACA 4415 wing section was modified in the leading edge region near the wing root. A 3° incidence was used and the leading edge radius and lower surface forward ramp were modified to provide the wing-to-body fairing. The transition between wing and body then represents intersecting planes to provide "flat" lower surfaces. The 15% chord thickness was retained for cruise engine and equipment installation. The end view on Figure 3-24 consists of a series of body station cuts and defines the forward body transition and body-to-wing fairing. The aft end boat-tail was reduced slightly as a result of subsequent boost engine installation requirements. This change, however, did not represent an appreciable configuration modification.

3.3.2 Orbiter Configuration - The 107 ft. long orbiter baseline configuration is shown in Figure 3-25. The lines defining vehicle body and aerodynamic surface are unchanged from those shown in Figure 3-13 as they were rigidly constrained.

Boost engine installation requirements dictated a slight modification to the upper surface fairing at the base of the vehicle. The relatively small base area between elevons imposed a rather severe limitation on available thrust. The required engine diameters, from the performance calculations, were superimposed on the base of the vehicle as shown in Figure 3-26. This illustrated the need

FIRST STAGE CONFIGURATION



WING DATA – MODIFIED NACA 4415

AREA PER WING	– 3400 FT^2 (EXPOSED PER WING) INCLUDING ELEVON
SPAN	– 151 FT
ASPECT RATIO	– 1.9 (THEORETICAL)
TAPER RATIO	– 0.18
THICKNESS RATIO	– 0.15
ELEVON AREA	– 960 FT^2 (PER WING)

DORSAL FIN DATA

AREA	– 1270 FT^2 (INCLUDING RUDDER)
SPAN	– 44.2 FT
ASPECT RATIO	– 1.63
TAPER RATIO	– 0.186
THICKNESS RATIO	– 0.0815
RUDDER AREA	– 319 FT^2

WETTED AREA (S_W)

BODY	– 19,500 FT^2
WING	– 13,600 FT^2
DORSAL FIN	– 2540 FT^2

TOTAL S_W = 33,800 FT^2

PLANFORM AREA – 13,700 FT^2

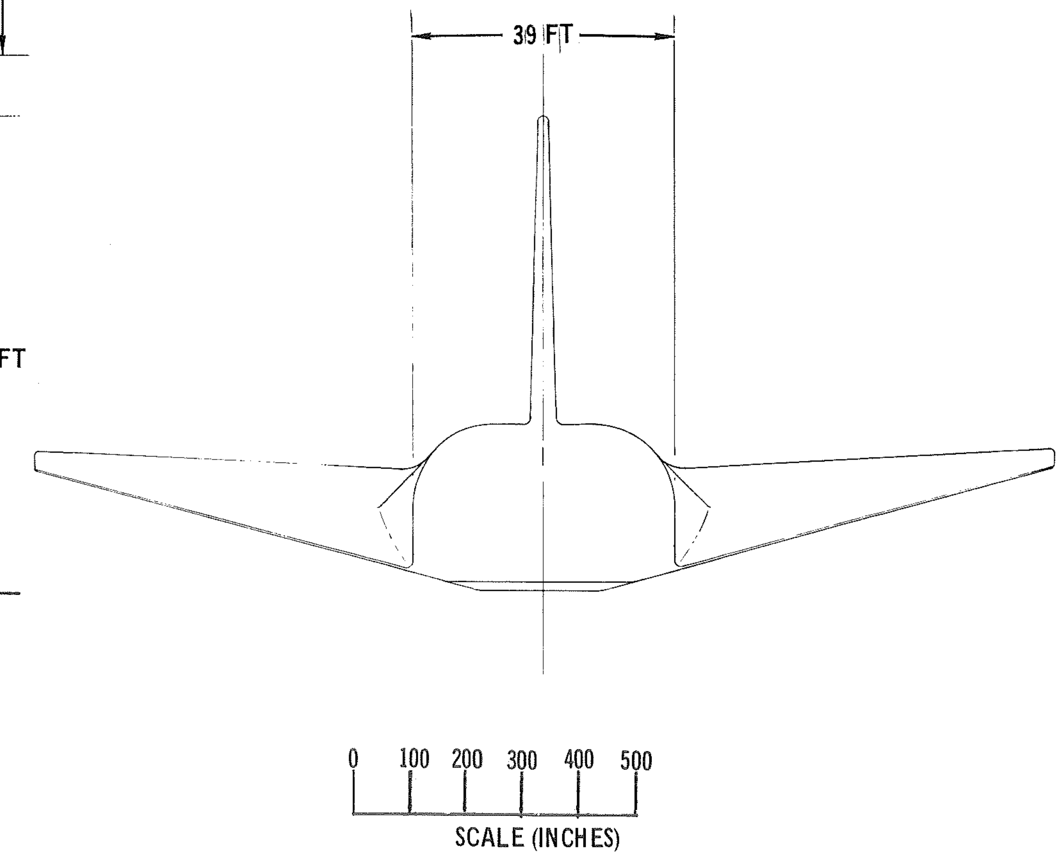
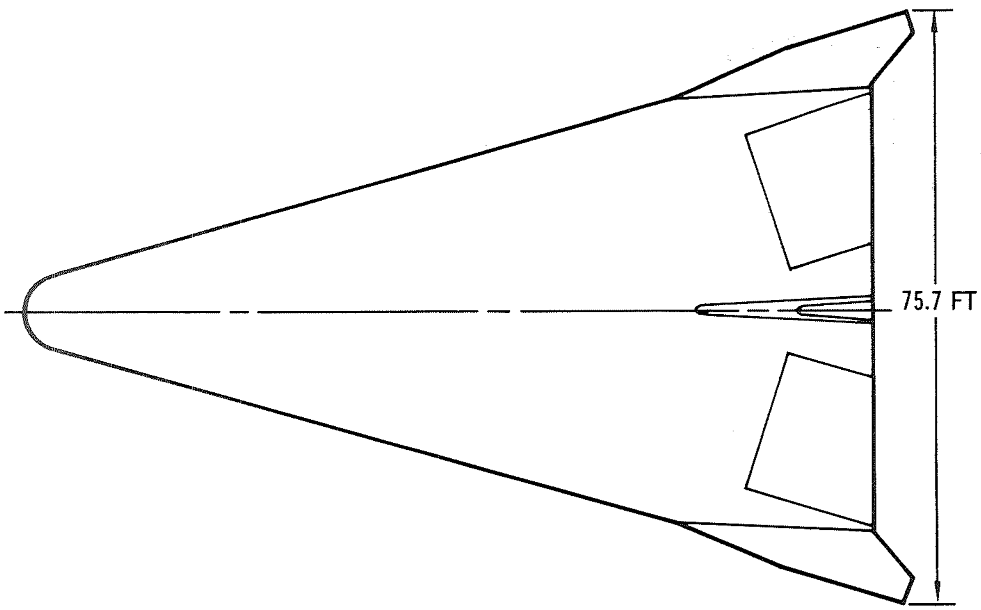


Figure 3–24

ORBITER BASELINE



ORBITER AERO SURFACES AREAS	
DORSAL FIN (INCLUDING RUDDERS)	395 FT^2
TIP FIN (INCLUDING FLAP)	346 FT^2
LOWER ELEVON	232 FT^2
UPPER ELEVON	226 FT^2
RUDDER	105 FT^2
TIP FIN FLAP (OUTER)	93 FT^2
TIP FIN FLAP (INNER)	66 FT^2
WETTED AREA	
OUTER ML (BODY ONLY)	10,700 FT
INNER ML (BODY ONLY)	9,600 FT
PLANFORM AREA	4,200 FT

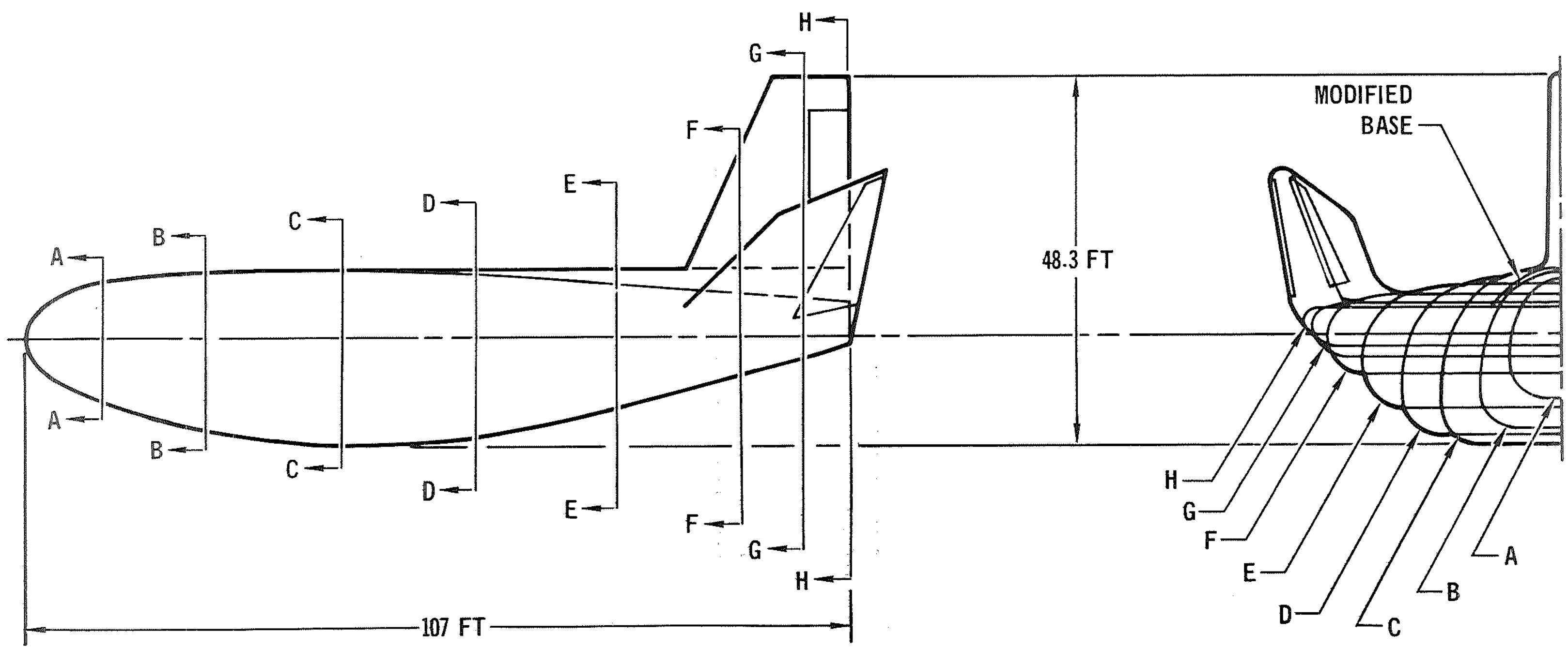


Figure 3-25

ORBITER BASE MODIFICATION

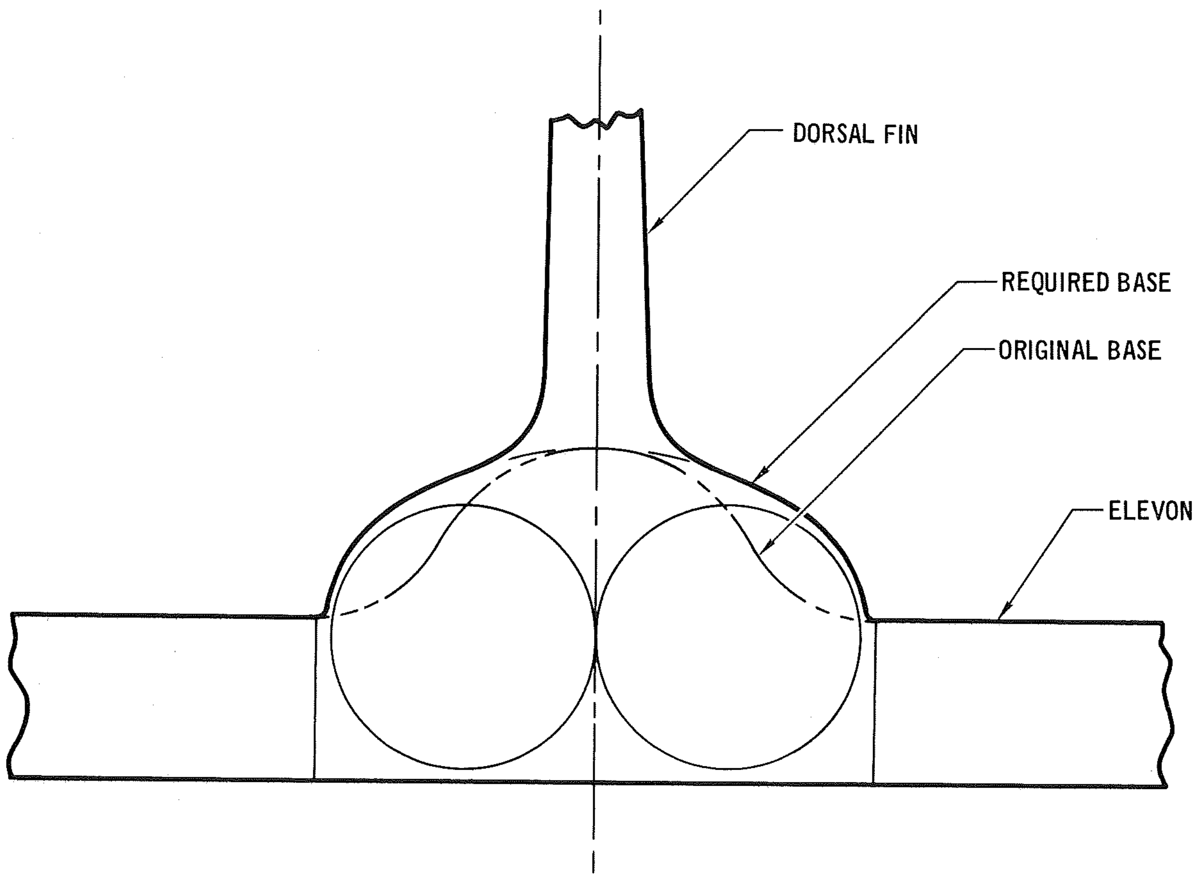


Figure 3-26

for some base area modification. However, it also showed that with proper engine orientation and careful packaging the elevons would probably require no changes. The base revision shown was therefore incorporated as a part of the baseline configuration.

3.3.3 Launch Configuration - The baseline launch configuration is shown in Figure 3-27. This arrangement shown has lower surfaces of both vehicles mated, with the bases of both vehicles in-plane. The problems discussed in Section 3.1.4, associated with this configuration, are shown in succeeding sections to be solved with relatively minor penalties.

The vehicle interface, internal structure and separation analyses were therefore performed for the configuration shown. This configuration was ultimately chosen because it provides low sensitivity to ground winds, lowest access height to payload and orbiter, and retains the option of firing second stage engines while mated.

BASELINE LAUNCH CONFIGURATION
Payload –25,000 Lb, 15' x 30'

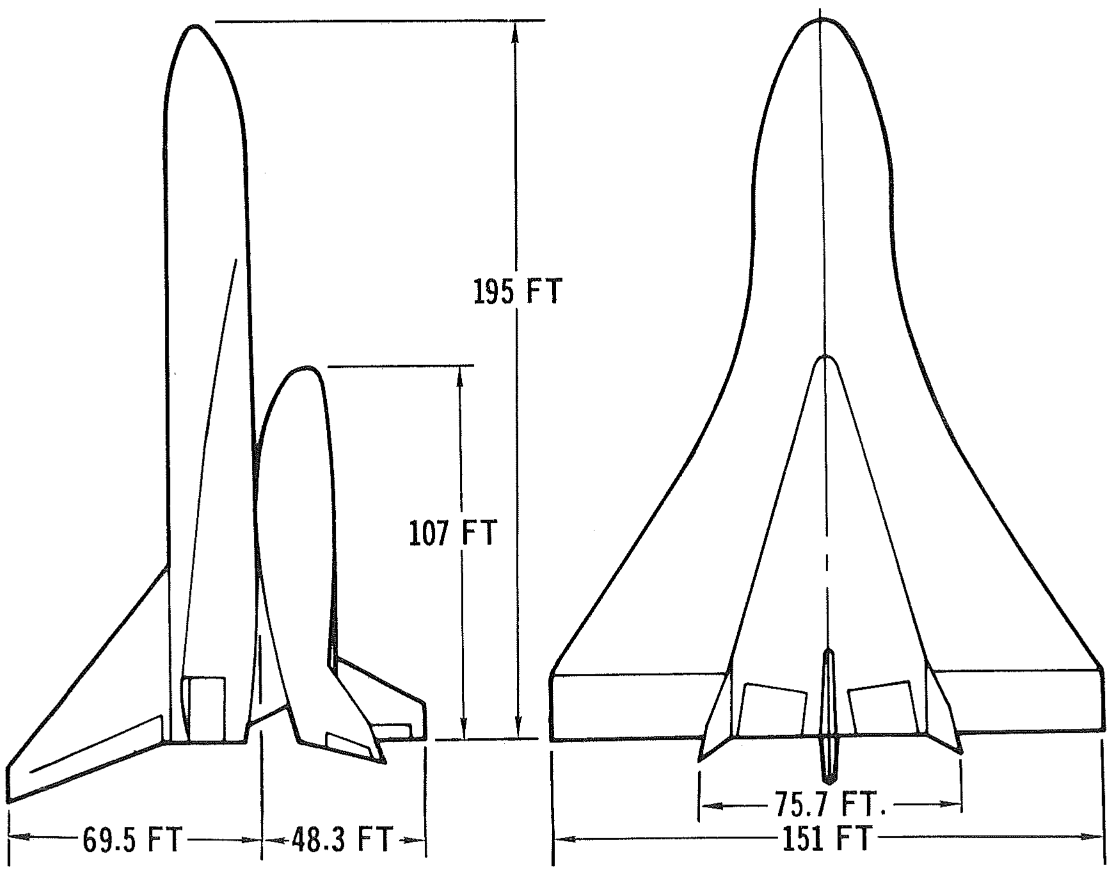


Figure 3–27

4.0 SUBSYSTEM ANALYSIS

This section includes analyses of structure, thermal protection system, avionics, electrical power and environmental control systems. The analyses of thermal protection system and avionics represent special emphasis areas.

Specific criteria and guidelines are presented in the applicable subsections. A baseline description is presented along with detail analyses, applicable trade studies and conslusions.

4.1 Structure - Included in this section are a description of the structural design criteria, structural loads and typical strength analyses. The system is a two stage vehicle with the orbiter being supported from the carrier lifting body surface (Reference Figure 4-1). A statically determinate three point attach arrangement is used for mating the two vehicles. The link at the aft attach point carries only direct tension or compression loads, all other loads are carried at the two forward attach points.

Carrier Structure - The general arrangement of the carrier airframe is shown in Figure 4-2. The airframe contains an insulated aluminum body shell structure with a titanium and Rene' 41 wing and vertical tail structure.

The body consists of an integral tank structure with both the forward portion of the airframe and thrust structure being unpressurized extensions of this integral structure. The shell structure contains integral longitudinal stiffeners and lateral flanges for attachment of frames (Reference Figure 4-3). The pitch, depth and gauge of the longitudinal stiffeners and gauge of the skin are varied to meet local strength requirements. The structural mold line is twelve inches inboard the external surface. Heat shield panels on the external surface are non-structural except for dynamic pressure loads and are attached so as to allow unrestrained thermal expansion. Frames supporting the heat shield panels and stiffening the shell are on twenty inch centers and are made of titanium to minimize conductance of heat to the inner structure. Space between the inner and outer surface contains fibrous insulation with a minimum two inch void maintained for purging this space.

The thrust structure consists of a semi-monocoque skirt, with a vertical keel web, extended from the integral tank structure, intercostals for local engine support and two major frames to support the intercostals. (Reference Figure 4-2). This arrangement leaves the center area open and easily accessible

CARRIER/ORBITER INTERFACE ARRANGEMENT

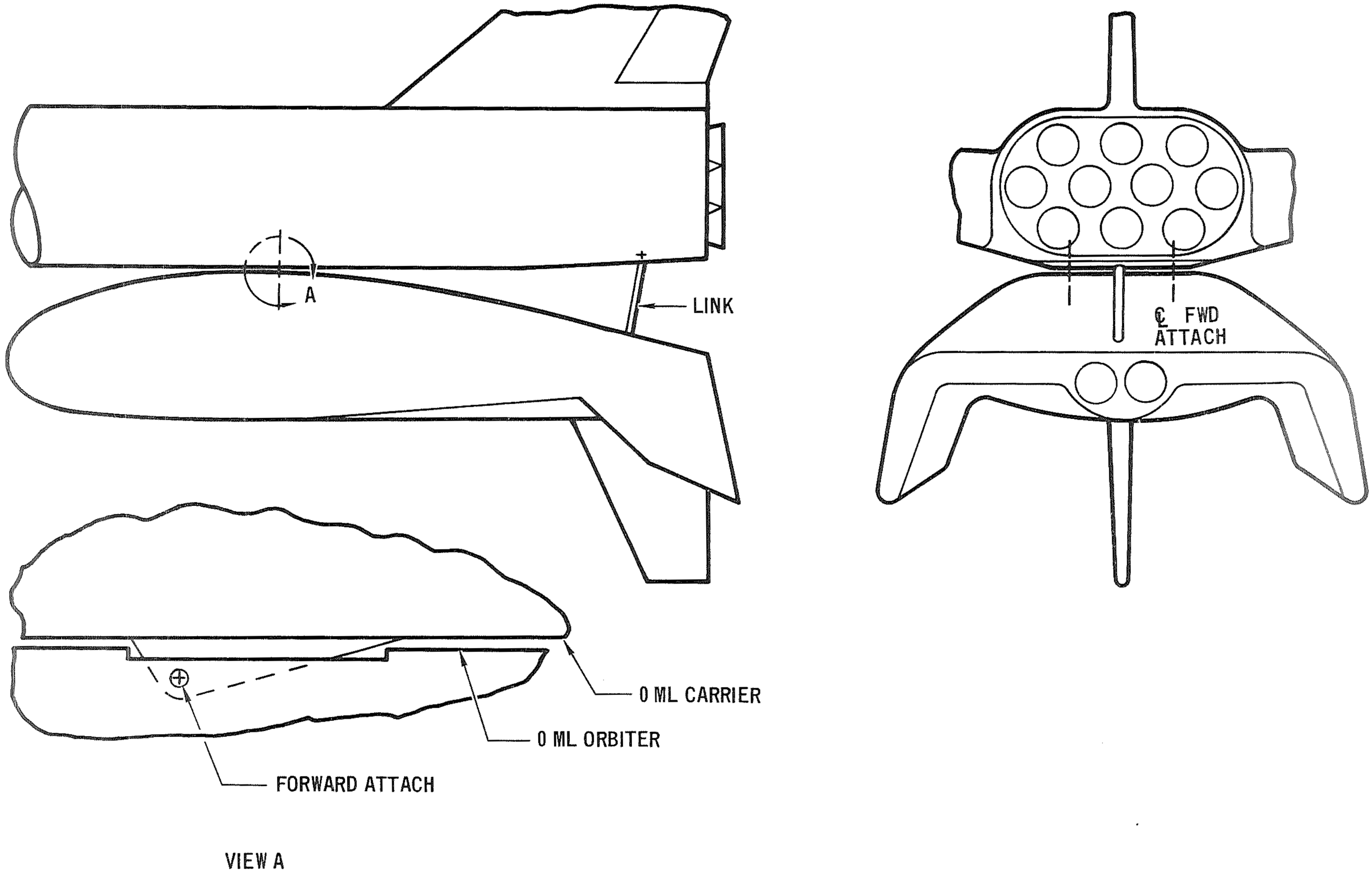


Figure 4-1

CARRIER GENERAL ARRANGEMENT

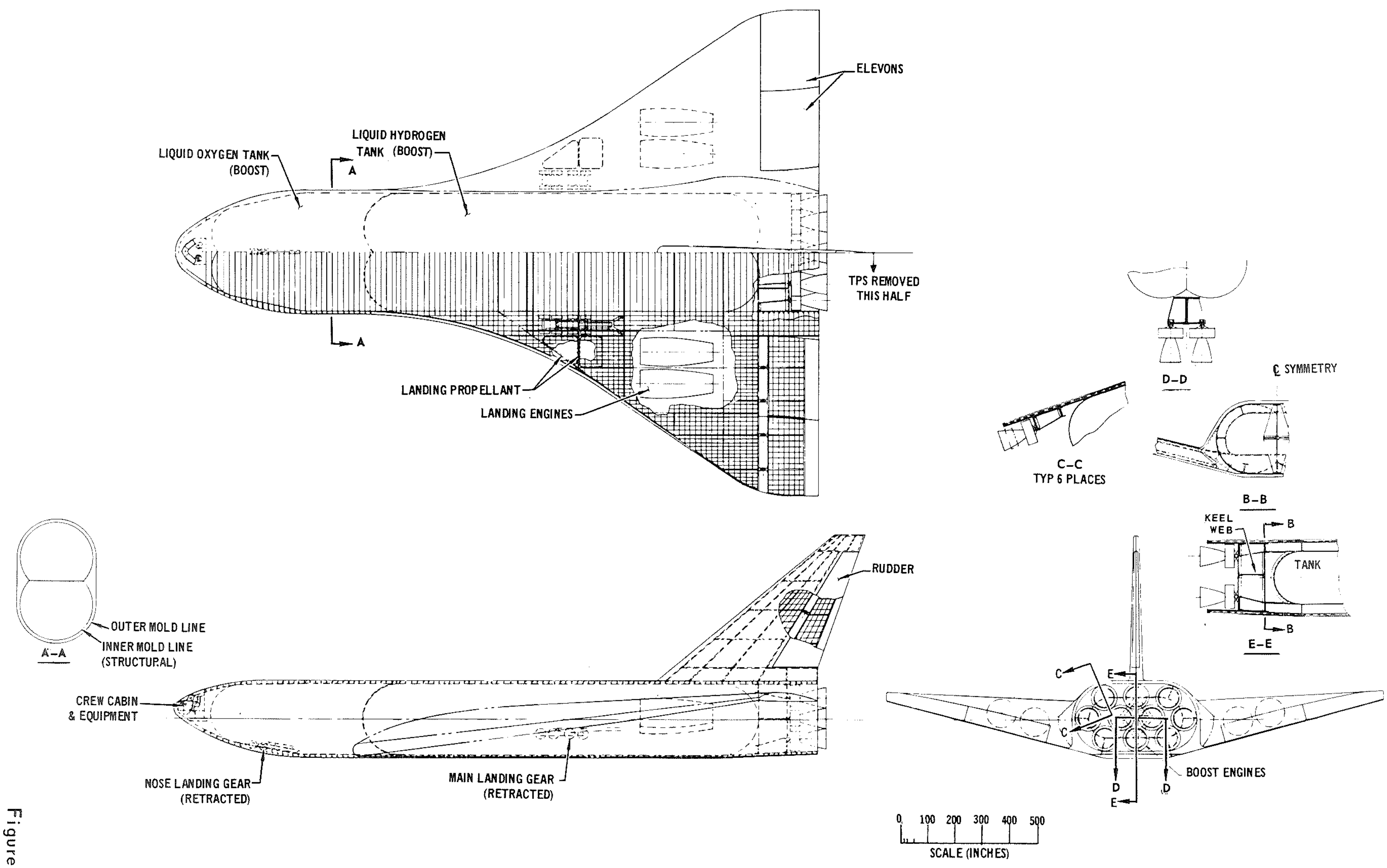


Figure 4–2

INTEGRAL TANK STRUCTURAL CONCEPT
Carrier

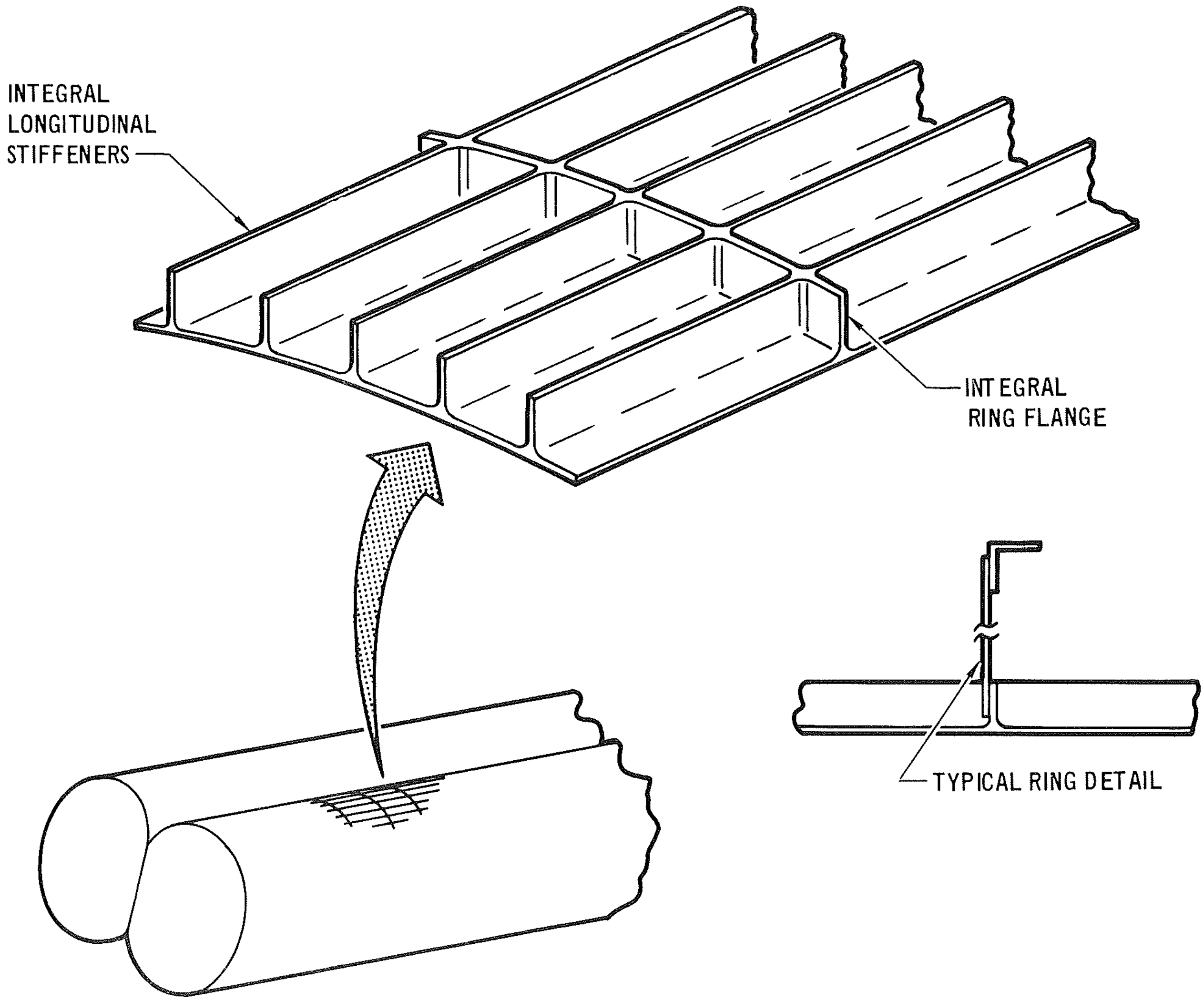


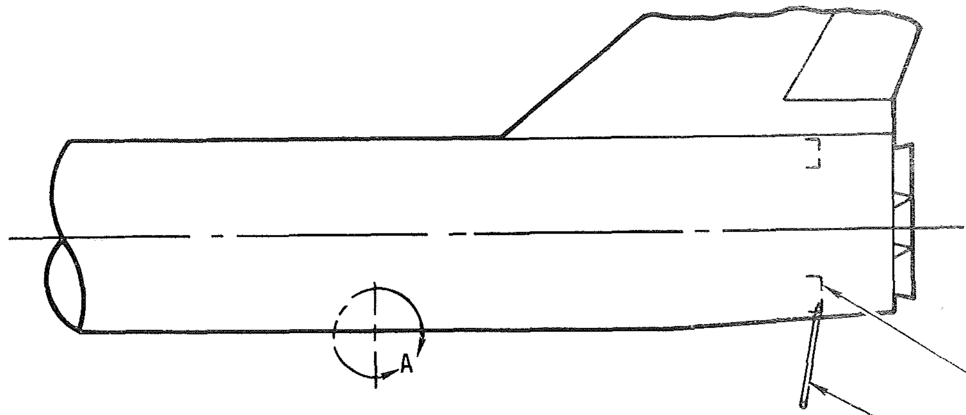
Figure 4-3

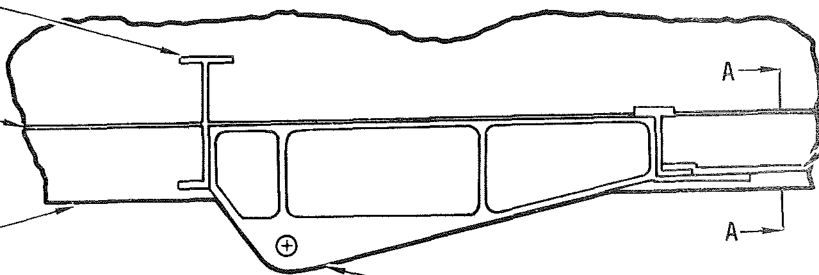
for installation of the propulsion system. Engine loads are reacted locally by the inboard intercostal cap and aft frame. Loads are sheared into the skirt and resulting overturning loads are carried by the two major frames. Loads are then redistributed by the skirt and introduced into the integral tank structure as distributed loads. The basic structure as designed for thrust loads provides a capability for launch pad tie down loads. Launch pad attach points coincide with the intercostals at the lower frame. Tie down loads are reacted locally by the outboard intercostal caps and frame and are in turn distributed to the shell structure.

Structure provided for the vehicle/vehicle attach loads include attach fittings, major frames to react the normal loads and longerons to react the drag loads. (Reference Figure 4-4). At the forward attach points, an attach fitting extends outboard of the outer surface moldline with the interconnect inboard of the orbiter moldline. This external structure is fixed and made from Rene' 41 alloy material because of reentry heating. Loads on the fittings are reacted by the frames and longerons. Normal loads on the frames are reacted by shears in the outer shell and centerline web. The required frame bending strength necessitates the addition of a beam cap inboard of the tank wall. Two titanium longerons are used to distribute drag loads to the integral body structure. The thrust structure is used to react the aft attach point loads.

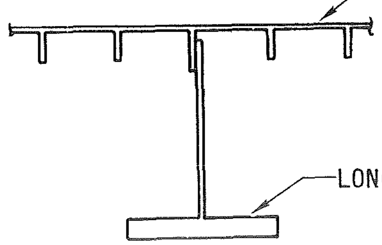
The wing and tail are designed as hot structures. Design temperatures are such that Rene' 41 and titanium alloy materials can be used for the structure. In general, Rene'41 material is used along the leading edges and forward portion of the lower wing surface with titanium material used over the remainder of the surfaces. Conventional multi-spar arrangements are used for both structures. Spars in the vertical tail have been located to coincide with wing carry through structure and thereby eliminate need for additional structural support members. Wing carry through structure at the rear spar is continuous through the thrust structure. Carry through structure at the intermediate and front spars is external to the integral tank structure (Reference Figure 4-5). The wing/fuselage intersection is faired in the root area to provide efficient load path continuity between the two structures. Stresses resulting from differences in wing and body strains due to pressure or temperature do not appear to be excessive and do not add to maximum stresses from mechanical loads. The inboard frame cap is an integral

CARRIER SUPPORT STRUCTURE
Carrier/Orbiter Attach





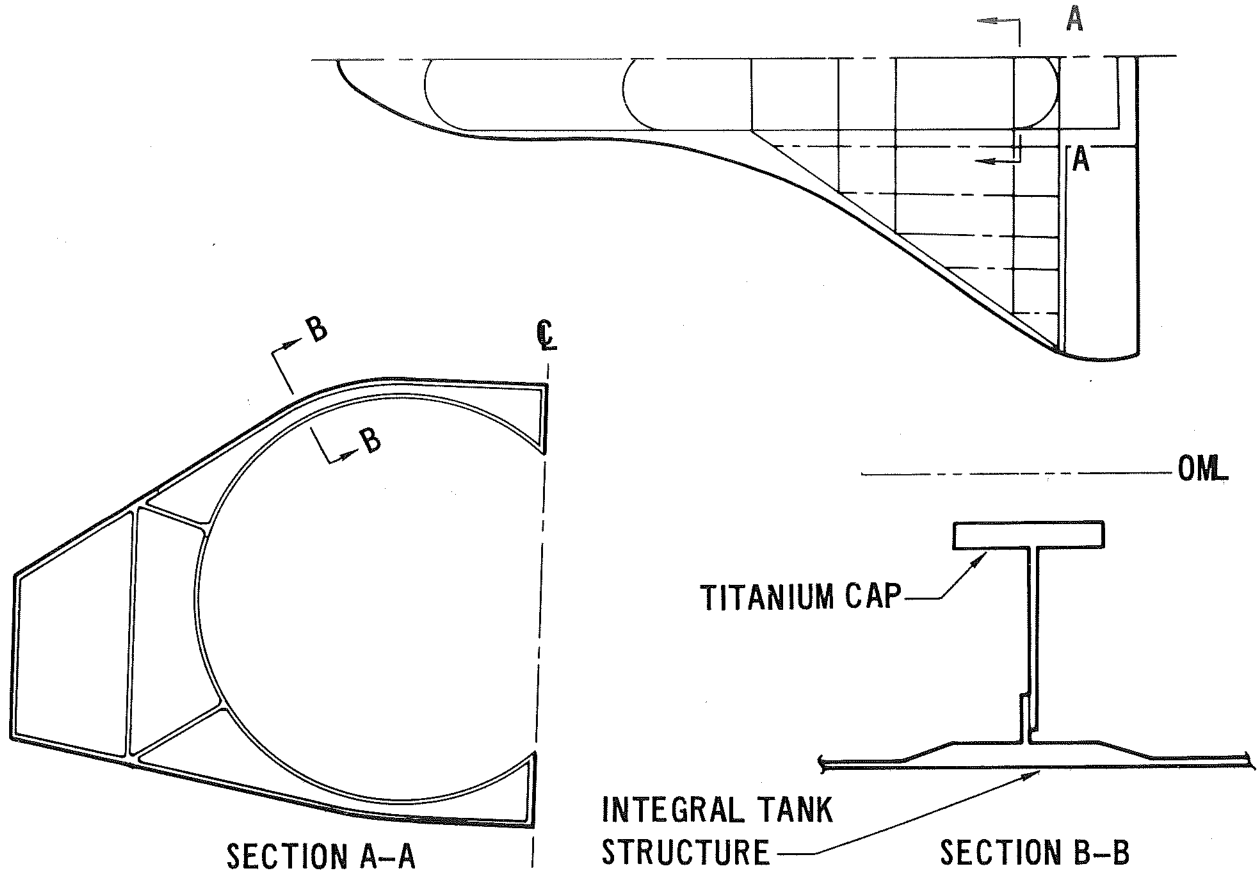
VIEW A



SECTION A–A

Figure 4–4

WING CARRY THROUGH STRUCTURE



ILRVS–250 F

Figure 4–5

part of the body structure. Titanium is used for the frame web and outboard cap because of its favorable strength weight ratio and to minimize conductance of heat to the inner structure.

Heat shield panels (shingles) block the bulk of the heat from the aluminum body shell structure (Reference Figure 4-6). Surface temperatures permit the use of radiation cooled shingles of titanium and Rene' 41 alloy materials. The panels are twenty inches long and on the lower surface and sides of the body are composed of an external smooth skin stiffened by longitudinal corrugations. Single thickness beaded panels are used on the upper shadowed surface in areas of low heating.

Panels distribute positive pressure loads directly to the frames by bearing on support channels. A pi shaped retainer reacts negative pressure loads from the corrugated panels and provides a gap for thermal expansion. Beaded panels are retained by round head screws with clamp up bushings. Oversize holes provide for thermal expansion.

Orbiter Structure - The general arrangement of the orbiter airframe is shown in Figure 4-7. The body consists of an insulated aluminum shell structure with external moldline heat shield panels. Closure bulkheads are provided at the forward end of the payload bay and aft end of the body structure. The structural moldline is twelve inches inboard the external surface. For efficient utilization of available volume, the main propellant tanks are integrated with the shell structure to form irregular shaped pressure vessels. This integral tank structure provides load paths for carrying both body bending, axial and shear loads simultaneously with tank pressure loads. With the irregular shaped pressure vessel, pressure loads are distributed to bi-axially loaded internal baffle/webs by bending the sidewall stringers. The shell contains integral longitudinal stiffeners and lateral flanges for attachment of external frames (Reference Figure 4-8). The pitch, depth and gauge of the longitudinal stiffeners and gauge of the skin are varied to meet local strength requirements. Spacing of frames supporting the non-strucutral heat shield panels and stiffening the shell varies from 12 to 15 inches. Frame outboard caps are made of titanium or Rene' 41 depending on the local surface temperatures. Thermal stresses due to frame temperature gradients would tend to relieve stresses due to mechanical loads at maximum bending and do not appear to be a problem. A detailed analysis is required to determine actual values throughout the frames. Space between the inner and outer surface contains fibrous insulation with a minimum two inch void maintained for purging this space.

SURFACE HEAT PROTECTION ARRANGEMENT
Carrier Body

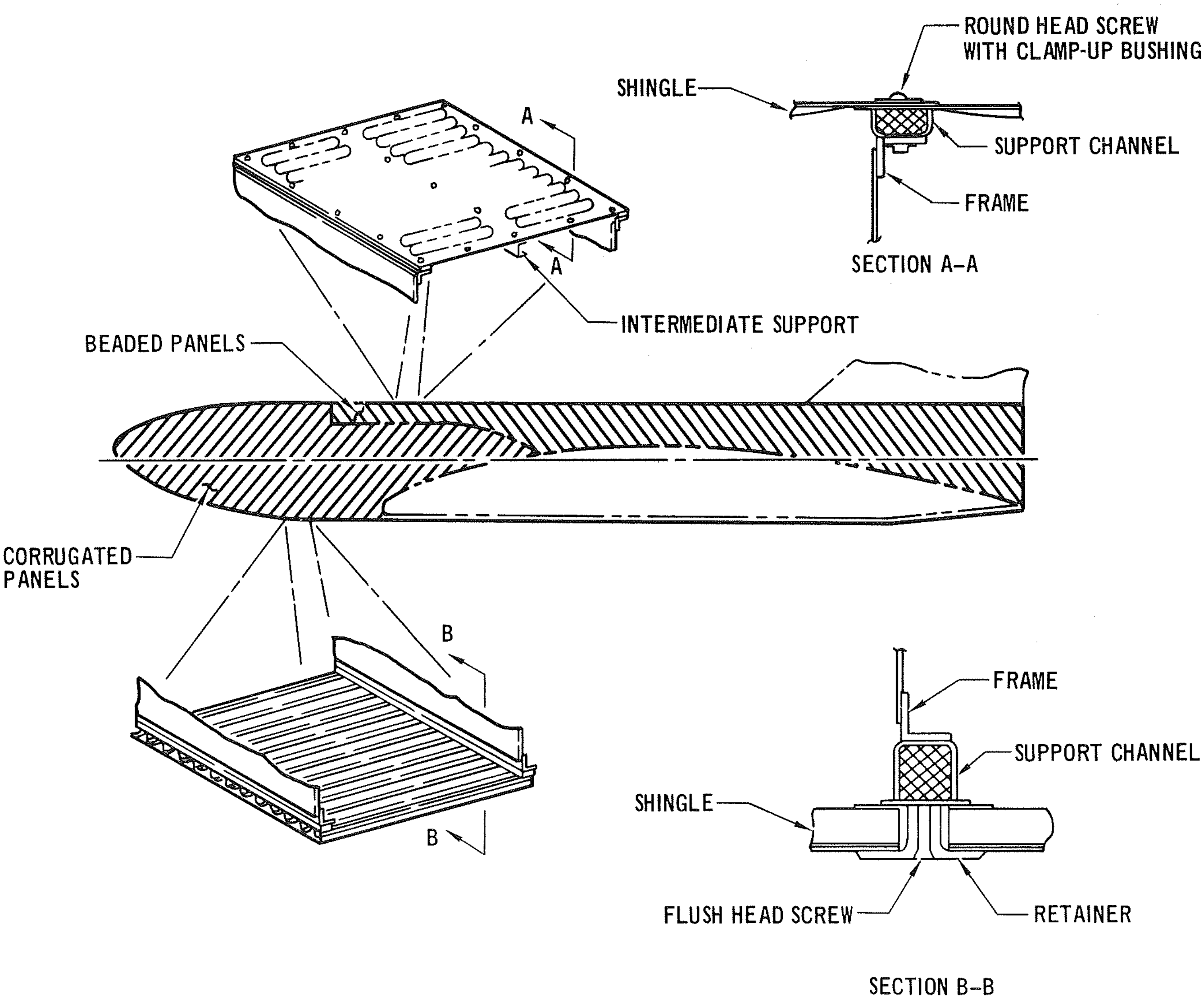


Figure 4–6

ORBITER INTEGRAL TANK CONCEPT

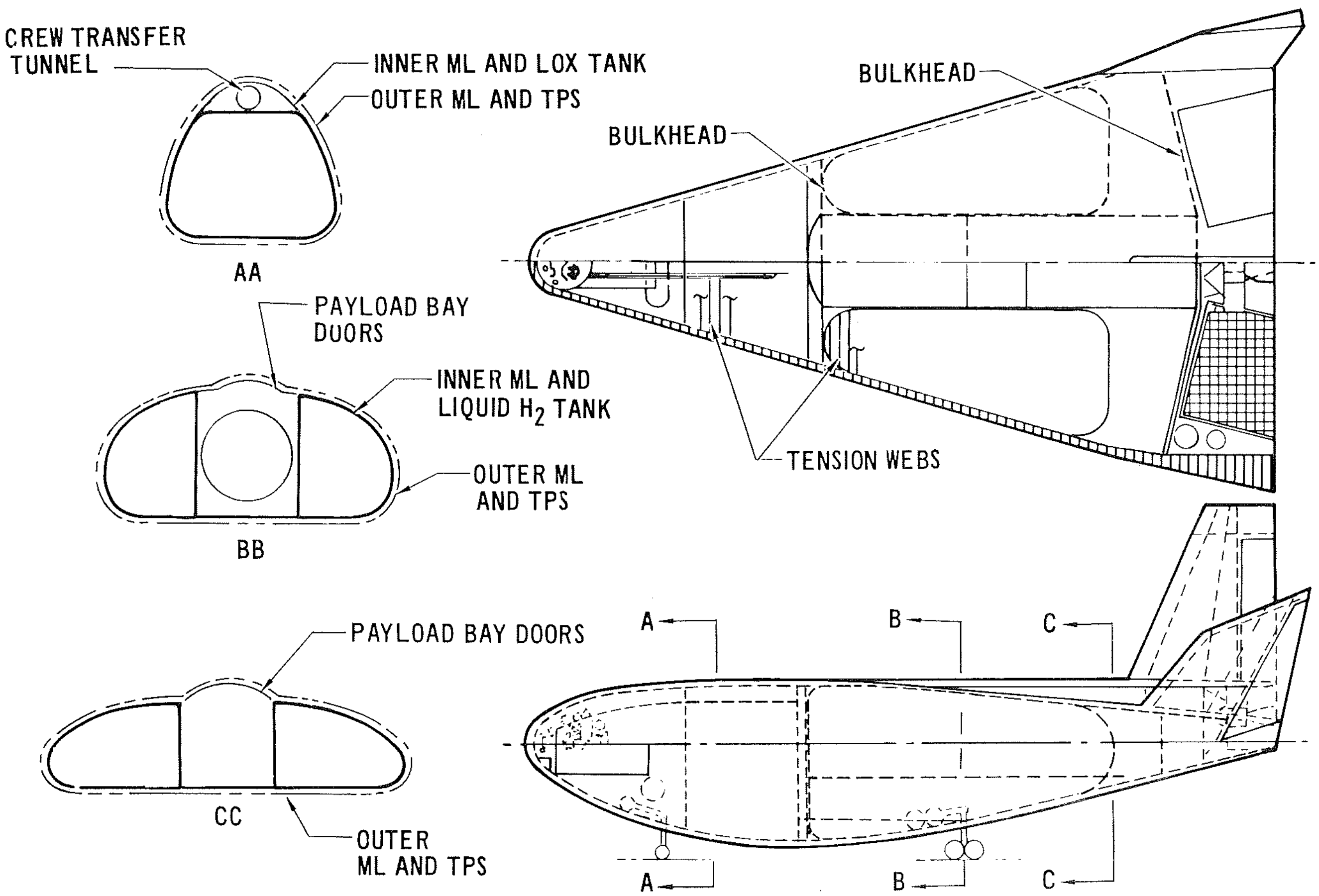


Figure 4-7

STRUCTURAL CONFIGURATION ORBITER INTEGRAL TANK

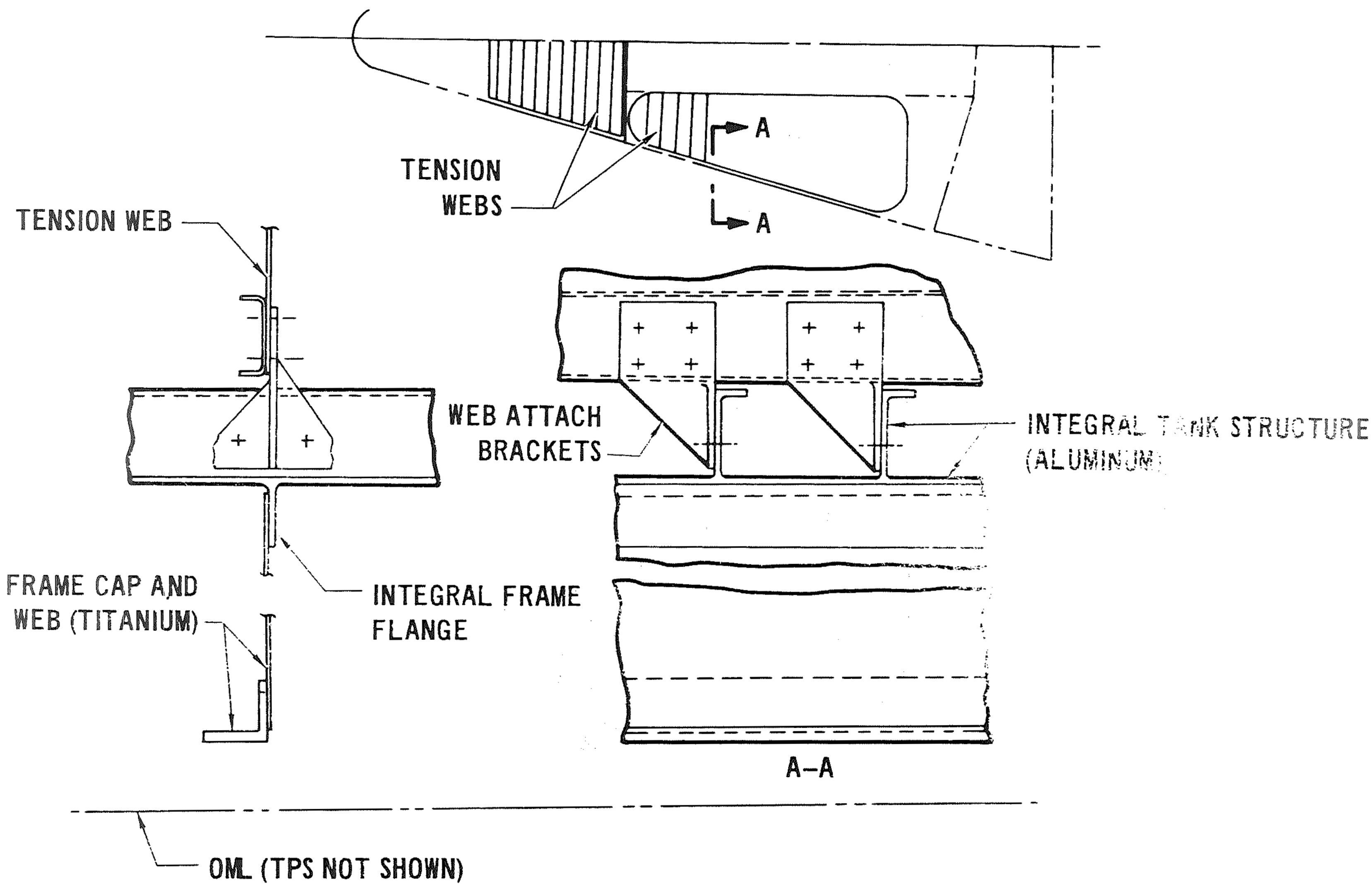


Figure 4-8

ORBITER SUPPORT STRUCTURE
Carrier/Orbiter Attach

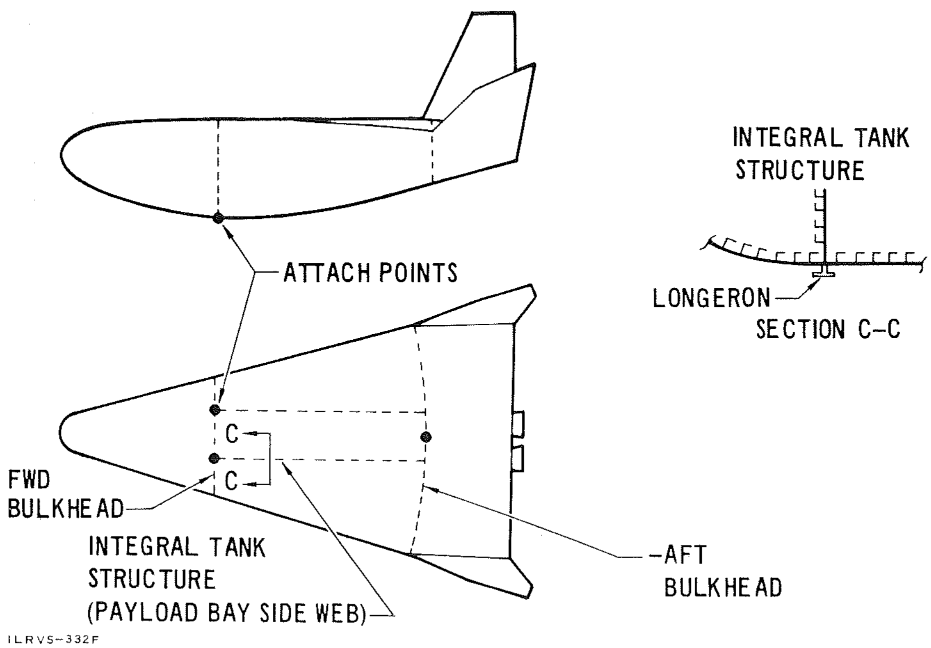


Figure 4-9

The closure bulkheads at the forward end of the payload bay and aft end of the body shell structure are utilized to redistribute vehicle/vehicle attach loads. (Reference Figure 4-9). Normal loads on the bulkheads are reacted by shears in the shell structure. Two titanium longerons are provided to distribute drag loads to the body structure. The upper attach points are located at the intersection of the payload bay side web and inner moldline web to take advantage of the multiple shear paths. The tip fins, elevons, and thrust structure are supported by the body shell and aft closure bulkhead. Torque boxes extending from the bulkhead support the tip fins. Thrust structure is extended from the two internal vertical web and enclosed moldline panels. The elevons are supported directly by the bulkhead and shell structure.

Heat shield panels (shingles) block the bulk of the heat from the aluminum body shell structure (Reference Figure 4-10). Surface temperatures require the use of radiation cooled shingles of titanium, Rene' 41, TD Ni Cr and columbium alloy materials. Panel lenghts vary from twelve to fifteen inches. Single thickness beaded panels are used on the upper shadowed surface in regions which experience low heating rates. Panels used on other areas of the body are composed of an external smooth skin stiffened by longitudinal corrugations. A pi shaped retainer reacts negative pressure loads from the corrugated panels and provides a gap for thermal expansion. Positive pressure loads are reacted by support channels. Beaded panels are retained by round head screws with clamp-up bushings. Oversize holes provide for thermal expansion.

4.1.1 Structural Design Criteria - The criteria summarized here were formulated to establish a basis for the study structural analysis tasks. Items usually found in a contract definition or acquisition phase structural design criteria were included only if necessary for the analysis planned for this stage of the development cycle. The scope and level of detail of the structural design criteria must be expanded as the ILRVS development cycle progresses.

Definitions

a) Structural Requirements - Structural requirements are values of specific design condition parameters such as loads and temperatures which satisfy conditions derived from the structural design criteria.

b) Design Conditions - The definitions of the combinations of natural and induced environments, based on the structural design criteria, which uniquely establish the structural design requirements.

SURFACE HEAT PROTECTION ARRANGEMENT
Orbiter Body

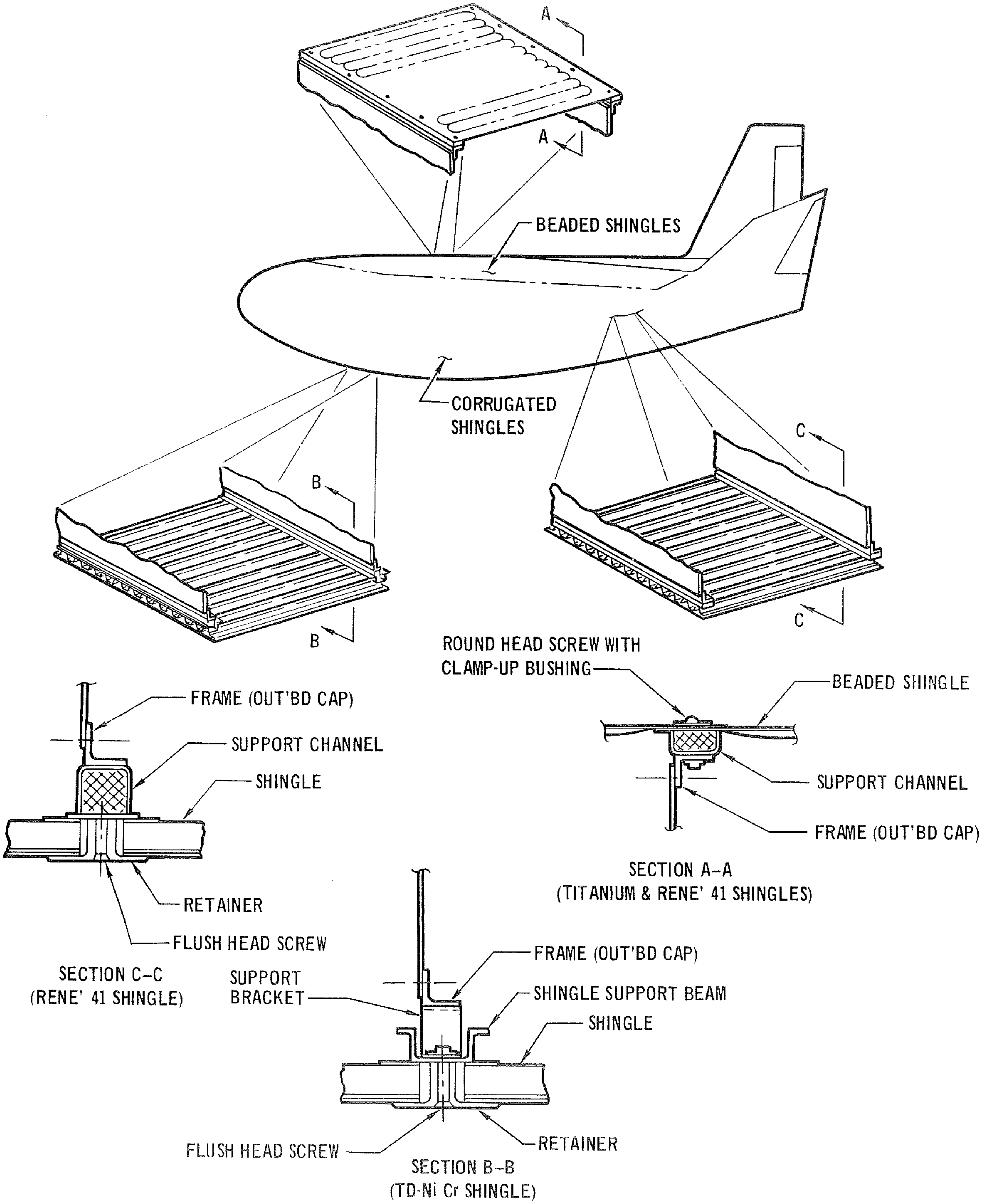


Figure 4-10

c) Factor of Safety - Ratio of allowable load (or stress) to limit load (or stress) at the temperature which defines the allowable and is used to account for uncertainties and variations from item to item in material properties, fabrication quality and details and internal and external load distributions.

d) Temperature Uncertainty Factor - The temperature uncertainty factor is an arbitrary factor applied to predicted temperature to account for uncertainties in the thermal analysis.

e) Limit Load - Limit load is the maximum load or combination of loads the structure is expected to experience in a specific condition.

f) Ultimate Load - The product of the factor of safety times limit load.

g) Nominal Heating Effects - Nominal heating effects are temperatures or heating rates the structure is expected to experience based on nominal environments, performance and trajectories.

h) Predicted Heating Effects - Predicted heating effects are temperatures or heating rates which the structure is expected to experience during a design mission. Predicted temperatures are analogous to limit loads and are assumed to include the effects of dispersions.

i) Design Heating Effects - Design heating effects are predicted heating effects with additional heating rate or temperature factors to account for analytical uncertainties.

Design Mass Properties - Design weights and centers of gravity used in the structural analysis are tabulated in Table 4-1 for both the carrier vehicle and the orbiter vehicle. Detailed weights data for each vehicle are presented in Section 5.

Design Environments - Design atmosphere, surface winds, and winds aloft for launch and ascent are based on the design environmental data in Reference 5. Ground winds for prelaunch conditions are for the worst month winds with a 99% probability of not being exceeded. Wind environment for the launch and ascent is 95% probability winds with 99 percentile wind shears.

Fundamental Criteria - The FAA (part 25), the applicable portions of the Military Specifications (8860 Series) and supersonic transport specifications are used as guidelines in establishing criteria for the vehicle. The intent is to merge the appropriate items of spacecraft criteria with well established air transport criteria, modified if necessary to reflect the ILRVS mission requirements. The following subsections define specific criteria related to the areas

Table 4-1

DESIGN MASS PROPERTIES

PHASE	CARRIER		ORIBITER	
	WEIGHT-LBS	CG - %L	WEIGHT-LBS	CG - %L
PRELAUNCH	2,689,000	32.4	730,000	37.0
LIFTOFF	2,672,000	32.6	730,000	37.0
MAX αq	2,028,000	38.1	730,000	37.0
SEPARATION	512,000	65.8	730,000	37.0
INJECTION	–	–	232,000	54.0
ENTRY	451,000	65.8	196,000	54.0
LANDING	451,000	66.8	186,000	56.6

of strength, stiffness, factors of safety and pressurization factors. These data are the minimum requirements for the design and structural analysis of the vehicle.

Design Factors - Spacecraft design is based on maximum reusability except for emergency conditions where only crew safety is considered to be mandatory. The design factors used in the structural analyses are summarized in Table 4-2. The factor of safety is applied to limit load to obtain ultimate load. The pressurization factors are applied to the maximum operating pressure to determine proof and burst pressure. Design heating effects are obtained by multiplying the temperature resulting from the design trajectories by the despersion and uncertainty factor. Aeroelastic and buffet effects are accounted for by multiplying normal loads by the aeroelastic and buffet factor. The dynamic amplification factors are applied to rigid body loads to account for the dynamic effects.

Strength - The structure is designed to withstand limit load combined with predicted heating effects, without experiencing detrimental deflections. The structure is designed to withstand the following ultimate conditions without failure: limit load combined with design heating effects or ultimate load combined with predicted heating effects, whichever is more critical. Structural re-usability is based upon loads, temperatures and other environments resulting from nominal flight trajectories.

The mechanical combinations are as follows:

a) Ultimate mechanical loads are combined with loads resulting from ultimate compartment pressure except that where compartment pressure loads relieve mechanical loads, limit pressure loads are used with ultimate mechanical loads. Compartment pressures are based on maximum vent pressure or minimum regulator pressure whichever is most severe.

b) The tank pressures are combined as indicated in (a) for mission phases in which the primary propulsion system is activated. For mission phases following ascent in which the primary propulsion system is not used the tanks are considered to be pressurized to the stand-by operating pressure or de-pressurized, whichever results in maximum loadings.

c) In addition to withstanding pressure differentials resulting from normal operations, common bulkheads shall be capable of withstanding loads resulting from a loss of 50 percent of the normal operating pressure in either tank, combined with inertia loads.

Mission Phase Requirements - Structural design criteria for specific mission phases are defined in the following paragraphs.

Table 4-2

DESIGN FACTORS

FACTOR OF SAFETY		
• FLIGHT CONDITIONS		1.40
• GROUND HANDLING CONDITIONS POTENTIALLY HAZARDOUS TO PERSONNEL		1.50
PRESSURIZATION FACTOR	PROOF	BURST
• MANNED CABINS	1.33	2.00
• PNEUMATIC VESSELS	1.67	2.22
• HYDRAULIC VESSELS	1.50	2.50
• MAIN PROPELLANT TANKS	1.00	1.40
• PYROTECHNIC DEVICES	1.20	1.50
• LINES AND FITTINGS	2.00	4.00
STRUCTURAL TEMPERATURE FACTOR		
• TEMPERATURE DISPERSION & UNCERTAINTY		1.10
AEROELASTICITY AND BUFFET FACTOR		
• ASCENT – NORMAL		1.4
DYNAMIC AMPLIFICATION FACTOR		
• LAUNCH – POGO EFFECT – AXIAL		1.1
• LANDING IMPACT – MAIN GEAR – NORMAL		1.2
• LANDING IMPACT – NOSEGEAR – NORMAL		1.6

Pre-Launch - The surface winds and gusts for design of the aerospace vehicle or the boost vehicle and orbiter vehicle separately are the 99 percent probability of non-exceedance values for the ETR launch site. The vehicles are mounted in a vertical position with propellant onboard. Reference 5 was used as a guide in defining the ground phase environments.

Ascent - The aerospace vehicle requires vertical lift-off as the primary ascent mode. Horizontal take-off at cruise design gross weight is a secondary ascent mode for use in the development testing, ferry, and training operations. The design winds aloft are 95% probability of non-exceedance for the ETR launch site.

The design winds are assumed to induce a maximum instantaneous angle of attack of 5 degrees at maximum dynamic pressure. This angle of attack is based on estimates of rigid body translation responses only (no pitch) which are then reduced by 50% to account for the effects of a load relief control system loop.

The design launch trajectory used is shown in Figure 4-11. Although this trajectory shows a maximum longitudinal load factor of 3.0, a maximum design longitudinal load factor of 4.0 was assumed for structural analysis to provide for mission with an all cargo payload where the engines may not be throttled. The factors applied to rigid body loads to account for aeroelasticity, buffet, and dynamic effects are defined in Table 4-2.

Entry - Structural requirements are based on the baseline design trajectories shown in Figures 4-12 and 4-13. Loads and structural temperatures based on these trajectories are limit and predicted, respectively. Entry vehicles are analyzed for a maximum normal load factor of 3.0.

Transition - Transition from the entry attitude to the airplane cruise attitude is made at a Mach Number of .8 or less. The design load factor for this phase is 3.0.

Cruise - The V-n and Design Speed diagrams for both vehicles are defined in Figures 4-14 and 4-15. The 2.5 load factor is common to both vehicles but stall lines and dive speeds are configuration dependent. Types of maneuvers required are based on applicable transport aircraft specifications.

Landing - Both vehicles are analyzed for landing sink speeds of 10 fps. Structural loads resulting from the landing conditions are neither limit or ultimate but are treated as "design" values. Landing gear yielding or minor

DESIGN ASCENT TRAJECTORY

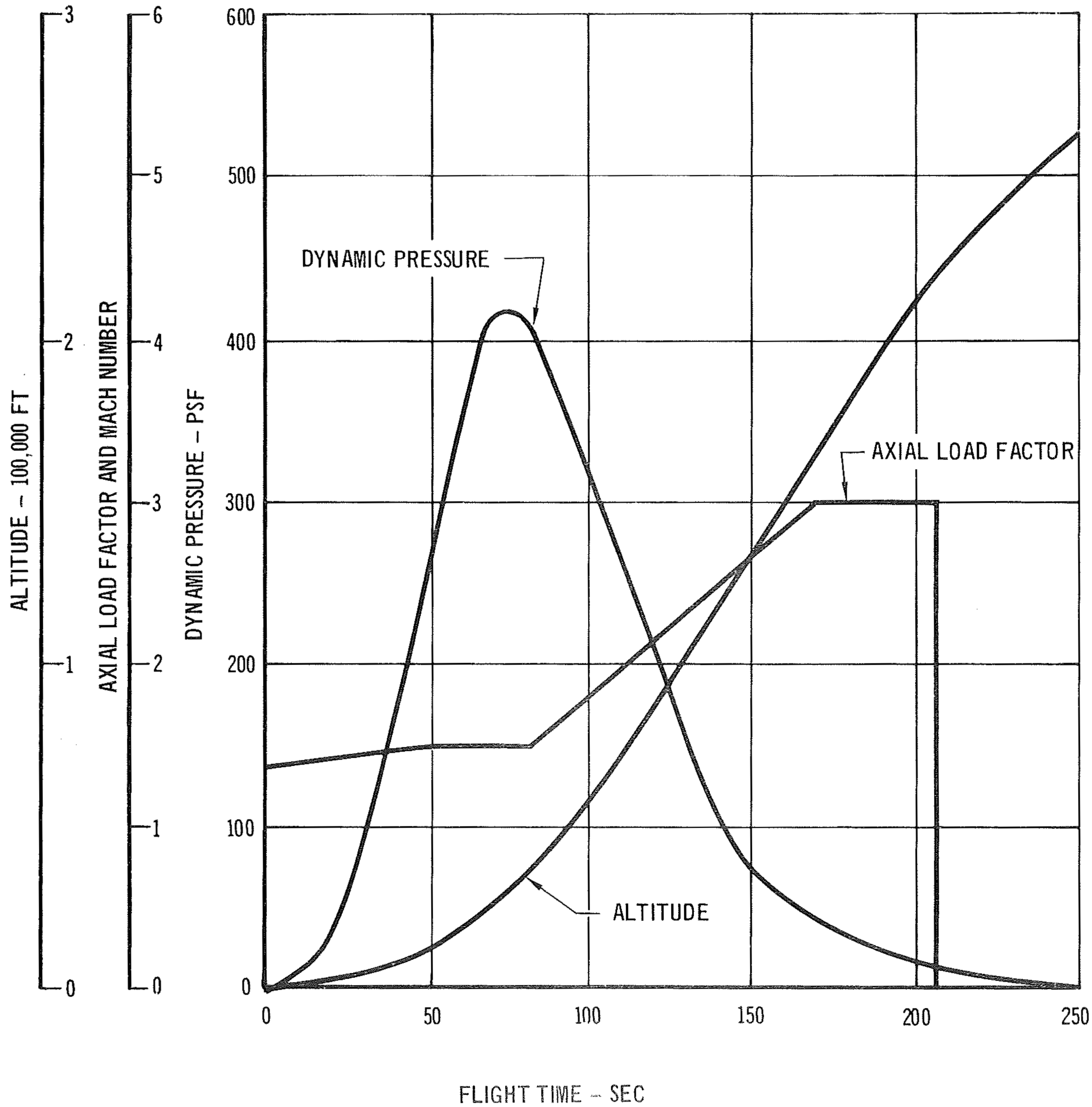


Figure 4-11

CARRIER DESIGN ENTRY TRAJECTORIES

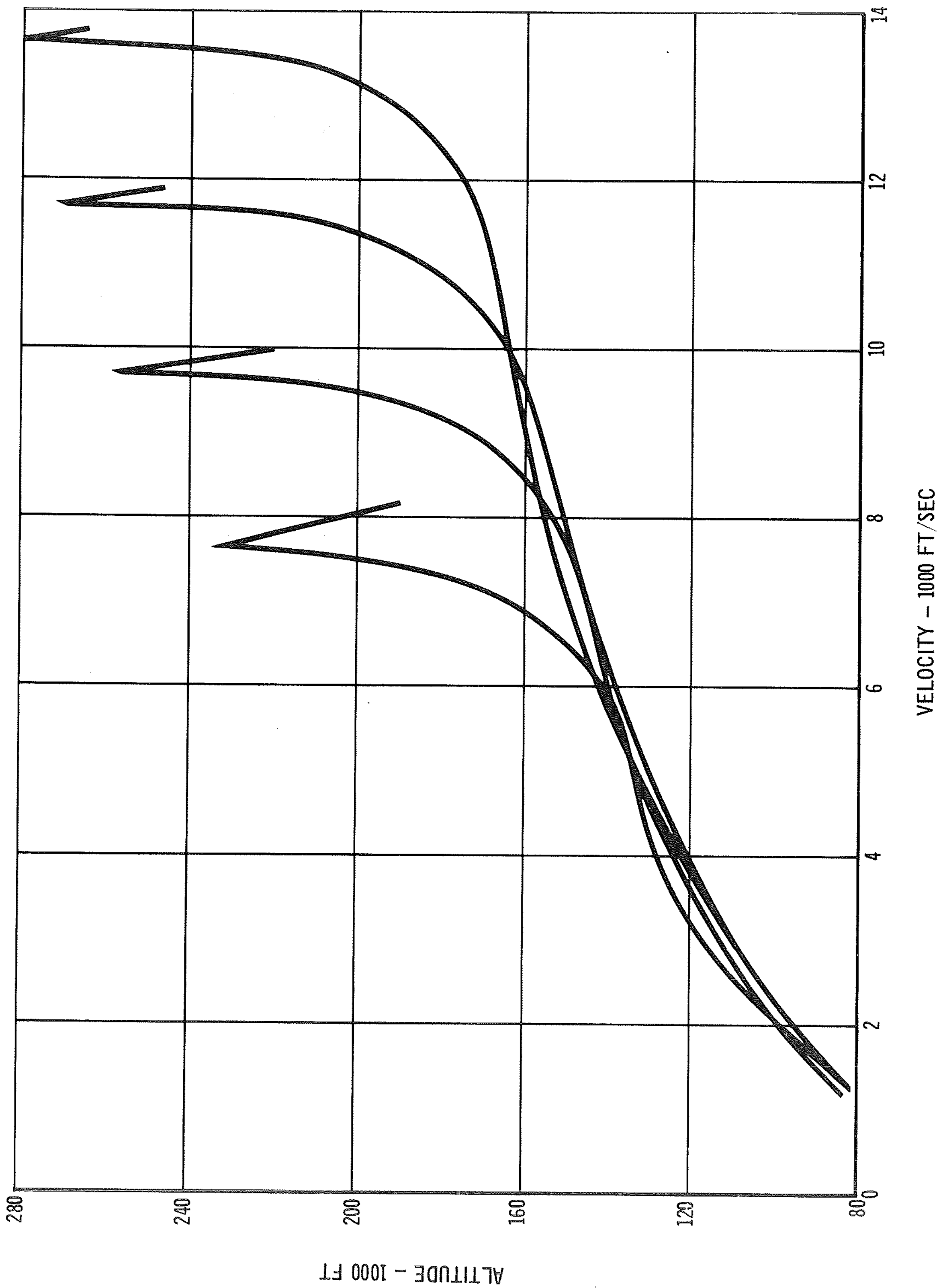


Figure 4-12

ORBITER DESIGN ENTRY TRAJECTORY

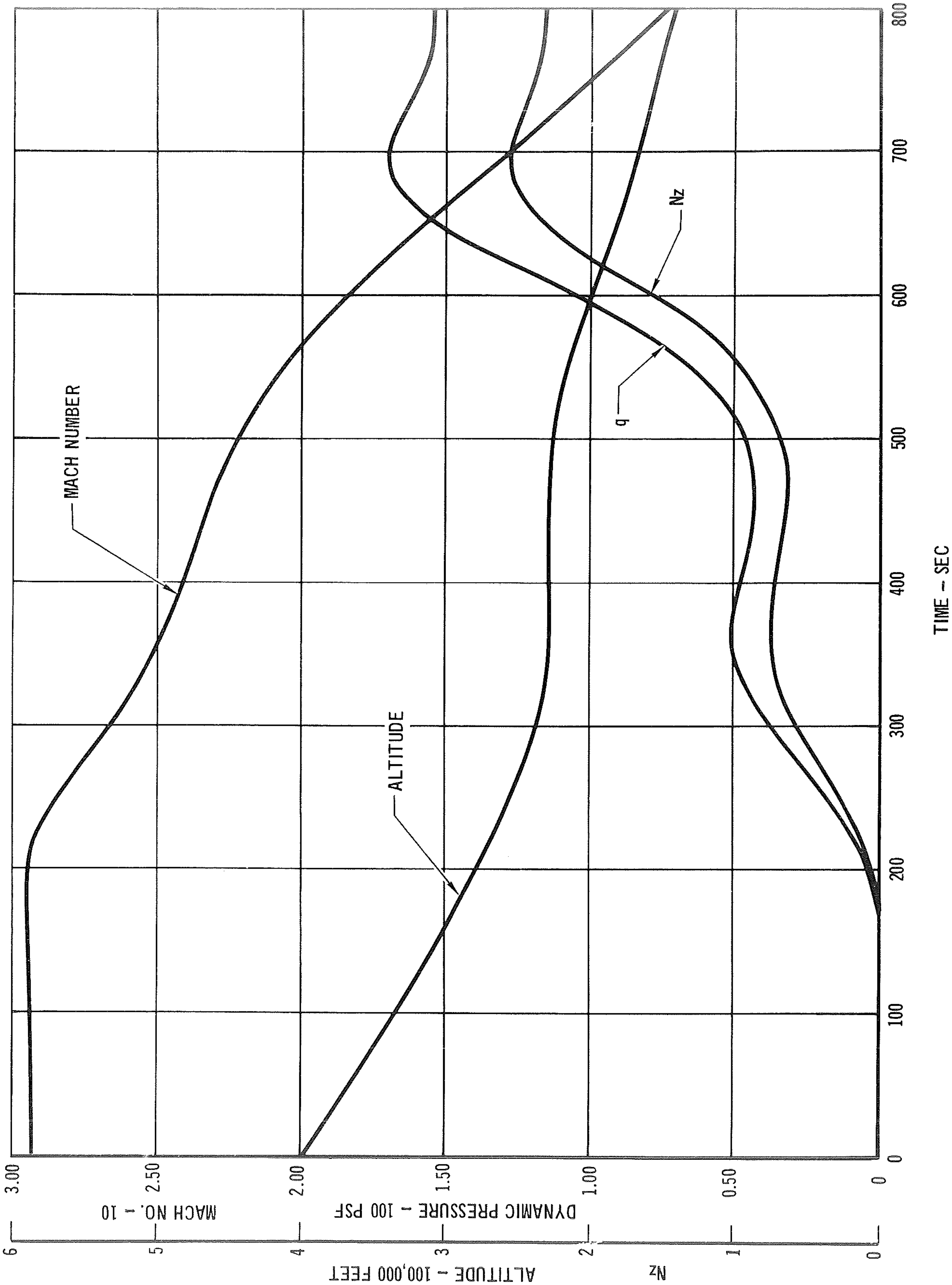


Figure 4-13

CARRIER AIRPLANE FLIGHT ENVELOPES

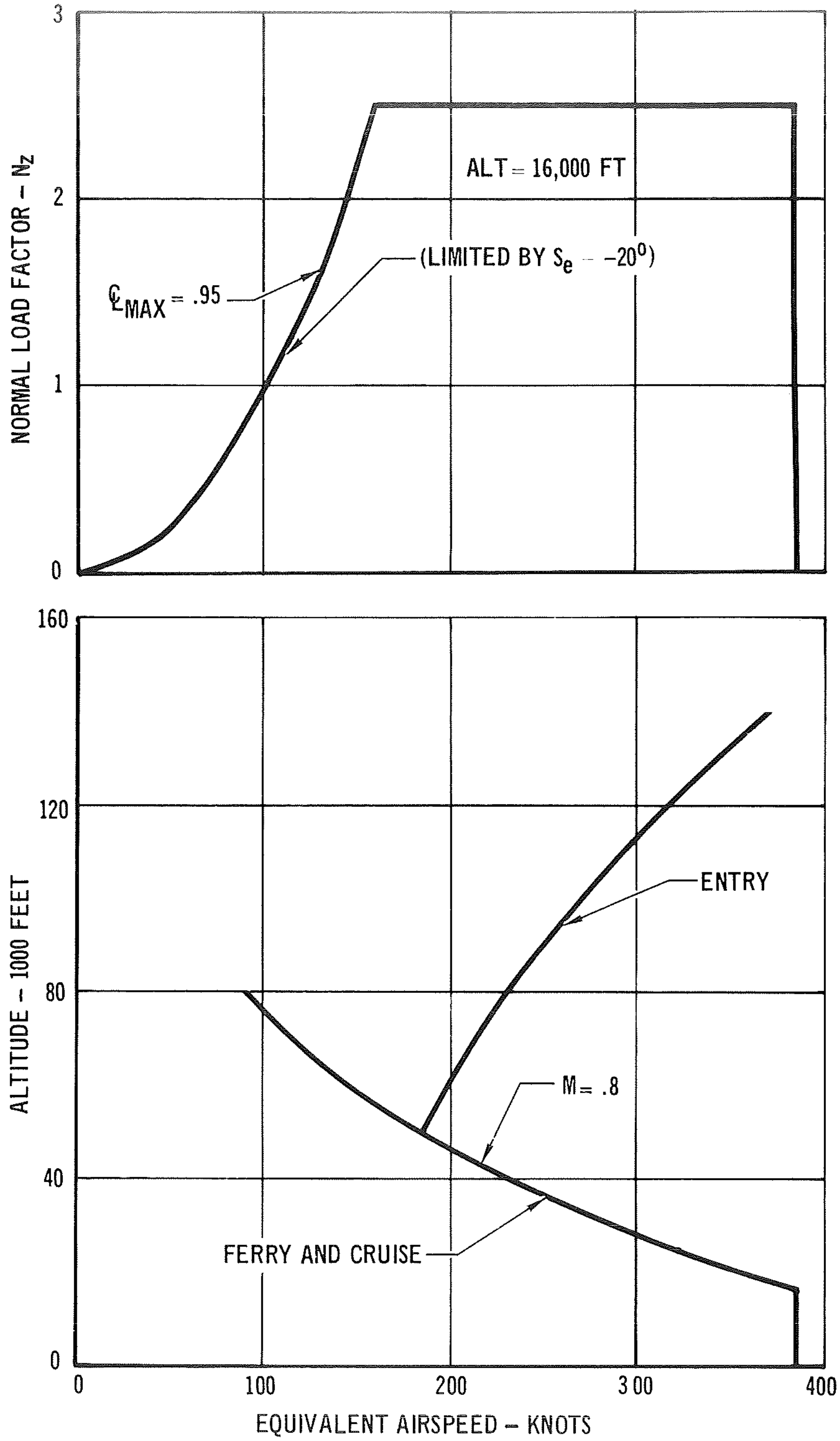


Figure 4-14

ORBITER AIRPLANE FLIGHT ENVELOPES

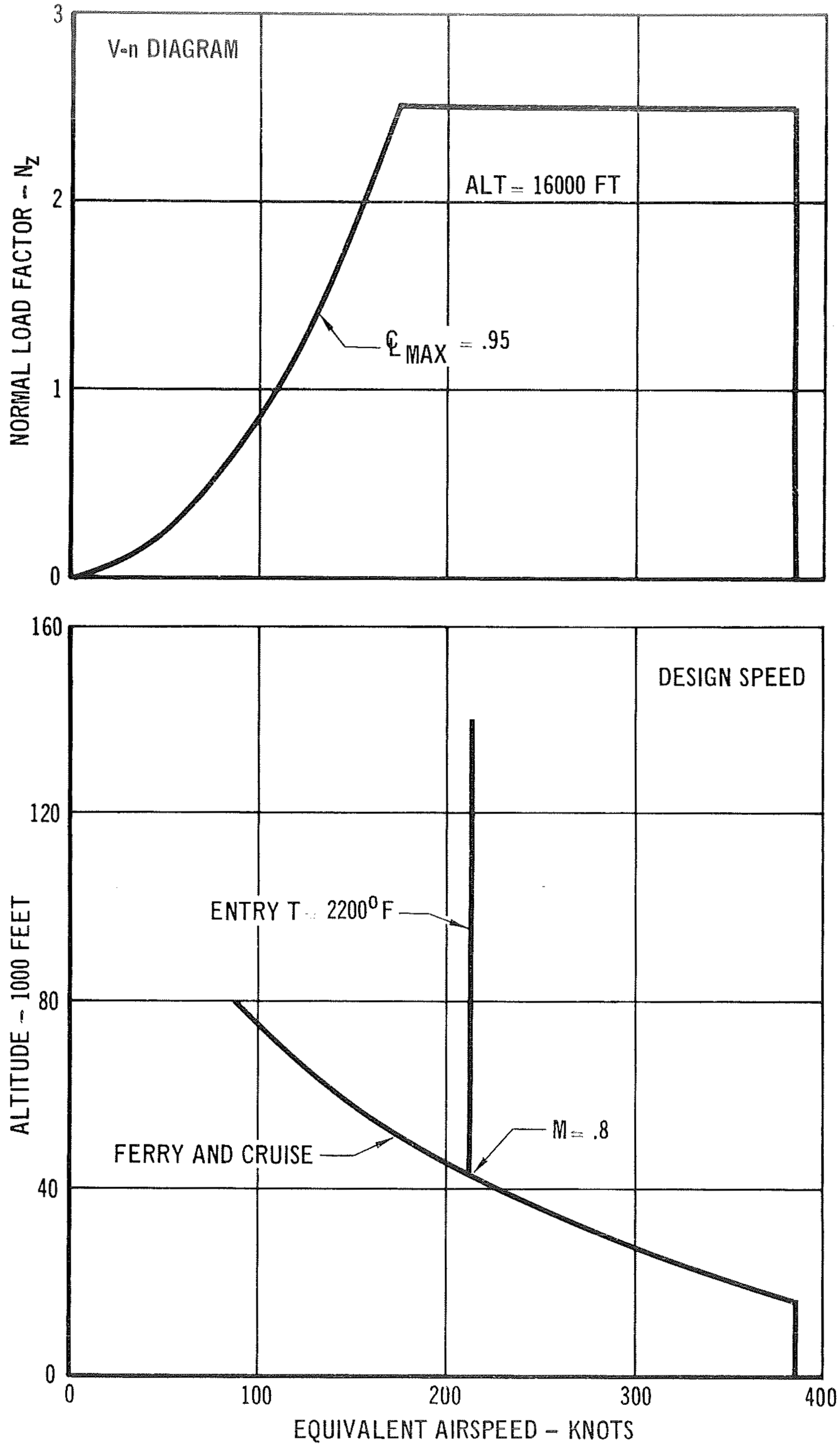


Figure 4-15

damage is acceptable at design levels provided the gear is functionally capable of one or more landings. The dynamic amplification factors which apply to the rigid body landing loads are defined in Table 4-2.

4.1.2 Loads - The structural loads presented herein are based on the structural design criteria of Section 4.1.1 and the baseline geometry described in Section 3. It is judged that the overall structural requirements presented are more than adequate to support the structural analyses necessary at this stage of the vehicle formulation cycle. These data also provide a basis for judgements relative to conditions that should be emphasized in the contract definition preliminary design phase.

The significant loading conditions which occur during the mission cycle are presented in this section. All loads are ultimate rigid body design values unless otherwise noted. Where factors other than factor of safety are used, it is noted on the figures.

Ground Phase - The ground wind condition results in severe loads on the aft portion of the carrier. The Eastern Test Range 99% ground winds are used. The vehicle is canted 2.75 degrees in pitch for the lift-off weight loading condition. Only winds in the pitch plane are considered. The resulting loads are presented in Figure 4-16.

Lift-off - The lift-off condition produces the most severe axial loads for most of the carrier vehicle. The reason is the lift-off load factor of 1.317 combined with a "pogo" effect of 1.1 and the fact that a large mass item (LOX) is high in the structure. The loads distribution for this condition is shown in Figure 4-17.

Ascent - The maximum aerodynamic loading during ascent usually occurs just prior to maximum dynamic pressure. In lieu of running wind shear response time histories, it was assumed that the maximum increment angle of attack experienced was 5 degrees. This angle is the result of an examination of the response of other vehicles to wind shear in conjunction with a load alleviating control systems. This angle is conservatively assumed to occur at maximum dynamic pressure. The resulting loads are shown in Figures 4-18 and 4-19 for conditions where the wind induces a positive and a negative increment of angle of attack.

Burnout - The maximum longitudinal acceleration occurs just prior to carrier burnout. At this time the orbiter is at maximum gross weight and most of the

CARRIER DESIGN LOADS

GROUND WIND
99% + GUST (ETR)

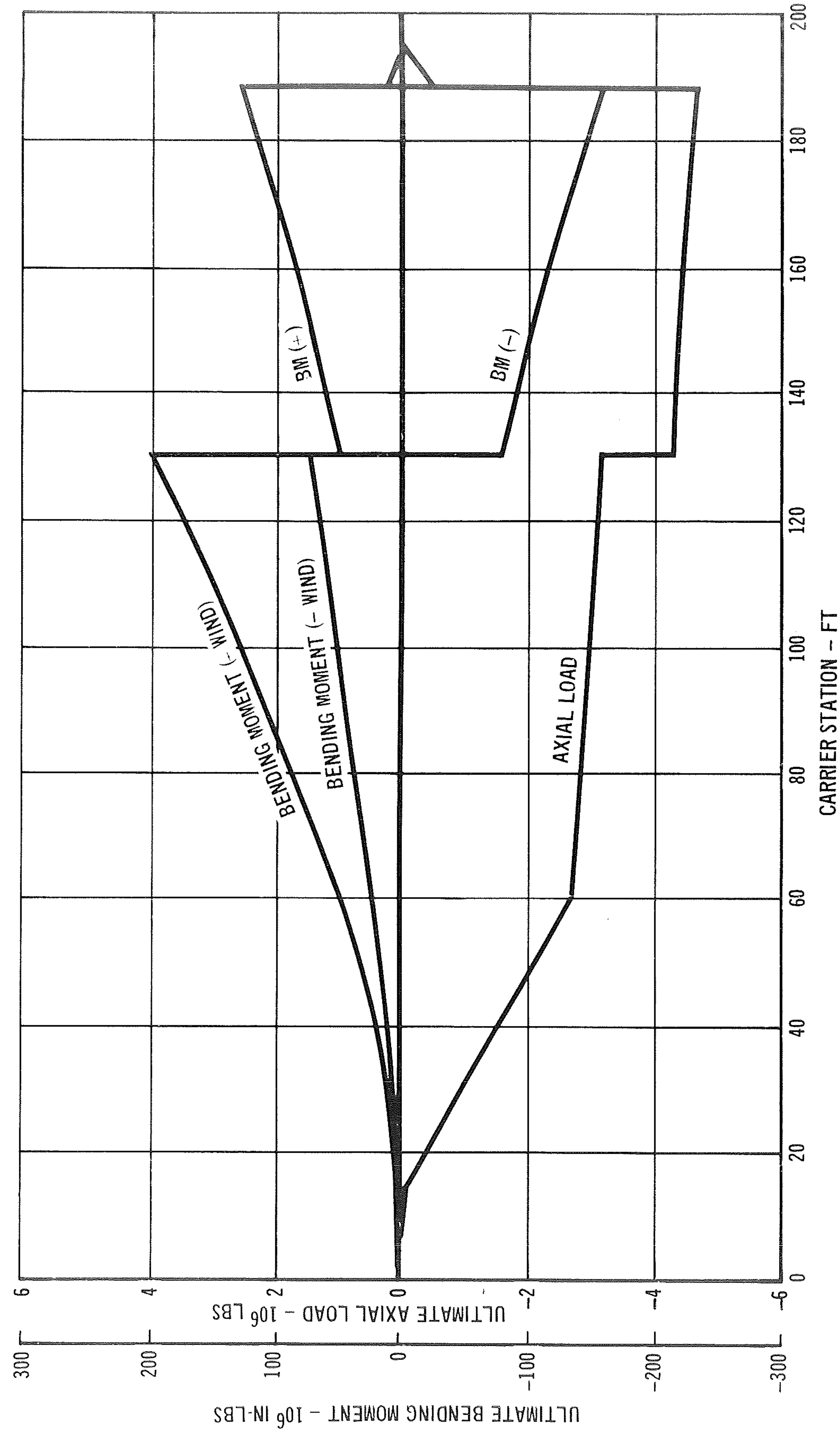


Figure 4-16

CARRIER DESIGN LOADS

LIFT OFF $N_x = 1.317$

POGO FACTOR = 1.1

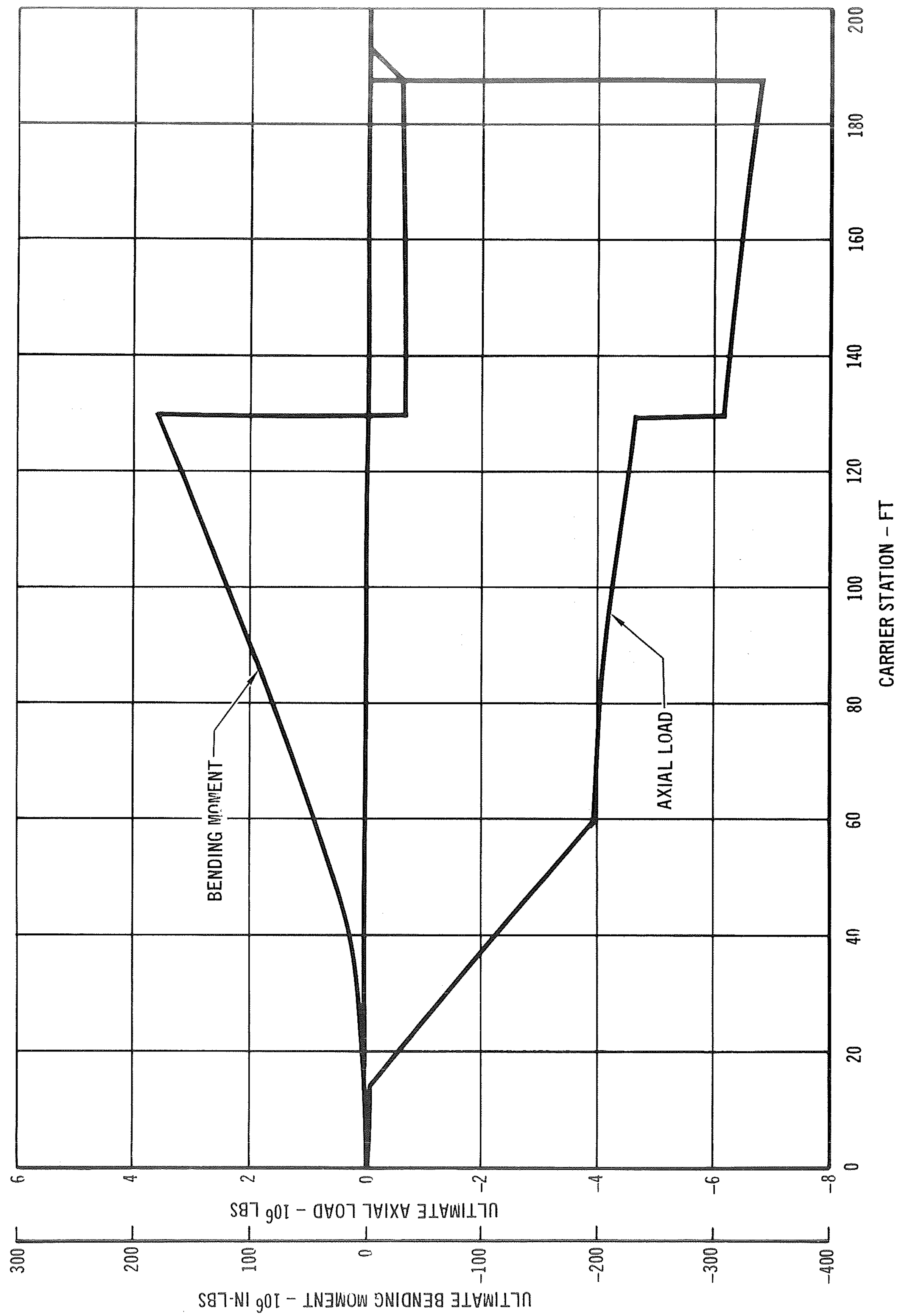


Figure 4–17

CARRIER DESIGN LOADS

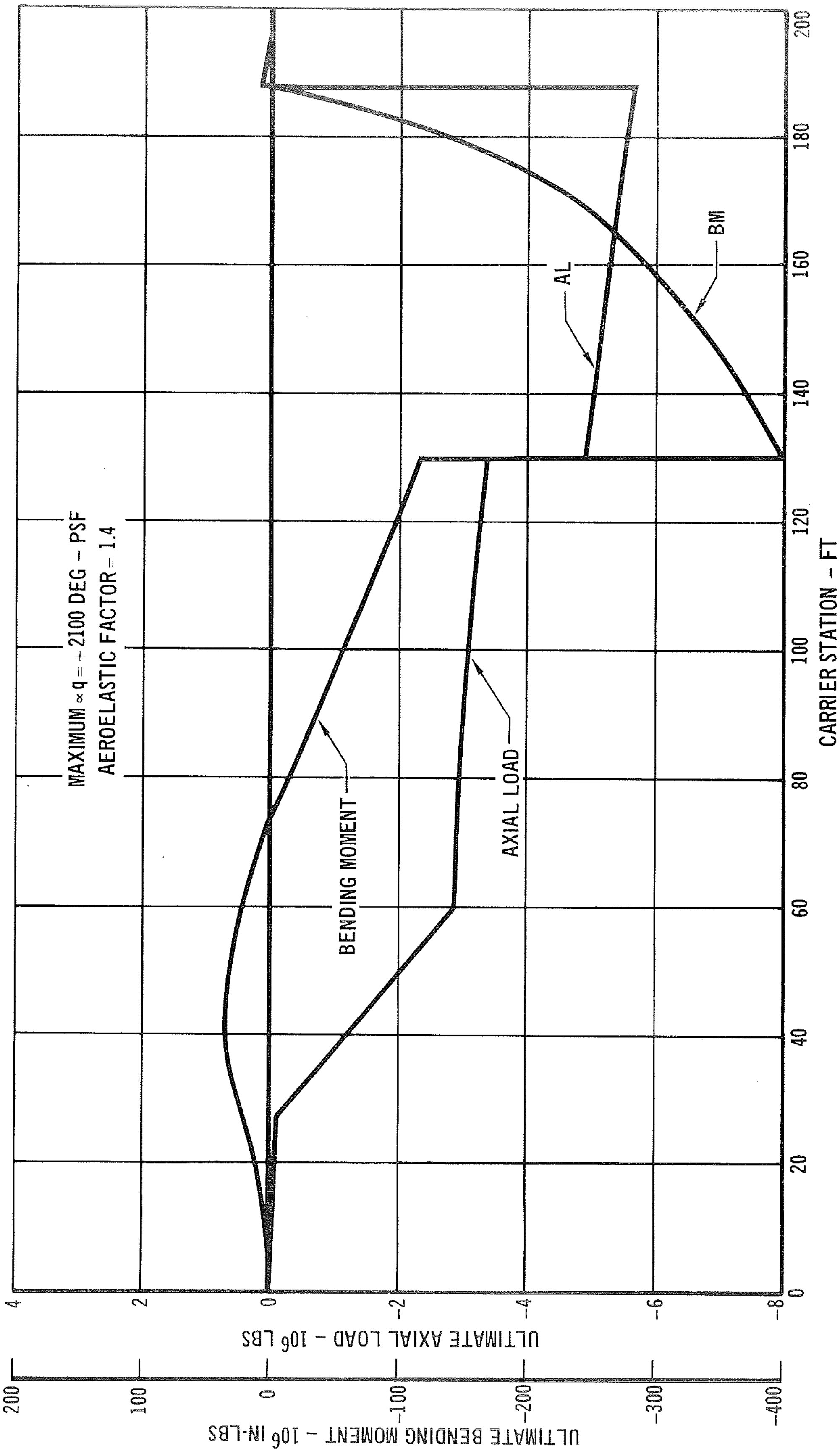


Figure 4–18

CARRIER DESIGN LOADS

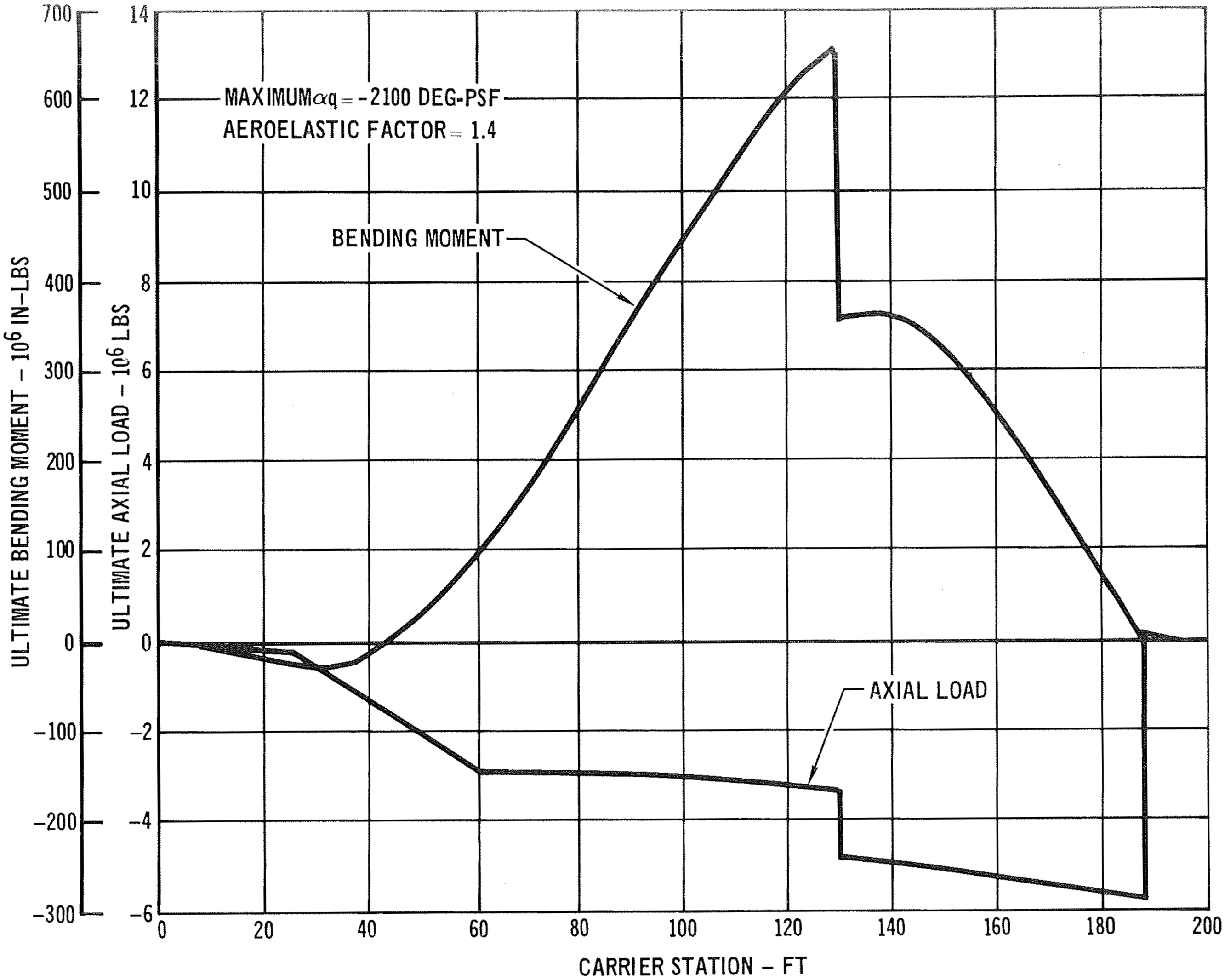


Figure 4-19

propellant has been expended in the carrier. The thrust force is directed to pass through the resultant center of gravity of the two vehicles requiring a thrust vector angle of 13.1 degrees. The loads resulting from this condition on both the carrier and the orbiter are shown in Figures 4-20 and 4-21 respectively. Orbiter body shell loads shown by the shear lag curve in Figure 4-21 are corrected for concentrated loads applied to lower longerons. Longeron loads are assumed to vary linearly from peak load at station 41 to zero at station 73.

Landing - The design sink speed for both the carrier and orbiter is 10 feet per second. The design loads on the fuselage during landing result from a 2 point landing with each main gear design load equalling the landing weight. This results in a normal load factor of 3.0 (including lift). The distributed loads for this condition are shown in Figures 4-22 and 4-23 for the carrier and orbiter, respectively.

Landing Gear Loads - Landing gear design loads for the main gear and nose gear of both the carrier and orbiter are summarized in Tables 4-3 and 4-4.

Pressures and Temperatures - The pressure and temperature history occurring on the bottom of the orbiter during entry are shown in Figure 4-24. A summary of pressure/temperature points on the carrier and orbiter are presented in Table 4-5.

4.1.3 Structural Analysis - Summaries of design conditions, selected materials and stress analysis of typical components of the body structure and heat protection shingles are presented in the following sections.

Carrier Structural Analysis, Integral Tank Structure - The structure is a ring stiffened shell with integral longitudinal stiffeners and is designed to carry both body loads and propellant tank pressure loads. Aluminum alloy 2021-T81 is selected for this application because of its excellent weld characteristics and good mechanical properties at cryogenic, room and elevated temperatures. The two loading conditions which design the shell are internal tank pressure forward of Body Station 70 feet and the launch maximum αq condition for portions of the body aft of B.S. 70 feet. Loads at B.S. 130 feet are used to illustrate a typical shell analysis as shown in Figure 4-25. Figure 4-26 shows the required equivalent sidewall thickness for other body stations.

Carrier Structural Analysis, Thrust Structure - The sidewall structure is a ring stiffened shell with longitudinal hat stiffenerrs. Aluminum alloy 7178-T6 is selected for this application because of its high strength/weight ratio at

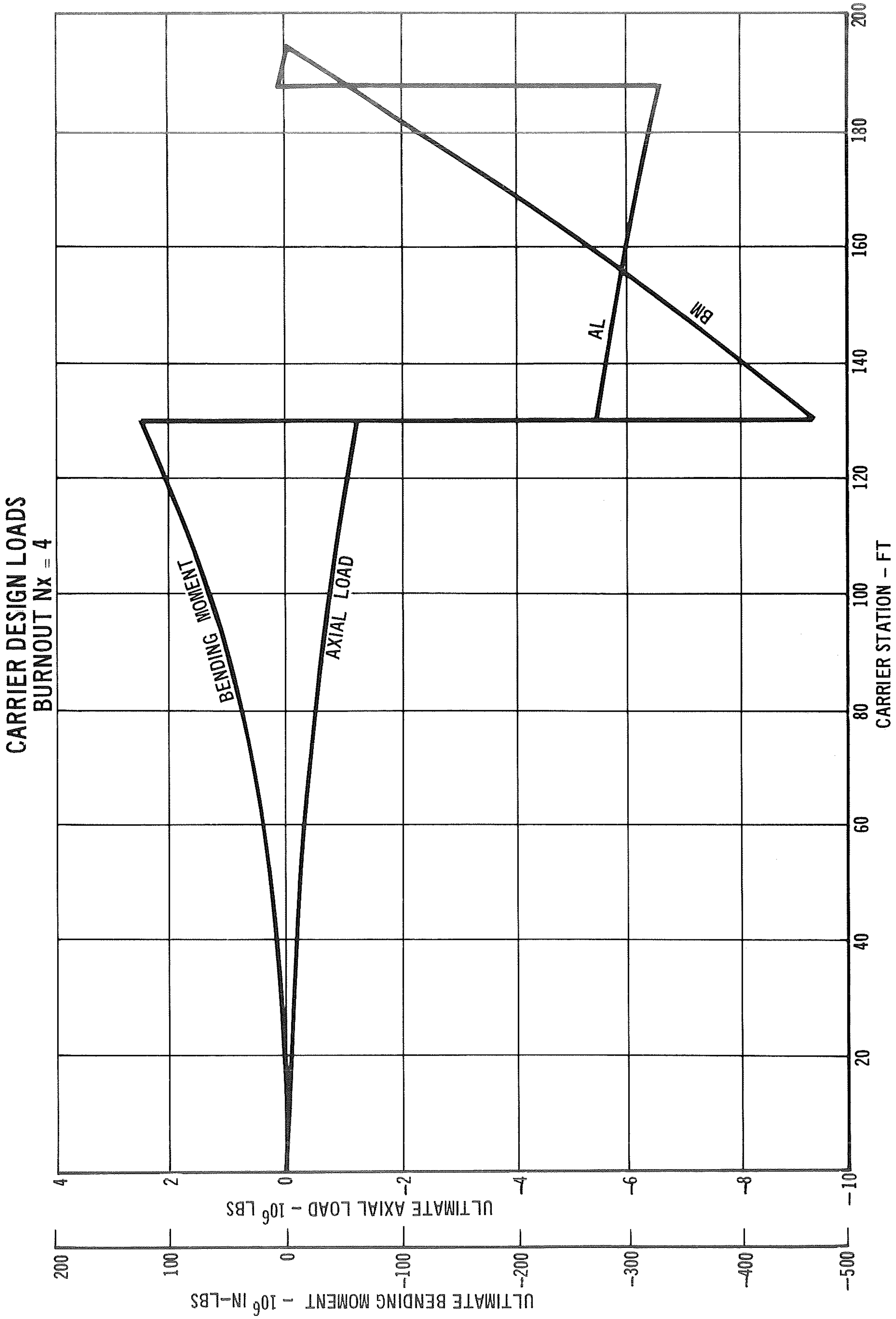

CARRIER DESIGN LOADS
BURNOUT Nx = 4
BENDING MOMENT
AXIAL LOAD
AL
BM
CARRIER STATION – FT
0
20
40
60
80
100
120
140
160
180
200
ULTIMATE AXIAL LOAD – 10^6 LBS
4
2
0
-2
-4
-6
-8
-10
ULTIMATE BENDING MOMENT – 10^6 IN-LBS
200
100
0
-100
-200
-300
-400
-500

Figure 4-20

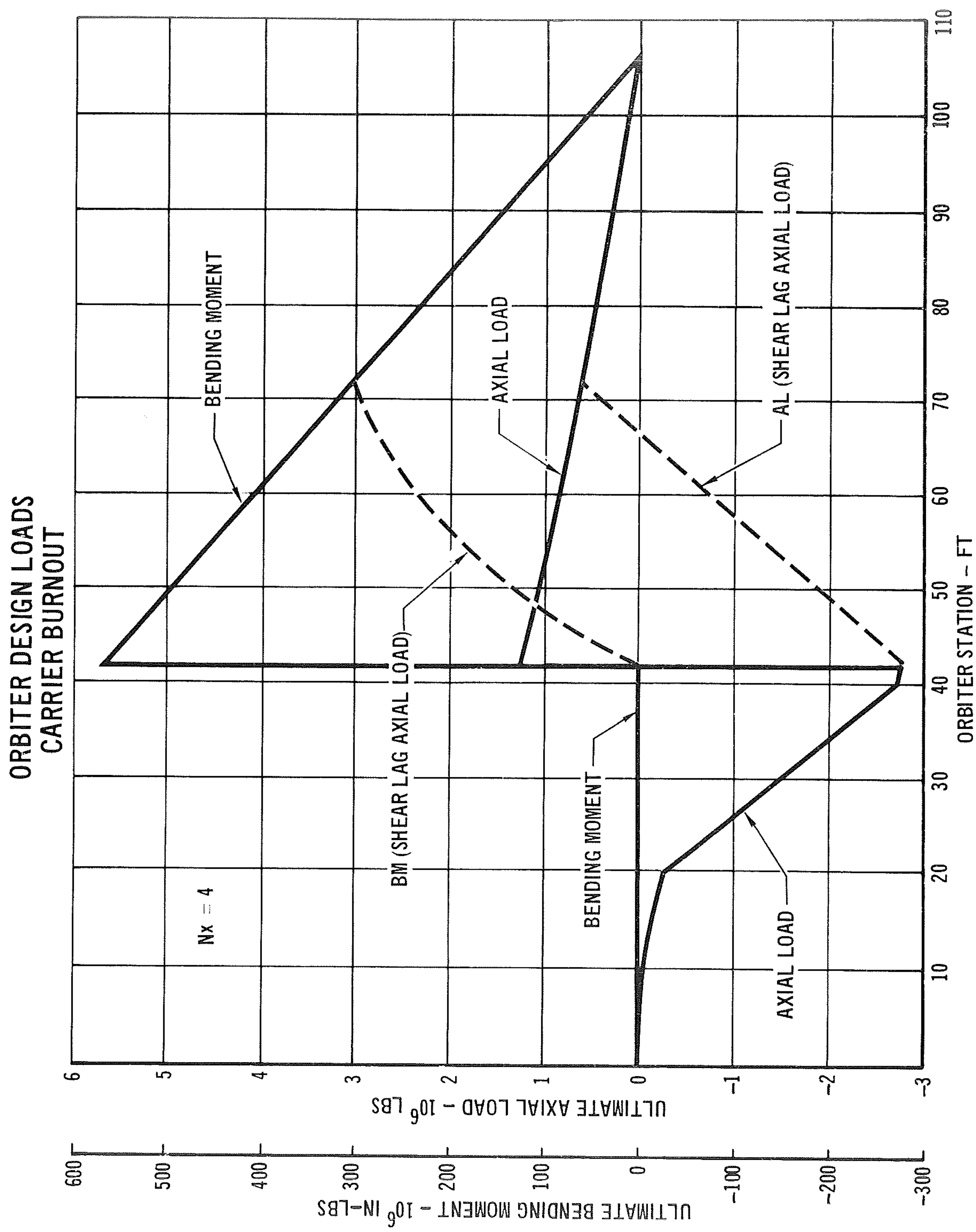

ORBITER DESIGN LOADS
CARRIER BURNOUT
Nx = 4
BENDING MOMENT
AXIAL LOAD
AL (SHEAR LAG AXIAL LOAD)
BM (SHEAR LAG AXIAL LOAD)
BENDING MOMENT
AXIAL LOAD
ORBITER STATION – FT
ULTIMATE AXIAL LOAD – 10^6 LBS
ULTIMATE BENDING MOMENT – 10^6 IN-LBS
10
20
30
40
50
60
70
80
90
100
110
6
5
4
3
2
1
0
−1
−2
−3
600
500
400
300
200
100
0
−100
−200
−300

Figure 4–21

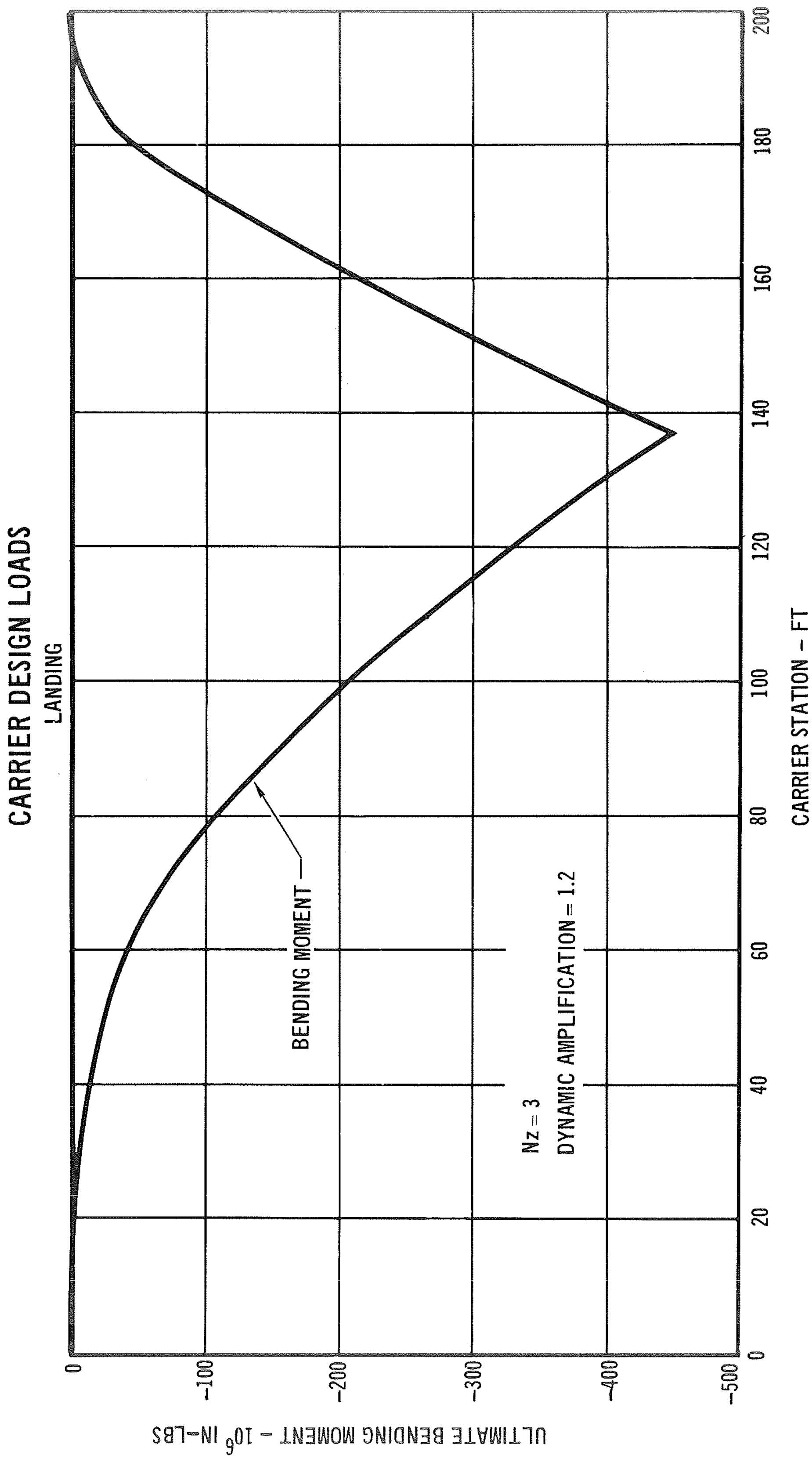

CARRIER DESIGN LOADS
LANDING
BENDING MOMENT
Nz = 3
DYNAMIC AMPLIFICATION = 1.2
CARRIER STATION – FT
ULTIMATE BENDING MOMENT – 10^6 IN-LBS
0
20
40
60
80
100
120
140
160
180
200
0
-100
-200
-300
-400
-500

Figure 4-22

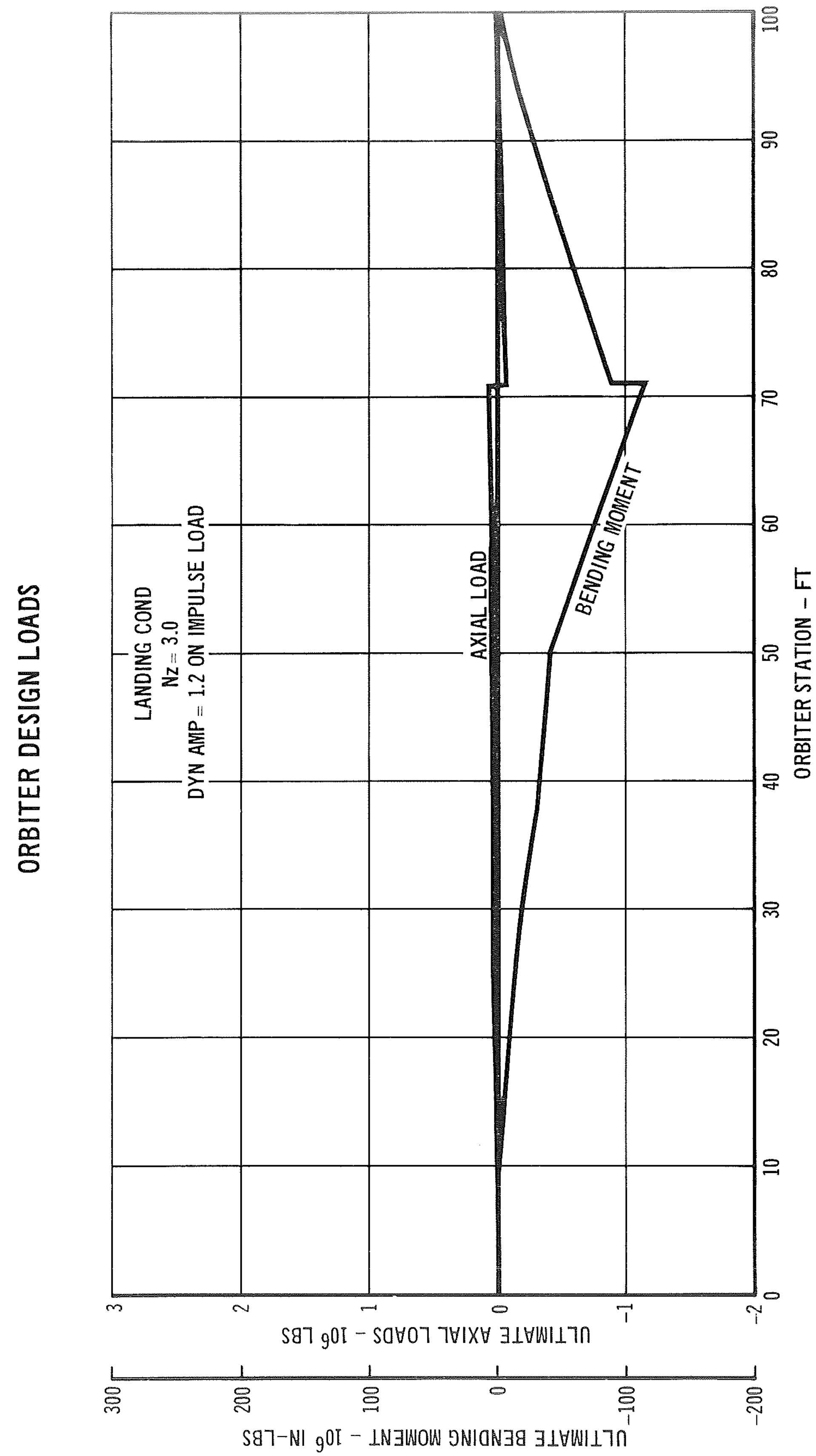

ORBITER DESIGN LOADS
LANDING COND
Nz = 3.0
DYN AMP = 1.2 ON IMPULSE LOAD
AXIAL LOAD
BENDING MOMENT
ORBITER STATION – FT
0
10
20
30
40
50
60
70
80
90
100
ULTIMATE AXIAL LOADS – 10^6 LBS
3
2
1
0
–1
–2
ULTIMATE BENDING MOMENT – 10^6 IN-LBS
300
200
100
0
–100
–200

Figure 4–23

Table 4–3

CARRIER LANDING GEAR DESIGN LOADS

MAIN GEAR

	2 Point Landing				Braking			Turning	Pivoting	Taxiing
	Spin Up	Spring Back	Drift In	Drift Out	2pt	Unsym	Reverse			
Vert	451000	451000	226000	226000	379000	194000	316000	361000	210000	420000
Drag	337000	-310000	0	0	303000	155000	-255000	0	0	0
Side	0	0	-180000	135000	0	220300	0	-181000	$T = \pm 13.8 \times 10^6$ in-lb	0

NOSE GEAR

	Fwd CG		Braking	Turning	Taxiing	Towing	
	Spin Up	Spring Back	Unsym			Fore & Aft	45°
Vert	226000	226000	252000	210000	420000	210000	210000
Drag	174000	-155000	0	0	0	$\pm$ 94600	$\pm$33500
Side	0	0	$\pm$ 40600	$\pm$105000	0	0	$\pm$33500

Table 4-4

ORBITER LANDING GEAR DESIGN LOADS

MAIN GEAR

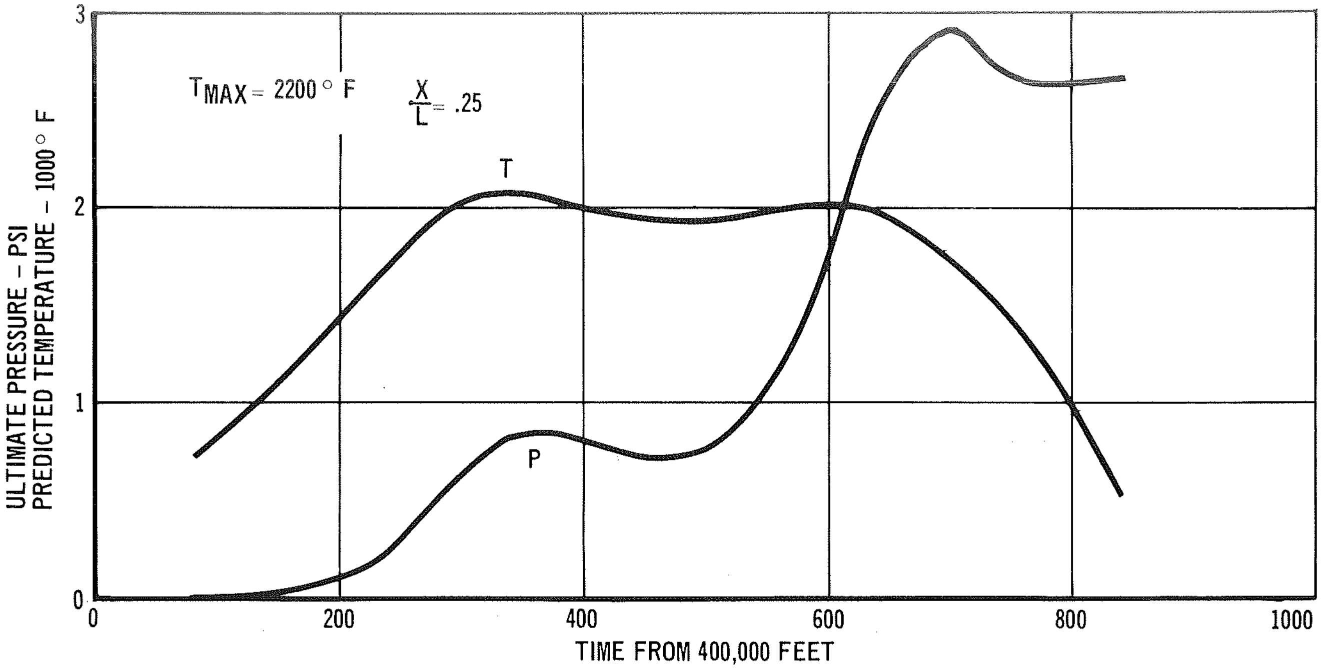
	2 pt Landing				2pt	Braking Unsym	Reverse	Turning	Pivoting	Taxiing
	Spin Up	Spring Back	Drift In	Drift Out						
Vert	186000	186000	93000	93000	156000	91000	130000	201000	103000	206000
Drag	143000	-128000	0	0	125000	72800	-104000	0	0	0
Side	0	0	-73400	558000	0	8000	0	-101000	$T=2.5x10^6$ in-lbs.	0

NOSE GEAR

	Fwd CG		Braking Unsym	Turning	Taxiing	Towing	
	Spin Up	Spring Back				Fore & Aft	45°
Vert	93000	93000	117000	78000	107000	53700	53700
Drag	71500	-64000	0	0	0	±39000	±13800
Side	0	0	±16000	±26800	0	0	±13800

ORBITER DESIGN ENTRY CONDITIONS

Design Entry Pressure/Temperature History



Design Entry Pressure Distribution At Maximum Dynamic Pressure

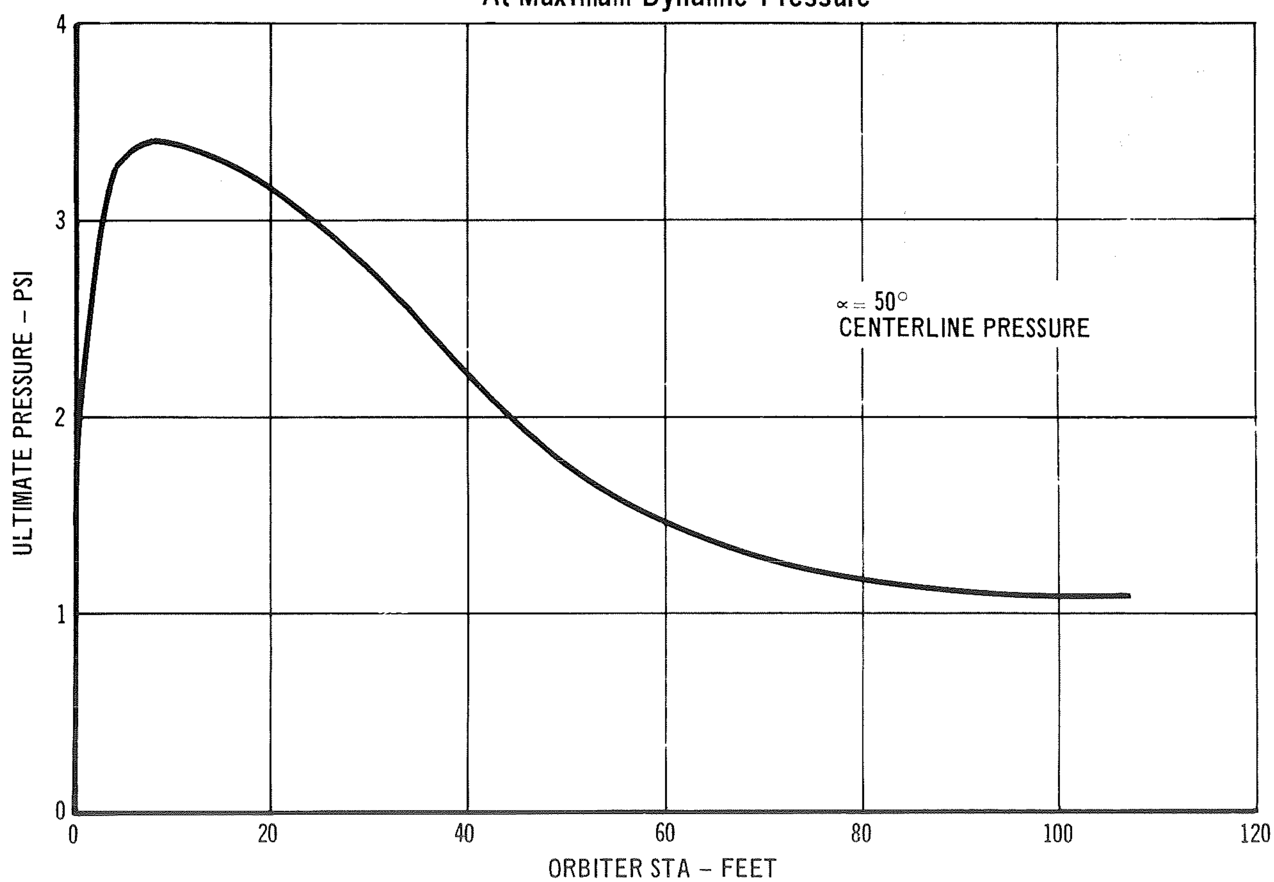


Figure 4-24

Table 4-5

SUMMARY OF DESIGN PRESSURES AND TEMPERATURES

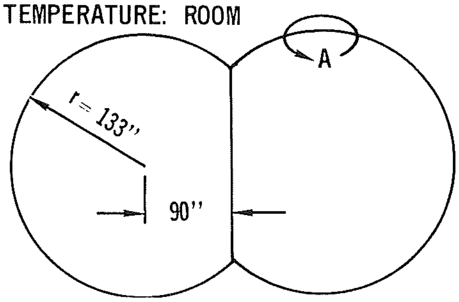
	CARRIER		ORBITER	
	ULTIMATE PRESSURE PSI	PREDICTED TEMPERATURE DEG F	ULTIMATE PRESSURE PSI	PREDICTED TEMPERATURE DEG F
TANK PRESSURES				
LOX	56	–	56	–
LH_2	42	–	42	–
EXTERNAL PRESSURES				
ASCENT				
BETWEEN VEHICLES	4.4	100	4.4	100
TOP SURFACE	2.2	100	2.2	100
ENTRY				
X/L =0.25	1.0	820	2.75	1940
X/L = 0.5	1.0	800	1.50	1560

STRENGTH ANALYSIS – INTEGRAL TANK STRUCTURE
Carrier

LOADS FOR B.S. 130 FROM FIGURE 4–19 FOR THE ASCENT MAXIMUM α q CONDITION ARE:

ULTIMATE BENDING MOMENT $M = 650 \times 10^6$ IN.-LB
ULTIMATE AXIAL LOAD $P = -3.4 \times 10^6$ LB

NOMINAL TANK PRESSURE $p = 30$ LB/IN.^{2}
TEMPERATURE: ROOM



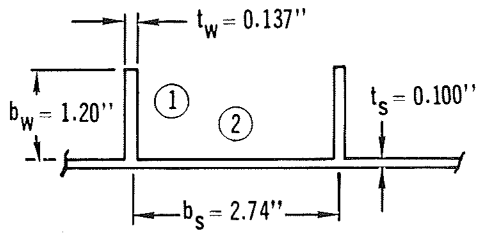
SHELL GEOMETRY

MATL: 2021–T81 ALUMINUM ALLOY
$F_{TY} = 66{,}000$ LB/IN.^{2}
$F_{CY} = 59{,}000$ LB/IN.^{2}
$E = 10.5 \times 10^6$ LB/IN.^{2}
SECTION MODULUS $= 0.985 \times 10^5 \bar{t}$
SECTION AREA $= 1220 \bar{t}$

MAXIMUM RUNNING LOAD/INCH $P^1 = \left(\frac{M}{Z} + \frac{P}{A}\right)\bar{t} = \frac{-650 \times 10^6}{.985 \times 10^5} - \frac{3.4 \times 10^6}{1220} = -9430$ LB/IN.

LIMIT PRESSURE RELIEF LOAD $= \frac{pr}{2} = \frac{30 \times 133}{2} = 2000$ LB/IN.

DESIGN RUNNING LOAD/IN $P_D = -9430 + 2000 = -7430$ LB/IN.



VIEW A

EQUIVALENT THICKNESS $\bar{t} = 0.160$ IN.
MOMENT OF INERTIA $I = 0.023$ IN.^{4}/IN.

$$f_c = \frac{P_D}{\bar{t}} = \frac{7430}{.16} = 46{,}500 \text{ LB/IN.}^2$$

CRIPPLING CHECK
(REFERENCE 1)

ITEM	b IN.	t IN.	b/t	b x t IN.^{2}	F_{CC} LB/IN.^{2}	b x t x F_{CC} LB
1	1.20	0.137	8.8	0.164	46,500	7,640
2	2.74	0.100	27.4	0.274	46,500	12,730
				0.438		20,370

$F_{CC_{AVG}} = \Sigma\, b \times t \times F_{CC} / \Sigma\, b \times t = 46{,}500$ LB/IN.^{2}

SHELL BUCKLING CHECK

$$F_{cr} = \frac{2E}{R\bar{t}} \sqrt{t_s\, I} = \frac{2 \times 10.5 \times 10^6}{133 \times 0.160} \sqrt{0.100 \times 0.023} = 47{,}200 \text{ LB/IN.}^2$$

$$M.S. = \frac{F_{CC}}{f_c} - 1 = \frac{46{,}500}{46{,}500} - 1 = 0.0$$

Figure 4–25

CARRIER INTEGRAL TANK STRUCTURE
REQUIRED EQUIVALENT THICKNESS VS. BODY STATION

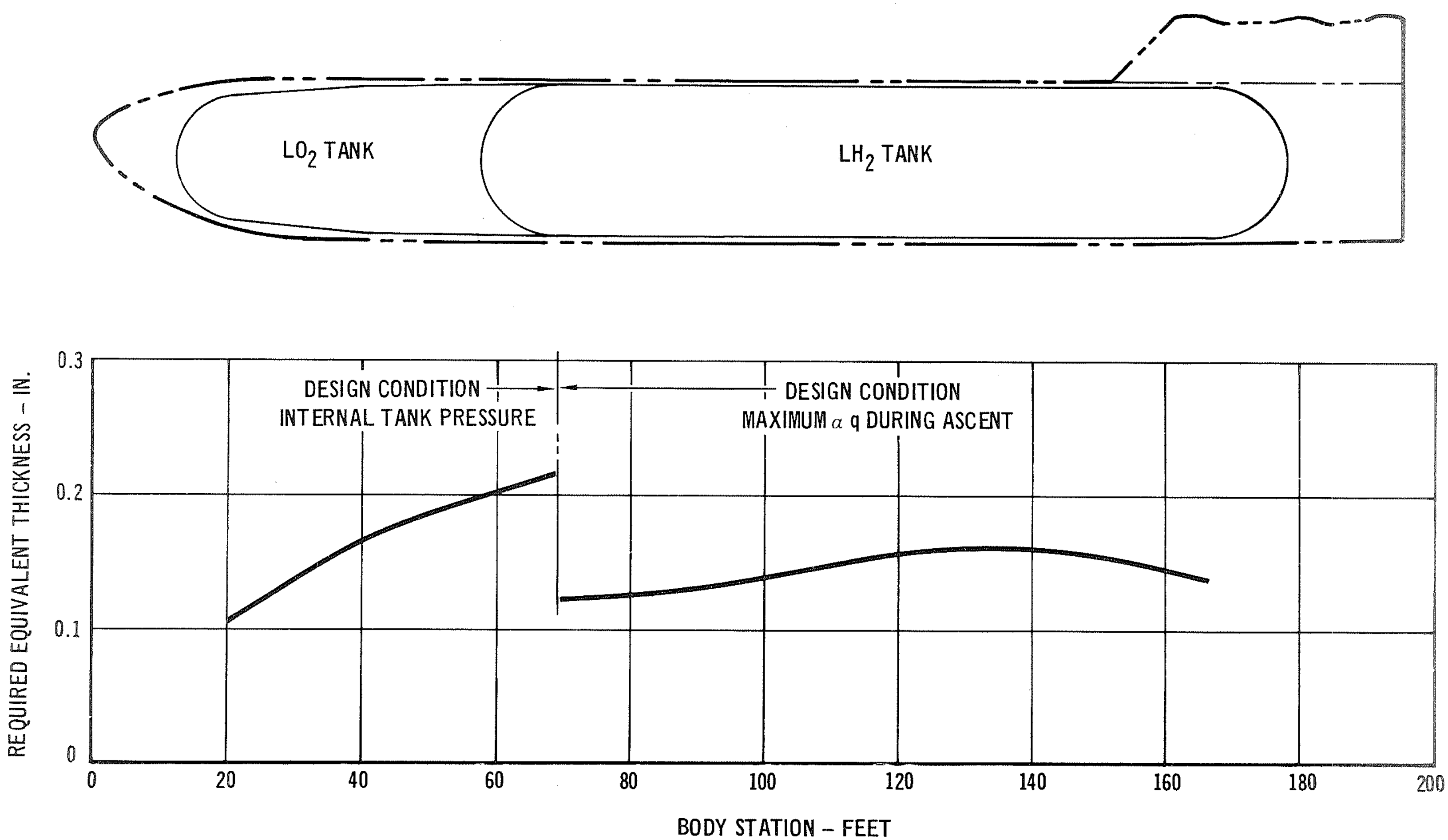


Figure 4-26

room and moderately elevated temperatures. The shell structure is designed by the ascent maximum acceleration loading condition. Each stiffener with effective skin is assumed to act as a short column supported at each ring. Loads at Body Station 170 feet are used to illustrate a typical stringer analysis as shown in Figure 4-27.

Carrier Structural Analyses, Heat Protection - The panel and support design must provide adequate strength to react surface pressure laods. Trade studies were conducted to determine an efficient panel configuration. These studies resulted in use of a smooth skin stiffened with trapezoidal shaped corrugations.

Studies were conducted to determine the optimum support frame spacing. A radiative titanium panel (8AL-1Mo-1V alloy) located on the bottom centerline near B.S. 70 feet is used to illustrate the optimization. The design condition occurs during the launch maximum αq condition when the external surface pressure is 2.2 psi ultimate and the surface temperature is 100°F. The corrugated panel does not have any restraint in the direction transverse to the corrugation and is therefore assumed to act as a beam simply supported at the frames. The lower portion of the frames are assumed to act as fixed end beams with a center support. The results of the study are shown in Figure 4-28. Twenty inch frame spacing is selected for design. This is slightly less than optimum, however, this spacing is more advantageous from the standpoint of panel deflections and requires smaller thermal expansion joints. A titanium panel with supports at twenty inch spacing is used to illustrate a typical sizing analysis shown in Figure 4-29. Past studies have indicated that internal pressure is not the design condition with this panel configuration.

Single thickness beaded titanium panels are used on the shadowed upper surface in regions which experience low heating rates. Analysis of a beaded panel is shown in Figure 4-30.

Orbiter Structural Analysis, Integral Tank Structure - The structure is a frame-stiffened, irregular-shaped shell with integral longitudinal stiffeners and is designed to carry body loads and propellant tank pressure loads simultaneously. Pressure loads are beamed to bi-axially loaded tension web/baffles. Similar to the carrier tanks, aluminum alloy 2021-T81 is selected for this application. The loading condition which designs the integral tank structure is maximum acceleration during ascent combined with tank pressures. Orbiter Station 720.00 inches is used to present a typical shell strength analysis as shown in Figure 4-31.

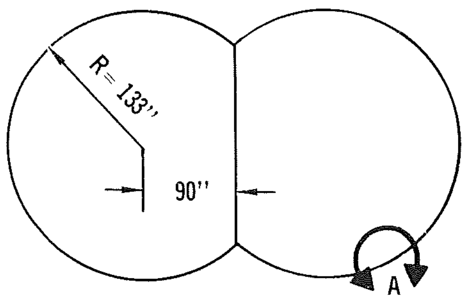
STRENGTH ANALYSIS – THRUST STRUCTURE STRINGERS

Carrier

LOADS FOR B.S. 170 FROM FIGURE 4-20 FOR THE ASCENT MAXIMUM ACCELERATION CONDITIONS ARE:

ULTIMATE BENDING MOMENT $M = -190 \times 10^6$ IN.-LB
ULTIMATE AXIAL LOAD $P = -6.1 \times 10^6$ LB

TEMPERATURE ROOM:

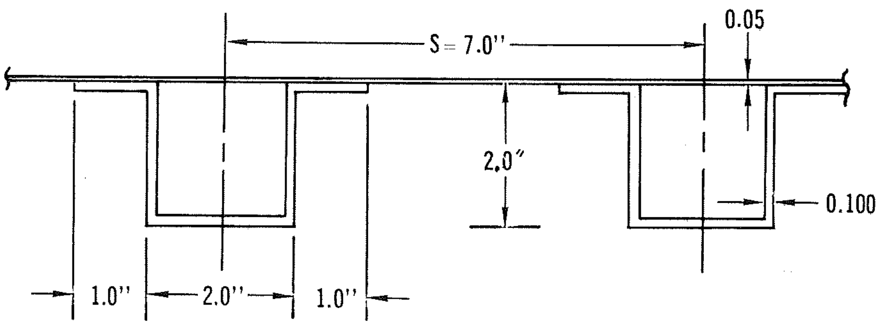


SHELL GEOMETRY

MATERIAL: 7178-T6 ALUMINUM ALLOY
$F_{TU} = 80{,}000$ LB/IN.^{2}
$F_{CY} = 71{,}000$ LB/IN.^{2}
$E_C = 10.5 \times 10^6$
(REFERENCE 3)

SECTION MODULUS $Z = 0.985 \times 10^5\,\bar{t}$
SECTION AREA $A = 1220\,\bar{t}$

RUNNING LOAD/INCH $P' = \left(\frac{M}{Z} + \frac{P}{A}\right)\bar{t} = \left(\frac{-190 \times 10^6}{.985 \times 10^5} - \frac{6.1 \times 10^6}{1220}\right) = -6930$ LB/IN.



VIEW A

STRINGER AREA = 0.92 IN.^{2}
(WITH EFFECTIVE SKIN)

RADIUS OF GYRATION = 0.81 IN.

$$F_{cr} = F_{cc} - \frac{F_{cc}^2}{4\pi^2 E}\left(\frac{L}{P}\right)^2$$

AVERAGE CRIPPLING STRESS $F_{cc} = 57{,}600$ LB/IN.^{2}
(REFERENCE 1)

SUPPORT SPACING L = 20 IN.

$$F_{cr} = 57{,}600 - \frac{\overline{57600}^2}{4\pi^2 \times 10.5 \times 10^6}\left(\frac{20}{.81}\right)^2 = 52700 \text{ LB/IN.}^2$$

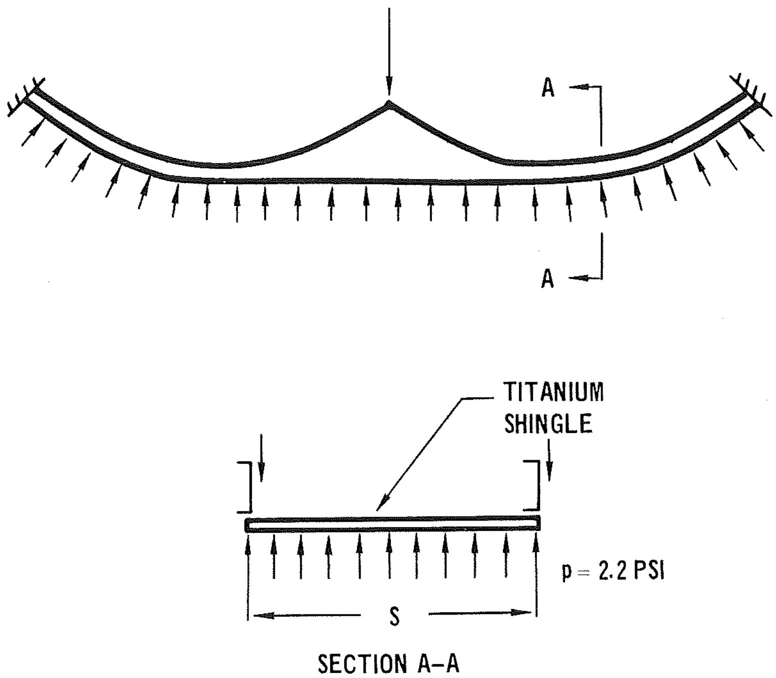
$P_{cr} = F_{cr}A = 52700 \times 0.92 = 48{,}500$ LB

APPLIED LOAD $P = P' \times S = 6930 \times 7.0 = 48{,}500$ LB

$$\text{M.S.} = \frac{P_{cr}}{P} - 1 = \frac{48{,}500}{48{,}500} - 1 = +0.0$$

Figure 4-27

HEAT PROTECTION STRUCTURE WEIGHT – CARRIER



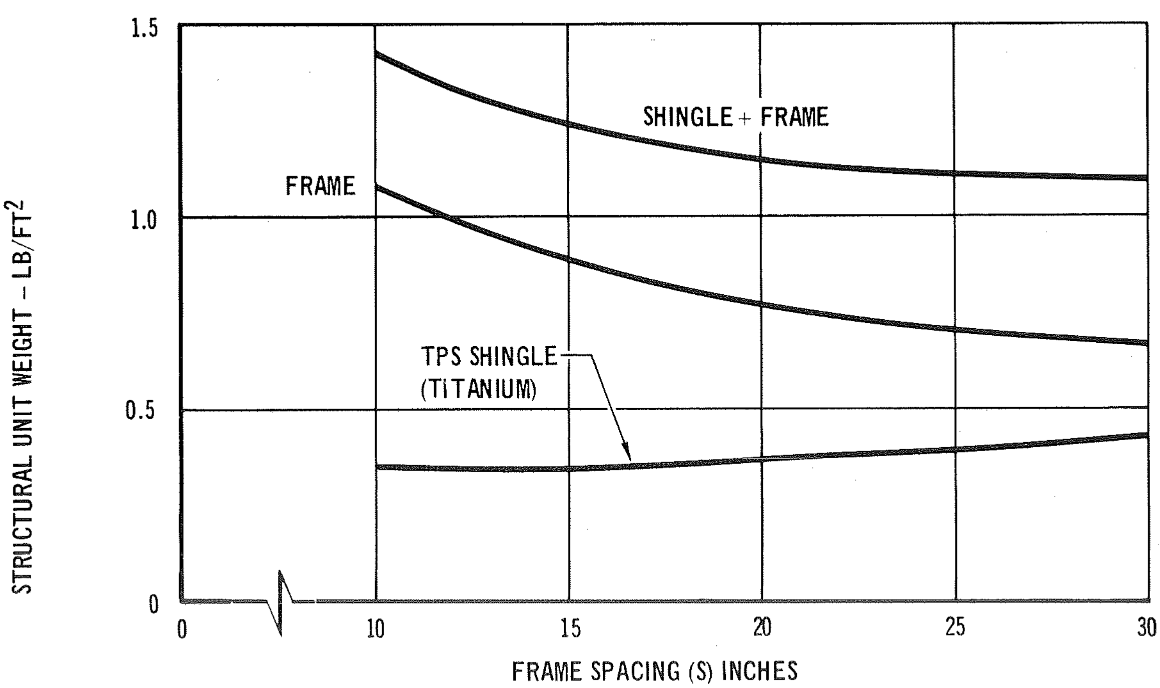
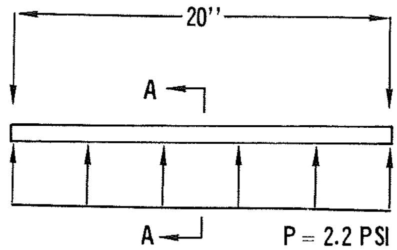


Figure 4–28

STRENGTH ANALYSIS – TPS SHINGLE
Single Faced Corrugation
(Carrier)

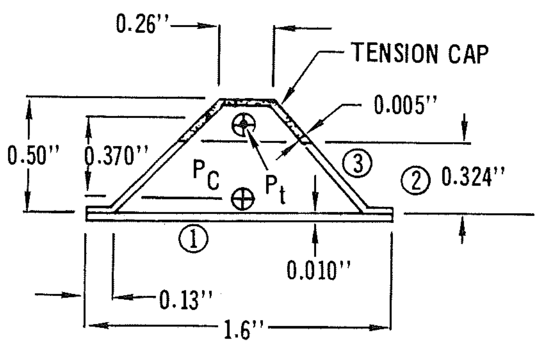
SURFACE TEMPERATURE AND PRESSURE FROM TABLE 4-5 FOR THE ASCENT MAXIMUM α g CONDITION AT B.S. 70 FEET ARE:

ULTIMATE SURFACE PRESSURE	P = 2.2 PSI
SURFACE TEMPERATURE	T = 100°F



MATERIAL: 8Al-1V-1Mo TITANIUM ALLOY
F_{TU} = 142,000 LB/IN.^{2}
F_{CY} = 141,000 LB/IN.^{2} AT 100°F
E = 17.8 x 10^6 LB/IN.^{2}
(REFERENCE 3)

BENDING CHECK OF SECTION A-A



$$M_{MAX} = \frac{wl^2}{8} = \frac{2.2 \times 1.6 \times 20^2}{8} = 176 \text{ IN.-LB}$$

$$P_t = P_C = M/h = \frac{176}{.37} = 475 \text{ LB}$$

COMPRESSION CAP CRITICAL

ITEM	b IN.	t IN.	b/t	AREA IN.^{2}	F_{cc} LB/IN.^{2}	P_{cc} LB
1	1.6	0.010	160	0.016	21,000	336
2	0.255	0.005	51	0.00127	54,000	69
3	0.480	0.005	96	0.0048(2)	32,500	156

$$P_{C(ALL)} = 561 \text{ LB}$$

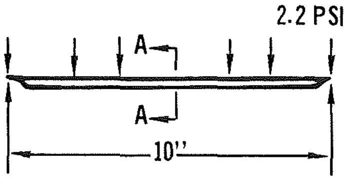
$$\text{M.S.} \frac{P_{C(ALL)}}{P_C} - 1 = \frac{561}{475} - 1 = +0.18$$

Figure 4-29

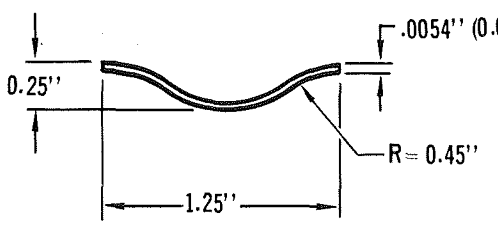
STRENGTH ANALYSIS – TPS SHINGLE
Beaded Panel
(Carrier)

SURFACE TEMPERATURE AND PRESSURE FROM TABLE 4-5 FOR THE ASCENT MAXIMUM α q CONDITION AT B.S. 70 ARE:

ULTIMATE SURFACE PRESSURE $P = 2.2$ PSI
SURFACE TEMPERATURE $T = 100^{\circ}F$



MATERIAL: 8 AL-1V-1Mo TITANIUM ALLOY
$F_{TU} = 142{,}000$ PSI
$F_{CY} = 141{,}000$ PSI AT $100^{\circ}F$
$E = 17.8 \times 10^{6}$ PSI
(REFERENCE 3)



SECTION MODULUS $Z = 6.25 \times 10^{-4}$ IN.^{3}/BEAD

$$M_{MAX} = \frac{wl^2}{8} = \frac{2.2 \times 1.25 \times 10^2}{8} = 34.4 \text{ IN.-LB/BEAD}$$

$$M_{ALL} = F_{ALL}\, Z$$

$$F_{ALL} = F_{cr} = \frac{0.3\, Et}{R} = \frac{0.3 \times 17.8 \times 10^6 \times 0.0054}{0.45} = 64{,}000 \text{ PSI}$$

$$M_{ALL} = 64{,}000 \times 6.25 \times 10^{-4} = 40.0 \text{ IN.-LB}$$

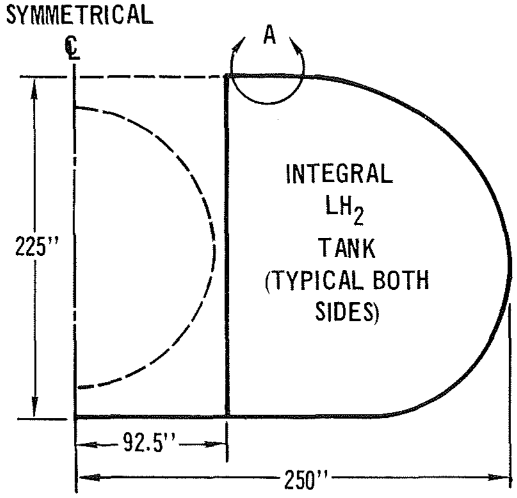
$$MS = \frac{M_{ALL}}{M} - 1 = \frac{40.0}{34.4} - 1 = +0.16$$

Figure 4-30

STRENGTH ANALYSIS – INTEGRAL TANK STRUCTURE
Orbiter

LOADS AT ORBITER BODY STATION 720.0 INCHES FROM FIGURE 4–21
FOR ASCENT, MAXIMUM ACCELERATION CONDITIONS ARE:

ULTIMATE BENDING MOMENT..... $M = 220(10)^6$ IN.-LB
ULTIMATE AXIAL LOAD.......... $P = -0.8(10)^6$ LB
ULTIMATE OPERATING PRESSURE $P = 42$ PSI
ULTIMATE HEAD PRESSURE...... $P_h = 3.0$ PSI



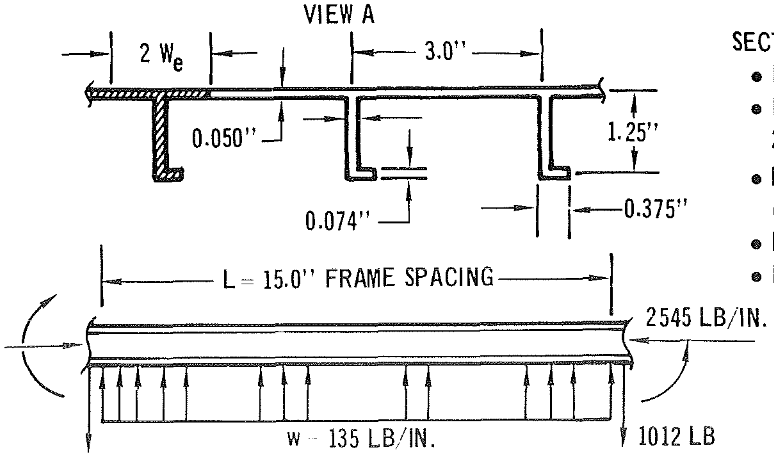
MATERIAL: 2021–T81 ALUMINUM AT 80°F
$F_{TU} = 66,000$ PSI
$F_{CY} = 59,000$ PSI
$E = 10.5(10)^6$ PSI

SECTION PROPERTIES: (REFERENCE FIGURE 4–32)
- SECTION MODULUS, $Z_T = 6.9(10)^4\ \bar{t}$ IN.^{3}
- SECTION AREA, $A_{CS} = 1510\ \bar{t}$ IN.^{2}
- TANK PERIMETER (1 SIDE), $S = 655$ IN.
- TANK CROSS-SECTIONAL AREA (1 SIDE), $A_T = 25,700$ IN.^{2}

$$\text{MAX. RUNNING LOAD PER INCH, } P' = \left(\frac{M}{Z_T} + \frac{P}{A_{CS}}\right)\bar{t} = \frac{-220(10)^6}{6.9(10)^4} - \frac{0.8(10)^6}{1510} = -3720 \text{ LB/IN.}$$

$$\text{LIMIT PRESSURE RELIEF LOAD, } P'_R = \frac{P_o A_T}{S} = \frac{30(25,700)}{655} = 1175 \text{ LB/IN.}$$

$$\text{DESIGN RUNNING LOAD PER INCH, } P'_D = P' + P'_R = -2545 \text{ LB/IN.}$$



SECTION PROPERTIES OF INTEGRAL STIFFENER
- EQUIVALENT THICKNESS, $\bar{t} = 0.090$ IN.
- EFFECTIVE SKIN WORKING WITH STRINGER, $2\ W_e = 24\ t_s = 1.20$ IN.
- MOMENT OF INERTIA OF INTEGRAL STRINGER (SHADED AREA), $I = 0.044$ IN.^{4}
- INTEGRAL STRINGER AREA (SHADED AREA) $A = 0.18$ IN.
- RADIUS OF GYRATION, $\rho_R = 0.495$ IN.

Figure 4–31

BEAM COLUMN ANALYSIS: THE FOLLOWING BEAM – COLUMN EQUATION (REFERENCE 1) IS UTILIZED TO DETERMINE THE ALLOWABLE COLUMN LOAD, P_{ALL}:

$$\frac{P_{cr}}{F_{cc}A}\left(\frac{P_{ALL}}{P_{cr}}\right)^2 - \left[\frac{P_{cr}}{F_{cc}A} + \frac{P_{cr}Y_0}{M_{ALL}} - \frac{M_0}{M_{ALL}} + 1\right]\left(\frac{P_{ALL}}{P_{cr}}\right) - \frac{M_0}{M_{ALL}} + 1 = 0$$

THE CRIPPLING STRESS, IS: $F_{cc} = 66{,}000$ PSI

THE CRITICAL COLUMN LOAD IS OBTAINED FROM JOHNSON'S FORMULAS

$$P_{cr} = F_{cc}A - \frac{F_{cc}^2 (L')^2 A^2}{4\pi^2 EI} = 11035 \text{ LB}$$

DEFLECTION AT THE BEAM-COLUMN'S CENTER IS $Y_0 = \frac{1}{384}\ \frac{wL^4}{EI} = 0.028$ IN.

THE BENDING MOMENT AT THE BEAM-COLUMN'S CENTER IS, $M_0 = \frac{wL^2}{24} = 1265$ IN.-LB

THE ALLOWABLE BENDING MOMENT IS: $M_{ALL} = 6100$ IN.-LB

SUBSTITUTING INTO THE GENERAL BEAM- COLUMN EQUATION AND SOLVING:

$P_{ALL}/P_{cr} = 0.71$, $P_{ALL} = 0.71\ (11{,}035) = 7850$ LB

$P'_{ALL} = P_{ALL}/3'' = 2610$ LB/IN.

THE MARGIN OF SAFETY IS, M.S. $= \frac{P'_{ALL}}{P'_D} - 1 = \frac{2610}{2545} - 1 = +.025$

Figure 4–31 Cont)

Figure 4-32 presents the orbiter geometric section properties utilized throughout the strength analysis. Maximum section modulus based on the distances from the horizontal centroidial axis (x-axis) to the top and bottom extreme fibers are represented by Z_T and Z_B respectively. Decreased values at orbiter Station 500.00 inches account for the structural shell becoming ineffective in the vicinity of payload bay doors. Primary body loads must be transferred around the cutout by shear lagging the loads to the sides of the cutout.

Summarizing the results of strength analyses on integral tank side wall structure, Figure 4-33 shows the required sidewall equivalent thickness for other body stations.

A sidewall configuration other than the integral skin-stringer concept used could prove to be attractive. An aluminum honeycomb sandwich was sized, using the loads and support spacing at Orbiter B.S. 720 inches (Reference Figure 4-31), for comparative purposes. Sandwich weights are nearly typical for other types of construction such as tubular or beaded panels and are used as an indicator for the group. A sandwich panel sized for B.S. 720 loads contains a one inch, 6.1 pounds per cubic foot honeycomb core (5052 alloy) with .033 inch face plates. Panel weight excluding edging members is 1.62 pounds per square foot. The equivalent thickness of the integral skin-stringer arrangement is .090 inches (Reference Figure 4-31) and the weight is 1.30 pounds per square foot. The incremental weight, .32 pounds per square foot, would vary for other load combinations however the value shown is considered indicative of the trend.

<u>Orbiter Structural Analysis, Heat Protection</u> - Similar to the carrier analysis, studies were conducted to define the optimum support frame spacing. Metallic shingles of TD-NiCr alloy material located on the bottom centerline at 25 and 50 percent of the vehicle length (X/L = .25; .50) are used to illustrate the optimization. The corrugated panels are assumed to act as beams simply supported at the frames. The lower portion of the frames are analyzed as fixed end beams. The design condition for both stations occurs during entry 40 seconds before maximum dynamic pressure. At X/L = .25 the external surface pressure is 2.75 psi ultimate and the surface temperature is 1940°F. External surface pressure and temperature at X/L = .50 are 1.5 psi and 1560°F respectively. Results of the study are shown in Figures 4-34 and 4-35. Optimum frame spacing varies from 12 inches at X/L = .25 to 15 inches at X/L = .5. A variable spacing is used between the two stations. A constant spacing of 12 inches and 15 inches is used on the forward and aft sections of the body respectively.

GEOMETRIC SECTION PROPERTIES OF ORBITER SIDEWALL STRUCTURE

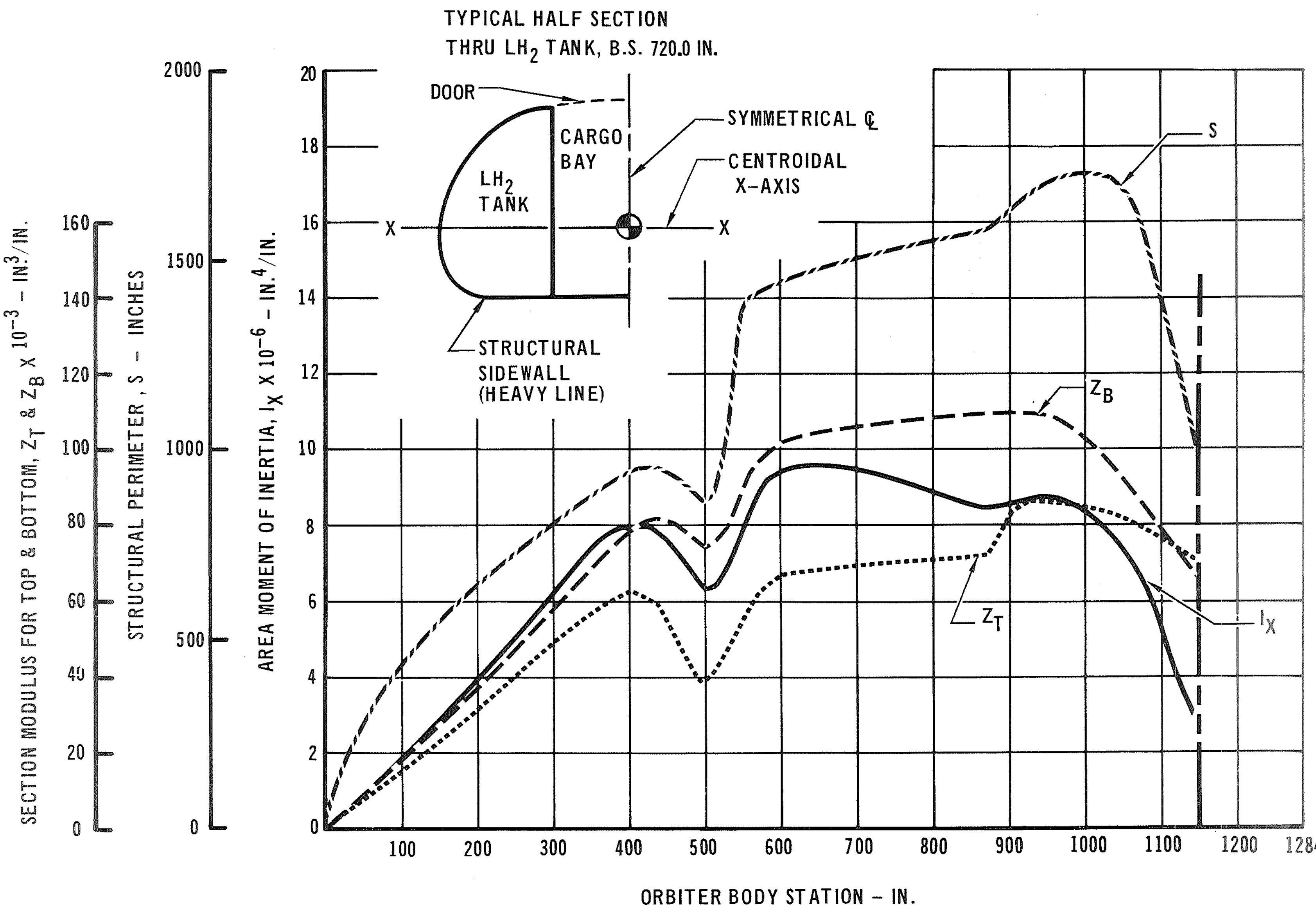


Figure 4-32

EQUIVALENT THICKNESSES OF ORBITER SIDEWALL STRUCTURE, UPPER SURFACE

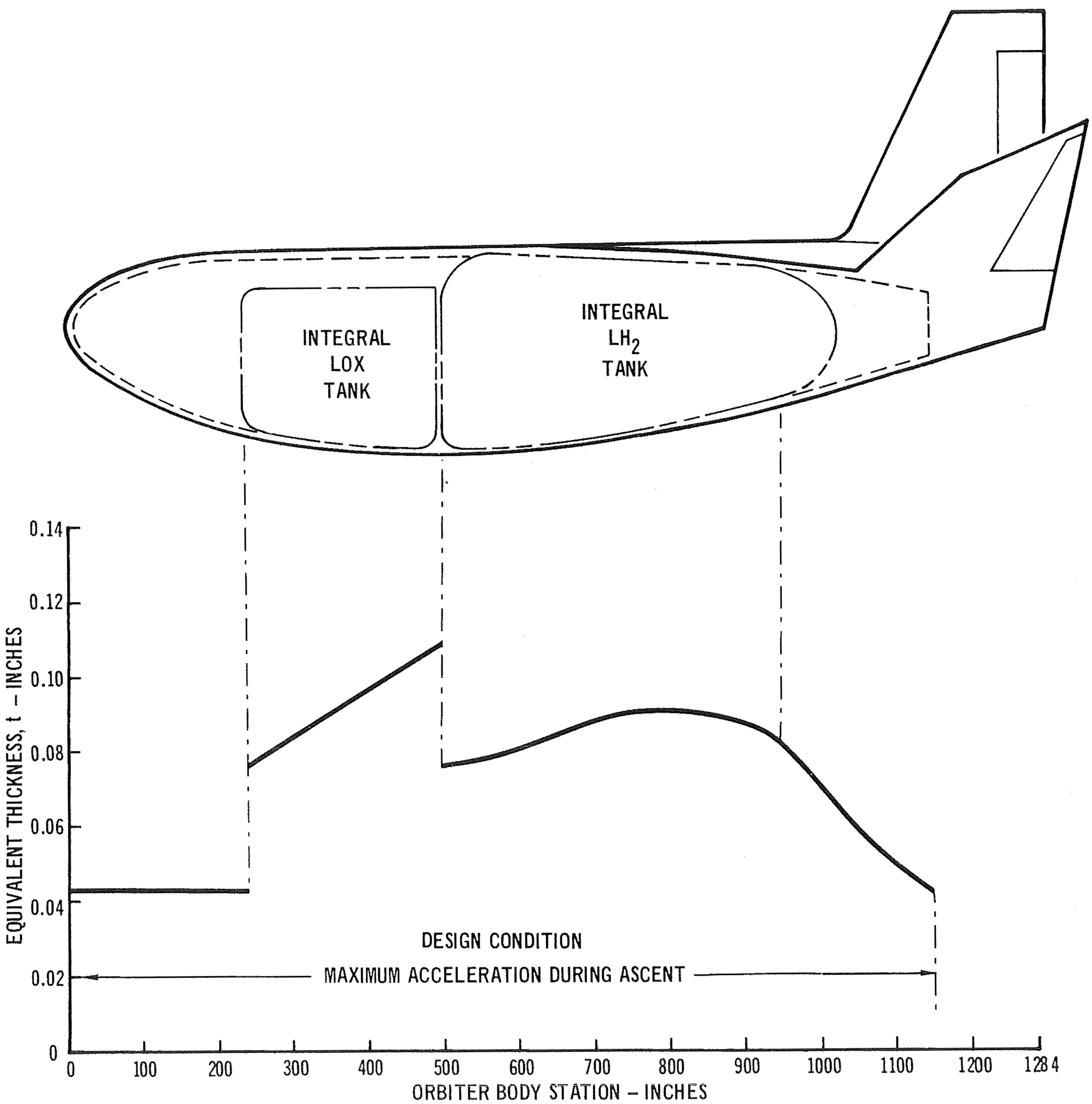


Figure 4–33

OPTIMUM FRAME SPACING FOR ORBITER
Once/Day Return Entry
(BOTTOM ℄ AT X/L = 0.25)

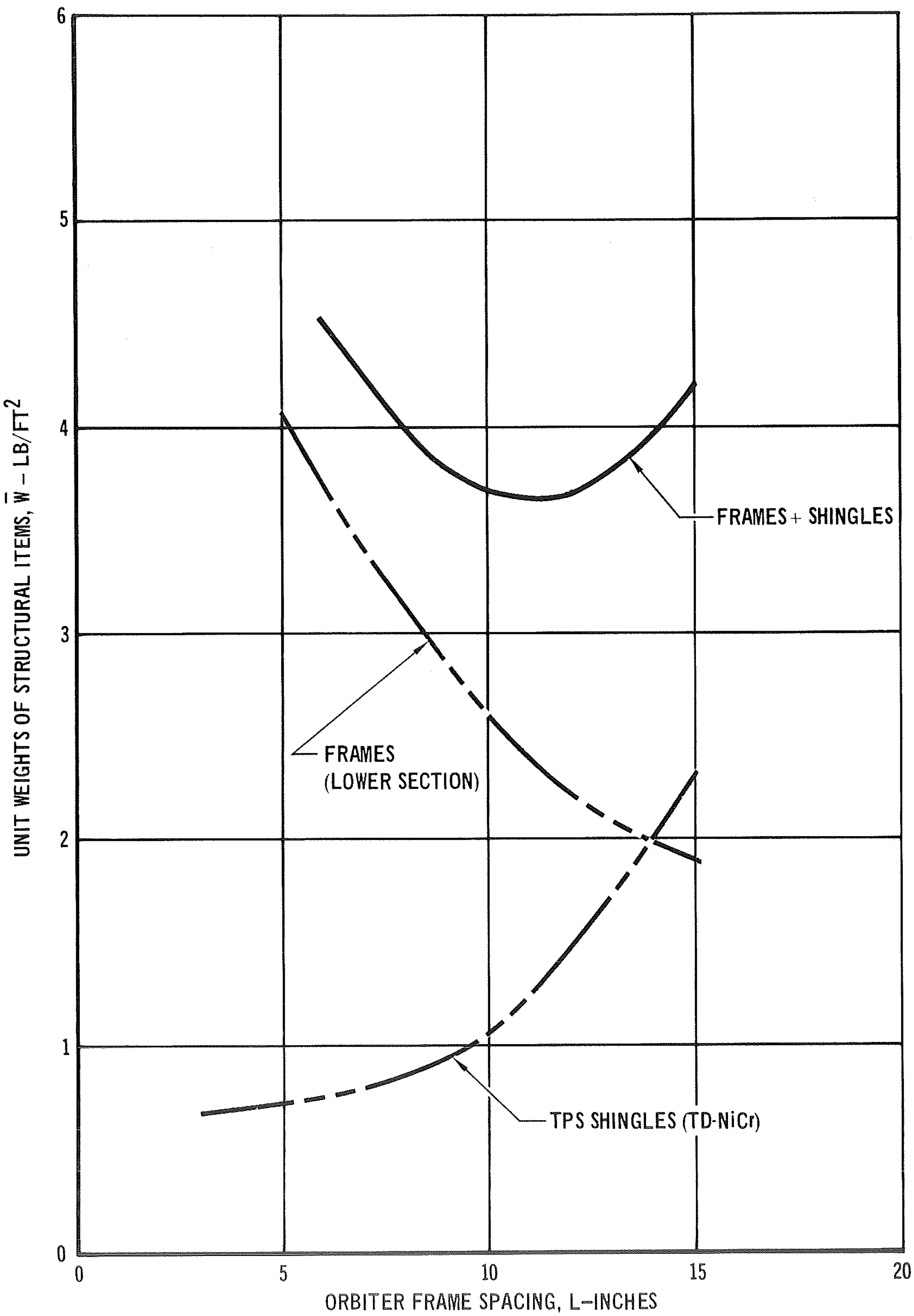


Figure 4-34

OPTIMUM FRAME SPACING FOR ORBITER

Once/Day Return Entry

(BOTTOM ℄ AT X/L = 0.50)

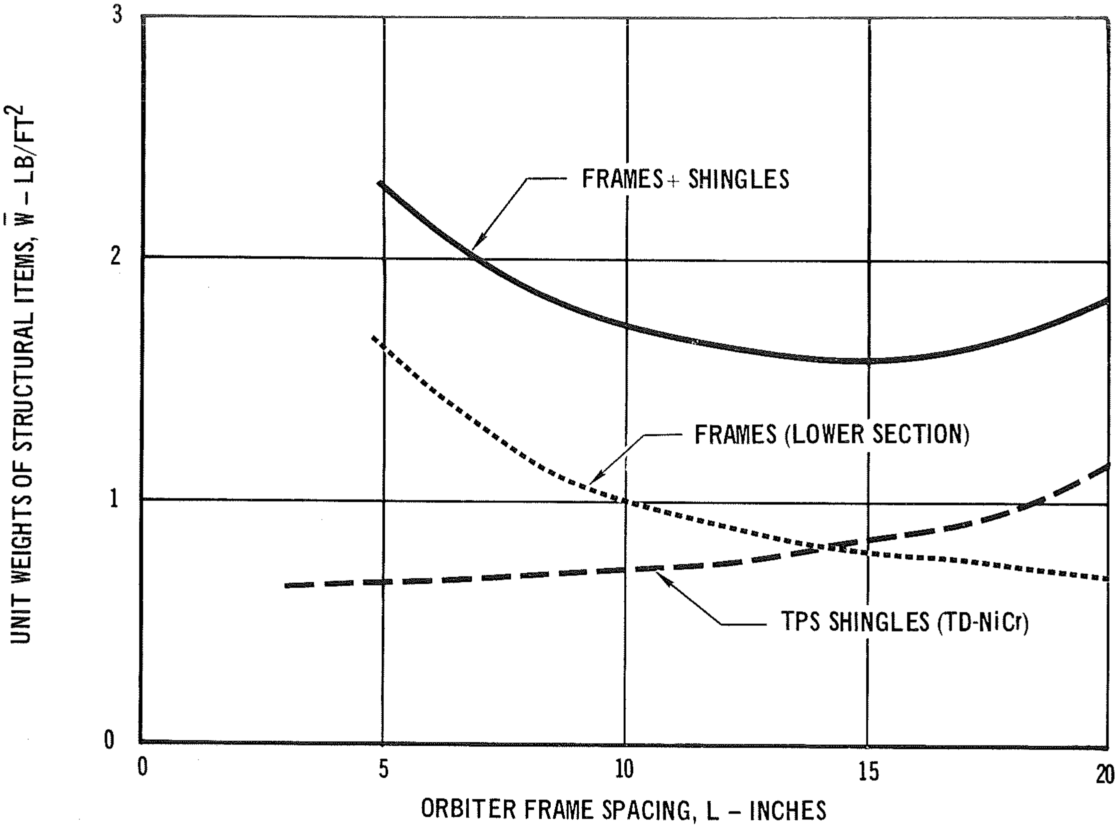


Figure 4-35

Strength analysis of the lower surface shingles are shown in Figures 4-36 and 4-37. Available strength data for the TD-NiCr alloy were reviewed during the study for determination of allowable stress levels through the expected temperature range of room temperature to 2200°F. Design allowables used are those proposed in Reference (2) and are shown in Figure 4-38. These allowables are based on no degradation of the base material mechanical properties after exposure to the given temperature. For the shingle geometry used, local crippling of individual elements does not occur. Single thickness beaded panels are used on the shadowed upper surface in regions which experience low heating rates. Analysis of a beaded panel is shown in Figure 4-39.

4.1.4 <u>Integral Cryogenic Tank/Structure Concepts</u> - A comparison of structural and cryogenic insulation arrangements was made to determine the relative merits of the various concepts and is shown in Figure 4-40. This study considered both internal and external insulation of the liquid hydrogen tank, plus primary structure external to, internal to, and on both sides of the cryo tank wall. This comparison was made for the baseline condition where the propellant tank wall also provides the vehicle body primary load carrying skin. The construction concept of the propellant tank wall is skin-stringer and this is held constant; other basic concepts such as single-faced corrugations, double faced corrugations and honeycomb would have similar structural arrangements. The outer moldline structure and thermal protection system was also assumed constant for each concept to avoid introducing another variable.

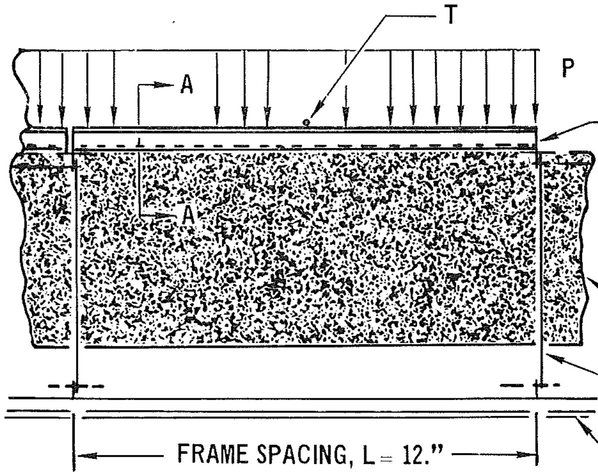
The structural arrangement with both rings and stringers outside the tank wall provide the maximum capability for support of wing, fins, landing gear and other external structural requirements. The internal rings and stringers however, provide the maximum ability for attaching internal structural webs, baffles, and required hardware.

Internal cryogenic insulation provides generally the minimum heat leak paths into the cryogenic fluid. This arrangement, with the rings and stringers external, presents the smoothest insulation surface, with the insulation discontinuities increasing as more structure is located inside the tank. A cryogenic foam insulation on the inside of the tank permits a higher tank wall temperature. The external cryogenic insulation requires a tank skin temperature of -423°F. The insulation which might be of a fibrous type then requires a mechanical attachment

STRENGTH ANALYSIS – TD NICR SHINGLE AT X/L = .25, BOTTOM ℄ (ORBITER)

CRITICAL DESIGN PRESSURE LOAD AND TEMPERATURE
OCCURING 40. SECONDS BEFORE MAXIMUM DYNAMIC PRESSURE
DURING ONCE/DAY RETURN ENTRY ARE (REFERENCE TABLE 4-5)

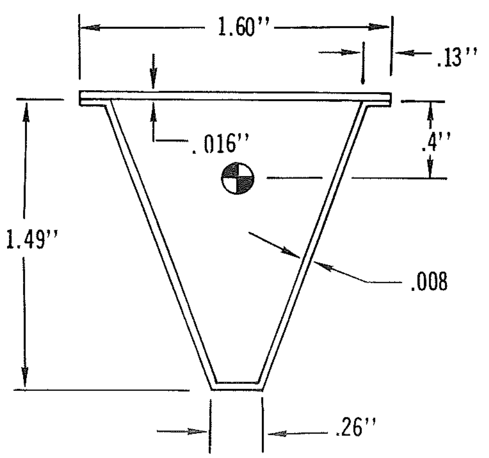
ULTIMATE PRESSURE .... p = 2.75 PSI, COLLAPSE
PREDICTED TEMPERATURE .... T = 1940.°F



MATERIAL: TD● NiCr (Ni – 20 Cr – 2ThO_2) @ 1940.°F
F_{all} = 6,000. psi (REFERENCE FIGURE 4-38)
DENSITY = .306 pci

DESIGN BENDING MOMENT PER INCH: $M_0 = \dfrac{pL^2}{8} = 1/8\,[(2.75)(12)^2]$

M_0 = 49.5 IN.-LBS./IN.



VIEW A–A

SECTION PROPERTIES:
SECTION AREA, A = .0552 IN^2
MOMENT OF INERTIA, I = .0145 $IN.^4$
MINIMUM SECTION MODULUS, Z_{MIN} = .0133 IN
CORRUGATION PITCH, P = 1.6 IN.

ALLOWABLE BENDING MOMENT PER INCH: $M_{ALL} = \dfrac{F_{ALL}\,Z_{MIN}}{P} = \dfrac{6000\,(.0133)}{1.6}$

M_{ALL} = 50. IN-LBS/IN.

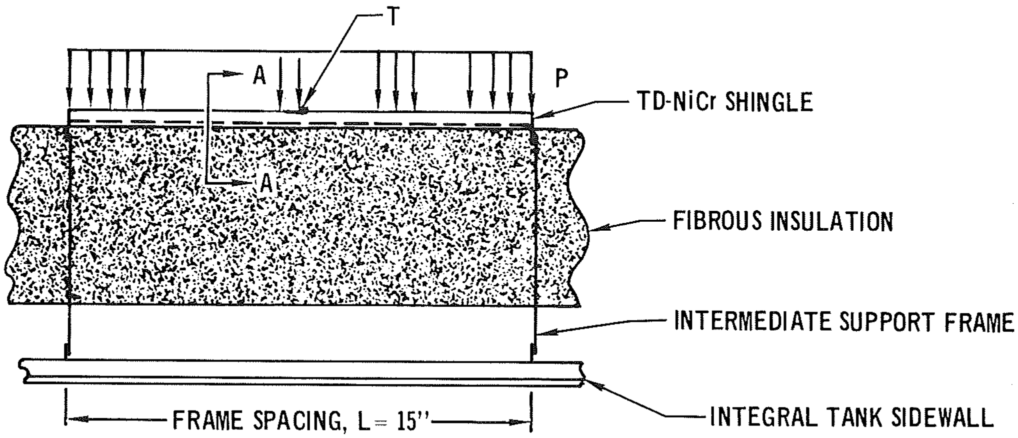
MARGIN OF SAFETY: $M.S. = \dfrac{M_{ALL}}{M_0} - 1. = \dfrac{50.}{49.5} - 1. = +.01$

Figure 4–36

STRENGTH ANALYSIS – TD-NiCr SHINGLE AT X/L = 0.50, BOTTOM ℄
Orbiter

CRITICAL DESIGN PRESSURE LOAD AND TEMPERATURE OCCURRING 40 SECONDS BEFORE MAXIMUM DYNAMIC PRESSURE DURING ONCE/DAY RETURN ENTRY ARE (REFERENCE TABLE 4-5)

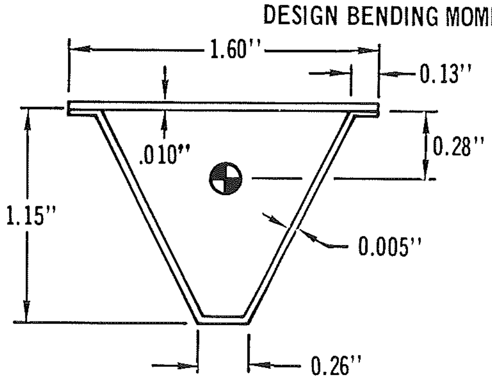
ULTIMATE PRESSURE P = 1.5 PSI, COLLAPSE
PREDICTED TEMPERATURE T = 1560°F



MATERIAL: TD–NiCr (Ni–20Cr– 2 ThO_2) AT 1560°F
F_{ALL} = 11,500 PSI (REFERENCE FIGURE 4–38)
DENSITY = 0.306 PSI

DESIGN BENDING MOMENT PER INCH: $\bar{M}_0 = \frac{1}{8} PL^2 = \frac{1}{8}(1.5)(15)^2$

$\bar{M}_0$ = 42 IN.-LB/IN.



VIEW A–A

SECTION PROPERTIES:
SECTION AREA, A = 0.0313 IN.^{2}
MOMENT OF INERTIA, I = 0.005 IN.^{4}
MINIMUM SECTION MODULUS, Z_{MIN} = 0.0058 IN.^{3}
CORRUGATION PITCH, P = 1.6 IN.

ALLOWABLE BENDING MOMENT PER INCH: $\bar{M}_{ALL} = \frac{F_{ALL} Z_{MIN}}{P} = \frac{11,500\ (0.0058)}{1.6}$

$\bar{M}_{ALL}$ = 42 IN.-LB/IN.

MARGIN OF SAFETY: $M.S. = \frac{\bar{M}_{ALL}}{M_0} - 1 = \frac{42}{42} - 1 = 0$

Figure 4–37

TD-NICKEL CHROME – DESIGN ALLOWABLE TENSILE STRENGTH VS TEMPERATURE

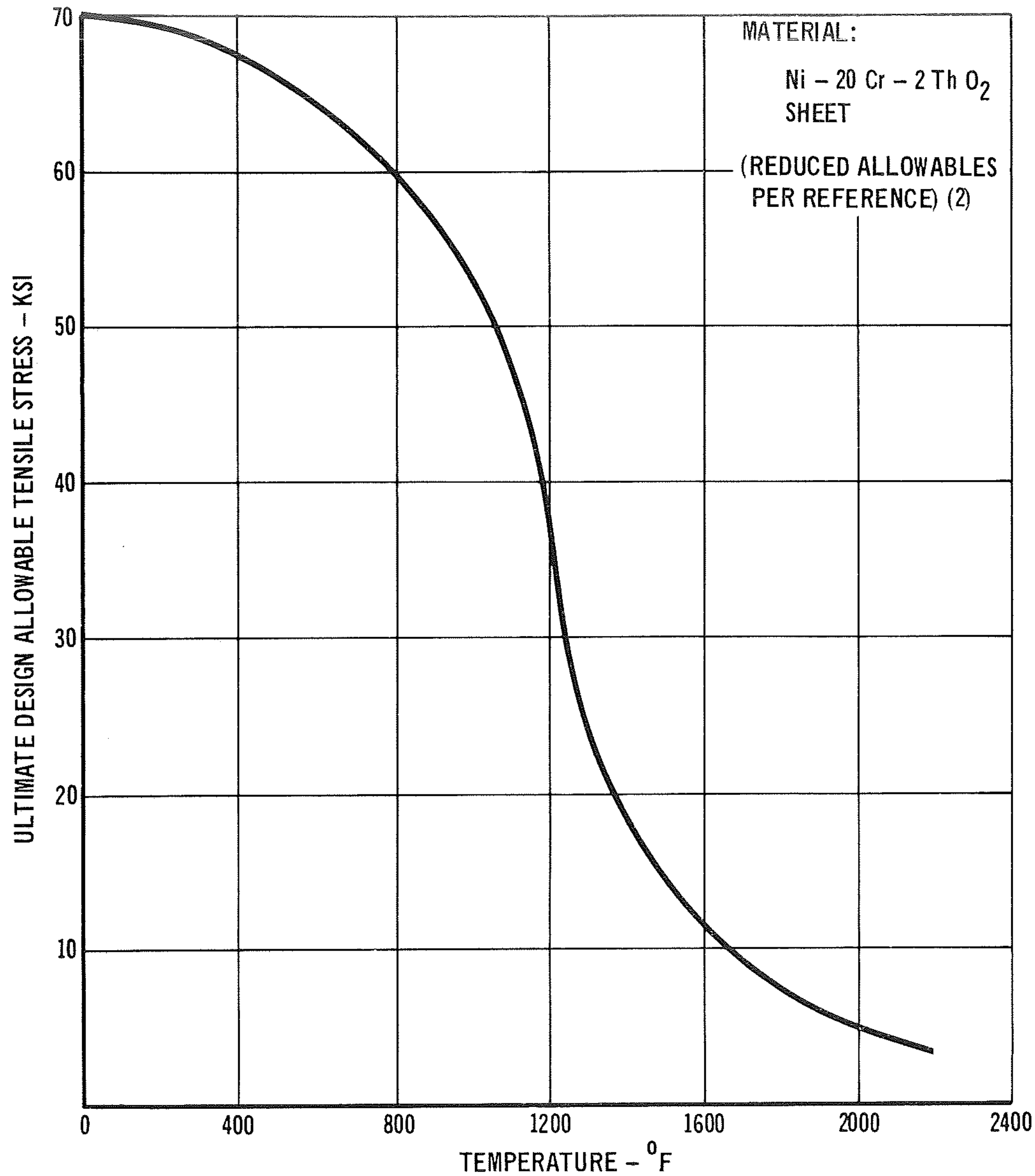
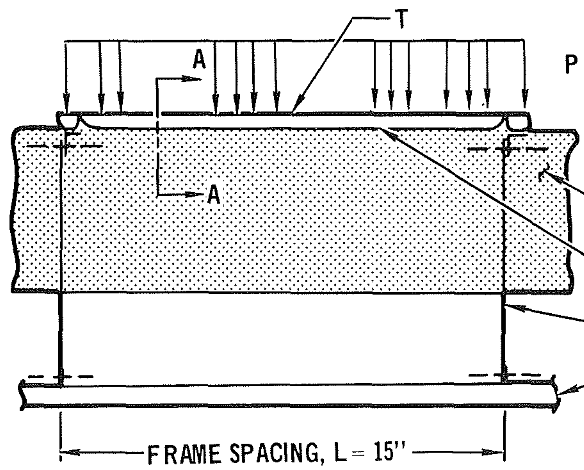


Figure 4–38

STRENGTH ANALYSIS – RENE' 41 SHINGLE ON UPPER SURFACE
Orbiter

PRESSURE LOAD AND TEMPERATURE ON TYPICAL, TOP SURFACE SHINGLE FROM TABLE 4-5 FOR ASCENT AT MAXIMUM DYNAMIC PRESSURE ARE:

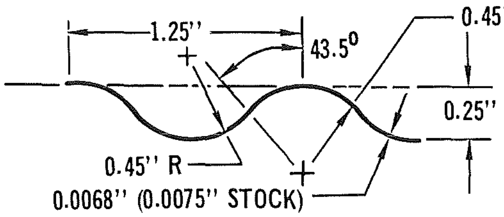
ULTIMATE PRESSURE – – – – – P = 2.2 PSI
CALCULATED TEMPERATURE – T = 100°F



MATERIAL: RENE' 41 AT 100°F
F_{TU} = 169,000 PSI
F_{CY} = 129,500 PSI
E_C = 31.3 $(10)^6$ PSI
(REFERENCE 3)

DESIGN BENDING MOMENT PER INCH: $\bar{M}_0 = \frac{1}{8} PL^2 = \frac{1}{8}(2.2)(15)^2$

$\bar{M}_0$ = 62 IN.-LB/IN.



VIEW A–A

SECTION PROPERTIES:
BEAD PITCH, P = 1.25 IN.
SECTION MODULUS, $Z = 7.65(10)^{-4}$ IN.^{3}

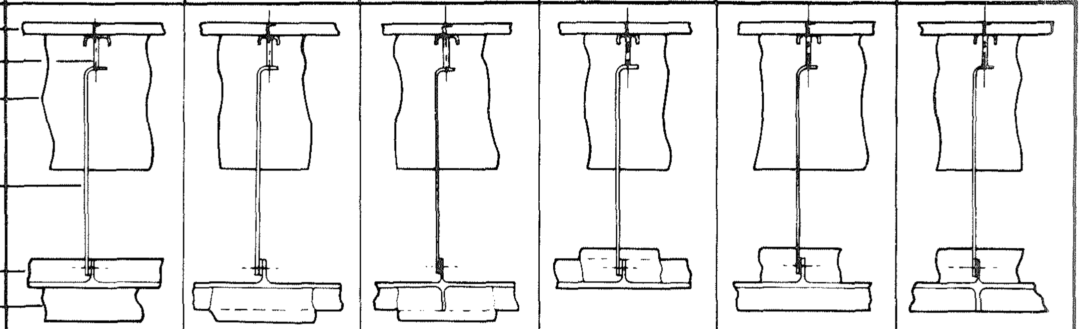
ALLOWABLE BENDING MOMENT PER INCH: $\bar{M}_{ALL} = \frac{F_{ALL} Z}{P}$; $F_{ALL} = \frac{0.3\ Et}{R} = \frac{0.3(31.3)(10)^6(.0068)}{0.45}$

$\bar{M}_{ALL} = \frac{1.42(10)^5(7.65)(10)^{-4}}{1.6}$ = 68 IN.-LB/IN. F_{ALL} = 142,000 PSI

MARGIN OF SAFETY: $M.S. = \frac{\bar{M}_{ALL}}{\bar{M}_0} \doteq 1 = \frac{68}{62} - 1 = +0.10$

Figure 4-39

INTEGRAL CRYOGENIC TANK/STRUCTURE CONCEPTS



CRYO INSULATION	INTERNAL			EXTERNAL		
RINGS/STRINGERS	EXT	EXT/INT	INT	EXT	EXT/INT	INT
PROVISION FOR EXT STRUCTURE & TPS (1 = LEAST PENALTY)	1	2	3	1	2	3
PROVISION FOR INT WEBS & BAFFLES (1 = LEAST PENALTY)	3	2	1	3	2	1
STRUCTURE HEAT LEAKS (1 LEAST PENALTY)	1	2	3	6	5	4
MECHANICAL CRYO INSUL SUPPORT (0 = NOT REQ, 1 = REQ)	0	0	0	1	1	1
COLD STRUCTURE (0 = NO, 1 = YES)	0	0	0	1	1	1
MOISTURE BETWEEN CRYO INSULATION & STRUCTURE (1 MIN PROBLEM)	1	1	1	2	2	2
STRUCTURE BUILD-UP COMPLEXITY (1 = LEAST PENALTY)	2	1	3	2	1	3
INSULATION INSTALLATION COMPLEXITY (CRYO) (1 LEAST PENALTY)	1	2	3	6	5	4
INSPECT AND REPAIR DIFFICULTY (1 = LEAST PROBLEM)	1	2	3	6	5	4

ILRVS-260F

Figure 4-40

for support. The cryogenic insulation also requires a vapor barrier to prevent frost and moisture buildup between the structure and insulation. A nitrogen purge may be used with the internal insulation, and a helium purge will probably be required with the external insulation since the area between cold tank skin and moldline panels is not pressure tight. The helium purge system is more complex and expensive than the nitrogen system.

The structural fabrication becomes most complex for the concepts where all the primary support structure is on one side of the tank wall. The major effect here is that additional structure must also be provided, either for external moldline structure support, or for internal webs and baffles. Fabrication of the cryogenic insulation installation is more difficult for the mechanically supported external insulation. It becomes simplest for the internal insulation with a relatively smooth surface present where all the primary structure is external. Inspection and repair of insulation and structure is most complex with the external insulation because the thermal protection panels must be removed to provide accessibility. The internal insulation may be maintained from within the tank by opening an access hatch. Structure external to the tank, with internal insulation, must still be inspected by removing moldline panels. The frequency of this service however, is less than that required for the insulation and the concept employing structure external and insulation internal is rated as the most optimum using this criteria.

An overall summation of relative merits of the various concepts indicates the first two cases as being most effective in terms of reducing the various penalties discussed. The choice between all external structure and divided structure, with internal insulation, then becomes a function of the amount of internal hardware required.

4.2 Reentry Heating/Thermal Protection System Analysis - This analysis represents a special emphasis area as a part of the overall study. Entry trajectories are defined in Volume II and the heating rates for 3 trajectories, together with temperature distributions are presented in this section. Thermal protection systems and material distributions are shown for the heating and temperature data of a nominal once-a-day return trajectory. Both metallic radiative and hardened compacted fiber outer surface thermal protection systems are shown, with support and attachment methods.

4.2.1 Trajectory Analysis - Trajectory simulations were performed to define the aerothermal environment for carrier reentry and orbiter reentry. Parametric studies were used to gain insight to principal variables for trajectory shaping. Subsequently, nominal mission parameters were defined for the final design configuration utilizing the flight commands derived from the parametric studies. Discussion of the results is contained in Volume II, Section 3.3. These sections also contain time histories of pertinent trajectory parameters.

4.2.2 Thermodynamics Analysis - This section presents the orbiter and carrier entry heating analysis and the thermal protection requirements during entry. The methods used in predicting orbiter heating are present in Volume II, Section 2.3.

Orbiter Entry Heating Analysis - Reentry heat pulses for the orbiter are shown in Figure 4-41 for the nominal once/day, minimum time (2600°F), and twice/day reentries. The trajectories were shaped such that the maximum surface temperature at 12-1/2 percent body length would be 2600°F for the minimum time and 2200°F for the once/day and twice/day reentries. The twice/day reentry incurs the largest stagnation point total heat load (46,200 BTU/ft^2) and the minimum time (2600°F) reentry incurs the smallest (13,200 BTU/ft^2). The minimum time (2600°F) reentry has the highest stagnation point heating rate (59 BTU/ft^{2}sec) and the twice/day reentry the lowest (41 BTU/ft^{2}sec).

The heating rates are calculated values based on the Fay and Riddell stagnation point theory for a nose radius of 62 inches and do not include an uncertainty factor.

Reentry temperatures on the orbiter lower surface centerline at 25% of vehicle length are shown in Figure 4-42 for the three reentries. These temperatures are radiation equilibrium values based on laminar flow and a surface

ORBITER REENTRY HEAT PROFILES

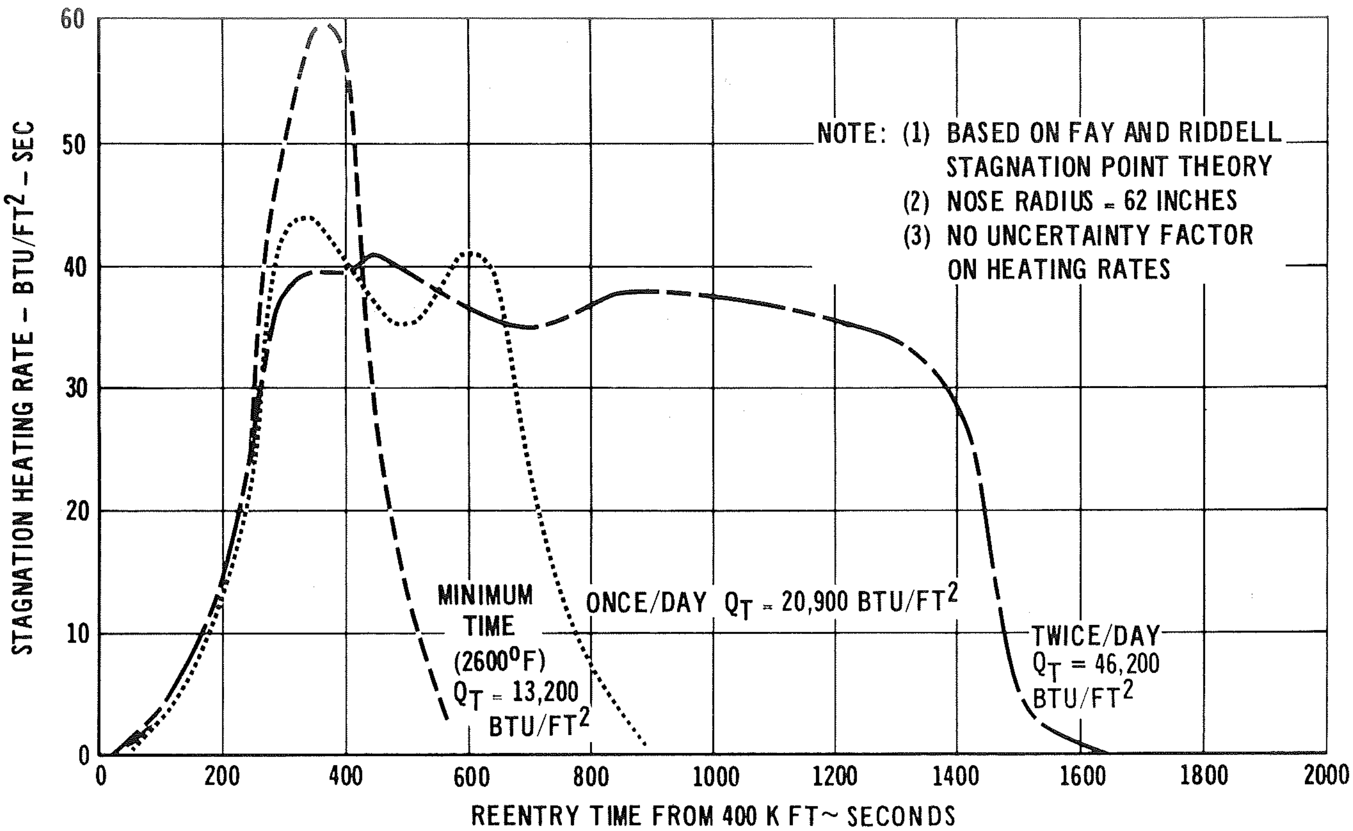


Figure 4-41

ORBITER REENTRY TEMPERATURES
Lower Surface Centerline
25% of Vehicle Length

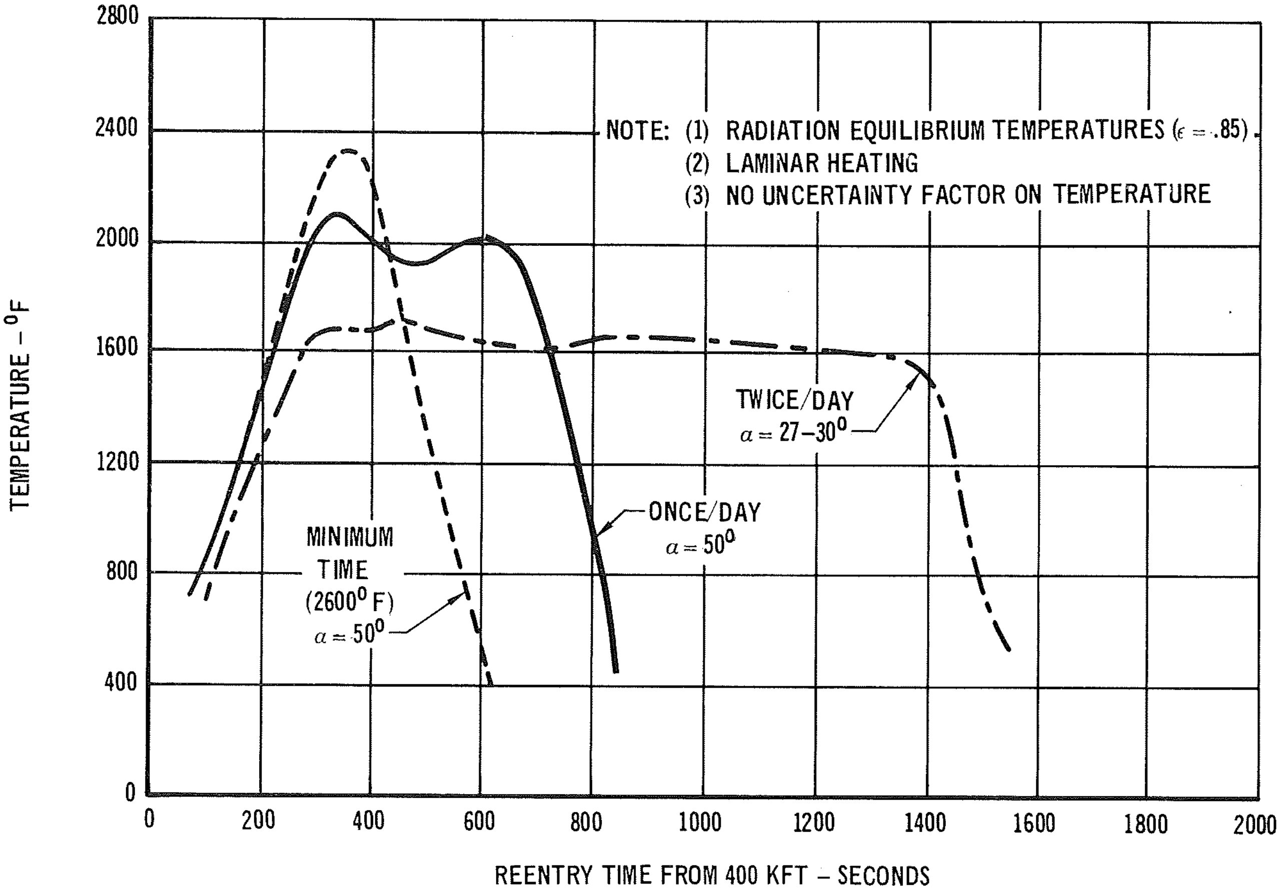


Figure 4-42

emittance equal to 0.85. Peak temperatures during reentry are 2325°F for the minimum time (2600°F), 2100°F for the nominal once/day, and 1725°F for the twice/day.

Maximum laminar radiation equilibrium orbiter surface temperature distributions during reentry are shown in Figures 4-43 through 4-45 for the three reentries at four orbiter body stations.

Figure 4-43 shows that maximum surface temperatures for the nominal once/day reentry range from 680°F on the upper surface to 2200°F on the lower surface. Maximum surface temperatures for the minimum time (2600°F) reentry, Figure 4-44, range from 780°F on the upper surface to 2600°F on the lower surface. Maximum surface temperatures for the twice/day reentry, Figure 4-45, range from 700°F on the upper surface to 2200°F on the lower surface.

An orbiter reentry heating comparison summary is presented in Table 4-6. The table shows the maximum stagnation point heating rate, the total stagnation point heat load and the range of maximum surface temperatures for the three reentries. It is apparent that the maximum temperature of 2200°F can be maintained by proper trajectory shaping during reentry at both high and low angles of attack. Reentry can be accomplished, however, in minimum time with the penalty of higher temperatures.

ORBITER MAXIMUM TEMPERATURES
NOMINAL ONCE/DAY REENTRY

NOTE:
(1) RADIATION EQUILIBRIUM TEMPERATURES (ε .85)
(2) LAMINAR HEATING
(3) NO UNCERTAINTY FACTOR
(4) α - 50^0

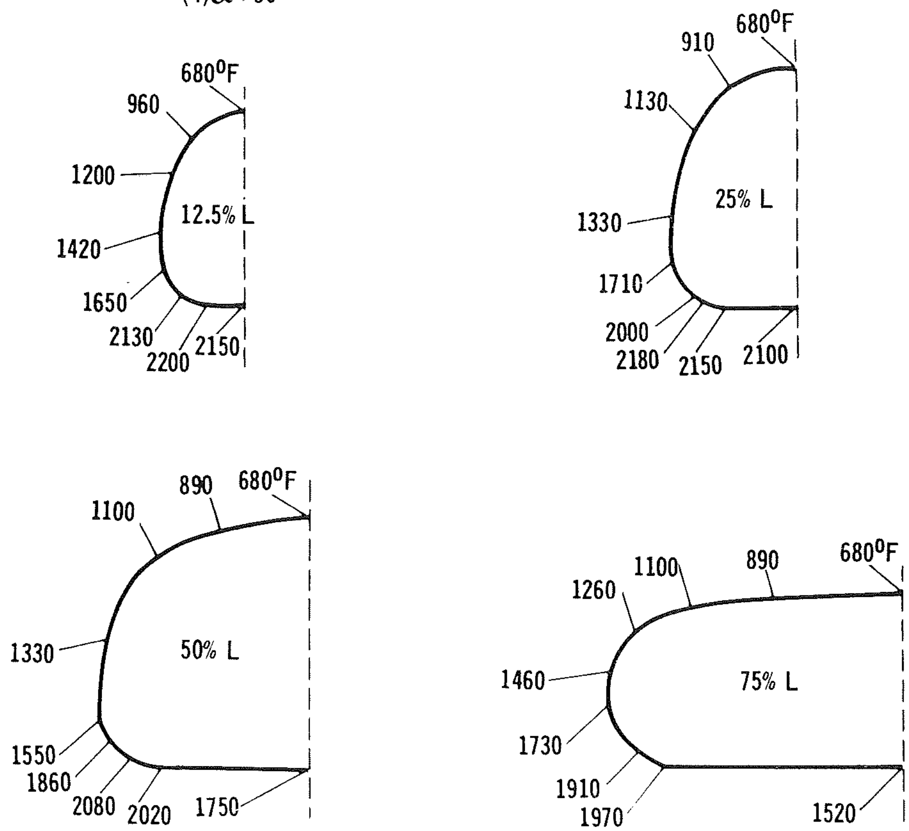
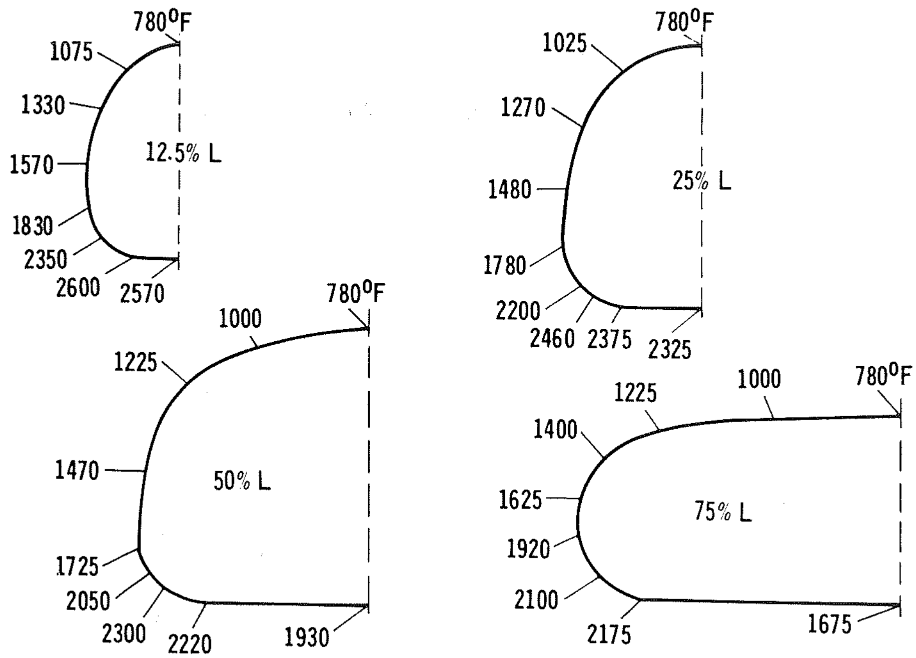


Figure 4–43

ORBITER MAXIMUM TEMPERATURES MINIMUM TIME (2600°F) REENTRY



NOTE:

(1) RADIATION EQUILIBRIUM TEMPERATURES(ε 0.85)

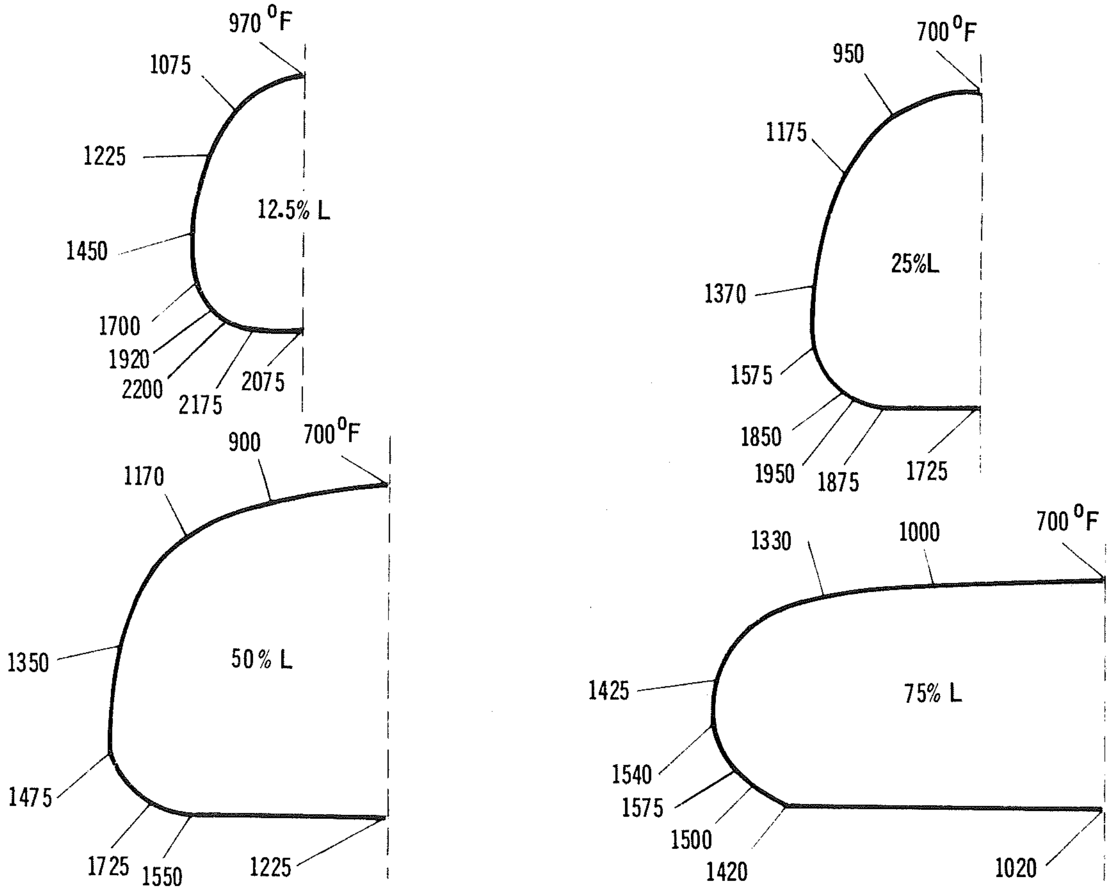
(2) LAMINAR HEATING

(3) NO UNCERTAINTY FACTOR

(4)α 50°

Figure 4-44

ORBITER MAXIMUM TEMPERATURES
NOMINAL TWICE/DAY REENTRY



NOTE:
(1) RADIATION EQUILIBRIUM TEMPERATURES ($\epsilon = 0.85$)
(2) LAMINAR HEATING
(3) NO UNCERTAINTY FACTOR
(4) $\alpha = 27-30^0$

Figure 4–45

Table 4-6

ORBITER REENTRY HEATING COMPARISON

REENTRY	$(q_0)_{MAX}$ (BTU/FT^2 SEC)	(Q_T) STAG (BTU/FT^2)	HEATING TIME (SECONDS)	MAXIMUM TEMPERATURE RANGE (^{0}F)
TWICE/DAY – $\alpha = 27 - 30^0$	41	46,200	1,550	700 – 2200
ONCE/DAY – $\alpha = 50^0$	44	20,900	830	680 – 2200
MINIMUM TIME (2600^{0}F) – $\alpha = 50^0$	59	13,200	540	780 – 2600

NOTE: (1) $(q_0)_{MAX}$ BASED ON RADIUS = 62 INCHES

(2) (Q_T) STAG AND HEATING TIME ARE FOR $q_0 > 1.0$ BTU/FT^2 SEC

(3) TEMPERATURES ARE PREDICTED LAMINAR RADIATION EQUILIBRIUM VALUES ($\epsilon = .85$)

The criterion used for prediction of turbulent heating rates is the onset of transition at a local Reynolds number of 10^6 and fully developed turbulent flow at 2×10^6. Velocities at which fully developed turbulent flow occurs along the orbiter lower surface centerline are shown for the three reentries in Figure 4-46. With exception of the aft ten percent of the vehicle transition occurs at higher velocities for the twice/day reentry than for either of the other two. Complete transition to turbulent flow occurs at 550,770 and 1410 seconds after initiation of reentry for the minimum time (2600°F), once/day and twice/day reentries, respectively. The methods used to predict turbulent heating are discussed more fully in Volume II, Section 2.3.

Table 4-7 shows that the effect of turbulent heating on total heat (Q_t) ranges from 7 percent increase for the once/day reentry to 17 percent increase for the twice/day reentry. Maximum surface temperatures are increased from 1225°F to 1440°F for the twice/day reentry and are not affected for the other two reentries. A temperature history of the lower surface centerline at 50% length is shown for the nominal once/day reentry is Figure 4-47.

Orbiter Thermal Protection - Thermal protection systems (TPS) for the once/day and twice/day orbiter reentries have been sized for the use of metallic shingles. The minimum time (2600°F) reentry temperatures exceed the Td-NiCr limitations on some areas of the orbiter and a hardened compacted fibrous insulation (HCF) thermal protection system has been sized for this reentry. TPS requirements presented will limit the cryogenic tank wall temperature to a maximum of 200°F until landing. Ground cooling is required after landing.

Insulation will be required underneath the orbiter metallic shingles to reduce the heat transfer to the cryogenic tank wall in order that the temperature does not exceed 200°F. Insulation requirements are shown in Figures 4-48 and 4-49 for the nominal once/day and twice/day reentries. Average insulation unit weights are shown as a function of body station for the lower surface, leading edge, side and upper surface. The unit weights shown are for microquartz insulation having a 3.5 lb/ft^3 density. A sketch of the thermal model used is shown on the figures to illustrate the modes of heat transfer considered. Microquartz unit weight requirements for the once/day entry vary from .72 lb/ft^2 on the lower surface to .18 lb/ft^2 on the upper surface and from .123 lb/ft^2 to .42 lb/ft^2 for the twice/day entry. Restriction of the tank wall to a maximum temperature of 200°F without ground cooling would require insulation unit weights approximately 70% higher than those shown.

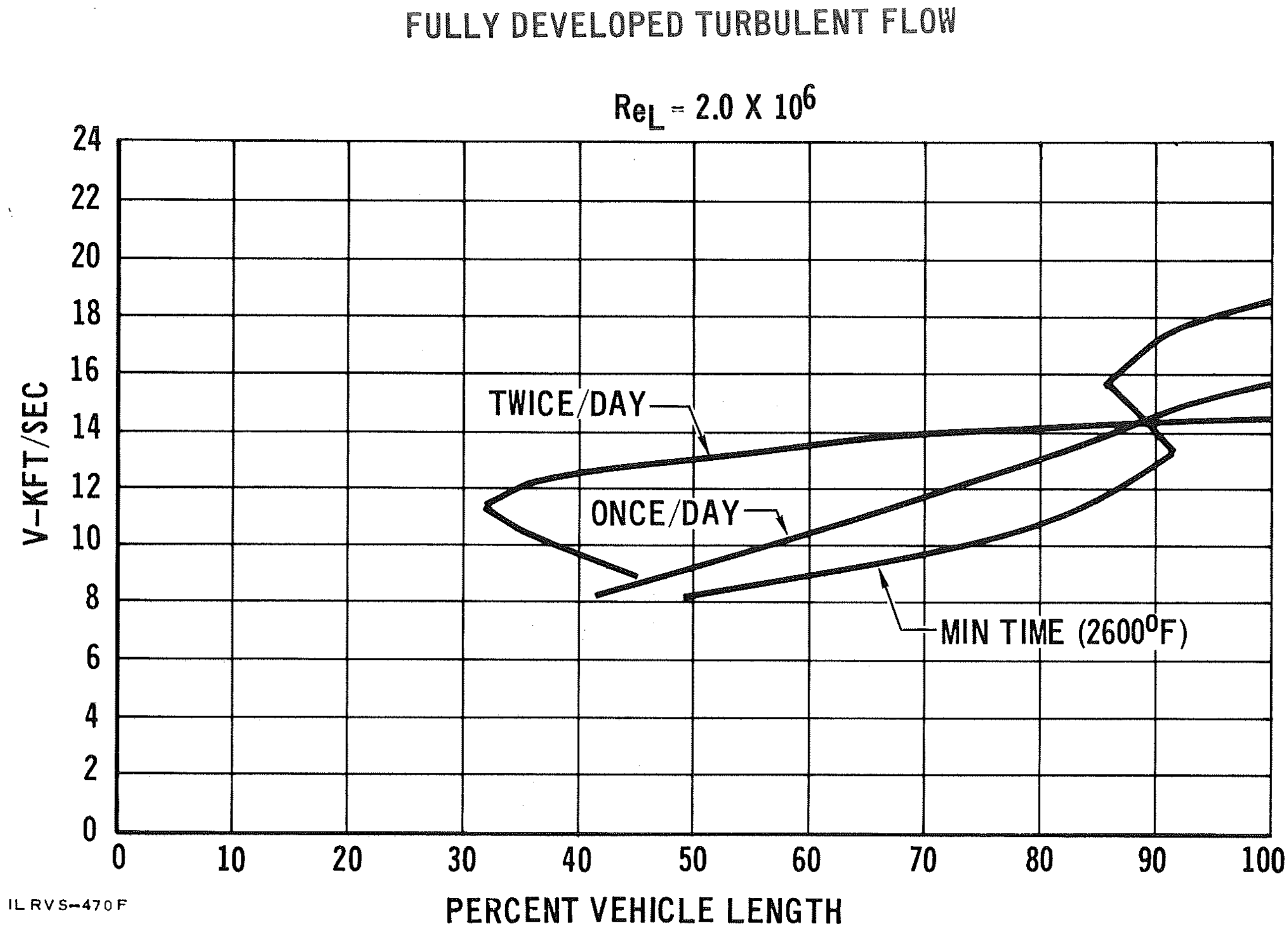

FULLY DEVELOPED TURBULENT FLOW
$Re_L = 2.0 \times 10^6$
24
22
20
18
16
14
12
10
8
6
4
2
0
V-KFT/SEC
TWICE/DAY
ONCE/DAY
MIN TIME (2600°F)
0
10
20
30
40
50
60
70
80
90
100
PERCENT VEHICLE LENGTH
ILRVS-470F

Figure 4-46

Table 4–7

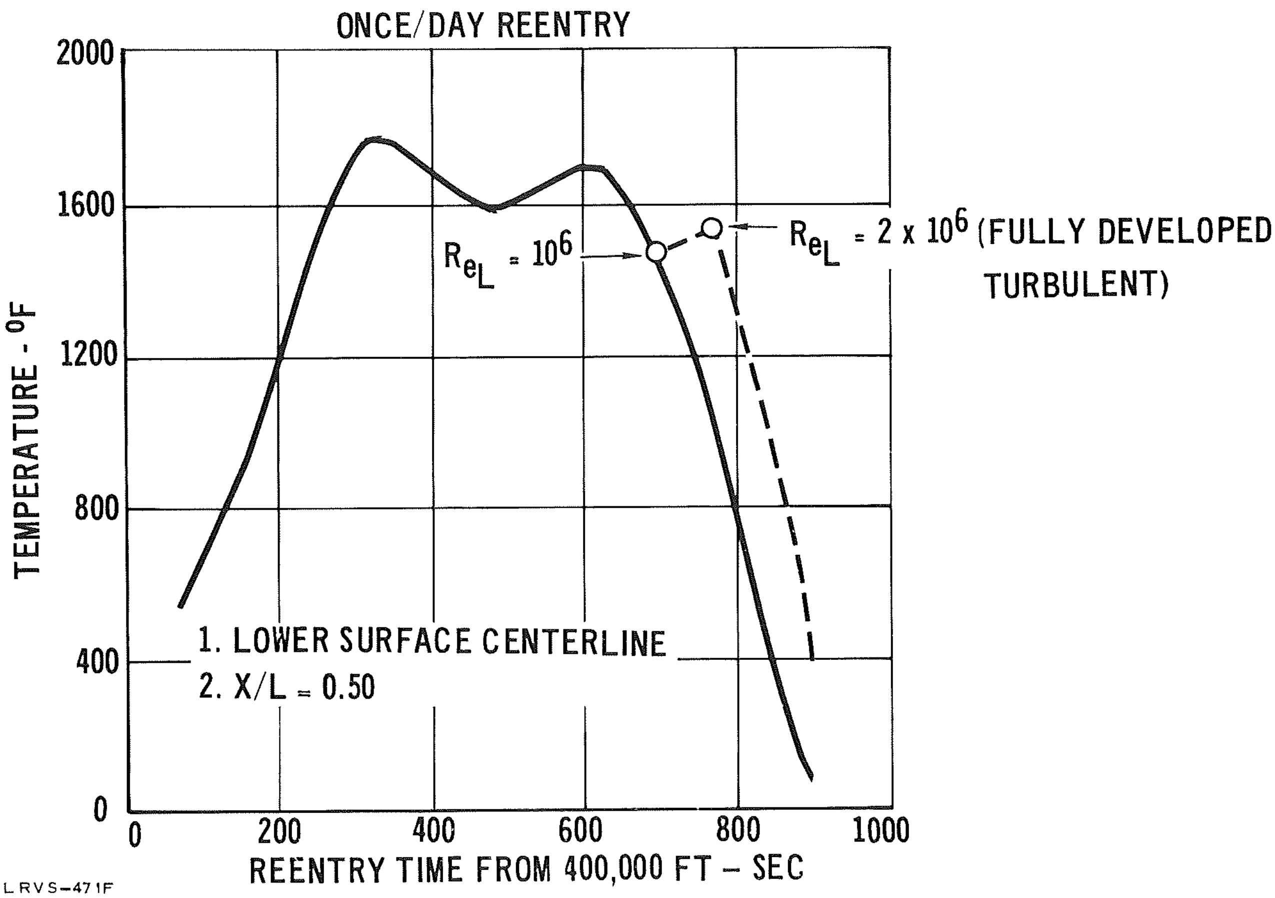
EFFECT OF TURBULENT HEATING ON ORBITER

Lower Surface Centerline, X/L = .50

REENTRY	T_{MAX}, LAMINAR	T_{MAX}, TURB $R_{E_L} = 2 \times 10^6$	Q_T, LAM (BTU/FT^2)	INCREASE IN Q_T DUE TO TURB HEATING (1)
ONCE/DAY	1750 °F	1540 °F	4280	7%
MIN TIME (2600 °F)	1930	1270	2710	16
TWICE/DAY	1225	1440	3700	17

NOTE: (1) TRANSITION ONSET AT $R_{E_L} = 10^6$, FULLY DEVELOPED TURBULENT AT $R_{E_L} = 2 \times 10^6$

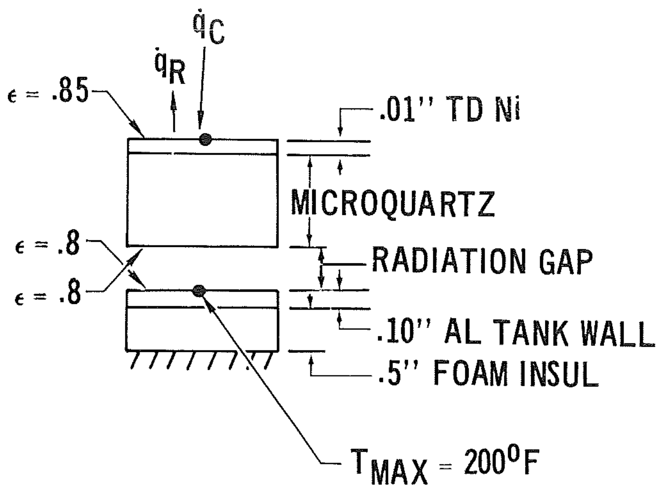
TRANSITION EFFECTS ON ORBITER TEMPERATURES



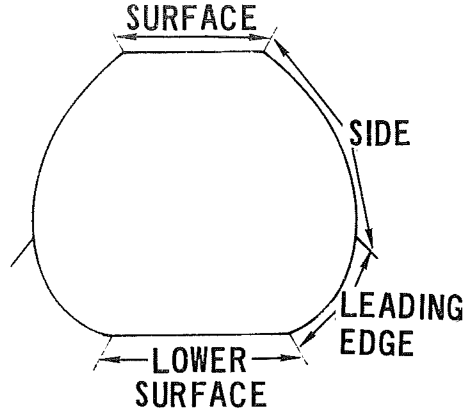
ILRVS-471F

Figure 4-47

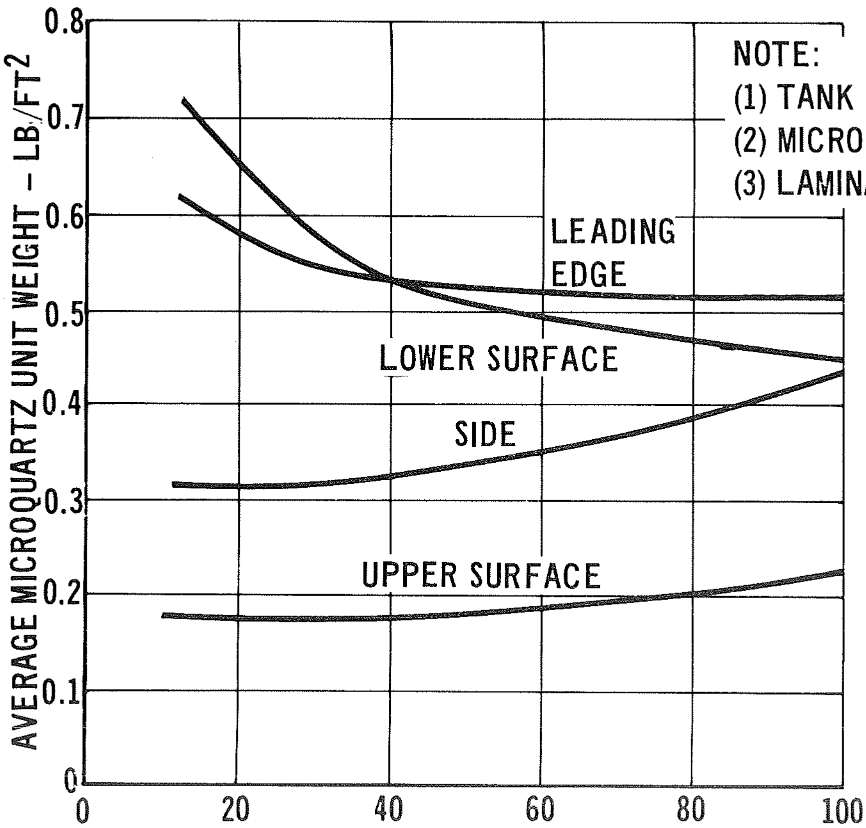
ORBITER MICROQUARTZ INSULATION REQUIREMENTS
NOMINAL ONCE/DAY



1–D MODEL
(TYPICAL)



TYPICAL



NOTE:
(1) TANK WALL TEMP 200°F AT LANDING
(2) MICROQUARTZ DENSITY = 3.5 LB/FT^3
(3) LAMINAR HEATING

Figure 4–48

ORBITER MICROQUARTZ INSULATION REQUIREMENTS TWICE/DAY REENTRY

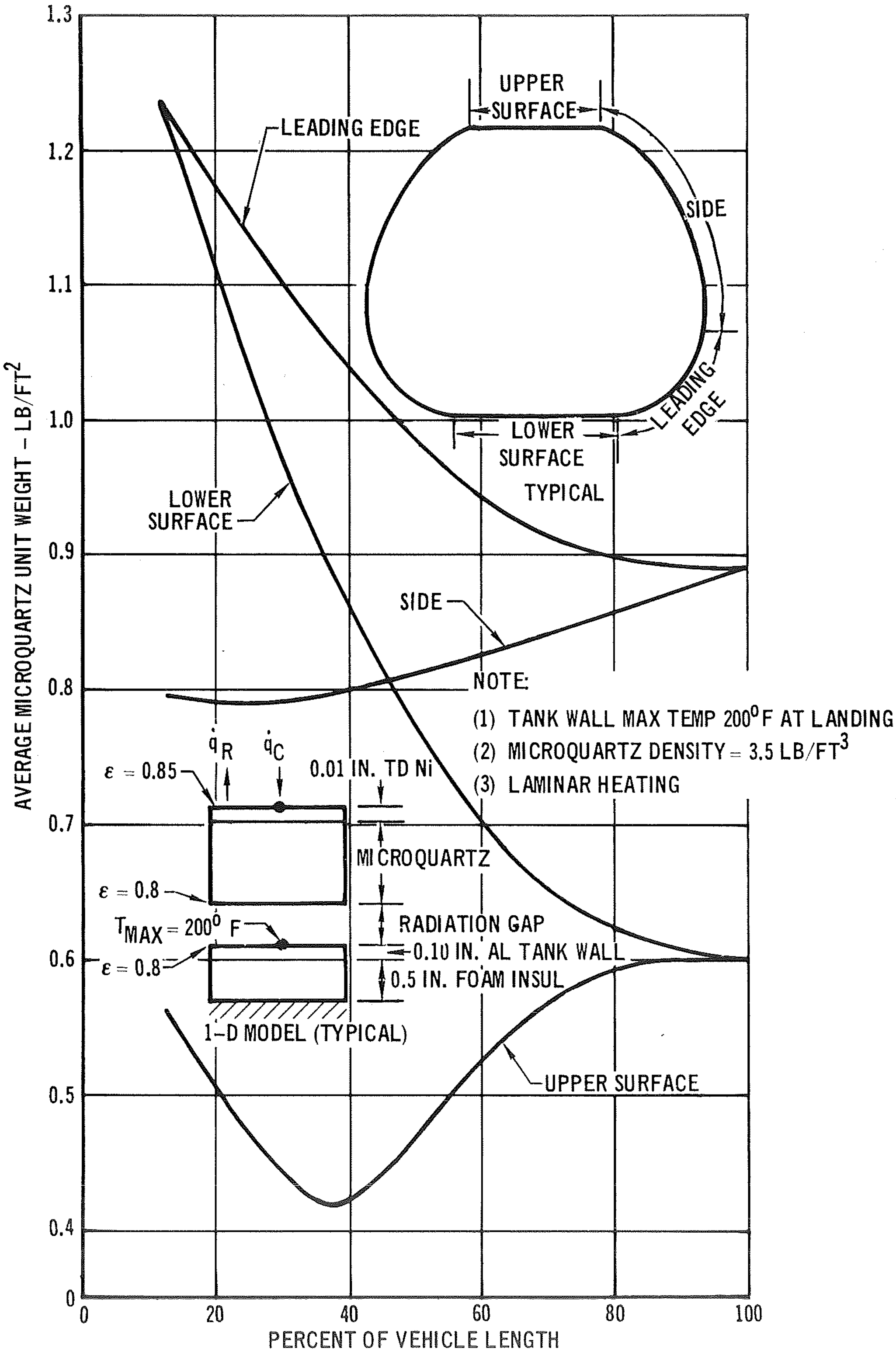


Figure 4-49

HCF thermal protection requirements are highly sensitive to the thermal properties of the material, especially the thermal conductivity. Figure 4-50 shows the HCF thermal conductivity used for preliminary design compared to that more recently obtained from data. Figure 4-51 shows the variation of HCF unit weight requirements based on the two thermal conductivity curves. This figure shows that the unit weight requirements differ considerably, especially at the higher local heating rates. For example, at a maximum local laminar heating rate of 10 BTU/ft^2 sec (T_{EQ} = 1750°F) the preliminary requirements are 39% higher than the requirements obtained with the benefit of data. Further tests are necessary to better establish the material properties of HCF.

Average HCF requirements for the minimum time (2600°F), based on the preliminary conductivity values are presented in Figure 4-52. The unit weights are based on laminar heating, a 15 lb/ft^3 density HCF, and a maximum bondline temperature of 500°F. The thermal model used is shown on Figure 4-52 to illustrate the modes of heat transfer considered.

The HCF unit weight requirements vary from 2.6 lb/ft^2 on the lower surface to 0.7 lb/ft^2 on the upper surface. Based on the more recent data these would be reduced to 1.8 and .47 lb/ft^2.

The thermal protection requirements shown in Figures 4-48, 4-49 and 4-52 would be increased by only a small amount for fully developed turbulent flow at $Re_L = 2 \times 10^6$ since thermal protection requirements are more strongly influenced by heating time than by total heat.

HCF THERMAL CONDUCTIVITY

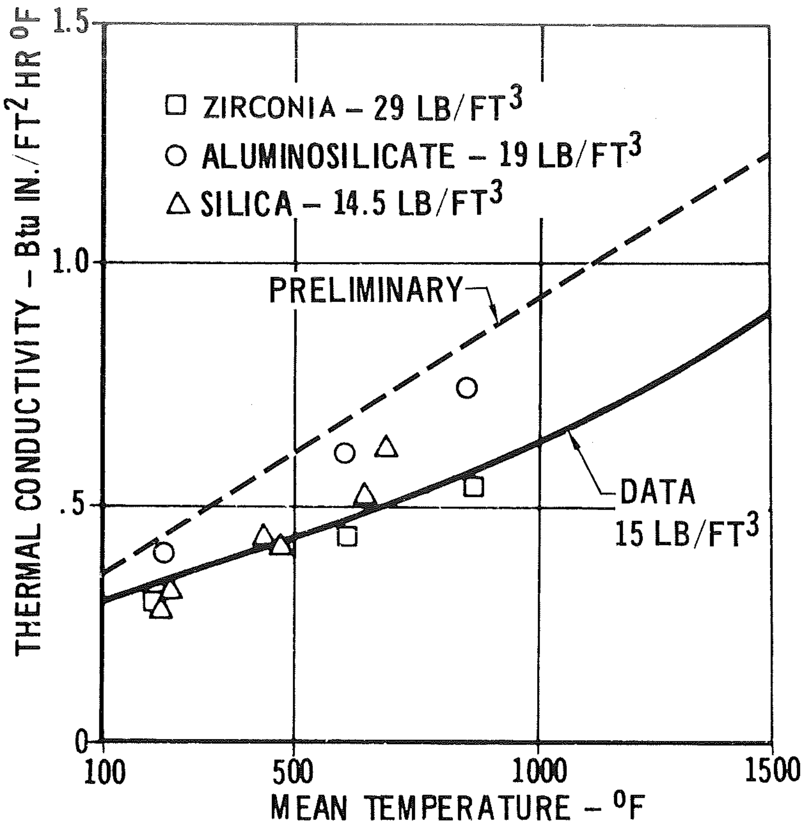


Figure 4-50

EFFECT OF THERMAL CONDUCTIVITY ON HCF UNIT WEIGHTS

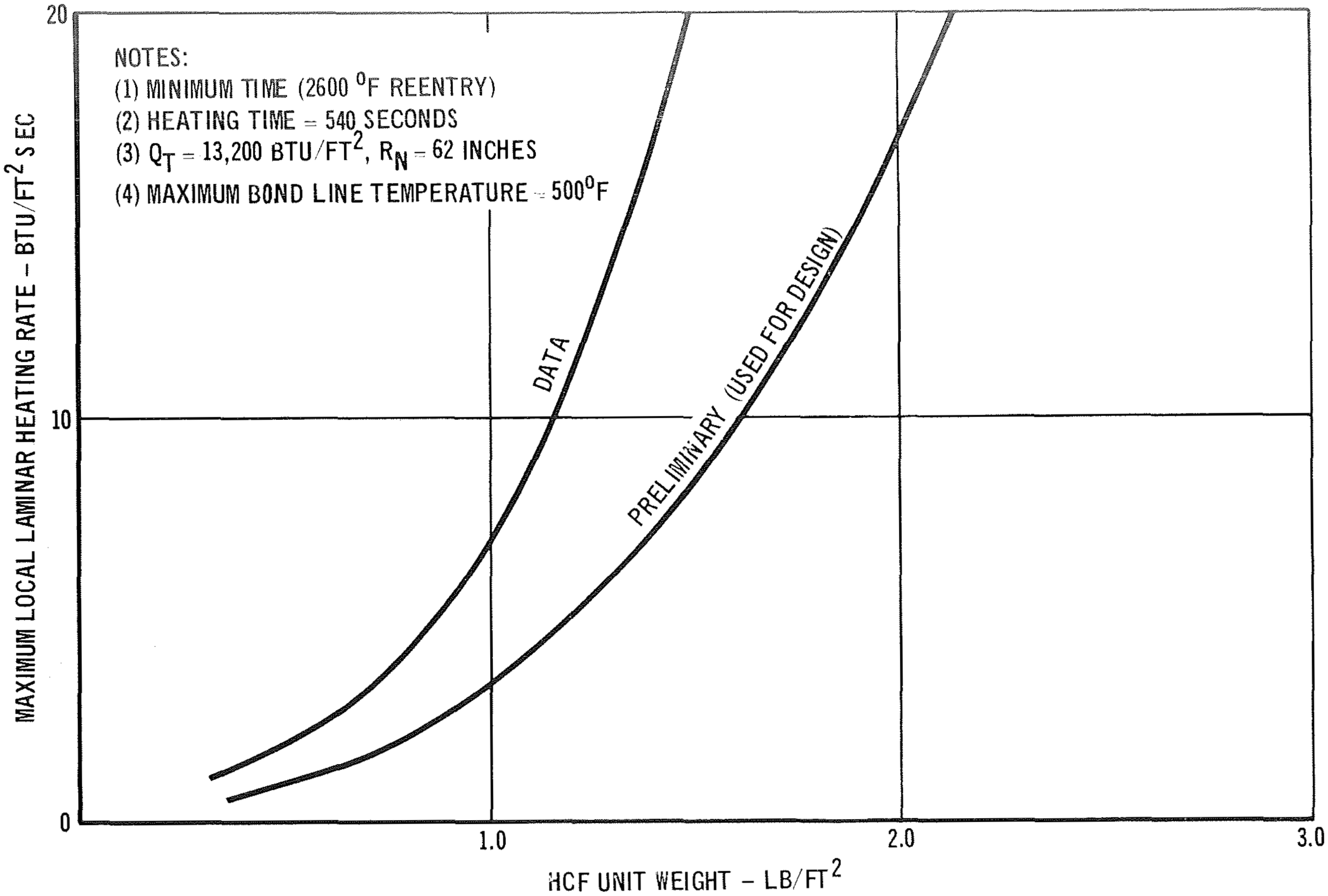
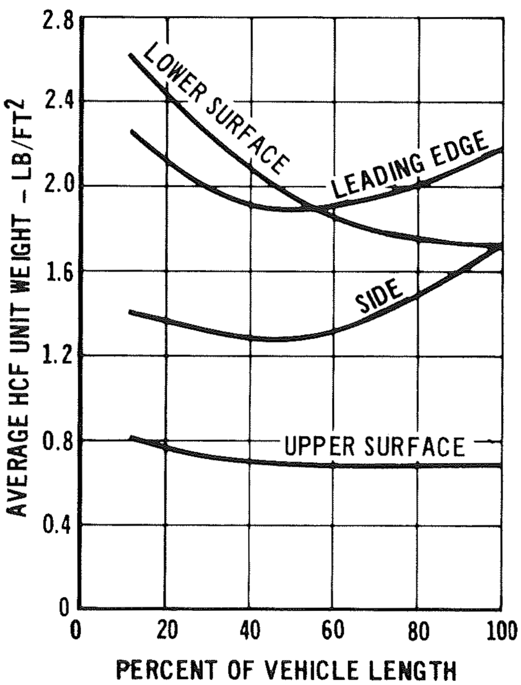
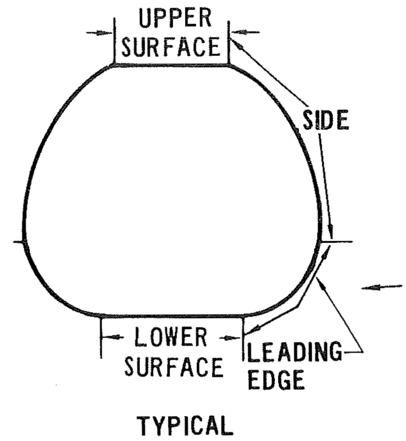


Figure 4-51

ORBITER HCF REQUIREMENTS
Minimum Time (2600°F)

NOTE:
(1) MAXIMUM BONDLINE TEMPERATURE = 500°F
(2) HCF DENSITY = 15 LB/FT³
(3) LAMINAR HEATING





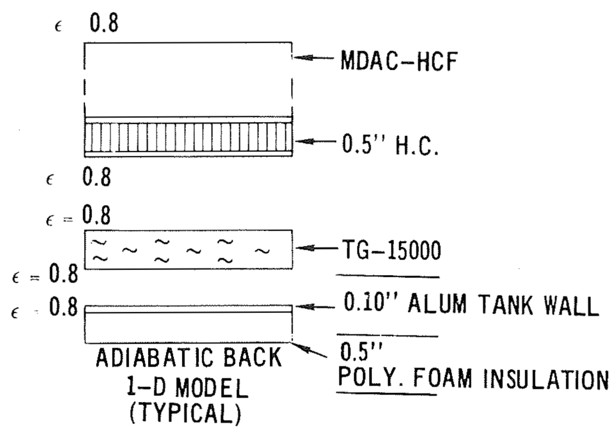


Figure 4-52

Carrier Heating Analysis - Maximum temperatures experienced by the carrier during a nominal launch and reentry for an impulsive velocity of 14,500 ft/sec (actual ΔV = 9170 ft/sec) are shown in Figure 4-53.

These temperatures are for laminar flow and, except for the leading edge of the vertical stabilizer, are based on heat transfer test data obtained from NASA-LRC. Experimental heat transfer tests were conducted on a low wing clipped delta configuration at a Mach number of 10.4 and a Reynolds number of 0.5×10^6 based on model length. The model was coated with a phase change material and local heating rates were determined by interpretation of photographic data. Lines of constant heating rates as interpreted from the photographic data are shown in Figure 4-54. Values shown are ratios of local heating rates to a calculated stagnation point heating rate on a hemisphere having a diameter equal to the vertical thickness of the model. Test results indicate that the high heating rates at the wing root-body juncture are reduced by the fairing incorporated in the baseline shape.

The dorsal fin is shielded at an angle of attack of 50° during reentry. Consequently, temperatures for the dorsal fin leading edge were determined from swept cylinder theory for ascent flight conditions at an angle of attack of zero degrees. Although the leading edge radius decreases with the distance from the base on the dorsal fin, estimated temperatures near the base are higher because of allowance for bow shock wave impingement.

As shown in Figure 4-53, temperatures for laminar flow along the bottom surface are in the range of 800 to 900°F. However, peak heating during reentry occurs at relatively low altitudes (i.e., less than 160,000 ft.) and the flow will be turbulent based on the criterion of onset at a local Reynolds number of 10^6 and fully turbulent flow at 2.0×10^6.

In order to investigate the influence of turbulent flow on maximum temperatures, lacking test data, blunt body modified Newtonian flow was assumed to define local flow properties. It was also assumed that streamline divergence or outflow has little influence on turbulent heating rates and equilibrium temperatures. Based on these assumptions, the variations of local Reynolds number and turbulent temperatures for a wetted length of 50 feet on the lower surface centerline are illustrated for carrier reentry flight conditions in Figure 4-55. It is seen that maximum temperatures along the bottom surface could approach 1100°F for a short time interval which is 300 degrees higher than the maximum laminar temperatures of approximately 800°F shown in Figure 4-53.

MAXIMUM CARRIER TEMPERATURES

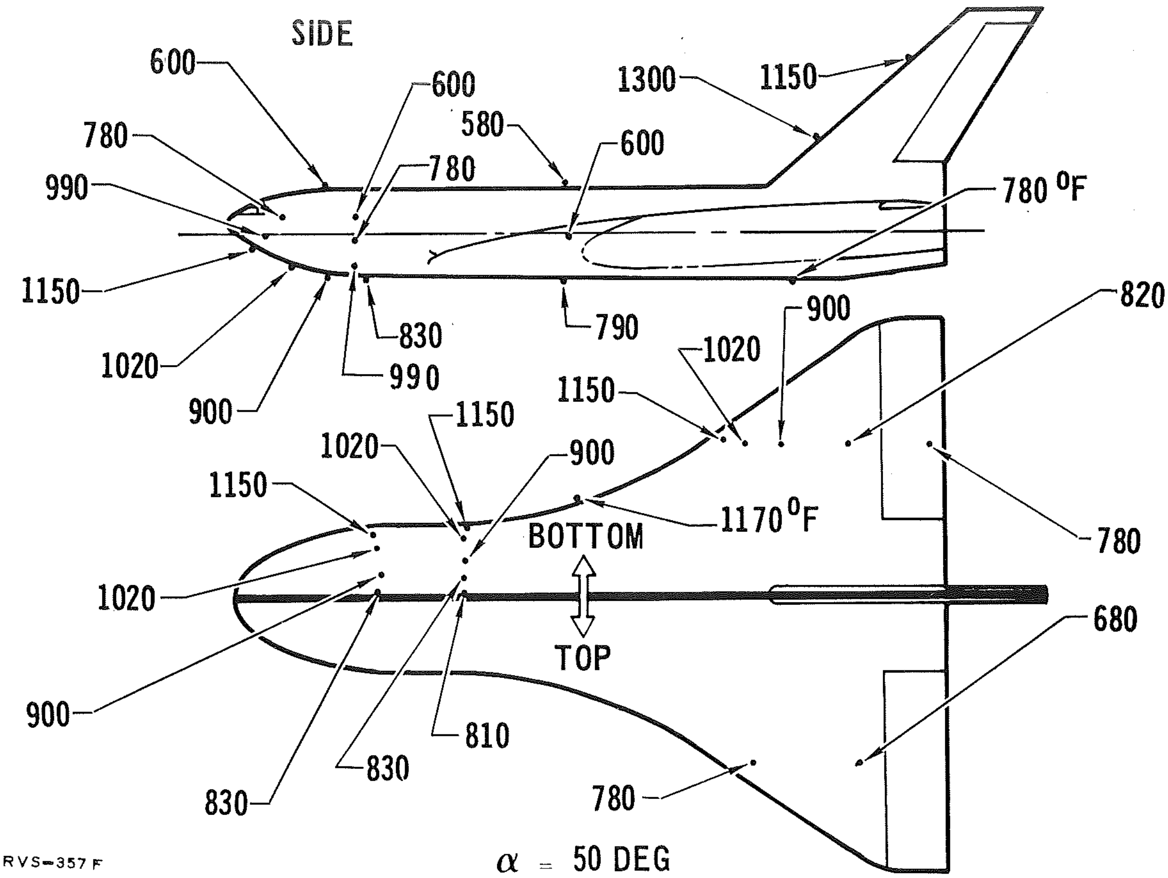


Figure 4-53

CARRIER EXPERIMENTAL HEATING

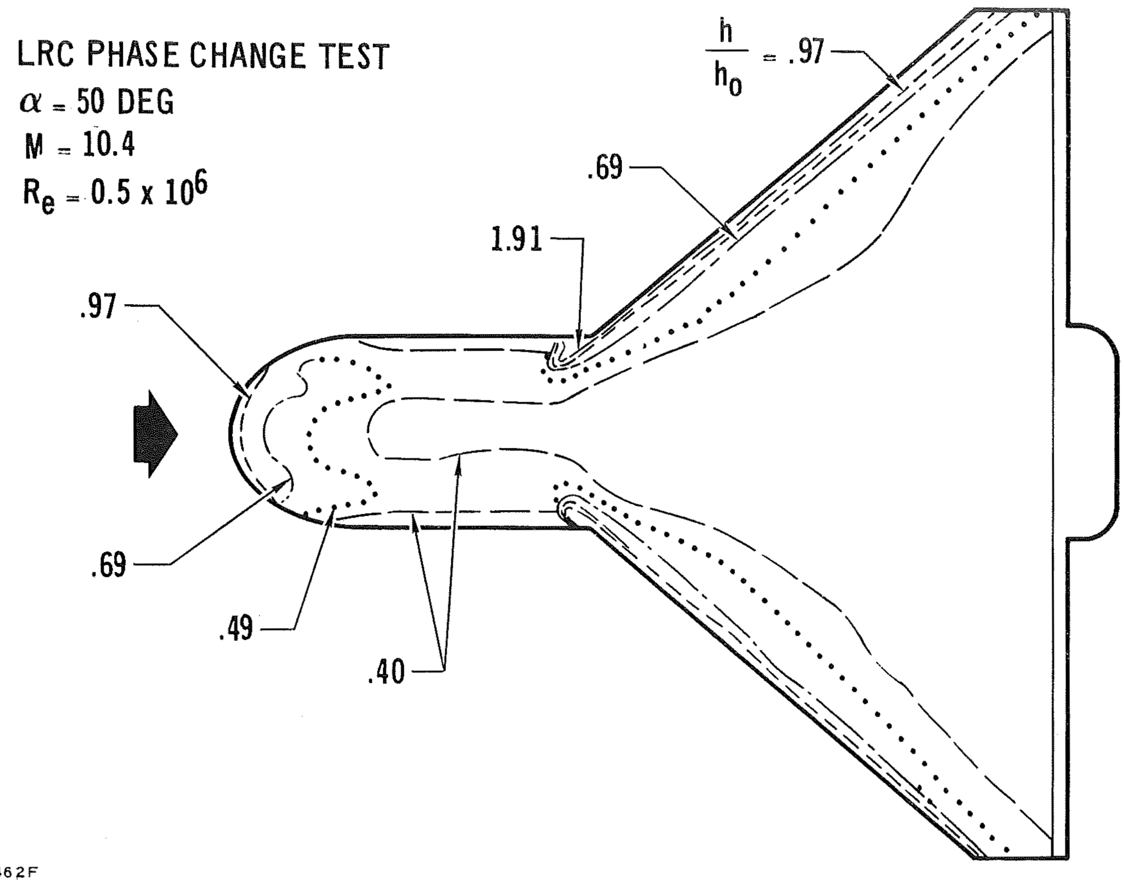
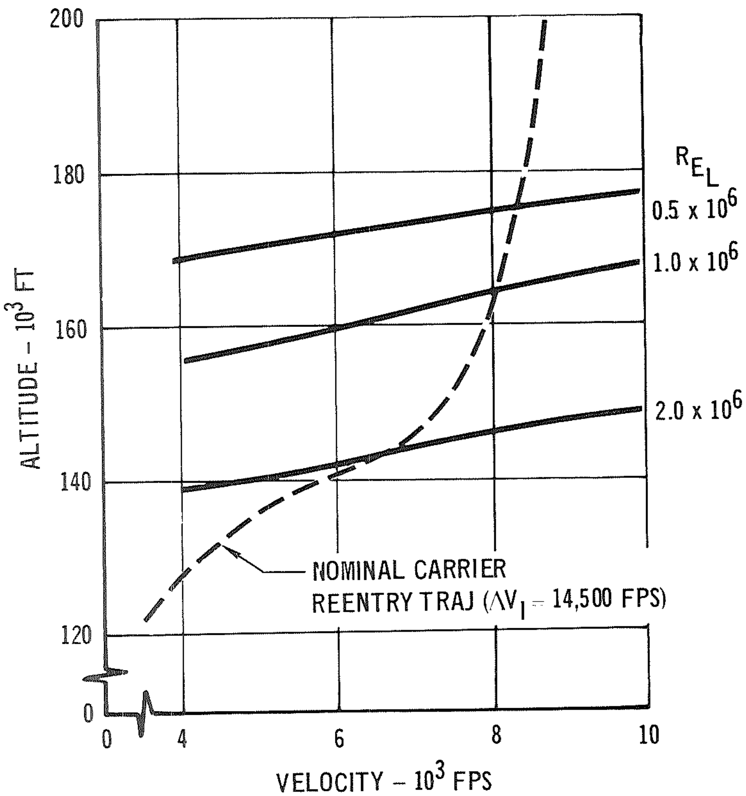


Figure 4-54

CARRIER – TURBULENT HEATING EFFECTS

NOTES:
1. MODIFIED NEWTONIAN FLOW
2. δ = 50 DEG
3. S = 50 FT
4. REF ENTHALPY
5. STRIP THEORY



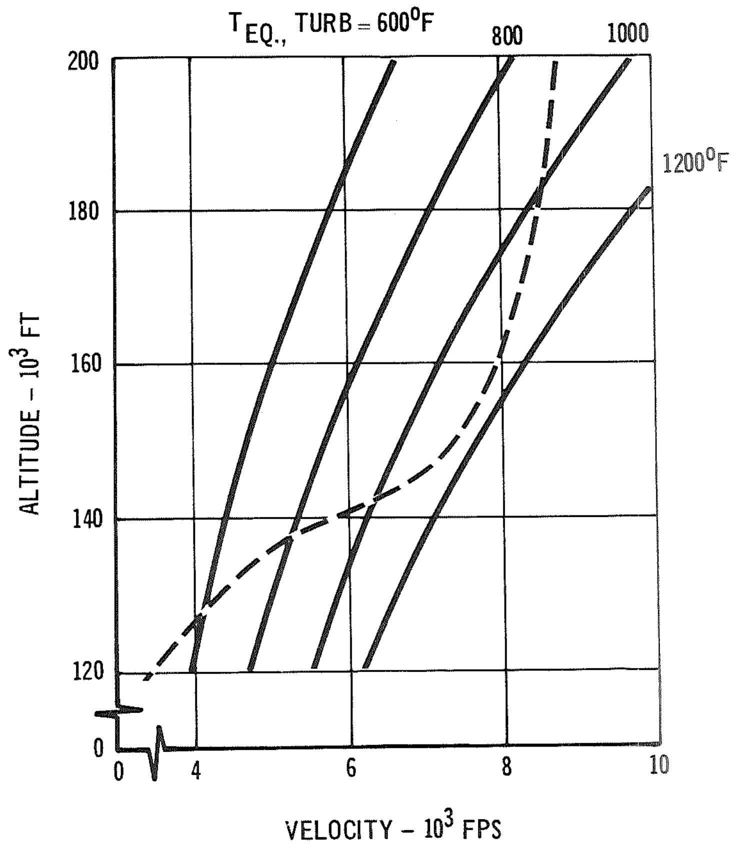


Figure 4-55

Carrier Thermal Protection - The structure and thermal protection arrangement for the carrier is shown in Figure 4-56. Heat transfer between the shingle and tank wall is minimized by insulation and a radiation gap to limit the maximum tank wall temperature to 200°F. It is necessary to limit the maximum tank wall temperature to 200°F so that the freon blown polyurethane foam insulation and NARMCO 7343 adhesive (foam to tank) temperature limit of 200°F is not exceeded.

TPS Shingle Arrangements

Corrugated Panels - Methods used for attachment and support of the corrugated panels are shown in Figure 4-57. A pi shaped retainer entraps the shingles and provides a gap for thermal expansion. The arrangement allows removal of individual panels.

Pressure loads are beamed by the corrugations to supports at the forward and aft edges of the panels. The supports are attached to body frames. The attachment and support concept are similar for radiation cooled shingles of titanium, René 41 and TD-NiCr alloys with the exception a support beam, isolated from frame caps by brackets, is used to react positive pressure loads from TD-NiCr panels whereas titanium and René 41 panels are used in lower temperature zones and bear directly on frame caps through support channels. The pi shaped retainer reacts negative pressure loads on the panels. These loads are introduced into the frames through support brackets. The standoff support brackets are used to minimize the conductive heat path from shingle to primary structure.

Beaded Panels - A typical beaded panel installation is shown in Figure 4-58. These panels are used on the shadowed surface in regions which experience low heating rates. In these areas the surface irregularities due to the beads do not significantly alter the heat inputs or aerodynamic characteristics. Panels are retained by round head screws with clampup bushings. Oversize holes provide for thermal expansion.

HCF Panels - This arrangement (Reference Figure 4-59) is an alternate to the metallic shingle and consists of a hardened compacted fiber (HCF) (Insulation) bonded to a fiberglass honeycomb substructure. The attachment concept is similar to that employed for the corrugated panels with the exception the panel is allowed to bear directly on the frame cap.

CARRIER STRUCTURE & TPS ARRANGEMENT
Integral Tanks – Titanium & Rene '41

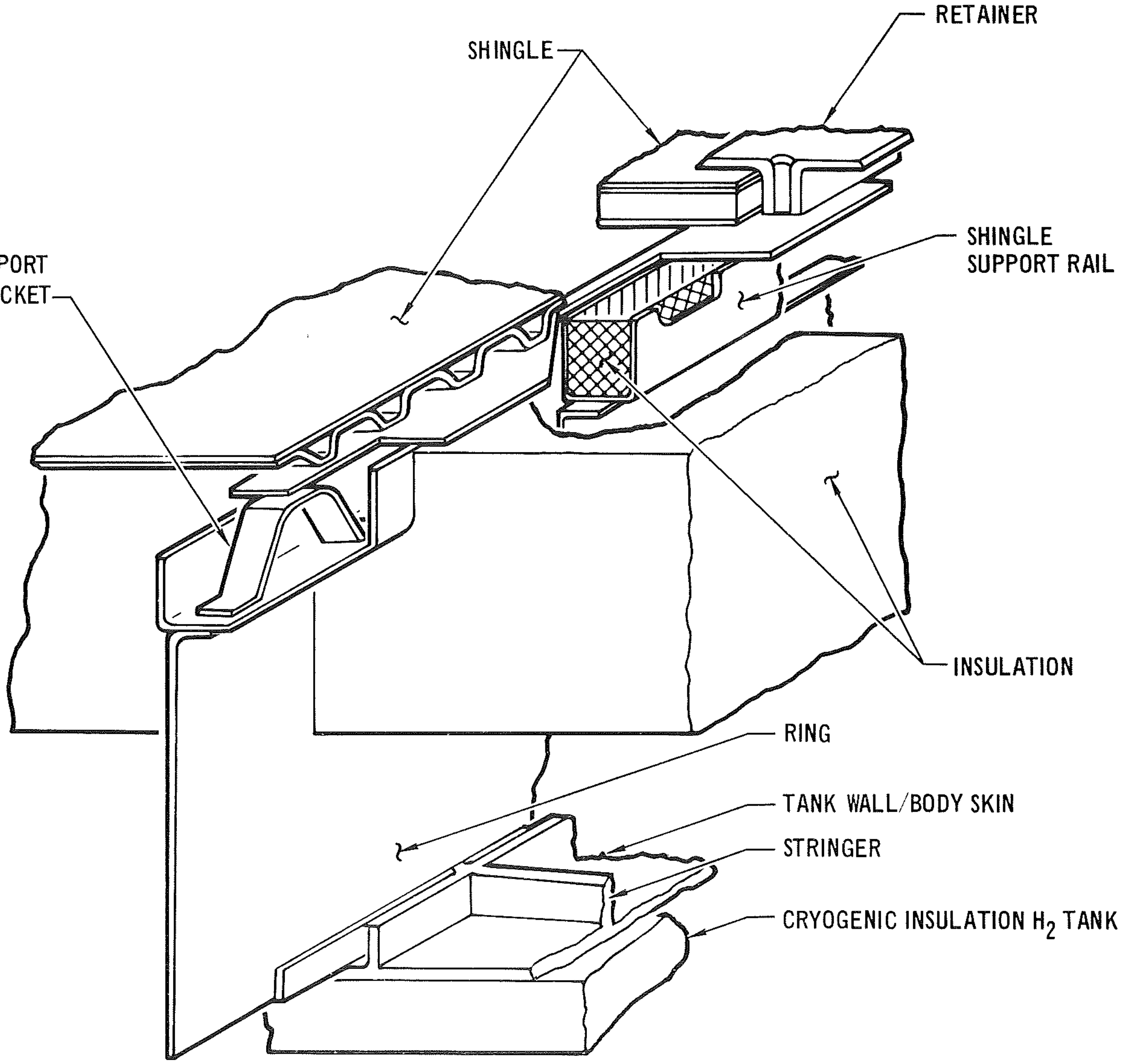


Figure 4–56

METALLIC SHINGLE TPS ARRANGEMENT

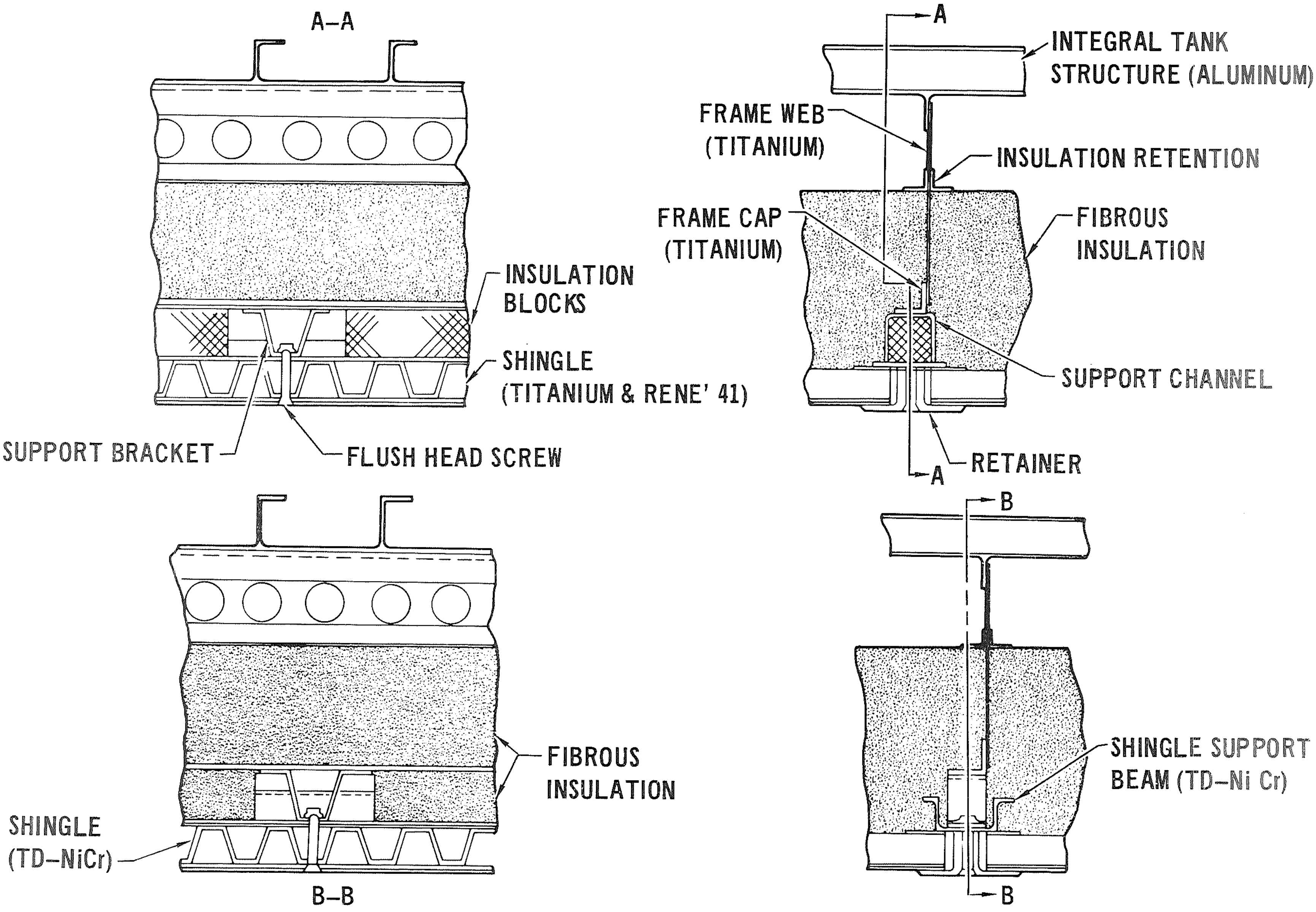


Figure 4-57

METALLIC SHINGLE TPS ARRANGEMENT
(Beaded Panels)

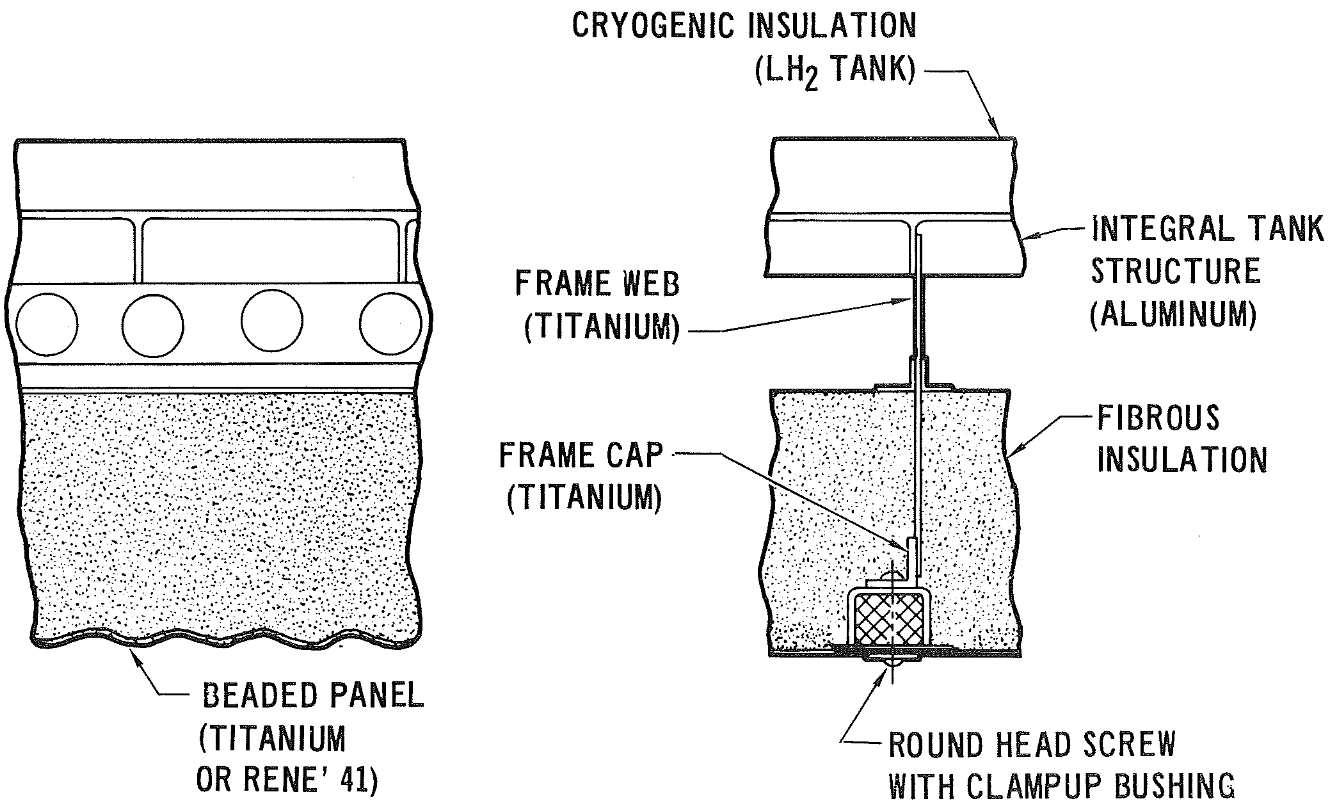


Figure 4-58

HCF SHINGLE TPS ARRANGEMENT

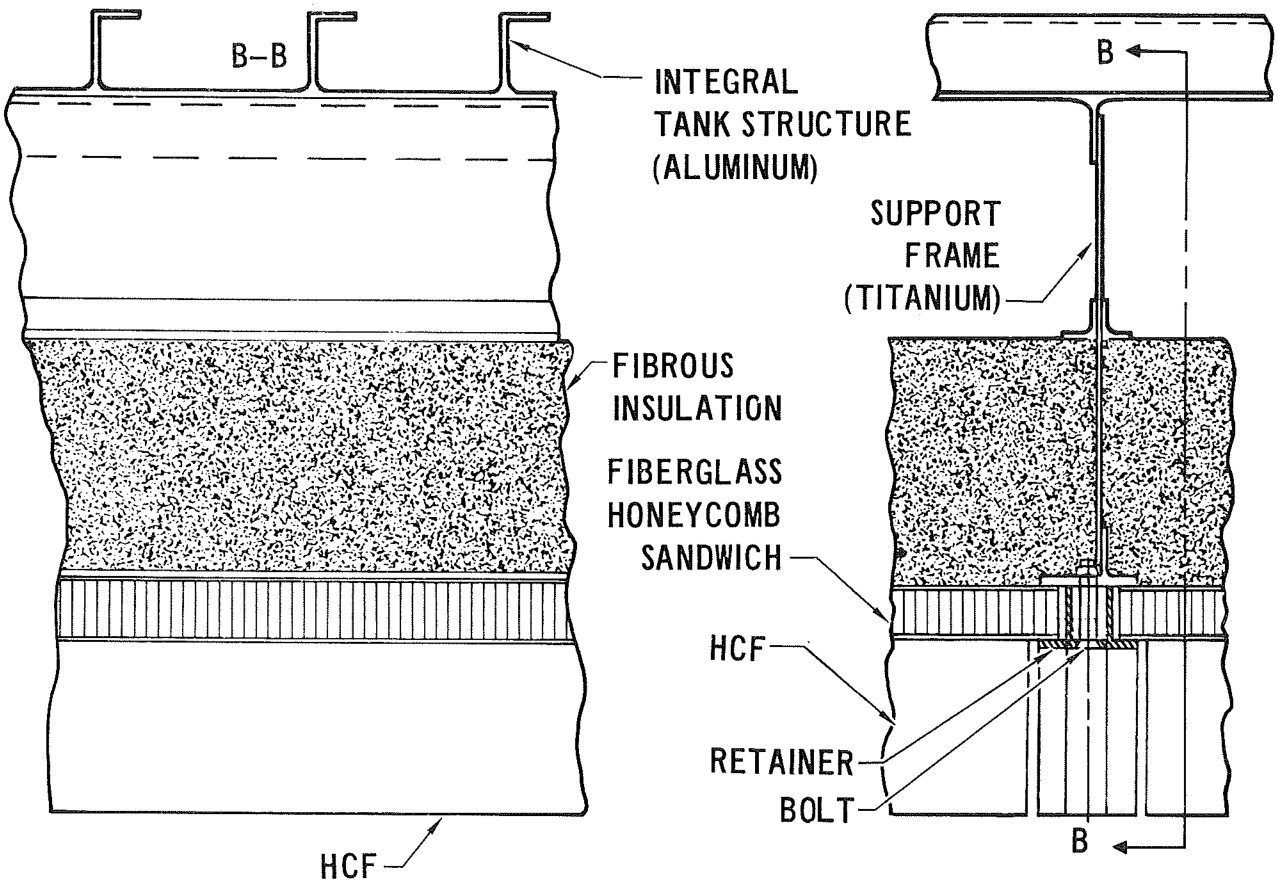


Figure 4–59

4.2.3 Material Distribution of TPS Shingles - Material distributions of TPS shingles for the carrier and orbiter are presented in Figures 4-60 and 4-61. The material distributions shown are derived from the temperature distributions shown in Figures 4-53 and 4-43 for laminar flow. A more detailed investigation including turbulent flow effects over the lower surface may require some deviations from the distributions shown in Figure 4-60. Material selection is based on the following temperature use ranges:

Titanium (8Al-1Mo-1V)	400 - 1000°F
Rene' 41	1000 - 1600°F
TD-NiCr	1600 - 2200°F
Columbium	2200 - 2800°F

The temperature use range upper bounds are based on material strength/density ratios, material metallurgical stability temperature limits and coating life. The metallurgical stability temperature limit is the temperature where a notable change in the metallurgical structure or significant reduction in mechanical properties occurs. If the temperature for metallurgical stability is exceeded, it is important to consider time dependent post heating effects. Exposure to higher temperatures for significant periods of time may result in subsequent reduction in both room and elevated temperature mechanical properties and material ductility. However, test data for some materials has indicated that accumulated temperature effects of recycling from room to peak temperature have considerably less degrading effect on mechanical properties than continuous exposure for the same total time at peak temperature.

The temperature limit of 1000°F employed for titanium alloy 8Al-1Mo-1V is based primarily on the reduction in mechanical properties above this limit. Accumulative exposures to 1000°F for short periods of time will not produce subsequent reduction in room and elevated temperature mechanical properties. Continuous exposure (10 hrs) of Rene' 41 above 1400°F has resulted in degradation of subsequent room and elevated temperature mechanical properties; however, it is felt that short time exposures to 1600°F can be tolerated with negligible effect on mechanical properties. Ten hours of accumulative 6 minute exposures to peak temperature per flight is representative of an orbiter with a 100-flight life. The temperature limit of 2200°F utilized for thorium-dispersed, nickel chrome (TD-NiCr) is based on the metallurgical stability limit. Columbium alloy upper bound is based on coating life for 100 flights.

MATERIAL DISTRIBUTION OF TPS SHINGLES CARRIER

NOTE: WING & VERTICAL TAIL SURFACE PANELS
ARE STRUCTURAL (TITANIUM WITH RENE LEADING EDGES)

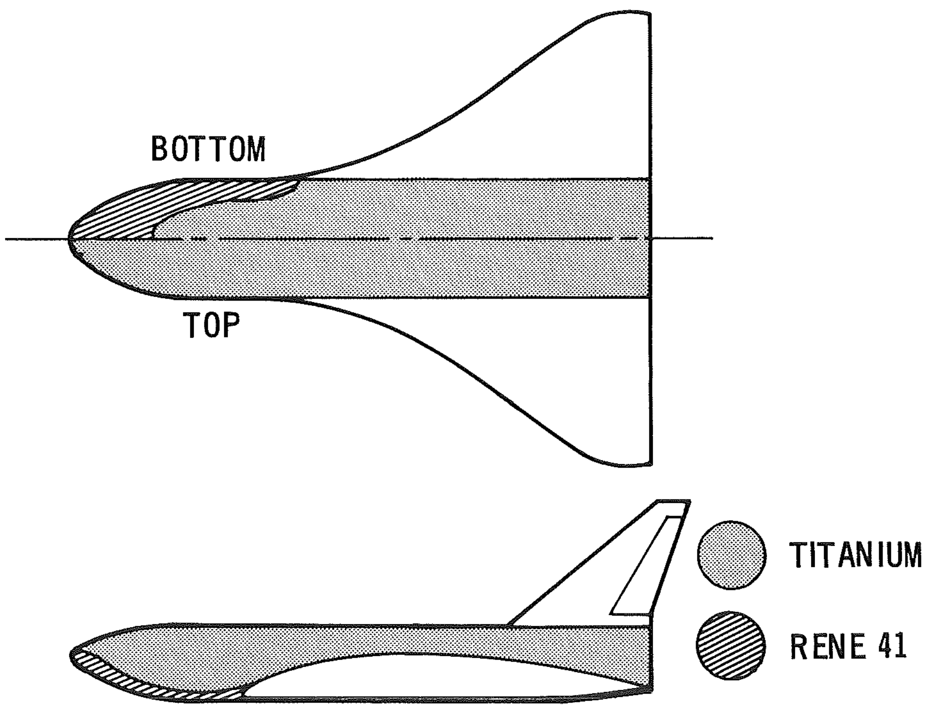


Figure 4-60

MATERIAL DISTRIBUTION OF TPS SHINGLES, ORBITER

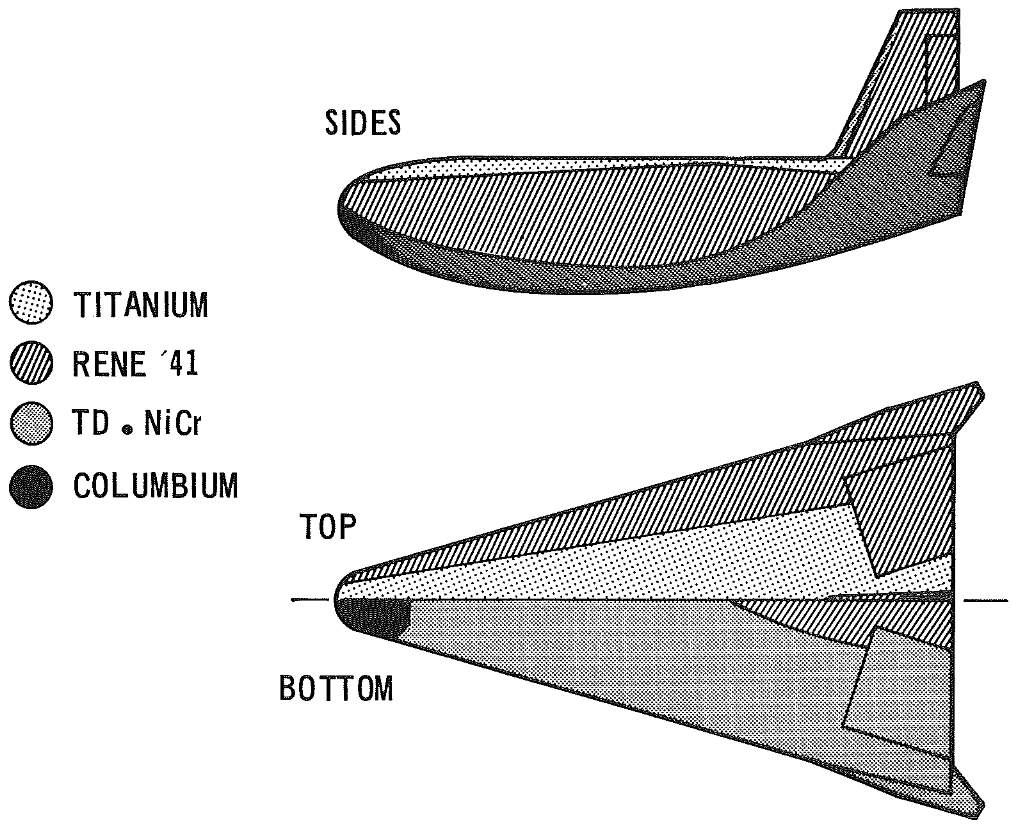


Figure 4-61

TPS shingle material distribution for an alternate orbiter configuration is shown in Figure 4-62. The arrangement is an alternative to the metallic shingle in regions between 1600°F and 2600°F. Corrugated panels are replaced by HCF bonded to a fiberglass honeycomb substructure.

MATERIAL DISTRIBUTION OF TPS SHINGLES ORBITER
Alternate Configuration

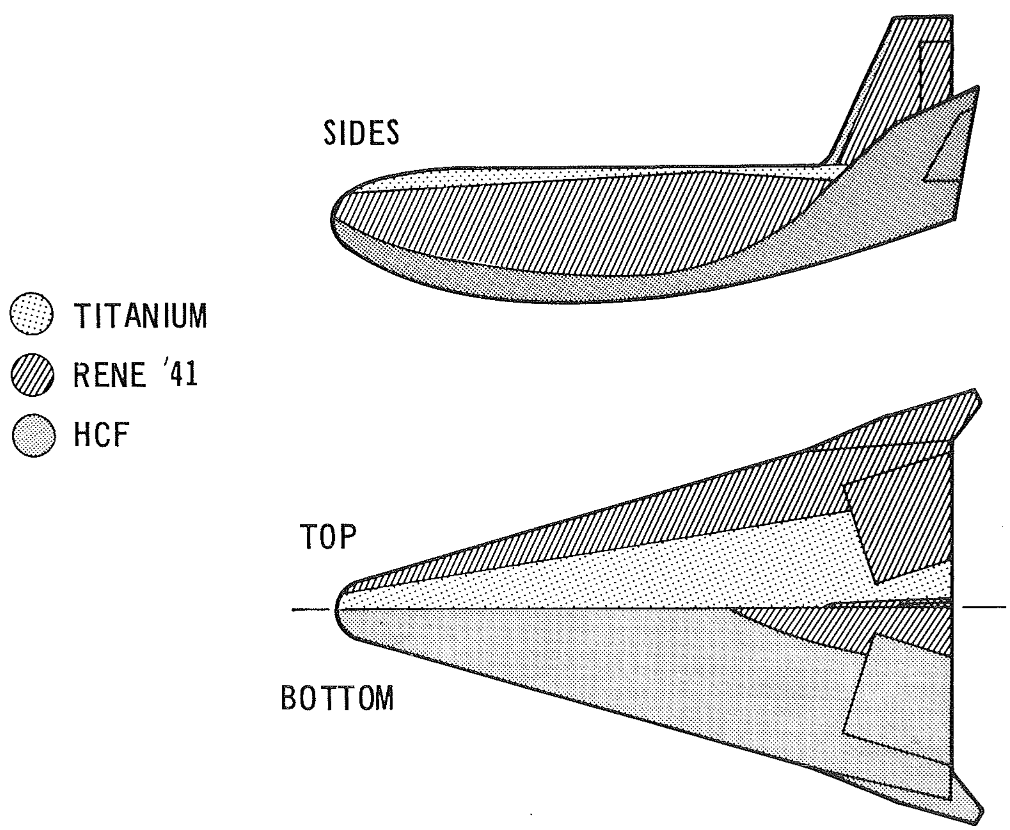


Figure 4-62

4.3 Integrated Avionics - The emphasis of the ILRV program is to achieve a high level of operational economy. This requirement, in conjunction with vehicle operation in the booster, spacecraft and aircraft flight regimes requires a new look at the design and implementation of the Avionics System. The new approach is called an "Integrated Avionics System". It considers all known functional requirements of the mission during initial vehicle system design.

The basic rationale for the use of Integrated Avionics is derived from the measures required to achieve economy of operation. These measures are a self-contained, crew controlled, prelaunch checkout capability, rapid turn around/reuse capability and a higher degree of mission success. Avionic capabilities must include self checkout, block and functional redundancy, and maintenance to a line replaceable unit (LRU). These capabilities produce a large amount of system status data. This data in conjunction with the system complexity due to the vehicle multiregime operation, requires an advanced Integrated Avionics capability. To ensure compatibility with manned control, the Integrated Avionics system will provide a highly efficient data management and display/control capability for compatibility with manned crew command and control. It will relieve the crew of excessive workload by automatically performing time critical functions and providing priority sorting and data compression of that information needed by the crew.

The general Avionic functions are:

- Vehicle Self Test and Warning
- Data Processing and Transfer
- Crew Command and Integrated Displays
- Target Tracking
- Autonomous Navigation and Flight Control
- Satellite Communications
- Supporting Energy Conditioning

More specific functions by mission phase are described in Figure 4-63.

The key questions to be answered in order to define the Integrated Avionics System are the means of implementing Data Management; On-Board Checkout; Display and Control; and Reliability as well as Reuse. The features that were evaluated in preliminary tradeoffs in this study are indicated in Table 4-8. These tradeoffs will be described and preliminary results indicated after the summary baseline system definition and description.

The integrated avionics study described in this section represents a special emphasis area. The automatic landing system discussion in Section 4.3.5 also constitutes a portion of the approach and landing special emphasis study. That study is described in Section 3.4.7 of Volume II.

AVIONICS – MISSION FUNCTIONS

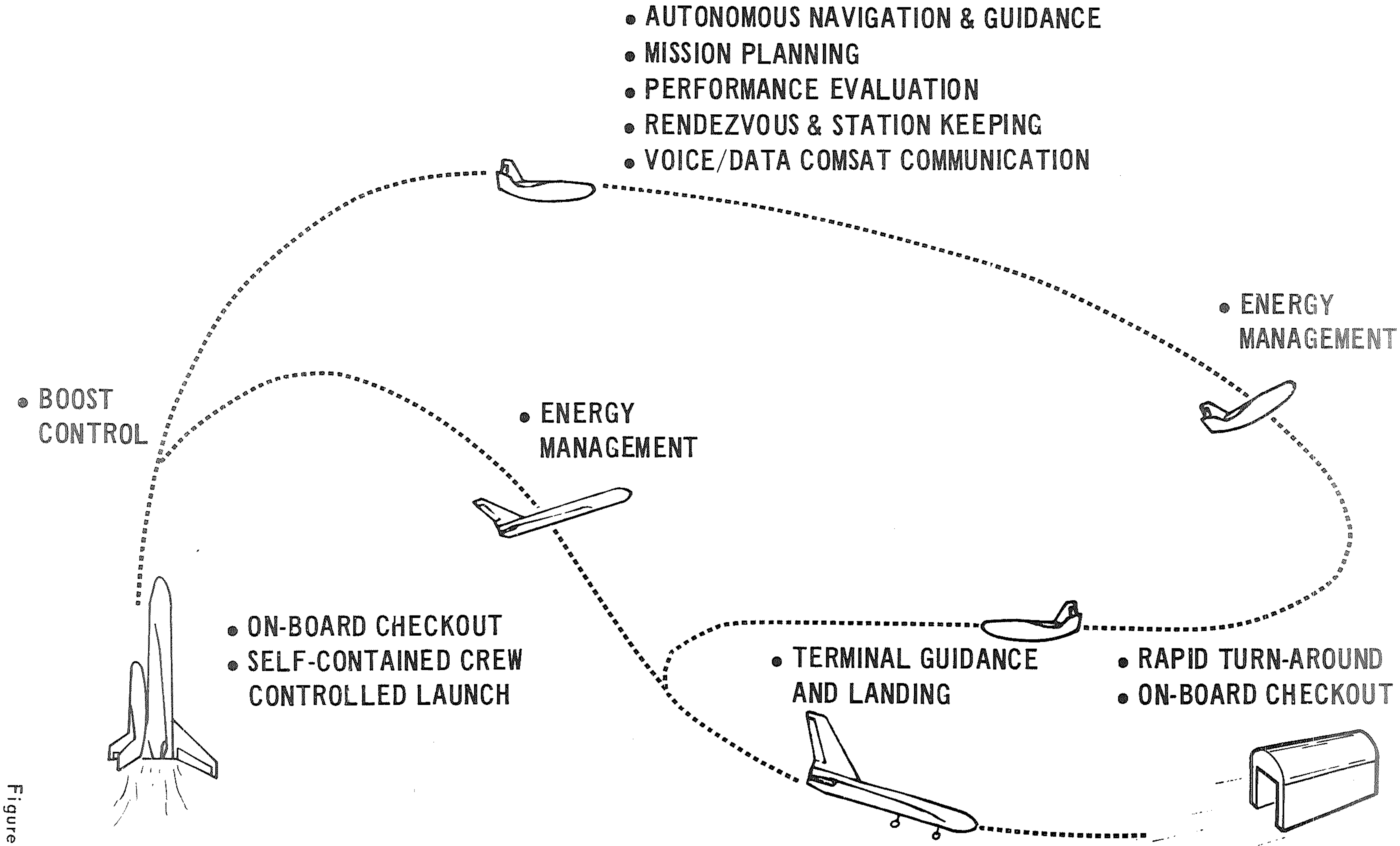


Figure 4-63

Table 4-8

KEY CONCEPTS AND TRADEOFFS

DATA MANAGEMENT	• DEGREE OF COMPUTATIONAL DISTRIBUTION (CENTRALIZED VS DECENTRALIZED) • DATA INTERFACE TECHNIQUE (MULTIPLEXED VS NONMULTIPLEXED)
ON-BOARD CHECKOUT	• SEPARATE CHECKOUT SYSTEM VS DECENTRALIZED BUILT-IN TEST • MANUAL VS AUTOMATIC • LEVEL OF FAULT ISOLATION AND MAINTENANCE
DISPLAY & CONTROL	• MULTIMODE INTEGRATED DISPLAYS VS SINGLE PURPOSE INDIVIDUAL DISPLAYS • UTILIZE SPECIAL HEADS UP DISPLAY
RELIABILITY & REUSE	• USE OF BLOCK VS FUNCTIONAL REDUNDANCY • INTERACTION OF REDUNDANCY VS SELF TEST • EVALUATION OF ACTIVE/PASSIVE/STANDBY REDUNDANCY • MALFUNCTION DETECTION AND SWITCHOVER

4.3.1 System Definition - The elements of the Integrated Avionics System are shown in Figure 4-64. Equipment and configuration selection was made on the basis of: (1) an estimate of the 1972 technology status and (2) use of concepts which provide small development risks.

Inertial sensors are used as the prime source of navigation data through all active mission phases. Choice of inertial systems in both the carrier and the orbiter were primarily dictated by the ascent guidance, entry to a predetermined landing site and automatic landing requirements. Star trackers and horizon sensors provide autonomous on-orbit attitude and navigational updates. The multi-mode rendezvous radar provides for rendezvous with either cooperative or non-cooperative vehicles. A dedicated navigation computer supplies the unique requirements of individual system sensors while permitting the central software programming tasks to be maintained at a manageable complexity level. This keeps sensor unique computational requirements from impacting the central computational requirements.

The UHF communication link is utilized for EVA, inter-vehicle voice or data and airport communication during the approach and landing phase. The Comsat-link provides nearly continuous communication capability between any ground station and the orbiter during the orbital phase of flight.

The display concept utilizing cathode ray tubes for multimode data presentation, permits crew decisions on important tasks while relieving them of the need to monitor a large number of displays and meters.

A common multiplexed data bus was selected to provide standardization of digital interfaces, and to reduce the complexity and weight of interconnecting systems. The intermix of computers, consists of a central data processor performing mission oriented functions and peripheral dedicated computers for sensor functions, navigation, flight control, and propulsion computations. This arrangement was chosen on the basis of commonality of requirements while maintaining equipment and software at manageable complexity levels. Thus, sensor oriented computational requirements both hardware and software, do not impact the central computer.

On-Board Checkout minimizes ground support and expedites maintenance and reuse. Decentralized built-in-test (BIT) was selected over a separate centralized test system to minimize interface complexity and provide subsystem functional autonomy. BIT provides self-test at all maintenance levels and permits

BASELINE ORBITER INTEGRATED AVIONICS SYSTEM

COMMUNICATIONS
- SHF* (2) & UHF (2) TRANSCEIVERS
- PROCESSOR (3)
- INTERCOM & HEADSETS (2)
- OMNI ANTENNAS (4)
- *SHF DISH ANTENNA (1)

GUIDANCE AND NAVIGATION
- INERTIAL MEASUREMENT (3)
- G&N COMPUTER (3)
- *INTEGRATED OPTICAL & IR (2)
- *RENDEZVOUS RADAR (2)

LANDING & NAVIGATION AIDS
- VORTAC TRANSCEIVER (2)
- RADAR ALTIMETERS (3)
- AIR DATA SENSORS (3)
- ADVANCED ILS (3)

ELECTRICAL
- *FUEL CELL (2 OUT OF 4)
- *REACTANT SUPPLY (25%)
- BATTERIES (2)
- INVERTERS (4)

CENTRAL MANAGEMENT
- CENTRAL COMPUTER (3)
- CREW

DISPLAY & CONTROL
- MODE CONTROL PANEL (2)
- HAND CONTROLLERS (2)
- MULTI-PURPOSE CRT DISPLAYS (6)
- HEAD-UP DISPLAY (2)
- *PRINTER (1)

CHECKOUT & MONITORING
- PROPULSION PROCESSOR (3)
- INSTRUMENTATION PROCESSOR (3)
- INSTRUMENTATION SENSORS
- REMOTE MULTIPLEXERS
- FLIGHT RECORDER (2)
- *REMOTE TV CAMERAS

FLIGHT CONTROL
- PROCESSOR (4)
- POWER SERVO AMPLIFIERS (4 PER FUNCTION)
- RATE GYRO (BACK-UP – 1)

*DENOTES SYSTEMS NOT USED ON CARRIER

Figure 4-64

identification of failures to the line replaceable units. Selective computer controlled access permits transmission of data pertinent to a particular mission phase, whether it be for flight caution and warning, or ground base checkout.

Table 4-9 shows size, power, and weight of the selected equipment. Carrier equipment is identical to that of the orbiter except that equipment utilized only for orbital operations is deleted. Such equipment is identified in Figure 4-64 as well as the level of equipment redundancy.

A more detailed system definition, including tradeoffs, recommendation, and conclusion, is contained in the following paragraphs.

Table 4-9

ORBITER INTEGRATED AVIONICS PHYSICAL CHARACTERISTICS

EQUIPMENT TYPE	WEIGHT (LB)	SIZE (CU FT)	OPERATING POWER (WATTS)
GUIDANCE & NAVIGATION	720	11.8	2270
LANDING & NAVIGATION AIDS	170	3.05	460
TELECOMMUNICATIONS	325	48.85	545
CENTRAL MANAGEMENT COMPUTER	180	3.0	500
DISPLAYS, CONTROL & SEQUENCING	477	8.25	1525
FLIGHT CONTROL	75	1.3	245
CONTROL AMPLIFIERS	122	2.25	870
INSTRUMENTATION	125	2.1	260
POWER GENERATION	1658	36.0	10 KW (CAPACITY)
PWR DISTR WIRE	700	14.0	
SIGNAL DISTR WIRE	1300	20.0	
TOTAL ORBITER AVIONICS	5852	151.0	(5765) (PEAK)
(TOTAL CARRIER AVIONICS)	(4065)	(66)	(5042) (PEAK)

4.3.2 Data Management System (DMS)

Summary - The Space Shuttle will utilize an onboard computerized data management system to provide the information processing and system control required for autonomous vehicle operation. A baseline system was selected after a conceptual study of promising candidate approaches. This system divides the computational requirements between a general purpose central computer for mission oriented functions and special purpose dedicated peripheral computers for sensor oriented functions. A redundant multiplexed data bus is employed to reduce the weight and installation complexity of wire bundles. Standard digital interface circuitry was selected to provide flexibility and simplify the interface design and management problem.

Requirements - The multitude of computational tasks that must be performed accurately and rapidly is beyond crew manual capability, and reliance on ground-based-computers is not compatible with the autonomous nature of the space shuttle. For these reasons an onboard data management system (DMS) is required. The DMS will provide the following functional requirements:

a) Computational capability required by other subsystems during all phases of the mission.
b) Standard electronic circuitry to interface with a redundant multiplexed data bus.

System Description - The data management system is involved with the total complement of hardware and software required for data acquisition, processing, analysis and distribution of information to the space shuttles crew and other using subsystems. The two major aspects of the DMS task are the computational requirements and the data bus implementation techniques.

Computational Requirements and Allocations - Table 4-10 presents a list of subsystems and their information/computational requirements. This figure provides an insight to the magnitude of the computational task. In addition to conventional spacecraft computations such as guidance/navigation we have unique requirements such as propulsion trend data analysis which will be used to expedite ground maintenance.

The majority of these calculations are performed in the central computer complex (CCC). However, some subsystems utilize dedicated special purpose computational devices to satisfy unique computational requirements. Figure 4-65 shows

Table 4-10

COMPUTATIONAL REQUIREMENTS

GUIDANCE	ASCENT, ORBIT, RENDEZVOUS, REENTRY, LANDING, ABORT
NAVIGATION	INERTIAL, AUGMENTED INERTIAL, AUTONOMOUS
FLIGHT CONTROL	ATTITUDE, STABILIZATION
ON-BOARD MISSION	TRAJECTORY GENERATION, OPTIMIZATION & SELECTION,
PLANNING	FLIGHT PROGRAM IDENTIFICATION, LOAD ALLEVIATION, ACCESSMENT OF UPLINK INFORMATION, CREW USAGE FOR SCIENTIFIC CALCULATIONS
CONFIGURATION AND SEQUENCING CONTROL	PAYLOAD PREPARATION & DEPLOYMENT, SYSTEM READINESS, SENSOR CONTROL, SAFING OPERATIONS, EXPERIMENT ACTIVATION & CONTROL, PILOT CHECK LIST
CREW DISPLAYS	SYMBOL GENERATION – PRIORITY & FUNCTIONAL SORTING
ON-BOARD CHECKOUT	STIMULUS GENERATION, PARAMETER TOLERANCE BAND COMPARISON, TREND DATA EVALUATION
PROPULSION	OPERATION, PROPELLANT UTILIZATION, MALFUNCTION DETECTION, TREND ANALYSIS
BIOMEDICAL	INFORMATION PROCESSING & EVALUATION – HEART RATE, BREATH RATE, FLUIDS ANALYSIS
DATA BUS MANAGEMENT	REQUEST/REPLY OPERATION, MESSAGE TRANSFER VERIFICATION

DATA MANAGEMENT SYSTEM BLOCK DIAGRAM

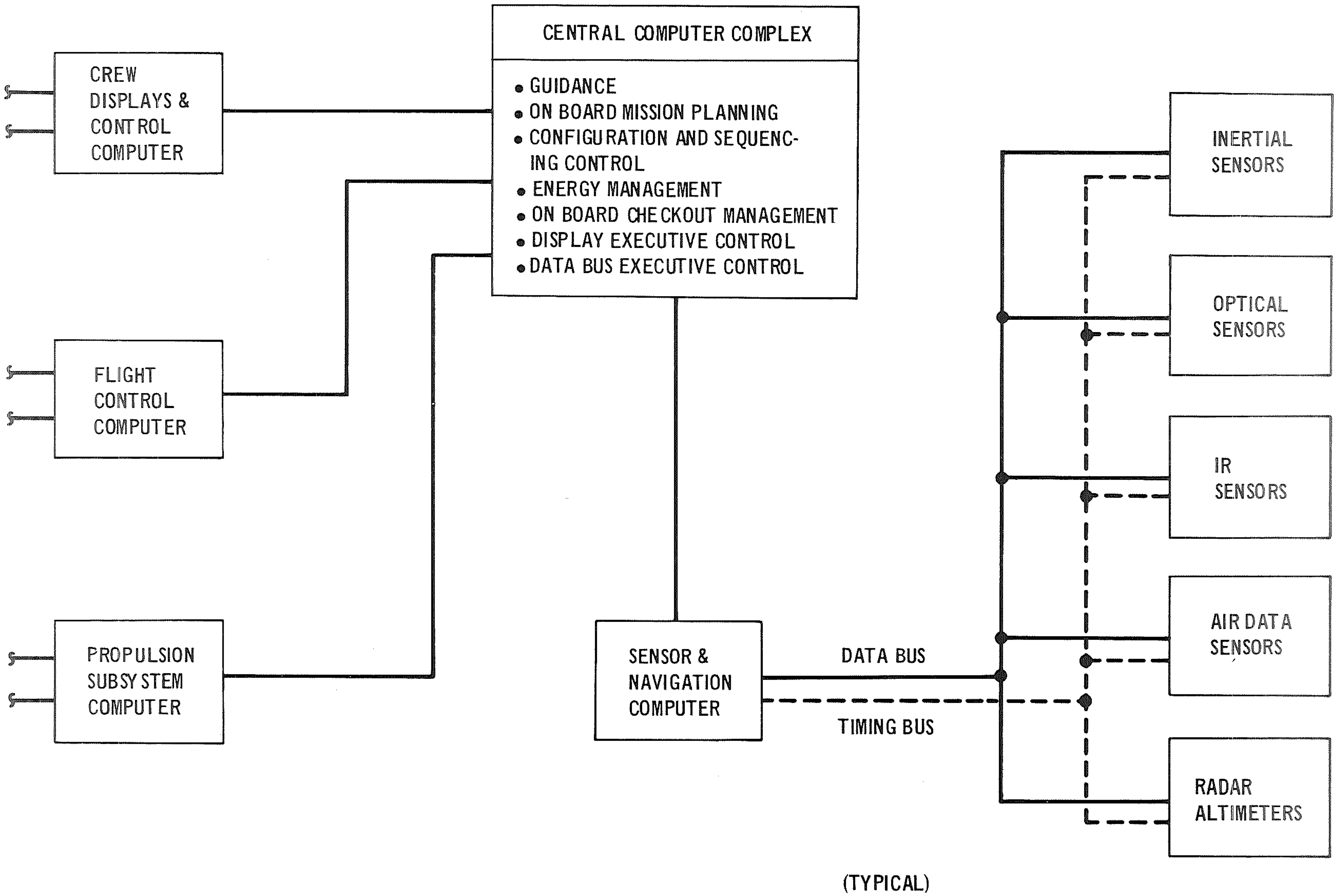


Figure 4-65

the inter-relationship of the assemblies and identifies the major signal interfaces with other vehicle subsystems.

- The central computer performs the mission oriented calculations such as those required for guidance and onboard mission planning. In general these are similar type computations and by grouping them in this same computer, software may be shared.
- The onboard checkout system utilizes built-in test (BIT). This requires that special logic and stimulus generation circuits be built into each line replaceable unit (LRU). The central computer continuously monitors the BIT control panel to determine the status of each LRU. The results of this routine are evaluated by CCC and display instructions are sent to the symbology generator for initiation of status displays to the crew.
- A special purpose dedicated computer will perform the calculations necessary for control of the propulsion subsystem. The propulsion subsystem main elements are jet engines, main propulsion boost engines, and ACS reaction jet engines. These engines are distributed throughout the vehicle and remotely located from the central computer. The large amount of data associated with propulsion calculations such as propellant utilization and the relatively remote location of propulsion equipment determines the need for a dedicated computer.
- The sensor and navigation subsystem has a number of high iteration rate and unique type computational requirements such as strapdown IMU coordinate determinations. A dedicated computer handles these requirements without impacting the central computer.
- A special purpose computer is assigned to the flight control subsystem. This subsystem provides high iteration rate control signals over a multitude of mission modes to a large number of control elements such as aerodynamic surfaces, thrusters, and brakes. The resultant large amount of data and diverse data traffic flow patterns justifies a dedicated computer.
- Cathode ray tubes were selected as the prime method for providing the crew with information displays because of their multimode capability. The implementation technique chosen for generation of cathode ray tube (CRT) displays requires extensive symbology memory capability and high speed calculations related to CRT beam deflection and blanking. A special data processor is assigned to the crew display subsystem for this purpose.

Data Bus Implementation Techniques - Current spacecraft and aircraft utilize individual hard wires as the transmission medium between black boxes and from subsystem to subsystem. The signal transmission system chosen for the space shuttle is a multiplexed data bus system. Equipments "share" this party line by use of standard interface circuitry and multiplexing techniques. This eliminates large, heavy and inflexible wire bundles. The resultant weight and space savings allow for the use of redundant buses to improve reliability. Data and signal interconnections between black boxes and between subsystems are via a two-wire twisted pair shielded cable. Selected analog signals and power will be routed by individual wires.

Figure 4-65 shows the navigation sensors connected to the navigation subsystem dedicated computer by means of a separate data bus. A timing bus is also shown for completeness. From preliminary estimates of data rates and data flow traffic patterns, it appears that separate buses will also be required for the flight control system and the propulsion subsystem. Intra subsystem information such as computational data, status information and control commands will be multiplexed on each subsystem bus. The peripheral computers will be connected to the central computer with the individual wires as opposed to a multiplexed bus. The reason for this is because computer-to-computer data rates are in excess of a single bus capacity. Simultaneous transmissions from computer-to-computer is also a requirement and this is not compatible with a "shared" party line bus concept.

The system employs serial digital time division multiplexing (TDM) and is computer controlled using a request/reply data flow control technique. Bi-phase (Manchester) digital coding and alternating current coupling methods were selected. The system timing reference (clock) required for synchronization is transmitted over a separate bus.

Management of the interface can be greatly simplified if the data bus system includes standard digital interface circuitry (SDIC) in addition to the transmission bus itself. Figure 4-66 depicts this. With one standard design each subsystem vendor does not have to invent the same circuit. Development of SDIC will provide isolation and facilitate interface management. Table 4-11 expands these thoughts.

DATA BUS MANAGEMENT

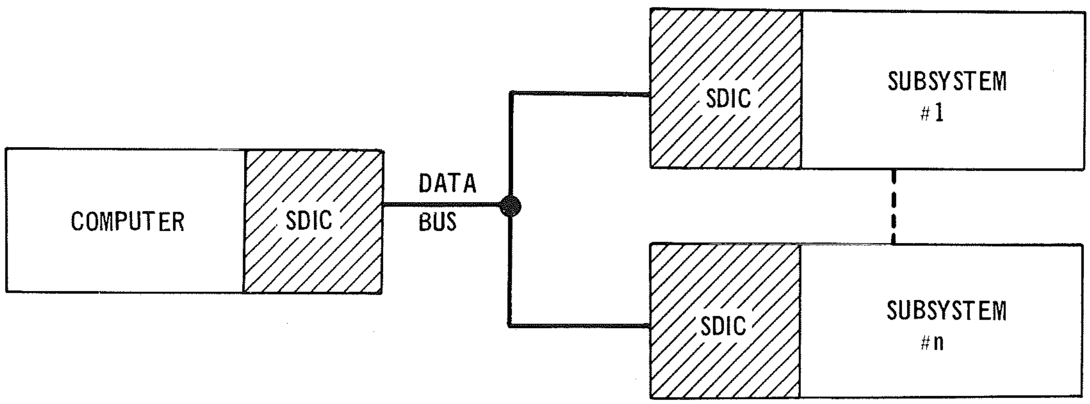


Figure 4-66

Table 4-11

DATA BUS CONSIDERATIONS

ISOLATION	• EQUIPMENT CAN'T CAUSE GLITCH ON DATA BUS • ELIMINATES MAJOR REDESIGN LATE IN PROGRAM CAUSED BY DISCOVERING OF PROBLEMS AT THE SYSTEM LEVEL TEST • ALLOWS PARALLEL DEVELOPMENT OF CEI EQUIPMENT WITH BUILD-AS-YOU-GO SYSTEM LEVEL TEST
INTERFACE MANAGEMENT	• ADDITIONAL INTERFACE SPEC (COMPARED TO CASE OF BOUNDARY AT THE WIRES), BUT IT IS EASY BECAUSE IT IS STANDARDIZED. • STANDARD DIGITAL INTERFACE REQUIREMENTS EASILY MET BY EACH CEI VENDOR USING WELL DEVELOPED ELECTRONIC TECHNOLOGY. • MAXIMUM EXPLORATION OF BUS WIRE CAPACITY • ASSURANCE THAT ALL TERMINALS PATCHING INTO BUS WIRES WILL LOOK EXACTLY ALIKE AND WILL THEREFORE PLAY TOGETHER. • GOOD FLEXIBILITY AND GROWTH CAPABILITY

A data rate of one million bits per second was selected because:

- o Most computation and control functions must be accomplished on a real time basis. This rate is fast enough so that the time between data samples or control functions is short enough not to affect system operation or to introduce system dynamic errors.
- o This rate is the upper limit for using simple data transmission techniques and state-of-the-art qualified electronics.
- o Data flow rates are estimated to be much lower than bus capacity. Thus, growth capability exists since additional black boxes or subsystems could be added at a later date.

The data bus transmission system described above will provide flexibility, simplify the interfaces, reduce the weight and installation complexity of wire bundles, reduce the time and complexity of the manufacturing and checkout operations, and simplify the installation and removal of equipment.

Alternate Concept Evaluation - The centralization versus decentralization of computational equipment is a major consideration in determining the design philosophy and subsequent design configuration of the data management system. Five alternate computational approaches were evaluated. Table 4-12 presents the results of this conceptual trade study. The selected allocation of computers consists of a central computer complex performing mission oriented functions and peripheral dedicated computers for sensor oriented functions and was chosen on the basis of commonality of requirements and physical location. As an example of the advantage of grouping like computations in the central computer, the guidance algorithms may be used for both guidance and mission trajectory planning. In addition, the software can be modularized to reduce costs and provide redundancy. This approach maintains the hardware and software at manageable complexity levels. This also provides flexibility by facilitating changes since the sensor oriented computational requirements, both hardware and software, do not impact the central computer.

Various interface implementation techniques were considered. Table 4-13 identifies the candidate approaches, baseline system selections and rationale.

Digital time division multiplexing requires precise synchronization of transmitter and receiver so that received data can be detected and decoded accurately. Synchronization can be obtained by use of an accurate timing reference (clock)

Table 4–12

DMS COMPUTER DISTRIBUTION

COMPUTATIONAL APPROACHES	SELECTION RATIONALE
CENTRALIZED – CENTRAL COMPUTER/ MULTIPROCESSOR	• COMPUTER REQUIREMENTS LARGE • MAXIMIZES DATA TRANSFER AND BUS REQUIREMENTS • MULTIPROCESSORS NOT DEVELOPED • SOFTWARE TOO COMPLEX
DECENTRALIZED – DEDICATED COMPUTER FOR EACH SUBSYSTEM	• UPWARDS OF 30 COMPUTERS REQUIRED (INCLUDING REDUNDANCY REQUIREMENTS) • EXECUTIVE CONTROL/INTERFACE VERY COMPLEX • MANY DIFFERENT SPECIAL PURPOSE COMPUTER DESIGNS DUE TO DIFFERENT SPEED, WORD LENGTH, STORAGE, AND SOFTWARE. REQUIRES DIFFERENT SPECIFICATIONS, VENDORS, ETC.
FUNCTIONAL COMMONALITY – OPERATIONAL COMPUTER, STATUS COMPUTER, DISPLAY & CONTROL COMPUTER, ETC.	• EXCESSIVE DATA BUS AND WIRES • DISSIMILAR OPERATIONAL CALCULATIONS DIFFERENT WORD LENGTHS, ITERATION RATES, SOFTWARE, ETC.
PHYSICAL LOCATION COMMONALITY – EQUIPMENT LOCATION DETERMINES COMPUTER ASSIGNMENT	• EQUIPMENT LOCATION IMPACTS DATA TRANSFER TASK AND CONSEQUENTLY PROCESSING
HYBRID APPROACH – BOTH COMMONALITY OF CALCULATION AND LOCATION – SENSOR ORIENTED SPECIAL PURPOSE COMPUTERS (SPC) WITH MISSION ORIENTED GENERAL PURPOSE CENTRAL COMPUTER COMPLEX (CCC) SELECTED	• UNIQUE HIGH RATE AND TYPE COMPUTATION FOR SENSORS PERFORMED BETTER BY SPC WITHOUT UNDULY COMPLICATING THE CCC. • SENSOR ORIENTED COMPUTATIONAL CHANGES (HARDWARE AND SOFTWARE) WILL NOT IMPACT THE CCC. • MISSION FLEXIBILITY PROVIDED BY SOFTWARE CHANGES IN THE CCC. • REMOTE SYSTEM WITH HIGH DATA REQUIREMENTS, (e.g. PROPULSION) JUSTIFIES SEPARATE PERIPHERAL PROCESSOR

Table 4-13

DMS INTERFACE IMPLEMENTATION

CANDIDATE APPROACHES	RATIONALE

• MULTIPLEXED DATA BUS (selected)
• NONMULTIPLEXED HARD WIRE
	• IMPLEMENTED WITH PARTY LINE OPERATION AND STANDARD DIGITAL INTERFACE CIRCUITRY
	  • REDUCES WIRING
	  • SIMPLIFIES INTERFACE

• MULTIPLEX MODULATION TECHNIQUES
  • ANALOG FREQUENCY DIVISION
  • ANALOG TIME DIVISION
  • DIGITAL TIME DIVISION (selected)
	• EFFICIENT TECHNIQUE FOR LARGE NUMBER OF LOW FREQUENCY SIGNALS
	• SIMPLE DIGITAL CIRCUITRY
	• INHERENTLY NOISE-IMMUNE

• TRANSMISSION LINE
  • COAXIAL CABLE
  • TWISTED PAIR SHIELDED CABLE (selected)
  • FIBER OPTIC BUNDLES
	• HIGH NOISE IMMUNITY
	• ALLOWS BALANCED DRIVE
	• LOW WEIGHT
	• GOOD HANDLING CHARACTERISTICS

• COUPLING METHODS
  • AC (selected)
  • DC
  • ELECTRO-OPTICAL
	• LOW AND HIGH FREQUENCY NOISE REJECTION
	• PROVIDES DC ISOLATION

• CODING METHODS
  • RZ
  • NRZ
  • BIPHASE (selected)
  • DIPHASE
  • ETC.
	• COMPATIBLE WITH OTHER SYSTEM PARAMETERS (e.g. AC COUPLING)
	• WIDELY USED TECHNIQUES AND CIRCUITS AVAILABLE

(ARROWS INDICATE SELECTED METHOD)

extracted from the data itself or transmitted over a separate line. A separate clock line was selected because its weight and cost penalties are offset by the saving in separate clock generating equipment required if the timing is extracted from the data.

Conclusions and Recommendations - The data management system described is the result of conceptual studies consistent with a Phase A effort. The baseline system selected satisfies the data management requirements of the space shuttle.

In the course of this study several areas requiring further detailed in-depth investigation were uncovered. These study recommendations are described below.

- Computer Organization - The centralization versus decentralization aspect of the computational task must be further evaluated. The amount of data, data rates, equipment locations, and data flow traffic patterns must be identified. This impacts both hardware and software configurations.
- Computer Configuration - Existing and proposed computer systems including multiprocessors should be examined for applicability to the space shuttle. If the centralized versus decentralized study determines the need for multiple computers, then most probably different generic types of computers will be required.
- Digital Interface Techniques - Both multiplexed data bus and non-multiplexed interconnection techniques should be studied. Equipment location and density of data flow between equipment are important considerations in determining the feasibility of multiplexing. Signals which may be multiplexed and which may not be multiplexed must be identified.
- Multiplexing Implementation - Assuming there will be some degree of multiplexing on the space shuttle the following parameters must be studied.
  - Modulation techniques
  - Coding/Decoding schemes
  - Word and message formats
  - Transmission lines
  - Signal coding and wave shapes
  - Coupling methods
  - EMI considerations

4.3.3 Self-Test and Warning

Summary - In past spacecraft programs, significant expense has been associated with prelaunch test complexes and associated operations support personnel. Significant time has been required for the planned series of prelaunch test activities. For the ILRVS vehicle, the objective is to accomplish this preflight testing on board the vehicle in order to reduce cost and minimize test time, which is especially important for a reusable vehicle. The on-board checkout approach and associated maintenance philosophy will be patterned after the approach followed for airliners and military aircraft. Some degree of on-board checkout is required in all aircraft and spacecraft to permit evaluation of vehicle performance during flight. Post-flight maintenance activity can be expedited and simplified by making the in-flight on-board checkout capability sufficiently thorough for fault isolation to line replaceable units. The prevailing philosophy for advanced military aircraft is to provide a comprehensive on-board checkout capability which is equally thorough for preflight testing, in-flight performance assessment, and in-flight testing for the purpose of expediting post-flight maintenance. The concept to be followed in the ILRVS vehicle will benefit from this previous spacecraft and aircraft experience. Two fundamentally different approaches to on-board automatic checkout have been utilized on military aircraft. In one approach, each subsystem incorporates the ability to perform a self-test. In the other approach, a central unit requests and obtains data from all subsystems and compares this data with established criteria in order to evaluate system performance. Varying degrees of combination of these two approaches are possible. For example, the inherent presence of certain stimuli within a given subsystem would make it undesirable to generate duplicate stimuli externally, even if a central unit was used for data acquisition and comparison. In some cases, only minor system additions are necessary to provide meaningful built-in self-test capability. It seems likely that an optimum system will utilize a large degree of built-in test capability in individual systems, but will also utilize some degree of centralization, at least for assembling, recording, and displaying test results.

Functional Requirements and Goals - On-board checkout is a group of status checks and tests which are conducted to assure operational readiness of the various subsystems of the vehicle without ground facility support. In this context, on-board checkout does not imply a subsystem specifically incorporated to perform the checkout function, since a limited amount of operational readiness data will inherently be displayed or built into the various subsystems.

The choice of the system to be used for on-board checkout is dependent on many factors, but the general goals to be met can be summarized as follows:

- o Provide crew controlled prelaunch and launch capability.
- o Provide rapid turnaround capability.
- o Improve probability of mission success.

The goals are most readily achieved by an on-board checkout system. A well designed on-board checkout system should consider the following desired characteristics:

- o Automatic continuous monitor.
- o Capability for crew initiation of supplemental tests.
- o All failure data available for crew display.
- o Provisions for permanent record of malfunctions.
- o Capability for monitoring trend data in appropriate cases.
- o Monitor all vehicle subsystems.
- o Essentially all preflight test capability available during flight.
- o Provisions incorporated for recognizing test system malfunctions.

System Concept - An evaluation of onboard checkout techniques between the use of a centralized system versus distributed built-in test (BIT) indicates the desirability of using self-contained built-in test circuitry in order to:

- o Minimize Interface Complexity
- o Provide Subsystem Autonomy
- o More easily fault isolate to a line replaceable unit.

The BIT system configuration is shown in Figure 4-67. The BIT control panel located in the pilot's compartment presents an indication of a faulty system by lighting the appropriate BIT control button and displaying on the status cathode ray tube (CRT), faulty equipment designation. For more detailed diagnostic data, the pilot presses the illuminated button to initiate a detailed diagnostic or fault isolation test within the faulty subsystem. The test results are fed to the central computer via multiplexed data line to be formated and accessed to the display system. This provides the crew detailed status analysis and allows an in flight decision how

BUILT IN TEST SYSTEM CONCEPT

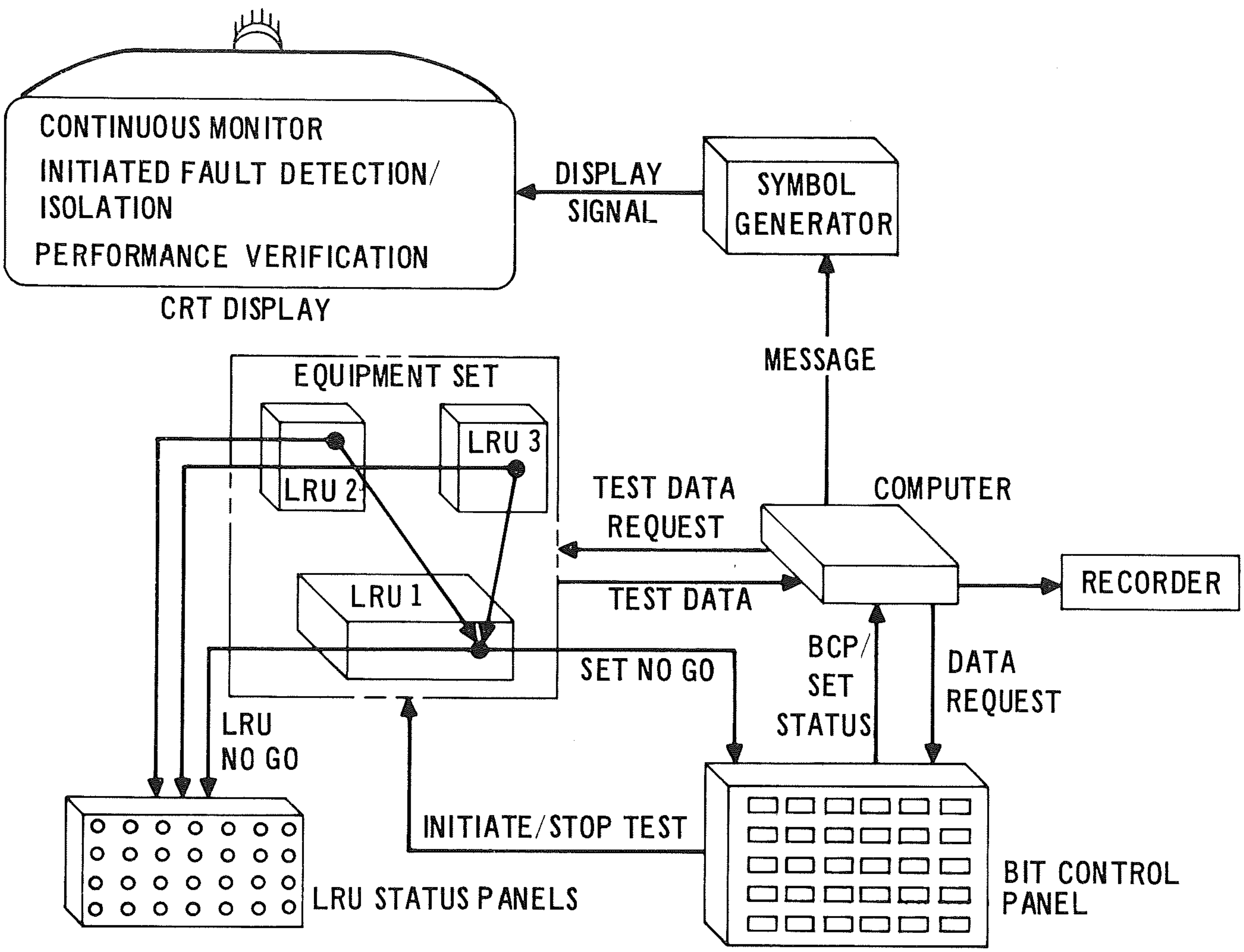


Figure 4-67

best to proceed, whether to continue with a degraded mode capability or switch to a redundant system.

To expedite the ground maintenance there is included a line replaceable unit (LRU) status panel which identifies the compartment in which the faulty LRU is located. Each LRU has its own latching indicator to identify the failed LRU. In addition, LRU diagnostic data is stored in an in-flight trend recorder to expedite repair.

Special features of the built-in test system are the following:

- The greatest practical amount of fault detection and fault isolation will be performed in flight; therefore aircraft mean time to return to service and maintenance costs are significantly and effectively reduced.
- BIT controls and displays consist of a control panel of switch lights and use of a status display CRT.
- Performance degradation is displayed to the pilot on the status CRT.
- BIT operation is part continuous and part initiated to reduce pilot tasks.
- BIT display messages have a significant impact on computer memory requirements. The selected approach minimizes memory requirements.
- The BIT interface is a hardwire and multiplex combination which has minimum weight, good maintainability, and maximum independence from the central computer complex (CCC).

Built-in Test Implementation - The ILRVS features three levels of self-test:

- Continuous monitor
- Initiated fault detection/isolation
- Diagnostic performance verification

All three levels can be employed in flight by the crew or on the ground by launch operations maintenance personnel. This design enables the flight crew to ignore detected faults in nonvital units (e.g., Instrument Landing System (ILS), antiskid, etc.) if he chooses, or to initiate further testing to determine the extent of failure in vital units such as radar, or the Inertial Navigation Set. The capability to initiate fault detection builds pilot confidence that essential units will operate during a critical phase such as entry.

To the greatest extent practical, all avionics are designed so that functionally associated components are contained within the same LRU. This feature vastly simplifies the BIT required for isolation to a faulty LRU.

Continuous Monitor Test - The continuous monitor BIT mode operates totally independent of operator or CCC control. On a continuous or periodic basis, test circuitry within each LRU monitors voltages, currents, impedance, voltage standing wave ratio etc., to determine that measured values are within preset tolerances. Faults are indicated on a cockpit BIT control panel. Since functional circuits are contained within a single LRU (for the majority of LRU's), detection of an out of tolerance condition also isolates the fault to the corresponding LRU. Independence from CCC control provides a test capability regardless of the CCC status; whether operating, inoperative, or removed from the vehicle. Depending on the complexity of specific units, continuous monitor fault defection/isolation capability will provide greater than 80 percent fault detection.

Initiated Fault Detection/Isolation Test - The initiated fault detection/isolation test increases pilot confidence that a set is functioning properly, or determines what functional capability has been lost in failed sets. The test may be initiated with a cockpit BIT control at any time, either in flight or on the ground. The CCC is required to be operating only if test results are desired to be displayed to the operator on the status display (latching fault isolation is made independent of the CCC). The fault detection/isolation capability is increased in this test mode to an average of 98 percent of all faults.

Diagnostic Performance Verification Test - The diagnostic test provides a virtually complete quantitative evaluation of the performance capability of the ILRVS individual sets, and provides fault isolation to a faulty LRU for 98 percent of all failures. In contrast to the continuous monitor and initiated fault detection/isolation tests, the diagnostic test utilizes the pilot or maintenance technician to exercise all modes of operation of the set, and is not limited to mode-in-use testing.

BIT Mechanization - BIT is implemented in three ways: (a) BIT controls and displays, (b) functional test circuitry within the LRU's of each set, and (c) software within the CCC. Human engineering principles have been employed to provide easily controlled testing, rapidly comprehended displays, and clearly indicated maintenance actions.

BIT Controls and Displays - BIT controls and displays are made up of three units whose sole function is BIT oriented, three display units functionally shared with other electronics operations and a trend data recorder. A cockpit installed built-in-test control panel displays the go/no-go status of each electronic equipment set in the orbiter, (both avionic and nonavionic), and controls start/stop of all initiated tests, either in flight or on the ground. One status panel installed in the equipment compartment provides a magnetically latching fault indication to indicate compartment location for each of about 100 LRU's which have self-test capability.

The display units shared with other functions are the master caution lights, used to indicate that a fault has been detected in essential sets; the warning/caution panel, used to display safety of flight faults; and the equipment status display, used to display avionic set no-go, functional capability loss, and diagnostic test operator instruction readout and fault isolation data display. Audible alarms are also generated for safety of flight faults and emergency conditions to immediately alert the crew to these conditions.

BIT Control Panel - The BIT control panel consists of lighted, alternate action, pushbutton switches which serve a dual function. When illuminated, the lighted portions of the switches serve as set failure indicators. Also the switches can be activated by an operator to alternately start and stop initiated fault detection/isolation or diagnostic testing. By means of a multiplex terminal, the BIT control panel is able to communicate digitally with the CCC. The CCC requests data from the BIT control panel on the test status of each set. When a set diagnostic test is desired, the test initiate signal from the BIT control panel is inhibited by the computer until the bulk storage tape is correctly positioned for the selected test.

Status Panels - One centrally located status panel provides a latching indication of failed LRU's compartment location for post-flight launch operation maintenance action. The indicators are activated by either a continuous or pulsed 28 VDC signal and are in parallel with the individual indicators mounted on each LRU containing BIT. The latching indicators are manually resettable after a faulty LRU has been replaced.

Trend Recorder - All BIT meaningful data is also routed to the trend data recorder for later evaluation and use by the launch operations maintenance crew. The data will enable flight analysis of all faults, failure prediction on the returned spacecraft, and contribute significantly to reducing failures on future flights.

BIT - Shared Cockpit Displays - Cockpit displays which share BIT with other display functions such as pilot alert or advisory displays are the following: (a) master caution lights, (b) warning/caution lights panel, and (c) equipment status display.

The master caution lights alert the pilot to vital equipment failure, and direct his attention to the warning/caution lights panel (safety of flight conditions) and the status display (all equipment failures).

The warning/caution lights panel, provides a failure indication for flight safety function such as the flight control system.

The equipment status display is used in all BIT test to advise of set failures by displaying a three or four-character alphanumeric mnemonic set name. When an initiated test is selected for a particular set, the word "TEST" also appears on the status display until the results of the test are decoded by the CCC, when any detected failures are displayed as three word messages describing the lost function. A second press of the set pushbutton stops the test, and erases the data written on the status display.

BIT Functional Circuit Integration - Figure 4-68 illustrates the application of BIT to an individual functional circuit. A typical functional circuit, the associated BIT circuit and corresponding BIT self-test (BST) circuit are interconnected as shown. "BIT" on a signal line indicates the built-in-test circuit has detected a functional circuit fault; "BST" denotes a BIT circuit failure. Either a "BIT" or a "BST" (logically denoted BIT + BST) causes a LRU fault to be indicated. However, a "BIT" without the "BST" (denoted BIT o BST) inhibits the digital data word validity bit, meaning the data is not valid.

<u>Central Computer Complex BIT Software</u> - The Central Computer Complex performs the following BIT functions:

- Continuous Monitor - The CCC continuously monitors the BIT control panel individual set lights (on/off) and set switches (on/off) in a predetermined

FUNCTIONAL CIRCUIT BIT INTEGRATION

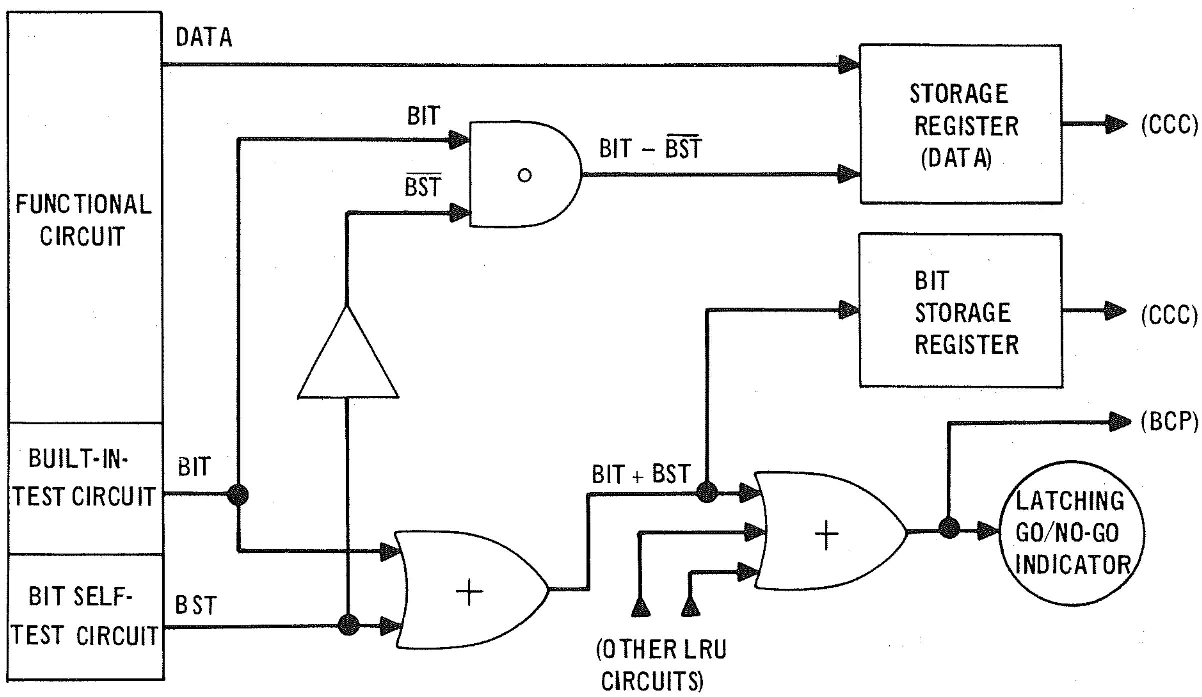


Figure 4-68

sequence to determine the status of all the sets. The results of this routine are evaluated by the BIT Module of the CCC and displayed by writing any failed set name(s) in an alphanumeric format on the status display.

- Initiated Fault Detection/Isolation - On command from the BIT control panel, the designated set initiates or stops self-contained fault detection/isolation testing. The CCC generated alphanumeric display messages for the status display are based on the BIT control panel status as evaluated by the BIT Module, set lights on/off, set switches on/off, and the individual LRU functional BIT data words as evaluated by the CCC BIT Data Module.
- Diagnostic Testing - On command from the BIT control panel the CCC initiates or stops set performance verification testing. When a diagnostic test is initiated, the CCC determines that bulk storage is interconnected and inhibits the particular LRU BIT circuit test until the BIT Monitor function reads diagnostic program data into the CCC. During this testing the LRU data bits are compared directly with the CCC by the BIT Data Module. This testing provides up to 98 percent fault detection, isolation and degraded performance information, as well as special alphanumeric displays, to indicate manual actions required and the results of the diagnostic tests.

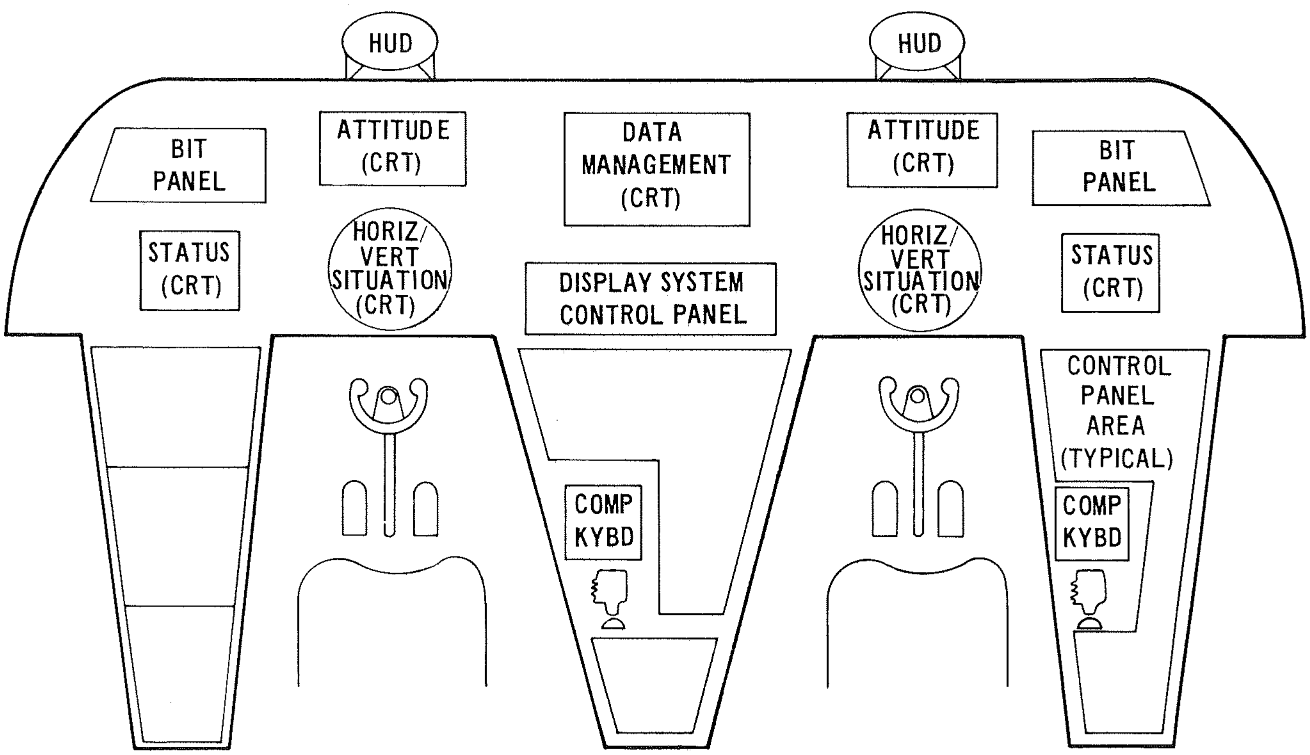
CCC BIT Sequencing and Control - The software program checks the test condition of each equipment set to determine present status. When test results are available the set name and status are displayed on the status display. Any messages that cannot be immediately used are sent to the deferred display table. The software routine also continually checks the deferred display table for any deferred messages that could be displayed during a new display period. Other functions of the program are to erase previously displayed messages when new ones are written and to determine if bulk storage data is available so that radar diagnostic testing can be done in place of the fault detection/isolation testing.

BIT Display Formatting - The BIT information is displayed in an alphanumeric format consisting of 15 characters per line. The display words are limited to four characters each, and describe functions such as set name, test and failed function. Messages are displayed starting with the bottom and continuing upward until the available space is occupied. Each message occupies only one line per set. When the available space is filled, new messages are written, again starting with the bottom line. However, previous messages indicating that a set is still in test are skipped over and not erased. When, on occasion, all lines are skipped during a display period, the new message is placed into the deferred display table for later display. When a message contains information involving a sequence of lost modes, the modes will be displayed and erased in sequence until the last mode is displayed and retained.

Installation - The BIT installation is subject to two constraints: (1) Separation between the status panel and the monitored units must be minimized for lowest practical weight penalty of the interconnecting wires; (2) the displays must be installed in an arrangement such that rapid cueing of status is provided to the pilot. An optimum separation between the status panel and the majority of the electronics has been provided by installing the status panel in the avionic equipment bay surrounded by the avionics units. This installation also provides quick access for the launch operations maintenance crew to view the status panel for LRU failure indications. The requirement for rapid pilot cueing has been satisfied by the philosophy shown in Figure 4-69. Failure of vital equipment is indicated by the master caution lights located in the pilot's central vision; the pilot responds by looking to the equipment status display for the name of the failed set.

Conclusion and Recommendation - The ILRVS on-board checkout system implementation is within the present day technology. Detailed studies are required to fulfill the operational objectives of the ILRVS Program. Effort should be expended in identification of the parameters required for determining a flightworthy subsystem, with special emphasis devoted to non-avionic subsystems.

MULTI-MODE DISPLAYS



- CRT AND CRT WITH SUPERIMPOSED SLIDE CAPABILITY
- HEAD-UP DISPLAY (HUD) FOR STATION APPROACH AND LANDING
- OVERHEAD PANEL AREA USED ALSO, BUT NOT SHOWN

Figure 4–69

4.3.4 Displays and Controls

Summary - The displays and controls for the space shuttle utilize state-of-the-art devices and techniques to provide a flexible display of multi-mode data with an acceptable work load for the crewmen. The space shuttle vehicles are both an aircraft and spacecraft, designed for autonomous mission operation. This, in conjunction with On-Board Checkout and redundant systems, results in a significant amount of mission data that must be compatible with the crewman capability. A high degree of display automation is required to provide an acceptable crew task work load and timeline. Integrated electronic multi-mode displays are required to present all the different flight regimes data in a limited cockpit area and pilot viewing cone. In addition, the data will be segregated according to function.

The required display information compression is provided by the use of multi-mode cathode ray tube (CRT) devices. These programmable devices allow the display of only that data pertinent to the present mission phase; all other data is relegated to the status monitor or caution/warning classification.

Cluttering of control devices is partially eliminated by mounting the jet aircraft engine throttles and rocket engines ΔV translational control stick on the pedestal between the two crewmen. At present the usual transport aircraft control yoke and rudder pedals is provided for aircraft flight control and a right-hand hand controller for space attitude control. It is hoped that present flight test programs on aircraft control with a hand controller will allow the future deletion of the bulky control yoke and rudder pedals.

Both control and display techniques and hardware for the space shuttle are being studied and evaluated in an in-house cockpit simulator. This continuing effort will be very instrumental in the design evolution of an optimum cockpit system, both in hardware selection and crewmen work load compatibility.

Requirements - The primary crew control and display system design guidelines and desirable features are summarized by:

a) Allowance for autonomous launch, orbital, reentry, and landing mission operations without crew task overload.
b) Provisions for two crewmen but flyable by a single crewman.
c) Maximum utilization of integrated electronic displays and controls over single purpose gauges and meters and toggle switches.

The inclusion of automated, multi-mode displays requires a continuing evaluation of control and display techniques and hardware features in a cockpit simulator. This experimental approach with empirical crewman performance evaluation is being used and will be continued to constantly refine the control and display system design.

Baseline Description

Displays - The displays provide the crewman information to monitor system/vehicle operation or status, assess control performance, and determine proper control actions. The basic mission operational data provided for each crewman includes vehicle attitude reference, horizontal or vertical situation, operational data from on-board systems, and status monitor of on-board systems. The display system functional block diagram of Figure 4-70 shows how these data are presented to each crewman by direct view of four CRT's and a head-up display (HUD). Three of the four direct view CRT's are "rear port" tubes which can optically project slide or film (microviewer) images in addition to the normal electron beam written image. These easily accommodate large quantities of diagrams or checkout procedure data, too voluminous for digital memory storage. The electronic attitude director indicator (EADI) CRT replaces the conventional electromechanical 8-ball attitude director indicator and airspeed, vertical sink speed, altitude needle gauges. One head-up display (CRT/optical) is provided for each crewman (2) to allow flight director symbology to be written upon the outside viewing reference to aid in space station or satellite docking and all weather landing approach.

All data received by the display system is routed through a standard interface. Here priority of display is established and the data is sorted to channel the display data storage symbology to the proper CRT. The display data storage provides the required high rate CRT image rewrite to eliminate flicker with a low input data rate to the system. The display system mode control is accomplished automatically through the self-contained autoprogrammer. A manual override capability is provided in case of mission change or equipment failure; for example, the crewman can switch a symbol generator to a different CRT via command to the CRT selector.

Table 4-14 summarizes the rationale used in selecting the baseline multi-mode display system design techniques from a field of candidate approaches based on the requirements and desirable features. Figure 4-71 depicts how these functional

DISPLAY SYSTEM CONCEPT

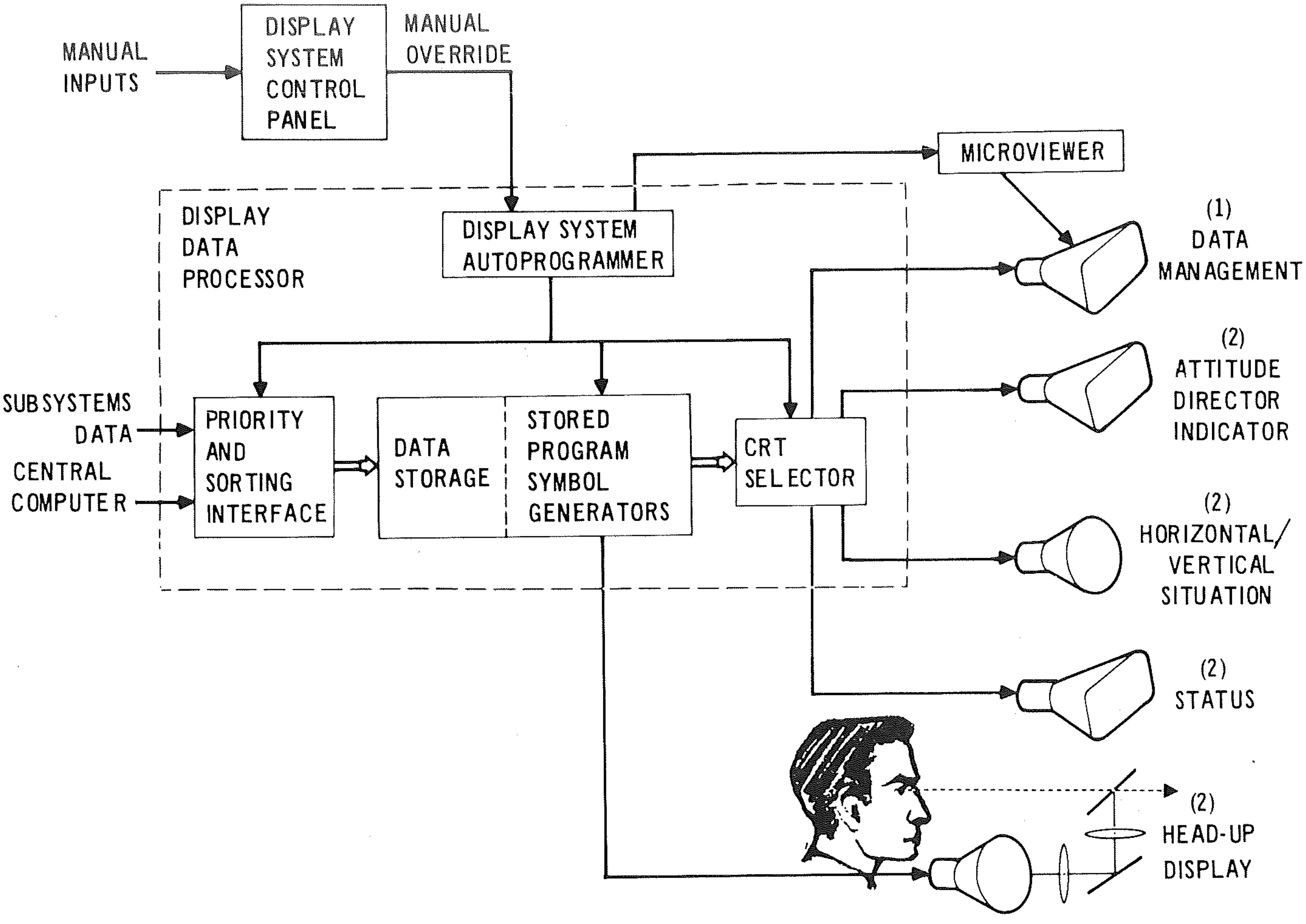


Figure 4-70

Table 4–14

MULTIMODE DISPLAYS STUDY SUMMARY

REQUIREMENTS/DESIRABLE FEATURES	CANDIDATE/BASELINE APPROACHES	BASELINE RATIONALE
• DISPLAY – C/O PROCEDURE AND DATA – CONTINGENCY MISSION PLANS – GUIDANCE/NAVIGATION DATA – HORIZONTAL/VERTICAL/ATTITUDE SITUATION DATA – STATUS, CAUTION & WARNING	• CATHODE RAY TUBE (CRT) • SLIDE/FILM PROJECTORS • CRT WITH SUPERIMPOSED SLIDE CAPABILITY • AUDIO, LIGHTS • PLASMA TUBE DISPLAY DEVICES • SCALE SCRIBES, DIALS, GAUGES	• CRT AND CRT WITH SUPER-IMPOSED SLIDE CAPABILITY – PROVIDES MULTIFORMAT DATA DISPLAY – ELIMINATES EXTRA SLIDE SCREEN, ATTITUDE 8-BALL, & SEPARATE GAUGES – SIMPLIFIES REDUNDANCY • FLASHING LIGHT/HEADSET AUDIO FOR CAUTION AND WARNING
• HEAD-UP (OUTSIDE) PROJECTION DISPLAY FOR DOCKING AND LANDING AID	• CRT/REFLECTIVE OR REFRACTIVE OPTICS • ELECTROMECHANICAL/OPTICAL	• BEST PHYSICAL CHARACTERISTICS AND RELIABILITY AND DESIGN EXPERIENCE
• SIMPLE SIGNAL INTERFACES	• ALL SOURCES INPUT DATA TO SINGLE DISPLAY SYSTEM SORTING/PRIORITY INTERFACE UNIT • MULTIPLE INTERFACES (G&C, ELECT PWR, PROP, ETC.) WITH DEDICATED DISPLAY DEVICES	• SIMPLIFIES DISPLAY MODE CONTROL, STANDARDIZES INTERFACE CIRCUITRY TO COMMON DISPLAY DEVICES, ELIMINATES MANY DEDICATED DISPLAY DEVICES
• CRT BEAM DEFLECTION/BLANKING COMMAND RATE HIGH ENOUGH (50–60 Hz) TO PREVENT FLICKER	• SAMPLE INPUT SOURCE DATA AT LOW RATE (1 Hz) AND DISPLAY SYSTEM MEMORY USED FOR 50–60 Hz CRT REFRESH • SAMPLE INPUT SOURCE DATA AT 50–60 Hz FOR CRT REFRESH	• BEST DESIGN EXPERIENCE • MINIMIZES REDUNDANT DATA GATHERING FROM SOURCE
• REDUNDANCY	• HARDWARE REDUNDANCY • DEGRADED MODE OPERATION	• SYMBOL GENERATOR TO TUBE CONNECT – SELECTABLE • MICROVIEWER CAPABILITY REDUNDANT

NOTE: BASELINE APPROACH UNDERLINED

DISPLAY ARRANGEMENTS

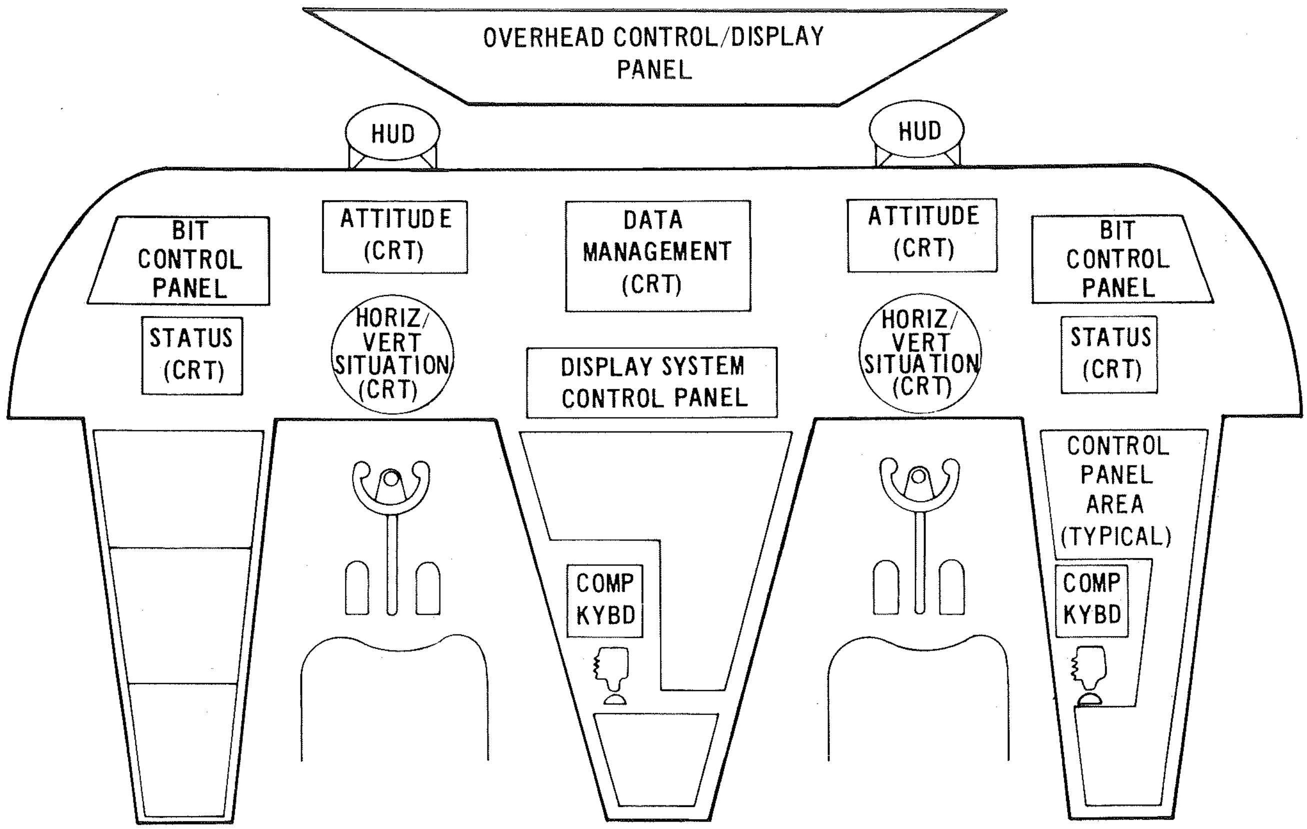


Figure 4-71

display devices might be integrated into the space shuttle cockpit. Note that the single data management CRT called out is shared by the two crewmen. The overhead area of the cockpit shown will be used for some of those displays requiring infrequent viewing.

All direct view CRT displays will contain contrast enhancement design features such as:

a) Built-in tube faceplate black layers, and/or
b) Tube faceplate attached filters (i.e., micromesh, neutral density, polaroid), and/or
c) Built-in panel photometer detectors with feedback beam current intensity control.

All these features are considered for enhancing pilot viewability during high ambient lighting phases of the mission.

Controls - The controls are those devices which provide for crew control of the subsystem's operational set-up and override and control of the vehicles 6 degrees-of-freedom. These are basically categorized as attitude and velocity control, central computer access, and subsystems selection or mode control.

The baseline cockpit functional layout of Figure 4-71 shows the conventional control yoke/rudder pedals system for aircraft attitude control and the hand controller for spacecraft regime attitude control. To be highlighted, is the potential removal of the aircraft systems control yoke and rudder pedals depending on flight test results of aircraft flight control by a hand controller. This would be one step in the elimination of controls clutter. The final decision will be based on the results of present and on-going flight tests on the McDonnell Douglas F-4 aircraft and the Cornell University variable stability aircraft. Center console (pedestal) mounting of the velocity control devices, aircraft jet engine throttles and ΔV translational rocket control stick, would allow the crewmen to share these devices and thus further reduce device clutter and eliminate duplication.

Each crewman is provided access to the on-board central computer via a computer keyboard. This allows data insertion for mission parameter update, subsystem commands via computer control, or control of data recording via the on-board printer for post-flight maintenance and quick turnaround.

Subsystem selection and mode control is provided primarily through several control panels containing a mixture of push buttons, thumb wheels, and twist knobs. Crewman programming of such control actions via the computer keyboard must be limited because rapid response is many times required and crewman memorization of control action codes should be minimal. Push button switches (mono and multi-function) will be used in the subsystem control panel areas to minimize toggle switches and levers used in the past. For example, landing gears extension and retraction can be push button initiated to eliminate bulky levers. Several thumb wheels and twist knobs will still be incorporated for such functions as communication channel select or manual slew of antennas or TV cameras. This allows inclusion of small devices with past pilot familiarity without unwieldy panel size. These single purpose devices can be subsystem grouped for quickness of location recognition. In many cases these devices can be shared between crewmen by center console (pedestal) mounting or overhead panel mounting.

Table 4-15 summarizes the above discussion by presenting the rationale used in selecting the baseline control devices from a field of candidate approaches, based on the requirements and desirable features.

Table 4-15

CONTROLS STUDY SUMMARY

REQUIREMENTS/DESIRABLE FEATURES	CANDIDATE/BASELINE APPROACHES	BASELINE RATIONALE
• ATTITUDE CONTROL	• BETWEEN-THE-LEGS CONTROL YOKE WITH RUDDER PEDALS FOR AIRCRAFT SYSTEMS CONTROL • FLY-BY-WIRE HAND CONTROLLER FOR SPACECRAFT SYSTEMS CONTROL • CONTROL YOKE/RUDDER PEDALS WITH SWITCHABLE OUTPUTS TO EITHER SYSTEM • HAND CONTROLLER WITH SWITCHABLE OUTPUTS TO EITHER SYSTEM	• PREVIOUS PILOT/ASTRONAUT EXPERIENCE • POTENTIAL CHANGE TO USE OF HAND CONTROLLER WITH SWITCHABLE OUTPUTS, BASED ON MDC F-4 AND CORNELL UNIV VARIABLE STABILITY AIRCRAFT FLY-BY-WIRE TEST PROGRAMS
• COMPUTER ACCESS	• ALPHANUMERIC KEYBOARD • TAPE, CARDS, ETC	• BEST FLIGHT EXPERIENCE, FLEXIBILITY, AND RELIABILITY
• CRT DISPLAY MODE CONTROL	• DISPLAY SYSTEM AUTOPROGRAMMER WITH OVERRIDE CAPABILITY • AUTOMATIC COMPUTER SELECT OF MODE • MANUAL ACCESS (KEYBOARD) TO COMPUTER TO SELECT MODE • MANUALLY SELECT MODE	• SIMPLIFIES PILOT TASK BUT LEAVES HIM AS MANAGER OF DISPLAY SYSTEM
• OTHER (I.E., CHECKOUT TEST OVERRIDE, SELECT COMMUNICATION CHANNEL, MANUAL SLEW OF ANTENNA OR TV CAMERA, ETC.)	• PUSHBUTTONS – MONO AND MULTIFUNCTION – COLOR CODED – OPERATION LOCK-OUT BY COMPUTER • THUMBWHEELS • TWISTKNOBS • COMBINATION OF ABOVE • PILOT PROGRAM THROUGH COMPUTER KEYBOARD	• PREVIOUS PILOT/ASTRONAUT EXPERIENCE • COMPUTER KEYBOARD PROGRAMMING REQUIRES EXCESSIVE CODE MEMORIZATION BY PILOT

NOTE: BASELINE APPROACH UNDERLINED

Alternate Concept Evaluations - Alternate control/display techniques and hardware are being studied and evaluated for both hardware simplicity and pilot acceptance in an in-house simulator. This simulator is presently using CRT's integrated into an existing airplane cockpit mockup. Figure 4-72 shows a schematic of the space shuttle (modified VSX airplane) control/display simulator to test variable approaches in all mission phases. Table 4-16 summarizes the possible uses for this simulator leading to the optimum cockpit design.

Technology Status and Recommendations

Control Devices - All necessary control type devices are in a satisfactory state of development. Special studies are in process to evaluate the practicality and reliability of such design approaches as replacing landing gear extension and retraction levers with push button controls. The push button technology is in an advanced state of development to include even non-contact switches employing the magnetic, hall effect, etc. principles.

CRT Development Status - Cathode ray tube displays are presently flying in the A-6A and F-111 aircraft. Their usage has also been proven acceptable for other planned aircraft such as the F-14, F-15, DC-10, and SST. The "rear port" CRT tubes as cited herein is available from several sources to include Westinghouse, Sylvania, Dumont (Division of Fairchild Camera and Instruments), and Raytheon. Conrac Corporation has also qualified a CRT display system to NASA space qualification standards. This is the dual CRT display devices to be flown on the Apollo Applications Apollo Telescope Mount (ATM). This program application together with present aircraft usage indicates no developmental problems for a wide environmental spectrum.

CRT Display Physical Characteristics - The size, weight, and power of most reviewed CRT display systems to date indicate a lack of miniaturization design approaches, primarily in the symbol generator units. These digital logic and digital-to-analog converter units need further development to reduce printed circuit board size, utilize low power logic, and improve electronic packaging design.

CRT Viewability - The visibility of cockpit CRT's in high external ambient lighting conditions is degraded by light transmitted through the cockpit window and subsequent reflections onto and from the CRT faceplate. The visibility of the CRT is not dependent upon image brightness alone, but on a combination of brightness and contrast.

SCHEMATIC OF CONTROL & DISPLAY SIMULATION

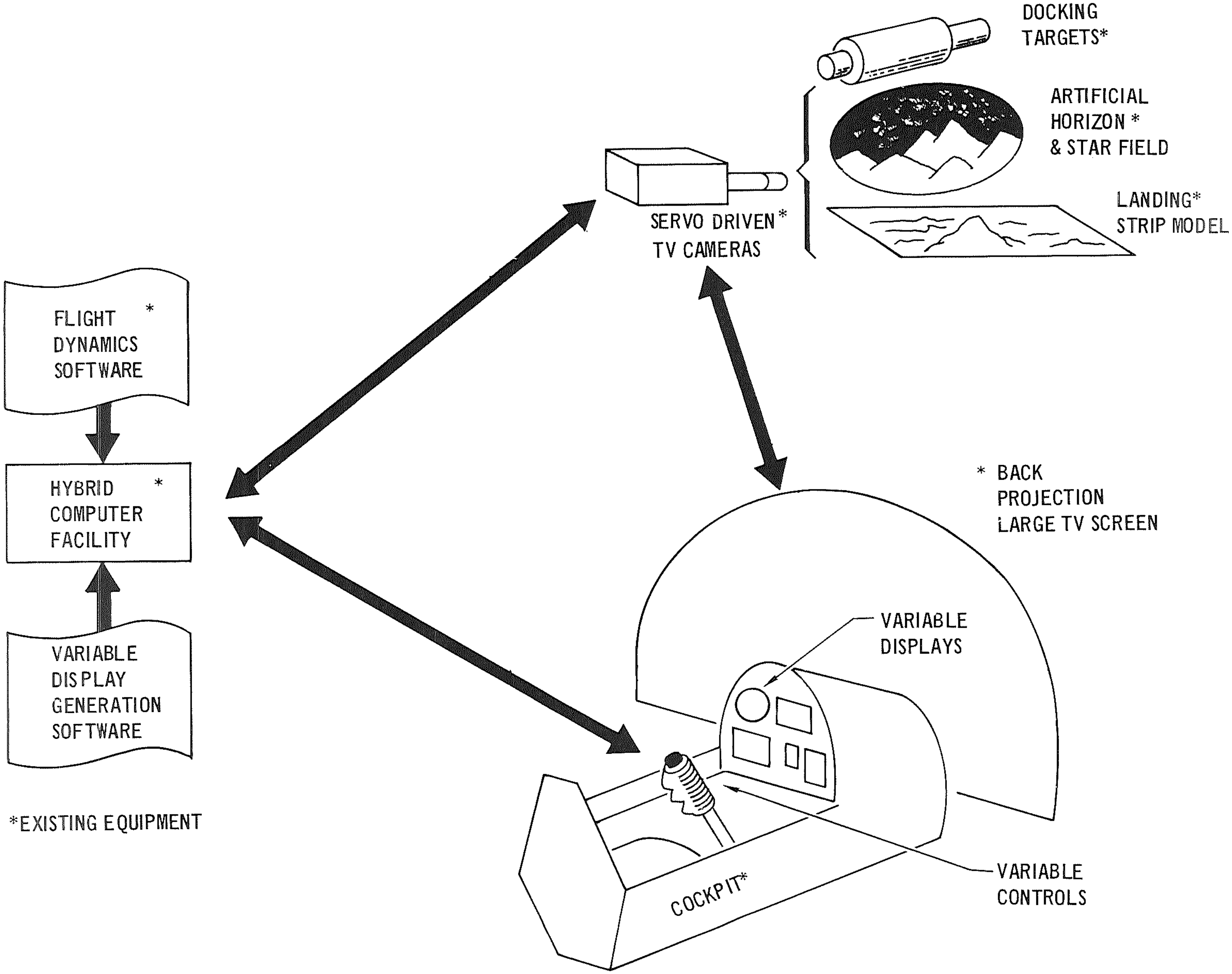


Figure 4-72

Table 4-16

POSSIBLE USES FOR SPACE SHUTTLE CONTROL AND DISPLAY SIMULATOR

1. IN GENERAL, REFINE CONTROL AND DISPLAY REQUIREMENTS THROUGH AN EXPERIMENTAL AND EMPIRICAL APPROACH
2. EVALUATE ACTUAL HARDWARE IN A REALISTIC CREW ENVIRONMENT
   - EQUIPMENT LAYOUT FEASIBILITY
   - VIEWING ANGLES, REACH TO TOUCH DISTANCES, TACTILE SENSE
   - AMBIENT LIGHTING CONDITIONS (VISUAL CONTRAST)
   - CRT REFRESH RATE TO ELIMINATE FLICKER
3. DETERMINE CRT DISPLAY REQUIREMENTS
   - SYMBOL SIZE, SHAPE, CLUTTER ELIMINATION
   - DIGITAL MEMORY CAPACITY, WORD LENGTH, BIT TRANSFER SPEED
4. EVALUATE ALTERNATE HARDWARE APPROACHES TO DISPLAY OF SAME DATA
   - SUBSYSTEMS DATA TO DISPLAY SYSTEM INTERFACE SIMPLICITY
   - SUBSYSTEMS DATA INTERROGATION RATE VS DISPLAY SYSTEM MEMORY CAPACITY FOR CRT IMAGE REFRESH RATE TO ELIMINATE IMAGE FLICKER
   - DISPLAY SYSTEM MODE CONTROL AND SWITCHING LOGIC
5. TEST FOR FEASIBILITY OF USING 3-AXIS HAND CONTROLLER FOR ALL FLIGHT REGIMES
6. DEVELOP CREW TASK TIMELINES

The use of CRT displays on the A-6A and F-111 aircraft, with wrap around cockpit windows, has been made possible by use of attachable filters (i.e., neutral density, polaroid, micromesh). These filter aided displays provide adequate image contrast even in the worst case ambient lighting conditions of 10,000 foot-lamberts at above 10,000 feet altitude. Electronic display devices with filters have been tested and found acceptable for viewability in the MDC design and cockpit simulator tests for the military F-14 and F-15 aircraft design competition programs and the commercial DC-10 aircraft.

Recent advances in increasing the tube image brightness from 200-500 foot-lamberts to 1500-2000 foot-lamberts has enhanced image viewability but has proven inadequate for all lighting conditions. The most interesting high contrast CRT developments in recent years have been the "optical diode filter" and "dark layer filter". These filters differ in they are actual material deposition on the CRT faceplate (i.e. layer denotation) and structurally carry the normal CRT phosphor. These tubes have been tested and shown viewable under direct impinging sunlight. The "dark layer filter" tube has been developed primarily by Hughes Aircraft and by a combined effort of Sigmatron Inc./Electro Vision Industries. The Hughes Aircraft Company actually modified existing Sony television tubes (Sony 140 CB4). The "optical diode filter" tubes were developed and tested primarily by Hartman Systems Company under NASA Electronic Research Center contract. This tube's image even under direct outdoor sunlight, is distinct and clear with high contrast.

MDC recommends the use of panel mounted photometers with feed back into the beam intensity control circuitry to automatically vary image brightness under varying lighting conditions. Kaiser Corporation includes this design feature in addition to normal manual override on their F-111 aircraft head-up display and EADI system.

4.3.5 Guidance Navigation and Control Requirements - The task of directing a space vehicle, to accomplish a given mission, is customarily discussed in terms of three functions: navigation, guidance, and control. As the boundaries between these functions are somewhat arbitrary, the terms, navigation, guidance and control, are used here in the following context:

- o Navigation is the determination of position and velocity of the vehicle from onboard measurements.
- o Guidance is the computation of maneuvers necessary to achieve the desired end conditions of a trajectory (e.g., an insertion into orbit).
- o Control is the execution of the maneuver (determined by the guidance command) by controlling the vehicle attitude and proper force producing elements.

Navigation, guidance and control requirements applicable to ILRVS include orbital insertion, rendezvous, station keeping, entry (includes cruise and landing to a pre-selected site) and the capability to ferry the booster and the orbiter between airports. In addition general requirements of particular significance to the G, N, & C design are: (1) autonomous operation during the ascent, orbital and entry phases of flight to minimize ground support and cost; (2) mission and growth flexibility, and (3) on-board checkout and failure detection. Table 4-17 shows their applicability as to carrier and/or orbiter. The basic requirement for navigation is similar for all mission phases. The accuracy of information and source of data, however, are dependent on the particular mission phase. The guidance and control requirements are highly dependent on mission phase or tasks to be performed. The equipment configuration for the selected G, N, & C system baseline is described in the following paragraphs.

Guidance Navigation and Control System Description - The baseline guidance, navigation and control configuration consists of the following:

- o A strapdown inertial measurement unit,
- o A dedicated inertial navigation computer,
- o A radar for rendezvous and station keeping,
- o An optical and IR tracker integrated into one gimballed head assembly,
- o Vortac and air data sensors as navigational aids,
- o A dedicated flight control computer with separate control element power amplifiers,

Table 4-17

GUIDANCE, NAVIGATION & CONTROL REQUIREMENTS

REQUIREMENT	APPLICABILITY	
	ORBITER	CARRIER
All azimuth launch capability	X	X
Information for termination by onboard system	X	X
Rendezvous and stationkeeping with passive or cooperative target	X	
One Axis Translation	X	X
Three Axis Attitude Control	X	X
Orbit Guidance and Navigation Functions Onboard	X	
Automatic Approach	X	
Return Guidance and Navigation Onboard	X	X
Manual landing complying with minimum FAA Requirements	X	X
Automatic, Zero-Zero weather landing	X	X

- An advanced all weather automatic landing system,
- An interface with the central management computer and the crew to provide guidance and mission oriented tasks.

During ascent, control steering signals are generated for the complete trajectory by the orbiter inertial navigation and guidance system. The carrier navigation system is active throughout its ascent phase and provides the basis for guidance during carrier entry and return to the landing site. During carrier cruise and return to the landing site, the air data sensors and Vortac provide data which can be used to enhance the long term accuracy of the inertial navigation system. The central management computer acts as an evaluator or filter to determine the best estimate of velocity and position from the various sources of navigational information. Carrier landing can be performed manually or automatically through use of the Advanced Instrument Landing System (AILS). If an abort were required, steering signal guidance command would be generated from the separate carrier and orbiter navigation systems in a manner similar to those used during a normal ascent.

Rendezvous and station keeping range and relative angular information is provided by a multimode radar. Range of the radar for passive targets is 30 miles. For cooperative transponding satellites, the range is increased to 400 miles. An alternate and backup capability is provided by the optical tracker. This backup capability includes all cooperative targets and sunlit uncooperative targets.

Attitude alignment and orbit ephemeris data is obtained from the optical and IR trackers. Accurate attitude information for inertial system alignment is obtained by tracking stars with the optical sensor. Earth edge tracking is provided by the IR sensor for navigational usage. The IR tracking head and the optical tracking head are integrated into a single gimballed assembly.

Retrograde attitude and time are determined by the central management computer. Energy management guidance during the entry phase is determined by the central management computer on the basis of navigational data provided by the inertial sensors. Attitude control is obtained by reaction jets, control surfaces or a blending of both. The cruise and landing phase is similar to the booster cruise and landing phase. In this phase, air data sensors and area navigational aids again are used to enhance the navigational accuracy. Landing can be either automatic or manual.

The carrier G, N, & C equipment is identical to the orbiter equipment except that those equipment required by the orbiter for the orbital phase are deleted.

System Evaluation and Trade-offs, Automatic Landing - A review of landing system was made to evaluate their applicability to the ILRV automatic landing requirements. Table 4-18 summarized the general characteristics of leading concepts applicable to the ILRVS needs. A description of these systems are contained below.

ILS (Instrument Landing System) is a term applied to an electronic system that is used at most large airports to provide a pilot with landing glide slope and runway centerline localizer signals. Many manufacturers supply the hardware for both the ground and airborne installations.

The ground glide path transmitter is located about 1000 feet down the rollout path from the start of the runway, and 400 feet to the side of the runway centerline. This system is generally applied to 10,000 foot runways and is used in conjunction with a localizer beacon (located 1000 feet behind the rollout end of the runway and on the runway centerline extension) and two "markers". The outer marker is located 4 miles from the start of the runway, and the middle marker is located 3500 feet from the start of the runway. (The inner marker at the start of the runway has been eliminated from recent systems.) The system transmits continuous (glide slope) information on the range of 329.3 to 335 MHz by modulating the transmission at 90 Hz and 150 Hz. The nominal glide slope is 2.5° to 3° and any deviation from the nominal slope causes the airborne equipment to receive either a 90 Hz or 150 Hz signal. This causes the airborne crosspointer display to show the deviation as a "fly-up" or "fly-down" error command or may be connected to an automatic control loop. Airborne acquisition of the ground transmitted guidance signal is 10 NM minimum for the localizer. Glideslope range is some 4-6 NM. The system has been in existence for many years, is well proven, and has seen many improvements and refinements, however the transmitted signal is subject to many errors. Since the system uses the 1 and 3 meter bands and Earth loaded antennas, the signal is topographically affected. The ILS at LaGuardia airport in New York is affected by the rise and fall of the tide. The hills surrounding the airport at Pittsburgh cause similar problems with ILS accuracy. Other aircraft (A/C) in the vicinity, particularly if they should cross the ILS beam, cause the received signal and its accuracy to degrade significantly. Additionally, due to the placement of the ground antenna, the transmitted signal is not readily usable below 100 to 200 feet.

Table 4-18

LANDING SYSTEM SURVEY

DESIGNATION	ACTUAL NAME	PRESENT USE	OPERATION	REMARKS
ILS	Instrument Landing System	Used at most commercial airports, some aircraft and facilities certified for Category II operation.	VHF Beam guides aircraft on approach from about 10 miles out. Can automatically land properly equiped aircraft. Uses localizer beam for roll out guidance. Performance is a function of beam quality and steering laws.	Useable for powered final approach and landing.
AILS	Advanced Instrument Landing System	In development flight test evaluated by FAA	Same as ILS except more accurate. Beam quality excellent. Ground display available.	Useable for powered final approach and landing.
AN/SPN-42	Automatic Control and Landing System	Capable of landing carrier based aircraft under zero-zero conditions, but lack of redundancy restricts bad weather operation to 200 ft. ceilings and 0.5 mile visibility. No flare, accommodates two aircraft simultaneously. 5 NM range capability. No roll out guidance.	Uses ship based precision tracking radar & guidance computer - up data link info supplied to aircraft.	Flare and roll out guidance need to be developed.

AILS refers to "Advanced Integrated Landing System". The System is built by Airborne Instrument Laboratories for the FAA. It is a new system which was at NAFEC in Atlantic City in February 1966 for evaluation. It is an evolutionary development from the former Flarescan equipment also built by Airborne Instrumentation Laboratories.

AILS automatically combines the features of ILS and ground control approach, providing guidance information through flare to TD of the A/C and providing a much improved Precision Approach Radar (PAR) function to the ground operator. The system combines two ground based antenna scanning arrays, one for elevation (glideslope), and the other for azimuth (localizer). The elevation antenna is located 1500 feet down the rollout path of the runway from the nominal touchdown point, has a beamwidth of 20° horizontal, and provides usable guidance to within 300 feet of its location. The localizer antenna has a beamwidth of 1/2° (half-power point) and gives $(\text{cosecant})^2$ coverage up to 10° with sharp cutoff on the bottom side. The localizer also serves as the transponder for the distance measuring equipment (DME) and is located at the rollout end of the runway. The system operates in the K_u-band (15.4 - 15.7 GHz) with circular polarization.

The localizer antenna oscillates at a very accurate rate of 5 Hz through a "torque-tube" arrangement which, like a tuning fork, oscillates at its natural frequency. Since two antennas are used and accurate synchronization is required, the elevation antenna "nodding" frequency is slaved to the azimuth antenna and is adjusted by a servo-driven mass to assure synchronization.

The elevation angle, localizer, and DME information are coded by the spacing between the two pulses making up a pulsed pair. The spacing between consecutive pairs of pulses is coded to give the glideslope angle or azimuth angle. For elevation guidance, a 40 microsecond pulse-pair spacing corresponds to zero degrees of glideslope (parallel to the ground). The pulse pair spacing increases by 8 microseconds per elevation degree, up to 10°, the maximum glideslope given. To assure airborne determination that the information is elevation guidance, the spacing between the pulses making up a pulse-pair is 12 microseconds.

For azimuth guidance, a 40 microsecond pulse-pair spacing corresponds to an azimuth location parallel to the runway centerline. The pulse-pair spacing increases by 8 microseconds per azimuth degree of deviation to the left or right of runway centerline, up to a maximum of $\pm 5°$, the maximum azimuth guidance given. To assure airborne unambiguous determination of the azimuth guidance information, a 14 microsecond spacing between the pulses of a pulse-pair corresponds to a fly-right command and 10 microseconds corresponds to fly-left. When DME information is transmitted, the spacing between the pulses of a pulse-pair is 8 microseconds.

Figure 4-73 depicts the azimuth and elevation antenna scanning, showing that only the central 10° of total travel is used for transmissions. This central 10° is the linear portion of the antenna total travel of 22°.

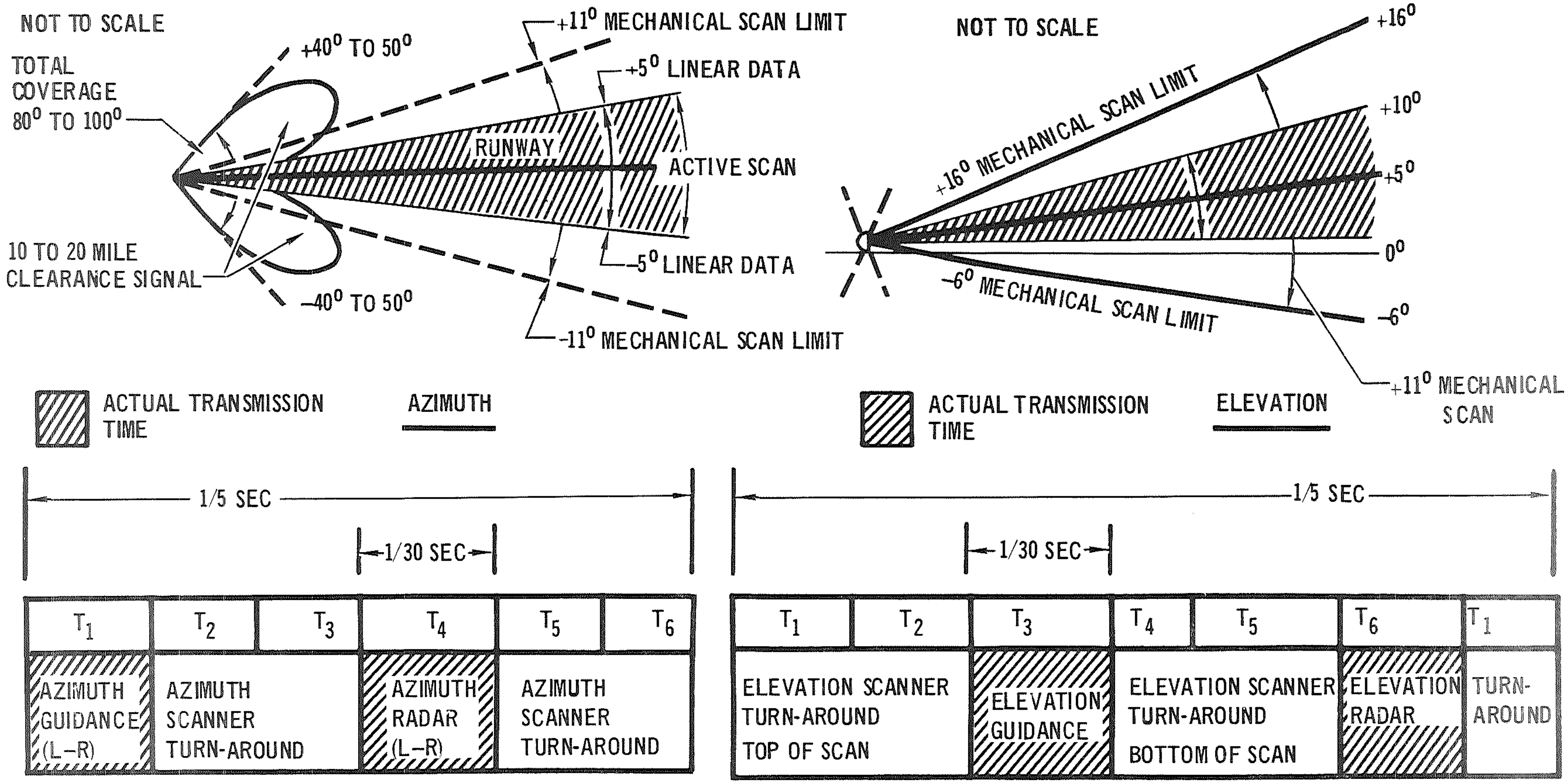
Unlike Flarescan which transmitted guidance information on both the up and down scan of the elevation antenna and on both the left and right scan of the azimuth antenna, AILS transmits guidance information during only one scan of each antenna. Figure 4-74 depicts this operation. Elevation guidance information is transmitted only during the down scan (T_3) and azimuth guidance information is transmitted only during the left-to-right scan (T_1). During the azimuth right-to-left scan (T_4) and the elevation up scan (T_6), the system performs precision approach radar (PAR) operation. This PAR information is presented to a ground controller so he can keep track of the approaching aircraft (A/C). Several A/C can thus be under simultaneous approach and the ground controller can differentiate between them while the pilots fly each of the A/C based upon received guidance and range information. The ground controller could still have to identify to the A/C their respective approach spacing. The DME information is furnished to the ground controller also, even after TD, thus providing the ground controller knowledge when the runway is clear for another A/C to land.

The approaching aircraft pilot can choose from a variety of glideslope angles, always knowing what glideslope he is following. The cockpit display is the conventional ILS crosspointer and DME range readout. The airborne units, besides incorporating a receiver, angle and distance decoders, and the necessary readout coupler circuitry, also include a computer for flare and control. The computer can be programmed to command progressively shallower angle of attack to the autopilot pitch channel. Since this concept is similar to ILS, little to no pilot retraining is required with this system for manual landing.

The AN/SPN-42 is manufactured by Bell Aerosystems for the Navy. The concept is a well-proven, fleet-operational, carrier-based, automatic landing system. It supersedes the AN/SPN-10.

The system consists of a precision dual tracking radar, shipboard computer, data link to and from the A/C, and the A/C autopilot and autothrottle. Three methods of landing are available; GCA (talkdown), semiautomatic (cross-pointer display, pilot nulls errors and manually lands the A/C), and fully automatic.

LOCALIZER AND ELEVATION ANTENNA PATTERNS AND OPERATIONS



SEQUENTIAL TIME-SHARED TRANSMISSION IN AILS FOR LOCALIZER AND GLIDE SLOPE ANTENNAS. EACH COMPLETE CYCLE IS DIVIDED INTO SIX TIME SLOTS OF 1 30TH SEC. DURATION. DURING THIRD SLOT (T_3). VERTICALLY SCANNING ANTENNA TRANSMITS VERTICAL GUIDANCE DATA AS IT SCANS DOWN. DURING T_6 IT OPERATES AS AN ELEVATION RADAR DURING ANTENNA UPSWING. SIMILARLY, AZIMUTH ANTENNA TRANSMITS GUIDANCE DURING T_1 AND SERVES AS AZIMUTH RADAR DURING T_4. TIME SLOT T_2 IS USED FOR DISTANCE INTERROGATION AND SLOT T_5 MAY BE USED ALSO FOR DME OR AN ADDITIONAL VERTICAL SCANNER. CLEARANCE SIGNAL IS TRANSMITTED AT START AND END OF T_1 SLOTS TO PROVIDE TURN LEFT RIGHT SIGNAL TO AIRCRAFT OUTSIDE MAIN BEAM.

Figure 4-73

AILS COMBINED OPERATION

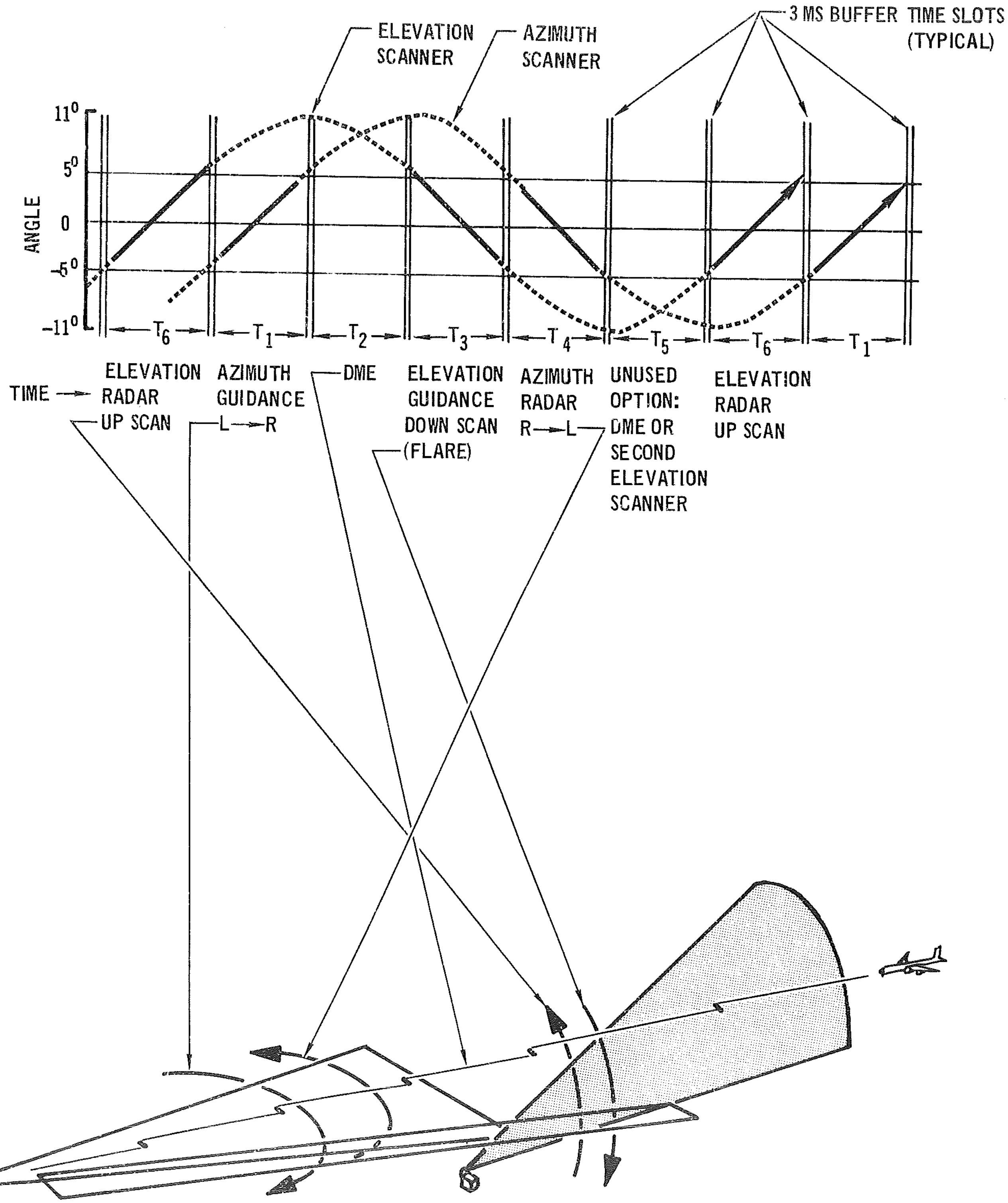


Figure 4-74

Automatic acquisition is at 4 NM range, although this may be manually increased to 8 NM. At 4 miles, the acquisition window is 11,000 wide by 700 feet high (12.0° x 2°), about 1200 feet deep, and is searched every 3 seconds by the carrier radar. Landing accuracy is ± 10 feet lateral and ± 40 feet longitudinal. The landing A/C is flown along a constant glide slope (3.5° to 4°) down to TD, without any flare.

The carrier-based equipment consists of a tracking and navigational computer, radar, signal data converter, ship motion monitor, UHF data link, control consoles, monitor displays, and associated power supplies.

The deck motion compensator measures the deck "heave" and for the last 12 seconds of the landing sequence, the A/C flight path is commanded to follow the deck motion. Landing sequence (automatic) is as follows: prior to 4 NM, the A/C is picked up by the AN/USQ-20 radar and the computer tells the SPN-42 the A/C type, range, correct altitude for acquisition gate, and time-to-go till the A/C reaches the acquisition gate. During this time, the pilot engages the autopilot coupler. At about 4 NM, the SPN-42 radar locks onto the A/C and transmits a lock-on discrete to the A/C. The pilot acknowledges lock-on and transmits a "pilot-ready" discrete. SPN-42 equipment then starts sending commands at 10 per second until TD or waveoff.

The airborne equipment consists of a Radar Signal Augmentor, high speed data link, autopilot, autopilot coupler, displays, and UHF voice and data communication link.

The accuracy of the ILS is not adequate under adverse conditions and only marginally acceptable under ideal conditions. The AILS and SPN-42 possess the basic accuracy for landing phase of the space shuttle. The SPN-42 has proven successful for many shipboard landings. The AILS has been flight tested by the FAA and was found acceptable for automatic landing. FAA Report RD 68-2 describes the results of the flight test evaluation. It is expected that the FAA will have certified an all weather automatic landing system by the mid-1970's. A system similar to AILS probably will be selected. Provided a system is selected in a time scale compatible with ILRVS development, this system should be the strongest candidate.

Conclusion and Recommendation - The ILRVS guidance, navigation and control, system implementation are in consonance with a technology capability of 1972. Detailed studies and special emphasis development are required to fulfill the operational objectives of the ILRVS program. Of particular significance to the G, N, & C systems are: flexibility in use, flexibility for growth, autonomous operation, a high level of on-board failure detection capability, and an efficient data management and crew participation concept. Study recommendations are described below.

Inertial Sensors - Past space programs have used gimballed planforms as the source of highly accurate navigation and attitude data. Development of strapdown IMU's show promise of attaining accuracy comparable to gimballed IMU's. The mechanical complexity of the platform gimbals, torque motors and sliprings, is replaced by the more reliable electronic computers in the strapdown configuration. A concept wherein strapdown gyro and accelerometers are aligned normal to the six faces of a regular dodecahedron is being developed. This concept provides a significant improvement in reliability over competing concepts which utilize redundant orthoganally mounted sensors. It is particularly applicable to the ILRV or any program where multiple redundancy is used.

Extensive testing and in some cases trend analysis is performed to determine satisfactory performance prior to flight. On-board checkout does not lend itself well to this detailed a test. A detailed study should be made to determine:

- o Equipment tolerances attainable on an operational basis
- o Penalties due to accuracy tolerances of concepts evaluated
- o Checkout concept which provide fault detection levels compatible with the ILRV requirements
- o Test and development required, if any, to utilize the most promising concept for the space shuttle.

Rendezvous - An optical tracking device was developed as an alternate means of obtaining rendezvous data for the Apollo program. Test and analysis of this concept showed that angular tracking data could be provided for a cooperative target at ranges up to 400 miles. Range information was obtained through use of a UHF transponder. Sunlit passive targets could be tracked at comparable ranges. Algorithms have been developed which permit rendezvous from angular data alone.

To use a radar for rendezvous with a passive satellite at 400 miles requires an excessive amount of power. Studies are required to determine the spread of rendezvous requirements, and the penalties associated with optical devices that can track only a sunlit target. In addition, IR tracking on the dark side of the Earth should be considered.

4.3.6 Telecommunications Subsystems - The telecommunications subsystem includes voice, data transmission and reception, TV, and flight recording equipment.

Requirements - The space shuttle requires a flexible telecommunication design capable of providing a variety of links to other space vehicles and ground bases. Because of the autonomous operation the data bandwidth needed is that required for voice or low data rate transmission. Nearly continuous communications capability is desired and contributes to improvement in safety, crew morale, mission reliability and permits real time control of unmanned spacecraft. Table 4-19 shows a detailed listing of the telecommunications system functional requirements by mission phase. The system implementation to meet these requirements is covered in the next section. Often one system can be used to meet several system requirements. This is desired to minimize telecommunication system complexity. The telecommunications RF link requirements are summarized in Figure 4-75.

System Description - The baseline includes two separate communications systems. One operates in the SHF band and is compatible with the Intelsat IV communications relay satellite system to minimize need for ground stations. The second system is a UHF system that provides direct communications with the space station, astronauts on emergency EVA, and the airports during landing. The block diagram of the communications system and estimated equipment size, weight and power is shown in Section 4.3.1.

Relay Communications - The relay communications link will provide communications capability virtually 100% of the time spent in orbit. This is an improvement over the Manned Space Flight Network that provides coverage only 10 to 25% of the time depending on orbit inclination. In addition, the relay satellite means of ground communications provides economical operation by deleting the need for the many ground stations now used for manned flights.

For the baseline system it is assumed that an Intelsat IV relay satellite will be used. This assumption was made because of potential economic advantages in using existing general purpose relay satellite systems rather than launch a dedicated relay satellite system for space use. Intelsat IV is currently being developed by the Communications Satellite Corporation, for operation in the early 1970's. Study has indicated that use of Intelsat IV is feasible but its use imposes stringent requirements on the shuttle communication system design. For example, a high gain (35 db) antenna (6 ft. parabolic disk) with a low noise

Table 4-19

TELECOMMUNICATION REQUIREMENTS

REQUIREMENTS	LAUNCH	IN ORBIT	CRUISE/ LANDING
• ONE DIRECT FULL DUPLEX VOICE CHANNEL BETWEEN THE SHUTTLE AND GROUND	(0-C)	(0)	(0-C)
• ONE RELAY FULL DUPLEX VOICE CHANNEL BETWEEN THE SHUTTLE AND GROUND		(0)	
• ONE DIRECT FULL DUPLEX VOICE CHANNEL BETWEEN THE SHUTTLE AND OTHER SPACE VEHICLES OR BETWEEN THE SHUTTLE AND OTHER AIRBORNE VEHICLES	(0-C)	(0)	(0-C)
• ONE DIRECT EMERGENCY EVA DUPLEX VOICE CHANNEL		(0)	
• DATA LINK FOR ROUTINE STATUS REPORTING TO GROUND OR SPACE STATION (3 KHz INFORMATION BANDWIDTH)		(0)	(0-C)
• DATA LINK FOR RECEIPT OF COMMANDS OR MAINTENANCE DATA FROM GROUND OR SPACE STATION (3 KHz INFORMATION BANDWIDTH)	(0-C)	(0)	(0-C)
• RECORD CRITICAL FLIGHT PARAMETERS	(0-C)	(0)	(0-C)
• VOICE INTERCOM	(0)	(0)	(0)
• EMERGENCY RECOVERY AID	(0-C)	(0)	(0-C)

NOTES: 0 - ORBITER
C - CARRIER
CARRIER IS ASSUMED TO BE MANNED IN THIS REQUIREMENT LIST.

TELECOMMUNICATIONS LINKS

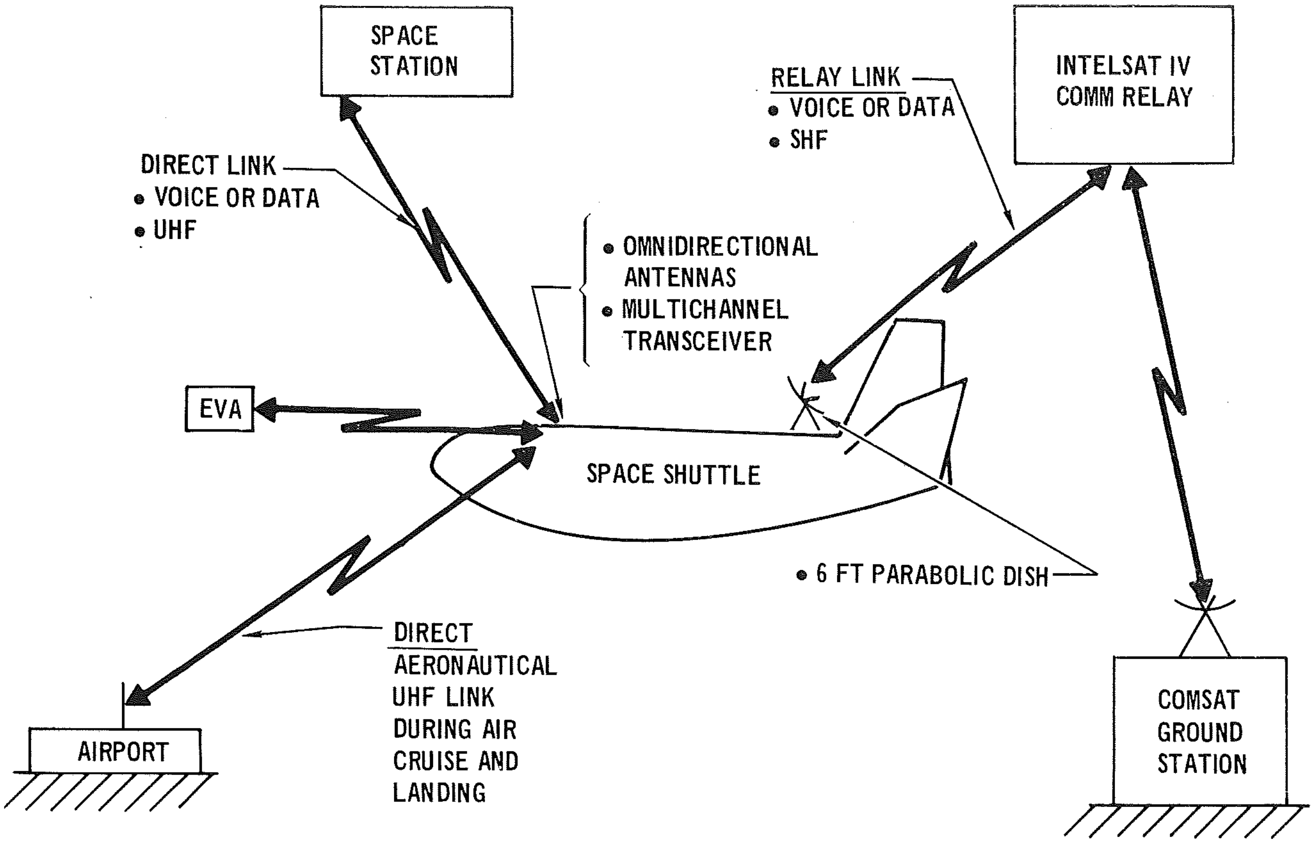


Figure 4-75

system (350°K) receiver is required for an information bandwidth of 3 KHz. Table 4-20 shows a signal to noise ratio margin analysis for the Intelsat IV to shuttle link. This is the critical link since the Intelsat IV effective radiated power is limited by fixed antenna beamwidth (global coverage is required) and fixed transmitter RF power output (6.3 watts). The Intelsat IV is operated with a ground station having a low noise receiver system (40°K) and a 90 foot or greater diameter antenna (gain > 59 db). This points out the disadvantage at which the shuttle craft is operating when using the Intelsat IV relay system. The low data rate requirement allows the shuttle to get by with a 6 foot diameter dish which is still a significant penalty.

Direct Communications - The direct communications link provides voice/data transmission between the shuttle and space station, between the shuttle and the airport, and between the shuttle and astronauts on emergency EVA. It is desirable to use the same type of transceiver for each of these functions to simplify the communications system. Therefore, a UHF system operating in the aeronautical UHF region (225 to 399.95 MHz) has been selected. However, the final decision for direct link equipment must be based on the entire operational environment including space station and space experiment telecommunication requirements. For example, experiment carriers operating in conjunction with the space station may require a S-band system for transmission of high rate experiment data to the space station. A multichannel S-band transceiver on the space station could therefore also be used for communications with the shuttle.

The UHF system uses a multichannel transceiver system and omnidirectional antennas. Any of the 3500 channels can be selected; however, several commonly used channels would be preset for ease of selecting these channels. Channel tuning is done electronically. RF power output of 20 to 100 watts is achieved by all solid state circuitry. The antenna system includes automatic antenna switches and flush mounted omnidirectional antennas. High temperature, flush mount, broadband annular slot antennas are used. Antenna switching is required to select the antenna that maximizes the received signal. If required, two transceivers can be operated simultaneously at 2 different sets of operating frequencies. Antenna switches are then used to connect both transceivers to a common antenna or to connect the two transceivers to different antennas. That is, each transceiver is connected to an antenna that will provide an adequate receive signal level.

Table 4-20

SHF COMMUNICATIONS RELAY LINK

	COMMUNICATIONS RELAY INTELSAT IV - 4 GHz
TRANSMITTED POWER RELAY TRANSMITTER LOSSES TRANSMITTER ANTENNA GAIN	48.2 dbm*
FREE SPACE LOSS (23,000 N.MI.)	-197.0 db
MISCELLANEOUS LOSSES	-1.0 db
ILRV ANTENNA GAIN	+ 35.db (6 FT DISH)
RECEIVED CIRCUIT LOSSES	-4.5 db
RECEIVED SIGNAL POWER	-119.3 dbm
NOISE SPECTRAL DENSITY (KT)	-175 dbm**
NOISE BANDWIDTH 30 KHz	44.8 db
RECEIVED NOISE POWER	-130.2 dbm
RECEIVED SIGNAL TO NOISE RATIO	+10.9 db
SIGNAL TO NOISE RATIO REQUIRED	9.0 db ***
SIGNAL TO NOISE RATIO MARGIN AT ILRV	1.9 db

* ASSUMES 3.8 db REDUCTION IN TOTAL RF POWER OUTPUT TO ALLOW FOR SUPPRESSION OF WEAKER CARRIER WHEN TWO CARRIERS ARE RELAYED BY THE SAME RELAY TRANSPONDER.

** ASSUMES 230^{o}K SYSTEM NOISE TEMPERATURE. AN UNCOOLED PARAMETRIC AMPLIFIER IS REQUIRED.

*** SUFFICIENT SIGNAL TO NOISE RATIO TO EXCEED THRESHOLD IN FM/FM SYSTEM.

Antennas - Table 4-21 summarizes the antenna requirements for all spacecraft systems.

A Voice Intercom system is used to enhance reporting to the passengers from the Earth, space station or crew.

The Communications Processor provides for voice and data signal processing, switching and routing. Included are decoding and formatting of received data, voice signal clipping, encoding of routine spacecraft status data prior to its transmission, and selection of the appropriate transceiver system.

The Flight Recorder monitors critical flight parameters which can be used for crash investigation and failure prediction. The recorder is crash proof and playback of data is done at the ground or space station.

Closed Circuit Television is used, as required, to provide visual accessibility to critical areas such as landing gear.

Alternate Concepts Evaluated - The key alternate concepts studies are listed below. Study results are summarized in Tables 4-22 thru 4-27.

a) Use of aeronautical UHF versus C-band for the communication relay link.
b) Mechanical scan parabolic dish antenna versus active electronic scan phased array antenna for the Intelsat IV relay link.
c) Separate antennas for rendezvous and communications versus a single antenna system for both functions.
d) Use of a mechanical scan parabolic dish antenna versus a mechanical scan passive planar array antenna.
e) Fuselage mounted high gain antennas versus dorsal fin mounted high gain antennas.
f) Radar mounted in nose behind radome versus a deployable radar.

Conclusions and Recommendations

Technology - The following are technology developments required for the baseline design.

- Reusable high temperature flush mounted antennas not requiring protection during launch/reentry
- Low noise receiver system for relay communications.

Table 4-21

ANTENNA SYSTEM REQUIREMENTS/SELECTION

ELECTRONIC SYSTEM	MINIMUM ANTENNA COVERAGE REQUIRED	NO. OF ANTENNAS REQUIRED	POLARIZATION REQUIRED	ANTENNA LOCATION	TYPE OF ANTENNA AND REMARKS
RELAY COMMUNICATIONS	HEMISPHERE	ONE (DUAL ELECTRONICS)	RHC–RECEIVE LHC–TRANSMIT	TOP OF FUSELAGE OR WITHIN DORSAL FIN	6 FT. PARABOLIC DISH. 3.7 TO 4.26 GHz RECEIVE. 5.925 TO 6.425 GHz TRANSMIT. DEPLOY AND USE ONLY IN ORBIT. UNFURLABLE IF LOCATED IN DORSAL FIN.
DIRECT COMMUNICATIONS	OMNIDIRECTIONAL	FOUR (2 PER SYSTEM)	VERTICAL	TWO ON TOP AND TWO ON BOTTOM OF FUSELAGE	FLUSH MOUNT ANNULATE SLOT. 225 TO 400 MHz 24"x24"x4.2" DEEP.
RENDEZVOUS RADAR	60^0 SOLID CONE ANGLE FORWARD OF SPACECRAFT	ONE (DUAL COMMON ELECTRONICS)	LINEAR	FORWARD AND TOP OF FUSELAGE	• DEPLOYABLE PARABOLIC DISH OR PASSIVE CORPORATE FEED PLANAR ARRAY • C-BAND
ADVANCED INSTRUMENT LANDING SYSTEM	FORWARD LOOKING $\pm 40^0$ PITCH $\pm 50^0$ AZIMUTH	THREE (1 PER SYSTEM)	CIRCULAR		• OPEN ENDED Ka BAND WAVE GUIDE • 15.4 TO 15.7 GHz BAND

ANTENNA SYSTEM REQUIREMENTS/SELECTION

ELECTRONIC SYSTEM	MINIMUM ANTENNA COVERAGE REQUIRED	NO. OF ANTENNAS REQUIRED	POLARIZATION REQUIRED	ANTENNA LOCATION	TYPE OF ANTENNA AND REMARKS
TACAN	OMNIDIRECTIONAL IN AZIMUTH ±45 DEG IN ELEVATION.	4 (2 PER SYSTEM)	VERTICAL	ONE ON BOTTOM AND ONE ON TOP CENTER LINE PER SYSTEM	ANNULAR SLOT 8.5" DIA., 2" DEEP 960–1220 MHz
RADAR ALTIMETER	40^0 SOLID CONE ANGLE. BEAM DIRECTED ALONG LOCAL VERTICAL	6 (2 PER SYSTEM) ONE RECEIVE ONE TRANSMIT	LINEAR	BOTTOM: NEAR FWD-AFT CENTER OF GRAVITY	HORN ANTENNA 7" DIA x 3" DEEP, 4.3 GHz
RECOVERY BEACON	HEMISPHERE ABOVE WATER OR LAND SURFACE	1	VERTICAL	DORSAL FIN	ANTENNA AND TRANSCEIVER THROWN FROM SPACECRAFT BY CRASH, HYDROSTATIC PRESSURE OR PILOT. 243 MHz
AIR TRAFFIC CONTROL	OMNIDIRECTIONAL IN AZIMUTH. ±45 DEGREES IN ELEVATION	2	VERTICAL	ONE ON BOTTOM AND ONE ON TOP CENTER LINE	ANNUAL SLOT 8.5" DIA., 2" DEEP 960–1220 MHz

Table 4–21 (Cont'd)

Table 4-22

UHF VS. C-BAND FOR RELAY LINK

	PROS	CONS
UHF	• USE OMNI ANTENNAS ON ORBITER • SIMPLE ORBITER SYSTEMS	• POTENTIAL INTERFERENCE FROM GROUND RADIATORS. • POTENTIAL MULTIPATH INTERFERRENCE • UHF SATELLITE MAY NOT BE AVAILABLE IN SHUTTLE TIME PERIOD (TACSAT 1 CURRENTLY AVAILABLE)
C-BAND	• USE EXISTING COMMERCIAL RELAY OF SHUTTLE TIME PERIOD (I.E. INTELSAT IV) • DEDICATED RELAY NOT REQUIRED	• REQUIRE HIGH GAIN (6 FT.) ORBITER ANTENNA • REQUIRES LOW NOISE RECEIVE SYSTEM ON ORBITER (3.5 Db NOISE FIGURE)

Conclusions: A C-band system was selected to be compatible with Intelsat IV. The aeronautical UHF band system offers simplicity of design and would allow common equipment to be used for all voice and data links. TACSAT I is an existing satellite relay that has a compatible UHF relay. However, the next generation TACSAT may not include a UHF relay. Also, the potential interference and channel available problems must be further analyzed before UHF (225 to 400 MHz) can be selected as the baseline system for the orbiter relay link.

Table 4–23

PARABOLIC DISH ANTENNA VS ACTIVE ELECTRONICALLY STEERED ARRAY FOR RELAY COMMUNICATIONS VIA INTELSAT IV

	PROS	CONS
DISH	• LOW NOISE SYSTEM PRACTICAL (2.5 TO 3.5 DB) • COMPARABLE SYSTEMS DEVELOPED AND USED SUCCESSFULLY IN SPACE	• MUST BE DEPLOYED • MOVABLE PARTS • LARGE STOWAGE SPACE REQUIRED; DEPTH $\approx$ DIAMETER/2
ARRAY	• NO DEPLOYMENT REQUIRED • FLUSH MOUNT • NO MOVING PARTS • DEPTH $<$ 6 INCHES	• SYSTEM NOISE TEMPERATURES 8–10 DB • EACH ARRAY LIMITED TO 120 DEGREE SOLID CONE SCAN ANGLE • GAIN DECREASES WITH SCAN OFF BORESIGHT (–3 DB AT + 60^{o}) • ARRAY EXPOSED TO LAUNCH/ENTRY HEATING

Conclusions: The parabolic dish is selected over active arrays because four active arrays are required to obtain spatial coverage equivalent to that obtainable with the dish. Aperture of each array needs to be 113 to 195 sq. ft. to obtain receive performance equivalent to a system with a 6 foot dish and a 3.5 db noise figure. Installation of four arrays with correct orientation (e.g. to achieve good forward coverage) is not practical. Weight of the array systems (4) is estimated at 1600 pounds vs 100 pounds for the dish system.

Table 4-24

SEPARATE VS COMMON ANTENNAS FOR COMMUNICATIONS AND RENDEZVOUS TRACKING

ANTENNA TYPE	PROS	CONS
COMMON	• ONE ANTENNA • ONE TRANSMITTER • ONE DEPLOYMENT MECHANISM • WITH SINGLE REDUNDANCY OMNI-DIRECTION COVERAGE CAN BE PROVIDED FOR EACH FUNCTION.	• TIME SHARING REQUIRED UNLESS SEPARATE ANTENNAS AND SEPARATE FREQUENCIES ARE USED FOR EACH FUNCTION.
SEPARATE	• TIME SHARING NOT REQUIRED • LESS COMPLEXITY OF EACH SYSTEM • HARDWARE MATCHES NORMAL ORGANIZATION GROUPING	• TWO DEPLOYABLE ANTENNAS WITH ASSOCIATED DOORS AND DEPLOYMENT MECHANISM

Conclusions: Separate communication and radar systems were selected. Each can be located to provide good coverage without interferring with the others operation. However, a combined rendezvous and communications system using a common transmitter, a common antenna, and separate receivers was found to be feasible. The system studied used interrupted CW for the radar mode. The commu-ications mode is compatible with Intelsat IV.

Table 4-25

USE OF A MECHANICAL SCAN PARABOLIC DISH ANTENNA VERSUS A MECHANICAL SCAN PASSIVE PLANAR ARRAY ANTENNA

ANTENNA	PROS	CONS
DISH	• MORE CONSISTANT WITH STANDARD PRACTICES • MINIMUM DEVELOPMENT	• DEPTH ≈ DIAMETER/2 • FURL ANTENNA TO INSTALL IN DORSAL FIN
PLANAR ARRAY	• < 6 INCH DEPTH • CAN MOUNT IN DORSAL FIN WITHOUT FURLING OR FOLDING	• MORE DEVELOPMENT REQUIRED

Conclusions: The dish antenna was selected as the baseline on the basis of minimum development. However, a passive array with a 4.5 x 4.5 foot aperture and 1300 crossed dipoles has been investigated. This array provides the same performance as a dish. It has less depth than a dish and therefore is more amenable to a dorsal fin installation. Hybrids and branch line couplers are used to obtain orthogonal polarization for transmit and receive. Orthogonal polarization is required by Intelsat IV.

Table 4-26

FUSELAGE MOUNT VERSUS DORSAL FIN MOUNT FOR HIGH GAIN ANTENNA

MOUNT	PROS	CONS
TOP FUSELAGE:	• FWD MOUNT: CLOSE TO ELECTRONICS BAY • PARABOLIC DISH OR PLANAR ARRAY CAN BE STOWED WITHOUT FURLING • MINIMUM DESIGN IMPACT	• LESS COVERAGE OVER THE SIDE • LESS COVERAGE FORWARD AND BELOW
DORSAL FIN	• BETTER OVER THE SIDE COVERAGE • BETTER COVERAGE FORWARD AND BELOW	• SIX FOOT DISH REQUIRES: FURLING OF ANTENNA AND WIDENING OF DORSAL FIN • BOTH DISH AND PLANAR ARRAY REQUIRE DOOR IN DORSAL FIN FOR DEPLOYMENT • REMOTE FROM ELECTRONICS BAY
BOTTOM FUSELAGE	• 4π STERADIANS COVERAGE WITH BOTH BOTTOM AND TOP MOUNT	• DOOR REQUIRED IN HIGH HEATING AREA

Conclusions: A top fuselage mount behind the payload was selected since it provides good coverage (>2π steradians) and has minimum spacecraft design impact. However, a dorsal fin mount should continue to be considered due to improved coverage capability. The installation of a mechanical steered passive array in the dorsal fin has advantages of fitting without widening fin structure.

Table 4-27

RADAR MOUNTED IN NOSE VS A DEPLOYABLE RADAR

MOUNT	PROS	CONS
NOSE (BEHIND RADOME)	• RADAR USABLE IN ORBIT AND AFTER ENTRY • NO DEPLOYMENT MECHANISM REQUIRED • MINIMIZE SPURIOUS ENERGY AT RECEIVER • GOOD FORWARD COVERAGE	• HIGH TEMPERATURE RADOME DEVELOPMENT REQUIRED. • HIGH TEMPERATURE EFFECTS ON REUSABLE RADOMES MUST BE DETERMINED.
DEPLOYABLE	• MINIMUM IMPACT ON SHUTTLE DESIGN • MINIMUM TECHNOLOGY DEVELOPMENT	• RADAR USABLE IN ORBIT ONLY; UNLESS SPECIALLY DESIGNED TO BE DEPLOYED DURING AERO CRUISE • FORWARD COVERAGE PROPORTIONAL TO LENGTH OF DEPLOYMENT BOOM

Conclusions: A deployable radar located forward and on top of the spacecraft was selected as baseline since the effects of high temperature on reusable radomes are unknown. The radar is used for both cooperative and noncooperative tracking in orbit. The use of a radar mounted behind a nose radome was also investigated. Of the radars studied, a C-band active phased array with electronic beam steering is the best suited for mounting behind the radome. The electronic steered array can be located very near to the radome thus reducing radome size. The array can produce multiple beams therefore doppler navigation mode or an altimeter mode could easily be added. At C-band the array can be made small and yet take advantage of relatively high efficiency components. A 15 inch diameter array drawing 1440 watts is estimated for a range of 30 nautical miles and a 5 sq. meter uncooperative target.

The following are technology developments recommended for refinements in baseline design:

- o Mechanical steerable planar array for easy mount in dorsal fin
- o High temperature multiple reuse radomes for multimode radar in nose sections
- o Multimode phased array radar for cooperative and non-cooperative rendezvous.

<u>Follow-On Study Recommended</u>

- o Study alternate concepts, items a, and c thru f listed above in greater depth.
- o Refine system requirements using a typical operational environment as a reference. Factor in preliminary space station study results and data relay system characteristics.

4.3.7 Integrated Avionics Reliability - The ILRV requirements of autonomy and economical operation dictates stringent reliability goals as shown in Table 4-28. The goals of (1) remaining operational after two failures and safe after the third failure, (2) avoiding minimum performance backups, (3) minimizing system transients due to failure, and (4) high mission success probability, all dictate redundancy. These goals require equipment and system designs which have sophisticated methods of failure detection and selection of properly functioning units.

To meet these goals, both modular and functional redundancy are being used. In some cases we are able to provide backup with equipment already required for other functions. For example, the optical sensor is primarily used for inertial alignment and as an orbital navigation sensor, but it can also be used to back up the radar as a target tracker for rendezvous.

Another area of concern is failure detection and switchover between redundant units. The requirement to minimize switching transients impacts the techniques to be used as well. With three data sources, active majority voting can be used to determine which output is in error and thus allow switchover to a monitored middle select output. Other techniques such as "Pair and Spare", where two systems are compared and for discrepancies in outputs, switched to a third unmonitored system, do not meet the switchover transient criteria. The use of fade in logic to control the rate of change of output signals would help. Another important factor in achieving a high probability of mission success is to have a ground maintenance analysis program. Trend data recorded on board, historical failure records, and periodic inspection data is used in a quantitative manner to program replacements of on-board equipment.

An example of equipment redundancy implementation for the guidance and control system is shown in Table 4-29.

Table 4–28

RELIABILITY GOALS

GOAL	APPROACH
• FIRST AND SECOND FAILURE – REMAIN OPERATIONAL • THIRD FAILURE – NON-CATASTROPHIC	• MODULAR AND FUNCTIONAL REDUNDANCY • FAILURE DETECTION AND SWITCHOVER
• AVOID MINIMUM PERFORMANCE BACK-UPS	• FUNCTIONAL REDUNDANCY PERMITTED ONLY WHEN MISSION PERFORMANCE IS NOT REDUCED • WHERE POSSIBLE USE EQUIPMENT ON BOARD FOR OTHER MISSION REQUIREMENTS
• MINIMIZE SYSTEM TRANSIENTS DUE TO FAILURE	• ACTIVE FAILURE DETECTION (e.g. MIDDLE SELECT) • FADE-IN LOGIC
• MISSION SUCCESS = 0.95	• HI-RELIABILITY EQUIPMENT • ON-BOARD FAULT DETECTION AND REDUNDANCY • PROGRAMMED GROUND MAINTENANCE

Table 4–29

TYPICAL REDUNDANCY APPLICATIONS
-For Orbiter G & C Functions

SUBSYSTEM ELEMENT	REDUNDANCY EMPLOYED	RELIABILITY ESTIMATE
I.G.S.		
COMPUTER	DEDICATED COMPUTER (TRIPLY REDUNDANT)	.99989
I.M.U.	STRAPDOWN INERTIAL UNIT (TRIPLY REDUNDANT)	.99998
RATE GYRO PACKAGE	BACKUP R.G. PACKAGE	.99997
RENDEZVOUS SYSTEM RADAR	DUAL RADARS – OPTICAL BACKUP	.99999
TIME REFERENCE SYSTEM	DUAL – ACTIVE REDUNDANCY	.99993
STAR TRACKER HORIZON SENSOR	DUAL REDUNDANCY	.99997
DISPLAYS AND CONTROLS	100% REDUNDANT– CRT & HEADS-UP DISPLAY	.99999
TERMINAL RENDEZVOUS OPTICS	DUAL REDUNDANT OPTICAL SUBSYSTEM	.99995
TOTAL (ALLOCATION)		.99967 (.9885)

4.4 Electrical Power

Summary - The characteristics of the electrical power subsystems for both the carrier and the orbiter are described in this section. The energy requirements and selected baseline power sources for the baseline vehicles are as follows:

Vehicle	Energy Required	Selected Power Source
Carrier	2.15 KWH	AgO-Zn Batteries
Orbiter	589.3 KWH	H_2O_2 Fuel Cells with peaking/emergency AgO-Zn batteries

4.4.1 Electrical Power Requirements - A seven day mission was used as a baseline for the orbiter load analysis. The mission consists of 26 hours for prelaunch through ascent and rendezvous, 120 hours orbital operation, and 24 hours for return, descent and landing. The orbiter load summary is shown in Table 4-30. The total energy required for the mission is 589.3 KWH. The overall average main bus power is 3.46 KW, with peaks of 6.94 KW during rendezvous operations. Figure 4-76 shows the variation in main bus average power for the various mission phases.

The baseline mission for the carrier consists of 2 hours for prelaunch, 10 minutes for liftoff through jet engine start, and 2 hours for cruise through landing. The carrier load summary is shown in Table 4-31. The carrier requires 21.5 KWH of energy to perform its mission. The average power level is 5.2 KW, with 5.83 KW peaks during cruise and landing. The variation of main bus average power with respect to carrier mission phase is shown in Figure 4-77.

All power quantities used in the load analyses were based on a 28 VDC bus. Inversion losses were added for equipment operating on AC.

The electrical power required for operation of the main propulsion engines has not been included in the load summaries. This power (6.2 KVA @ 115V 400 Hz per engine) will be supplied by turbine driven auxiliary power units (APU). These units also provide backup hydraulic power for engine gimbal and prime hydraulic power for the aerodynamic control surface prior to turbojet operation.

4.4.2 Electrical Power Subsystem (EPS) Baseline - The baseline electrical power subsystem configurations for the orbiter and the carrier are described in the following paragraphs. The main power sources for the orbiter are H_2-O_2 fuel cell

Table 4–30

ORBITER ELECTRICAL LOAD SUMMARY
Electrical Energy in Watt-Hours

EQUIPMENT \ MISSION PHASE	PRELAUNCH 2 HOURS	ASCENT 1 HOUR	ORBITAL PHASING 20 HOURS	RENDEZVOUS 3 HOURS	ORBITAL OPERATIONS 120 HOURS	RETURN PHASING 22 HOURS	ENTRY & LANDING 2 HOURS
INERTIAL SENSORS	1,500	750	15,000	2,250	90,000	16,500	1,500
COMPUTERS	2,200	1,100	22,000	3,300	132,000	24,200	2,200
FLIGHT CONTROL AMPLIFIERS	740	408	600	110	1,800	825	713
3-AXIS RATE GYROS	90	45	900	135	5,400	990	90
COMMUNICATIONS	525	355	5,670	1,050	–	4,088	635
RENDEZVOUS RADAR	–	–	–	800	–	–	–
DISPLAYS & CONTROLS	2,670	1,335	27,500	4,179	32,400	30,720	2,750
NAVIGATION AIDS	–	–	800	120	–	880	–
LANDING AIDS	–	–	–	–	–	–	644
DATA HANDLING	540	350	5,400	810	18,000	5,940	540
TV CAMERAS	–	–	160	80	–	175	80
EC/LS	1,218	609	12,180	1,822	36,500	13,410	1,218
LIGHTING	500	250	5,000	750	–	5,500	500
MISC & LOSSES	599	312	5,713	924	18,966	6,194	652
TOTAL ENERGY (WATT-HR)	10,582	5,514	100,923	16,330	335,066	109,422	11,522
AVERAGE POWER (WATTS)	5,291	5,514	5,046	5,443	2,792	4,974	5,761

TOTAL ENERGY FOR 7 DAY MISSION – 589.3 KWH
AVERAGE POWER FOR 7 DAY MISSION – 3.46 KW
PEAK POWER (DURING RENDEZVOUS) – 6.94 KW

ORBITER MAIN BUS AVERAGE POWER
Total Mission Energy: 589.3 KWH

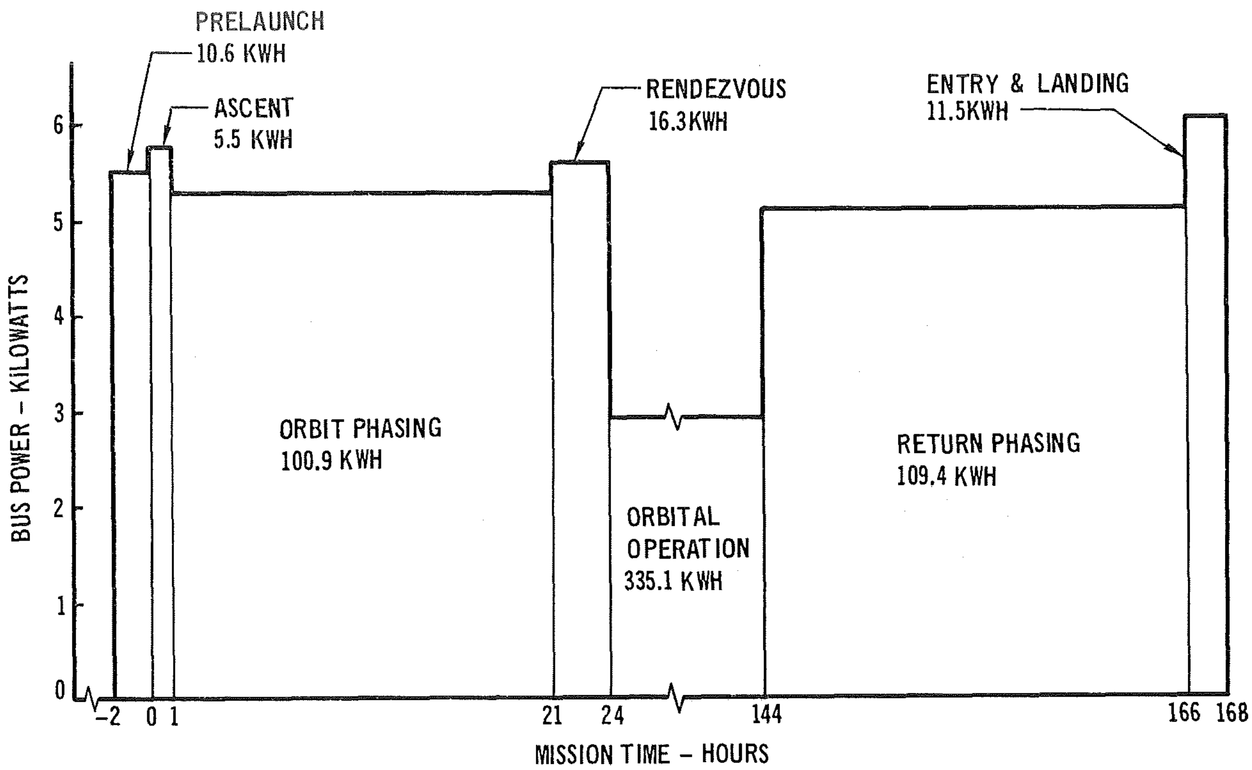


Figure 4-76

Table 4-31

CARRIER ELECTRICAL LOAD SUMMARY

Equipment \ Mission Phase	Prelaunch 2 Hours	Ascent 10 Minutes	Cruise & Landing 2 Hours
Inertial Sensors	1,500	125	1,500
Computers	2,200	183	2,200
Flight Control Amplifiers	740	62	683
3-Axis Rate Gyros	90	7	90
Communications	525	61	635
Displays & Controls	2,830	243	2,910
Landing Aids	--	--	544
Data Handling	380	32	380
TV Cameras	--	7	80
EC/LS	988	82	988
Lighting	125	11	125
Misc. & Losses	563	49	608
Total Energy	9,941 W-HR	862 W-HR	10,743 W-HR
Average Power	4,970 W	5,172 W	5,372 W

Total Mission Energy 21.5 KWH
Average Mission Power 5.2 KW
Peak Power (During Cruise and Landing) 5.83 KW

CARRIER MAIN BUS AVERAGE POWER

Total Mission Energy: 21.5 KWH

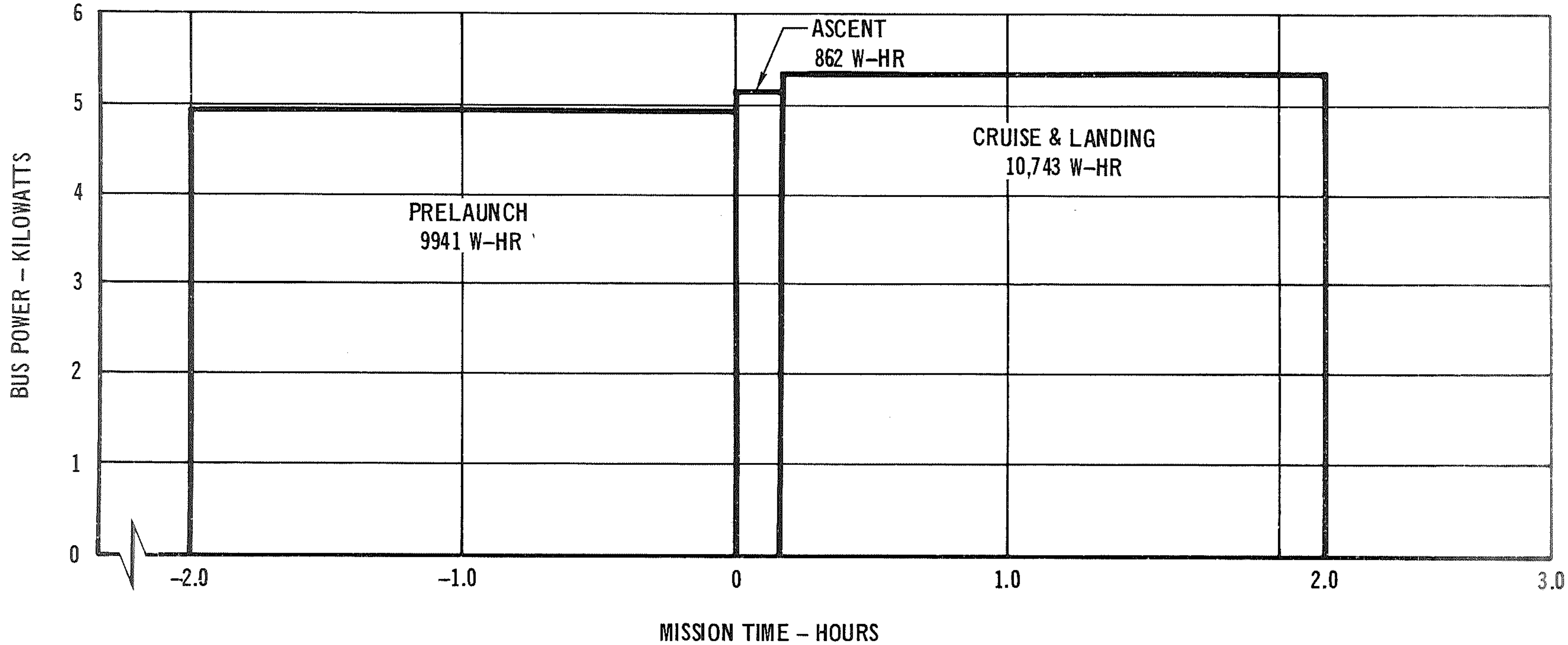


Figure 4-77

modules. For the carrier, rechargeable AgO-Zn batteries are used. Except for the power sources, the subsystems are essentially identical for both the orbiter and carrier.

Figure 4-78 and Figure 4-79 show the EPS configurations for the orbiter and carrier, respectively. The design philosophy used is an adaptation of that used in the design of commercial aircraft such as the DC-9 and the DC-10. The components of the EPS (for both orbiter and carrier) are interconnected to form two separate power source channels. These prime source channels can be operated either independently, or in parallel. Paralleling of the DC buses is accomplished by closing the DC bus tie relay No. 3 (DCBTR3), and the AC buses can be paralleled by closing the AC bus tie relay No. 3 (ACBTR3). The inverters are timed by a common clock located in the inverter frequency reference. This common clock synchronizes the inverters so parallel operation is possible. The inverter frequency reference contains sufficient redundancy to maintain the desired system reliability.

Both the DC and the AC buses are further divided into essential and non-essential buses. Only that equipment that is absolutely essential for crew and vehicle survival is connected to the essential buses - all other equipment is connected to the non-essential buses. Although circuit protection components are not shown, unprotected circuits will be kept to an absolute minimum consistent with safety.

Orbiter Power Source - Prime power for the orbiter is supplied by four H_2-0_2 matrix type fuel cell modules. Each module is rated at 2.0 - 2.5 KW, for a total capability of 8 - 10 KW at the buses. All four fuel cell modules are operated simultaneously for reactant economy as well as continuity of power in the event of a module failure. The peaking/emergency batteries are rated at 6.0 KWH each. These serve two purposes, (1) they improve the bus transient response characteristics (the battery voltage is slightly below the nominal bus voltage), and (2) they will provide power up to two hours for emergency deorbit, entry and cruise in the event of a catastrophic failure of the fuel cell system.

The orbiter power source is sized so that a safe return is possible with two fuel cell modules failed.

Table 4-32 shows the major components and their estimated weight for the orbiter EPS (excluding mounting provisions and radiators).

ORBITER ELECTRICAL POWER SUBSYSTEM

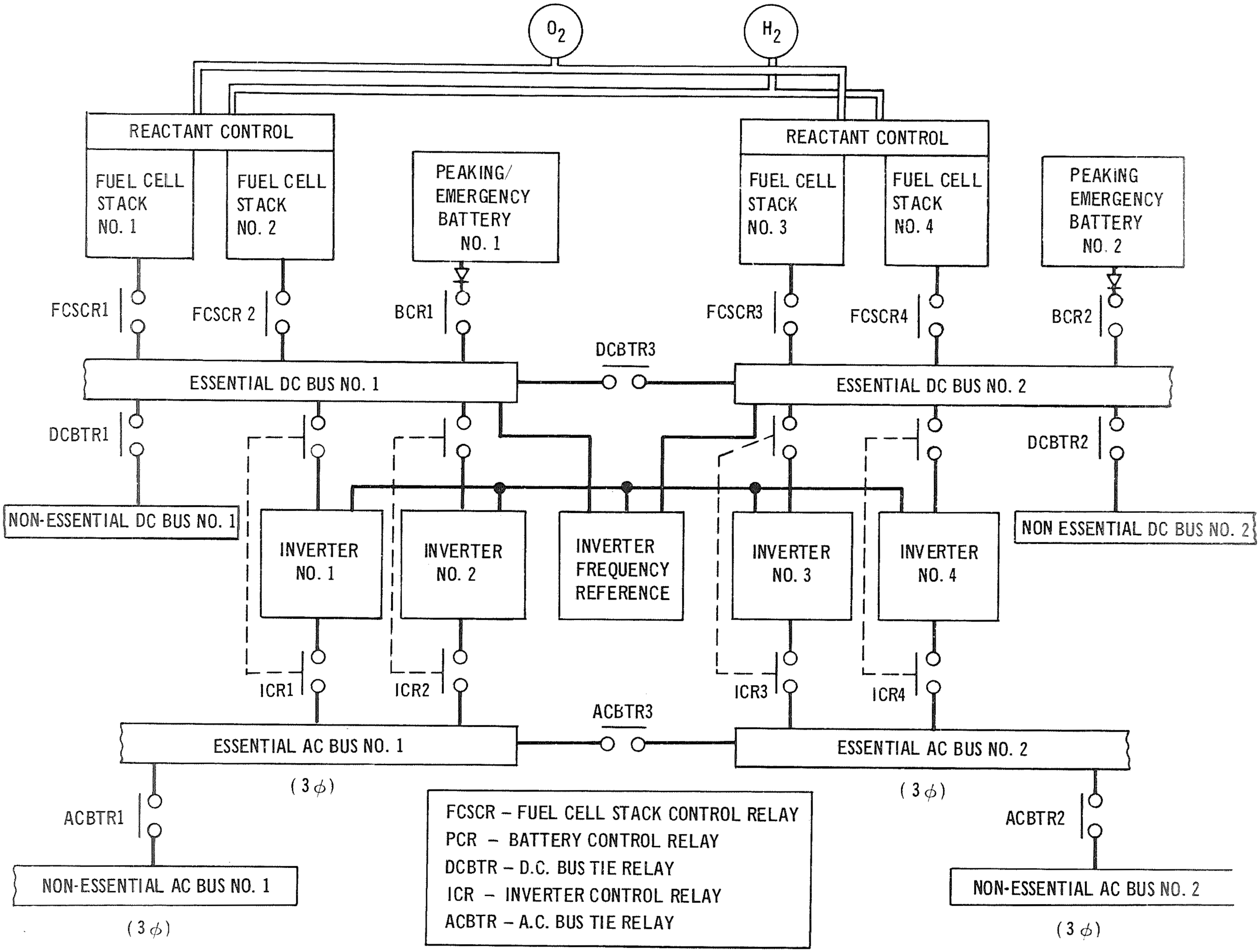


Figure 4-78

REPORT NO. MDC E0049
NOVEMBER 1969

CARRIER ELECTRICAL POWER SUBSYSTEM

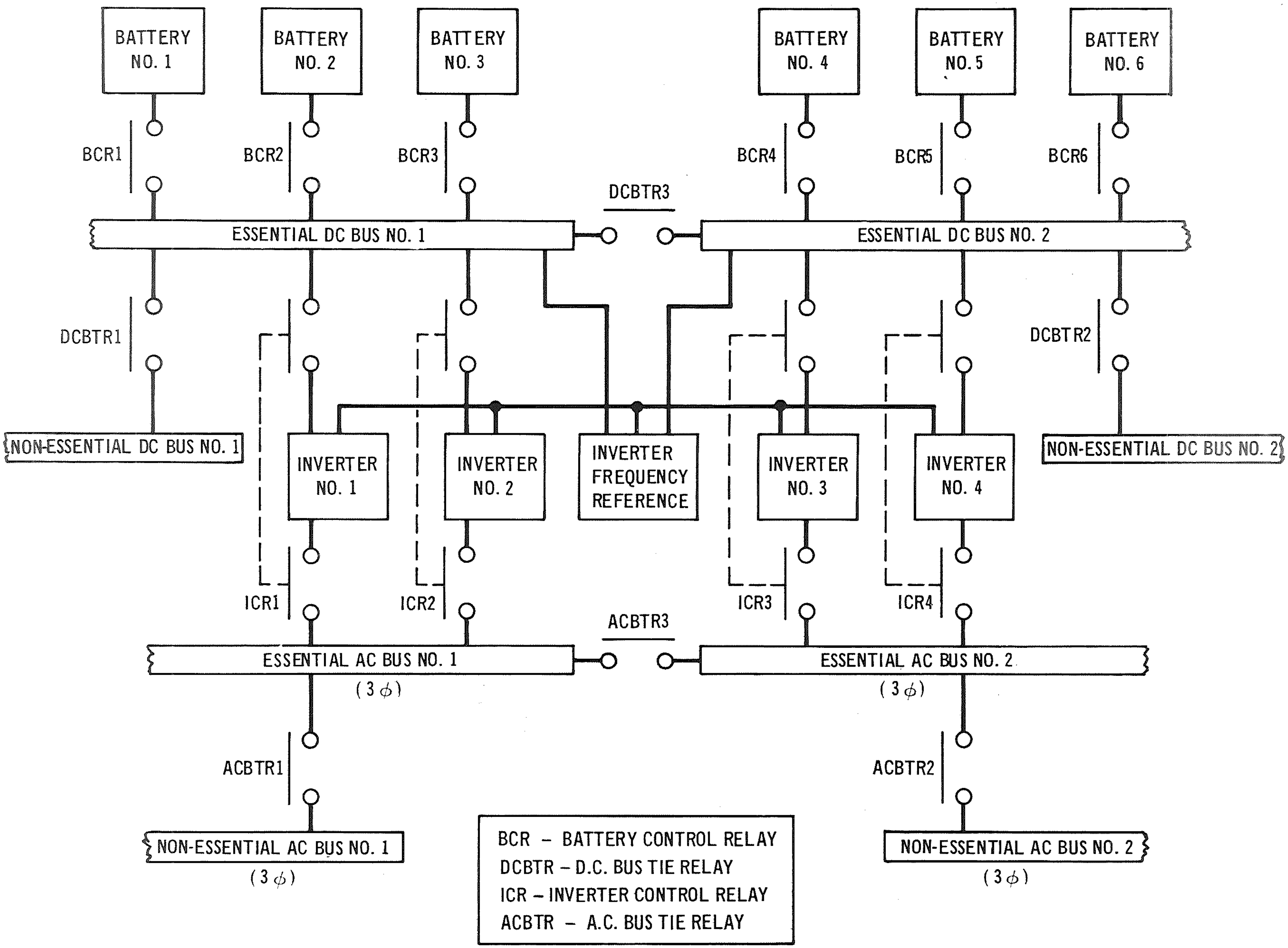


Figure 4-79

REPORT NO. MDC E0049
NOVEMBER 1969

Table 4-32

ORBITER EPS WEIGHT

Item	Qty	Unit Wt (lb)	Total Wt (lb)
Fuel Cell Module	4	100	400
Reactant Control Assy	2	15	30
Thermal Control Unit	1	40	40
Product Water Subsystem	1	40	40
Control Subsystem	1	40	40
Hydrogen Tank	1	90	90
Hydrogen	-	-	59
Oxygen Tank	1	98	98
Oxygen	-	-	471
Inverter	4	40	160
Peaking/Emergency Battery	2	115	230
Power Distribution Subsystem	-	-	700
Total			2358 lb

Carrier Power Source - Prime power for the carrier is supplied by six 6.0 KWH rechargeable AgO-An batteries, for available energy totaling 36 KWH. The battery control relays (BCR) are reverse current sensing as well as control relays to prevent degradation of the remaining batteries in the event of a battery failure.

The carrier power source is sized so that the mission can be completed with two battery failures.

Table 4-33 shows the major components and their estimated weight for the carrier EPS (excluding mounting provisions).

4.4.3 Alternate Concepts - During the course of the study, several different power sources were investigated for potential use in the space shuttle vehicle. These are listed in Table 4-34 along with the advantages and disadvantages for each candidate.

A turbo alternator power source may be competitive with batteries for the carrier, due to the relatively short flight duration. This is especially true if the same turbines are used to drive hydraulic pumps as well as alternators. Further study is required in this area with more complete analysis of the electrical and hydraulic load requirements.

4.4.4 Reliability - The electrical power subsystems for both the orbiter and the carrier are designed for mission completion with two power sources failed (orbiter - 2 fuel cell modules failed, and booster - 2 batteries failed). The busing is arranged for maximum utilization of remaining power sources in the event of a failure, and redundant using equipment is divided between the separate buses. Fault isolation devices will be utilized to prevent bus degradation from failures in loads or short circuits in interconnecting wiring. Further definition of the vehicle configuration is required to define the fault isolation scheme to be used.

Table 4-33

CARRIER EPS WEIGHT

ITEM	QTY	UNIT WT (LB)	TOTAL WT (LB)
200 A-H AgO-Zn Battery	6	115	690
Inverter	4	40	160
Power Distribution	-	-	700
Total			1550 lb

CANDIDATE ELECTRICAL POWER SOURCES

POWER SOURCE	ADVANTAGES	DISADVANTAGES
AgO–Zn BATTERIES (RECHARGEABLE)	• FLIGHT PROVEN • RELIABLE • REUSEABLE • DEVELOPED • SELF CONTAINED	• WEIGHT AND VOLUME INCREASE ESSENTIALLY LINEARLY WITH REQUIRED ENERGY (55–60 WATT-HOURS PER POUND AND 3–5 WATT HOURS PER CUBIC INCH) • RECHARGE PROCEDURE IS COMPLEX WHEN LARGE NUMBER OF BATTERIES ARE INVOLVED. • WET-LIFE LIMITED (1 YEAR OR LESS)
Ni-Cd BATTERIES	• FLIGHT PROVEN • RELIABLE • REUSEABLE • DEVELOPED • SELF CONTAINED	• WEIGHT AND VOLUME INCREASE ESSENTIALLY LINEARLY WITH REQUIRED ENERGY (10–12 WATT-HOURS PER POUND AND 1–1.5 WATT-HOURS PER CUBIC INCH). • RECHARGE PROCEDURE IS COMPLEX WHEN LARGE NUMBER OF BATTERIES ARE INVOLVED.
H_2–O_2 FUEL CELLS	• CONCEPT FLIGHT PROVEN • RELIABLE • REUSEABLE • LONG OPERATING LIFE – CURRENT LIFE 3000 HOURS, DESIGN GOAL 10,000 HOURS • HIGH ENERGY DENSITY (400–450 WATT-HOURS PER POUND, INCLUDING TANKAGE FOR ORBITER ENERGY AND POWER RANGE)	• HIGH PURITY CRYOGENIC REACTANTS REQUIRE TANKAGE SEPARATE FROM PROPULSION REACTANTS • LIMITED TO DC GENERATION. • MATRIX TYPE FUEL CELLS REQUIRE FLIGHT QUALIFICATION.
TURBOALTERNATOR (H_2–O_2 FUEL)	• LIGHT WEIGHT EQUIPMENT • FUEL SOURCE CAN BE COMMON WITH MAIN PROPULSION TANKS • OPTION OF AC OR DC GENERATION • OPTION OF HIGH OR LOW VOLTAGE GENERATION	• HIGH FUEL CONSUMPTION (2.5–4 POUNDS PER KWH) • COMPLEX CONTROL SYSTEM. • TURBINE EFFICIENCY IS POWER SENSITIVE. • TURBINE EFFICIENCY IS ALTITUDE SENSITIVE. • EXHAUST GAS CAN CAUSE VEHICLE ATTITUDE CHANGE • SHORT DEMONSTRATED OPERATING LIFE (250 HOURS) • DEVELOPMENT REQUIRED.
TURBOALTERNATOR (MONOPROPELLANT HYDRAZINE WITH CATALYST BED)	• LIGHT WEIGHT EQUIPMENT • CONTROL LESS COMPLEX THAN H_2–O_2 UNIT • OPTION OF AC OR DC GENERATION • OPTION OF HIGH OR LOW VOLTAGE GENERATION	• HIGH FUEL CONSUMPTION (5–10 POUNDS PER KWH). • SEPARATE FUEL TANK REQUIRED. • TURBINE EFFICIENCY IS POWER SENSITIVE • TURBINE EFFICIENCY IS ALTITUDE SENSITIVE • EXHAUST GAS CAN CAUSE VEHICLE ATTITUDE CHANGE • SHORT DEMONSTRATED OPERATING LIFE (250 HOURS) • DEVELOPMENT REQUIRED.

4.5 Environmental Control System - The function of the Environmental Control System (ECS) is to provide a habitable shirtsleeve environment in the vehicles. The orbiter requires an ECS that will provide this environment for two men for a flight as long as seven days. The carrier requires an ECS that will provide the desired environment for a brief launch flight or a long ferry flight. The systems to provide these functions are discussed below. The functional concepts and baseline characteristics are given in Tables 4-35 and 4-36 respectively and a weight summary is given in Table 4-37.

4.5.1 Carrier - The Carrier ECS must provide the atmosphere supply, and cabin and equipment temperature control. The ECS consists of four subsystems: the Oxygen Supply, the Heat Transport Circuit, the Air Cycle, and the Hydraulic Cooling subsystems. These subsystems, with the exception of the Hydraulic Cooling Subsystem, are shown schematically in Figure 4-80. The operation of each subsystem is summarized in the succeeding paragraphs.

Oxygen Supply - The oxygen supply subsystem provides an emergency supply of oxygen. In normal flight, the cabin will be pressurized with air to the equivalent of an 8000 ft. altitude and additional oxygen will not be necessary. If the cabin pressure is lost, then the oxygen supply will provide oxygen to the crew until the vehicle is brought down to an altitude where supplementary oxygen is not necessary. This system is similar to aircraft systems and is used for its simplicity and low cost.

The Heat-Transport Circuit - The system uses redundant coolant loops, and dual passage coldplates for the thermal control of electronic equipment. The secondary loop is used if a failure occurs in the primary loop. Redundant coolant pumps in each loop circulate the heat transfer coolant. Waste heat is rejected by an air cycle refrigeration package during subsonic cruise flight or during ferry flights. Prior to launch the air cycle machine is powered by a ground supply of high pressure air. During the boost phase of flight, heat dissipated by the electrical equipment is absorbed by equipment, coolant fluid, and circuit component temperature increases. Subsequent to boost the air cycle is powered with bleed air from the jet engine compressor.

Air Cycle - The air cycle subsystem serves a dual function, providing cabin air conditioning and pressurization, and providing cooling for the Heat Transport Circuit. Jet engine compressor bleed air is cooled by heat exchange with ram air, is compressed, again is cooled by ram air and then is further cooled by expansion

Table 4-35

ENVIRONMENTAL CONTROL SYSTEM FUNCTIONAL CONCEPT

MISSION PHASE	ORBITER	CARRIER
PRELAUNCH	SYSTEM COOLING BY AIR CYCLE – GROUND SUPPLY HIGH PRESSURE AIR.	SYSTEM COOLING BY AIR CYCLE – GROUND SUPPLY HIGH PRESSURE AIR.
ASCENT	SYSTEM COOLING BY WATER BOILER.	HEAT SINK IN COMPONENTS, COOLANT CIRCUIT.
ORBIT	SYSTEM COOLING BY SPACE RADIATOR – CRYOGENIC GAS SUPPLIES – CO_2 ABSORPTION BY LiOH – CREW WATER FROM FUEL CELLS.	NOT APPLICABLE.
DESCENT/LANDING	NOT APPLICABLE.	SYSTEM COOLING BY AIR CYCLE – ENGINE BLEED SUPPLIES HIGH PRESSURE AIR – PILOTS IN PARTIAL PRESSURE SUITS.
ENTRY	SYSTEM COOLING BY WATER BOILER.	NOT APPLICABLE.
CRUISE/LANDING	SYSTEM COOLING BY AIR CYCLE – ENGINE BLEED SUPPLIES HIGH PRESSURE AIR.	NOT APPLICABLE.

Table 4-36

ENVIRONMENTAL CONTROL SYSTEM CHARACTERISTICS

ORBITER REQUIREMENTS • SHIRTSLEEVE ENVIRONMENT FOR TWO MAN CREW • SEVEN DAYS IN ORBIT • CAPABLE OF SUBSONIC FERRY FLIGHT • DISSIPATE 5 + KW EQUIPMENT WASTE HEAT	ORBITER BASELINE SYSTEM • SEA LEVEL ATMOSPHERE – NO PRESSURE SUITS • STORE GASES AS SUPERCRITICAL CRYOGEN • CONTROL CO_2 WITH LITHIUM HYDROXIDE • CONTROL EQUIPMENT TEMPERATURE WITH LIQUID COOLANT CIRCUIT AND COLDPLATES • AIR CYCLE COOLING PACKAGE FOR FERRY/CRUISE • DISSIPATE WASTE HEAT WITH SPACE RADIATOR AND WATER BOILER • SUPPLY DRINKING WATER FROM FUEL CELLS • DUMP URINE – STORE FECAL WASTE • HYDRAULIC COOLING BY WATER BOILER
CARRIER REQUIREMENTS • MAXIMUM PRESSURE ALTITUDE OF 8000 FT • CAPABLE OF SUBSONIC FERRY FLIGHTS • DISSIPATE 5 + KW EQUIPMENT WASTE HEAT	CARRIER BASELINE SYSTEM • CABIN TEMPERATURE CONTROLLED BY AIR CYCLE MACHINE • CONTROL EQUIPMENT TEMPERATURE WITH LIQUID COOLANT CIRCUIT AND COLDPLATE • DISSIPATE WASTE HEAT WITH AIR-CYCLE MACHINE • HIGH PRESSURE EMERGENCY OXYGEN – FACE MASKS – PARTIAL PRESSURE SUIT • HYDRAULIC COOLING BY WATER BOILER OR RAM AIR

Table 4-37

ENVIRONMENTAL CONTROL SYSTEM WEIGHT SUMMARY

ORBITER ENVIRONMENTAL CONTROL SYSTEM SUBSYSTEM	SALIENT FEATURES	WT (LB)
GAS PROCESSING	CO_2 ABSORPTION WITH LiOH	52
GAS SUPPLY & CONTROL	SUPERCRITICAL CRYOGENIC STORAGE	353
HEAT TRANSPORT	SPACE RADIATOR (680 LB), WATER BOILER (110 LB) AIR CYCLE COOLING PACKAGE (50 LB)	1022
CREW WATER SUPPLY	TANK ONLY - WATER SUPPLIED BY FUEL CELL REACTANTS	11
HYDRAULIC SYSTEM COOLING	WATER BOILER COOLING	409
MISC CIRCUITRY, LINES, FTGS.		90
TOTAL ENVIRONMENTAL CONTROL SYSTEM		1937
CARRIER ENVIRONMENTAL CONTROL SYSTEM SUBSYSTEM	SALIENT FEATURES	
OXYGEN SUPPLY	HIGH PRESSURE SUPPLY - MASKS AND PARTIAL PRESSURE SUIT FOR EMERGENCY	25
HEAT TRANSPORT	HEAT SINK UNTIL AIR CYCLE OPERABLE	176
AIR CYCLE PACKAGE	POWERED BY ENGINE BLEED AIR OR GROUND SUPPLY	50
HYDRAULIC SYSTEM COOLING	WATER BOILER COOLING, RAM AIR	176
TOTAL ENVIRONMENTAL CONTROL SYSTEM		427

CARRIER ENVIRONMENTAL CONTROL SYSTEM SCHEMATIC

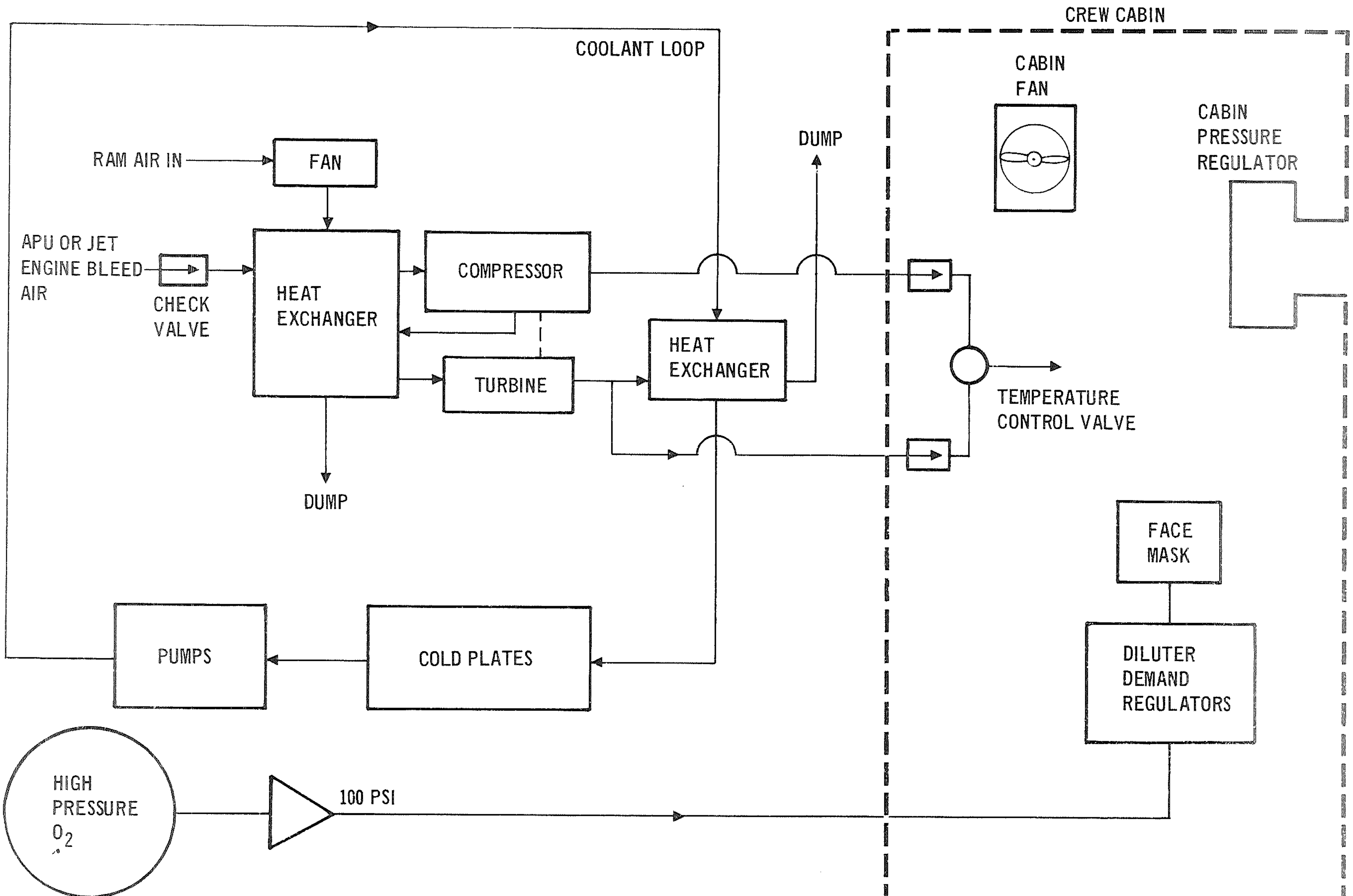


Figure 4-80

in a turbine that drives the compressor. The cold air removes heat from the coolant circuit and then is mixed with hot air from the compressor to control the cabin temperature. Ground and launch operation of this subsystem is described in the preceeding paragraph.

Hydraulic Cooling - This subsystem prevents overheating of the fluid in the hydraulic subsystem which powers the aerodynamic control surfaces. Heat is removed by a water boiler that exchanges heat with the hydraulic fluid to prevent temperatures in excess of 212°F. This subsystem is completely independent of the heat transport circuit and contains a total of 61 lbs. of water for cooling in hypersonic flight. For subsonic ferry flight a ram air heat exchanger is provided.

4.5.2 Orbiter ECS - The functions to be provided by the ECS are atmosphere supply, atmosphere processing, cabin and equipment temperature control, water supply and waste management. The ECS consists of the Gas Supply and Control, the Gas Processing, the Heat Transport circuit, the Water and Waste Management, and Hydraulic Cooling subsystems. These subsystems are briefly described below and, with exception of the Hydraulic Cooling Subsystem, are shown schematically in Figure 4-81.

Gas Supply and Control - This subsystem supplies the oxygen and nitrogen for breathing and cabin pressurization. Redundant supercritical cryogenic tanks in the equipment bay contain a total of 66 lbs. of oxygen and 148 lbs. of nitrogen. The cabin pressure is maintained at 14.7 psia by a cabin pressure regulator which is supplied from either the nitrogen or the oxygen supply. Initially, if the oxygen partial pressure is below the upper limit (3.0 psia), the solenoid valves in the nitrogen supply remain closed and only oxygen is added to the cabin. When the oxygen partial pressure reached 3.0 psia, the controller opens a solenoid valve (redundant). The nitrogen which is regulated to 150 psig, then backpressures a check valve in the 100 psig oxygen supply line, closing it, so that only nitrogen is supplied. When the oxygen partial pressure drops to the lower limit (2.7 psia) the nitrogen valves are closed and oxygen is again supplied. System partial and total pressures were selected to match postulated space station pressures.

Gas Processing - The system provides crew ventilation, atmosphere constituent control and atmosphere cooling. Cabin fans and gas inflow and outflow distribution ducts are provided at selected locations to circulate the cabin atmosphere. The cabin atmosphere gases are circulated throught system components to filter, remove the carbon dioxide by reaction with LiOH, remove odors and trace contaminants with

ORBITER ENVIRONMENTAL CONTROL SYSTEM SCHEMATIC

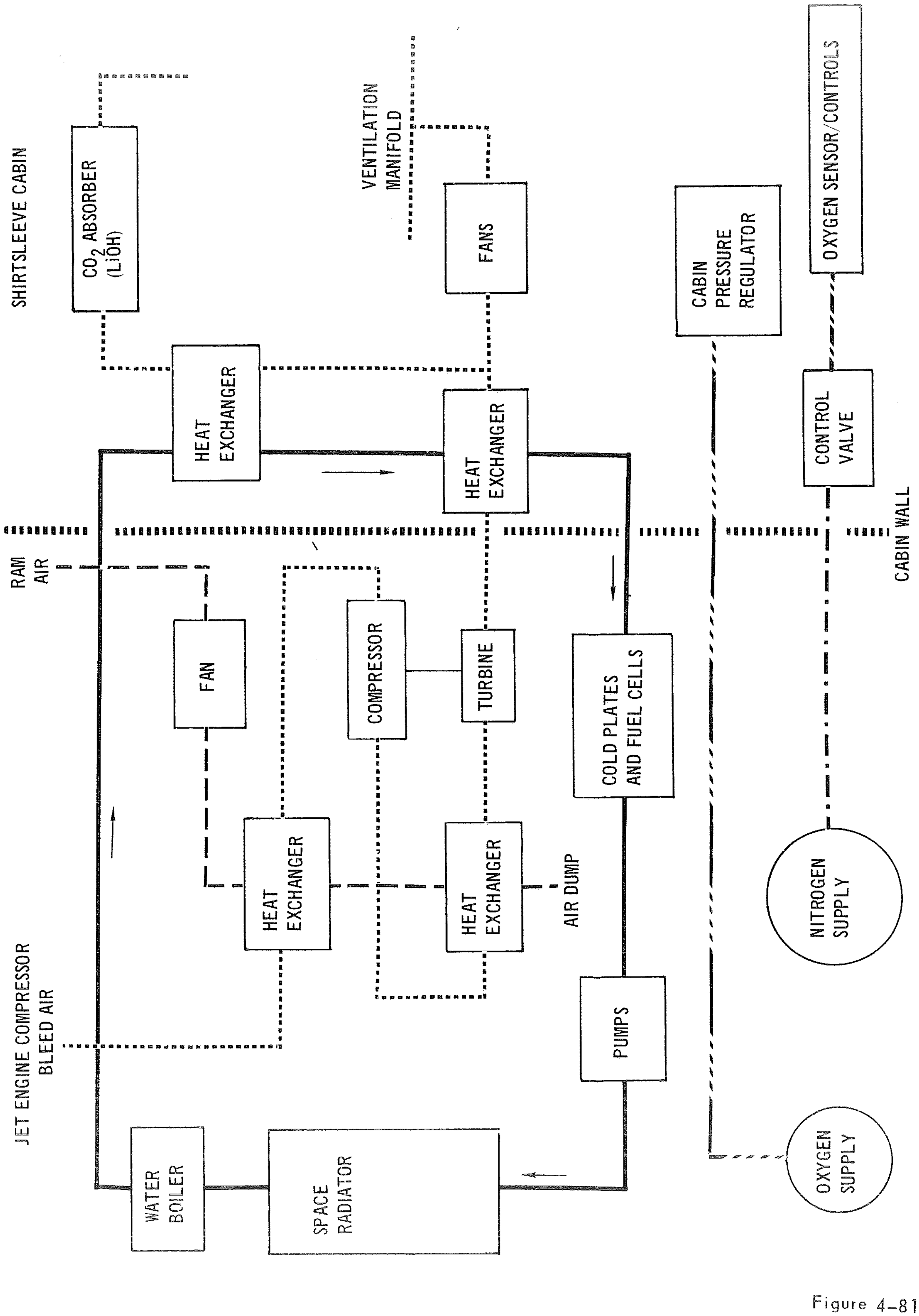


Figure 4-81

activated charcoal, and cool and control the relative humidity by a heat exchanger.

The Heat-Transport Circuit - The system uses redundant coolant loops and dual coldplates for the thermal control of electronic equipment, a space radiator, and a water boiler for heat dissipation. The secondary loop is used if a failure occurs in the primary loop. Redundant coolant pumps in each loop circulate the heat transfer coolant. Waste heat is rejected by the orbiter radiator and water boiler in orbit and by the water boiler during atmospheric entry. An air cycle refrigeration package removes waste heat during subsonic cruise flight or during ferry flights.

Water and Waste Management - The subsystem provides: drinking water to the crew; a source of water for heat dissipation by evaporation; storage and disposal of condensate from the cabin heat exchanger and fuel cell product water; collection, storage or disposal of waste materials generated during the mission. Because of the short flight mission, water condensed in the cabin heat exchanger/ water separator does not supplement the drinkable water supply, but is routed directly to the water boilers. The water supplied by the fuel cells is temporarily stored in a bladder type tank until it is used for drinking or heat dissipation. Urine is collected in GFE urine receivers and then dumped overboard. The fecal wastes, meal time food residue, and expendables are collected in sealable bags. The bags are treated with a bactericide and then stored in a wast storage container.

Hydraulic Cooling - This subsystem prevents overheating of the fluid in the hydraulic system which powers the aerodynamic control surfaces. Heat is removed by a water boiler that exchanges heat with the hydraulic fluid to prevent temperature in excess of 212°F. This subsystem provides a total of 306 lbs. of water for cooling the hydraulic fluid from a supply tank that is independent of other subsystems.

5.0 WEIGHT ANALYSIS

Weight estimating techniques for the LRC vehicle are based on detail analysis of major components, off-the-shelf hardware and semi-empirical weight equations. Factors are employed in the analysis to assure that realistic rather than optimistic weights are quoted.

In estimating weight, non-optimum allowances must be made for excess weight that is added to the component optimum design. The nonoptimum factor is the ratio of the component total weight to its optimum weight.

As an example, items that contribute to nonoptimum structural weight are joints between structural members and the use of nontapered and standard gage sheet metal.

An example of the excess weight added to the structure by nontapered, standard gage sheet metal is shown in Figure 5-1. When the structure is made from nontapered sheets, the required thickness curve is approximated by "steps", as shown by the dashed lines. The non-optimum weight is obviously reduced by increasing the number of steps.

Weight estimates based on empirically derived equations include allowance for non-optimum effects. When detail analysis of a structural component is performed more refined allowances are made for such factors as material tolerance, splices, and overlap.

In the preliminary design phase, time does not permit a detail analysis of all structural components. Therefore, empirically derived weight estimates for various components are combined with detail weight estimates of the major structural members. Using this method, the total weight of body structure can be written:

$$W_{TOTAL} = K[W_{OPT.}] + W_{SEC.}$$

where: K = overall non-optimum factor

W_{OPT} = optimum structure weight

W_{SEC} = remaining structure weight based on empirical weight data

EFFECT OF NONTAPERED SHEET

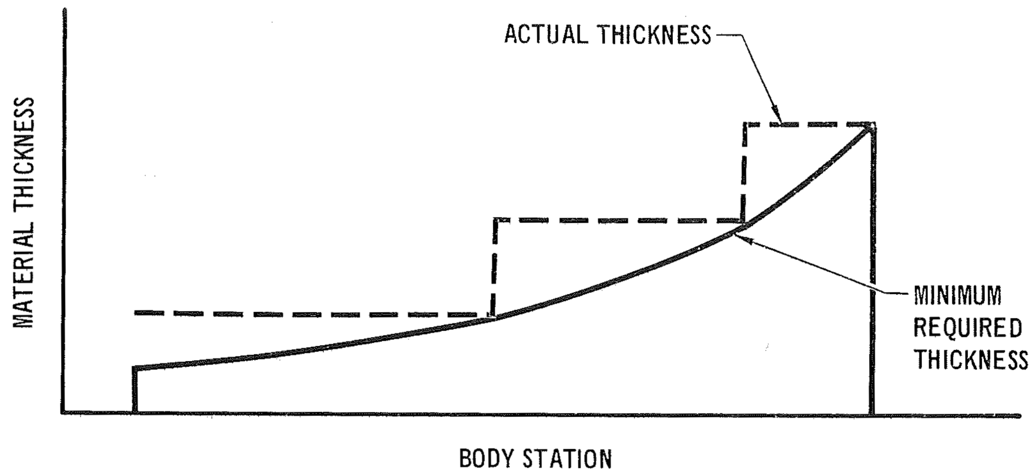


Figure 5–1

The non-optimum factor, K, can be further separated as follows:

$$K = K_1 + K_2$$

where: K_1 = non-optimum factor for material tolerance and overlap plus minimum gage material thickness.

K_2 = non-optimum factor for miscellaneous structural components, such as fasteners, sealing compound and complex joints.

In the analysis of the structural shell of the orbiter 15 percent was added to the skin weight for material tolerance, thickness and overlap, resulting in a factor of 1.15 for K_1. A value of 1.15 was used for K_2 to allow for miscellaneous structural components and complex joints. A similar analysis was performed on the carrier and it resulted in a $K_1 = 1.10$ and a $K_2 = 1.10$.

5.1 <u>Weight Derivations</u> - The weight analysis for this study was conducted as a point design. Significant inputs were received and utilized from strength, thermo and aerodynamic disciplines. In cases where the design paralleled existing hardware data, weight estimation techniques and empirical equations were used. The primary example of this was the use of aircraft wing weight estimation equations for the carrier wing and tail and the orbiter tail section.

<u>Subsystem Weight Estimates</u> - Subsystem weight estimates for the LRC preliminary design spacecraft are presented in this section. Component weights are coded in accordance with MIL-38310.

<u>Aerodynamic Surfaces</u> - Aerodynamic surfaces on the orbiter include tip fins (plus flaps), elevons, and a dorsal fin (plus rudder). The carrier's surfaces are comprised of a dorsal fin (plus rudder) along with delta wings (plus flaps). The following components were considered in the weight estimates:

- Tail Sections
  - Torque box
  - Body attachment
  - Leading edge
  - Trailing edge
  - Flap or rudder provisions
  - Flaps or rudders

- Delta Wing
  - Torque box
  - Carry through structure
  - Leading edge
  - Trailing edge
  - Engine provisions
  - Fuel provisions
  - Landing gear provisions
  - Expanded root provisions
  - Control surface provisions

Table 5-1 presents the weight estimates for the aerodynamic surfaces for each spacecraft.

The weight of the aerodynamic surfaces is estimated using semi-empirical methods which estimates the various weight penalties for the various functions. McAir has correlated the aerodynamic surface weights. Some 50 airplane surfaces from hi load factor fighters to low load factor transports, high and low aspect ratio wings, etc. have been correlated using some 68 parameters in 30 equations for 13 special design features. The results of this correlation are presented in Figure 5-2.

The equations used in the LRC analysis including material selection, geometry, and structural concept conversion factors are:

1. Torque box

Element	Exposed Surface(s)	Carrythru (Ct)
Bending Shell	$\gamma_1 = 2\ C_B\ S_{TBe}$	$\gamma_2 = 2\ C_B\ S_{TB\ CT}$
Bending	$\gamma_3 = \frac{\rho\ L_w\ k}{3\ F_a\ .8t_r} \left(\frac{b_e}{2\cos\theta}\right)^2$	$\gamma_4 = \frac{b_e\ b_{ct}\ L_w\ (2\lambda + 1)}{6\ F_a\ .8t_r\ (\lambda + 1)}$
Shear Shell	$\gamma_5 = 2\rho\ (C_S) \left[\frac{.8t_r\ (1 + m)}{2}\right]^2 \left(\frac{b_e}{\cos\theta}\right)$	$\gamma_6 = 2\rho\ (C_S)\ (.8t_r)^2\ b_{ct}$
Shear	$\gamma_7 = \frac{\rho L_w\ (2\lambda + 1)}{\tau_a\ 6\ (\lambda + 1)} \left(\frac{b_e}{\cos\theta}\right)$	
Ribs	$\gamma_8 = C_R\ t_r\ (1 + m)\ S_e \times 10^{-2}$ (Note: S_e in sq. ft.)	$\gamma_9 = C_R \times 2t_r\ S_{ct} \times 10^{-2}$ (Note: S_{ct} in sq. ft.)

Table 5-1

AERODYNAMIC SURFACE WEIGHTS

Carrier		Orbiter	
Dorsal Fin	(6860)	Dorsal Fin	(1770)
Torque Box	2830	Torque Box	860
Attach. to Body	1180	Attach. to Body	230
Leading Edge	1340	Leading Edge	200
Rudder Provisions	300	Trailing Edge	70
Rudder	1210	Rudder Provisions	80
		Rudder	330
Delta Wing	(83430)	Tip Fins	(5320)
Torque Box	28630	Torque Box	2360
Carry Through	28690	Attach to Body	1350
Leading Edge	6400	Leading Edge	420
Trailing Edge	3110	Trailing Edge	60
Engine Provisions	8650	Flap Provisions	240
Fuel Provisions	270	Flap	890
Main Landing Gear Prov.	340	Elevons	(2060)
Expanded Root Prov.	450		
Control Surface Prov.	1840		
Elevons	5050		

WING TORQUE BOX WEIGHT

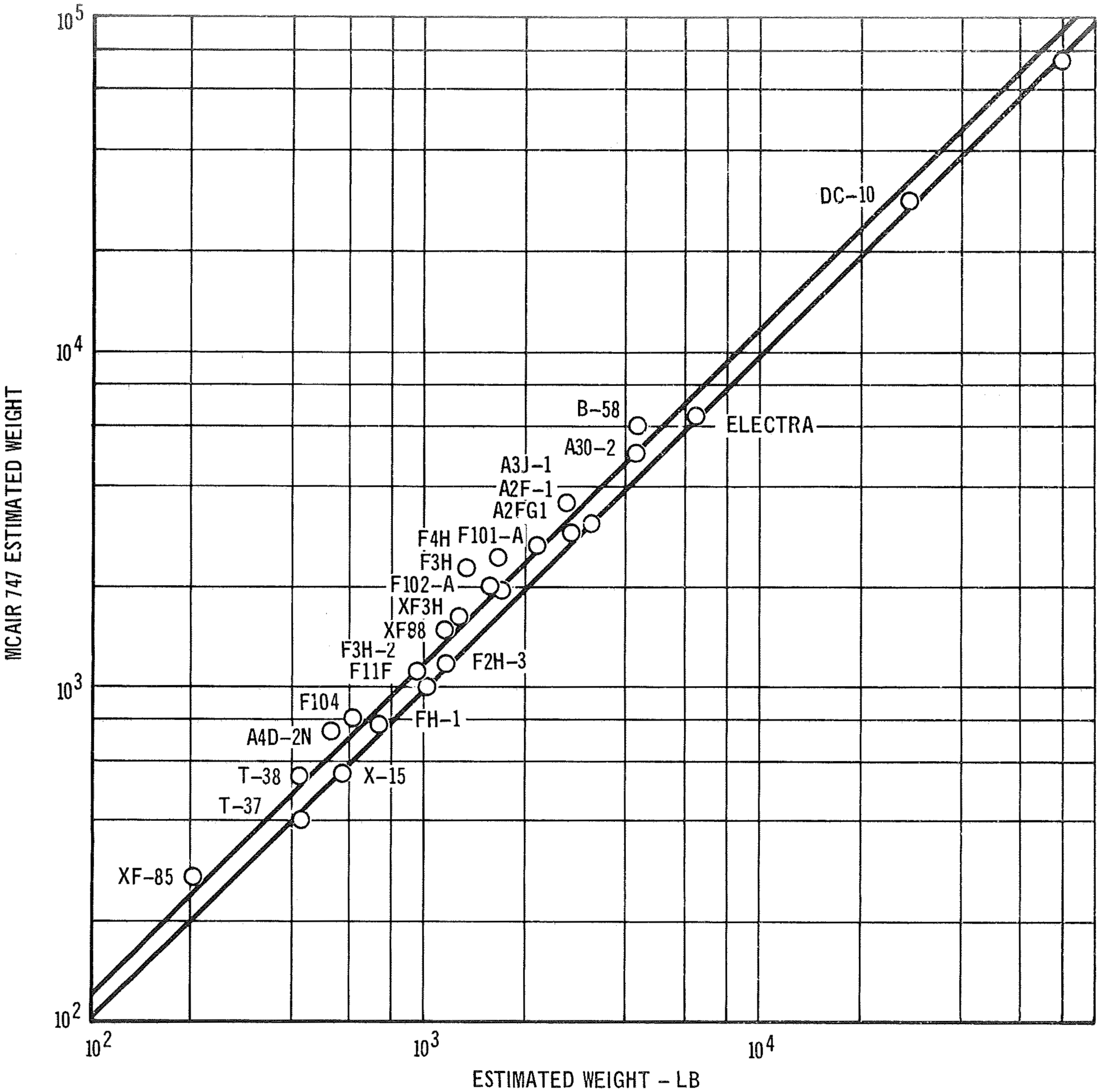


Figure 5-2

Joints $\gamma_{10} = (\gamma_1 + \gamma_3 + \gamma_5 + \gamma_7 + \gamma_8).10$ $\gamma_{11} = (\gamma_2 + \gamma_4 + \gamma_6 + \gamma_9).10$

Std Gages $\gamma_{12} + .14\ S_{TBe}$ $\gamma_{13} = .14\ S_{TBCT}$

where:

C_B = unit weight bending shell psf

S_{TB} = area torque box (sq. ft.)

ρ = density (pci)

L_w = load on exposed surface (lbs.)

k = integration factor for planform (λ) and thickness (m) taper ratio (dimensionless)

λ = tip chord/root chord

m = tip thickness/root thickness

b = span (inches)

F_a = artificial compression stress (psi)

t_r = root thickness (inches)

cos θ = cosine of sweep angle nominally measured at 50% chord

C_S = thickness coefficient of shear shell (dimensionless)

use .003 for aluminum

use .002 for Titanium and steel

ttof shear material shell = $C_S\ .8t_r$ (inches)

τ_a = artificial shear stress (use .31 ft_{tu})

C_r = unit weight ribs (psf)

use .7 for aluminum r.t

for other materials

use $\dfrac{(\gamma_5 + \gamma_9)}{(\gamma_5 + \gamma_7)\text{Alum r.t}}\ .7$

S_e = expose wing area = sq. ft. (Note - Not TB area)

S_{CT} = Carry thru wing area - sq. ft.

2. Engine provisions

$= 3.85\ (T)^{.6}\ (N_1) + 3.24\ (L_1D)^{.9}\ (N_1)$

3. Fuel System Provisions
   $= K_1 \ (G/N_2)^{.75} \ (N_2)$

4. Main Landing Gear Provisions
   $= .43 \ (q)^{.3} \ (S_1) + .077 \ (FgL_2)^{.9}$

5. Expanded Root Thickness
   $= .0048 \ (Wghzb/\cos\theta)^{.9} (ta/t_r)^{.9} (b_r/\cos\theta)^{.5}$

6. Control Surface Provisions
   $= 1.1 \ (W_{ft})^{.8}$

7. Leading Edge
   $3.38 \ (Wghz/sg)^{.3} \ (Sb)$

8. Trailing Edge Surfaces
   (Basic Shell) $= 1.75(S_{cs}) + 1.30 \ (S_{hc}) + 1.5(S_{hs}) \quad N_3$

   (Drive Ribs and Chrod Wise Bending) $= K_3(PM/t_m)^{.75} \ (C_m)N3$

   (Hinges and Front Beam and Supports) $= .40(PM)^{.2} \ (b_a)N_3 + K_4$ (Leading Edge surface) (wt.)

9. Trailing Edge
   $1.87 \ (W_g \ n_z/S_g)^{.2} (S_{te})$

10. Leading Edge Surfaces
   (Flap Structure) $= 13.12 \ (W_g \ n_z/S_g)^{.33} \ (b_a)(C_m)^{1.2}$

where:

T = thrust/engine = lb x 10^{-3}

N_1 = Number of engines

L_1 = Length, engine compartment - Ft.

D = Diameter, Engine Compartment - ft.

K_1 = Constant, fuel storage

G = Wing fuel capacity - gallons

N_2 = Number of fuel tanks

q = Maximum dynamic pressure - psf

S_1 = Area, main landing gear door - ft^2

Fg = Load, Maximum ultimate vertical - lbs x 10^{-3}

L_2 = Length, main gear extended - in.

h_z = Load factor, ultimate vertical

W_g = Landing weight - lbs. x 10^{-3}

$b/\cos\theta$ = Span - ft.

t_a = root thickness (actual root) - Ft.

t_r = Root thickness - ft.

W_{rt} = Weight trailing edge flap - lbs.

S_g = Area - ft^2

Sle = Area of leading edge - ft^2

S_{cs} = Area of surface - ft^2

S_{hc} = area, Honeycomb Structure - ft^2

S_{hs} = Area, half shell structure - ft^2

N_3 = Number of control surfaces

K_3 = Constant, control surface actuation point

M = Hinge Moment - in. lb x 10^{-3}

t_m = Hinge line thickness

C_m = Chord - ft

b_a = Span, hinge line - ft

K_4 = constant

Ste = Area trailing edge

Body Structure - This section accounts for all the primary and secondary body structure. The structural design concept of the LRC vehicles is described in Section 4.1.

The carrier and the orbiter utilize an integral tank concept where the propellant tank walls are the primary structural members. The basic body structure of the carrier is composed of an aluminum skin with integral longitudinal stiffeners and wing flanges. Circumferential frames are added to support the thermal protection system. This structural concept is illustrated in Figure 4-3.

The orbiter also uses an aluminum ring stiffened shell with longitudinal stiffeners. However, a small amount of titanium was used on the frame caps and webs where the temperature requirement exceeds the aluminum capability. Figure 4-8 illustrates this concept.

Cabin structure, which is the same for both stages, is based on a semi-empirical weight equation for the weight prediction of aircraft cockpit weight (MAC Report 747, page 5.1).

$$W_c = 1.54\ (V_c)^{.78}\ (1 + P_c)^{.34}$$

Parameters

V_c = cabin volume - cu. ft. = 180
P_c = cabin pressure - psi ult. = 22
W_c = cabin structure weight - lb. = 260

Weight prediction for the orbiter cabin to payload tunnel is based on hardware data from the Gemini B-MOL program. The unit weight of the hardware is 0.79 lb/in of tunnel length for a 32 inch diameter tunnel. The LRC tunnel is 48 inches in diameter and 400 inches long. The weight prediction is scaled as follows:

$$\text{Tunnel Weight} = \frac{48 \text{ in. dia.}}{32}\ (.079 \text{ lb/in})(400 \text{ in}) = 470 \text{ lb.}$$

The carrier has three bulkheads which form the integral LOX and LH_2 tanks. The weights of these bulkheads are based on a correlation with Atlas, Titan III, Saturn II and Saturn IV bulkheads as shown in Figure 5-3. The more important parameters used were radius and design pressure. A weight summary for the three bulkheads is as follows:

BULKHEAD WEIGHT

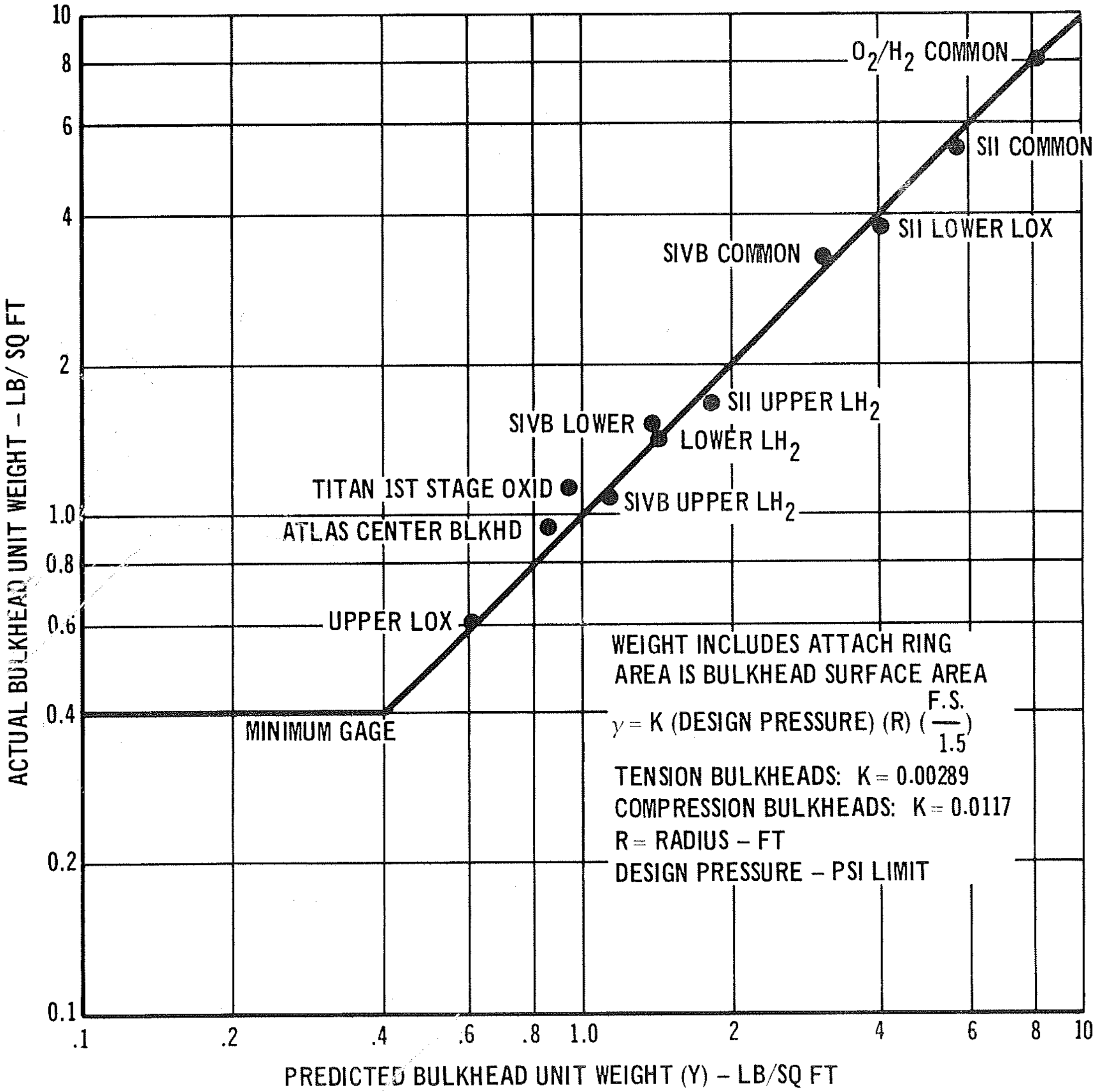


Figure 5–3

- o upper lox bulkhead 1000 sq ft @ 0.6 lb/sq ft - 600 lb.
- o common LOX/LH_2 bulkhead 1589 sq ft @ 8.2 lb/sq ft = 13,030 lb.
- o lower LH_2 bulkhead 1634 sq ft @ 1.34 lb/sq ft = 2190 lb.

The seemingly outsized weight of the common bulkhead is due to the large head pressure created by the LOX. The baffles in the propellant tanks are .020 aluminum webs spaced at 24 inch intervals. It was assumed that the baffles would have numerous holes for propellant passage equal to 35% of the net area. This total weight is 9690 lbs.

The frames are spaced at 20 inch intervals. They have titanium outer caps and webs with aluminum inner caps. A cross section of a typical frame is also shown in Figure 4-3. The total weight of the frame is 13,510 lbs. which is .68 lb/sq ft of body surface area.

The skin stiffener combination has an equivalent thickness ranging from .11 inches near the nose to .22 inches 70 ft aft and then decreasing to .14 at the tail. The total weight of the skin and stiffeners is 50730 lb which is 2.56 lb/sq ft of body surface area.

The cryogenic insulation used on the hydrogen tank walls is polyurethane foam. It is applied at the rate of .395 lb/sq ft. A typical cross section is shown in Figure 5-4.

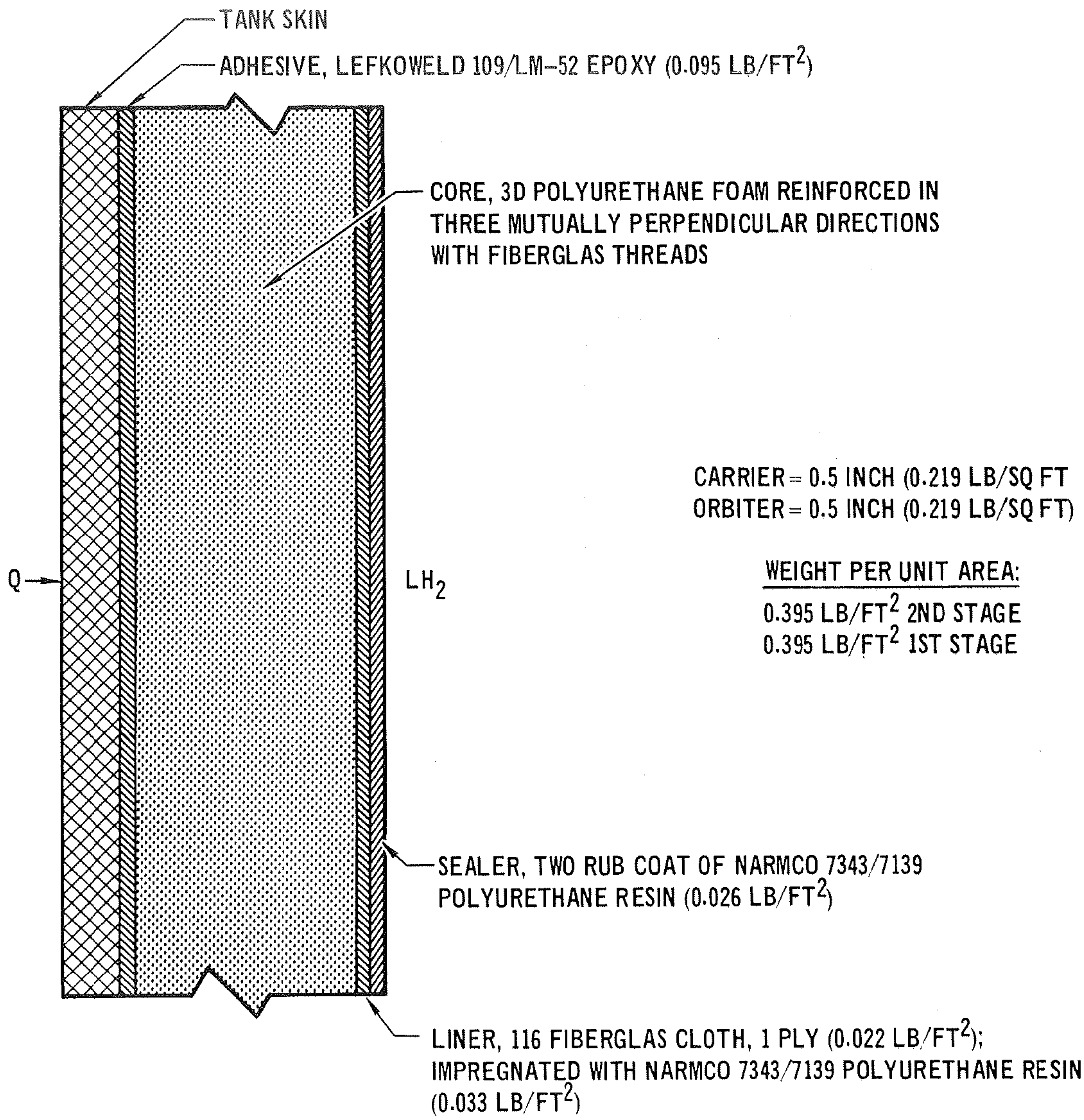
A center vertical web runs the length of body propellant tanks. The equivalent thickness in the LO_2 tank is .14 inches and .12 inches in the LH_2 tank. The weights are 3450 and 7450 pounds respectively.

The total weight of structure and tank insulation is 112990 lbs. This is equivalent to 5.89 lb/ft^2 over the body surface area.

The orbiter has an average material thickness which ranges from .025 inches at both the nose and tail to its thickness (.110 inches) at body station 490 inches. The average unit weight of the skin is 1.64 lb/sq ft with a surface area of 9780 sq ft. This skin forms all exterior body moldlines as well as the inner tank walls.

The technique used to derive the weights of the skin-stringer structures for the outer moldine and tank walls was as follows. Strength analysis determined the equivalent thickness required as a function of body station. A plot of the values used is shown in Figure 4-33 of Section 4.1. Several section cuts were then taken at key locations as determined from the plot. A sample section cut is shown in Figure 5-5 for body station 720 inches. The weight per linear inch was

SELECTED TANK INSULATION



CARRIER = 0.5 INCH (0.219 LB/SQ FT
ORBITER = 0.5 INCH (0.219 LB/SQ FT)

WEIGHT PER UNIT AREA:

0.395 LB/FT^2 2ND STAGE
0.395 LB/FT^2 1ST STAGE

Figure 5-4

TYPICAL BODY CROSS SECTION

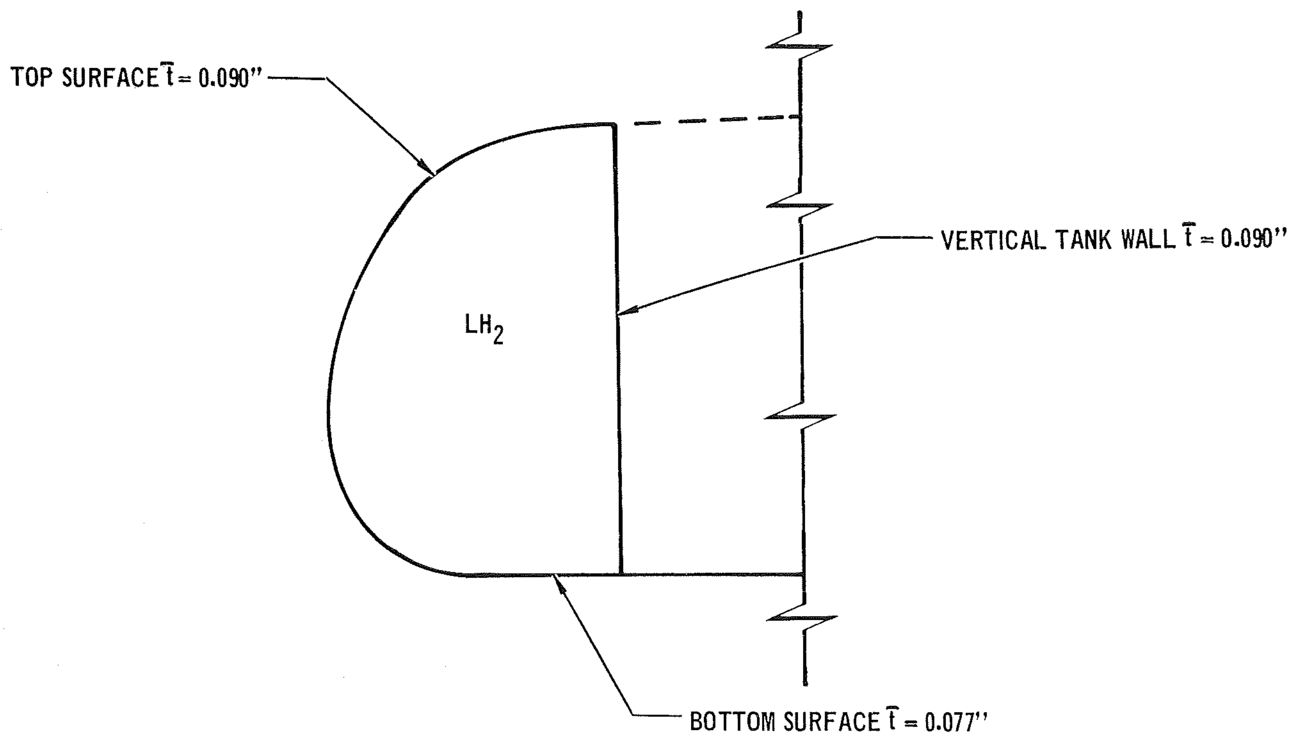


Figure 5–5

calculated for each section and integrated over the length of the vehicle to obtain the design estimated weight. Non-optimums were applied to this weight to obtain the total of 16070 lbs.

The frames are spaced at intervals between 12 and 15 inches around the outer surface depending on the pressure. Due to temperature requirements, the outer cap and web are titanium with an aluminum inner cap. A typical frame is shown in Figure 4-8. The frame weight based on body surface area, is .74 lb/sq ft or 7820 lbs/vehicle.

Two major bulkheads form the forward and aft closures for the liquid oxygen tank. The forward bulkhead is a flat plate with a waffle pattern stiffener. This construction is illustrated in Figure 5-6. Due to the large pressure head the rear LOX bulkhead is constructed quite differently. Its basic construction is shown in Figure 5-7. The unit weight of the bulkhead is 12.0 lb/sq ft for a total weight of 3600 lbs. A summary of the bulkhead weights is presented below:

- o rear LOX bulkhead - 500 sq ft @ 7.2 lb/sq ft = 3600 lb
- o forward LOX bulkhead - 400 sq ft @ 1.05 lb/sq ft = 420 lb
- o LH_2 bulkheads - 156 sq ft @ 3.8 lb/sq ft = 680 lb

To help the lower bulkhead distribute the pressure loads, tie rods are added, connecting to the upper bulkhead. A total of 75 are used having .49 sq inch cross section area. The total weight including attachment provisions is 1040 lbs.

The pressure loads inside the LH_2 tanks are taken out by biaxially loaded tension webs which also can act as baffles. They are spaced at 15 inch intervals and are assumed to have 35% of the surface area removed for propellant passage. The baffles in the LOX tank are .040 inch aluminum and weigh 1380 lbs. The hydrogen tank baffles are .026 inches thick at the forward end increasing to .029 in the aft end. Their weight is 3360 lb.

A small hydrogen tank is located in the rear of the vehicle and weighs 1290 lbs complete. This includes walls, center web and baffles. The end domes are .032 inch aluminum while the side wall tapers from .049 to .060 inches from forward to rear.

Polyurethane foam is used to insulate all LH_2 tank walls and the rear LOX bulkhead. It is applied at the rate of .395 lb/ft^2 for a total weight of 3620 lbs.

FORWARD BULKHEAD LOX TANK

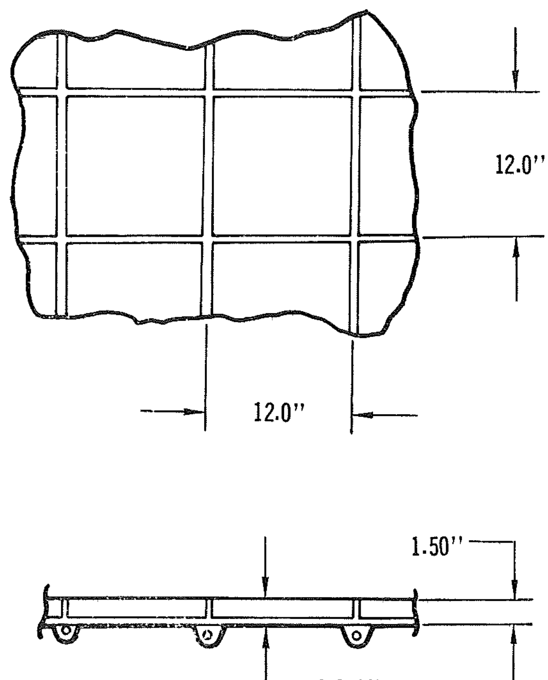


Figure 5-6

AFT BULKHEAD – LOX TANK

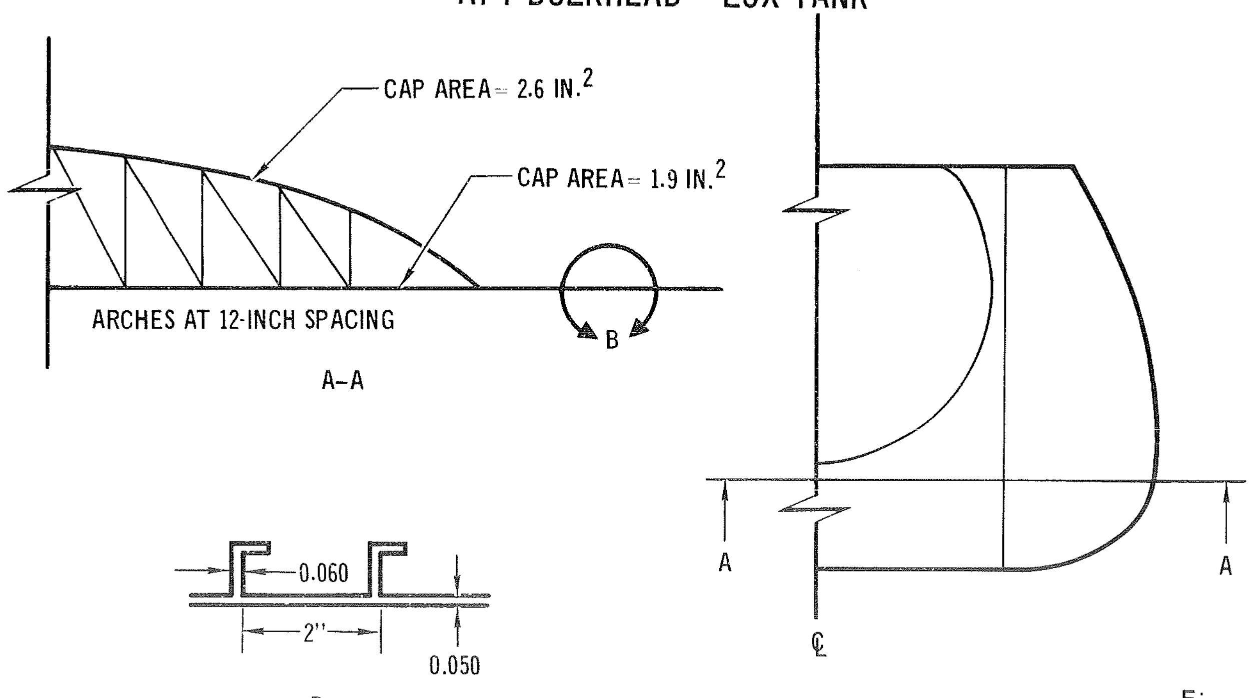


Figure 5-7

The total structure weight (excluding aero surfaces) is 46240 lbs. The unit weight including cryogenic tank insulation is 4.72 lb/sq ft.

Landing Gear - Landing gear back-up structure weight accounts for material to react and distribute landing loads. The weight predictions for the orbiter are based on aircraft semi-empirical weight equations.

$$W_n = .039\ (F_n L_4)^{.9}$$
$$W_m = .077\ (F_m L_5)^{.9}$$

where:

F_n = Nose gear vertical load/1000-max ultimate
F_m = Main gear vertical load per strut/1000-max. ultimate
L_4 = Nose gear extended strut length - inch
L_5 = Main gear extended strut length - inch
W_n = Weight of nose gear back-up - lb.
W_m = Weight of main gear back-up - lb.

The weight penalty for the carrier main gear back-up is included in the delta wing estimation. Landing gear system weight prediction for the LRC vehicle is based on current state of the art materials. For current aircraft the landing gear system is 4.5 percent of the landed weight. The correlation of LRC landing system predicted weight with hardware data is shown in Figure 5-8.

Vehicle to vehicle attachment weight prediction is based on the following equations:

$$\text{Carrier} = 0.63\ (W_2)^{.65}$$
$$\text{Orbiter} = 0.43\ (W_2)^{.65}$$

W_2 = Orbiter gross launch weight

The carrier and orbiter structural components weights are shown in Table 5-2.

HORIZONTAL LANDING GEAR WEIGHT

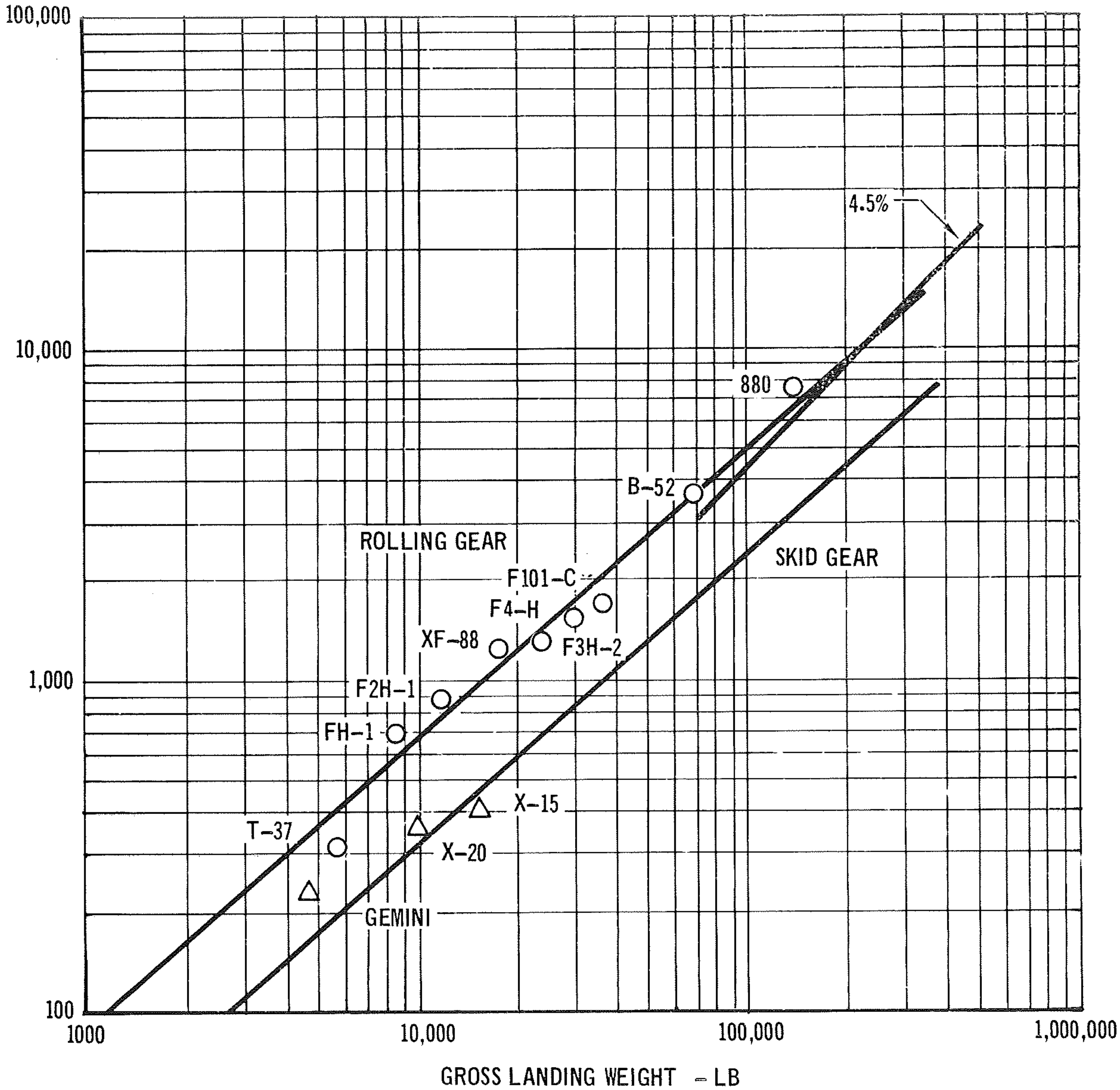


Figure 5-8

Table 5-2

STRUCTURAL COMPONENTS

Item	Weight (Lbs.)	
	Carrier	Orbiter
Sidewalls	50730	16070
Frames	13510	7280
Bulkheads	15820	4700
Baffles	9690	4740
Small LH_2 Tank	--	1290
Vertical Tank Webs	9880	--
Stage Attachment	4000	3180
LOX Tank Tie Rods	--	1040
Crew Cabin Walls	260	260
Tunnel	--	630
Thrust Structure Beef-up	900	210
Landing Gear Provisions	530	1050
Landing Engine Door Penalty	--	500
Landing Gear Door Penalty	370	600
Payload Door Penalty	--	1070
Tank Insulation	7300	3620
Total Weight	112990	46240

Thermal Protection (TPS) - The TPS includes weight allowances for the vehicle body, base and empennage heat protection. Derivations of the heating profiles and insulation requirements are discussed in Section 4.2 of this volume. Outer heat shield panel sizing is defined in Section 4.1. The component weights are summarized in Table 5-3. Maximum temperature distributions are shown in Figures 4-53 and 4-43.

Material selections for TPS shingles is based on the following temperature use ranges:

Titanium	400-1000°F
Rene-41	1000-1600°F
TD-NiCr	1600-2200°F
Columbium	2200-2900°F

The thermal protection system used for the bodies of the LRC vehicles consists of outer surface panels backed by fiberous microquartz insulation enclosed in a waterproof blanket.

The upper one-half of the carrier consists of titanium panels (66 percent-beaded, 33 percent corrugated). The lower one-half is comprised almost solely of titanium corrugated panels with Rene panels in the nose area. The base heat protection system consists of columbium panels.

The orbiter has titanium beaded panels over 61 percent of the top while the remaining top, the entire sides and 10 percent of the bottom utilizes Rene panels. TD-NiCr shingles cover the remaining 90 percent of the bottom and leading edges. Columbium shingles cover the base of the vehicle as well as the nose cap.

The aero control surfaces on the carrier are structural (titanium with Rene leading edges). The only TPS occurs in the delta wings where fibrous microquartz insulation at .5 lbs/ft^2 has been added in the areas of the landing engines, landing fuel and landing gear. The average thickness of this insulation is 1 inch and weighs 4000 lbs.

On the orbiter the tip fin outer surface and outer flaps and the lower surface of the lower elevons have TD-NiCr panels with TD-NiCr leading edges. The tip fin inner surface and inner flap are covered with Rene panels. The dorsal fin and rudder is covered with Rene panels with TD-NiCr leading edges.

Table 5-3

TOTAL THERMAL PROTECTION WEIGHT

Body

Carrier

Item	Material	Panel.Wt.	Back-up Wt.	Insul. Wt.	Total
Top 1/2 of vehicle					
2/3 of top 1/2	Tit.(beaded)	2100	1690	900	4690
Lwr 1/3 of top 1/2	Tit.(corrugated)	1220	990	500	2710
Lwr 1/2 of vehicle					
	Titanium	200	130	70	400
	Rene 41	1030	520	300	1850
	Titanium	3320	2690	1340	7350
Base Heat Prot.	Columbium	890	260	300	1450
Total System Wt.		8760	6280	3410	18450

Orbiter

Item	Material	Panel Wt.	Back-up Wt.	Insul. Wt.	Total
61% of top	Tit.(beaded)	900	560	560	2020
37% of top & sides	Rene 41	5050	2240	2550	9840
90% of bottom	Td-NiCr	5680	1070	1620	8370
Base heat & nose cap	Columbium	400	110	160	670
Total System Wt.		12030	3980	4890	20900

Empennage

Carrier	
Wing	4000
Total System Wt.	4000

Orbiter

Tip Fins	3160
Dorsal Fin	1810
Elevons	2100
Total System Wt.	7070

A cross section of the TD-nickel chrome panels is shown in Figure 5-9.

Body leading edge unit weights are based on the curve shown in Figure 5-10. The slope of the curve is determined using MDC flight proven hardware as a guide.

A value of 10 percent non-optimum has been added to the predicted weights of the thermal protection system.

Boost Propulsion - The rocket engine selection is discussed in detail in Book II of this volume.

High PC bell engines were used on both the carrier and the orbiter. The optimum number selected for the carrier was 10 engines while 2 were chosen for the orbiter.

Engine gimbal weights are derived using Figure 5-11. The weights shown on the Figure include the gimbal package on the thrust structure plus the hydraulic actuator system. A gimballed engine weight is approximately 5 percent heavier than a non-gimballed engine which results in 15 percent of the engine weight for the gimbal increment. The total system weight is:

	Carrier	Orbiter
Engine wt. - lb.	47740	9760
Gimbal wt. - lb.	7160	1460

Boost engine feed system lines are 16 inch, ID stainless steel pipes. The fuel lines are vacuum jacketed. LOX line unit weight is 17 lb/ft and fuel lines are 25 lb/ft. Line weights include a 30 percent factor for fittings and 2 lb/ft for supports. The remaining system components are derived from Figure 5-12 using the curve of system weight without suction lines.

Thrust structure weight predictions utilized a correlation with existing hardware data as shown in Figure 5-13. The weight equation is:

Thrust Structure Weight = .003 (Fvac) total

The total feed system and thrust structure weight is:

	Carrier	Orbiter
Feed System wt. - lb.	23840	6300
Thrust structure wt. - lb.	15250	3120

TD – NICKEL CHROME SHINGLE PANEL WEIGHT

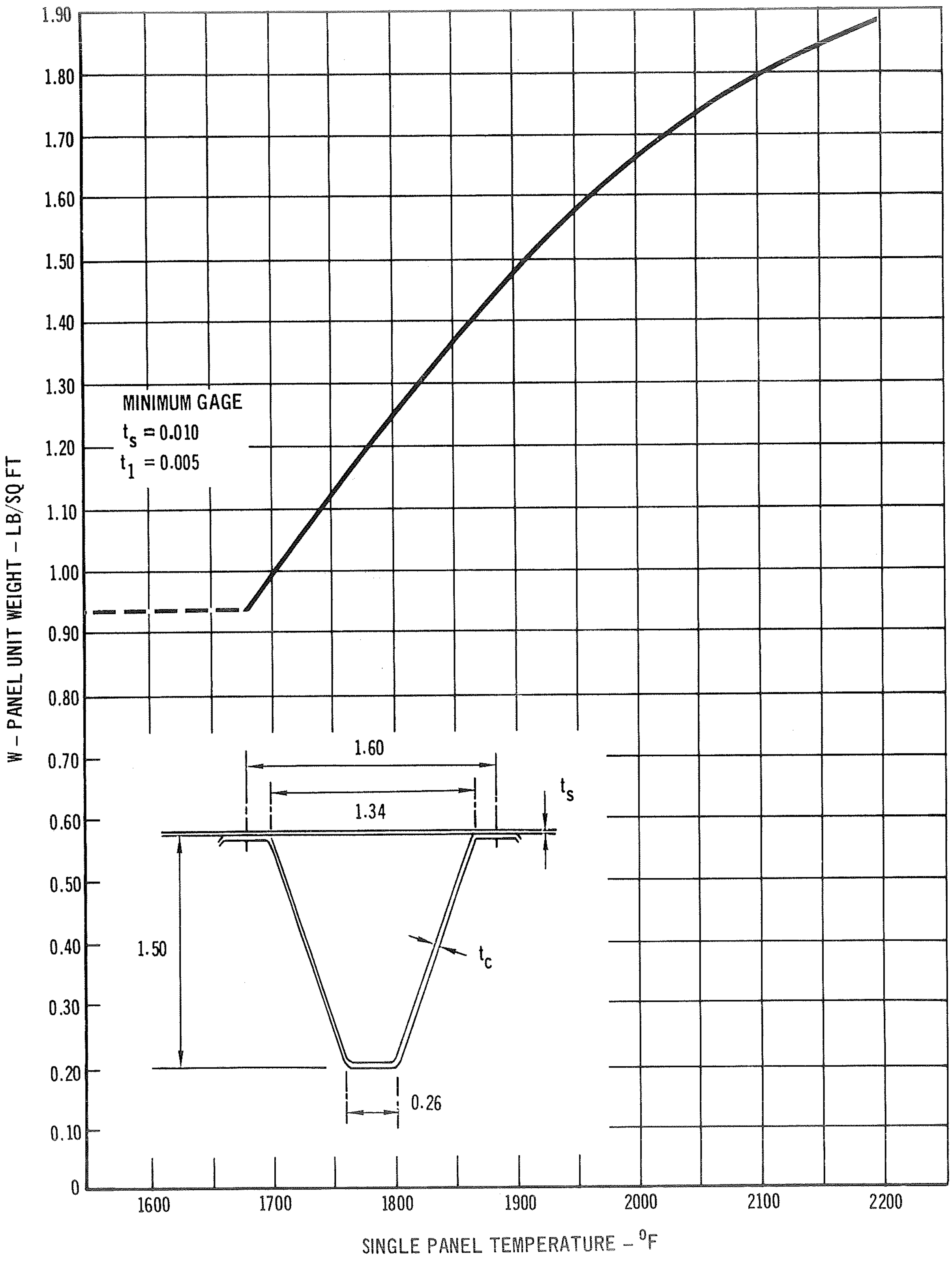


Figure 5–9

LEADING EDGE WEIGHT ESTIMATION

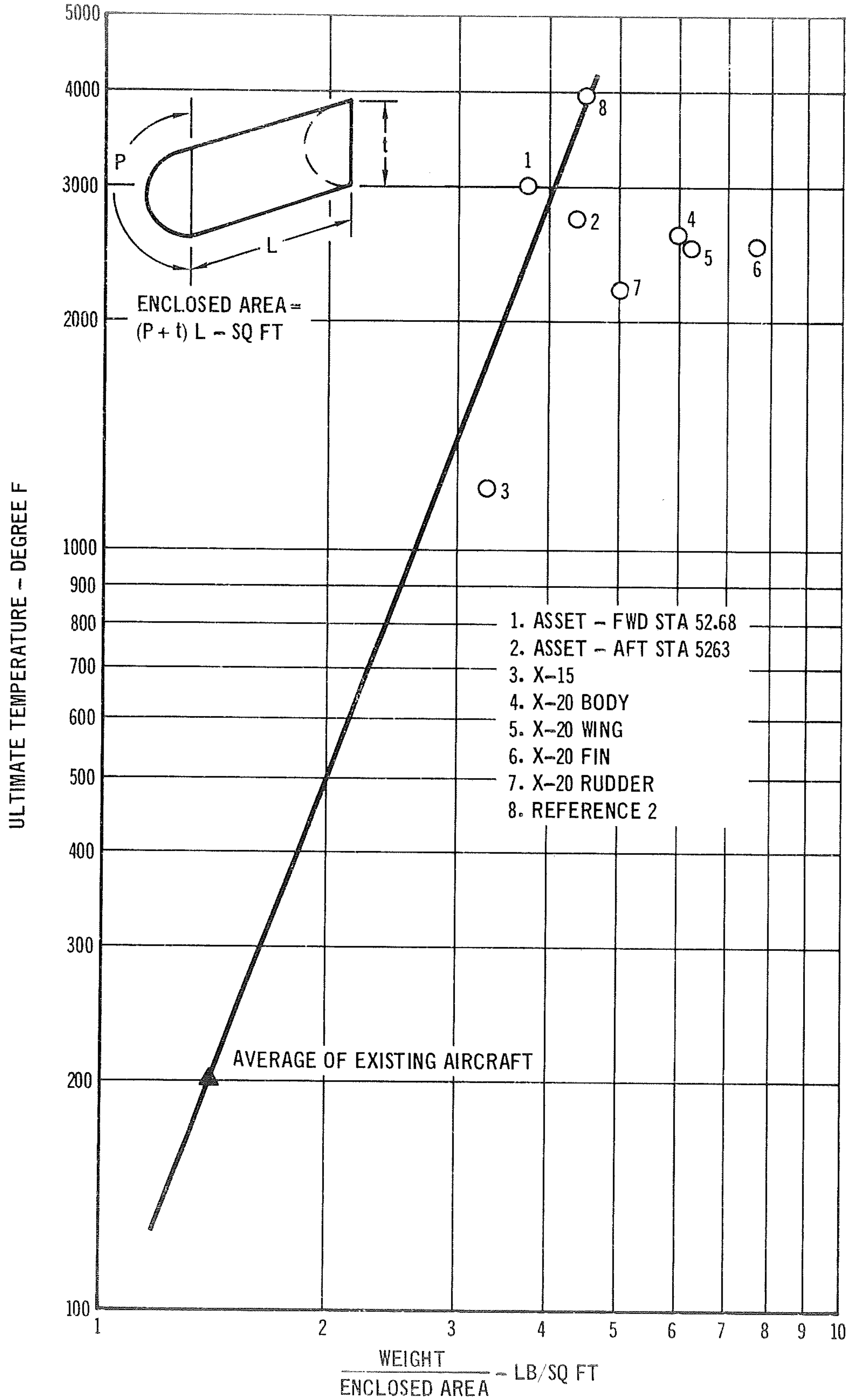


Figure 5–10

GIMBAL SYSTEM WEIGHT

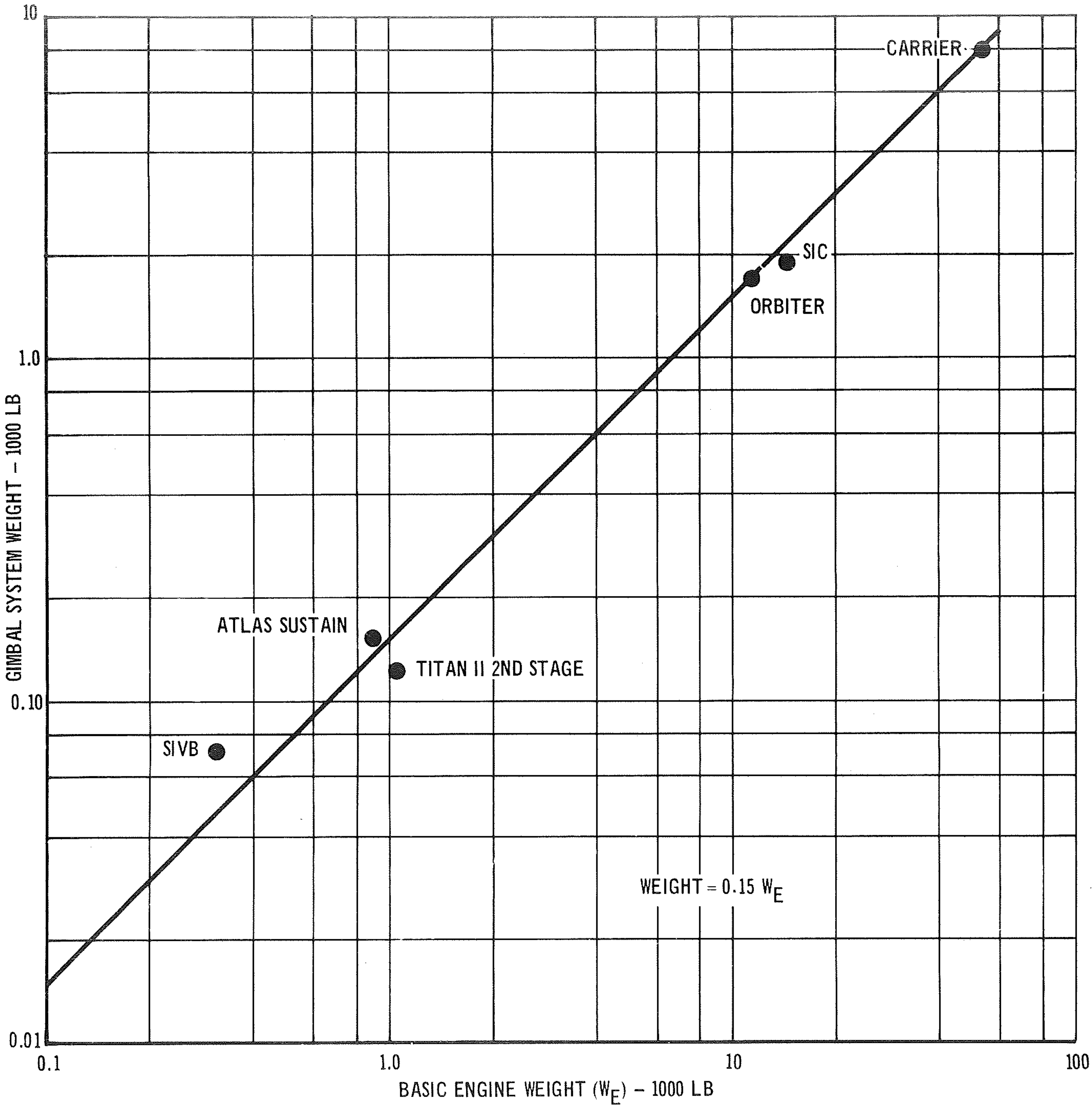


Figure 5-11

INTEGRAL BOOST
MISCELLANEOUS PROPULSION WEIGHT

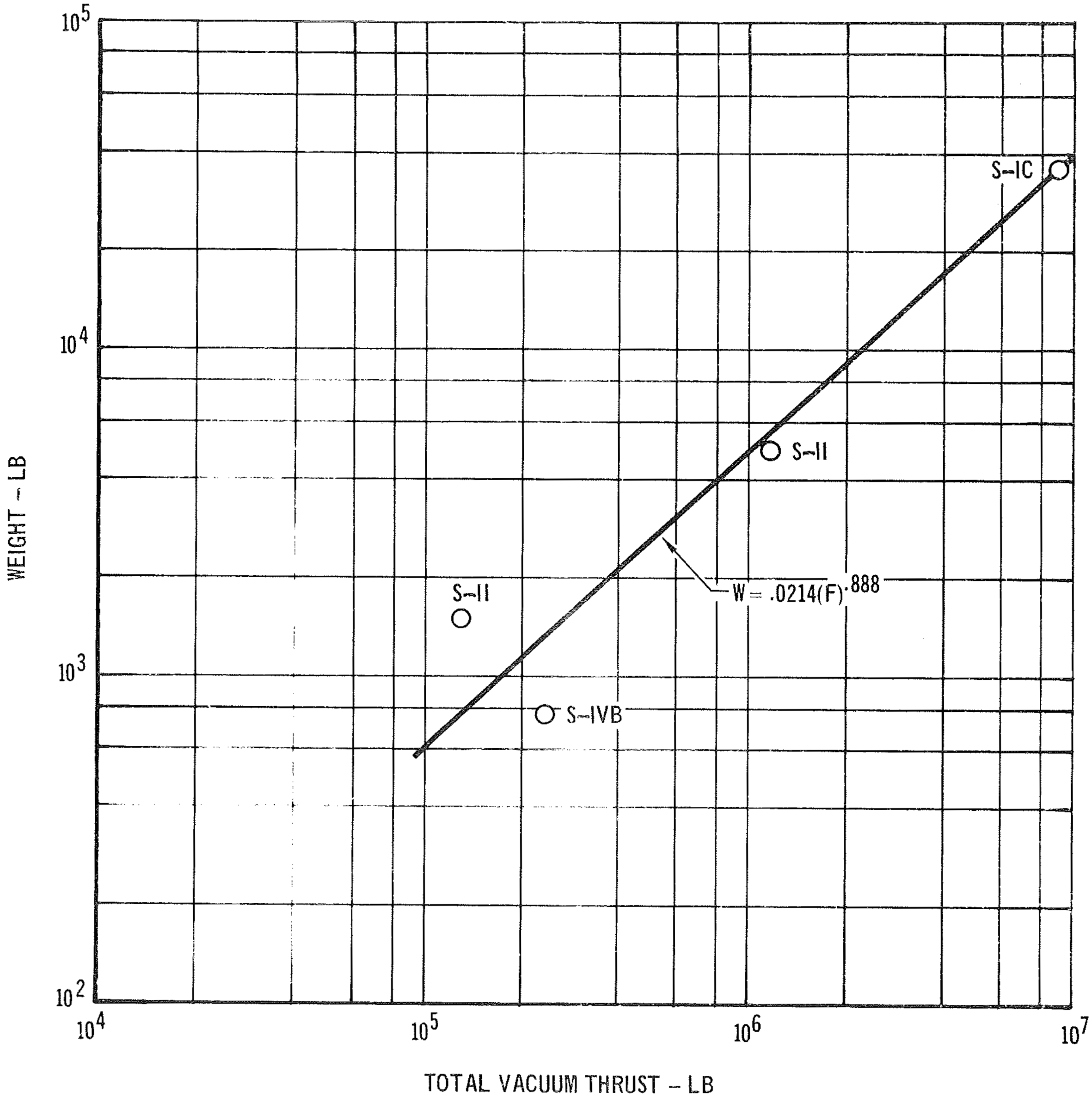


Figure 5–12

THRUST STRUCTURE WEIGHT

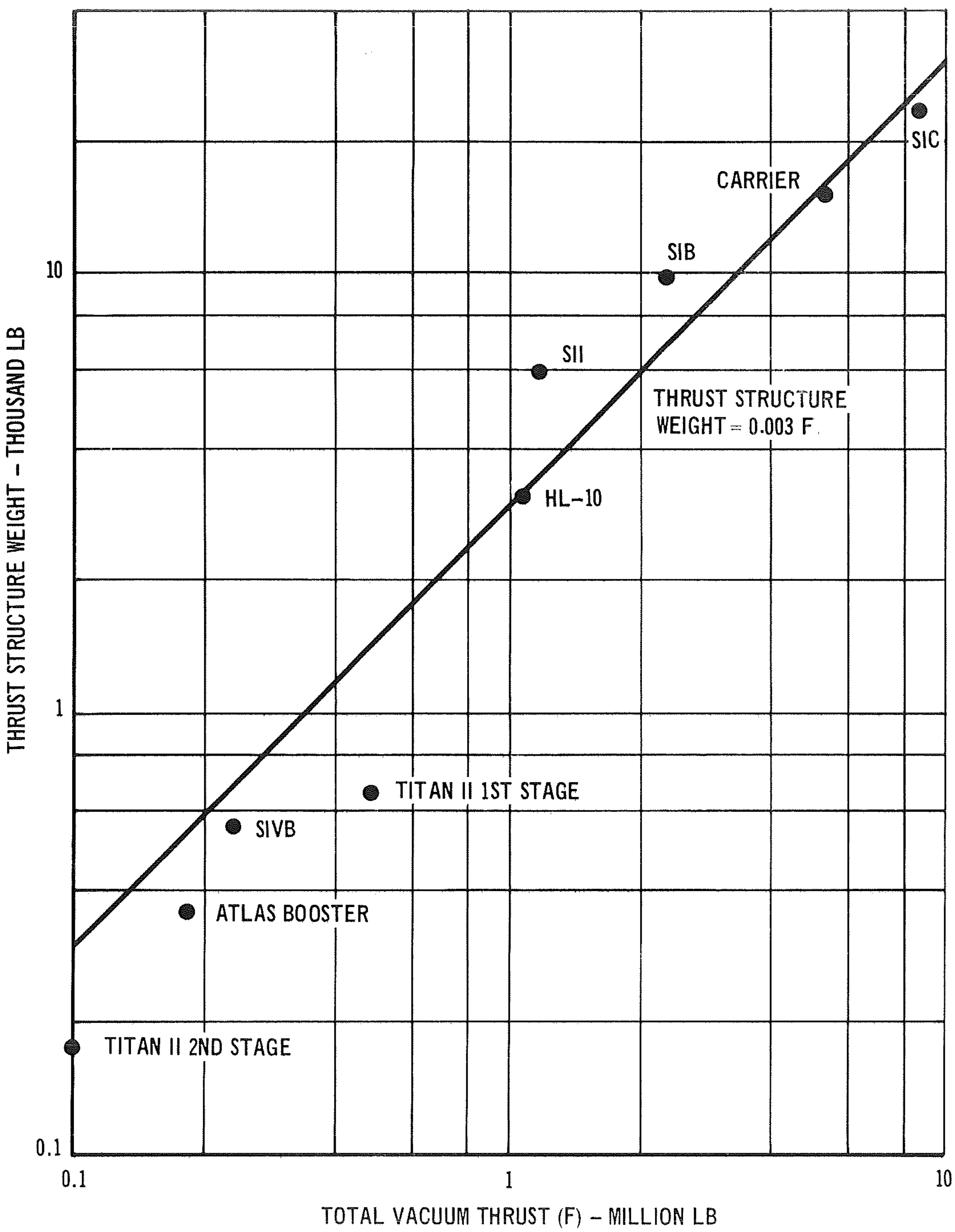


Figure 5-13

Propellant tank pressurization is accomplished by engine bleed off. The vaporized propellant is transferred to the main propellant tanks by lines. The weight allowances for these lines, values and fittings is .005 (usable propellant). The value of .005 (usable propellant) was also used for the residuals in the system. The total system weight for the Boost Propulsion System is shown in Table 5-4.

Table 5-4

BOOST PROPULSION SYSTEM WEIGHT

Item	Carrier Lbs.	Orbiter Lbs.
Engine	47740	9760
Gimbal	7160	1460
Thrust Structure	15250	3120
Feed System	23840	6300
Pressurization	10890	2500
Residual Propellant	10890	2500
Usable Propellant	2160060	499980
Hold Propellant	17040	---
	2292870	525620

A single system for both the attitude control and maneuvering is selected as the baseline because of its simplicity, minor development and installation effects. The major criteria used in the weight predictions are:

Carrier

Attitude Control

- 12 engines
  F_V/engine = 4000 lb.
- GO_2/GH_2 propellant
- Turbopump feed

Orbiter

Attitude control and Maneuver

- 20 engines
  F_V/engine = 4000 lb.
- GO_2/GH_2 propellant
- Turbopump feed

The total system weight for the Secondary Propulsion systems and the Attitude Control System is shown in Table 5-5.

Table 5-5

ATTITUDE CONTROL AND MANEUVERING SYSTEM WEIGHT

Item	Carrier Lb.	Orbiter Lb.
Engines & Accessories	1120	3050
Residual Propellant	30	1030
Usable Propellant	900	34380
Total System Weight	2050	38460

Subsonic cruise for the carrier and go-around for the orbiter is accomplished with the airbreathing propulsion system. The system is defined in detail in Book II of this Volume. Major design parameters are:

Carrier

- 4 Deployable Turbofan Engines
  F_{SL5}/Engine = 45,000 lb. (TF39)
- JP-4 fuel

Orbiter

- 4 Deployable Turbojet Engines
  F_{SL5}/Engine = 23,000 lb (JT11)
- JP-4 fuel

System component weights are shown in Table 5-6.

Table 5-6

AIRBREATHING PROPULSION SYSTEM WEIGHT

Item	Carrier Lb.	Orbiter Lb.
Engines & Accessories	35,590	13,320
Residual Propellant	1,220	200
Usable Propellant	60,940	9,970
Total System Weight	97,750	23,490

Crew and Furnishings - This system includes crew, seats, and their associated life support equipment. Weight estimates for the system are presented in Table 5-7 and they are the same for both the carrier and the orbiter.

Criteria for the weight estimates are:

- o Crew weights are based on 95 percentile men. (200 lb/man)
- o Seat weight estimates are based on modified Apollo web seats.
- o Survival kit weight allowance is 15 lb/man, to provide a one day habitable sustenance level for all personnel after landing.
- o Weight estimates for food are based on 1.8 lb/man/day and 0.2 lb/man/day for containers.
- o A minimum weight of 9 lb/man is used for drinking water, the rest if required is taken from fuel cell reactants.
- o Accommodations for living quarters during the seven day mission duration are included in the payload weights.

Environmental Control (ECS) - The ECS includes components to control internal environmental conditions of temperature, pressure, humidity, and atmospheric constituents for personnel and equipment. A detailed discussion of the system operation and source for component weights is contained in Section 4.5. Component weights for both the carrier and the orbiter are presented in Table 5-8.

Prime Power - The prime power and distribution system includes electrical power for the vehicle electronic equipment and an APU system for aerodynamic surface controls and hydraulics. The system weight summaries for both the carrier and the orbiter are presented in Table 5-9. Prime power for the carrier is supplied by six 6.0 KWH rechargeable AgO-Zn batteries, for available energy totaling 36 KWH. The battery control relays (BCR) are reverse current sensing as well as control relays to prevent degradation of the remaining batteries in the

Table 5-7

PERSONNEL AND PROVISIONS
System Weight

Item	Lbs.
Crew	400
Seat & Installation	80
Survival Kit	30
Food	28
Water	18
Misc. (Personnel Acc. & Mtg.)	44
Total System Weight	600

Table 5-8

ENVIRONMENTAL CONTROL SYSTEM WEIGHT

Orbiter

Item	Lbs.
Gas Management and Processing	52
Gas Supply and Controls	353
Heat Transport	1022
Crew Water Supply	11*
Hydraulic System Cooling	409
Circuitry, Lines, Fittings	93
Total System Weight	1940

Carrier

Item	Lbs.
Air Cycle	50
Coolant Loop	215
Hydraulic System	120
O_2 Supply	25
Circuitry, Lines, Fittings	20
Total System Weight	430

* Tank Only - Water supplied by fuel cell reactants.

Table 5-9

PRIME POWER WEIGHT SUMMARY

Carrier

Items	Lbs.
200 A-H AgO-Zn Battery	690
Inverter	160
Total System Weight	850

Orbiter

Items	Lbs.
Fuel Cell Module	400
Reactant Control Assy.	30
Thermal Control Unit	40
Product Water Subsystem	40
Control Subsystem	40
Hydrogen Tank	90
Hydrogen	59
Oxygen Tank	98
Oxygen	471
Inverter	160
Peaking/Emergency/Battery	230
Total System Weight	1658

event of a battery failure. The carrier power source is sized so that the mission can be completed with two battery failures.

Prime power for the orbiter is supplied by four H_2-O_2 matrix type fuel cell modules. Each module is rated at 2.0-2.5 KW, for a total capability of 8-10 KW at the bases. All four fuel cell modules are operated simultaneously for reactant economy as well as continuity of power in the event of a module failure. The peaking/emergency batteries are rated at 6.0 KWH each. These serve two purposes, (1) they improve the bus transient response characteristics and (2) they will provide up to two hours power for emergency deorbit, entry and cruise in the event of a catastrophic failure of the fuel cell system. The orbiter power source is sized so that a safe return is possible with two fuel cell modules failed.

The APU systems are sized by the following parameters:

	Carrier	Orbiter
Peak horsepower	1310	343
Average horsepower	286	91
Specific fuel consumption lb/hp. hr.	7	5.5

and the weights are presented in Table 5-10.

Aerodynamic Controls and Hydraulics - This system accounts for aerodynamic surface controls and hydraulics.

Weight predictions for the surface control group and hydraulics group are based on aircraft semi-empirical weight equations. Correlations with existing hardware data points are shown in Figures 5-14 and 5-15. The surface control group includes the weight for actuators, plumbing, fluid, mechanisms and supports. The weight equation is:

$$SCG - (K)(Sq)^{.317}(Wre)^{.602}(nz)^{.525}(g)^{.345} + C$$

where:	Carrier	Orbiter
K = constant for conventional aircraft	.355	.355
Sq = surface projected area sq. ft.	2350	1200
Wre = reentry weight/1000	450	185
nz = ultimate vertical load factor	5.6	5.6
g = free stream maximum dynamic pressure - psf	110	400
c = pedals and miscellaneous	100	100

Table 5-10

APU Weight Summary

Carrier

Items	Lbs.
APU	715
Fuel	575
Tanks	110
Total System Weight	1400

Orbiter

Items	Lbs.
APU	280
Fuel	540
Tanks	80
Total System Weight	900

SURFACE CONTROL GROUP

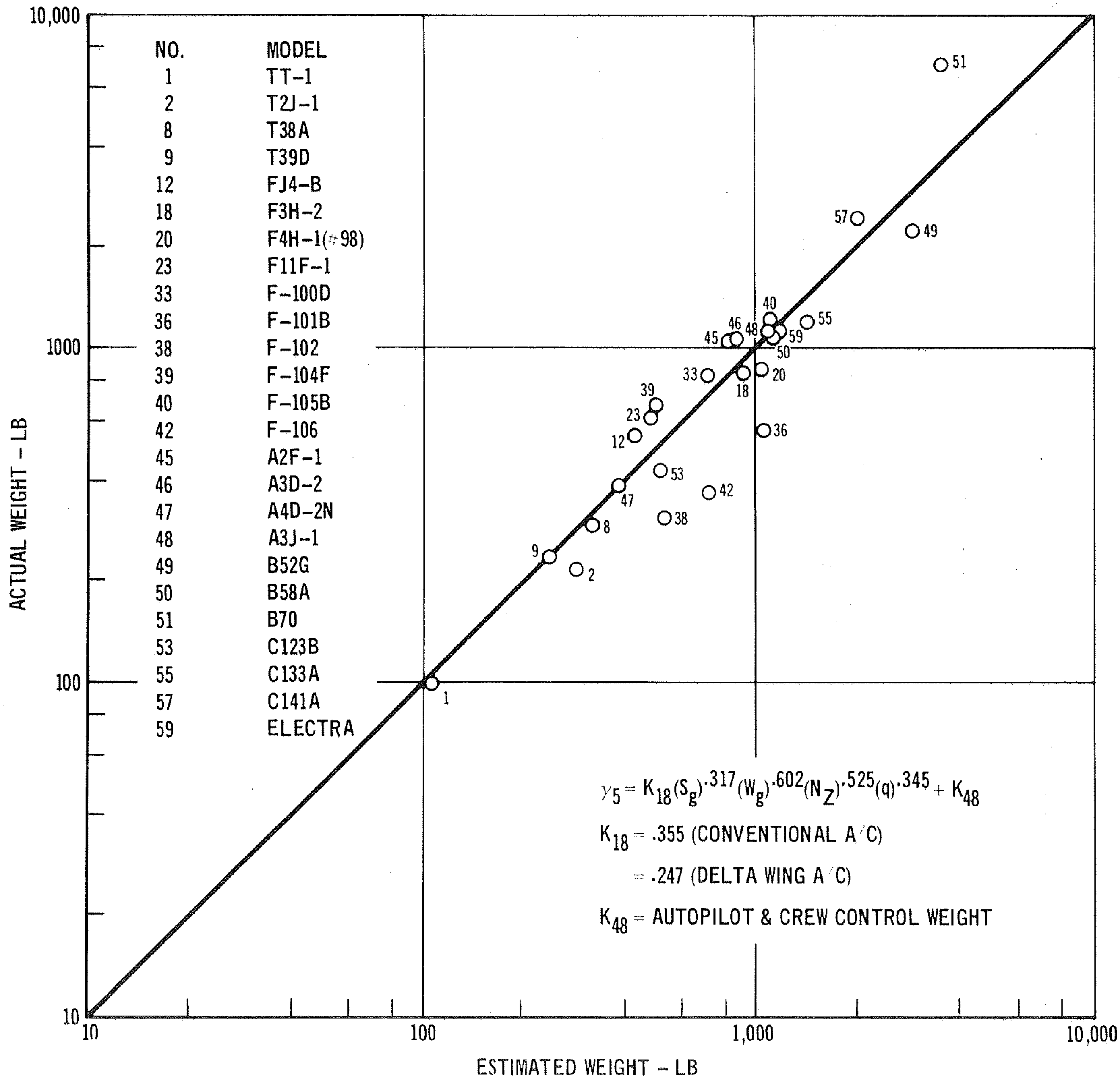


Figure 5-14

HYDRAULIC & PNEUMATIC GROUP WEIGHT

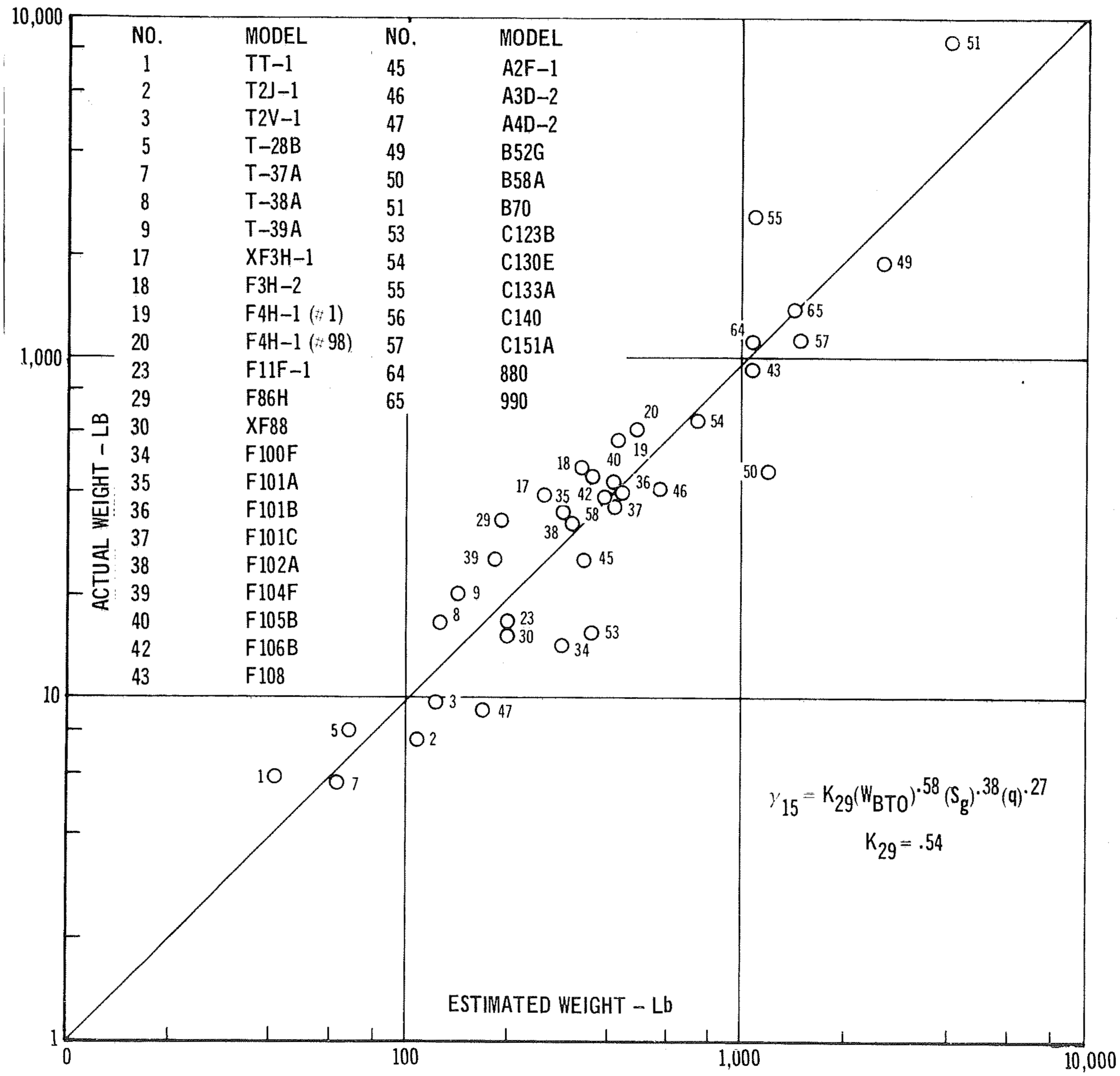


Figure 5-15

The hydraulic and pneumatic group includes reservoirs, pumps, accumulators, filters, plumbing, fluid and supports. The weight equation is:

$$HPG = K(W_{re})^{.58}(Sq)^{.38}(g)^{.27}$$

where	Carrier	Orbiter
K = constant for conventional aircraft	.54	.54

Derived weights are shown in Table 5-11.

Table 5-11

AERODYNAMIC CONTROLS AND HYDRAULIC SYSTEM WEIGHTS

Carrier

Item	Lbs.
Surface Control Group Weight	2180
Hydraulic & Pneumatic Weight	1250
Total System Weight	3430

Orbiter

Item	Lbs.
Surface Control Group Weight	1620
Hydraulic & Pneumatic Weight	830
Total System Weight	2450

Avionics - This section accounts for the electronic component weights for guidance, navigation, flight controls, on-board checkout, data management, communications and displays. The component weights are summarized in Table 5-12.

Ordnance and Separation - The ordnance and separation weight estimates include allowances for relay panels, guillotines and pyrotechnic ignitors. Weight estimates are based on Gemini data and an allowance of 200 lbs. was included for each vehicle.

Contingency - A contingency of 10 percent on all items (excluding propellant, ballast, crew and payload) is included as required by the SOW.

Ballast - Balance calculation for both stages were performed in minute detail due to the potential ballast problem on the orbiter. Both stages were balanced for the reentry condition. The carrier and the orbiter were balanced for 66 percent and 54 percent respectively and neither required ballast. Figures 5-16 and 5-17 presents the mass property distribution for each vehicle.

5.2 Carrier Weights - The weight presented in Table 5-13 represents two vehicles which conform to a large number of guidelines, performance objectives, design constraints, etc. To better understand the mass properties derived in this report, a listing of the more important characteristics is shown in Table 5-14.

The weight distribution of each vehicle in the dry weight condition is shown in Figure 5-18. The dry weight is defined as having no propellant, cargo or crewmen. Each grouping's contribution to the weight makeup is easily determined. This information is useful in helping weight reduction, cost reduction, etc., efforts concentrate on the areas which have the most promise for success.

The sequenced mass properties and detailed weight summary for the carrier are presented in Table 5-15 and 5-16. Table 5-16 includes a summary of the 50000 lb payload configuration in addition to the baseline. The mass properties include weight, three axis center of gravity and three axis moment of inertias for each important mission phase. The carrier meets the reentry center of gravity requirement without the aid of ballast.

5.3 Orbiter Weights - The sequenced mass properties and detailed weight summary of the orbiter are presented in Tables 5-17 and 5-18. Table 5-18 also includes weights for the 50000 lb payload configuration. The mass proeprties include weight, three axis center of gravity and three axis inertias. The only removals from gross weight thru landing are the usable propellants for the four (main, secondary, entry attitude and landing) propulsion systems. The dry weight

Table 5-12

Integrated Avionics System Weight

Orbiter

Item	Lbs.
Guidance & Navigation	720
Landing & Navigation Aides	170
Telecommunication	325
Central Management Computer	180
Displays, Control & Sequencing	477
Flight Control	75
Control Amplifiers	122
Instrumentation	125
Power Distribution Wire	700
Signal Distribution Wire	1300
Total System Weight	4194

Carrier

Item	Lbs.
Integrated Avionics Subsystem	1570
Power & Signal Distribution Wire	1645
Total System Weight	3215

WEIGHT DISTRIBUTION
CARRIER
REENTRY CONDITION

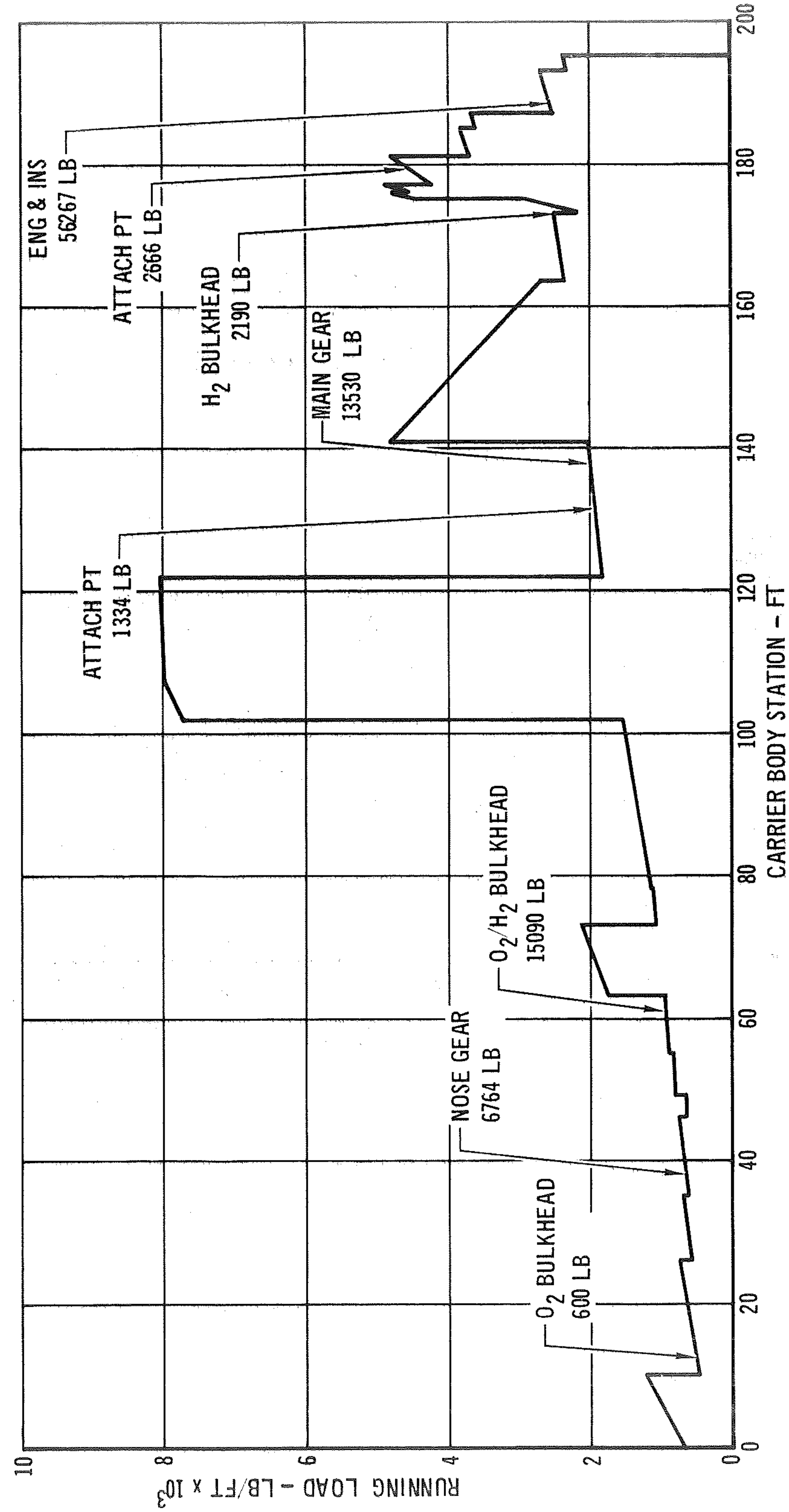


Figure 5-16

WEIGHT DISTRIBUTION
ORBITER
REENTRY CONDITION

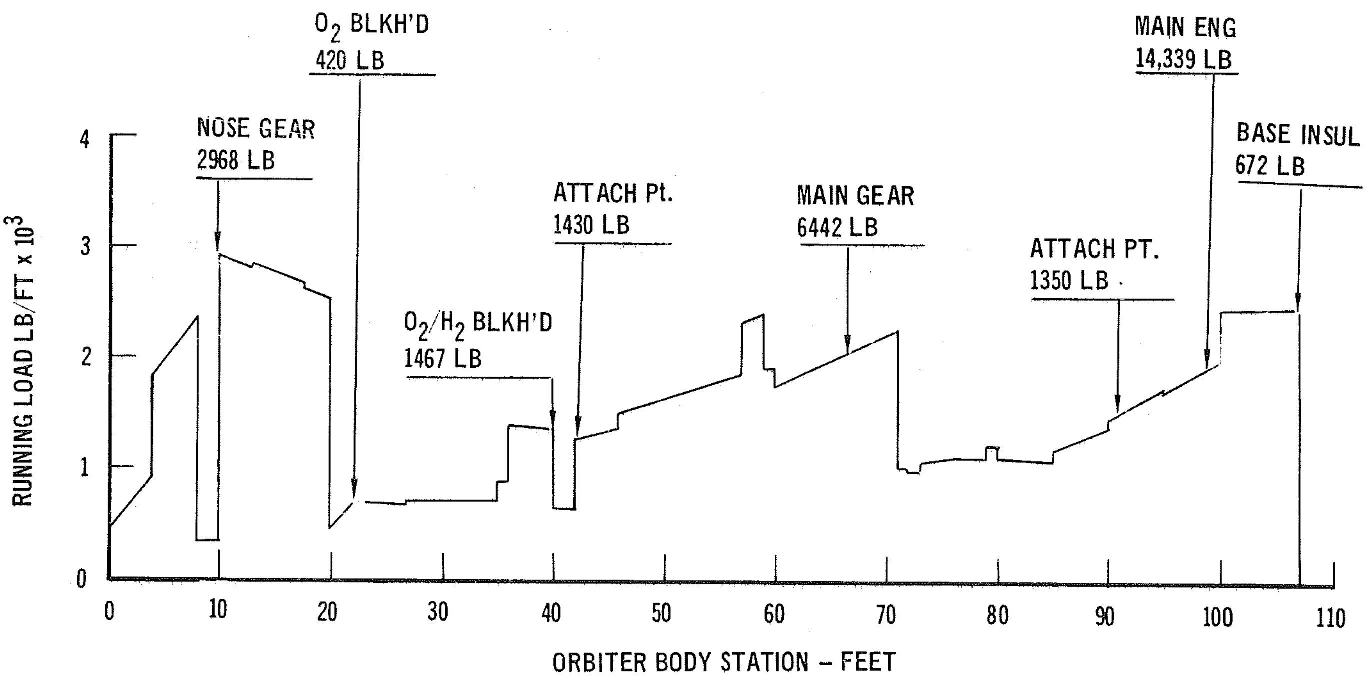


Figure 5-17

Table 5-13

TWO STAGE WEIGHT SUMMARY
Baseline System

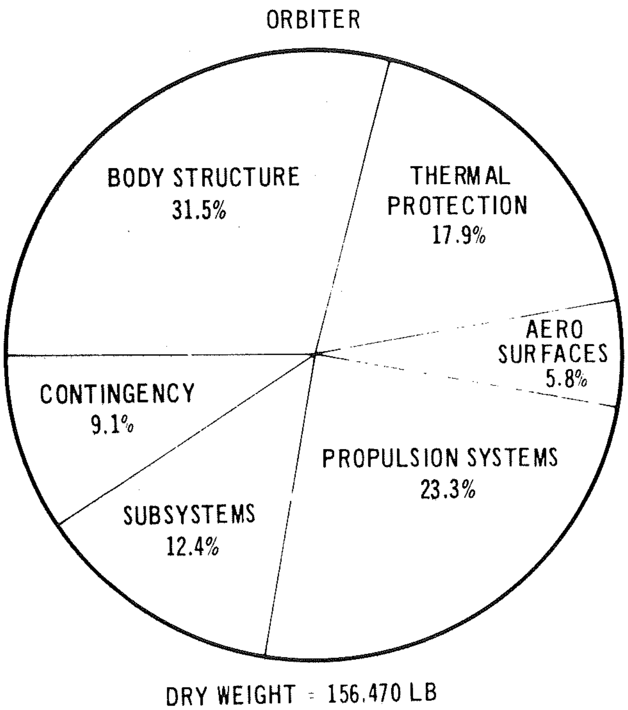
VEHICLE	ORBITER	CARRIER
LENGTH-FT	107	195
THERMOSTRUCTURES	91,470	248,485
LANDING SYSTEM	9,190	22,310
MAIN PROPULSION SYSTEM	527,750	2,302,860
SECONDARY PROPULSION SYSTEM	38,790	2,250
AIR BREATHING PROPULSION SYSTEM	24,710	100,900
SUBSYSTEMS AND CREW	13,310	11,070
CARGO	25,000	–
GROSS PAD WEIGHT	730,220	2,687,875
GROSS LIFTOFF WEIGHT	3,401,055	

Table 5-14

BASELINE DESIGN AND OPERATIONAL CHARACTERISTICS

CHARACTERISTIC	FIRST STAGE (CARRIER)	SECOND STAGE (ORBITER)
CONFIGURATION	LOW WING CLIPPED DELTA	HL-10
LENGTH	195 FT	107 FT
GROSS LAUNCH WEIGHT	2.69×10^6 LB	0.730×10^6 LB
INERT WEIGHT CONTINGENCY	10%	10%
CARGO SIZE	–	15' DIA x 30 FT LENGTH
CARGO WEIGHT	–	25,000 LB
CREW	2	2
MISSION TIME	2 DAYS	7 DAYS
BOOST ΔV	31,250 FPS	–
BOOST PROPELLANT	LOX/LH_2	LOX/LH_2
PROPELLANT TANKAGE	INTEGRAL	INTEGRAL
THERMAL PROTECTION SYSTEM	TITANIUM-RENE´	TD-NiCr
LAUNCH THRUST MODE	SERIES BURN –	NO IDLE MODE
REENTRY CG LOCATION	66% L	53–55% L

DRY WEIGHT DISTRIBUTION



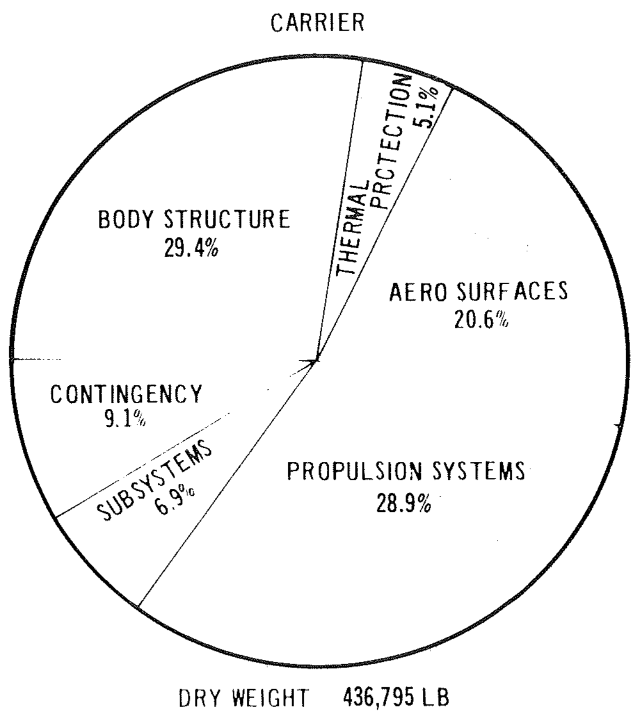


Figure 5-18

Table 5-15

SEQUENCED MASS PROPERTIES

CARRIER

MISSION POINT	WEIGHT (LB.)	CENTER OF GRAVITY (FT)			MOMENT OF INERTIA (SLUG - FT^2)		
		X	Y	Z	ROLL	PITCH	YAW
GROSS WEIGHT	2,687,875	63.2	0	.2	11,660,000	202,405,000	210,160,000
LIFT OFF WEIGHT	2,670,835	63.5	0	.2	11,659,000	201,215,000	208,970,000
INJECTED WEIGHT	510,775	128.7	0	.8	11,381,000	42,658,000	50,422,000
RETROGRADE WEIGHT	510,775	128.7	0	.8	11,381,000	42,658,000	50,422,000
ENTRY WEIGHT	510,775	128.7	0	.8	11,381,000	42,658,000	50,422,000
LANDING WEIGHT	448,935	130.7	0	.9	9,420,000	42,322,000	48,137,000
DRY WEIGHT	436,795	132.5	0	.9	9,378,000	40,262,000	46,039,000

Table 5-16

SPACECRAFT DETAILED SUMMARY WEIGHT STATEMENT

CONFIGURATION - CARRIER

PAYLOAD TO ORBIT	25000	50000
1.0 Aerodynamic Surfaces	(99320)	(115420)
1.1 Fixed Surfaces		
1.1.1 Dorsal Fin	5670	6590
1.1.2 Delta Wing	78390	91100
1.2 Movable Surfaces		
1.2.1 Dorsal Fin Rudder	1190	1380
1.2.2 Delta Wing Flap	5040	5860
1.19 Contingency	9030	10490
2.0 Body Structure	(141060)	(169190)
2.1 Structural Fuel Tank	57815	69330
2.2 Structural Oxidizer Tank	25125	30140
2.3 Structural Propellant Tank	13030	15630
2.6 Str. Fwd. of Integral Tanks	4200	5040
2.7 Str. Between Integral Tanks	---	---
2.8 Str. Aft. of Integral Tanks	7660	9190
2.9 Thrust Structure	15250	18290
2.10 Interstage/Spacer/Vehicle Inst.	4000	4800
2.11 Pressurized Compartment	260	310
2.12 Non-Pressurized Compartment	900	1080
2.19 Contingency	12820	15380
3.0 Induced Environment Protection	(24700)	(30220)
3.2 Thermal Protection (Passive)		
3.2.4 Body	17000	21230
3.2.6 Airbreathing Engine	4000	4640
3.2.7 Base and Nose Cap	1450	1600
3.8 Contingency	2250	2750
4.0 Launch Recovery and Docking	(22320)	(27380)
4.3 Landing Gear		
4.3.1 Main Landing Gear	18260	22400
4.3.2 Nose Landing Gear	2030	2490
4.19 Contingency	2030	2490
5.0 Main Propulsion	(130440)	(170300)
5.1 Liquid Rocket Engine and Acc.		
5.1.1 Main Engines and Acc	47740	66330
5.1.2 Att. and Maneuver Engines	1120	1220
5.6 Airbreathing Engine and Acc.	35590	41410
5.9 Fuel System	23840	33050
5.10 Pressurization System	10890	13640
5.19 Contingency	11260	14650

Table 5-16 (Continued)

6.0 Orientation Controls, Separation	(11650)	(17310)
6.1 Thrust System		
6.1.1 Gimbal System	7160	11700
6.3 Aerodynamic Controls	2180	2550
6.3.1 Hydraulic and Pneumatic	1250	1490
6.19 Contingency	1060	1570
7.0 Prime Power Source	(2300)	(2300)
7.2 Power Source - Fuel Cell	--	---
7.3 Power Source - Batteries	690	690
7.4 APU - System	1400	1400
7.19 Contingency	210	210
8.0 Power Conversion and Dist.	(1985)	(1985)
8.1 Power Conversion - Electrical	1205	1205
8.6 Power Distribution - Hyd. and Pneu.	600	600
8.19 Contingency	180	180
9.0 Guidance and Navigation	(606)	(606)
9.1 Guidance - Source - Evaluation - Output	550	550
9.19 Contingency	56	56
10.0 Instrumentation	(201)	(201)
10.1 Sensors	181	181
10.19 Contingency	20	20
11.0 Communications	(206)	(206)
11.1 Communications Equipment	186	186
11.19 Contingency	20	20
12.0 Environmental Control System	(470)	(470)
12.1 ECS Equipment - Personnel - Coolant	430	430
12.19 Contingency	40	40
14.0 Personnel Provisions	(200)	(200)
14.1 Accommodations for Personnel	200	200
14.19 Contingency	--	--
15.0 Crew Station Controls and Panels	(557)	(557)
15.1 Crew Station Controls	507	507
15.19 Contingency	50	50
16.0 Range Safety and Abort	(--)	(--)
17.0 Personnel	(400)	(400)
17.1 Crew	400	400
18.0 Cargo	(--)	(--)
19.0 Ordnance	(220)	(220)
20.0 Ballast	(--)	(--)

Table 5-16 (Continued)

21.0 Residual Propellant	(12140)	(14870)
21.1 Main Propulsion System	10890	13640
21.2 Att. and Maneuver System	30	50
21.3 Airbreathing System	1220	1180
22.0 Reserve Propellant	(12200)	(11800)
22.3 Fuel Main Engine Reserves	--	--
22.7 Fuel Reserves - RCS	--	--
22.11 Fuel Reserves - Cruise	12200	11800
23.0 Inflight Losses	(*)	(*)
24.0 Thrust Decay Propellant	(--)	(--)
25.0 Full Thrust Propellant	(2209700)	(2750810)
25.1 Main Propulsion System	2160060	2702660
25.2 Att. and Maneuver System	900	1180
25.3 Jet Fuel - Air Breathing Engine	48740	46970
26.0 Thrust Build Up Propellant	(17040)	(25190)
27.0 Pre-Ignition Losses	(--)	(--)
Gross Launch Weight	2687875	3339795
Gross Lift Off Weight	2670835	3314605

* Included in Residual Propellant

Table 5-17

SEQUENCED MASS PROPERTIES

ORBITER

MISSION POINT	WEIGHT (LB.)	CENTER OF GRAVITY (FT)			MOMENT OF INERTIA (SLUG - FT2)		
		X	Y	Z	ROLL	PITCH	YAW
GROSS WEIGHT	730,220	42.4	0	-3.1	1,626,000	12,762,000	13,659,000
LIFT OFF WEIGHT	730,220	42.4	0	-3.1	1,626,000	12,762,000	13,659,000
SECOND STAGE SEPARATION	730,220	42.4	0	-3.1	1,626,000	12,762,000	13,659,000
INJECTED WEIGHT	230,230	57.9	0	-1.8	1,106,000	6,606,000	7,090,000
RETROGRADE WEIGHT	197,690	58.0	0	-.5	1,025,000	6,468,000	7,028,000
ENTRY WEIGHT	195,760	58.0	0	-.4	1,020,000	6,458,000	7,024,000
LANDING WEIGHT	185,800	60.6	0	-.4	1,019,000	5,851,000	6,416,000
DRY WEIGHT	156,470	61.8	0	-.2	983,860	5,629,000	6,200,000

Table 5-18

SPACECRAFT DETAILED SUMMARY WEIGHT STATEMENT

CONFIGURATION - ORBITER

PAYLOAD TO ORBIT	25000	50000
1.0 Aerodynamic Surfaces	(10070)	(14870)
1.1 Fixed Surfaces		
1.1.1 Tip Fins	4430	6560
1.1.2 Dorsal Fin	1440	2130
1.2 Movable Surfaces		
1.2.1 Tip Fin Flaps	890	1310
1.2.2 Dorsal Fin Rudder	330	480
1.2.3 Elevons	2060	3040
1.19 Contingency	920	1350
2.0 Body Structure	(54300)	(79760)
2.1 Structural Fuel Tank	18361	26970
2.2 Structural Oxidizer Tank	11224	16490
2.3 Structural Propellant Tank	1290	1890
2.6 Str. Fwd. of Integral Tanks	1716	2520
2.7 Str. Between Integral Tanks	1070	1570
2.8 Str. Aft of Integral Tanks	6359	9340
2.9 Thrust Structure	3120	4580
2.10 Interstage/Spacer/Vehicle Inst.	3180	4670
2.11 Pressurized Compartment	890	1310
2.12 Non-Pressurized Compartment	2150	3160
2.19 Contingency	4940	7260
3.0 Induced Environment Protection	(30770)	(45430)
3.2 Thermal Protection (Passive)		
3.2.1 Tip Fins and Flaps	3160	4670
3.2.2 Dorsal Fin and Rudder	1810	2670
3.2.3 Elevons	2100	3090
3.2.4 Body	20230	29880
3.2.5 Base and Nose Cap	670	990
3.8 Contingency	2800	4130
4.0 Launch Recovery and Docking	(9200)	(14050)
4.3 Landing Gear		
4.3.1 Main Landing Gear	7520	11490
4.3.2 Nose Landing Gear	840	1280
4.19 Contingency	840	1280
5.0 Main Propulsion	(38138)	(56368)
5.1 Liquid Rocket Engine and Acc.		
5.1.1 Main Engines and Acc	9760	13570
5.1.2 Att. and Maneuver Engines	3050	4660
5.6 Airbreathing Engine and Acc.	13320	20350
5.9 Fuel System	6300	8730
5.10 Pressurization System	2500	4340
5.19 Contingency	3208	4718

Table 5-18 (Continued)

6.0 Orientation Controls, Separation	(4300)	(6550)
6.1 Thrust System		
6.1.1 Gimbal System	1460	2390
6.3 Aerodynamic Controls	1620	2320
6.3.1 Hydraulic and Pneumatic	830	1240
6.19 Contingency	390	600
7.0 Prime Power Source	(2638)	(2638)
7.2 Power Source - Fuel Cell	1268	1268
7.3 Power Source - Batteries	230	230
7.4 APU - System	900	900
7.19 Contingency	230	230
8.0 Power Conversion and Dist.	(2380)	(2380)
8.1 Power Conversion - Electrical	1760	1760
8.6 Power Distribution - Hyd. and Pneu.	400	400
8.19 Contingency	220	220
9.0 Guidance and Navigation	(1180)	(1180)
9.1 Guidance - Source - Evaluation - Output	1070	1070
9.19 Contingency	110	110
10.0 Instrumentation	(555)	(555)
10.1 Sensors	505	505
10.19 Contingency	50	50
11.0 Communications	(135)	(135)
11.1 Communications Equipment	120	120
11.19 Contingency	15	15
12.0 Environmental Control System	(2130)	(2130)
12.1 ECS Equipment - Personnel - Coolant	1940	1940
12.19 Contingency	190	190
14.0 Personnel Provisions	(200)	(200)
14.1 Accommodations for Personnel	200	200
14.19 Contingency	--	--
15.0 Crew Station Controls and Panels	(544)	(544)
15.1 Crew Station Controls	494	494
15.19 Contingency	50	50
16.0 Range Safety and Abort	(--)	(--)
17.0 Personnel	(400)	(400)
17.1 Crew	400	400
18.0 Cargo	(25000)	(50000)
19.0 Ordnance	(220)	(220)
20.0 Ballast	(--)	(--)

Table 5-18 (Continued)

21.0 Residual Propellant	(3730)	(6220)
21.1 Main Propulsion System	2500	4340
21.2 Att. and Maneuver System	1030	1580
21.3 Airbreathing System	200	300
22.0 Reserve Propellant	(14840)	(16820)
22.3 Fuel Main Engine Reserves	7720	5940
22.7 Fuel Reserves - RCS	7120	10880
22.11 Fuel Reserves - Cruise	--	--
23.0 Inflight Losses	(*)	(*)
24.0 Thrust Decay Propellant	(--)	(--)
25.0 Full Thrust Propellant	(529490)	(919460)
25.1 Main Propulsion System	492260	862630
25.2 Att. and Maneuver System	27260	41610
25.3 Jet Fuel - Air Breathing Engine	9970	15220
26.0 Thrust Build Up Propellant	(--)	(--)
27.0 Pre-Ignition Losses	(--)	(--)
Gross Pad Weight	730220	1219910

* Included in residual propellant

removes inert propellants, cargo and crew. The orbiter reentry center of gravity is at 54 percent which is within the specified range (53-55%L) and therefore has no ballast. However, if it were deemed necessary to relocate the C.G. to 53% ballast would be required. Obviously, the amount of ballast required is a function of how far the C.G. must be moved. Figure 5-19 illustrates this relationship for the baseline orbiter. A requirement to have the reentry C.G. at 53 percent L would require 5400 lbs. of ballast to be added.

During the course of a study and/or production run the weight of a vehicle usually increases due to a number of factors (increased performance, revised mission times, revised estimates, etc.). It is then common to conduct weight reduction campaigns. It is at this time that an interesting thing occurs with vehicles that have a C.G. requirement as the orbiter and carrier. Figure 5-20 illustrates the effects of localized weight increments on the baseline orbiter and the importance of their location.

From the figure it is seen that any element whose C.G. is at the C.G. of the vehicle will be 100% effective. However, if an element is to be removed whose C.G. is in front of the desired location it will be necessary to add back ballast to reestablish the desired C.G. condition. The weight saving will therefore be less that 100% effective. As an example, if 1200 lbs. is to be removed with a C.G. at .3L it is necessary to add 630 lbs. of ballast resulting in only 47% effectiveness.

By the same token, if weight is added behind the desired C.G., it will be more than 100% effective, again because of the need to add ballast. The addition of 1200 lbs. at .7L results in a total weight increase of 1550 lbs. for an effectiveness of 129%.

CENTER OF GRAVITY LOCATION BALLAST EFFECT

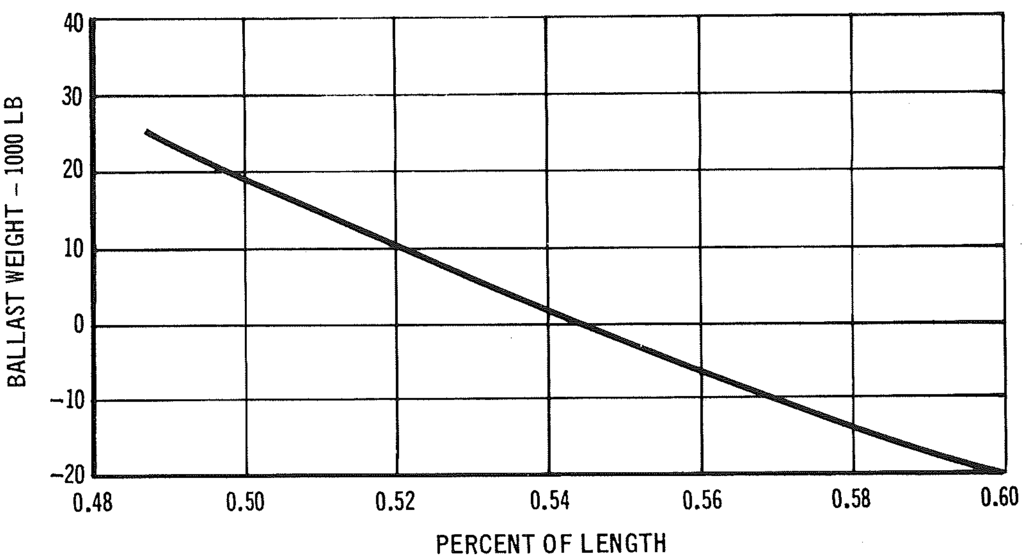


Figure 5–19

EFFECTS OF LOCALIZED WEIGHT INCREMENTS
Orbiter

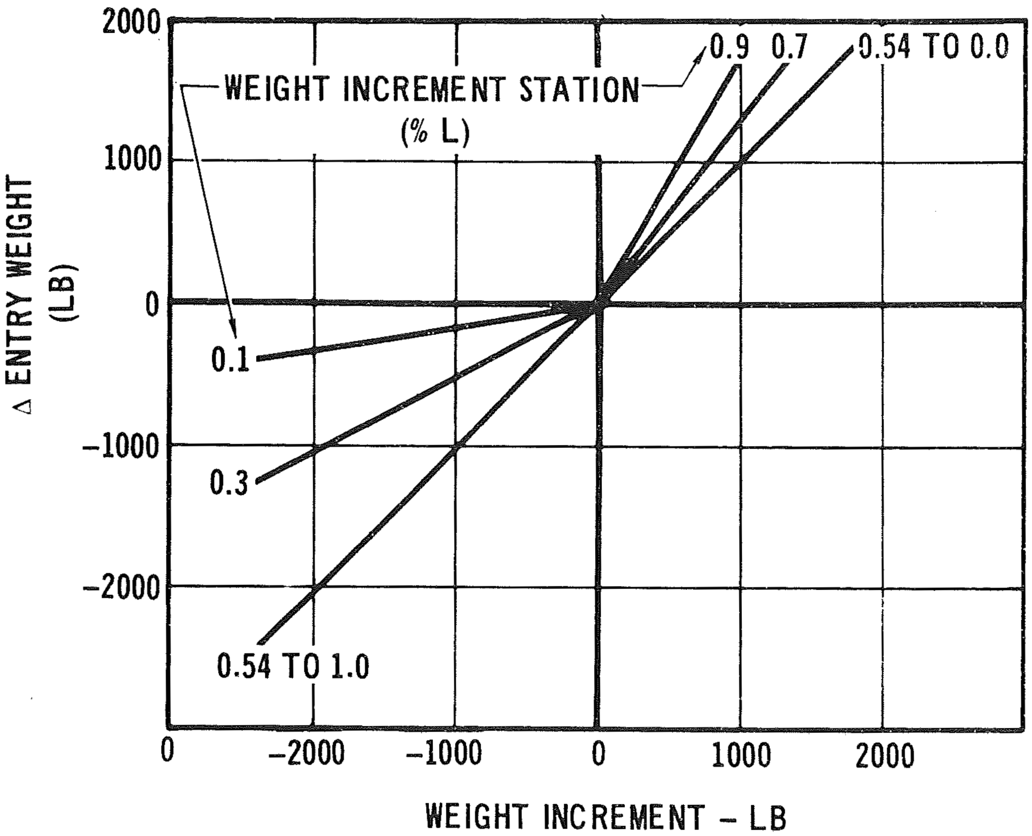


Figure 5-20

5.4 Launch Configuration - The combined vehicles sequenced mass properties is presented in Table 5-19. The table covers gross launch through first stage burn out only. For mission phases subsequent to burn the properties are the same as shown for the individual vehicles.

Of some interest to several disciplines was the center of gravity travel during the first stage burn. This travel is shown on Figure 5-21. The C.G. travels a distance of 52.3 feet longitudinally and 8.5 feet in the pitch plane. No lateral movement is experienced. This is somewhat unusual as most launch vehicles have a symmetrical travel about the longitudinal axis. However, the solution is simply a larger gimbal angle capability.

5.5 Sensitivities - The sensitivities as well as the final baselines were obtained using "SWAP-IT" (Simplified Weight and Propulsion Iterative Techniques). This program was developed during the course of the study and it proved an invaluable tool.

Using a tool such as SWAP-IT enabled the operator to do the following:

- o Control the environment (keeping a consistent set of ground rules)
- o Look at the effects of performance criteria (thus getting the optimum answers for the baseline)
- o Repeat answers (insuring the same degree of accuracy in the answer with each perturbation of the vehicle)
- o Monitor the study (giving each discipline a quick insight into the development of the baseline and the effects of their inputs)
- o Evaluate other configurations to the same ground rules.

This program can be broken down into two basic blocks, the weights section (which incorporates all subsystems inputs) and the propulsion section as shown in Figure 5-22. The weights section incorporates all of the empirical weight equations discussed in this section along with a ballast routine for each vehicle. A spacecraft weight is then calculated and sent to the propulsion section which uses this weight to size its subsystems. Data is passed between the two sections and iterated on internally until the desired tolerances are met. SWAP-IT iterates on cruise range, ballast, landing gear weight, aero controls, hydraulic and pneumatic systems, and carrier length. A sample output of SWAP-IT is presented in Tables 5-20, 5-21 and 5-22.

Table 5-19

COMBINED SEQUENCED MASS PROPERTIES
Orbiter and Carrier

MISSION PHASE	WEIGHT (LBS.)	CENTER OF GRAVITY			MOMENT OF INERTIA		
		X	Y (FT)	Z	ROLL	PITCH (SLUG-FT^2)	YAW
GROSS LAUNCH	3,418,095	77.6	0	- 4.9	23,239,000	305,825,000	304,523,000
LIFT OFF WT.	3,401,055	77.8	0	- 4.9	23,226,000	303,835,000	302,546,000
1/4 BURNED	2,861,000	87.2	0	- 5.8	22,678,000	248,023,000	247,214,000
1/2 BURNED	2,321,000	98.0	0	- 7.2	21,908,000	191,244,000	191,139,000
3/4 BURNED	1,781,000	111.2	0	- 9.4	20,710,000	131,807,000	132,832,000
BURNED	1,241,000	129.9	0	-13.4	18,530,000	60,959,000	64,097,000

LAUNCH CONFIGURATION C.G. TRAVEL
First Stage Burn

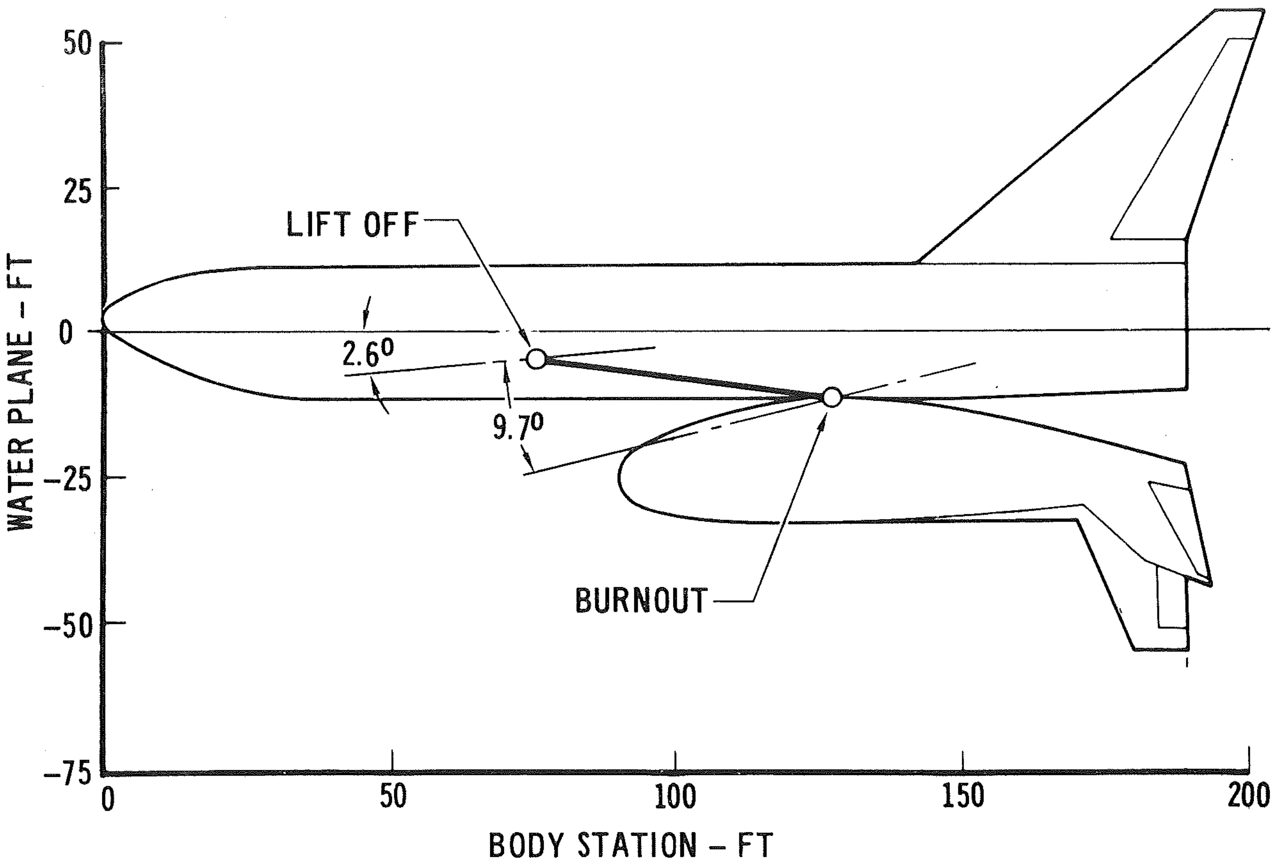


Figure 5-21

SWAP IT BLOCK DIAGRAM

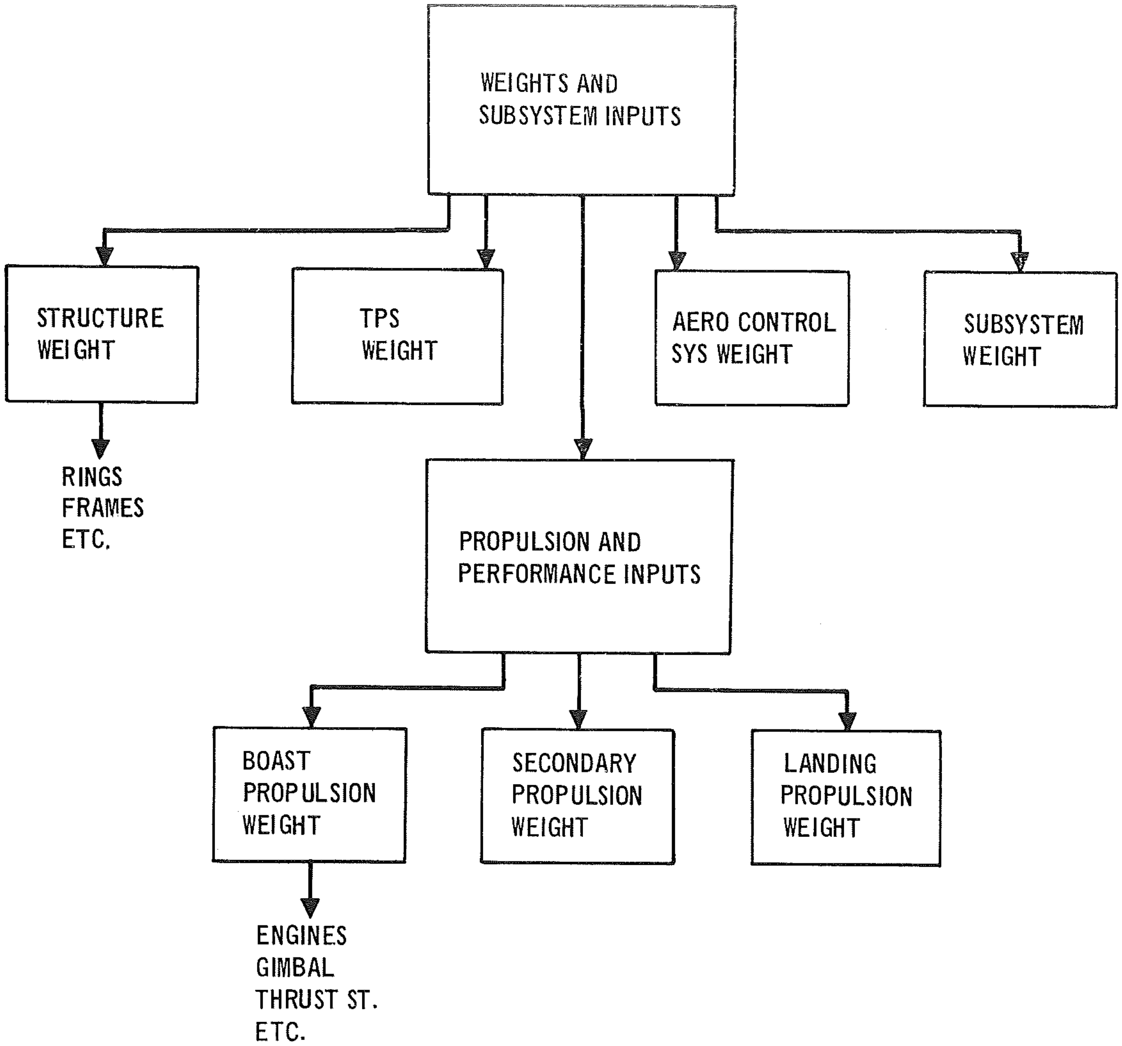


Figure 5–22

Table 5-20

SWAP-IT OUTPUT

VARIABLE
.71291 10
DELTA V IN CARRIER= 14473.2
508182. 520577.
3160.96

VARIABLE	1ST STAGE	2ND STAGE
GROSS LAUNCH WEIGHT	2.67192E 6	730216.
INJECTED WEIGHT	511877.	230229.
ENTRY WEIGHT	511877.	195764.
LANDING WEIGHT	450938.	185797.
DRY WEIGHT	439538.	157697.
BASIC S/C WEIGHT	282946.	133955.
CARGO WEIGHT	0	25000
CARGO AT LANDING	-------	25000
CREW WEIGHT	600	600
BOOST ENGINE SYS. WGT.	70136.8	14339.2
BOOST FEED SYS. WGT.	23842.4	6295.14
CORE PROP. INERTS WGT.	21771.2	4999.88
ORBIT MAN. SYS. WGT.	-------	34892.3
ACS WEIGHT	2047.51	3568.51
LANDING SYS. WGT	97745.9	23491.6
CONTINGENCY WEIGHT	13379.2	3635.42
NUMBER OF ENGINES	10	2
LENGTH OF VEHICLE	195.066	107
EXPANSION RATIO	57	80
VACUUM SPECIFIC IMP.	446.5	451.8
ISP,SL/ISP,VAC	.882	.761
BOOST PROP. WGT.	2.16005E 6	499888.
PROPELLANT VOLUME	105175.	24154
HOLD PROPELLANT	17072.2	0
IDLE PROPELLANT	-------	0
TOTAL SL THRUST	4.48217E 6	896433.
TOTAL VAC. THRUST PER ENG.	508182.	520577.
F/W,SL	1.31746	1.22763
F/W,VACUUM	-------	1.42582
F/W,ALT WITH ONE ENG. OUT	-------	.71291
DELTA V	14473.2	16776.7
OMS DELTA V	-------	1460
CRUISE FACTOR	.190956	.12
CONTINGENCY FACTOR	.1	.1

TOTAL VEHICLE
GLW DV

3.40213E 6 31249.9

ORBITER WEIGHT SUMMARY Table 5-21		
AERODYNAMIC SURFACES	9153.28	
VERTICAL TAIL	1441.3	
VERTICAL RUDDER	326.48	
SIDE FINS	4440	
SIDE FLAPS	886.94	
UPPER ELEVONS	1016.96	
LOWER ELEVONS	1041.6	
BODY STRUCTURE	42384.8	
THERMAL PROTECTION	31594.	
BODY TPS SYSTEM	20235.9	
VERTICAL FIN + RUDDER	1805.76	
SIDE FINS + FLAPS	3164.64	
UPPER ELEVONS	1035.12	
LOWER ELEVONS	1060.2	
TANK INSULATION	3620	
BASE HEAT PROTECTION	672.328	
LANDING SYSTEM	8360.88	
MAIN PROPULSION SYSTEM	525622.	
ENGINES + ACC.	11215.7	
PRESSURIZATION	2499.94	
THRUST STRUCTURE	3123.46	
RESIDUAL PROPELLANT	2499.94	
USABLE PROPELLANT	499988.	
IDLE PROPELLANT	0	
HOLD PROPELLANT	0	
SYSTEM LINES + ETC	6295.14	
SECONDARY PROPULSION SYSTEM	34892.8	
ENGINES + ACC.	1468.99	
RESIDUAL PROPELLANT	973.509	
USABLE PROPELLANT	32450.3	
ENTRY ATTITUDE CONTROL SYSTEM	3568.51	
ENGINES + ACC.	1583.7	
RESIDUAL PROPELLANT	57.8098	
USABLE PROPELLANT	1926.99	
LANDING PROPULSION SYSTEM	23491.6	
ENGINES + ACC.	13326.2	
RESIDUAL PROPELLANT	199.323	
USABLE PROPELLANT	9966.15	
INTEGRATED AVIONICS	2200	
POWER DISTRIBUTION GROUP	7217.19	
ELECTRICAL POWER	3860	
AERODYNAMIC CONTROLS	1623.35	
HYDRAULIC SYSTEM	833.838	
AERO APU SYSTEM	900	
ECS	1940	
PERSONNEL + PROVISIONS	600	
RANGE SAFETY + ORD.	200	
CARGO AT LANDING	25000	
CARGO OAT LAUNCH	25000	
CONTINGENCY	10305.	3686.42
BALLAST	0	

Table 5-22

CARRIER WEIGHT SUMMARY		
AERODYNAMIC SURFACES	90235.5	
VERTICAL TAIL	5669.35	
VERTICAL RUDDER	1185.25	
DELTA WING	78389.2	
WING FLAPS	5041.71	
BODY STRUCTURE	105760.	
THERMAL PROTECTION	30862.8	
BODY TPS SYSTEM	18262.4	
VERTICAL TAIL + RUDDER	0	
DELTA WING + FLAP	3995.31	
TANK INSULATION	7229.62	
BASE HEAT PROTECTION	1375.53	
LANDING SYSTEM	20291.8	
MAIN PROPULSION SYSTEM	2.29287E 6	
ENGINES + ACC.	54891.3	
PRESSURIZATION	10885.6	
THRUST STRUCTURE	15245.5	
RESIDUAL PROPELLANT	10885.6	
USABLE PROPELLANT	2.16005E 6	
HOLD PROPELLANT	17072.2	
SYSTEM LINES + ETC	23842.4	
ENTRY ATTITUDE CONTROL SYSTEM	2047.51	
LANDING PROPULSION SYSTEM	97745.9	
ENGINES + ACC	35587.8	
RESIDUAL PROPELLANT	1218.79	
USABLE PROPELLANT	60939.3	
INTEGRATED AVIONICS	1570	
POWER AND DISTRIBUTION GROUP	7253.45	
ELECTRICAL POWER	2430	
AERODYNAMIC CONTROLS	2176.89	
HYDRAULIC SYSTEM	1246.56	
AERO APU SYSTEM	1400	
ECS	400	
PERSONNEL + PROVISIONS	600	
RANGE SAFETY + ORD.	200	
CONTINGENCY	25722.4	13379.2
BALLAST	0	
28.7 SECS READY		

Sensitivities are an important phase of the mass property analysis. This involves analyzing the net effect on the vehicle gross launch weight and payload capability resulting from changes in various parameters. Seldom, if ever, does a one-to-one relationship occur as a result of a change. Rather, it is multiplied, sometimes quite dramatically, due to its effect on other subsystems within the vehicle and on other vehicles. For example, any increase in landing weight on the second stage is multiplied not only by its effect on the landing, retrograde, and entry attitude control system, but, by its effect on the first stage propulsion system as well. Therefore, sensitivities allow a quick assessment of the desirability of proposed changes which effect any of the eight parameters chosen for study. The changes in payload were analyzed while holding the gross weight constant and changes in gross weight were analyzed while holding the payload constant. The sensitivities for the baseline vehicle, 3.40 million pounds, and 25,000 pounds payload are presented in Table 5-23.

In addition, the sensitivities for a 4.53 million pound, 50,000 pound payload vehicle were determined and are shown on Table 5-24. Comparison of the values on the two tables reveals that the sensitivities vary with vehicle size and weight. This requires that the sensitivities be redetermined for different vehicles.

In addition to the sensitivities, changes which are in play during the entire design phase are larger, one time changes called incremental effects. Incremental effects result from specific modification in the design or operational mode of the vehicle. These include such changes as: no flyback to launch site, incorporate idle burn, remove inert contingency, etc. In assessing the net weight effect of each operational or design change, the basic vehicle dimensions were held constant. Also, the gross launch weight is held constant at 3.4 million pounds and the net weight changes associated with the design or performance parameter change is reflected solely in the orbiter payload capability. It should be remembered that these effects are really interdependent and the incorporation of two or more will not result in the total net effect being the sum of the individual increments shown. The incremental effects for the baseline vehicle are shown in Table 5-25.

As in sensitivities, the effect on a 4.53 million pound, 50,000 pound cargo vehicle is of interest. The incremental effects for this vehicle are shown in Table 5-26.

The values shown differ slightly from those presented for the 3.4 MLB configuration, and are generally of a greater magnitude. The only exceptions to this

Table 5-23

SENSITIVITIES

25,000 Lb Payload

REFERENCE:

P/L SIZE	15' D x 30' L
GLOW	3.40 MLB

PARAMETER	CONST. GLOW $\frac{\Delta W_{PL}}{\Delta \text{ PARAMETER}}$		CONST. P/L $\frac{\Delta W_{GL}}{\Delta \text{ PARAMETER}}$	
1ST STAGE INERT WT.	− 0.16	LB/LB	+ 6.5	LB/LB
2ND STAGE INERT WT.	− 1.0	LB/LB	+ 38.0	LB/LB
ΔV − LAUNCH	− 12.8	LB/FPS	+ 528	LB/FPS
ΔV − ORBIT	− 14.0	LB/FPS	+ 560	LB/FPS
PROPELLANT I_{SP}	+ 843	LB/SEC	− 35,400	LB/SEC
CRUISE RANGE	− 152	LB/NM	+ 5,790	LB/NM
GO AROUND (ORBITER)	− 377	LB/MIN	+ 12,400	LB/MIN
RETURN CARGO	− 0.27	LB/LB	+ 10.5	LB/LB

ILRVS — 385F

Table 5-24

SENSITIVITIES
50,000 Lb Payload

REFERENCE:

P/L SIZE	15' x 60' L
GLOW	4.53 MLB

PARAMETER	CONST. GLOW $\Delta W_{PL} / \Delta$ PARAMETER	CONST. P/L $\Delta W_{GL} / \Delta$ PARAMETER
1ST STAGE INERT WT.	− 0.16 LB/LB	+ 5.5 LB/LB
2ND STAGE INERT WT.	− 1.0 LB/LB	+ 32.6 LB/LB
ΔV − LAUNCH	− 18.7 LB/FPS	+ 614 LB/FPS
ΔV ORBIT	− 21.0 LB/FPS	+ 690 LB/FPS
PROPELLANT I_{SP}	+ 1,262 LB/SEC	− 41,200 LB/SEC
CRUISE RANGE	− 180 LB/NM	+ 5,860 LB/NM
GO AROUND (ORBITER)	− 499 LB/MIN	+ 16,200 LB/MIN
RETURN CARGO	− 0.27 LB/LB	+ 9.0 LB/LB

ILRVS-384F

Table 5-25

INCREMENTAL EFFECTS
25,000 Lb Payload

REFERENCE:

P/L SIZE 15'D x 30'L
GLOW 3.40 MLB

	CONST. GLOW	CONST. P/L
Δ EFFECT	ΔW_{PL}	ΔW_{GL}
• NO FLYBACK TO LAUNCH SITE	+10,200	-312,000
• SERIES BURN TO IDLE MODE	- 5,700	+283,000
• PARALLEL BURN WITH CROSS FEED	+ 3,200	-124,000
• NO 1ST STAGE INERT CONTINGENCY	+ 6,400	-235,000
• NO 2ND STAGE INERT CONTINGENCY	+14,000	-524,000
• NO INERTS (BOTH STAGES) CONTINGENCY	+20,500	-695,500
• NO 0.75% ΔV RESERVE	+ 3,000	-124,000
• H_2 CRUISE FUEL	+ 6,210	-252,000

NOTE: INCREMENTS ARE NOT ADDITIVE

ILRVS-473F

Table 5-26

INCREMENTAL EFFECTS
50,000 Lb Payload

REFERENCE:

P/L SIZE 15' D x 60' L
GLOW 4.53 MLB

	CONST. GLOW	CONST. P/L
Δ EFFECT	ΔW_{PL}	ΔW_{GL}
• NO FLYBACK TO LAUNCH SITE	+ 9,800	–268,000
• SERIES BURN TO IDLE MODE	– 5,700	+211,000
• PARALLEL BURN WITH CROSS FEED	+ 4,700	–144,000
• NO 1ST STAGE INERT CONTINGENCY	+ 7,700	–229,000
• NO 2ND STAGE INERT CONTINGENCY	+20,000	–615,000
• NO INERTS (BOTH STAGES) CONTINGENCY	+28,000	–787,000
• NO 0.75% ΔV RESERVE	+ 4,400	–144,000
• H_2 CRUISE FUEL	+ 6,000	–204,000

ILRVS-474F

are the flyback and series burn increments. The staging velocity of the 4.53 MLB vehicle is less than the 3.4 which results in a shorter distance to flyback and therefore less sensitivity. In the case of series burn, each second stage expends approximately the same propellant. However, this is a smaller percentage of the total in the 4.53 MLB vehicle and again results in less sensitivity.

Several of the parameters or design features effects were of a greater interest. Accordingly, they are presented in greater detail in the following four charts. The first is the effect of return payload weight on ascent payload weight and is shown in Figure 5-23. Returning no payload would result in a 127% increase in ascent payload capability.

The baseline spacecraft was designed to return all the ascent payload. However, if only a portion of this is returned, the amount of ascent payload could be increased as shown. This tradeoff study was conducted holding the total gross lift off weight constant. The increase in ascent payload capability is a reflection of the weight reduction in those systems which are designed or influenced by retrograde, reentry and landing weight such as the retrograde and landing propulsion systems and the landing gear.

The second effect illustrated is the effect of payload weight and size on gross launch weight. The variation in gross launch weight for changes in payload weight for four payload sizes is shown in Figure 5-24. Each of the vehicles is point designed for a specific payload size and weight capability. The payload size (diameter and length) is given in feet at the left of each curve. The 15' x 17' vehicle is designed for 10,000 lbs. payload while the 15' x 30' and 15' x 60' use 25,000 lbs. and the 22' x 60' is 50,000 lbs.

The curve illustrates that a larger vehicle suffers a smaller weight change in gross weight per pound of payload than one designed for a smaller payload. The 15' x 17' vehicle increases at the rate of 44 lb/lb payload but the 22' x 60' only increases at the rate of 20 lb/lb payload.

Note, however, that the payload geometry has a greater effect than payload weight. This is shown by the 15' x 60' payload which has a larger gross launch weight than the same payload in a 15' x 30' container.

The five circled points on this chart represent the baseline and four alternate payloads which were analyzed as required by the study contract.

EFFECT OF RETURN PAYLOAD CAPABILITY

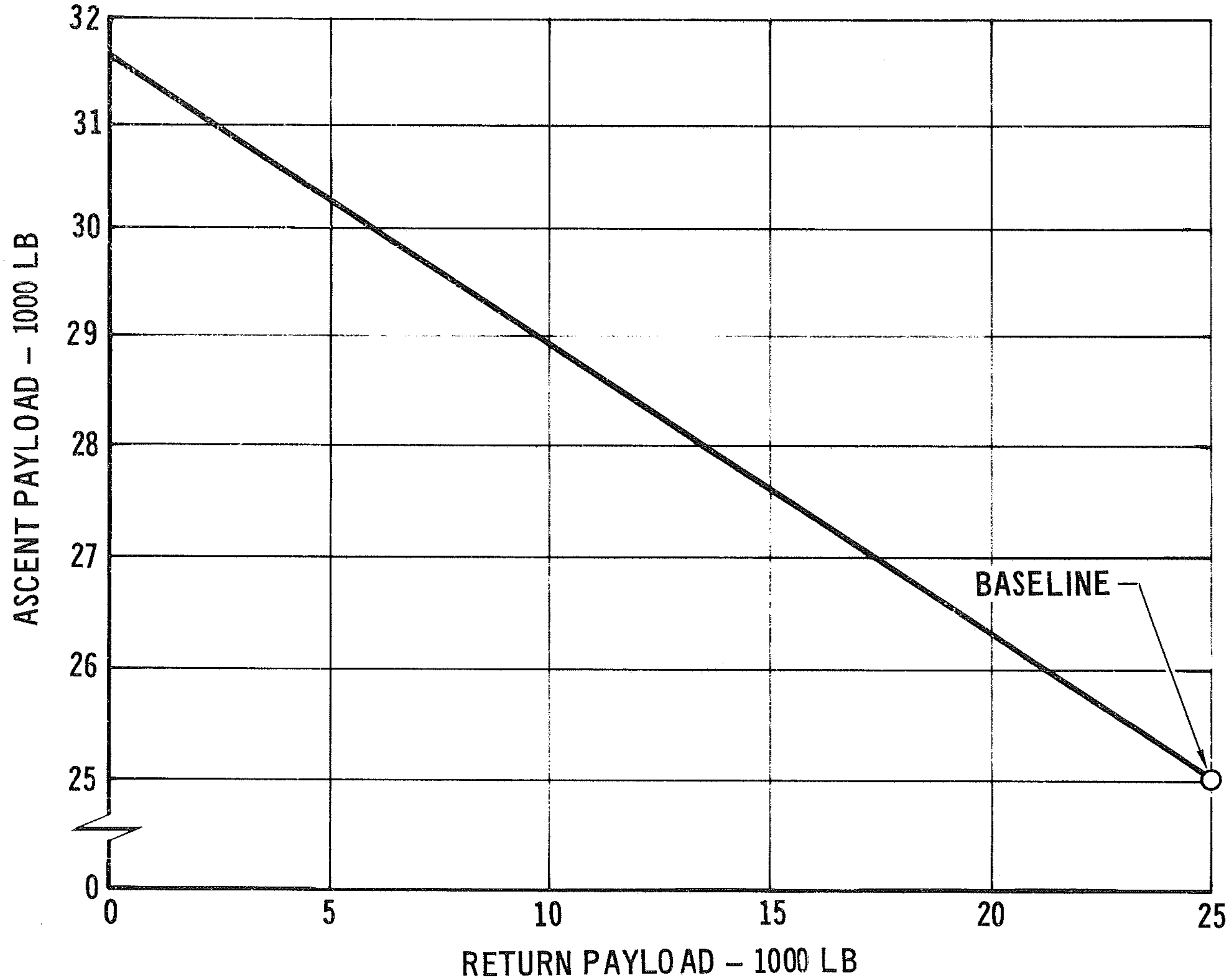


Figure 5–23

EFFECT OF PAYLOAD WEIGHT AND SIZE

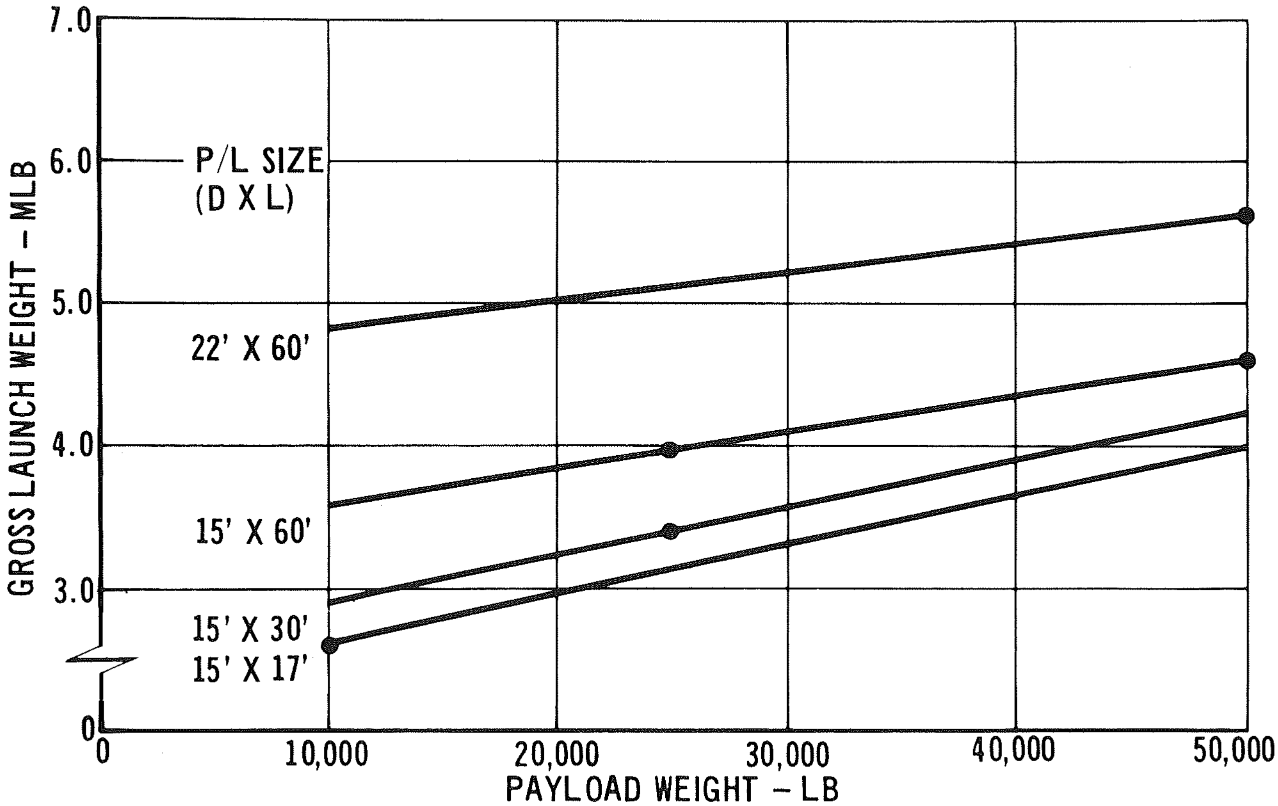


Figure 5-24

The third effect is the effect of velocity apportionment between stages on the vehicle gross launch weight. The values are shown in Figure 5-25.

The design point is predicated on use of the shortest length 2nd stage that will accommodate the 15' x 30' payload canister. The maximum amount of boost propellant is loaded into the excess volume that exists even with a minimum length vehicle. The resulting 2nd stage ΔV is then subtracted from the total required to determine the 1st stage velocity increment. As indicated, this design point is very near the optimum for minimum gross launch weight. To increase the 1st stage ΔV required off-loading the 2nd stage, an inefficient approach from the standpoint of volume utilization and, hence, gross launch weight. Decreasing the 1st stage ΔV necessitates an increased length orbiter to obtain the additional ΔV required. Although a small weight reduction results, the increased cost of a larger orbiter plus the potential increase in the engine out ΔV penalty are expected to more than offset this weight advantage. Hence, the ΔV split for the baseline vehicle is that resulting from the use of a minimum length orbiter.

The fourth figure (Figure 5-26) illustrates the effect of efficient utilization of volume available for boost propellant on gross lift-off weight. The figure clearly shows the advantage of the baseline concept (integral tankage, 80% utilization) over a non-integral concept which has only 55% utilization. As indicated, the weight difference is so pronounced that even the addition of a 10,000 lb. inert weight would not increase the integral tankage gross weight above the non-integral value.

First generation weight summaries are shown in Table 5-27 and 5-28 for the baseline payload and four alternates. These summaries present the carrier and orbiter characteristics respectively.

VELOCITY SPLIT DETERMINATION
Baseline System

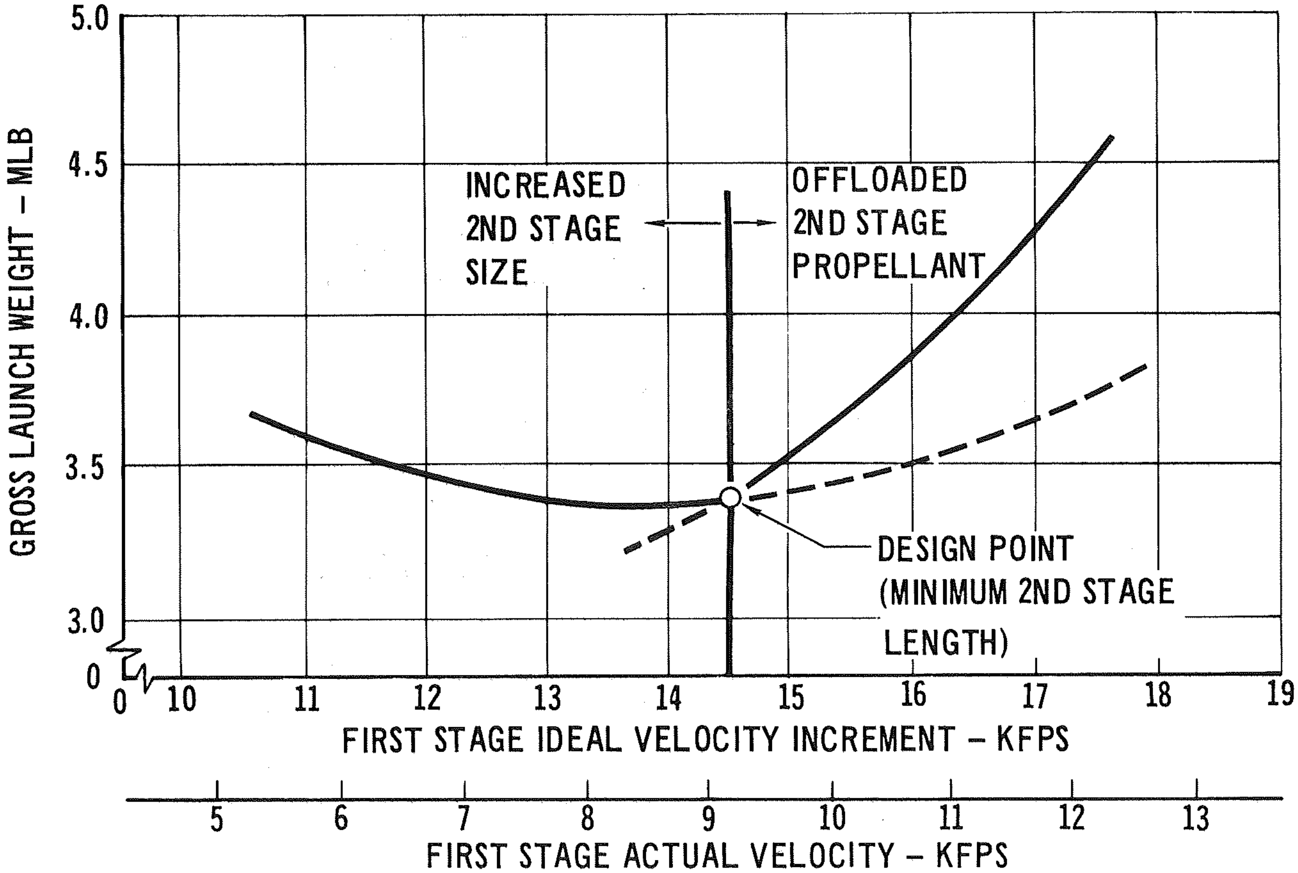


Figure 5-25

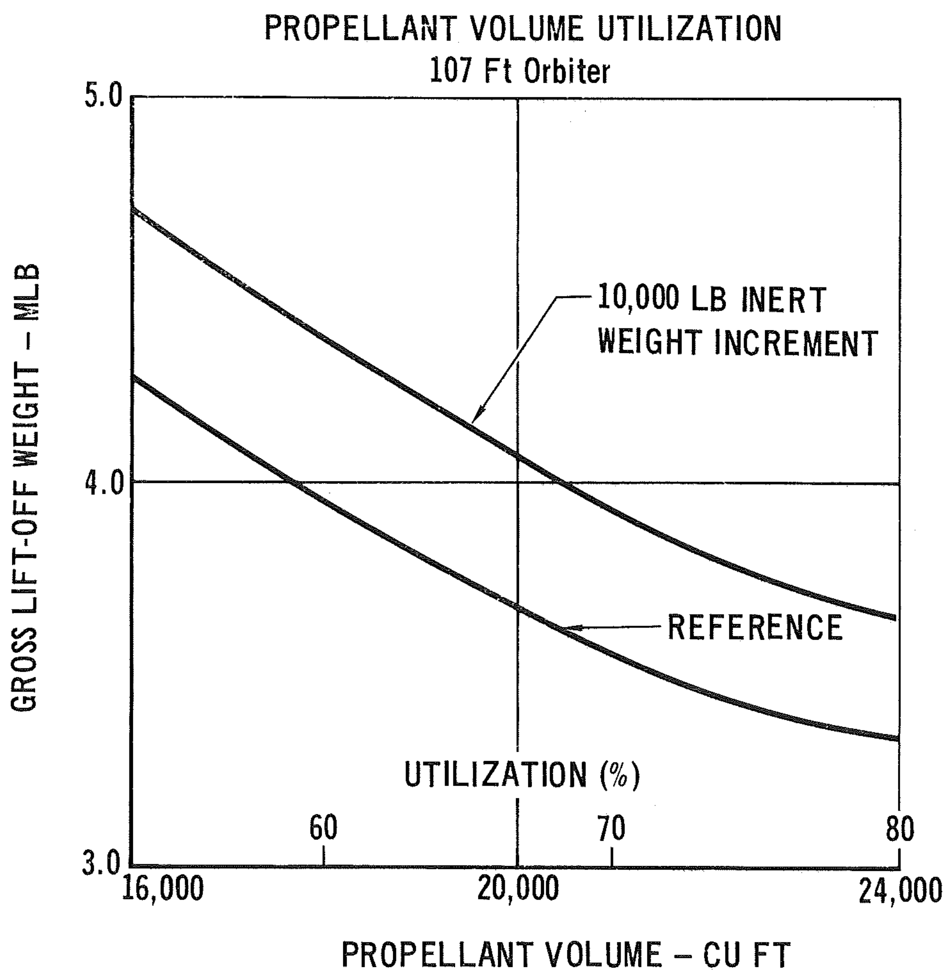

PROPELLANT VOLUME UTILIZATION
107 Ft Orbiter
5.0
4.0
3.0
GROSS LIFT-OFF WEIGHT – MLB
10,000 LB INERT
WEIGHT INCREMENT
REFERENCE
UTILIZATION (%)
60
70
80
16,000
20,000
24,000
PROPELLANT VOLUME – CU FT

Figure 5-26

Table 5-27

SPACECRAFT SUMMARY WEIGHT STATEMENT
Configuration: Carrier

Length - Ft.		170	195	200	210	209
Code	System					
1.0	Aerodynamic Surfaces	75910	99320	103500	115420	114520
2.0	Body Structure	109050	141060	149310	169190	173030
3.0	Induced Envir Protection	19870	24700	27090	30220	29980
4.0	Launch Recovery & Docking	17470	22320	23680	27380	28420
5.0	Main Propulsion	104180	130440	140760	170300	187050
6.0	Orient Control Sep & Ull	10880	11650	14250	17310	19880
7.0	Prime Power Source	2300	2300	2300	2300	2300
8.0	Power Conv & Distr	1985	1985	1985	1985	1985
9.0	Guidance & Navigation	666	666	666	666	666
10.0	Instrumentation	221	221	221	221	221
11.0	Communication	226	226	226	226	226
12.0	Environmental Control	470	470	470	470	470
13.0	(Reserved)	---	---	---	---	---
14.0	Personnel Provisions	200	200	200	200	200
15.0	Crew Sta Control & Pan	617	617	617	617	617
16.0	Range Safety & Abort	---	---	---	---	---
Subtotals		344045	436175	465275	536505	559565
17.0	Personnel	400	400	400	400	400
Subtotals		344445	436575	465675	536905	559965
18.0	Cargo	---	---	---	---	---
Subtotals		344445	436575	465675	536905	559965
19.0	Ordnance	220	220	220	220	220
20.0	Ballast	---	---	---	---	---
21.0	Resid Prop & Serv Items	8000	12140	12500	14870	13940
Subtotals		352665	448935	478395	551995	574125
22.0	Res Prop & Serv Items △3	7300	12200	8800	11800	4600
Subtotals		359965	461135	487195	563795	578725
23.0	Inflight Losses	△2	△2	△2	△2	△2
24.0	Thrust Decay Propellant	---	---	---	---	---
Subtotals		359965	461135	487195	563795	578725
25.0	Full Thrust Propellant	1470600	2209700	2332180	2750810	2685160
Subtotals		1830565	2670835	2819375	3314605	3263885
26.0	Thrust Prop Buildup	14300	17040	20150	25190	30250
27.0	Pre-Ignition Losses	---	---	---	---	---
28.0	LES Effective Wt	---	---	---	---	---
Totals	(Lb)	1844865	2687875	2839525	3339795	3294135

Notes & Sketches:

1. Items 9, 10, 11, 15 were artificially allocated from the Integrated Avionics weight estimate in order to accommodate this reporting format.

△2 Included in Item 21.

△3 Includes .25 x required cruise propellant.

Table 5-28

SPACECRAFT SUMMARY WEIGHT STATEMENT
Configuration: Orbiter

Code	Length - Ft. / Cargo Dimension - Ft. / Cargo Weight - Lb.	98 / 15 x 17 / 10000	107 / 15 x 30 / 25000	130 / 15 x 60 / 25000	130 / 15 x 60 / 50000	168 / 22 x 60 / 50000
Code	System					
1.0	Aerodynamic Surfaces	8440	10070	14870	14870	24820
2.0	Body Structure	45330	54300	79760	79760	132940
3.0	Induced Envir Protection	25810	30770	45430	45430	75890
4.0	Launch Recovery & Docking	7190	9200	12570	14050	21120
5.0	Main Propulsion	31360	38138	53920	56368	90090
6.0	Orient Control Sep & Ull	3810	4300	6300	6550	10110
7.0	Prime Power Source	2638	2638	2638	2638	2638
8.0	Power Conv & Distr	2380	2380	2380	2380	2380
9.0	Guidance & Navigation	1180	1180	1180	1180	1180
10.0	Instrumentation	555	555	555	555	555
11.0	Communication	135	135	135	135	135
12.0	Environmental Control	2130	2130	2130	2130	2130
13.0	(Reserved)	---	---	---	---	---
14.0	Personnel Provisions	200	200	200	200	200
15.0	Crew Sta Control & Pan	544	544	544	544	544
16.0	Range Safety & Abort	---	---	---	---	---
Subtotals		131702	156540	222612	226790	364732
17.0	Personnel	400	400	400	400	400
Subtotals		132102	156940	223012	227190	365132
18.0	Cargo	10000	25000	25000	50000	50000
Subtotals		142102	181940	248012	277190	415132
19.0	Ordnance	220	220	220	220	220
20.0	Ballast	---	---	---	---	---
21.0	Resid Prop & Serv Items	3090	3730	6020	6220	12270
Subtotals		145412	185890	254252	283630	427622
22.0	Res Prop & Serv Items 3	7100	14840	16950	16820	45110
Subtotals		152512	200730	271202	300450	472732
23.0	Inflight Losses	2	2	2	2	2
24.0	Thrust Decay Propellant	---	---	---	---	---
Subtotals		152512	200730	271202	300450	472732
25.0	Full Thrust Propellant	450920	529490	912220	919460	1945610
Subtotals		603432	730220	1183422	1219910	2418342
26.0	Thrust Prop Buildup	---	---	---	---	---
27.0	Pre-Ignition Losses	---	---	---	---	---
28.0	LES Effective Wt	---	---	---	---	---
Totals	(Lb)	603432	730220	1183422	1219910	2418342

Notes & Sketches:

1. Items 9, 10, 11, 15 were artificially allocated from the Integrated Avionics weight estimate in order to accommodate this reporting format.

2. Included in Item 21.

3. Includes propellant for .0075 x boost ΔV plus 500 fps orbit maneuver contingency

6.0 RELIABILITY AND SAFETY ANALYSIS

The mission success and crew safety required that stringent design guidelines be introduced and followed during the course of the study.

The failure modes of the major contributing subsystems in each mission phase were carefully analyzed. Redundant components, either passive or active, or functional path redundancies were incorporated. Single point failure areas were either eliminated during design or controlled in such a manner as to remove serious impact on the success of the mission or safety of the crew.

One of the basic groundrules followed during the conceptual design, was that mechanical, electro-mechanical and fluid subsystems should be fully redundant; i.e., the first critical component failure allows continuance of function and the second such failure permits safe subsystem operation. Avionics design requires a fail operational, fail operational, fail safe sequence, which can be readily accomplished with present day hardware and redundancy techniques. With the advent of large scale integrated circuits, this criteria should be met with even lesser penalties for weight, power consumption and complexity.

The total program concept requires operational performance of the shuttle vehicle to be comparable to that of commercial airlines. To achieve reliability and safety attained by the commercial airlines many of the tried and proven techniques of design, manufacture, operation and servicing have been incorporated into the shuttle program with only minor changes required because of unique operational environments.

6.1 Reliability Criteria and Goals - The reliability requirement is that the operational vehicle has a .95 probability of successfully completing the mission. With the redundancy techniques available and applied to the subsystem design and with the use of present day, hi-reliability components, this goal is feasible and can be achieved.

The operational requirements dictate a low cost, fully reusable spacecraft to be operated as an air transport, with minimum turnaround, minimum maintenance between missions. Using these operational constraints, subsystem designs were reviewed with the failure-tolerant criteria in mind, i.e., fail operational, fail safe sequence for mechanical and fluid subsystems, and fail operational, fail operational, fail safe sequence, for the integrated avionics subsystems.

6.2 Reliability Analysis - The reliability analysis of subsystem designs includes review of the design concept, the redundancy techniques employed in the design, and the effort to eliminate single point failures and control hazardous flight conditions. A vehicle gross failure analysis was also performed based on a typical mission, and the guidelines and groundrules listed below. The major mission events are outlined and the subsystem functions required for success of the event are listed along with the major modes of failure of the subsystem. The results of this analysis are presented in Table 6-1.

As an aid in completing this analysis, certain guidelines and groundrules were set forth which are typical for all mission phases.

a) General
   - Provisions will be made for safe mission termination from all phases of the mission.
   - Preflight and inflight subsystem checkout will be performed by an on-board system.
   - Each vehicle will have a two man crew; either crewman capable of flying the vehicle.
   - Vehicle subsystems designs will employ multiple redundancy concepts.
   - Electronic subsystems designed so that they remain operational after failure of two critical components and safe after failure of the third critical component.
   - All other subsystems will be designed so that following an initial failure the system remains operational; following the second failure, the system remains safe.
   - Landing sites will be located near the launch sites.

b) Pre-Launch Phase
   - Subsystem ground checkout equipment will be minimized.
   - Vehicle launch will be possible within 2 hours from a standby status.
   - Cargo loading and preflight stores loading (fuel, oxidizer, oxygen) shall be simplified operations.

c) Launch Phase
   - Minimum ground support and facilities will be required for launch. For rendezvous missions, the schedule will permit launching within a 60 second window.

Table 6–1

GROSS FAILURE ANALYSIS

MISSION EVENT	MAJOR SUBSYSTEM FUNCTIONS 1ST STAGE	2ND STAGE	MAJOR MODE OF FAILURE	IMPACT OF FAILURE ON MISSION/SAFETY	METHOD OF CONTROL OR MINIMIZING EFFECT
1. PRE-LAUNCH	ECLS	ECLS	UNCONTROLLED LEAKAGE–CABIN ATMOSPHERE CONTAMINATION	SCRUB MISSION NORMAL EGRESS	MONITOR PRESSURE DURING PAD OPERATIONS LOCATE HI-PRESSURE BOTTLES OUT OF CREW COMPARTMENT
SYSTEM CHECKOUT (BOTH VEHICLES)	G & N ELECTRONICS		LOSS OF VEHICLE CONTROL SIGNALS	SCRUB MISSION NORMAL EGRESS	REDUNDANT SYSTEMS IN EACH VEHICLE
	ELECTRICAL	ELECTRICAL	POWER INTERRUPTION – POWER LOSS – FIRE	SCRUB MISSION NORMAL EGRESS	RAPID EXIT – PURGE CABIN WITH INERT GAS
LOAD CARGO	N/A	AGE AND MECHANICAL LOADING DEVICE	CARGO DROPPED – DAMAGE TO S/C EXTERIOR – RADIOACTIVE CARGO HAZARDS	HOLD LAUNCH DETERMINE EXTENT OF DAMAGE – SCRUB MISSION	POSITIVE MEANS OF CARGO HANDLING – PROTECT S/C DURING LOADING – PROVIDE RADIATION PROTECTION OF S/C AND OCCUPANTS
FUELING OF S/C (BOTH STAGES)	FUEL AND OXIDIZER UMBILICALS – FUEL DUMPING AT DISCONNECTS		LEAKAGE – FAILURE TO SHUTOFF – AUTOGENOUS IGNITION OF FUEL	MISSION HOLD CRITICAL EVENT COULD DESTROY BOTH VEHICLES	STANDARD FUELING PROCEDURES – PURGE EQUIPMENT AVAILABLE AT LAUNCH SITE – CREW EGRESS AND ESCAPE MODES ACTIVATED
CREW BOARDING	CABIN ENVIRONMENT FINAL SYSTEMS CHECK		FAILURE TO SECURE HATCHES	HOLD LAUNCH – DEFUEL AND REPAIR LATCH MECHANISM	
IGNITION	PROPULSION BOOST ENGINES		FAILURE TO IGNITE – TO DEVELOP FULL THRUST	ABORT MISSION-ENGINE(S) SHUTDOWN NORMAL EGRESS	REDUNDANT PATHS PROVIDED FOR ENGINE IGNITION – HOLD-DOWN MODE
2. LAUNCH/ASCENT INITIAL BOOST	PROPULSION ENGINES		LOSS OF ENGINE LOW THRUST LEVEL	RELEASE AND CONTINUE MISSION – PREPARE TO SEPARATE AND ORBIT STAGE I AND RETURN TO LANDING SITE WITH STAGE II	CAPABLE OF SUCCESSFUL LAUNCH WITH AN ENGINE OUT – EACH ENGINE HAS THRUST POTENTIAL GREATER THAN MINIMUM REQUIRED
HOLD DOWN RELEASE	LAUNCH OPERATIONS AGE		HOLD DOWN RELEASE FAILS TO RELEASE	ENGINE SHUTDOWN	REDUNDANT RELEASE DESIGN
GUIDANCE AND CONTROL OF COMBINED VEHICLES AND SEPARATED VEHICLES	G & C (AVIONICS)		IMU MALFUNCTION PLATFORM DRIFT – LOSS OF SIGNAL TO COMPUTER	LOSS OF VEHICLE CONTROL IMPROPER ORBIT ENTRY	TRI-REDUNDANT AVIONICS PROVIDED IN EACH VEHICLE CROSS-OVER LINK BETWEEN VEHICLES
	ENGINES GIMBALLING		HARD OVER CONTROL PROBLEMS – FAILURE OF ENGINES TO REACT	LOSS OF VEHICLE CONTROL INTACT ABORT AFTER SEPARATION	REDUNDANT CONTROL SIGNALS FROM BOTH VEHICLES REDUNDANT CAPABILITY FOR CONTROL IN EACH SEPARATE VEHICLE
SEPARATION OF STAGES (1) AMD (2)	HYDRAULIC SUBSYSTEM, PYROTECHNICS AND SEQUENTIALS		FAILURE TO ACTUATE PIN PULLERS, THRUSTER MALFUNCTION HANG UP OF LINES, CABLES, STRUCTURE	CATASTROPHIC EVENT EJECT*	MULTIPLE REDUNDANT PATHS FOR SEPARATION DEVICES – REDUNDANT INITIATORS FOR THRUSTERS – EJECT FOR CREW SAFETY
	MECHANICAL RELEASE		BINDING, SEIZING MISALIGNMENT	CATASTROPHIC IF SEPARATION NOT COMPLETED	POSITIVE – ACTING RELEASE DESIGN – BACKUP SPRINGS
COMMUNICATIONS BOTH VEHICLES AFTER SEPARATION	GROUND CONTACT – BOTH STAGES COMMUNICATION WITH SECOND VEHICLE		SIGNAL LOSS FROM GROUND STATION INABILITY TO RECEIVE INFORMATION FROM OTHER VEHICLE	MINIMUM IMPACT ON MISSION SUCCESS DUE TO MULTIPLE REDUNDANT PATHS	MULTIPLE REDUNDANCY PROVIDED BOTH ACTIVE AND FUNCTIONAL PATHS
PROVISION OF BREATHABLE ATMOSPHERE & TEMPERATURE CONTROL IN CREW AND PASSENGER COMPARTMENTS – ECLS	O_2 SUPPLY AND CABIN TEMPERATURE CONTROL		LOSS OF O_2 SUPPLY (REDUCED PRESSURE)	FIRST STAGE – MINIMAL EFFECT– RETURN TO BASE SECOND STAGE – DETERMINE URGENCY OF LOSS CONTINUE MISSION OR ABORT	REDUNDANT O_2 SUPPLIES AVAILABLE REDUNDANT O_2 SUPPLIES AVAILABLE
3. ORBIT MANEUVERING STAGE II	N/A	ATTITUDE CONTROL ELECTRONICS AND THRUSTERS	FAILURE OF ATTITUDE CONTROL – ELECTRONICS TO PROPERLY SEQUENCE THRUSTERS	INABILITY TO MAINTAIN PROPER ATTITUDE – LOSS OF FIX ON TARGET – USE BACK UP MODE OR EFFECT REPAIR	MANUAL OVERRIDE TO CONTROL – THRUSTER REDUNDANCY PROVIDED
		FUEL/ OXIDIZER SUPPLY SYSTEMS	EXCESSIVE LEAKAGE TANK OR LINE RUPTURE	ABORT MISSION – SWITCH TO ALTERNATE SUPPLY FOR SAFETY-ISOLATE LEAKAGE	REDUNDANT SUPPLY SOURCES AND REDUNDANT LINES RUN ON OPPOSITE SIDES OF THE FUSELAGE

*EJECTION SEATS OR POD ESCAPE PROVIDED RDT&E FLIGHTS ONLY

Table 6–1 (Continued)

GROSS FAILURE ANALYSIS (Continued)

MISSION EVENT	MAJOR SUBSYSTEM FUNCTIONS 1ST STAGE	2ND STAGE	MAJOR MODE OF FAILURE	IMPACT OF FAILURE ON MISSION/SAFETY	METHOD OF CONTROL OR MINIMIZING EFFECT
		CABIN ATMOSPHERE AND TEMPERATURE CONTROL	LOSS OF CABIN PRESSURE BY FLOW RESTRICTION – RUPTURE OF CABIN WALLS – PRESSURE VALVE MALFUNCTION	PREPARE TO ABORT MISSION DETERMINE EXTENT OF MALFUNCTION AND ACT ACCORDINGLY	SECONDARY O_2 SUPPLY AVAILABLE EMERGENCY SUITS AVAILABLE TO CREW
4. DE-ORBIT STAGE II	N/A	ATTITUDE HOLD FOR RETRO	ATTITUDE CONTROL LOSS – DUE TO ELECTRONICS FAILURE	COULD BE CATASTROPHIC	REDUNDANT ACS PACKAGES PLUS MANUAL BACKUP FOR CONTROL
			THRUSTER MISFIRING FAILURE TO RETRO AT PROPER TIME OR ATTITUDE	PROLONGED ENTRY PERIOD – MISS ENTRY WINDOW FOR RECOVERABLE LANDING AT PLANNED SITE	MANUAL CONTROL BACK-UP TO REDUNDANT ELECTRONICS
	N/A	RETRO MOTORS FIRE	FAILURE TO FIRE AT REQUIRED THRUST FUEL EXPENDED	PROLONGED ENTRY PERIOD – MISS ENTRY WINDOW FOR RECOVERABLE LANDING AT PLANNED SITE	FUNCTIONAL REDUNDANCY PROVIDED IN ENGINES AND CONTROLS RESERVE FUEL SUPPLY PROVIDED FOR RETRO ONLY
5. CRUISE TRANSITION	JET ENGINE DEPLOYMENT		UNSYMMETRICAL DEPLOYMENT, SEIZING BINDING OF MECH LINKAGES –	LOSS OF S/C POSSIBLE LOSS OF CRUISE AND GO-AROUND CAPABILITY	REDUNDANT DEPLOYMENT METHODS– AERO AID ONCE DEPLOYMENT STARTS
(BOTH STAGES)	ENGINE IGNITION		FAILS TO TURN-OVER FUEL LINE RESTRICTION	LOSS OF S/C POSSIBLE LOSS OF CRUISE AND GO-AROUND CAPABILITY	AIR START WITH CARTRIDGE BACKUP
	WING DEPLOYMENT		UNSYMMETRICAL DEPLOYMENT – FAILS TO FULLY EXTEND –DOES NOT LOCK	LOSS OF S/C DUE TO HIGH LANDING SPEEDS REDUCED SUBSONIC L/D	REDUNDANT ACTUATORS ACTING ON BOTH WINGS SIMULTANEOUSLY
	LANDING GEAR EXTENSION		DOORS FAIL TO OPEN LOSS OF HYDRAULIC ACTUATOR FOR GEAR EXTENSION	S/C DAMAGE MISSION SUCCESS DEGRADED	REDUNDANT ACTUATORS FOR DOORS AND GEAR
			GEAR FAILS TO LOCK DOWN	LOSS OF CREW POSSIBLE – S/C LOSS	PYRO BACK UP FOR DOOR REMOVAL
6. LANDING (BOTH STAGES)	ATTAIN PLANNED LANDING AREA		FAILURE TO REACH PLANNED LANDING SITE	HARD LANDING ON UNPREPARED SURFACE	SUBSONIC CRUISE CAPABILITY – ALTERNATE SITES PROVIDED
	LANDING GEAR ACTUATED		LANDING GEAR HANG UP OR BUCKLES UNDER LOAD	EXTENDS REFURBISHMENT TIME – DAMAGE TO S/C	CRASH WORTHINESS OF S/C DESIGN
	FLIGHT CONTROLS		HYDRAULIC ACTUATION FAILS – BINDING, SEIZING	DEGRADED RELIABILITY MOMENTARY CONTROL CONDITION	QUAD-REDUNDANT CONTROLS ALL AXIS – FLY-BY-WIRE CAPABILITY
	INSTRUMENTATION		ALTIMETER, DIRECTIONAL INDICATION, BLIND LANDING	DEGRADED MISSION SUCCESS	GROUND CONTROL AS AID TO LANDING AVAILABLE
	COMMUNICATIONS		LOSS OF VOICE AND BEACONS	DEGRADED MISSION SUCCESS	REDUNDANT LOSSES
	TOUCHDOWN		LAND SHORT OR LONG ON RUNWAY – HARD IMPACT	DEGRADED MISSION SUCCESS JEOPARDIZE SAFETY OF CREW	GO AROUND CAPABILITY OR GLIDE EXTENSION USING JET ENGINES
	LANDING ROLL		VEER OFF RUNWAY	S/C DAMAGE	ENGINE STEERING DURING ROLL – BRAKING PROVIDED

- o Stage separation shall be accomplished with a minimum disturbance to both vehicles. A positive separation will be performed within a minimum time period, without damage to either stage.
- o The launch/ascent shall be capable of completion with one engine out.

d) On-Orbit Phase

- o All guidance and navigation functions shall be performed on board with simplified systems.
- o Approach and rendezvous maneuvers and/or cargo transfer to a space station, shall be an automatic, single operation for crew.
- o Transfer of a 15' diameter by 30' cylindrical payload is required.

e) De-orbit Phase

- o Multiple methods for performing de-orbit will be provided. The opportunity to return to a pre-selected site will be available once per 24 hours.

f) Landing

- o Go-around capabilities will be provided for both vehicles.
- o Landing speeds and the visibility will be provided comparable to present-day land based multi-engine aircraft.
- o All weather, capability for low visibility, automatic landings will be provided.

g) Post-Flight Phase

- o Both vehicles will be capable of horizontal takeoff, flight and landing from an emergency/alternate landing site.
- o Post landing vehicle systems safeing capability shall be provided to the flight crew.

6.2.1 Subsystem Apportionments - Reliability goals for subsystem designs were based on the mission success requirement (.95), and are presented in Table 6-2. The 12% contingency in the mission reliability goal over the mission requirement is available to account for operational and equipment details unknown at this point.

The goal was also apportioned to establish subsystem reliability requirements for both the carrier and orbiter vehicles during the typical mission. The results of this second task are shown in Table 6-3 for the carrier vehicle and Table 6-4 for the orbiter. The total subsystem requirements for mission success is a combination of the requirements of each phase. For example, the total ECLSS apportionment for the carrier would be .9984 and for the orbiter .9957.

Table 6-2

ILRV MISSION RELIABILITY APPORTIONMENT

Mission Reliability Required = 0.95

MISSION PHASES (MAJOR ONLY)	RELIABILITY APPORTIONMENT STAGE 1 (CARRIER)	STAGE 2 (ORBITER)	TOTALS
1. LAUNCH (BOTH STAGES)	0.995	0.995	0.990
2. SEPARATION (BOTH STAGES)	0.995	0.995	0.990
3. ON-ORBIT (STAGE 2)		0.990	0.990
4. ENTRY (STAGE 2)		0.990	0.990
5. LANDING (STAGE 2)		0.999	0.999
6. SUB-ORBITAL MANEUVER (STAGE 1)	0.998		0.998
7. LANDING (STAGE 1)	0.999		0.999
TOTAL	0.987	0.969	0.956

MISSION RELIABILITY DESIGN GOAL = 0.956

Table 6-3

RELIABILITY APPORTIONMENTS BY SUBSYSTEM
CARRIER RELIABILITY REQUIREMENTS = 0.9870

SUBSYSTEM IDENTIFICATION	MISSION PHASE AND MAJOR EVENTS			
	LAUNCH	SEPARATION	SUB-ORBITAL MANEUVER	LANDING
ECLSS	0.9986	0.9999	0.9999	1.0
ELECTRICAL POWER	0.9999	0.9999	0.9998	1.0
PROPULSION	0.9980	0.9999	0.9995	0.9999
GUIDANCE AND CONTROL	0.9990	0.9970	0.9999	0.9998
TELECOMMUNICATION	0.9997	0.9999	0.9997	1.0
LANDING SYSTEM	N/A	N/A	N/A	0.9996
ONBOARD CHECKOUT	0.9999	0.9999	0.9999	1.0
AERO CONTROL	N/A	N/A	0.9994	0.9998
SEQUENTIALS, HYDRAULICS, THRUSTERS AND MECHANICAL	0.9999	0.9985	0.9999	0.9999
RELIABILITY PHASE REQUIREMENTS	0.995	0.995	0.998	0.999

Table 6-4

RELIABILITY APPORTIONMENTS BY SUBSYSTEM
ORBITER RELIABILITY REQUIREMENTS = 0.969

SUBSYSTEM IDENTIFICATION	MISSION PHASE AND MAJOR EVENTS				
	LAUNCH	SEPARATION	ON-ORBIT	ENTRY	LANDING
ECLSS	0.9986	0.9999	0.999	0.9982	1.0
ELECTRIC POWER	0.9999	0.9999	0.9997	0.9995	1.0
PROPULSION	0.9980	0.9999	0.9949	0.9985	0.9999
GUIDANCE AND CONTROL	0.9990	0.9970	0.9980	0.9947	0.9998
TELECOMMUNICATION	0.9997	0.9999	0.9994	0.9996	1.0
LANDING SYSTEM	N/A	N/A	N/A	N/A	0.9996
ONBOARD CHECKOUT	0.9999	0.9999	0.9990	0.9999	1.0
AERO CONTROL	N/A	N/A	N/A	0.9989	0.9998
SEQUENTIAL, SEPARATION THRUSTERS, AND MECHANICAL DEVICES	0.9999	0.9985	N/A	0.9987	0.9999
RELIABILITY PHASE REQUIREMENT	0.995	0.995	0.990	0.990	0.999

The difference between the carrier requirements and that of the orbiter is due primarily to the number of systems functioning, the longer operational time for the orbiter, and the reentry environment that the carrier does not experience in the normal mission.

6.2.2 Subsystem Estimates - Preliminary subsystem designs have been examined for feasibility of concept, compliance with redundancy requirements, and single point failure elimination. To the extent permitted by design definition, preliminary reliability estimates have been made and compared to the subsystems' reliability requirement. An example of the method used in developing an estimate is shown for the electrical power subsystem (EPS) of the orbiter vehicle.

Subsystem Description - The Electrical Power Subsystem for the orbiter vehicle is a fully redundant, four stack fuel cell design with peak/emergency power requirements backed up by either of two batteries. Two primary DC busses operate in parallel with a bus tie relay providing the crossover path. Each primary bus distributes power to dual inverters and a secondary (non-essential) DC bus. Both pairs of inverters provide 3Ø, AC power to redundant AC primary busses with a bus-tie relay provision. All elements are easily isolated from the system in the event of malfunction by power relays. Inverter frequency signals provide reference of output back to both DC primary busses. Power distribution beyond this point, to avionics, propulsion, instrumentation and EC/LSS subsystems, is not included in this analysis.

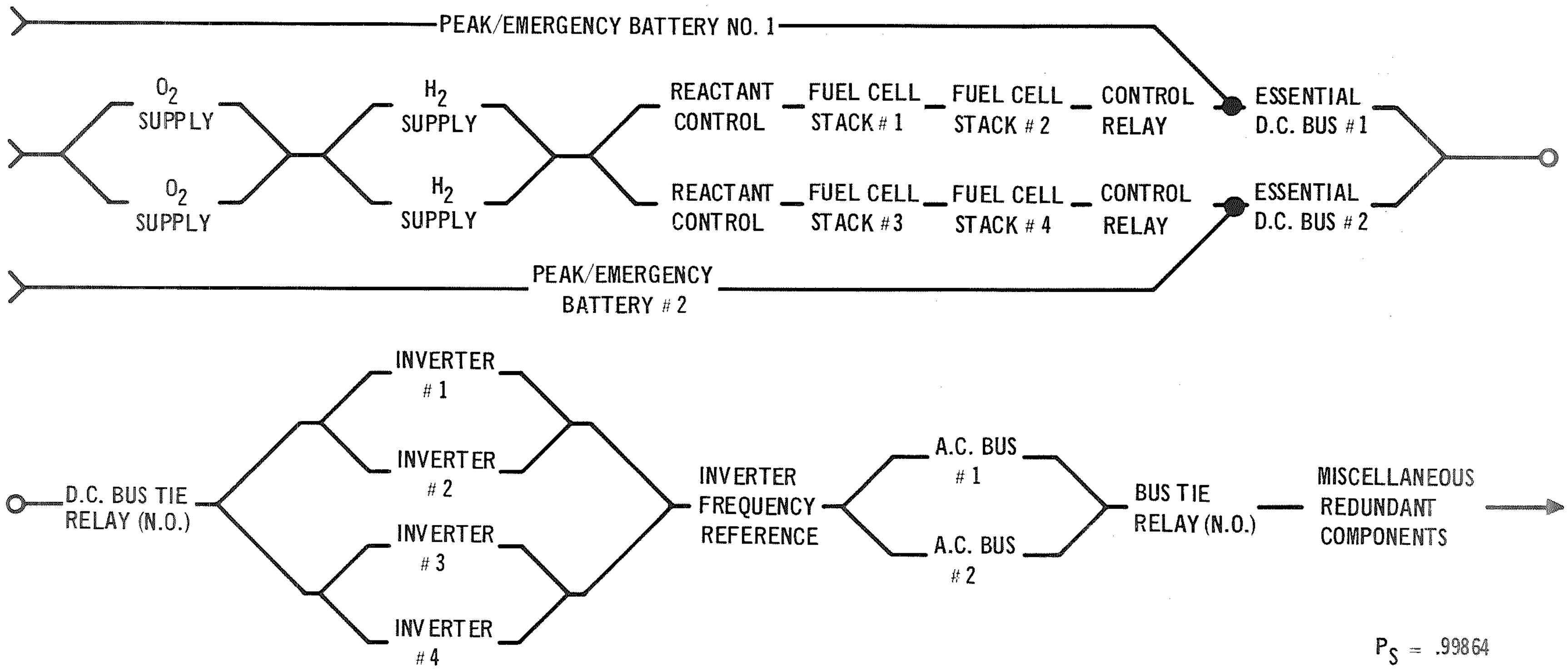
The preliminary reliability estimate for mission success is .99864 which closely approximates the total EPS subsystem goal established for the orbiter. Figure 6-1 is a reliability logic diagram of this system.

Table 6-5 lists the major components of the system and the failure rates or success probabilities used in this analysis. The equipment application factors (Kapp), listed in Table 6-6 were applied to the equipment with time considerations of launch/ascent equal to 8 hours, orbit 160 hours, and the remaining 2 hours for entry and landing. Table 6-7 is the element mission reliability total with all redundant paths considered in each elements' estimate.

The hardware associated directly with the landing operation and considered in the gross estimate of this subsystem is listed below.

- o Hydraulic subsystem that include reservoirs, plumbing actuators, servos, accumulators.
- o Aerodynamic control surfaces.
- o Flight instrumentation
- o Mechanical landing gear, wheels, brakes steering equipment.

RELIABILITY DIAGRAM
ORBITER – ELECTRICAL POWER SUBSYSTEM



P_S = .99864

Figure 6-1

Table 6-5

COMPONENT RELIABILITY DATA

COMPONENT	FAILURE RATE X 10^5 HOURS	PROBABILITY OF SUCCESS	DATA* SOURCE
1. O_2 TANKS AND PLUMBING	.20		M
2. H_2 TANKS AND PLUMBING	.20		M
3. REACTANT CONTROL	5.50		V
4. FUEL CELL STACK	12.30		V
5. CONTROL RELAY		0.99999978/CY	M
6. DC BUS		0.99999	M
7. DC TO AC INVERTER	10.70		M
8. AC BUS		0.99999	M
9. BUS TIE RELAY		0.99999934/CY	M
10. BATTERY (6.0 KWH EACH)		0.9950	M
11. MISCELLANEOUS COMPONENTS (WIRING, CONNECTORS, SWITCHES)		0.99999	
12. INVERTER FREQUENCY REFERENCE (INTERNALLY REDUNDANT)		0.99999	M

* DATA SOURCE:

M = MDAC EXPERIENCE

V = VENDOR DATA

Table 6-6

APPLICATION FACTORS (K_{APP})

EQUIPMENT TYPE	MISSION PHASE			
	LAUNCH	ORBITAL	ENTRY	LANDING
MECHANICAL	500	1	100	100
ELECTRO-MECHANICAL	100	1	20	1
ELECTRONIC	15	11	3	1

Table 6-7

ELEMENT RELIABILITY

COMPONENT	MISSION RELIABILITY ESTIMATE*
O_2 SUPPLY	.99980
H_2 SUPPLY	.99980
REACTANT CONTROLS	.99951
BATTERY (REDUNDANT)	.99995
RELAY (20 REQUIRED)	.99999
FUEL CELL REDUNDANT (BATTERY BACKUP)	.99995
INVERTER FREQUENCY REFERENCE	.99999
DC TO AC INVERTERS (REDUNDANT)	.99967
MISCELLANEOUS SWITCHES, WIRING AND CONNECTORS	.99999
BUS (AC AND DC) REDUNDANT	.99999
ESTIMATE TOTAL P_S =	.99864

*EQUIPMENT REDUNDANCIES INCLUDED IN ESTIMATE

The estimate for probability of successful operation of this subsystem is .99972.

6.3 Safety Analysis - A qualitative safety assessment has been made of the subsystem designs and operational requirements of the baseline vehicle. A preliminary identification of catastrophic and critical operational hazards for the candidate booster and orbiter vehicles was prepared.

As a part of the analysis, a comparison of safety considerations for a commercial transport and the ILRV spacecraft was made, based on a typical mission for each type system. The correlation between systems is close except for the differences in launch attitudes and the on-orbit and entry phase environments that the orbiter experiences. The comparison of the safety provisions for both systems in their normal operational mode is shown in Table 6-8.

Some key points followed during the safety analysis were averting the cascading effect of rocket engine/fuel system failures, the elimination of abort-forcing escape-precluding failures, and providing for ample warning time in the event of potentially catastrophic failures.

6.3.1 Goals and Guidelines - The crew-safety requirement is .999 or one loss per 1000 missions. This goal can be attained with current safety-of-operations criteria applied during the design and planning stage. In qualitative terms, the safety level for the ILRV spacecraft must approach that level exhibited by commercial transports. To accomplish this, several guidelines have been established and followed during the preliminary safety analysis.

- o Mission safety to be commensurate with current FAA regulations.
- o Identified hazards will be eliminated or reduced and controlled by use of current MDC commercial aircraft design practices and airline procedures.
- o Provisions are made for rapid on-pad egress and escape paths for crew and passengers.
- o Design must provide for rapid dump/usage of fuel following ascent phase abort.
- o Separation devices, such as pyrotechnics, mechanical pistons, hydraulic or electrical actuators and releases, are fully redundant and easily inspected or functionally checked prior to a mission.
- o Dual, triple and quad-redundancy techniques are employed in design, dependent upon criticality of function.

Table 6–8
AIRLINE VS SPACECRAFT SAFETY CONSIDERATIONS

AIRLINE OPERATIONS	SPACECRAFT PROVISIONS
1. GROUND OPERATIONS –	1. PRE-LAUNCH
EQUIPMENT CHECKOUT	FUNCTIONAL CHECKOUT – ALL SUBSYSTEMS
SYSTEM PERFORMANCE MONITORING	ONBOARD CHECKOUT SUBSYSTEM
MALFUNCTION DETECTION SYSTEM	MDS
CREW(NORMAL) EGRESS – NORMAL PASSENGER EGRESS	NORMAL TOWER EGRESS PROVISIONS
EMERGENCY ESCAPE (CREW) & PASSENGERS, HATCHES, CHUTES, & STEPS	SLIDE WIRE – ELEVATOR – MULTIPLE HATCHES SWING-ARM PICKUP
"SINGLE-SWITCH" SHUTDOWN CAPABILITY	COMPARIBLE CAPABILITY – ABORT SWITCH
FIRE-FIGHTING EQUIPMENT AVAILABILITY	GROUND BASED EQUIPMENT
–EXPLOSION PROTECTION PROVIDED GROUND CREW	BUNKERS AND VAULTS PROVIDED AT LAUNCH SITE
2. TAKE-OFF & CLIMB-OUT –	2. LAUNCH ASCENT (ROCKET ENGINES)
DEVELOP ENGINE THRUST PRIOR TO BRAKE RELEASE	HOLD-DOWN CAPABILITY ON PAD
ABORT PRIOR TO LIFT-OFF – BRAKE & SHUTDOWN	ENGINE SHUTDOWN & EGRESS FROM VEHICLE
ABORT AFTER LIFT-OFF – GO-AROUND, ALTERNATE SITE LANDING	SEPARATION-INTACT ABORT MODE – BOTH STAGES
ENGINE-OUT CAPABILITY, TAKE-OFF, CLIMB W/O FLAPS	ONE ENGINE OUT – CONTINUE MISSION
	TWO ENGINES OUT – ABORT MISSION
REDUNDANT FLIGHT CONTROLS & PILOTS	TRIPLE REDUNDANT AVIONICS – EITHER CREWMAN
REDUNDANT COMMUNICATION LINKS	TRIPLE REDUNDANT AVIONICS
REDUNDANT GUIDANCE INSTRUMENTATION	TRIPLE REDUNDANT AVIONICS
GROUND-BASED FLIGHT STATUS CONFIRMATION	GROUND COMMUNICATIONS AVAILABLE – TRACKING & VOICE
FUEL DUMP PROVISIONS	DESIGN SAFETY MARGIN ADEQUATE – CONSTRUCTION
STRUCTURAL INTEGRITY OF FUSELAGE DURING CRASH	TECHNIQUE RINGS & LONGERON
FUEL & HYDRAULIC SUPPLIES LOCATED REMOTE FROM PASSENGER COMPARTMENT FOR WHEELS-UP LANDING	VOLATILE STORES LOCATED EXTERNAL TO CREW & PASSENGER COMPARTMENT
CABIN PRESSURE & O_2 SUPPLY – INDIVIDUAL O_2 MASKS	REDUNDANT O_2 SUPPLIES – SPACE SUITS AVAILABLE
	REDUNDANT BATTERIES, BUSSES, WIRING, & FUEL CELL SECTION
REDUNDANT POWER SUPPLIES	RESTRAINT STRAPS, CONTOURED SEATS COUNCHES
RESTRAINT SYSTEM PROVIDED (CREW & PASSENGER)	PROVIDED
3. INFLIGHT –	3. ON-ORBIT & SUB-ORBITAL MANEUVERING
ENGINE-OUT CRUISE CAPABILITY –	ENGINE-OUT CRUISE CAPABILITY
REDUNDANT FLIGHT INSTRUMENTS –	TRI-REDUNDANT INSTRUMENTATION
REDUNDANT FLIGHT CONTROLS & PILOTS	COMPARABLE TO COMMERCIAL TRANSPORT
GROUND STATION DIRECTIONAL AIDS – TRAFFIC CONTROL	GROUND CONTROL & NAVIGATIONAL AIDS – TRAFFIC CONTROL
FUEL-TANKS SEPARATED (CROSS-FEED PROVIDED)	100% REDUNDANCY PROVIDED IN FUEL SUPPLY
ALTERNATE BASES FOR EMERGENCY LANDINGS	ALTERNATE LANDING SITES – UNDER STUDY
REDUNDANT ENGINE – DRIVEN GENERATORS, FUEL PUMPS, ETC.	FUEL PUMPS CONSIDERED AS PART OF ROCKET ENGINE INTERNAL REDUNDANCY PROVIDED
ALTERNATE PATHS OF COMMUNICATION	S-BAND COMM. SYSTEM & UHF & VHF SYSTEMS
SYSTEM PERFORMANCE MONITORING –	ON-BOARD CHECKOUT PLUS REDUNDANT INSTRUMENTS
	EJECTION SEATS OR ESCAPE CAPSULE PROVIDED FOR CREW RDT&E FLIGHTS
4. APPROACH & LANDING	4. APPROACH & LANDING
ENGINE THROTTLING – GLIDE EXTENSION	ENGINE (JET) THROTTLING FOR GLIDE EXTENSION
GLIDE EXTENSION – FLAPS – SPOILERS	AERO LIFT PROVIDED BY VEHICLE SHAPE
FUEL SUPPLIED FOR GO-AROUND & OR	WHEEL BRAKING AND THRUST REVERSAL OF JET ENGINES
ALTERNATE BASE SELECTOR	COMPARABLE STEERING TO COMMERCIAL VEHICLES
GROUND CONTROL OF GLIDE ANGLE & PATH DIRECTION	FUEL SUPPLY AVAILABLE FOR GO-AROUND
BRAKING – THRUST REVERSERS	GROUND CONTROL FOR LANDING ASSIST
STEERABLE NOSE WHEEL LOCK UNLOCK	EMERGENCY GROUND ESCAPE PATHS PROVIDED
EMERGENCY EGRESS – HATCHES, DOORS, STEP, CHUTE	QUICK OPENING HATCHES, DOORS, STEPS & CHUTES

- o A single failure should not cause mission abort and preclude escape.
- o An inadvertant abort initiation will not result from a single failure.
- o Explosives and hi-energy storage facilities will be located remotely from crew compartments.
- o Abort, escape and recovery paths will be available to crew members at all times.

6.3.2 Design Evaluation for Safety - The evaluation of available subsystem designs was completed in conjunction with the inspection of the hazardous events that must occur during the normal mission.

A gross failure analysis was made to identify the major modes of failure of the operating subsystems for each mission phase. The impact of the failure on mission success or crew safety and the design methods for controlling or minimizing the effect of the failure are included in the Gross Vehicle Failure Analysis, Table 6-1.

Single point hazard areas are identified in Table 6-9, to pin-point critical components of the subsystem's operating during certain emergency situations. For example, the loss of electrical power emergency and the identification of the critical components, item (5) of Table 6-9, provide a basis for design correction action options shown in the following chart, Table 6-10.

Table 6–9

CRITICAL COMPONENTS IDENTIFICATION

EMERGENCY TYPE	MAJOR SUBSYSTEM(S) OPERATING	PRIMARY CAUSE OF EMERGENCY	CRITICAL COMPONENTS REQUIRING DESIGN CONCERN
1) FIRE	ELECTRICAL POWER	• ELECTRICAL ARCING DURING SWITCHING, SHORTS, OPEN WIRES	• WIRING, BATTERIES, BUSSES, SWITCHES & POWER CONSUMING DEVICES
	ECLSS	• SUPPORTS COMBUSTION BY LEAKAGE OR NORMAL CABIN O_2 SUPPLY	• O_2 SUPPLY TANKS SHUT-OFF VALVES & PLUMBING
2) NON-HABITABLE ENVIRONMENT	ECLS	• LOSS OF O_2 SUPPLY • LOSS OF PRESSURE & TEMPERATURE & HUMIDITY CONTROL • ATMOSPHERE CONTAMINATION • SOLAR RADIATION	• O_2 TANKS, VALVES, PLUMBING • TANKS, VALVES, PLUMBING, COLD PLATES, BOILERS, FILTERS • FILTERS, EMERGENCY O_2 SUPPLY • STRUCTURAL SHIELDING, LOCATION OF PERSONNEL
3) EXPLOSION	ALL PROPULSION SUBSYSTEM	• FUEL TANK OR OXIDIZER TANK RUPTURE, RUPTURE, PLUMBING LEAKAGE • FUEL & OXIDIZER TRANSFER	• SUPPLY TANKS, VALVES, PLUMBING, JOINTS • TRANSFER HOSES, LINES, VALVES, PUMPS
	ECLS	• SUPPLY TANK RUPTURE, EXCESSIVE HI-PRESSURE LEAKAGE	• TANKS, VALVES, LINES
4) LOSS OF ATTITUDE CONTROL	ATTITUDE CONTROL ELECTRONICS	• LOSS OF REFERENCE • POWER FAILURE	• GYROS, IMU, COMPUTER, DISPLAYS • POWER SUPPLY, WIRING BUSS CONNECTIONS
	ATTITUDE CONTROL & MANEUVER PROPULSION	• THRUSTER FAILURE • FUEL DEPLETION	• THRUSTER, VALVES, PLUMBING • TANKS, PLUMBING, S/O VALVES
	BOOST PROPULSION	• ENGINE FAILURES • LO-THRUST DEVELOPED • HARD-OVER GIMBALING	• FUEL PUMPS, COMPRESSOR BEARINGS & BLADES • FUEL CONTROL, NOZZLE CONTROL, THROTTLING • GIMBAL ACTUATORS, MECHANICAL LINKAGES
5) LOSS OF ELECTRICAL POWER	ELECTRICAL POWER SUPPLY & DISTRIBUTION	• BATTERY FAILURE • SHORT CIRCUIT • LOSS OF FUEL CELL GAS SUPPLIES • CONTROL RELAY OPEN	• BATTERIES, CONNECTORS, POWER BUSS RELAYS • SWITCHES, WIRING, CONNECTORS, INVERTERS • TANKS, PLUMBING, CELLS
6) MECHANICAL SYSTEMS MALFUNCTION	SEPARATION SYSTEM	• BINDING OF LINKAGES • GAS GENERATOR FAILURE	• MECHANICAL ATTACH POINTS, BEARING SURFACES • GAS GENERATORS, BACK-UP SPRINGS
	HATCH LATCHING	• FAILURE TO LATCH & SEAL CREW COMPARTMENT DURING ASCENT PRESSURE CHANGE – FAILURE TO UNLATCH IN EMERGENCY	• LATCHES, SEALS, LOCKING MECHANISMS, DOORS, PORTS, SERVICE HATCHES, GEAR DOORS
	LANDING GEAR EXTENSION	• LOSS OF HYDRAULIC POWER • FAILURE OF GEAR TO POSITION & LOCK	• HYDRAULIC SUPPLY, PLUMBING, ACTUATORS, SEALS • DOWNLOCK MECHANISM, PIVOT BEARINGS
	CONTROL SURFACES	• BINDING/SEIZING OF SURFACE • LOSS OF CONTROL SURFACE THRU HI-TEMPERATURE EXPOSURE	• BEARINGS, SHAFTS, LINKAGES, ACTUATORS • THERMAL PROTECTION AND STRUCTURE

Table 6–10

CORRECTIVE ACTION OPTIONS FOR EPS FAILURES ON ORBITER VEHICLE

Options For Corrective Action

FAILURE TYPE	ASCENT	RETURN PHASING	DESCENT
LOSS OF MAIN BUS POWER	• SWITCH TO REDUNDANT BUS	• SWITCH TO REDUNDANT BUS. DEFER RETROGRADE FOR A MORE DESIRABLE POINT WITHIN THE EXISTING ORBIT.	• ISOLATE DEFECTIVE CIRCUIT • ISOLATE DEFECTIVE ELEMENT AND CONTINUE NORMAL OPERATION WITH SHORTENED POSTLANDING CAPABILITY
LOSS OF MAIN BUS POWER	• SWITCH TO REDUNDANT ESSENTIAL BUS	• SWITCH TO REDUNDANT BUS. DEFER RETROGRADE FOR A MORE DESIRABLE POINT WITHIN THE EXISTING ORBIT	• ISOLATE DEFECTIVE CIRCUIT • ISOLATE DEFECTIVE ELEMENT AND CONTINUE NORMAL OPERATION WITH SHORTENED POSTLANDING CAPABILITY
LOSS OF OUT-PUT OF FUEL CELL	• ISOLATE DEFECTIVE UNITS AND CONTINUE ASCENT • ABORT MISSION FOR DE-SIRABLE RETURN TRAJECTORY.	• ISOLATE DEFECTIVE UNITS AND CONTINUE RETURN PHASING	• FUEL CELLS USED DURING THIS PERIOD BACKED-UP WITH BATTERIES ON LINE.

6.3.3 Critical Subsystem Analysis - A most hazardous required function in the normal ILRV mission is the separation of the two stages.

For this study, the aerodynamic interface between the two bodies and firm requirements for propulsion and G&C avionics during the separation have not been clearly defined. Once these problems have been analyzed further, the event may become a state-of-the-art function that has been accomplished with slight variation on many manned and unmanned spacecraft flights.

The structural attach points are assumed to have a reliability of unity, i.e., they are able to withstand all environmental factors associated with the launch without degradation. The mechanical separation devices such as hydraulic pin pullers, actuators, gas operated pistons, pyrotechnic bolts or MDF are fully redundant and have operated very successfully on previous programs. Trades performed to date for separation methods favor the hydraulic or electric pin pullers concept. Separation propulsion is provided by four, 4,000 lb. thrusters mounted on the carrier and firing normal to the plane of separation.

A preliminary reliability estimate for this function exceeds the established goal of 0.990 by an order of magnitude, 0.999.

A second critical subsystem is the launch/ascent propulsion which consists of ten rocket engines mounted on the boost vehicle and two engines on the orbiter operating in series burn. Both vehicles have the capability of completing the mission with a single engine out, and at some altitudes, with 20% overspeed for the remaining engines, the booster can perform satisfactorily with a double engine malfunction. With the pad hold down capability, approximately 30% of the normal engine start failures are eliminated. This capability provides assurance that all engines are operating satisfactorily prior to launch, or if not, the mission may be scrubbed with minimum risk. Quick egress and escape provisions have been made for crew and passengers to reduce the personnel risks associated with fueling, engine ignition, and system checkout during the pre-launch phase.

Based on vendor information for engines in the 1/2 million pound thrust class, the reliability range for operational engines will lie between .992 and .999 for start. The catastrophic failure rate is estimated to be less than 1% of the normal operating rate, or the probability of not experiencing a catastrophic engine failure will range between .99992 and .99999 per engine. Although the use of ten booster engines increases this risk probability by an order of magnitude, the goal can be attained

For booster reliability, the mid point of the range of single engine reliability was used (.9955) to estimate the probability of launch success with hold-down and engine-out capabilities. The estimate is .9991, which is an improvement of the launch reliability goal of (.995) by approximately 80%.

7.0 BASELINE DESIGN DEFINITION

Designs of the selected baseline vehicles are defined in the following paragraphs. The first and second stages are defined individually with internal arrangement and detail definitions. The structural interface between the two stages is treated in a separate section. Major trade studies affecting vehicle performance are described in other sections of this report. Some of the subsystem analyses are described and illustrated in this section. Selection of some of the systems was based on a conventional approach to minimize the secondary trade study requirements. This permitted definition, to a level of detail commensurate with the scope of the study, of all the major subsystems.

7.1 Carrier Design Definition - Definition of the design elements of the carrier are separated into three major sections. An overall general arrangement, including the structural concept, is followed by a description of propulsion systems. The vehicle equipment not defined by structure and propulsion systems is presented in the third section. The selected baseline systems are defined in each case although some of results are based on analysis of interim vehicles. Derivations not shown in other sections are also presented for definition of some of the components.

7.1.1 General Arrangement Carrier - The structure and subsystems arrangement for the carrier are shown in Figure 7-1. The planform view shows the external configuration on the right hand side of centerline and the internal arrangement on the left side.

The vehicle body contains a dual lobe boost propellant tank with the oxidizer forward and the hydrogen in the aft portion. The walls of this tank, with integrally machined stringers and ring flanges, form the primary structural skin for the vehicle body. The area between the tank skin and outer moldline contains the vehicle thermal protection system, body rings, and external structural supports. The forward end of the body is formed by an extension of the tank walls and provides a transition to the nose radius. This volume encloses the pressurized crew cabin, avionics, E.C.L.S. and power supply. The nose landing gear is housed in the forward end of the vehicle in the cavity between the oxidizer tank lobes. Boost engines, thrust structure, propellant utilization system and auxiliary power system are located aft of the boost propellant tankage and are supported by a structural extension of the tankage. Thrust loads from

CARRIER GENERAL ARRANGEMENT

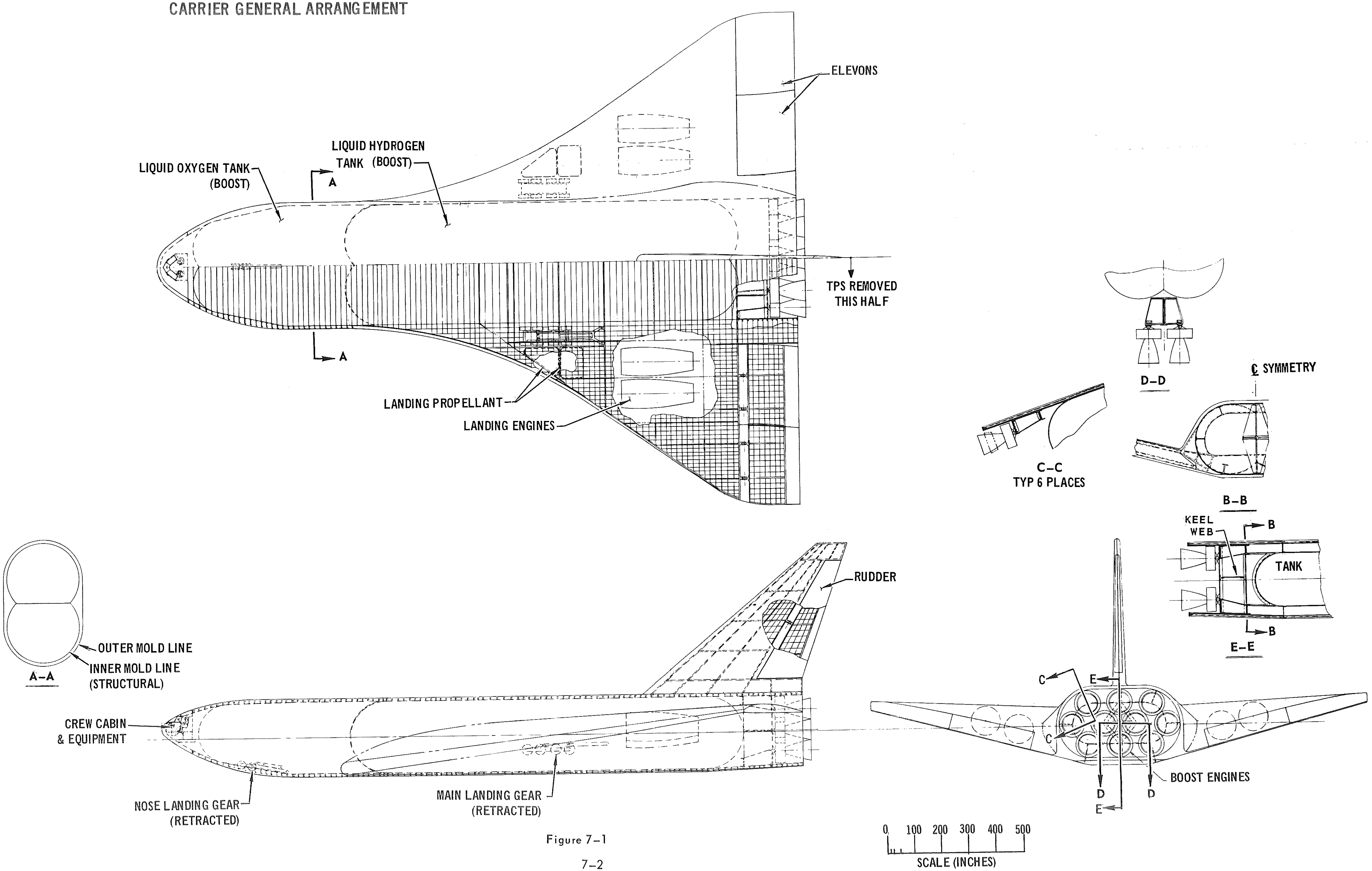


Figure 7-1

six peripheral boost engines are transferred to the skin by longerons. The overturning moment introduced by this reaction is restrained by two main body rings. Thrust loads from four interior boost engines are transferred into the skin and the web between the tank lobes. Induced moments are reacted by the same rings as for the peripheral engines. The forward thrust structure ring is located in line with, and is a continuation of, the wing spars at the forward side of the elevons.

The landing engines, landing propulsion system and main gear are enclosed in the wing, eliminating the need for separate fairings on the body or wing to enclose these systems. The forward wing spar lies along a constant percent of the chord. The other spars are normal to the body sides. This provides a transition to the body rings and a load path for wing carry-through without the necessity for penetrating the propellant tank walls with primary structure. The main landing gears are located on the inboard side of the wings to keep them as low as possible minimizing the landing gear length required to provide a 12° touchdown angle of attack. The landing engines are centered longitudinally at the maximum wing thickness and at an inboard location that still clears the landing gear in the deployed condition. The landing propellant tanks are located as far forward in the wing as possible to help achieve vehicle balance.

The dorsal fin spars intersect structural rings. The rudder hinge support spar ties into the forward thrust structure ring.

7.1.2 Carrier Propulsion Systems - The principal propulsion systems study was that of the boost propulsion system. This is described first, followed by the secondary propulsion systems. These systems are depicted in Figure 7-2.

Boost Propulsion - The paragraphs below treat the carrier vehicle cryogenic systems tankage configuration, fill and feed systems, vent systems, pneumatics, and tank pressurization systems.

The LOX tank configuration consists of two parallel and intersecting cylinders joined to two intersecting cones whose forward domes are formed by two intersecting externally convex hemispheres and whose aft common bulkhead is formed by two intersecting externally concave hemispheres. Access to the LOX tank interior is provided by a removable cover in each of the forward domes. Analysis indicates that insulation is not required for the main propulsion LOX tank.

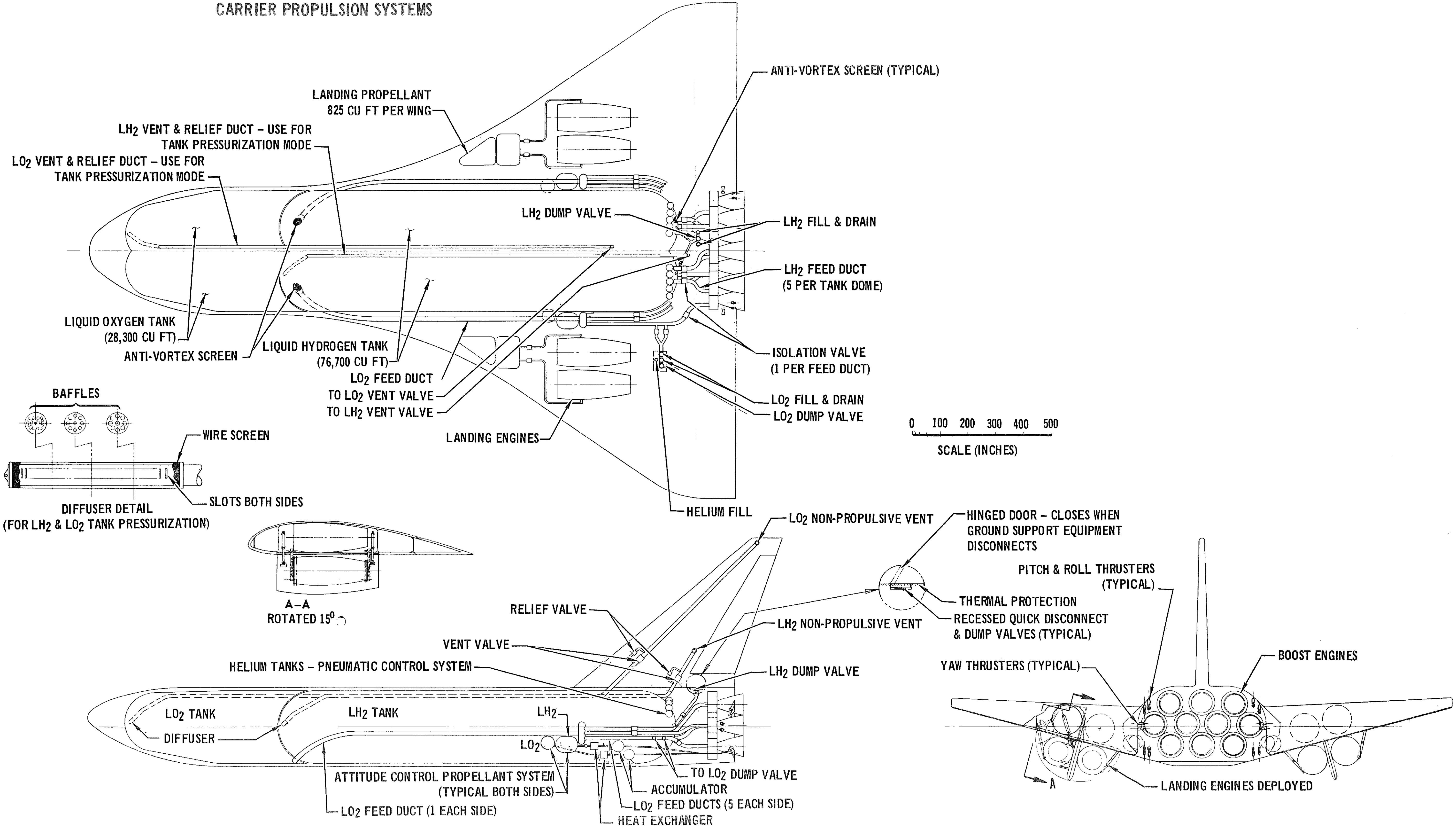

ANTI-VORTEX SCREEN (TYPICAL)
LANDING PROPELLANT
825 CU FT PER WING
LH_2 VENT & RELIEF DUCT - USE FOR
TANK PRESSURIZATION MODE
LO_2 VENT & RELIEF DUCT - USE FOR
TANK PRESSURIZATION MODE
LH_2 DUMP VALVE
LH_2 FILL & DRAIN
LH_2 FEED DUCT
(5 PER TANK DOME)
LIQUID OXYGEN TANK
(28,300 CU FT)
ANTI-VORTEX SCREEN
LIQUID HYDROGEN TANK
(76,700 CU FT)
LO_2 FEED DUCT
TO LO_2 VENT VALVE
TO LH_2 VENT VALVE
ISOLATION VALVE
(1 PER FEED DUCT)
LO_2 FILL & DRAIN
LO_2 DUMP VALVE
0 100 200 300 400 500
SCALE (INCHES)
BAFFLES
WIRE SCREEN
LANDING ENGINES
SLOTS BOTH SIDES
DIFFUSER DETAIL
(FOR LH_2 & LO_2 TANK PRESSURIZATION)
HELIUM FILL
LO_2 NON-PROPULSIVE VENT
HINGED DOOR - CLOSES WHEN
GROUND SUPPORT EQUIPMENT
DISCONNECTS
PITCH & ROLL THRUSTERS
(TYPICAL)
A-A
ROTATED 15°
RELIEF VALVE
THERMAL PROTECTION
RECESSED QUICK DISCONNECT
& DUMP VALVES (TYPICAL)
LH_2 NON-PROPULSIVE VENT
VENT VALVE
HELIUM TANKS - PNEUMATIC CONTROL SYSTEM
LH_2 DUMP VALVE
YAW THRUSTERS (TYPICAL)
BOOST ENGINES
LO_2 TANK
LH_2 TANK
LH_2
DIFFUSER
LO_2
A
ATTITUDE CONTROL PROPELLANT SYSTEM
(TYPICAL BOTH SIDES)
TO LO_2 DUMP VALVE
ACCUMULATOR
LANDING ENGINES DEPLOYED
LO_2 FEED DUCT (1 EACH SIDE)
LO_2 FEED DUCTS (5 EACH SIDE)
HEAT EXCHANGER

Figure 7–2

The tank volume of 28,400 cu. ft. includes a 10 percent allowance to account for ullage volume and propellants reserve. The total weight of the LOX loaded aboard includes.5 percent that is set aside for unusable propellants, startup propellants, performance reserves, and pressurant gas. A series of annular ring baffles are recommended to induce LOX slosh damping.

Consideration was given to the O_2/H_2 bulkhead configuration from a propellant feed standpoint, residuals, and strength. The ideal strength configuration of the common bulkhead was that of a concave dome relative to the LOX tank. The final configuration selected, however, was that of a convex common bulkhead, relative to the LOX tank, which was the result of the heavy influence of the propellant feed capability and the effect upon residuals. This configuration permitted almost 100% propellant utilization, whereas the concave bulkhead would have presented a penalty.

The LOX feed system consists of two main feed ducts (16 inch ID) that run from their LOX tank interfaces approximately 92 feet down the vehicle. At this point, they are manifolded into 10 feed ducts (12 inch ID) that run the remaining 42 feet to each engine. The individual engine feed lines were designed to provide sufficient volume (40 ft^3) to attenuate the high pressure -- two phase backsurge that results from an emergency engine shutdown while operating near 100% thrust.

The LOX tank fill and drain system ties into one of the 12 inch engine feed ducts near its engine interface. Filling and draining of the tank will be accomplished through a checking quick disconnect (QD) located near the vehicle aft end. The QD will be recessed and covered with a hinged door which is torsion loaded, opens when GSE is connected and automatically closes whenever GSE is disconnected. This protects the QD from the high heat input which occurs on the exterior surfaces. This is necessary to inhibit heat from being conducted to the LOX feed line and also to protect the QD for reuse. A combined fill and drain shutoff and relief valve will be downstream of the QD to isolate the unusable LOX trapped in the fill and drain line from the LOX feed line. A normally closed dump valve will be located parallel to the QD and is for the purpose of dumping propellants subsequent to completion of burn. It is assumed that the engine flow resistance would be excessive and consequently high liquid flowrates would not be available through the engines.

The two main propellant feed ducts will have antivortex screens located at the feed duct interface with the propellant tank. Vortexing in the feed lines is undesirable since it inhibits mass flow and could result in LOX pump cavitation. Since propellant loading will be accomplished through one of the antivortex screens, its design must include a "flip-top" feature with a baffle. This fliptop design allows unfiltered tank filling through the feed lines and the integral baffle inhibits propellant spraying during fill operations. Subsequently, the fliptop screen performs a filtering function during stage burn.

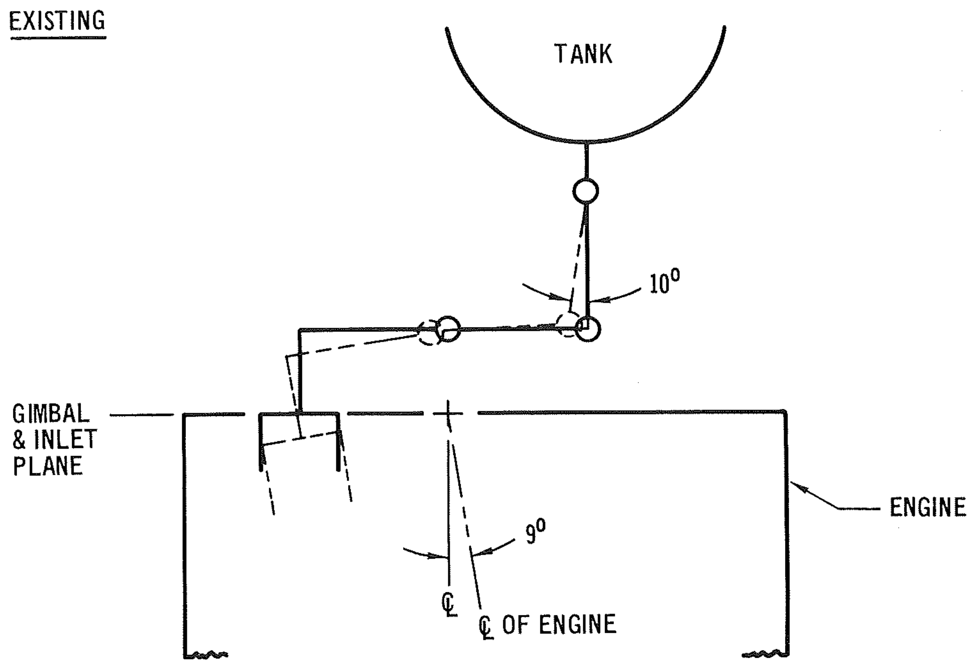
All feed ducts will have to be constructed using bellows to allow for thermal expansion and contraction. The bellows will also help to compensate for manufacturing tolerance buildup during installation. The location of the bellows can best be determined once the structure support locations are known.

Preliminary analyses indicated need for a 9% engine gimbal requirement in all directions. Designing a feed duct configuration which permits the engine to gimbal in a 9° square pattern is a definite problem. The prime obstacle is to design a duct configuration which will have "3 planes" of freedom within the available space and not interfere with the thrust structure. Also, a preliminary layout showed an angular movement of approximately 20° in the gimbal bellows, which exceeds the 6-9 degree angulation of the present "state-of-the-art". The proposed P&W engine has the turbo-pump inlets and engine gimbal point on the same fore and aft station plane. By moving the engine inlet port aft with respect to the engine gimbal point (Figure 7-3), an additional duct gimbal joint could be employed, thereby reducing the duct angulation to an acceptable level. In addition to the above, the bending moment of the gimbal bellows must be such that vibration and flow will have no detrimental effect on the bellows or the ducting. It is assumed that the turbo-pump housing can withstand loads transmitted to it from the bellows and duct.

An isolation valve will be installed in each LOX feed duct upstream of its gimbal joints. The valve will serve as a backup for the engine's propellant shutoff valve. Actuation of either of these two valves will isolate the tank and engine in the event of an emergency.

The non-propulsive vent exit assemblies are two 180° opposed orificed nozzles. They are positioned on the vehicle to prevent the impingement of the exhausted gases upon the vehicle's surfaces. The lines and ducts of the vent

FEED DUCT GIMBAL REQUIREMENTS



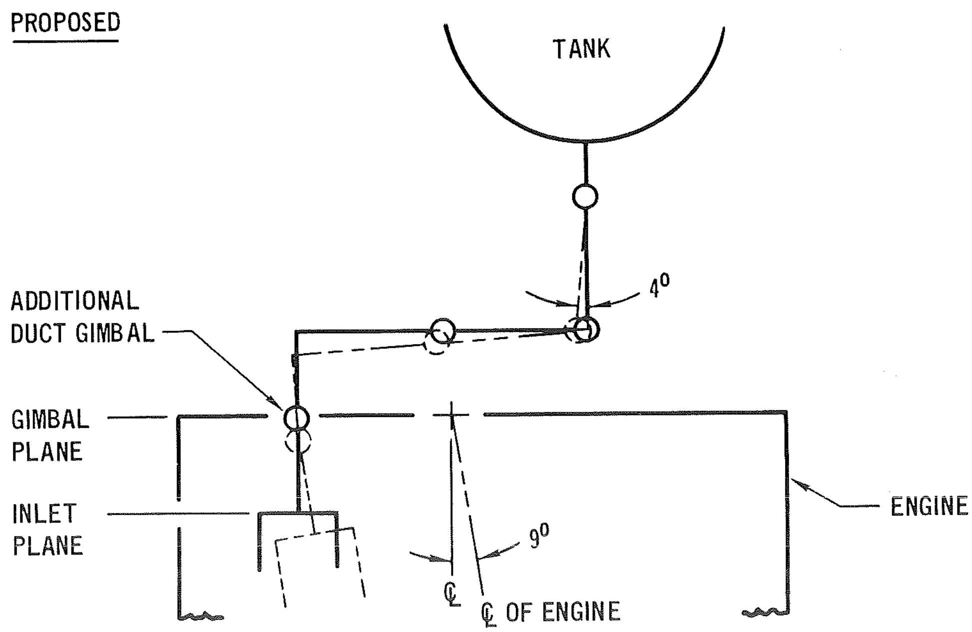


Figure 7-3

system are sized to flow the maximum system requirement and to maintain a flow mach number less than 0.15. This mach number will assure low line flow velocities and pressure drops which will assure equal and cancelling thrust.

The vent duct is routed from the LOX tank forward dome to the upper portion of the dorsal fin where it terminates in the 180° opposed non-propulsive vent nozzles. This location was dictated by the desire to prevent GOX plume impingement on the vehicle exterior surface since this would create a highly flammable environment during re-entry. The two parallel combination vent and relief valves were placed in the lower portion of the vertical stabilizer near the LH_2 tank vent and relief valves. This location was selected to utilize the available space in this area and to establish a common design vibration level for all the vent and relief valves. This system is very similar to that used by existing flight vehicles and system development can progress using present hardware.

The carrier vehicle LH_2 tank configuration is parallel, intersecting cylinders, with their forward ends being closed by two intersecting common bulkheads. The aft ends are closed by two intersecting hemispheres and the concave aft domes allow the propellant to be utilized more efficiently, i.e., with less residual. The LH_2 tank's forward dome is nested in the LOX tank aft bulkhead. Required volume, including a 10 percent ullage, is 77,600 ft^3. To provide access to the interior of the tank, the aft domes will have removable sumps.

The LH_2 feed system will consist of 10 main feed ducts interfaced with 10 separate tank outlets located on the LH_2 tank aft domes. These ducts have an inside diameter (ID) of 12 inches and are approximately 25 feet long. The feed ducts are routed directly to the engines. The entire LH_2 tank volume will act as a suppressor to dampen out the high energy reverse flow surge caused by an instantaneous engine shutdown while at or near 100% thrust.

The 10 tank feed duct outlets will be covered with antivortex screens. The separate tank outlets are necessary to prevent backflow of high energy fluid to other engines in the event of a premature shutdown. The screens are necessary to eliminate any vortexing effect which may be induced by the flow characteristics at tank outlets. Vortexting at low residual levels could result in vapor ingestion, which could cause cavitation of the pump, and consequent cutoff or engine damage. An isolation valve will be installed in each LH_2 feed duct upstream of its gimbal joints. The valve will serve as a backup for the engine propellant shutoff valve. Actuation of either of these two valves will isolate the tank and engine in the event of an engine shutdown.

Pressurization of the tank will be accomplished using the same ducting that is used for the vent system. A diffuser will be used to prevent the pressurization gas from impinging on the surface of the LH_2. Impingement of pressurization gas could cause high liquid-ullage heat transfer which would rapidly cool the ullage gas causing tank pressure to decrease.

Filling and draining of the tank will be through a quick disconnect (QD) located near the vehicle's aft end. The QD will be recessed and covered with a hinged door which is torsion loaded, opens when GSE is connected and automatically closes whenever GSE is disconnected. This protects the QD from the temperature changes which occur on the exterior surface. A fill and drain shutoff valve will be downstream of the QD to prevent loss of propellant when GSE is disconnected.

Routing of the ducting shall be directly into the two aft domes. The ducting will be insulated to prevent excessive heat loss. Bellows will be used to compensate for temperature induced loads and tank wall deflections caused by propellant weight and flight loads.

The LH_2 tank vent system is comprised of two parallel vent and relief valves, a diverter valve, a non-propulsive vent tee nozzle with check valve, and the ducting required to transport the gaseous hydrogen (GH_2) overboard. The ducting is routed from the LH_2 tank forward dome interior area to an isolated point on the dorsal fin. The ducting terminates in a "TEE" venting nozzle that has its open ends 180° apart.

The pneumatic system consists of 10 helium bottles (3000 psi), associated hardware and an umbilical connection. The bottles are located near the point of usage and the umbilical connector is located near the aft end of the vehicle for ease of servicing. The miscellaneous hardware for the system is not shown in Figure 7-2.

Airbreathing Propulsion - The airbreathing propulsion system consists of four deployable turbofan engines which provide thrust for cruise, go-around, and landing.

Two turbo fan engines are installed in each wing and are deployed below the wing for operation. At completion of the engine deployment sequence a fairing closes the cavity on the wing lower surface to restore mold line smoothness. Fuel feed lines have swivel joints to provide the capability of extending with the engine during deployment. Isolation valves are provided on fuel feed lines to stop fuel flow in case of engine shut down. JP-4 fuel is stored in tanks forward of the engines. These tanks are insulated for protection from re-entry heating.

Attitude Control Propulsion System - An attitude control system provides vehicle orientation in the high altitude flight. Roll control is provided by coupled thrusters and moments for pitch and yaw are provided by thrusters supplying unidirectional impulse. These thrusters are located at the base of the vehicle to provide the maximum moment arm and to provide an installation which is protected from entry heating. Hydrogen and Oxygen are stored in cryogenic tanks in each wing and are pumped through heat exchangers into accumulators to provide gaseous propellant for the thrusters. Separate propellant storage, conditioning systems, and thruster groups are on either side of the vehicle. These systems are interconnected to provide redundancy in the event of component failure.

7.1.3 Carrier Equipment - The vehicle equipment shown in Figure 7-4 includes subsystems not described in the sections on propulsion and structure subsystems.

The pressurized crew cabin is sized for two men based on the dimensions shown in Figure 7-5. This envelope requires approximately 40 cu. ft. and was initially established for a pressure-suited man to allow the 66° shown between spine and legs. The substitution of a crew-man in a shirtsleeve environment permits him to sit in a more upright position, or 90° between spine and legs. An airplane type environmental control system provides a shirtsleeve environment and occupies approximately 20 cu. ft. The avionics equipment items and their battery power supply are located at the forward end of the vehicle to help achieve vehicle balance around the required c.g. Part of the avionics components are included in the crew cabin as controls and displays. The avionics equipment occupies 30 cu. ft. and the power supply system requires 8 cu. ft. These systems are assumed to be packaged onboard with a 65% efficiency for vehicle volume requirements. Power distribution requires an additional volume, however, no spatial allocation was reserved because this item is spread throughout a relatively large portion of the vehicle.

CARRIER EQUIPMENT

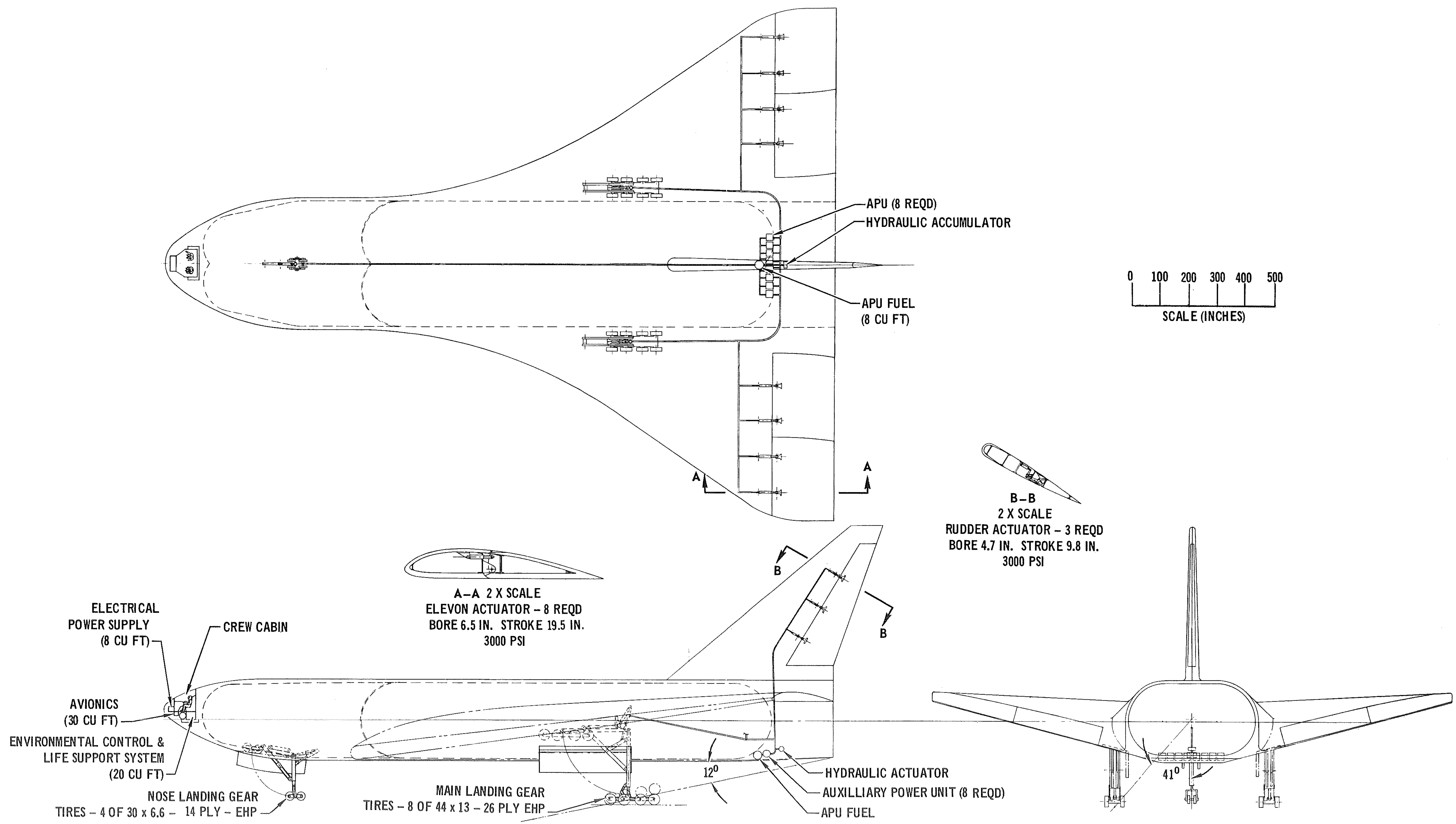


Figure 7–4

CREWMAN SPATIAL ENVELOPE

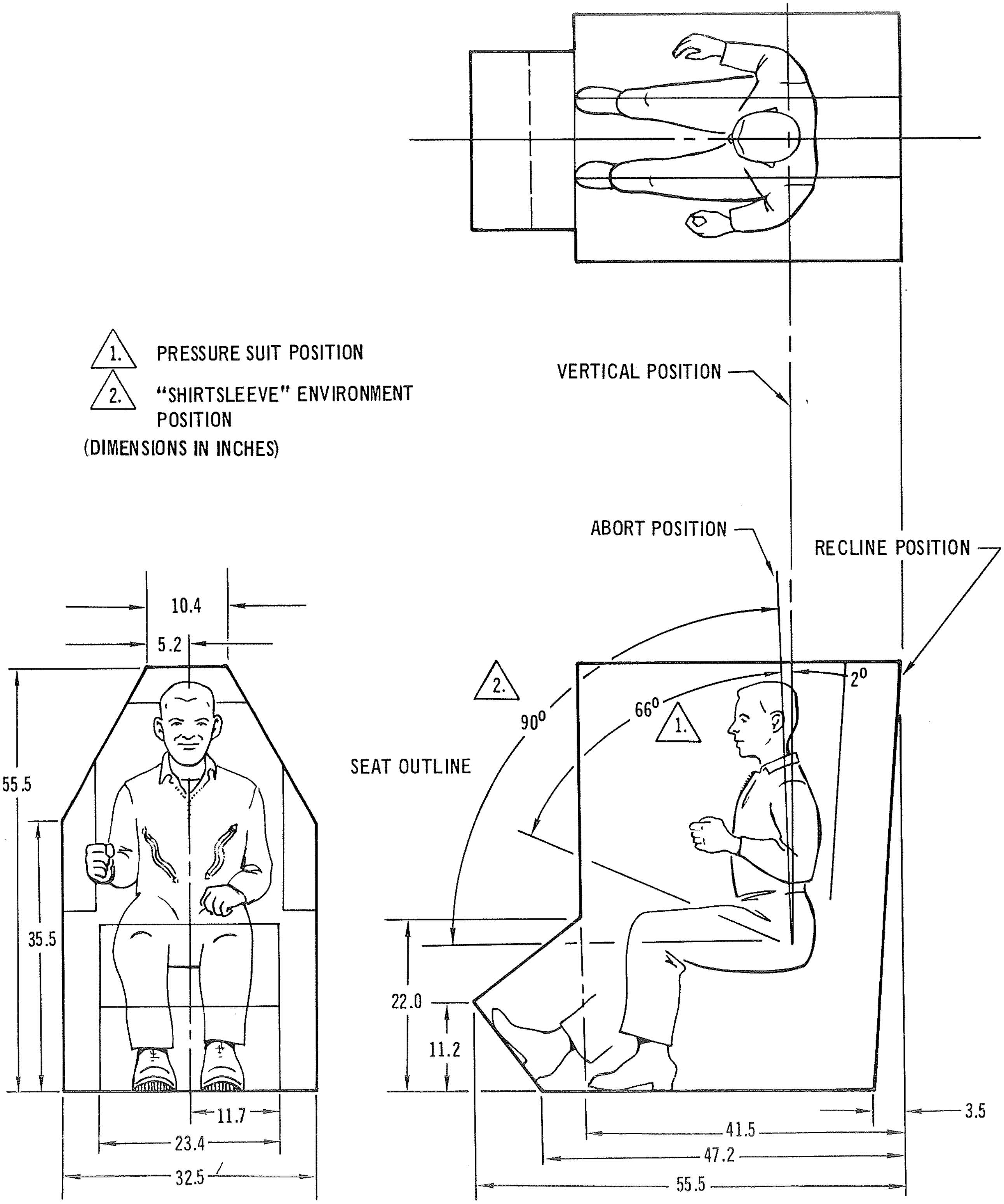


Figure 7-5

The aerodynamic control system components shown in Figure 7-4 were sized by assuming values pertinent to the derivation of equipment requirements. Hydraulic system pressure is 3000 psi, and actuators were assigned a stroke to diameter ratio of 3:1.

Values for hinge moments, surface deflection rates, total deflection, and nominal time are shown in Table 7-1. The nominal time is for the period when hydraulic power must be supplied by an internal power source. The stated values are based on estimates of required performance for the vehicle. The actuators were sized using the formula:

$$V = (H.M.)(\delta)/P$$

where:

H.M. = Hinge Moment (in-lb),

δ = deflection (radians),

V = fluid displacement (cu. in.)

P = pressure (lbs per sq. in.)

The volume was then reduced to physical dimensions using the specified proportions. Power and energy were determined to provide a basis for sizing the required power supply system. System non-operating power was determined as 15% of maximum power. This is the power required for maintenance of system pressure, leakage, etc. Nominal power, representative of an average value, is 25% greater than the system non-operating power. These values are based on previous analyses to establish interim requirements prior to derivation of actual load-time curves.

Table 7-1

195 FT CARRIER AERO CONTROL SURFACE SYSTEM

Hinge Moment (Ft-Lb)	Rotation Rate (°/sec)	Deflection (Deg.)	Time (Min.)	Actuator No.	Actuator Dia.	Actuator Length	Max. Power (Ft-lb/hr)	Nominal Energy H.P.-Hr.
Elevon 717000	15	50	10	4	6.5	19.5	6.75×10^8	10.68
Rudder 243000	15	30	10	3	4.7	9.8	2.29×10^7	3.78

NOTE: All numbers based on one aero surface.

A total of 8 actuators are used for elevon control and 3 actuators are used for rudder rotation.

The 8 auxiliary power units (APU) use hydrazine fuel which is stored in an 8 cu. ft. sphere located just aft of the main propellant tank. A possible alternate to the hydrazine powered APU system is one powered by H_2 and O_2. The reaction control system may include such a system to provide pump power for liquid transfer. That system might then provide the driving force for both reaction control and aero control systems.

The main landing gear is located just aft of landing c.g. with the tire size and number determined by landing loads and concrete runway requirements. An assumed value of 35,000 #/tire is used to determine the required number of tires. This value is then adjusted to allow one tire per carriage to blow out and the remaining tires to take the landing load. The main landing gear was then sized for 32,200 #/tire (for 14 acting tires) with 8 tires per carriage or 16 tires total. Reference 4 was used to determine a required tire size of 44 x 13, 26PLY, extra high pressure (35,800 allowable). The nose wheel tires were sized using the formula:

$$\frac{(W)\ (A)}{(B)\ (N)} + \frac{10\ (W)\ E}{(32.2)\ B\ (N)} = \text{Load/tire, where;}$$

W = landing weight

A = longitudinal distance from center of main landing carriage to c.g.

B = longitudinal distance from center of nose landing gear to c.g.

N = number of tires

E = vertical distance from c.g. to ground level

10 = 10 Ft. per sec^2 braking deceleration

The above equation using 4 tires yielded a value of 18,520 #/tire.

A tire size of 30 x 6.6 (14 PLY) Extra High Pressure with 12,950 #/tire static load rating and 19,400 lbs per tire dynamic (braking) load rating is required. Deployment of the main gear with an 18 in. load deflection, provides clearance for a 12° maximum angle of attack at touchdown. The main gear shock absorber has a diameter of 11 in., with an internal pressure of 2500 psi.

7.2 Orbiter Design Definition - Definition of the design elements for the orbiter is divided into five sections. The first section provides an over-view of an interim vehicle. The orbiter design effort was devoted to this concept prior to the baseline revision resulting from the special study introducing the integral tanks concept. The next three sections describe general arrangement, propulsion systems and vehicle equipment similar to the carrier design definition. Selected baseline systems are described, along with derivation of subsystem components. The last section defines a ferry configuration.

7.2.1 Interim Arrangement - The iterim orbiter concept employed separate structure for boost propellant tanks and for carrying vehicle loads. The basic arrangement and structural concept are represented by Figure 7-6.

This arrangement utilizes structural full depth webs on either side of the payload bay to transfer the launch thrust loads into the vehicle skin. The main propellant tanks consist of 3 hydrogen tanks and 1 oxygen tank. The two larger H_2 tanks are located along either side of the payload bay. Each tank consists of two cones which contain a fuel volume of 4840 cu. ft. The third hydrogen tank is aft of the cargo bay and consists of two cones joined together to obtain a volume of 2500 cu. ft. The total hydrogen volume is then 12,180 cu. ft. and the three tanks are inter-connected to provide one source for propellant feed purposes. The main oxygen tank is formed by two intersecting cylinders, located forward of the cargo bay, and has a volume of 4550 cu. ft. Each of these tanks has spherical ends to complete the pressure vessel shape. Maneuvering propellant is contained in two cylinders having a volume of 600 cu. ft. each. These two tanks are located on either side of the payload bay above the main hydrogen tanks.

The spacecraft volume forward of the main oxygen tank includes provisions for crew cabin, avionics, power supply, and landing systems. To provide thrust for approach and go-around, two turbojet engines are deployed from each side of the vehicle. They are stowed between the crew cabin and main oxygen tank and are supplied with fuel from a tank located between the stowed engines. The tank has a capacity of 200 cu. ft. A transfer tunnel inside the spacecraft connects the crew cabin and payload container.

Investigation of this vehicle concept was terminated at the initiation of the integral tank concept study. However, most of the data on subsystems, operational

ORBITER INTERIM ARRANGEMENT

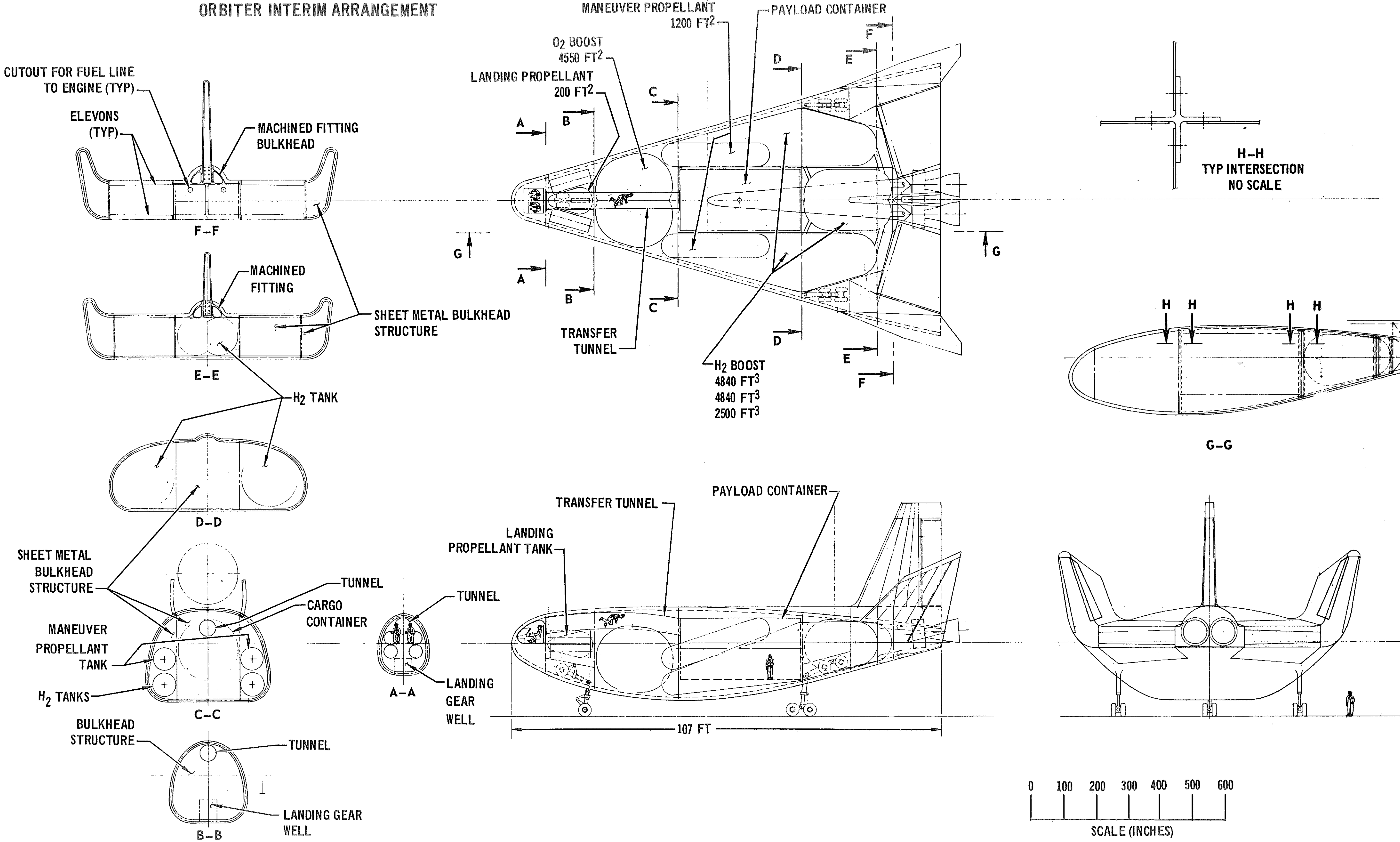


Figure 7–6

capability, and spacecraft volume utilization were carried from this concept to the latter one. The systems are described in more detail in subsequent sections of this volume.

7.2.2 Orbiter General Arrangement - The structural concept and internal arrangement of the 107 ft. orbiter are shown in Figure 7-7.

The payload bay shows the 15 ft. dia. by 30 ft. long payload container with one foot allowed on each end of container and 6 inches on each side for installation clearance and mounting provisions. The boost propellant oxidizer tank is forward of the payload bay and a hydrogen tank is on either side of, and aft of, the payload bay. The walls of these tanks are made to conform to the inner moldline of the vehicle whenever possible. The tank walls then become the primary load carrying skin for vehicle loads. The area between the inner and outer moldline will provide insulation for the tanks and equipment. The inner moldline skin forms an extension of the tank walls forward of the oxygen tank, between oxygen and hydrogen tanks and aft of the hydrogen tanks.

The forward compartment encloses pressurized crew cabin, and unpressurized area for avionics, power supply, E.C.L.S., nose gear and landing propulsion system. The landing engines are mechanically deployed out the sides of the vehicle to the required operating position. Additionally, a pressurized tunnel is provided between the crew cabin and payload bay to permit transfer of the crew to a pressurized payload container during orbit operations. This tunnel is inside the moldline and on the vehicle centerline above the oxygen tank.

Propellant for 2000 fps in-orbit maneuvering capability is provided by interconnected tanks mounted below the forward end of the payload bay. The main landing gear is positioned on either side below the aft portion of the payload bay.

Thrust loads from the 2 boost engines are transferred through a truss structure to the two longitudinal webs, which continue forward to form the LH_2 tanks. A transverse bulkhead forward of the engines extends to either side of the vehicle and provides elevon hinge support. The main ribs in the tip fins, and the ribs in the dorsal fin are tied into this bulkhead; then the loads are distributed to propellant tank walls and the body skin.

7.2.3 Orbiter Propulsion Systems - The primary propulsion systems effort was concerned with the boost propulsion system. The following paragraphs and Figure 7-8 describe this system with a subsequent description of secondary propulsion.

Integral Launch and Reentry Vehicle System

ORBITER GENERAL ARRANGEMENT

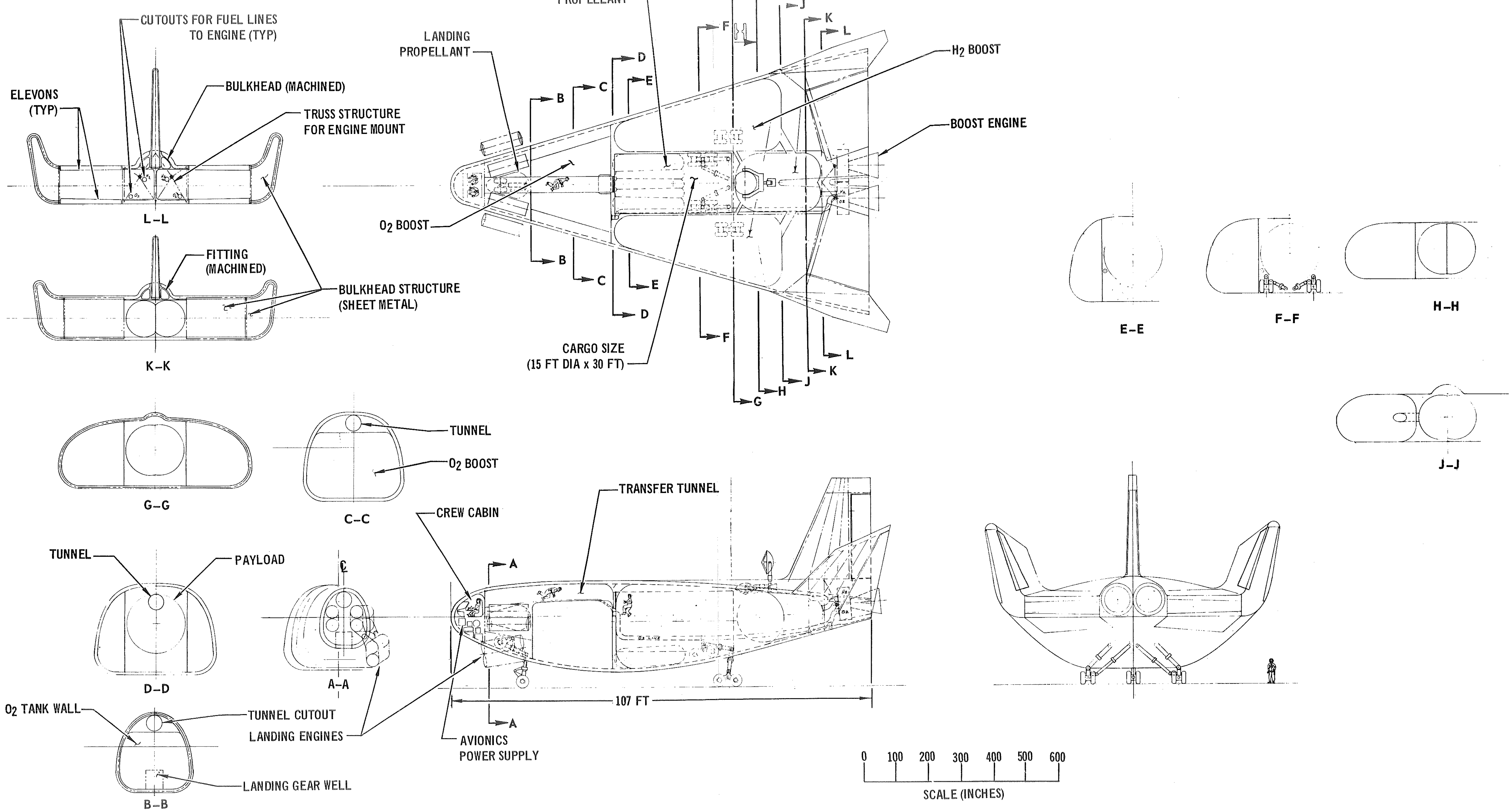


Figure 7–7

ORBITER PROPULSION SYSTEMS

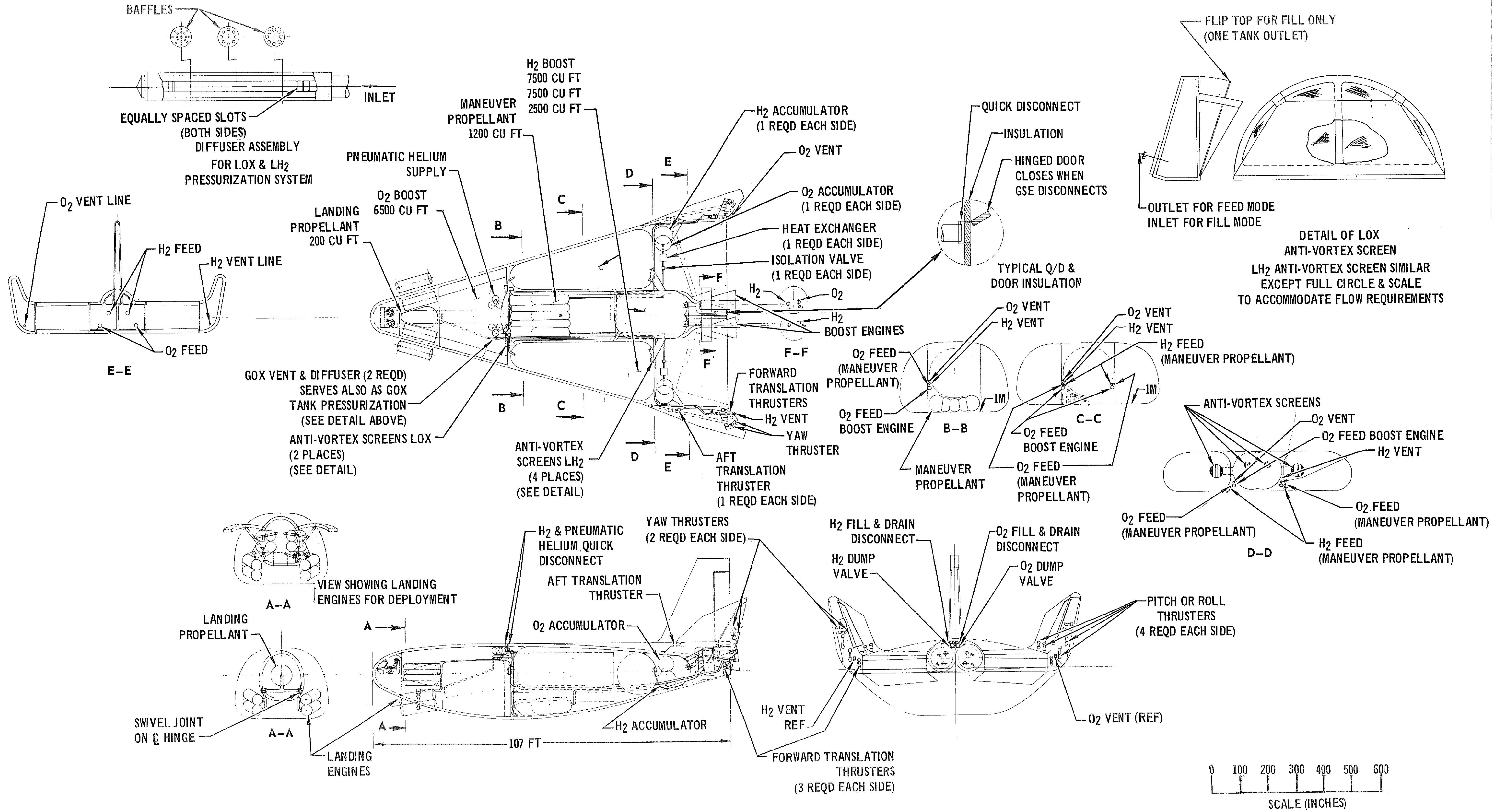


Figure 7–8

Boost propulsion - The propellant storage, feed, pressurization, and venting systems for the orbiter boost system are described below.

Two feed ducts (12 inch ID and 63 ft. long) are routed from the integral LOX tank (6500 cu. ft.), parallel to the payload area before entering their respective engine interfaces. Isolator valves are located at the propellant tank outlet as a backup to the main engine propellant valve. The LOX tank fill and drain system ties in with one of the 12 inch engine feed ducts near its engine interface. Filling and draining of the tank will be accomplished through a checking quick disconnect (QD) located near the aft end of the vehicle. The QD will be recessed and covered with a hinged door which is torsion loaded shut. The door is opened when GSE is connected and automatically closed whenever GSE is disconnected, thus protecting the QD from the high heat input that occurs on the exterior surface. This is necessary to inhibit heat conduction to the LOX feed line and to protect the QD for reuse. A combined fill and drain shutoff and relief valve (normally closed) downstream of the QD will isolate the unusable LOX trapped in the fill and drain line from the LOX feed line. A normally closed dump valve will be located parallel to the QD.

The two propellant feed ducts have antivortex screens located at the aft end of the orbiter vehicle LOX tank. Vortexing in the feed lines is undesirable since it inhibits mass flow and could result in LOX pump cavitation. Since propellant loading will be accomplished through the antivortex screen, its design must include a "flip-top" feature with a baffle. This fliptop design allows unfiltered tank filling through the feed line, and the integral baffle inhibits propellant spraying during fill operations and engine shutdown, thus minimizing the possibility of ullage pressure collapse.

All feed ducts will have to be constructed using bellows to allow for thermal expansion and contraction. The bellows will also help to compensate for manufacturing tolerance buildup during ducting installation. The location of the bellows can best be determined once the structure support locations are known.

Designing a feed duct configuration which permits the engine to gimbal in a 9° square pattern is a definite problem. This problem is discussed in Section 7.1.2.

The common LOX tank vent and pressurization duct (5 in. I.D.) runs from the forward end of the LOX tank to the LOX tank penetration point in the forward bulkhead. After exiting from the LOX tank, it runs down the side of the LOX tank and enters parallel combination vent and relief valves. The LOX tank vent duct then runs along the payload area to a pair of opposing non-propulsive nozzles in the aft end of the vehicle. The overboard venting point was selected in order to eliminate any GOX plume impingement on the exterior surfaces of the vehicle.

The LH_2 tanks (17500 cu. ft.) have integral skins on the vehicle sides. The tanks on either side of the payload bay have longitudinal webs the full depth of the vehicle. These webs also become an integral part of vehicle structure. A double conical tank with a longitudinal intersection and spherical ends is located aft of payload.

The orbiter LH_2 feed system consists of two 12 inch diameter, 6-foot long feed ducts that exit at the aft section of the double center tank and enter the engine at the LH_2 inlet. The 12-inch diameter crossover ducts connect the aft end of the side tanks to the aft end of the center double tanks. Antivortex screens are located in the aft end of side tanks at the crossover duct and in the aft end of the center double tank at the engine feed duct outlets. They are designed to filter the LH_2 exiting from the tank and to prevent vortex action. A hinged fliptop on the side tank screen is required to permit fluid flow into tanks during fill mode. It then closes for filtering during engine operation. The fliptop opening travel is restrained to divert the incoming flow horizontally, preventing fluid from penetrating the upper surface of the liquid during fill.

Isolator valves are installed at the tank outlets as a backup to the engine propellant shutoff valves. After thrust and tank support structure have been determined, the duct near the engine can be designed to permit 9 degrees of engine gimbaling.

The emergency dump and fill and drain functions are incorporated in a single system. The 12-inch diameter duct runs from the aft end of center tank, near the engine feed outlet, to the aft end of vehicle and provides for both fill and drain and dump.

A shutoff valve (fill and drain) is located in the line adjacent to the tank. A positive check valve (allowing flow towards the tank) is in the line at the rear of the fuselage. An emergency dump valve is located in a duct joining the fill and drain line between the check valve and the shutoff valve and permits rapid dump in a flight abort situation by opening the fill and drain and dump valves and bypassing the QD. At liftoff, a torque loaded door covers the recessed

QD to protect it from excessive heating. The QD incorporates a relief valve to bleed off gas or liquid trapped in the line. During reentry, the fill and drain valve opens to ensure positive pressure in the duct and to preclude explosive mixing due to external leakage.

The 5-inch vent duct system on the orbiter consists of a crossover from the forward end of the side tanks to a common entrance into the parallel redundant vent and relief valves and then into the directional control valve. Prior to liftoff, the gas exits overboard into ground vent lines through a QD. At liftoff, a torque loaded door covers the recessed QD and duct to protect them from excessive heat and provide a flush exterior surface.

After liftoff, vent gas enters the non-propulsive duct which parallels the payload area and exits through a non-propulsive tee at the aft end of the vehicle. Outlets are 180° opposed and directed up and down with respect to fuselage surfaces to provide least impingement forces on the fuselage and to prevent dumping of vent gas into the engine area. A check valve at the NPV exit prevents the entry of atmosphere into the duct during non-venting reentry periods.

The pneumatic system consists of 3000 psi Helium bottles, associated hardware and an umbilical connector. The umbilical connection is near the LOX fill connector for ease of servicing.

Airbreathing Propulsion - The landing system shown has a powered landing capability supplied by 4 turbo-jet engines with 23,000 lbs. thrust per engine. These engines are mounted so they can be deployed out the sides of the vehicle with a powered mechanism to rotate the engines into a canted position with the forward end down. This aligns their centerlines with the airflow at an angle of attack during operation to permit more efficient intake. The landing propellant tank mounted between the stowed engines contain 200 cu. ft. of JP-4 fuel. The fuel is supplied to the engines through a duct arrangement which incorporates a swivel joint on the rotation centerline.

Attitude Control and Maneuvering Propulsion - The orbit maneuvering system has the propellant located below the payload bay in a multi-lobe dual propellant tank with 285 cu. ft. of oxygen and 915 cu. ft. of hydrogen. Separate but interconnected propellant conditioning systems are positioned on either side of the vehicle. They include pumps, heat exchangers, and O_2 and H_2 accumulators. This system supplies the gaseous H_2 and O_2 for twenty 4,000 lb. thrusters. Attitude control thrusters are mounted in the two tip fins. Roll control is provided by coupled thrusters. Pitch and yaw are provided by thrusters firing in one direction to provide the proper moments. Forward translation is provided by six aft firing thrusters (3 per tip fin) and aft translation is provided by two forward firing thrusters. The six aft firing thrusters are used for main impulsive maneuvers and two of these are used for incremental in-orbit translation maneuvers.

7.2.4 Orbiter Equipment - The spacecraft systems shown in Figure 7-9 cover only systems which are not defined by structure and propulsion subsystems.

The sizing of the two man pressurized crew cabin was based on the envelope shown in Figure 7-5 like the carrier crew compartment. The crew function in a shirtsleeve environment which is provided by a two gas (O_2-N_2) environmental control and life support (ECLS) system. The ECLS system occupies 50 cu. ft. A 4 ft. dia., 32 ft. long tunnel, also provided with a shirtsleeve environment, permits internal transfer of the crew to the payload container. The payload container contains its own support provisions. The payload container is deployed by opening doors on the spacecraft upper surface and mechanically translating the container upward until accessible to a space tug or space station interface. A cooling radiator is incorporated in the payload bay doors for dissipating internal heat energy in orbit. The radiator, on the inside of the doors, is protected from launch and entry heating.

The power supply and avionics being relatively high in density are located as far forward in the vehicle as possible for c.g. considerations. Spacecraft avionics equipment occupies 40 cu. ft., part of which is included in the crew cabin as flight controls and displays. Power supply, provided by batteries and fuel cells for the electrical load, requires 33 cu. ft. A 6-ft. diameter SHF dish antenna is located aft of the payload bay near the upper surface. Doors on the vehicle upper surface are opened and the antenna is rotated upward for operation. A rendezvous radar antenna, aft of the crew cabin, is also deployed for orbital operations.

Power for the aero control system is provided by three auxiliary power units using hydrazine fuel. The hydrazine fuel is located in the aft end of the vehicle in an 8 cu. ft. sphere. This system is positioned near the point of usage to

ORBITER EQUIPMENT

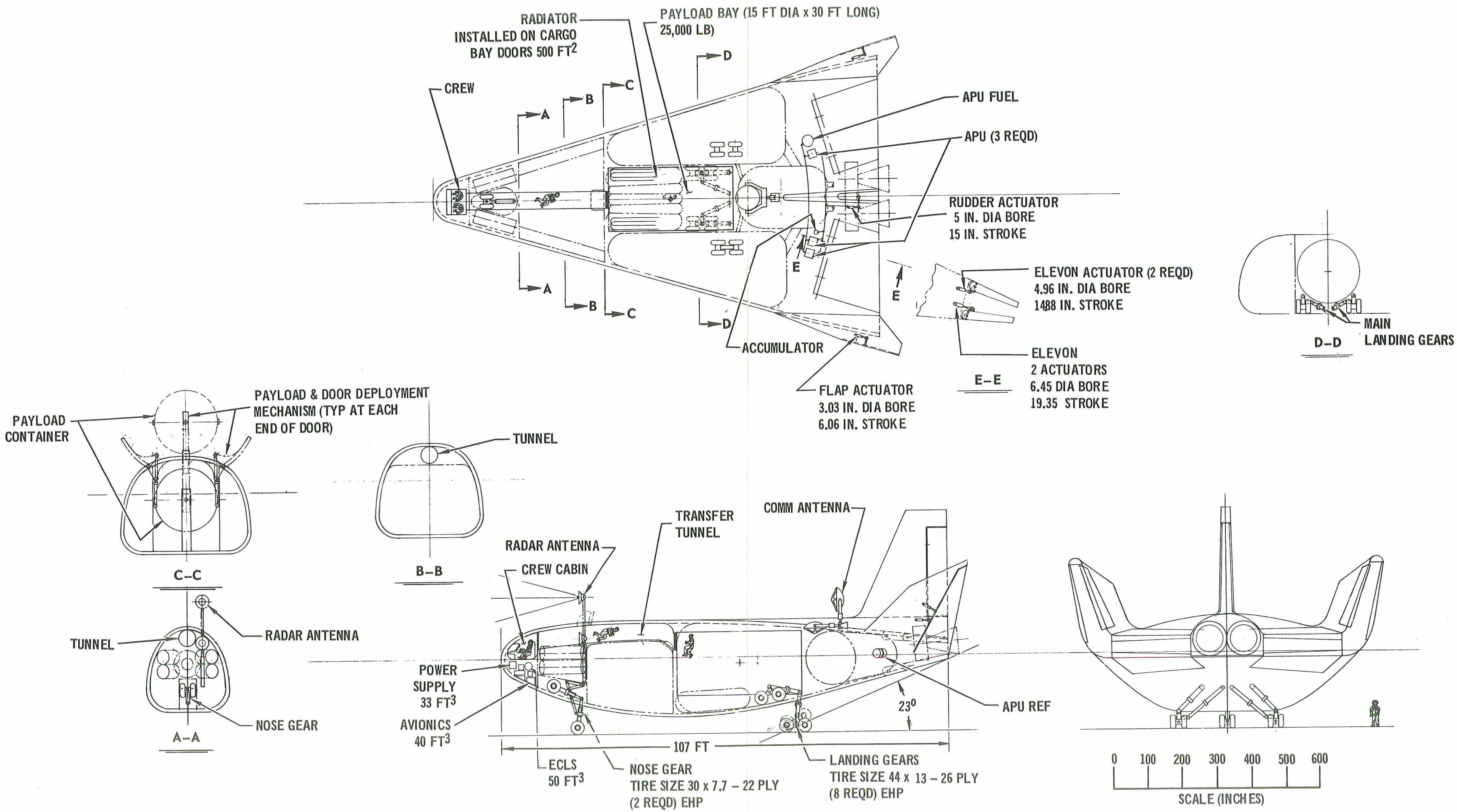


Figure 7-9

minimize the length of high pressure hydraulic lines required. Dual redundant hydraulic actuators impart the required rotational control to 4 elevons, 2 tip fin flaps, and the rudder.

The sizing of aero control system components is based on the same format as that used in the carrier equipment, Section 7.1.3. The orbiter APU's are the same size units as those in the carrier. The values used for component sizing are shown in Table 7-2.

The main landing gear and nose gear sizing analysis follows the same format as that of the carrier equipment, Section 7.1.3. The main landing gear loads are 31,000 lbs/tire for 8 tires based on 6 tires not blown out and sized at 44 x 13 (26 ply) extra high pressure (35,800 lbs/tire allowable). The nose landing gear loads are 30,250 lbs/tire for 2 tires and sized at 30 x 7.7 (22 ply) extra high pressure 21,300 lbs/tire static load rating and 31,900 lbs/tire dynamic load rating. The main landing gear has an 18 in. load deflection stroke which provides clearance for a maximum 23° angle of attack at touchdown. The shock absorbing piston diameter is 7-inches using a system pressure of 2500 psi.

Table 7-2

107 FT ORBITER AERO CONTROL SURFACE SYSTEM

	HINGE MOMENT (Ft-Lb)	ROTATION RATE (^{o}/sec.)	DEFLECTION (Deg.)	TIME (Min.)	ACTUATOR			MAX POWER (Ft-Lb/Hr)	NOMINAL ENERGY (H.P.-HR)
					No.	DIA (In.)	LENGTH (In.)		
Elevon	360,000	15	50	40	2	6.45	19.35	6.38×10^{7}	21.5
Elevon Flap	157,000	5	52	40	2	4.96	14.88	9.25×10^{6}	3.12
Rudder	163,000	15	30	40	1	5.25	15.75	2.88×10^{7}	9.74
Tip Fin Flap	41,300	5	15	40	1	3.03	6.06	2.42×10^{6}	.817

NOTE: All numbers based on 1 Aero Surface

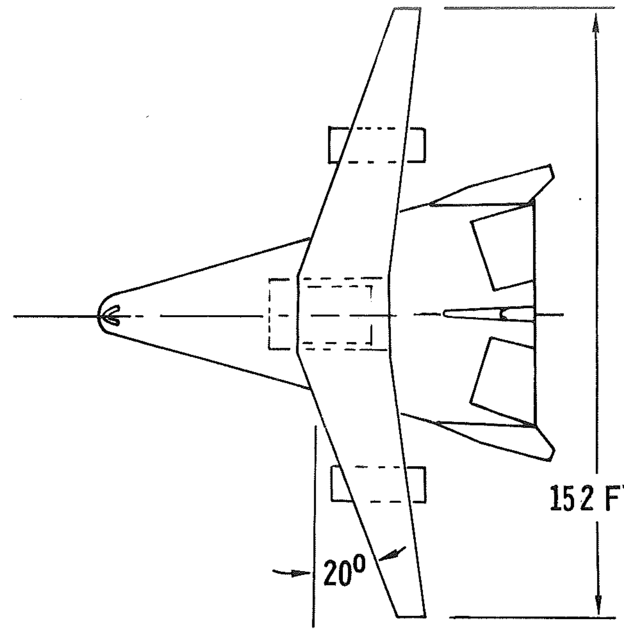
7.2.5 Ferry Configuration - Landing at the launch site cannot be assured for every return from a mission. Weather conditions or emergencies such as an aborted flight may require orbiter return to an alternate airfield with subsequent transportation to the primary launching area. The physical size of the orbiter precludes the transportation by conventional cargo transports. Disassembly of the orbiter into major components which could be transported by current aircraft is in opposition with the concept of quick turnaround for subsequent missions. Thus, self-ferry appeared to be the most rational approach to the problem. However, the subsonic maximum L/D of 4 and the on-board propulsion system capability do not yield a great deal of cruise range. Thus, it was necessary to define a modified configuration which would improve the subsonic cruise capability of the vehicle.

The orbiter configuration shown in Figure 7-10, conceived by NASA/LRC, has its payload bay doors removed and an assembled package of wing, cruise engines and cruise propellant is attached with the propellant tank in the payload bay. The wing has a ST CYR 156 airfoil with a 20° sweep angle, 152 ft. span and .3 taper ratio. The basic orbiter control system is used for aerodynamic control. Similar configurations have been wind tunnel tested at LRC and shown to be feasible. Subsonic test data shows that this configuration has a maximum trimmed L/D of 8.5. Figures 7-11 and 7-12 show the variation of the trimmed lift coefficient and the trimmed L/D with trim angle of attack.

Two turbofan engines are mounted on the lower surface of the wing to provide the required thrust for the 200,000 lb. vehicle to take off in 5,000 ft. These engines are similar to the cruise engines on the carrier. Turbofan engines were selected to take advantage of their superior specific fuel consumption. The propulsion system, including 19,000 lbs. of propellant, provides 330 nautical miles cruise at an altitude of 10,000 ft. The system for engine control is the only physical interface required between the orbiter and ferry kit, aside from the structural interface. The existing vehicle structure will accept the kit installation. No landing gear penalty is imposed as the take-off weight of this configuration is about 20,000 lbs. greater than normal landing weight from orbit.

ORBITER FERRY CONFIGURATION

MAX L/D (TRIM) ≈ 8.5 (LRC TEST DATA)
FERRY KIT DRY WT ≈ 31,000 LB
FERRY PROPELLANT WT ≈ 19,000 LB
ENGINE THRUST (TF39) = 45,000 LB (SLS)
FERRY RANGE ≈ 330 N M



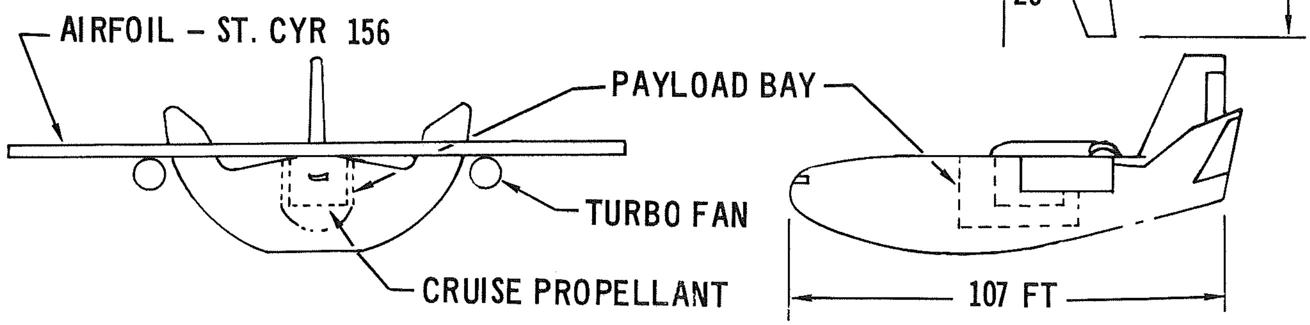


Figure 7-10

ORBITER FERRY CONFIGURATION SUBSONIC TEST DATA
TRIM LIFT COEFFICIENT

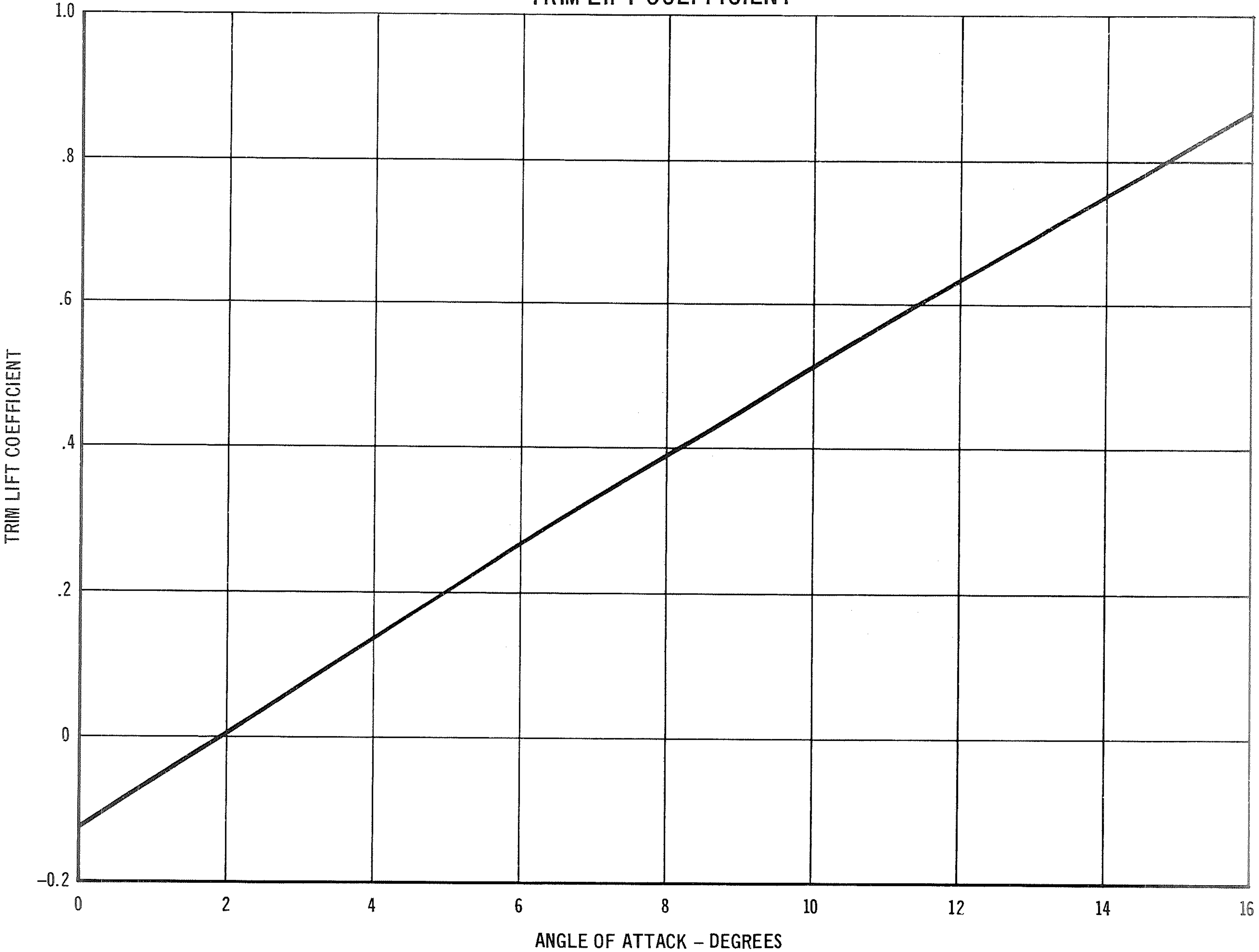


Figure 7-11

ORBITER FERRY CONFIGURATION SUBSONIC TEST DATA
TRIM LIFT-DRAG RATIO

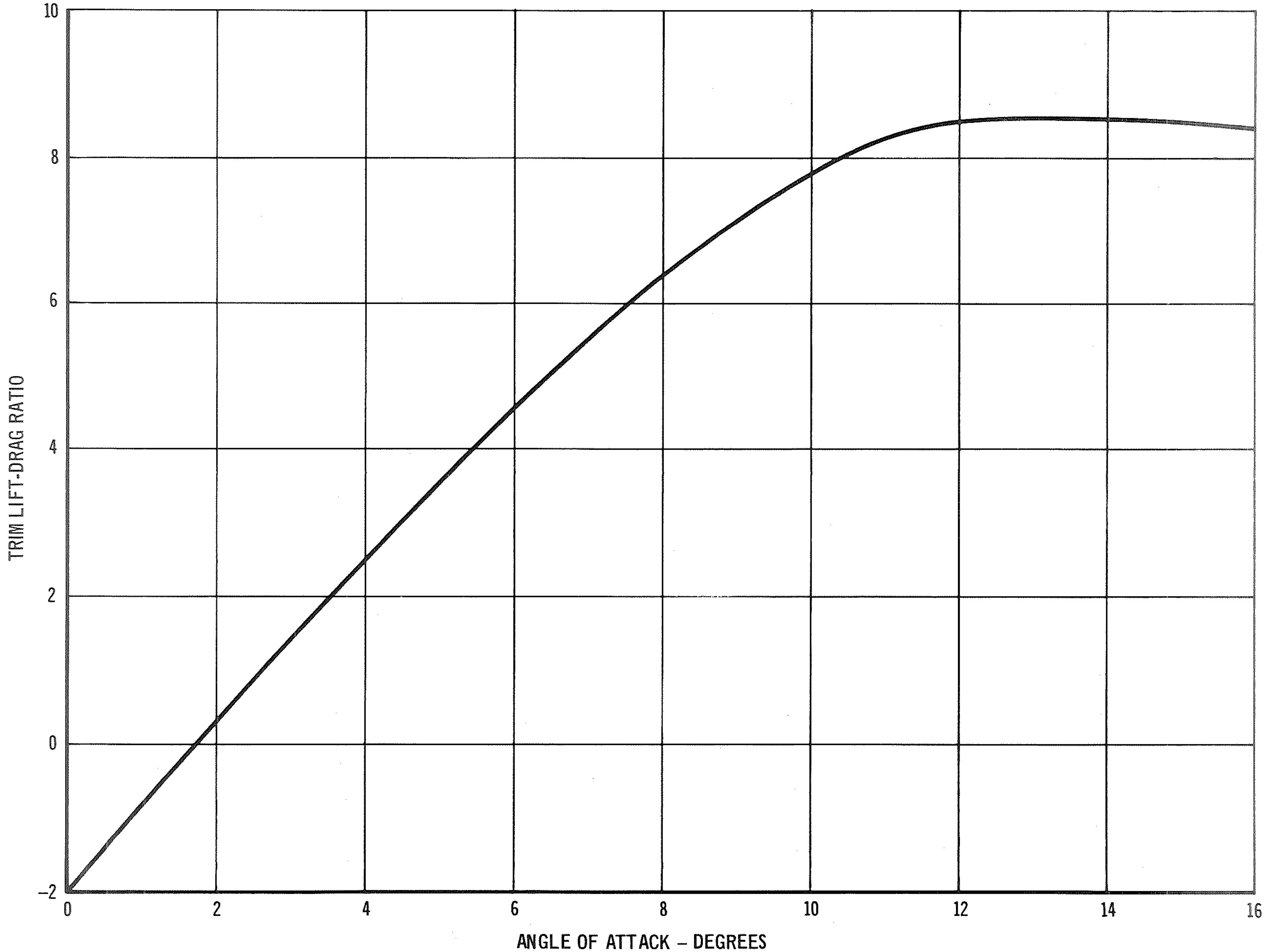


Figure 7-12

7.3 Vehicle Interface - The structural tie between carrier and orbiter consists of a three point attachment as shown in Figure 7-13.

Two support points are provided at the intersection of the orbiter forward payload bay bulkhead and payload bay side webs. All side loads and longitudinal loads are transferred at these two points. These fittings are near the longitudinal C.G. of both vehicles at separation, and make maximum utilization of the existing vehicle structure.

A separate support arm is provided near the aft end of the vehicles to complete the structural interface. This arm is allowed to take loads along it's axis, but offers no resistance to side loads and vehicle axial loads. This arm is hinged on the carrier at the aft carrier thrust structure ring and is attached to a fitting in the orbiter at the orbiter thrust structure bulkhead by a pin. For separation, this pin is retracted and the arm is rotated into the area between carrier inner and outer moldline structure. A thermal protection surface is installed on one surface of the arm so that retraction of the arm into the carrier provides a smooth carrier outer moldline surface. A closure door on the orbiter provides a smooth moldline.

Two methods of attachment were considered for the two forward interface points. The first case as shown in Figure 7-14 utilizes a fitting mounted in the orbiter and attached to a fitting in the carrier by a pin. For separation this pin is retracted and the orbiter fitting is rotated into the vehicle. Closure doors on both vehicles restore a smooth moldline. An alternate approach for this method is to install the retractable fitting in the carrier, with the fixed fitting in the orbiter. The method of attachment chosen for this case would be the method showing the least weight penalty for the orbiter. In the second case, as shown in Figure 7-13, a fixed fitting is attached to the carrier and extends external to the carrier outer moldline into the area between the orbiter inner and outer moldlines. It is attached to a fitting on the orbiter by a pin. For separation this pin is retracted. A closure door restores a smooth outer moldline on the orbiter. The carrier outer moldline structure attaches directly to the carrier fitting.

VEHICLE INTERFACE

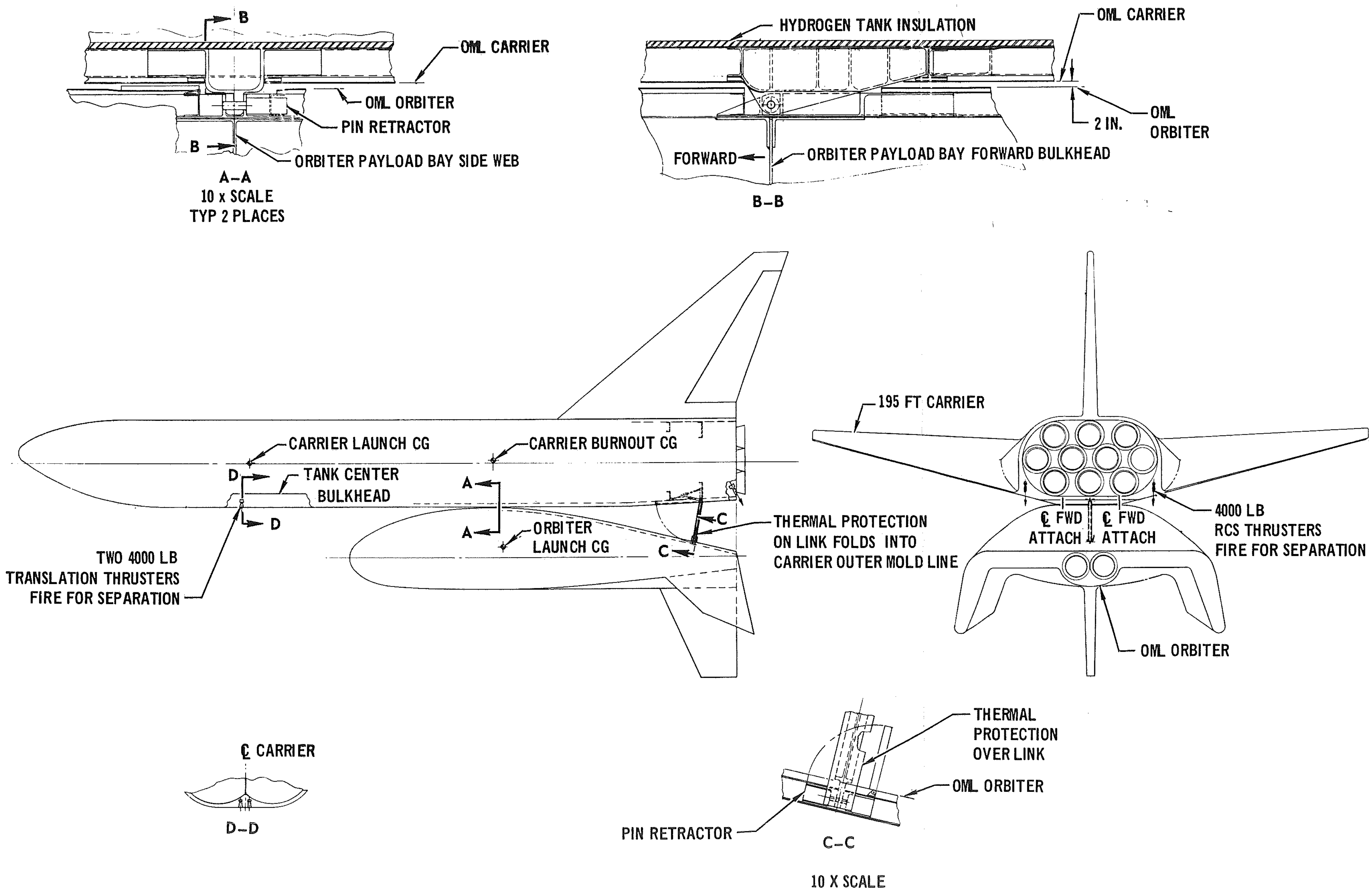
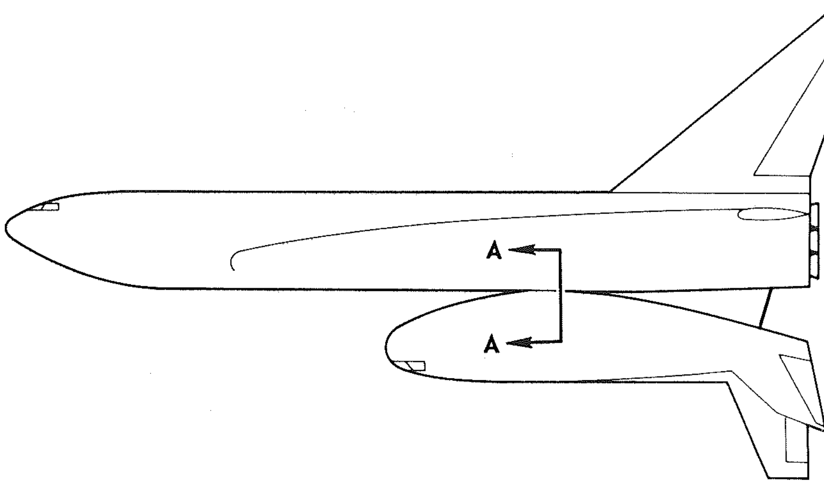
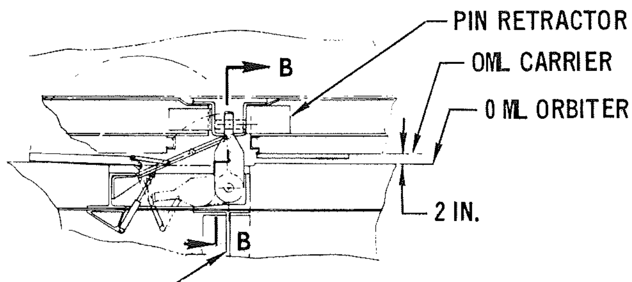


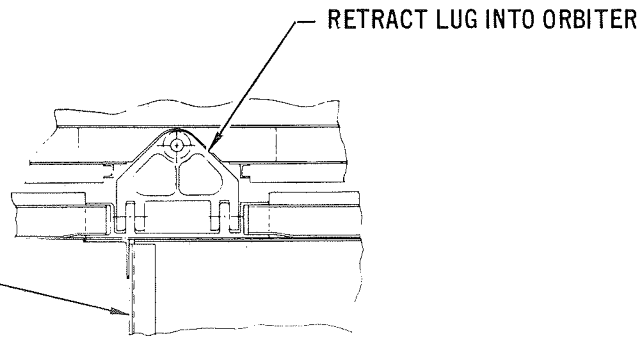
Figure 7-13

VEHICLE INTERFACE
(Alternate)





A–A
(TYP 2 PLACES)



B–B

Figure 7–14

The second case, fixed exterior moldline fitting on the carrier with fixed fitting in the orbiter is preferred since heating of the carrier external fitting would not be excessive and this method eliminates the complexity of retractable fittings. It also minimizes the orbiter weight penalty and maximizes performance.

The axis of the connecting pins at all three points is normal to the vertical plane through the vehicles so that differential expansion during launch between the vehicles is accommodated by a rotation of the aft attachment arm.

Vehicle separation is provided by translation thrusters. Two 4000 lb. reaction control system pitch down R.C.S. thrusters near the base of the carrier fire in conjunction with two 4000 lb. translation thrusters installed in the carrier forward of its launch c.g. to translate the carrier away from the orbiter without disturbing the orbiter flight path.

8.0 REFERENCES

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