CATALOGUED BY ASTIA
AS AD No. 333326

AD-333 266

(UNCLASSIFIED TITLE)
Environmental Control Systems Selection
for Manned Space Vehicles
Volume II: Appendix I -- Missions, Vehicles, and Equipment

TECHNICAL REPORT NO. ASD-TR-61-240, PT. II. VOL. II

October 1962

Flight Accessories Laboratory
Aeronautical Systems Division
Air Force Systems Command
Wright-Patterson Air Force Base, Ohio

Project No. 6146, Task No. 614609

(Prepared under Contract AF33(616)-8323
by Los Angeles Division, North American
Aviation Inc., Los Angeles, California
Author: R. J. Oberto

62ASRM-2390

NOTICES

When Government drawings, specifications, or other data are used for any purpose other than in connection with a definitely related Government procurement operation, the United States Government thereby incurs no responsibility nor any obligation whatsoever; and the fact that the Government may have formulated, furnished, or in any way supplied the said drawings, specifications, or other data, is not to be regarded by implication or otherwise as in any manner licensing the holder or any other person or corporation, or conveying any rights or permission to manufacture, use, or sell any patented invention that may in any way be related thereto.

This document contains information affecting the National defense of the United States within the meaning of the Espionage Laws, Title 18, U.S.C., Sections 793 and 794. Its transmission or the revelation of its contents in any manner to an unauthorized person is prohibited by law.

Qualified requesters may obtain copies of this report from the Armed Services Technical Information Agency, (ASTIA), Arlington Hall Station, Arlington 12, Virginia.

Copies of this report should not be returned to the Aeronautical Systems Division unless return is required by security considerations, contractual obligations, or notice on a specific document.

FOREWORD

This appendix to "Environmental Control Systems
Selection for Manned Space Vehicles" has been separated
from the main volume and classified principally because
of the possibility of suggestion or revealing portions
of Air Force planning programs or underlying concepts.
This report, as well as the main report to which it is
appended, is one of a series on space vehicle thermal
and atmospheric control systems.

This report is classified SECRET because it describes an
offensive weapon system.

ABSTRACT
This abstract is Unclassified

Four versions of manned orbital reentry basepoint vehicles are developed for the purpose of providing reference points for determination of the thermal and atmospheric control requirements of realistic vehicles. Two of the vehicles (i.e., Vehicles 1A and 1B) were developed in Phase I of this study series (ASD Technical Report 61-240, Pt I) and will only be summarized here. The remaining two vehicles (Ballistic Reentry and Lenticular Reentry) are presented in greater detail in this report.

In addition to the development of specific vehicles, general data have been compiled on the more important aspects of manned space vehicle design, (i.e., flight vehicle power, structures, effects of meteoroids, mission equipment, and examination of these general data for environmental requirements.

PUBLICATION REVIEW

This report has been reviewed and is approved.

FOR THE COMMANDER:

WILLIAM C. SAVAGE
Chief, Environmental Branch
Flight Accessories Laboratory

TABLE OF CONTENTS

(CONTINUED FROM VOLUME I)

ILLUSTRATIONS

LIST OF SYMBOLS

A	-	area	ft^2
C_D	-	drag coefficient	--
C_L	-	lift coefficient	--
L	-	lift	lbs
D	-	drag	lbs
d	-	diameter	ft
h_w	-	wall enthalpy	Btu/lb
h_s	-	stagnation enthalpy	Btu/lb
h_{eff}	-	effective heat of ablation	Btu/lb
$\dot{q}_o$	-	stagnation point heating rate	Btu/ft^2 sec
s	-	ablator thickness	ft
T	-	temperature	$°R$
t	-	time	seconds
V	-	vehicle velocity	ft/sec
W	-	weight	lbs
γ_e	-	flight path angle	degrees
ρ_∞	-	free stream (ambient) density	lbs sec^2/ft^4

Appendix I

MISSIONS, VEHICLES, AND EQUIPMENT

As stated in Section II of Volume I, the purpose of developing specific hypothetical vehicles is to provide tangible reference points for determination of the thermal and atmospheric control system requirements of realistic manned space vehicles. These vehicles serve as a means for: (1) identification of environmental factors such as cabin heat rejection, solar and aerodynamic heating, cabin pressure losses, and cabin atmospheric contamination; (2) establishment of environmental requirements of crew and equipment; (3) integration of thermal and atmospheric control systems into realistic vehicles; and (4) development of trade data useful in selecting and sizing thermal and atmospheric control systems.

The scope of this portion of the study is limited to the time period 1965 to 1975. Thus, projects Mercury and Dyna Soar are considered to be pre-1965, while planetary entry and landing missions are considered to be post-1975 (References 1 and 2).

Primary emphasis to date has been placed on the manned orbital reentry vehicle whose mission would be global surveillance and/or bombardment. This has been done inasmuch as such a vehicle is probably of the greatest immediate military interest and inasmuch as such a vehicle also serves as an excellent model for thermal and atmospheric control system design studies. Four variations of this orbital reentry vehicle were developed to establish the influence of crew size, mission duration, mission equipment, and flight vehicle power on thermal and atmospheric control systems. These three uses of the manned reentry vehicles were developed in detail sufficient to accomplish the purposes stated at the beginning of this appendix.

For convenience, the three subclasses of the manned orbital reentry vehicles have been designated as follows:

Vehicle 1A -- Five-man, 6-week, full-surveillance version

Vehicle 1B -- Two-man, 1-week, full-surveillance version

Vehicle 2A -- Five-man, 6-week, full-surveillance version

Vehicle 3A -- Four-man, 6-week, bombardment version

- 1 -

All four versions are boosted, orbital configurations
with pilot controlled reentry trajectory. Vehicles 1A and 1B
are winged, Vehicle 2A, ballistic, and Vehicle 3A, lenticular
shaped. Vehicles 1A and 1B are briefly outlined here. A
detailed description will be found in Reference 3. Vehicles
2A and 3A are discussed in more detail in this section of
the appendix.

LENTICULAR REENTRY VEHICLE

The overall weapon system concept results in a requirement for three basic orbiting components. First, there is a requirement for a manned bombardment vehicle which houses the basic control function in space. Secondly, a weapon cluster is required. This is an unmanned weapon carrier which combines and integrates several weapons into a common orbiting package to facilitate handling and servicing. The third requirement is the weapon itself (Reference 6).

Manned Bombardment Vehicle

The bombardment and control vehicle is the primary element of the system, since it actually contains weapons within itself as well as exercises control of the remainder of the weapons clustered near it. This vehicle, housing the human element and the basic operational equipment, is the only portion of the system which is required to effect repeated reentries through the earth's atmosphere. Thus, the design criteria for this element of the system is much more severe than those criteria imposed on the other vehicles. The lenticular control vehicle, then, becomes the system of primary concern in describing conceptual designs for thermal and atmospheric control subsystem studies.

The structural criteria presented here are primarily based upon the temperature and loading requirements associated with the reentry phase of the mission and upon the injection boost loading. Secondary requirements include adequate insulation characteristics to assist in the control of the internal environments, and special situations, such as meteoroid encounter and crew escape provisions.

General Arrangement

The manned bomber is a lenticular shaped reentry vehicle as shown in Figure 13. The disc-shaped configuration was chosen for its greater usable volume available for weapon storage and crew accommodations and for other advantages. The volumetric efficiency of the disc with respect to a cylindrical body is shown graphically in Figure 14. It has a basic diameter of 40 feet and a gross launch weight of about 45,000 pounds. The vehicle functions as a manned orbital bombing system with an internal armament load of four winged reentry weapons and also acts as an orbital control and maintenance center for additional unmanned weapon clusters.
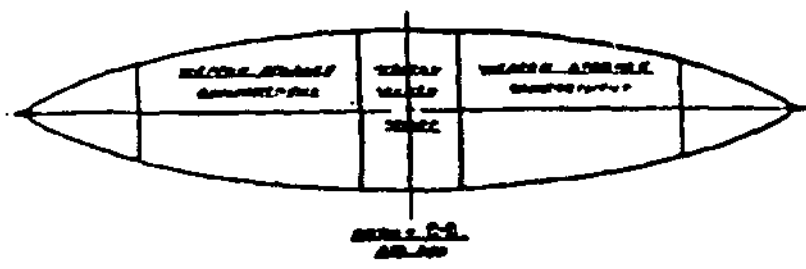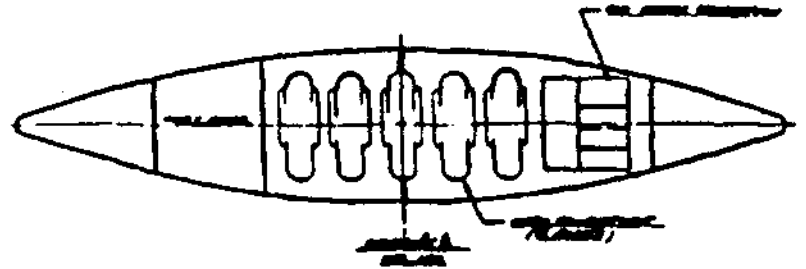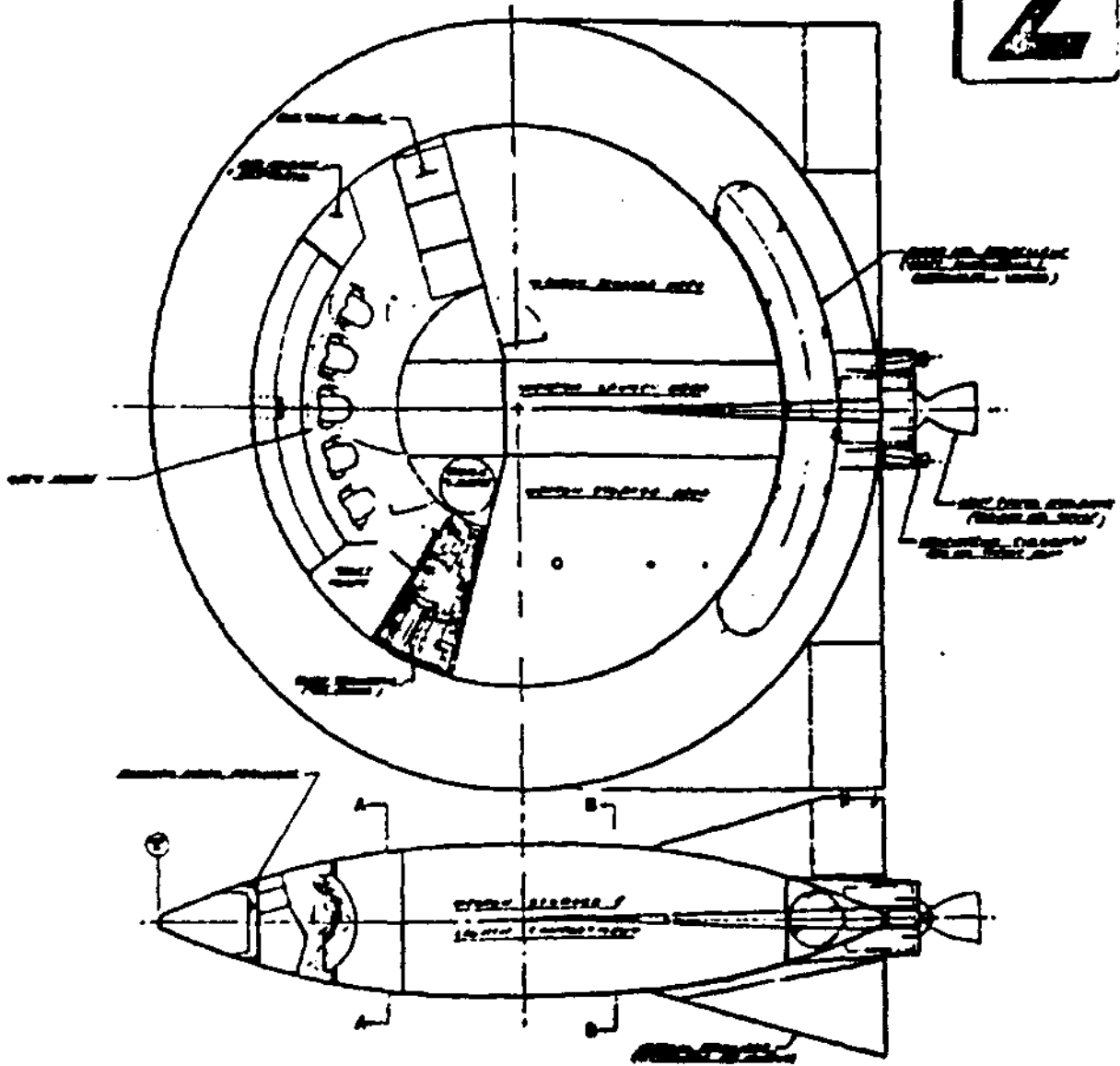
Figure 13   Disc Shape Design Study
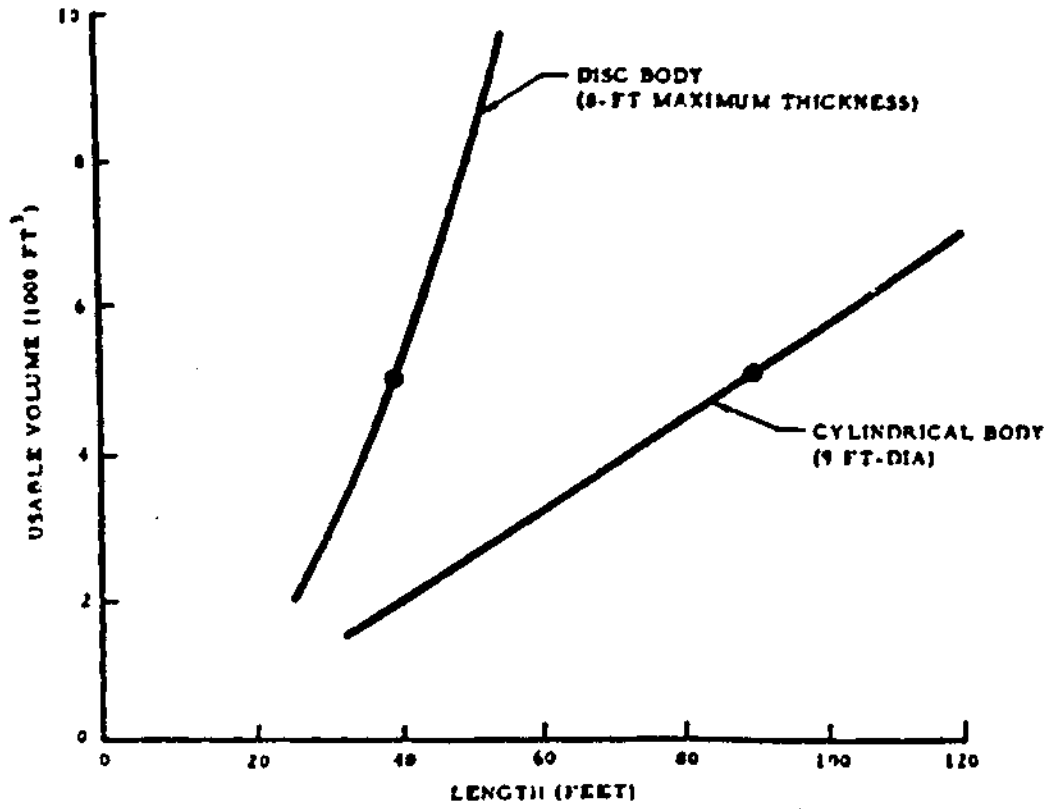
Figure 14. Volumetric Efficiency

The operational mission design is 6 weeks duration at a nominal orbital altitude of 300 nautical miles, with a crew of four men. The vehicle incorporates a jettisonable crew compartment capsule for emergency crew escape and a space shuttle service vehicle for inspection and maintenance of the weapon clusters.

External Configuration

The external configuration of the manned bomber is shown in Figure 15. The total projected planform area is 1548 square feet which results in a normal reentry and landing wing loading of approximately 23 pounds per square foot. The projected planform area noted includes the horizontal area between the disc and the straight trailing edge, which provides the flap area and also includes the movable horizontal stabilizer surface located outboard of the vertical stabilizer surfaces. The hinged trailing edge surfaces inboard of the verticals function as elevators during subsonic flight and landing. The all movable horizontal surfaces located outboard of the verticals can be actuated differentially to provide roll control as well as simultaneously to provide pitch control, and are effective throughout the entire flight regime. The landing gear consists of retractable skids mounted on the lower fuselage surface.

With the gross launch weight of the manned bombardment vehicle at 45,000 pounds, the useful load is 27,958 pounds, including 8056 pounds for four winged weapons. A weight summary of the manned vehicle including a weight breakdown of the major components is presented in Table 9.

Internal Arrangement

The internal arrangement of the vehicle is shown in Figure 16. The vehicle is designed with four primary internal compartments; the living quarters, work area, armament bay, and crew escape capsule. Each compartment can be individually isolated from the other three, should a puncture or leak make one temporarily unusable. Repairs could then be made to the damaged compartment while the crew used the remaining compartments for living and working quarters.

Crew Escape Capsule

The crew capsule functions as the vehicle control center during normal operation and as the crew escape capsule for an emergency abort of the mission. The escape capsule is actually a breakaway nose section of the vehicle, approximately 17 feet long and 6 feet wide. The general arrangement of the escape capsule is shown in Figure 17, and a weight summary is presented

1

TARGET/CARGO SECTION

WING AREA (INCL NOSE AND BODY)
ASPECT RATIO - 1.98
TC - .04
AIR FOIL - NACA 65 UPPER 010

ALL MOVABLE SURFACE (ELEVON)
BOTT SA (2)

C/4

STATIC GROUND LINE

AO

MAC

Figure 1

SHUTTLE

SECT. B-B

Figure 16. Manned Bombardment Vehicle Internal Arrangement

FINAL APPROACH CONDITION
(NOSE SECTION JETTISON &
RELEASED PARACHUTE FOR
ABORT CONDITION)

Figure 17. Crew Escape Capsule
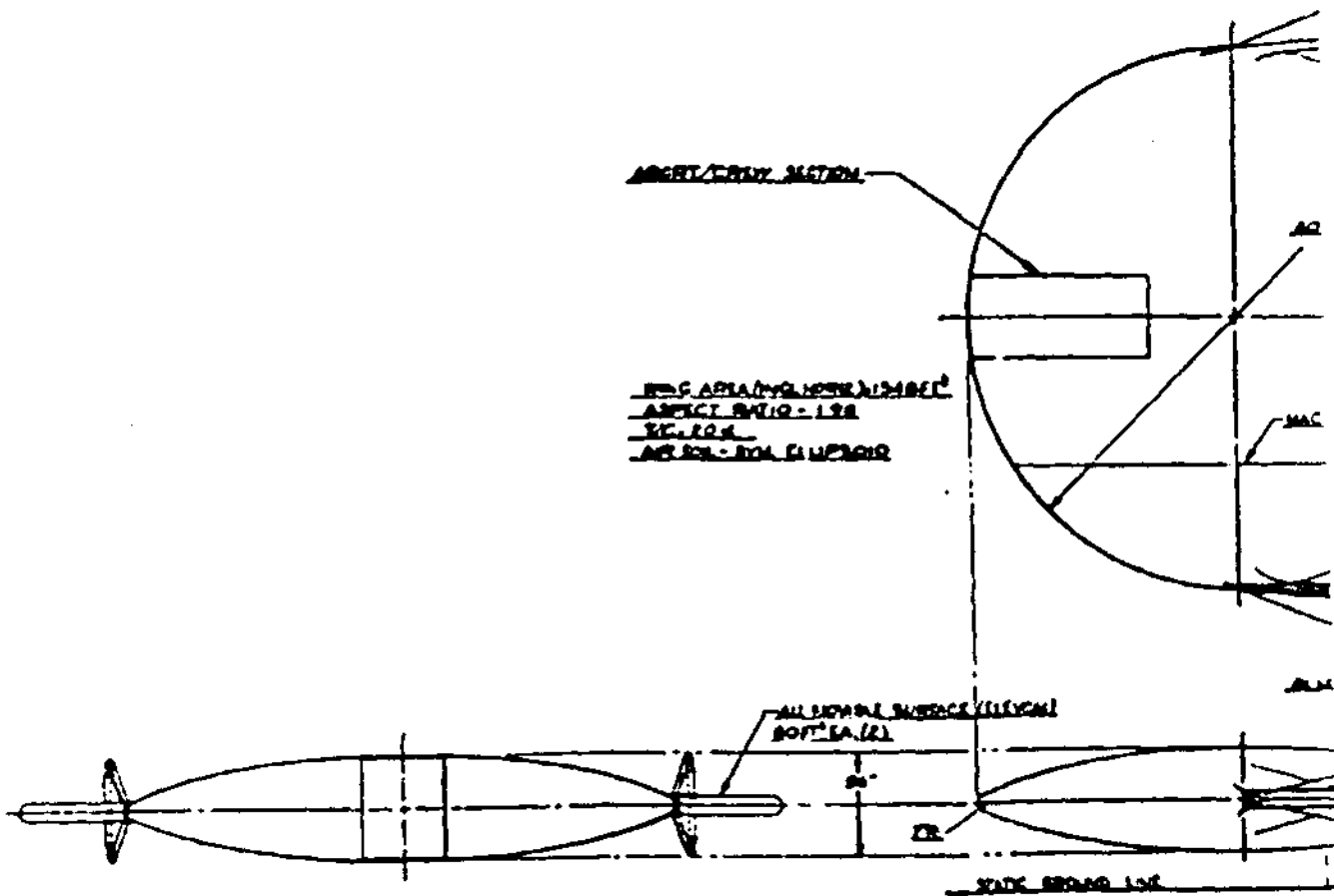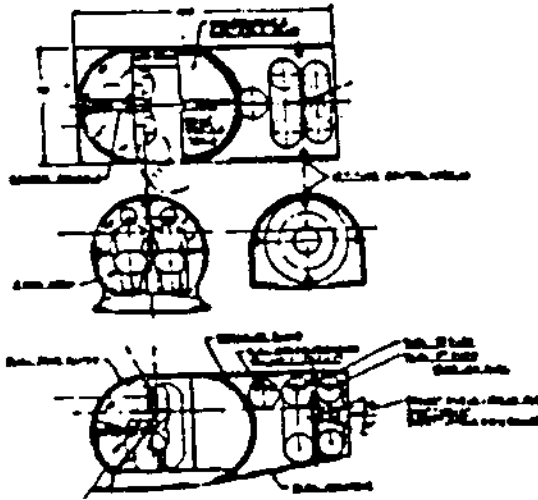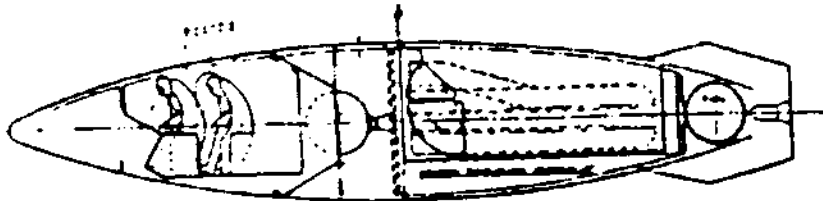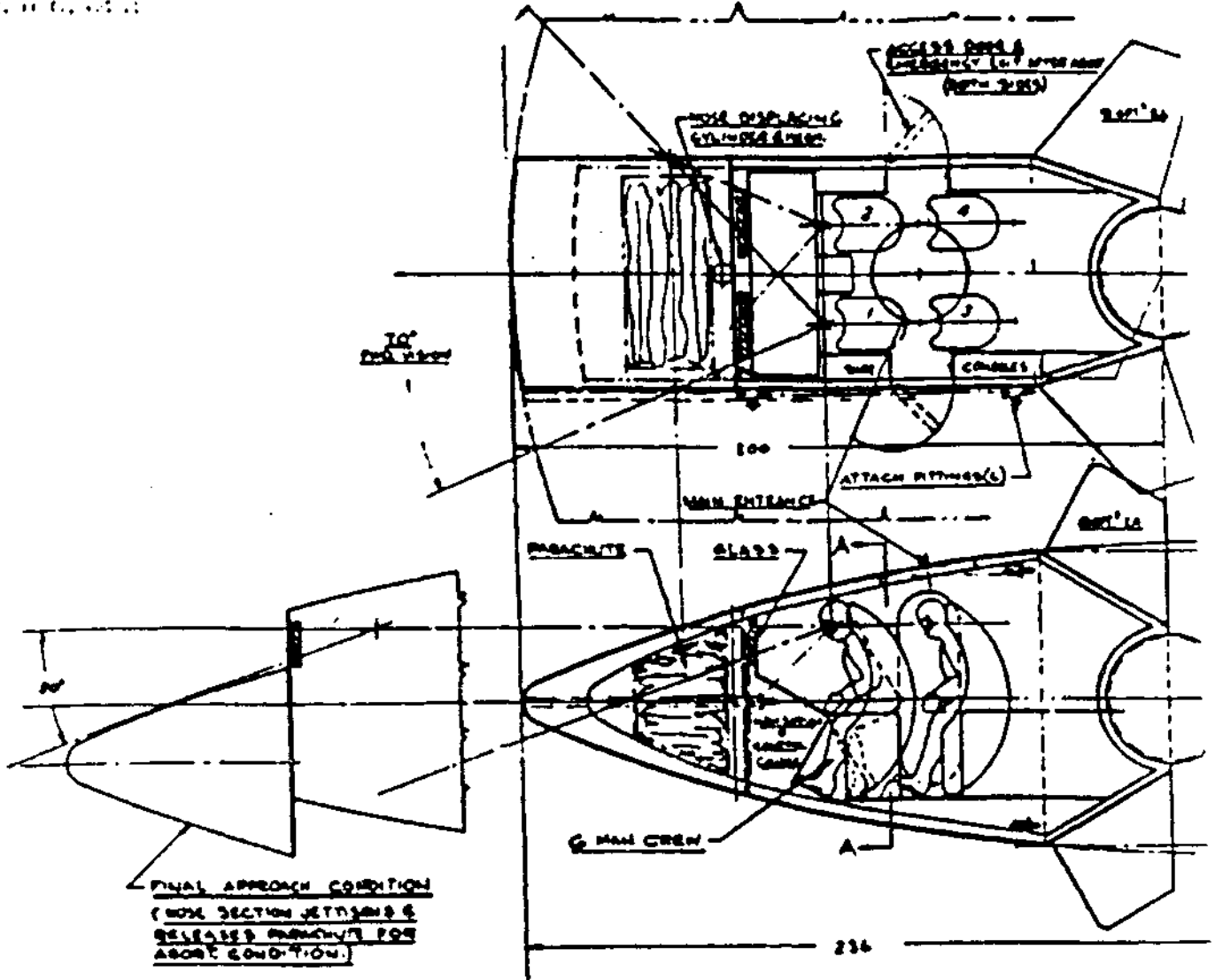
TABLE 9.

MANNED BOMBARDMENT VEHICLE WEIGHT SUMMARY

Item	Weight (pounds)	
Structure	8,740	
Propulsion	2,602	
Fixed equipment	5,700	
Weight empty		17,042
Crew (4)	1,000	
Propellant (usable)		
Retro and maneuver	9,129	
Attitude control	250	
Trapped propellant	190	
Abort vehicle (less crew)	5,000	
Service vehicle	3,756	
Pressurizing helium	51	
Water	330	
Weapons (4)	8,056	
Tools and spare parts	200	
Useful load		27,958
GROSS LAUNCH WEIGHT		45,000
Propellant	9,375	
Service vehicle propellant	1,900	
Water	330	
Expendables		11,605
LANDING WEIGHT		33,395

in Table 10. The capsule separates the crew living quarters from the work area and provides access to either compartment through a door in each side of the capsule. A sealant material around the doors prevents leakage of the cabin air during normal operation and is broken only when the capsule is separated from the primary vehicle. The abort capsule contains the crew stations for boost and reentry, the vehicle control consoles, and the emergency power supply and life support equipment needed for an abort mission. Exit from the capsule after an abort is provided by the doors on each side which serve as the passageway doors between the compartments in normal operation. Separation of the capsule from the main bomber vehicle is accomplished with a 50,000-pound thrust, spherical, solid propellant rocket engine located on the aft section of the escape capsule. The engine provides a burning duration of about 10 seconds. The abort engine imparts an initial acceleration of approximately 8.5 g to the capsule which is sufficient to escape the 5-psi overpressure wave resulting from an explosion in the first stage booster during launch from the pad. The 10-second burning duration ensures sufficient altitude above the pad at burnout to permit recovery of the capsule with a parachute located in the forward nose section of the capsule, which is enclosed with a jettisonable nose fairing. The recovery chute is sized to limit the impact velocity of the capsule to approximately 25 feet per second.

Stabilization of the capsule is accomplished with four foldout fins located on the aft section. Over-nose vision for a normal vehicle landing is provided by translating the nose section downward and exposing a flat plate windshield in the forward pressure bulkhead of the capsule. Utilization of this window for observation from the crew compartment during the orbital mission is also permissible. The environment of the crew capsule, living quarters area, and working area compartments is maintained for "shirt-sleeve" operation during the orbital mission; however, all compartments contain sufficient volume for crew operation in space suits.

Off-Duty Area

The off-duty area is the living quarters for the crew and is located on the starboard side of the vehicle. This area is designed with sufficient volume and flow area to provide comfortable living quarters and recreational area for the crew during the extended mission duration of 6 weeks. The compartment contains sleeping and sanitation provisions, food storage and separation facilities, storable table, chairs, and exercise equipment. Sleeping nets with storage space below for

TABLE 10
ESCAPE CAPSULE WEIGHT SUMMARY

Item	Weight (pounds)	
Structure	1886	
Propulsion	109	
Fixed equipment	920	
Weight empty		2915
Crew (4)	1000	
Propellant	2085	
Useful load		3085
GROSS LAUNCH WEIGHT		6000
Propellant	2085	
Expendables		2085
LANDING WEIGHT		3915

personal gear, books, etc., are located along the wall. Food storage and preparation facilities, plus miscellaneous storage provisions, are located along the other wall. An air lock compartment provides access from the pressurized off-duty area to the unpressurized weapons bay, or to the outside of the vehicle. Access to the on-duty work area is through the vehicle control compartment.

On-Duty Area

The on-duty work area is located on the opposite side of the vehicle. The work area contains all the display consoles and control equipment for launching, monitoring, and controlling the weapons and controlling the unmanned weapon clusters, the primary electronic equipment, and miscellaneous tools and maintenance equipment. A generous allowance of volume for the display consoles and electronic weapon control equipment permits easy access to all the equipment for maintenance and repair. The primary environmental control, power supply and miscellaneous vehicle support equipment is located around the forward periphery of the off-duty and on-duty compartments. The central floor area of the work area is kept open to permit freedom of movement for the crew for maximum efficiency and convenience. Access to the outside and to the weapon storage area is provided with an inter-connecting air lock compartment.

Weapon Storage Area

The weapon storage compartment is located in the aft section of the vehicle on each side of the weapon launch bay and provides storage space for four winged weapons - two on each side of the displacing mechanism which is located on the lift coefficient of the vehicle. The weapon storage area is an unpressurized compartment and is accessible from both the on-duty and off-duty crew compartments through air lock compartments.

Sufficient space is provided for access to the weapons in the storage area for maintenance, repair, checkout, and launch procedures. The weapons are supported on rails in the storage area which permit manual transfer of the weapons from the stored position to the displacing mechanism for launch. During the orbital phase of the mission, all weapons will be transferred, via the displacing mechanism, after the shuttle vehicle is removed, and attached to the external surface of the vehicle in ready position. The shuttle is then returned to the displacing mechanism and the vehicle is ready to initiate a strike using the attached weapons or using remotely located clustered weapons. The vehicle is also in position to effect an immediate reentry, in case of emergency, by simply detaching

the four weapons and leaving them in space to be recovered at a later time.

An alternate use of the weapon storage area will be as a logistic cargo compartment for transporting fuel and replacement parts as needed for service and repair of the unmanned weapon clusters.

Flight Performance

Aerodynamics

Primarily because of its excellent surface area-volume-weight relationship, the lenticular shape has been chosen as its satellite-reentry configuration for the manned bomber. The basic disc shape is inherently unstable assuming a representative center of gravity location. However, control surfaces, flaps, and speed brakes suitably located and configurations tailoring can make the lenticular shape stable and, with other desirable characteristics, a very satisfactory manned reentry and landing configuration will evolve. A Global Surveillance System using this same basic lenticular concept is described in Reference 7.

Subsonic L/D_{max} is approximately 9, which will result in excellent landing characteristics. The L/D_{max} at supersonic speed is about 2.0, and at hypersonic speeds this value would decrease to about 1.5 which is ample for reentry maneuvering.

Entry Performance

For all the entry trajectories, it was assumed that the maximum lift coefficient capabilities of the vehicle was 0.70 occurring at an angle of attack of 51 degrees. The maximum L/D of 1.5 occurs at $\alpha = 14$ degrees which corresponded to a lift coefficient of 0.28. Four basic entry trajectories are presented in Figures 18 through 21. In Figure 18 a manned entry is shown initiated by a retrograde impulse of 200 feet per second below circular orbit speed at 400,000 feet. The entry is flown at a constant maximum lift-coefficient and results in the coolest leading-edge temperatures of those trajectories studied. Two alternate entries are shown in Figures 19 and 20. These trajectories were initiated with similar conditions, but lift is arbitrarily varied to represent a maneuvering entry. Leading-edge temperatures encountered during these entries are considerably higher and have been accounted for in the structural design. Figure 21 presents an entry initiated by a 500-foot-per-second retrograde impulse in order that entry time may be reduced. This may be desirable due to system malfunctions or solar flare and radiation warnings.

Figure 28. Reentry From Orbit ($C_{L_{max}}$ = 0.7)
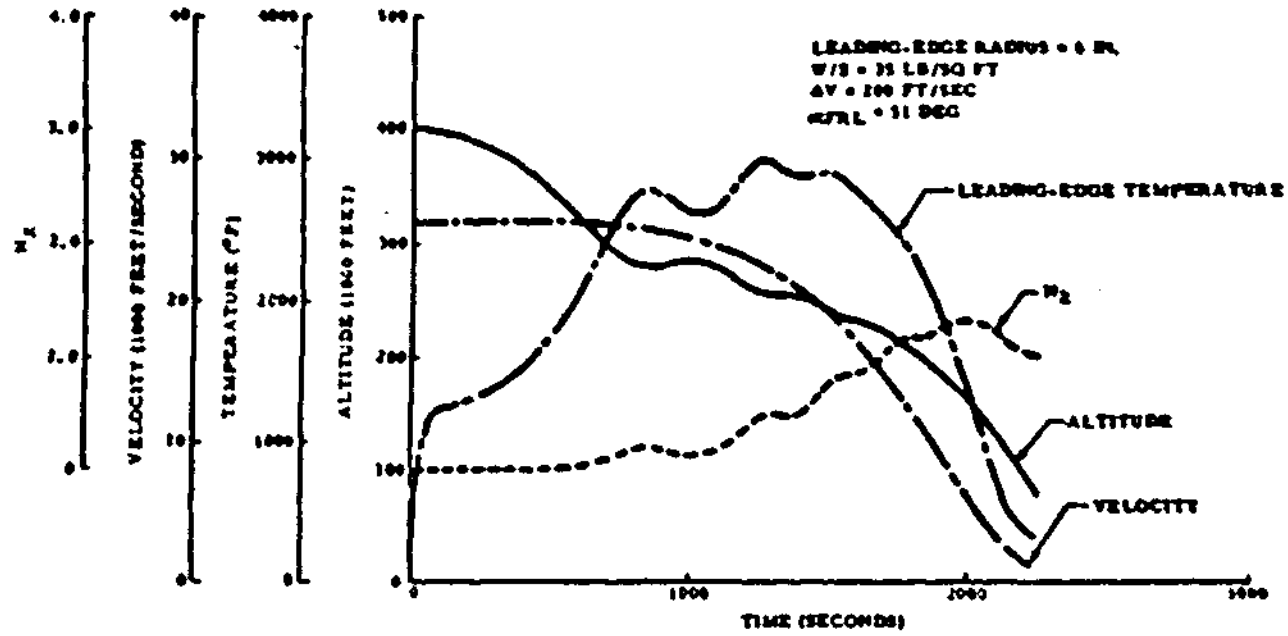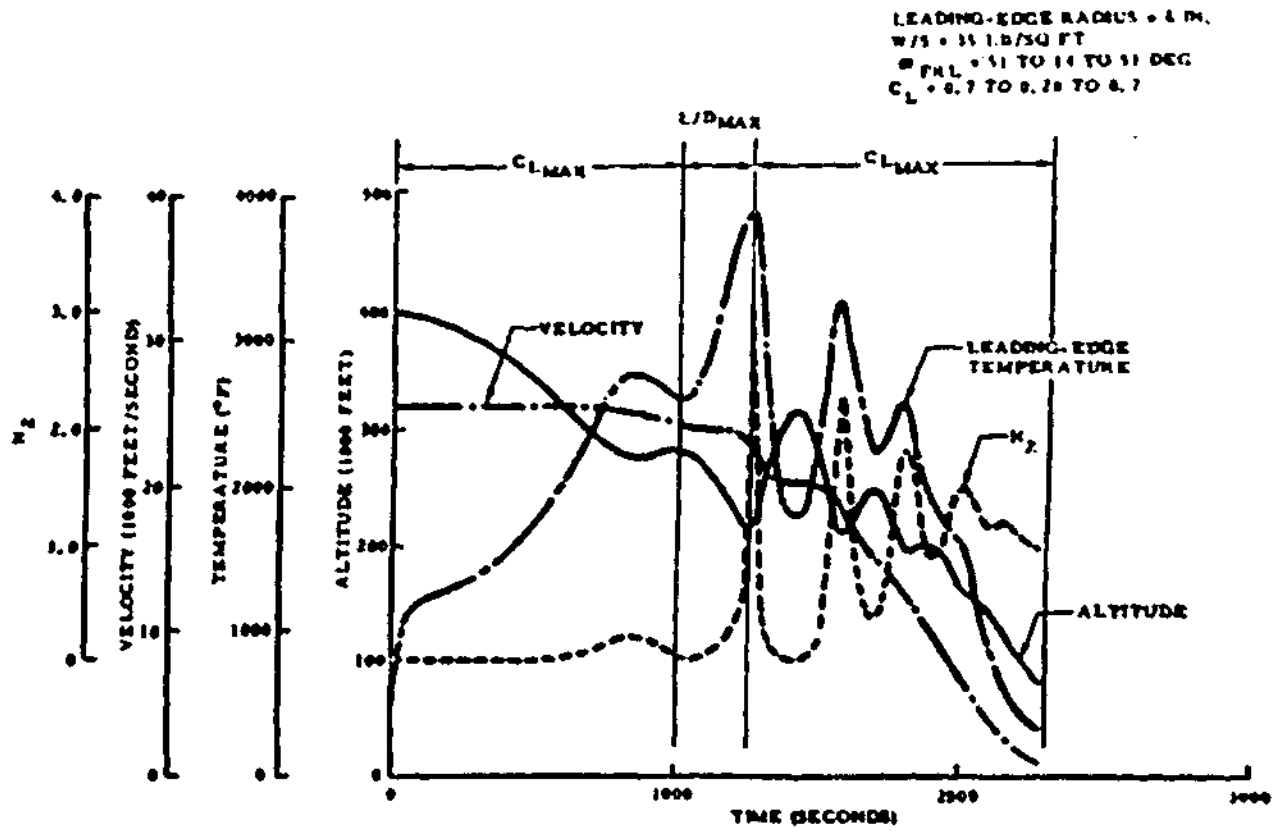
Figure 19. Reentry From Orbit ($C_{L_{max}}$ = L/D_{max} to $C_{L_{max}}$)
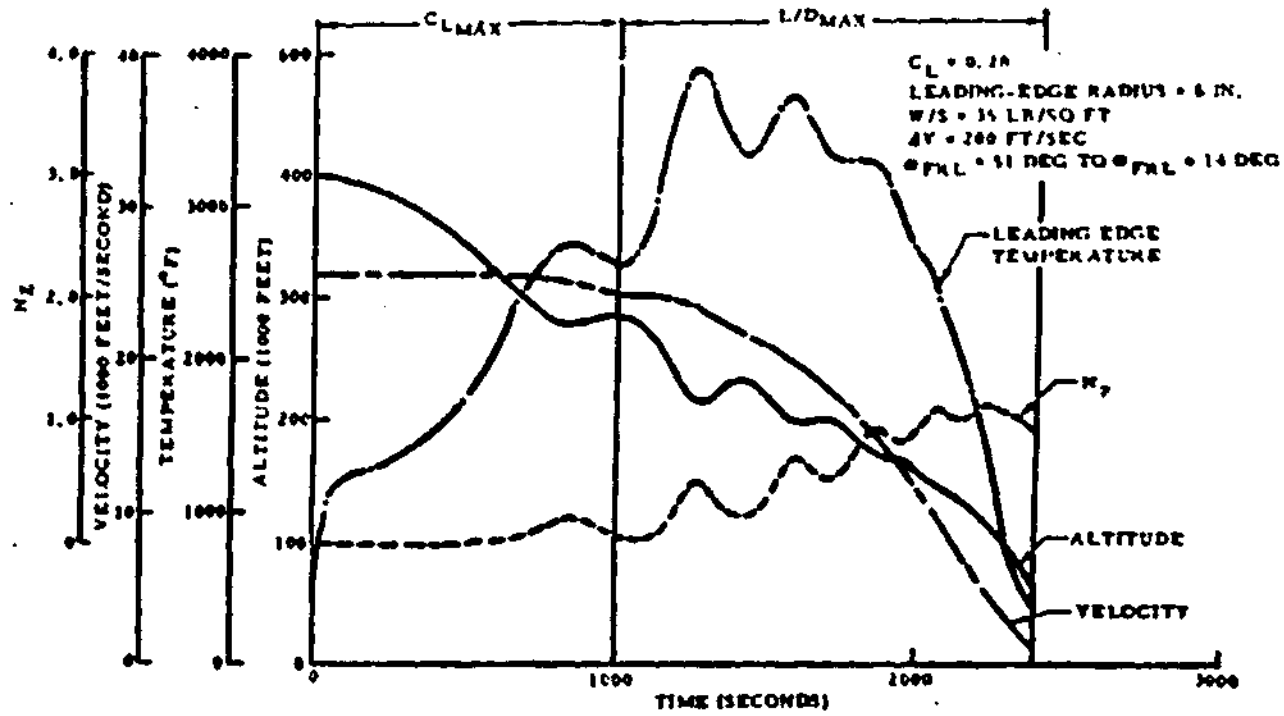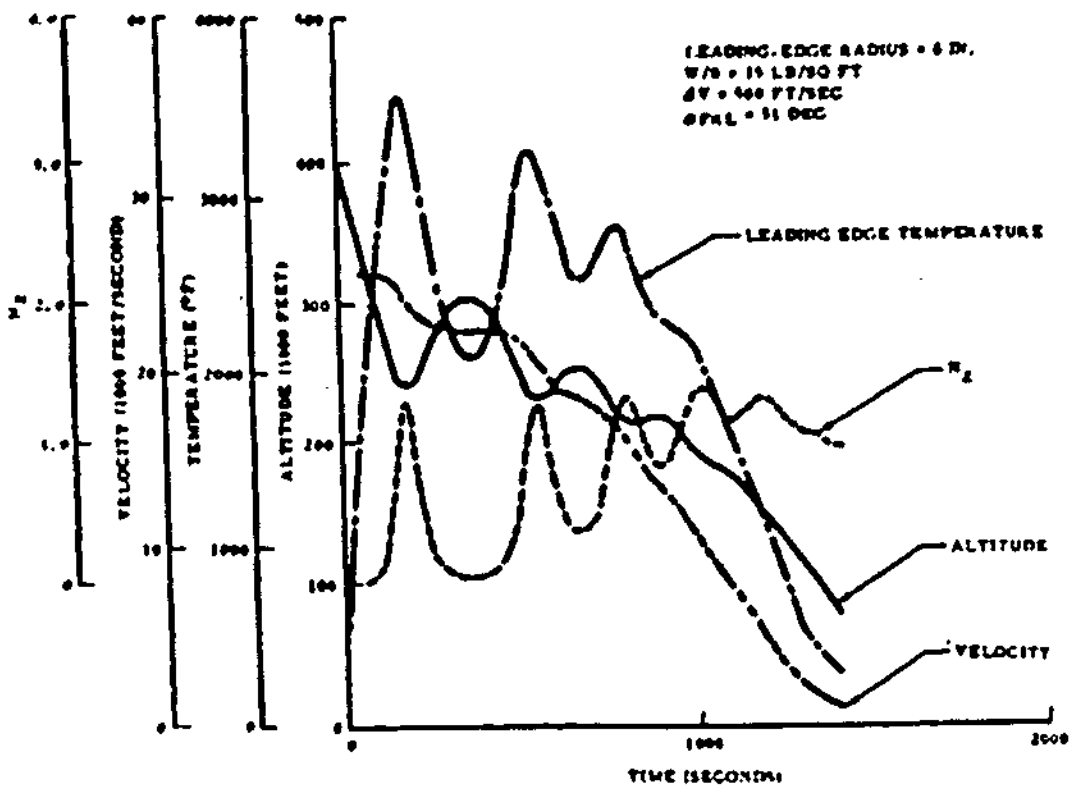
Figure 20. Reentry From Orbit ($C_{L_{max}} = 0.7 \; L/D_{max}$)

Figure 21. Reentry From Orbit ($C_{L_{max}}$ = 0.7; 500 Feet Per Second ΔV)

Thermodynamics

The lenticular-shaped reentry vehicle is favorable from a heating standpoint, when compared to other reentry vehicle configurations. The leading edge of the disc may be treated as a cylinder normal to the flow, which results in two-dimensional flow is a reduction of the aerodynamic heating rate to the leading edge of $1/\sqrt{2}$ or 0.707 of the heating rate that would normally reach a hemisphere or blunted cylinder during a similar entry mission.

The disc-shaped configuration with control surfaces on the aft portion of the vehicle eliminates the problem of high heating due to low shock interactions between conventional fuselage nose and wing leading-edge surfaces. This problem is common to winged body lifting vehicles.

Another advantage of the disc configuration is that large leading-edge radii may be achieved. This results in a reduction of the aerodynamic heating rate that would be experienced by thin-winged entry vehicles.

Structural Design

The manned bombardment vehicle must be capable of supporting its flight and dynamic loadings throughout the mission profile without sustaining permanent damage. The mission profile of this vehicle starts at the launching pad, continues through 6 weeks in orbit, through reentering the earth's atmosphere, and concludes with maneuvering to a landing. After each mission, the airframe will require visual inspection to detect any meteoroid damage or surface erosion. These minor damages should be the only airframe service required prior to the next flight.

In order to provide a reliable vehicle for the accomplishment of the above mission the structural design must follow a rigorous program involving criteria, materials, and structural arrangement. The criteria is studied in four phases that are consistent with the operational mission of the vehicle.

Ground Launch — A maximum axial load factor of $NX = 8.0$ g (limit) is used. This load factor also accommodates engine thrust overshoot.

The structural temperature is unspecified, but is assumed to be less critical than the reentry temperature. The above load factor is applied to the vehicle by the interstage structure through the main fore and aft longitudinal beams that are adjacent to the personnel escape module. These beam caps are reinforced locally at the junction to the interstage and the webs also serve as pressure isolation members in the vehicle cabin area.

Wind shears associated with the boost phase induce the maximum bending moment in the interstage structure.

Orbital Flight — During this phase of the mission there are essentially no loads on the vehicle other than control thrusts. These loads are so small that the structure, other than the immediate attachment, is designed by some other criteria. The primary structural problem during orbital flight is the provision of a non-leaking pressure-tight cabin.

Reentry — Preliminary reentry trajectory studies, including the effects of vehicle geometry, showed that a leading-edge radius of 6 inches was sufficient to keep the temperature within the limits of coated graphite. Using this radius, time-temperature-load factor profiles were established for several different reentry trajectories. The limits of these trajectories were the maneuvering range from $C_{L\ max}$ to L/D_{max}. The $C_{L\ max}$ reentry generates minimum structural temperatures, but does not allow the maneuverability of lower C_L profiles. Emergency reentry, utilizing maximum retro thrust, generates higher temperatures which are compatible with L/D_{max} reentry temperatures. Maneuvering from L/D_{max} to $C_{L\ max}$ provides design points for combinations of high temperature and high loads. The reentry profile for this condition is shown in Figure 19 and the resulting fuselage lower surface and upper surface temperature profiles are shown in Figures 22 and 23, respectively.

The cabin is pressurized to 10 psi limit continuously from 10,000 feet altitude in the boost phase through the 6-week orbiting flight and reentry back to 10,000 feet altitude.

Landing — The normal landing load factor is limited to 2.0 g limit by pneumatic landing gear struts.

$C_L = 0.15$
LEADING EDGE RADIUS = 6 IN.
W/S = 35 LB/SQ FT
θ_{FRL} = 14 DEG
ALTITUDE = 216,126 FT
VELOCITY = 23,118 FT/SEC

Figure 22. Lower Surface Temperature

L/ D$_{max}$

C_L = 0.15
LEADING EDGE RADIUS = 6 IN.
W/ S = 35 LB/ SQ FT
ΘFRL = 14 DEG

ALTITUDE = 216,126 FT
VELOCITY = 23,118 FT/ SEC

BP 216

BP 180

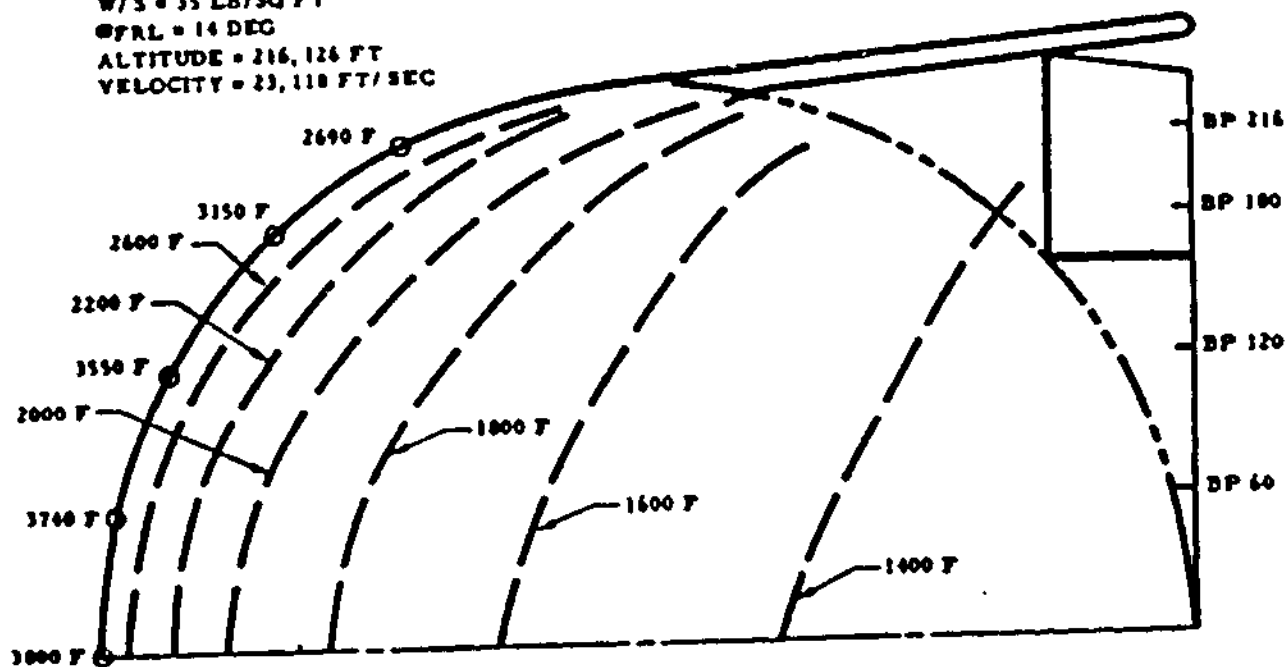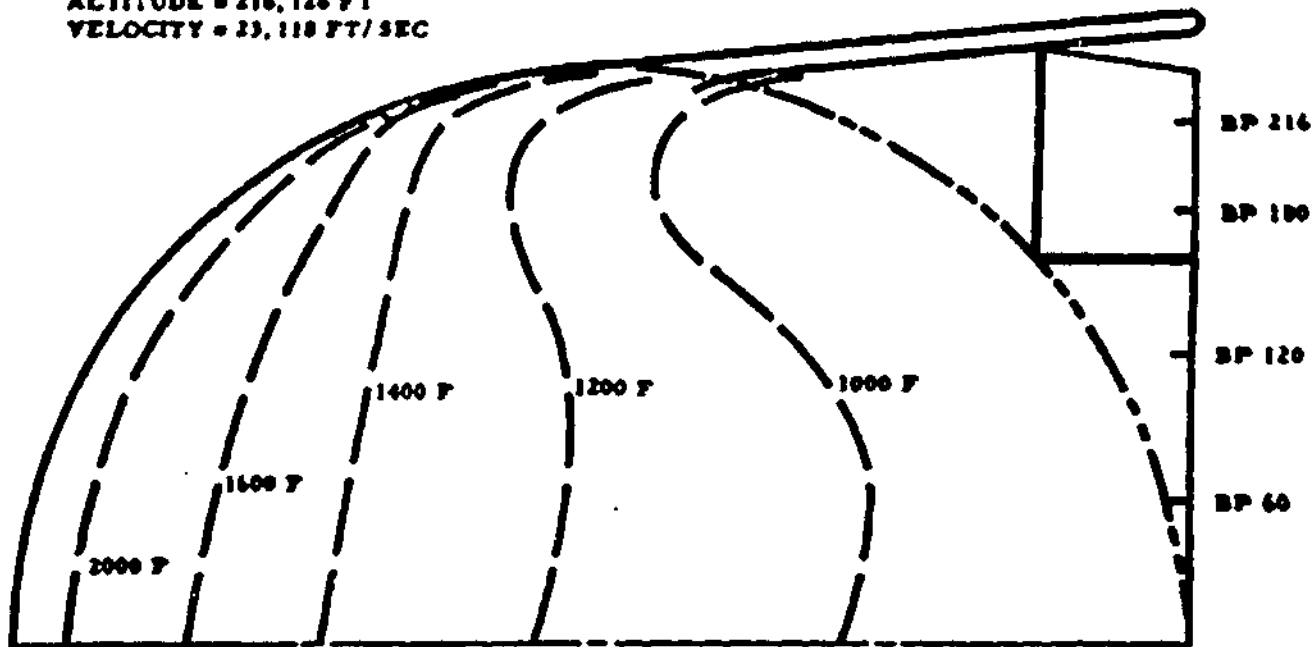BP 120

BP 60

1400 F   1200 F   1000 F

1600 F

2000 F

- 35 -

Figure 23. Upper Surface Temperature

Propulsion Systems

There are three basic propulsion systems contained within the manned bomber; the basic vehicle maneuvering system, the shuttle vehicle maneuvering system, and the crew abort system. The crew abort system is a single-purpose system while the manned vehicle maneuvering system is integrated with the shuttle vehicle system, as well as with the propulsion systems contained in the unmanned weapon carrier or cluster, and in the weapons themselves.

All of these systems (manned vehicle, shuttle vehicle, clusters, and weapons) utilize the same storable, hypergolic propellant combination; nitrogen tetroxide and hydrazine, N_2O_4/N_2H_4. The main propellant storage is in the manned vehicle, which has a capacity for 9375 pounds of propellant. The shuttle vehicle capacity is 1900 pounds, while the cluster and weapon (each) capacities are 700 pounds and 200 pounds, respectively.

Because the manned vehicle contains the majority of the propellants the shuttle vehicle will be refueled from this main supply, as required. The shuttle vehicle, in turn, will top-off the supply in each cluster as it makes its periodic servicing trips. The cluster fuel supply is continuously connected to the propellant tanks of each weapon in the cluster and thus maintains the proper fuel level. If a weapon is fired at a target, or otherwise disconnected from the cluster, automatic check valves will prevent fuel leakage.

Top-off propellant requirements for the cluster and for the weapons will be small; estimated to be less than 50 pounds per complete cluster per 6 weeks period. The ratio of absorptivity to emissivity of the external surface of either spherical or cylindrical tankage can readily be established such that, in the spatial environment, the propellants can be maintained between limits of +40 and +70 F. In the case of spherical tankage this ratio will be between 1.0 and 1.4, while in the case of cylindrical tankage it will be between 4.5 and 7.0.

The basic vehicle maneuvering system is normally used to establish precision orbits immediately after injection, and to provide retrograde impulse prior to reentry. Alternate uses of the maneuvering system will include a limited orbital maneuver capability for the manned vehicle which can be used to correct its ephemeris after long period drift, or can provide maneuvering functions which may be required as a result of special situations, such as a rendezvous requirement.

This engine provides a nominal 2000 pounds of thrust at a very low chamber pressure of about 60 psi. Recent tests indicate that throttling to 5 percent of nominal thrust will not be adversely affected by this low pressure since stable burning can be attained at chamber pressures as low as 1 psi under vacuum conditions. This engine will be radiation cooled and will provide unlimited restart capability. The expansion area ratio will be about 40:1 which will give very good specific impulse characteristics as well as providing extensive area for cooling.

The propulsion system required for the shuttle vehicle develops a nominal thrust of 200 pounds. Detail characteristics of this engine are identical to those of the manned vehicle system except for thrust level.

The abort engine is a conventionally designed, spherical, solid propellant system producing 50,000-pound thrust at sea level. This is a "one-shot" non-reusable system with a compromise nozzle area ratio of about 8:1, because abort must be performed throughout the entire altitude range of the earth boost trajectory.

Secondary Power

The manned bomber requires two separate power systems; one for the boost and reentry phases and another for the normal 6-week orbital operation. Unfortunately, it is not feasible to provide one system which can supply the energy for both requirements.

Energy for the orbital operation can most feasibly be supplied from nuclear or solar sources because of the long flight duration and relatively high continuous power level required. Approximately 7 kilowatts will be required continuously. Neither of these systems are suitable for the boost and reentry phases.

The nuclear reactor cannot be activated until the vehicle is in orbit, and on reentry, would probably be left in space to avoid the possible hazards associated with a hot reactor should a crash occur on landing. Similarly, the solar collector of a solar turboelectric system could not be erected until in orbit, and would have to be dismantled or discarded prior to reentry. Although it might be possible to devise a heat storage reservoir of sufficient capacity with the solar system to provide boost and reentry power, this could not reasonably be placed in the control vehicle because of weight and volumetric penalties.

Orbital Power

The manned bomber will use a solar turboelectric system for its orbital phase because of present nuclear reactor shielding problems. The total weight will not exceed 800 pounds including a 27-foot diameter, hinged solar collector-radiator. This estimate is based on a 90-minute orbit with a mean 35-minute dark period.

The solar collector will be similar in construction to the SUNFLOWER design in that it will consist of hinged petals which can be expanded after an orbit is achieved. However, it will also have a 500 F radiator on the back side to which a circulating fluid will carry waste heat to be radiated into space. Orientation of the collector toward the sun within ±0.75 degree is accomplished by a reaction jet attitude control system and a sun-seeking system which are integral with the collector itself.

Light from the collector will be focused on a cavity receiver in which a lithium hydride storage unit is interposed between the interior absorbing surface and mercury boiler tubes. This storage unit will have sufficient capacity to produce the required energy for the vehicle during the dark periods of the orbits.

A binary Rankine-cycle conversion system will be utilized to change the solar heat into mechanical energy because it has high efficiency with minimum weight and size. Mercury and steam are two possible fluids for the heat exchangers.

The secondary fluid could lubricate the bearings on the hermetically sealed package which contains the primary and secondary turbines, the alternator and the pump, all of which rotate on the same shaft. Although the maximum operating temperature of this system will be 1200 F, the alternator and bearings will be subjected to a maximum temperature of 250 F.

Included in the heat engine will be a compact mercury vapor condenser, a secondary fluid boiler, a steam condenser and assorted piping, fittings, and supports.

A 7-kilowatt, 1000-csp, three-phase, 110-volt per phase alternator will transform the turbine power into electric power. Voltage and speed control regulating units will automatically maintain voltages and turbine speeds within prescribed limits.

Boost and Reentry Auxiliary Power

A silver-oxide zinc cell battery will supply the peak reentry load of 12 kilowatts for 10 minutes plus an average load of 7 kilowatts for a maximum time of 2 hours. This battery will weigh approximately 200 pounds and have a volume of 1-1/2 cubic feet. After each 6-week mission, the battery will be replaced with a new unit.

Weightwise, a cryogenic hydrogen-oxygen motor and generator would be better than a battery, and a hydrazine turbine would be as good as a battery. However, considering cryogenic storage for periods of as long as 6 weeks, a static battery system is to be preferred over a rotating turbine, on the basis of reliability and maintainability.
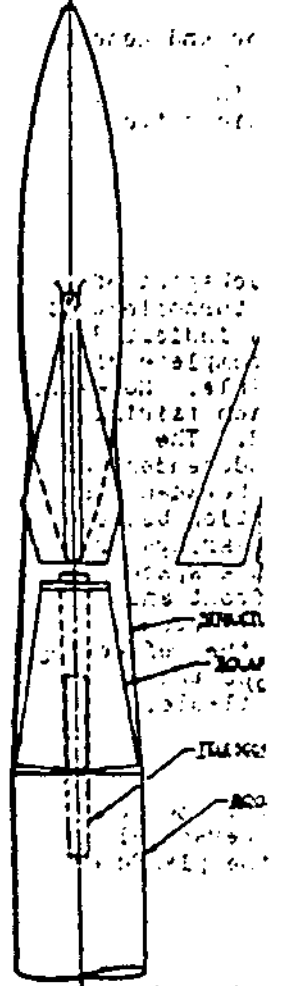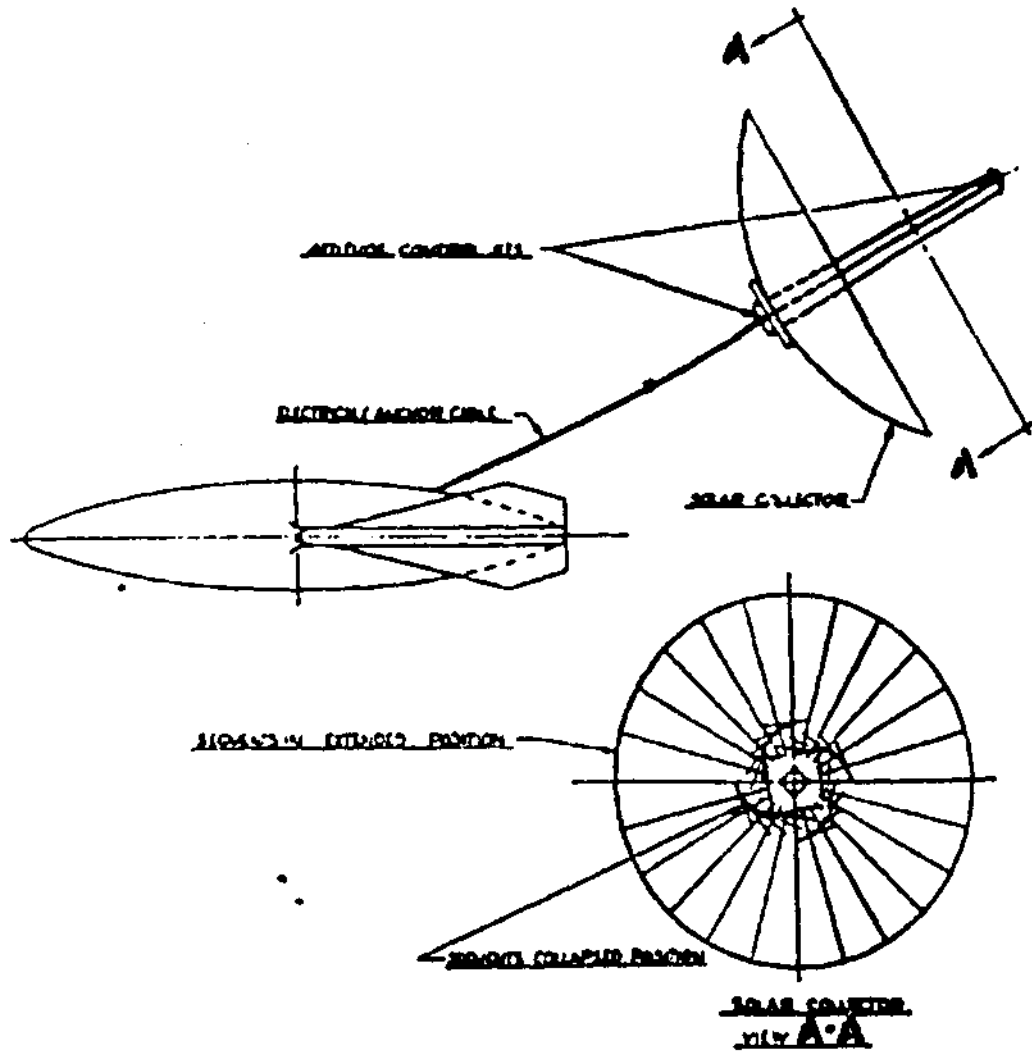
Installation

A number of difficult problems, including packaging of the array, must be solved when designing a solar turboelectric system for space. A possible packaging scheme is indicated in Figure 24. This scheme does not require the complete dismantling of the solar collector and appears feasible. However, external packaging will minimize the booster attach fairing length and may be the optimum installation method. The position and attitude of the collector will be independent of the vehicle except for the electrical connection between the collector and the vehicle. The electrical connection between the collector-radiator and the vehicle requires bearings or flexible couplings, all of which pose problems in a space environment. Temperature gradients between the front and back of the collector-radiator can distort the collector structure and degrade the optical performance of the reflecting surface. However, it is believed that solutions to these problems are possible in the development time available.

Structural Arrangement

The inner section of the vehicle (approximately 30 feet long and 30 feet wide) housing the equipment and personnel is defined as the fuselage and the remainder of the planform is defined as the wing.

Wing Structure

The wing structure is a multirib design with the ribs oriented normal to the leading edge toward the forward half of the vehicle and appearing as spars toward the rear half of the vehicle. The ribs and spars are concentrated cap design and, where needed, skin shear stiffness is achieved by adding a

ATTITUDE CONTROL JETS

ELECTRICAL ANCHOR CABLE

A

A

SOLAR COLLECTOR

STORAGE IN EXTENDED POSITION

STORAGE COLLAPSED POSITION

SOLAR COLLECTOR
VIEW A-A

STRUCT

SOLAR
TUBE

PACKAGE

ACCU

Figure 24. Solar Collector Installation

corrugated stiffening skin. The wing skins, ribs and spars are fabricated from coated columbium alloy that is allowed to heat up and reradiate. The wing structural arrangement is presented pictorially in Figures 25 and 26.

The wing leading-edge radius is 6 inches. The nose portion of the leading edge is fabricated from coated graphite and is segmented to minimize thermal strains.

Fuselage Structure

The fuselage structure is designed to function simultaneously as a load transmission structure and as a thermal protection system for the pressurized crew compartment. This structure is basically a triple wall design consisting of an outer radiation shield, a honeycomb panel primary structural shell, and an inner cabin shell. Each of these layers of the fuselage shall are separated by a layer of insulation. A typical structural arrangement of this shell is shown in Figure 27.

The outer radiation shield (reradiating surface) consists of many small independent panels that are attached to the primary shell by minimum attachment to minimize heat shorts. These panels distribute the aerodynamic loads to the primary structural shell and reach a temperature of approximately 2000 F on the vehicle lower surface and 1600 F on the vehicle upper surface.

The primary structural shell is made of honeycomb sandwich panels that are fabricated of brazed nicle base alloy (Rene' 41) face sheets and core. Panel boundaries are afforded by beams rather than tension webs to the far surface because with the more optimum panel configurations, the tension webs would render the fuselage space nearly unusable. This shell is used to react and redistribute all flight and landing loads and will, in general, be designed by cabin pressure with reinforcements and fittings as are necessitated by localized or concentrated loads. The general design of this primary shell is based on a fuselage pressure of 10.0-psi limit and an operating temperature of 1000 F. This design is shown in parametric form in the curve of Figure 28. This figure shows that the smallest feasible square panel will yield the optimum structural design. However, other criteria such as compartmentalization and equipment installation or crew movement and accessibility for service and maintenance of equipment during orbital flight necessitates a departure from optimum structural design.
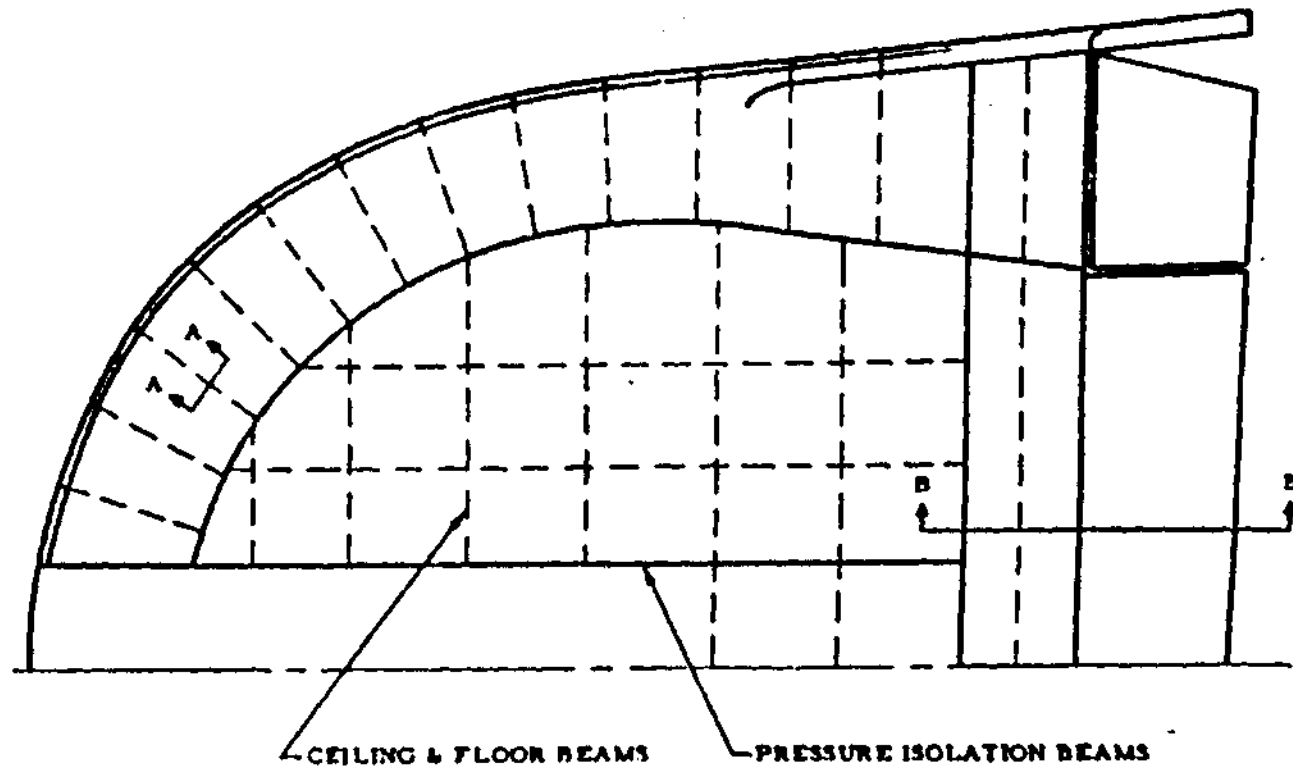
CEILING & FLOOR BEAMS   PRESSURE ISOLATION BEAMS

Figure 25. Wing Structural Arrangement

SKIN WITH MILLED LANDS
FOR RIB CAP

CORRUGATED
RIB WEB

SECTION A-A

COLUMBIUM SKINS WITH MILLED
LANDS FOR SPAR CAPS

CORRUGATED
SPAR WEBS

SECTION B-B
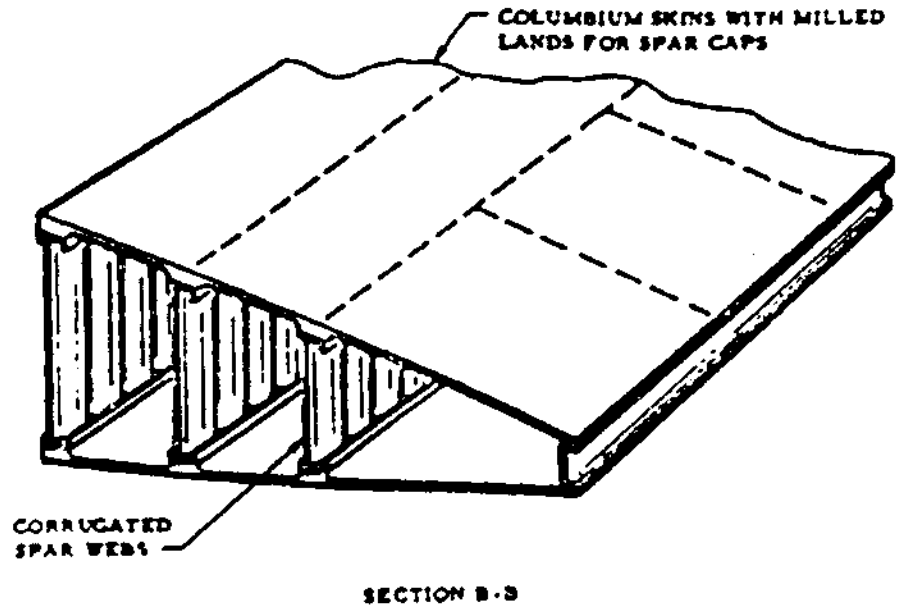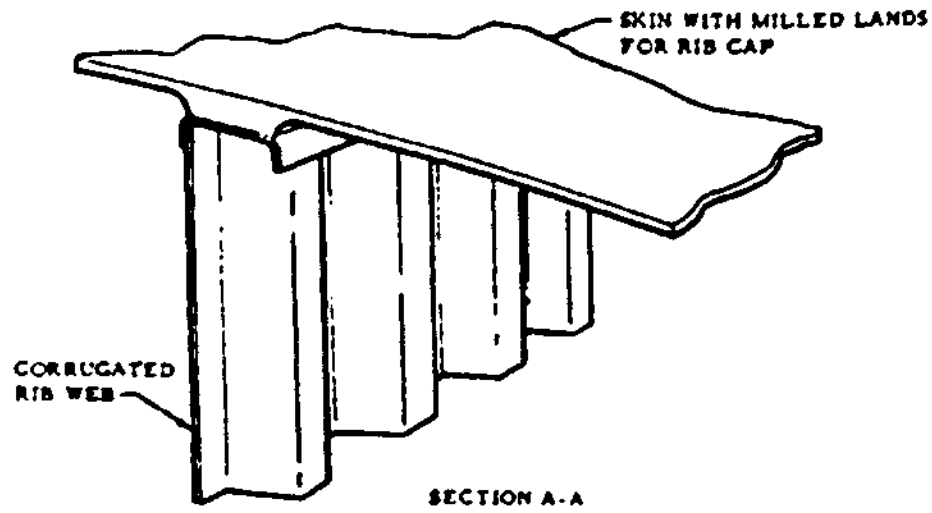
Figure 26. Wing Structural Details
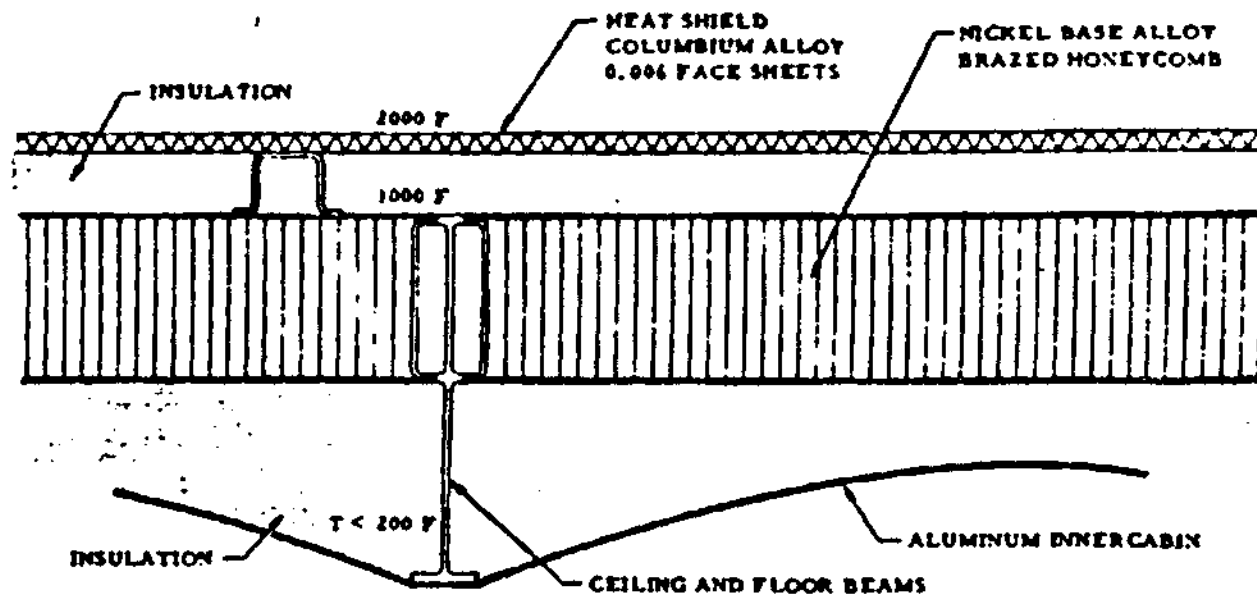
Figure 27. Fuselage Wall Construction

INSULATION

HEAT SHIELD
COLUMBIUM ALLOY
0.006 FACE SHEETS

NICKEL BASE ALLOY
BRAZED HONEYCOMB

1000 F

1000 F

INSULATION

T < 200 F
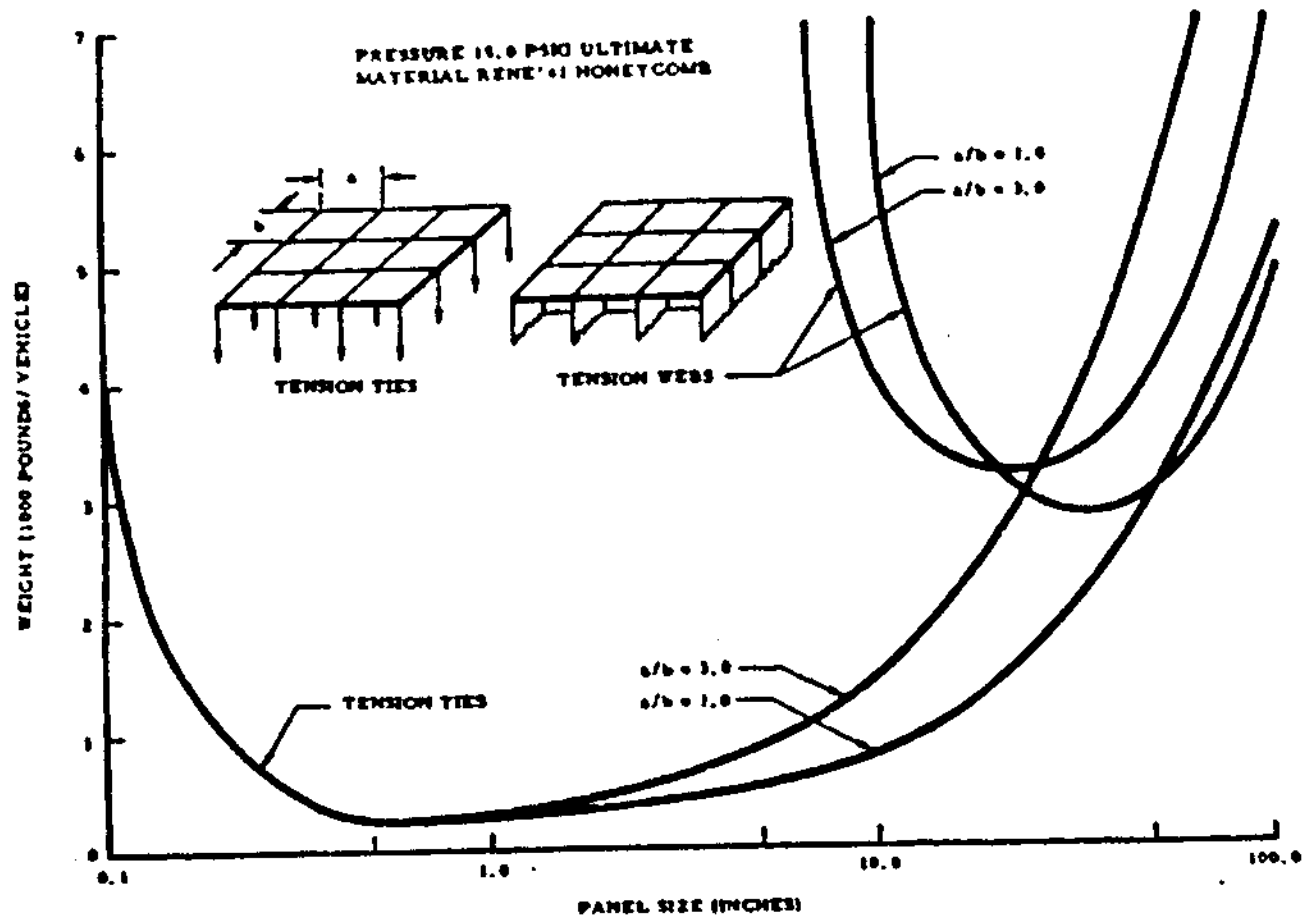
ALUMINUM INNER CABIN

CEILING AND FLOOR BEAMS

Figure 28. Fuselage Structural Shell

Thermal Shielding

Structural and environmental problems, due to the high temperatures induced during reentry, require the use of thermal shielding to control the peak temperature reached in the personnel area. In the case of wing structure, where there are no internal temperature limits, the use of shielded structure is precluded unless it proves lighter than hot radiating structure. Because the wing of this vehicle is designed as a hot reradiating structure this discussion is concerned with the fuselage portion of the vehicle.

The above problem is solved, in part, by the use of shingle-type heat shields. These shields are 24- by 4-inch rectangular sandwich panels of corrugated annealed columbium (F-48) designed to peak temperatures of about 2600 F. The face sheets are 0.006 inch thick and the shear member is a 60 degree corrugated web that is 0.003 inch thick. Between these heat shields and the primary structure is a layer of insulation which limits the temperature of the primary structure to about 1200 F. A number of heat resistant materials and alloys, among which is the nicle base alloy that was used in the design of the primary structure, have good structural efficiency in this temperature range. A layer of insulation between the primary structure and the inner cabin further reduces the temperature to something less than 200 F; depending on the type, thickness, and whether or not it is evacuated.

Because it is mandatory, due to the presence of personnel and temperature sensitive equipment, to provide a controlled temperature in the fuselage, the dependence on these parameters is studied in greater detail. The simplest and most reliable insulation is a dry-type system. If the insulation is operating in a vacuum, the thermal conductivity is approximately 1/5 of that at sea-level pressure. Studies of the reentry profile indicate that if the insulation chamber is vented to the outside, the peak temperatures occur at an altitude where the thermal conductivity is high approaching the sea level conductivity value. This makes it desirable to evacuate the insulation chambers.

Evacuation of the outer insulation compartment does not appear feasible because of the extreme temperature gradients of the outer radiating surface. Because of these high gradients, the outer surface heat shield is segmented and the accompanying numerous joints makes sealing impractical.

The above considerations for cabin areas of the vehicle leads to the conclusions that:

Inside wall maximum temperature - 200 F

Maximum primary structure temperature - 1200 F

Maximum outside skin temperature - 2600 F

Insulation chambers should be evacuated

Outside surface should be segmented

From the above conclusions, it appears to be desirable to split the required insulation, placing part of it outside of the primary structure and the balance inside the primary structure. The inboard insulation chamber can be evacuated by producing a sealed primary structure capable of supporting cabin pressure and then constructing an inner pressure vessel that forms the cabin wall. This inner vessel can be idealized for pressure loading only; the outer primary structure supports the flight and landing loads. By producing two pressure vessels one inside of the other, each capable of supporting the cabin pressure, the insulation chamber between these two vessels can be evacuated thus taking advantage of the improved insulation characteristics. This inner insulation chamber can be evacuated prior to reentry by controlled venting to space and then closing the vents for reentry.

A sketch of this type of cabin wall construction is shown in Figure 29. The curves of Figures 30 and 31 show the resulting inner wall temperature as a function of insulation thickness for this type of cabin wall construction.

Meteoroid Encounter

The foregoing paragraphs of this section have delineated the criteria and design philosophy based on the temperature and loading conditions which would normally be encountered on the specified mission. Unusual environmental situations which will develop, such as meteoroid encounter, must also be considered in the design.

Figure 32 shows the probability of at least one meteoroid encounter, as a function of meteoroid diameter. For example, it is noted that an encounter with a meteoroid of 0.12 inch diameter would be anticipated with a 0.88 probability.
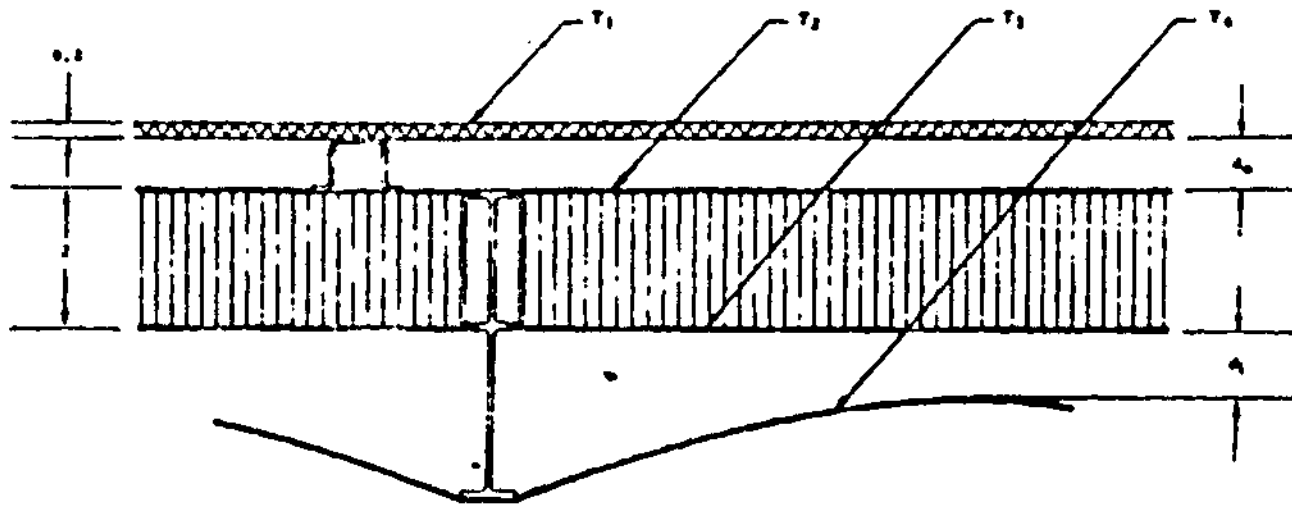
Figure 29. Cabin Wall Construction

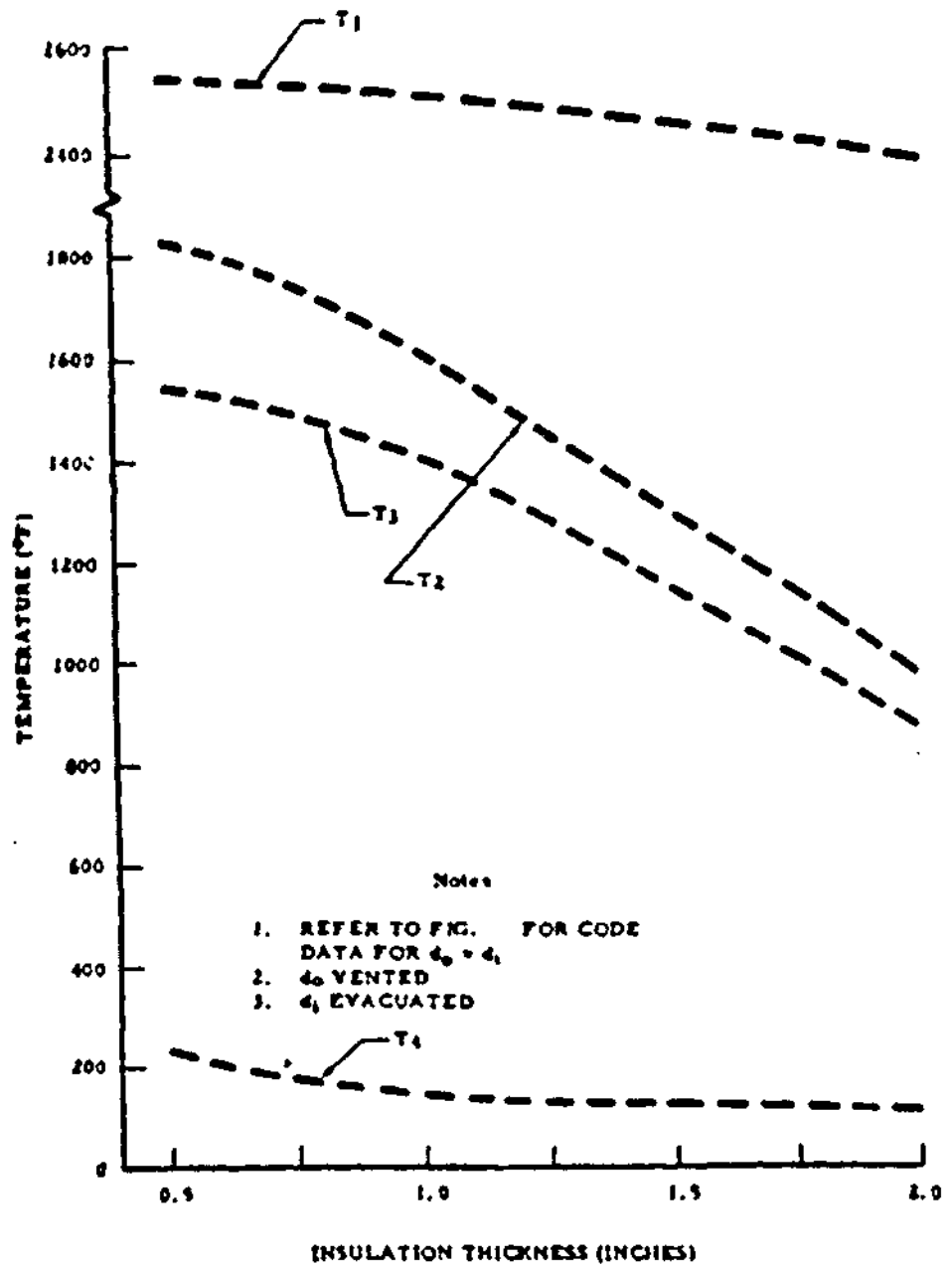Figure 30. Temperature Versus Insulation Thickness,
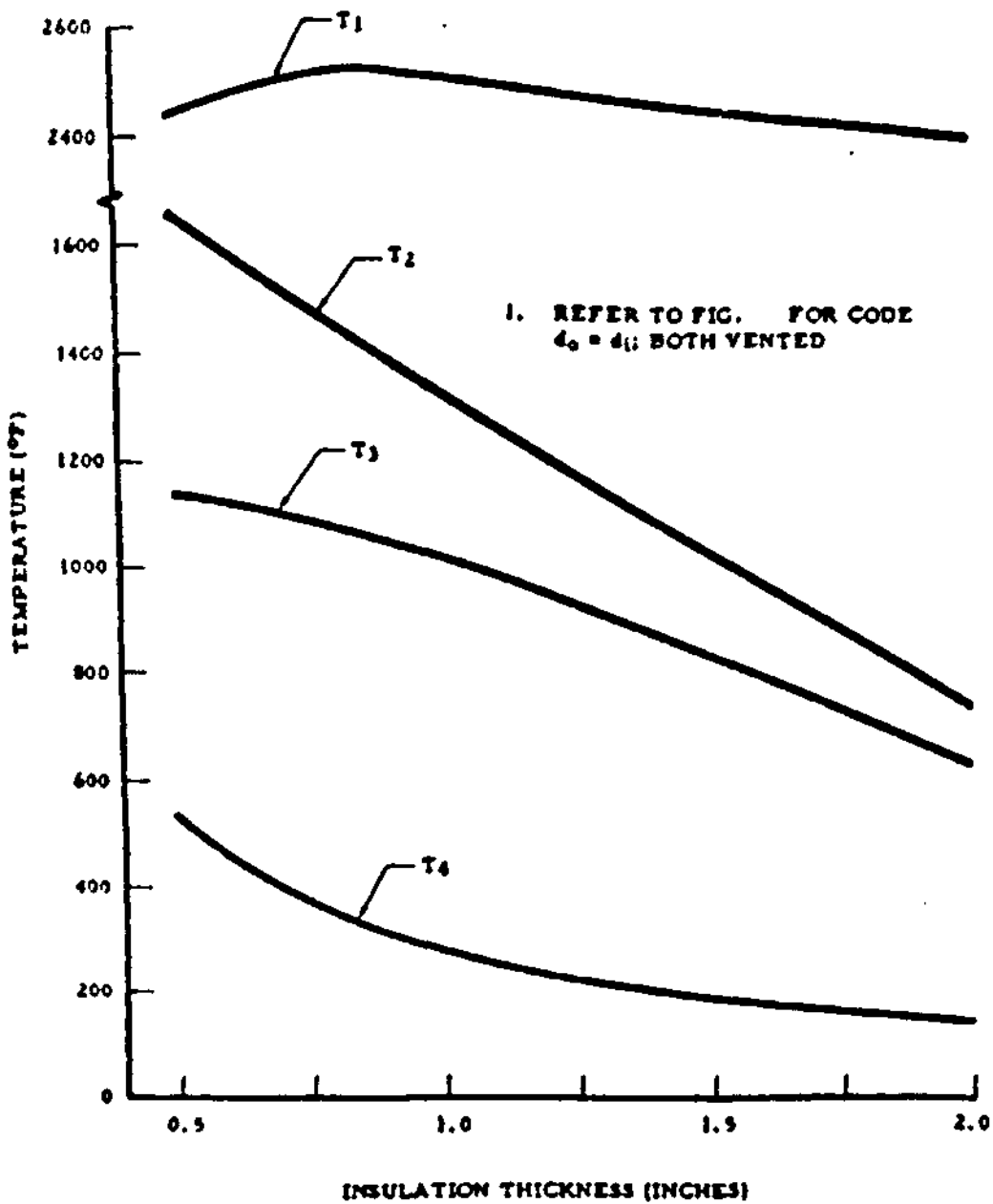Vented and Evacuated Walls

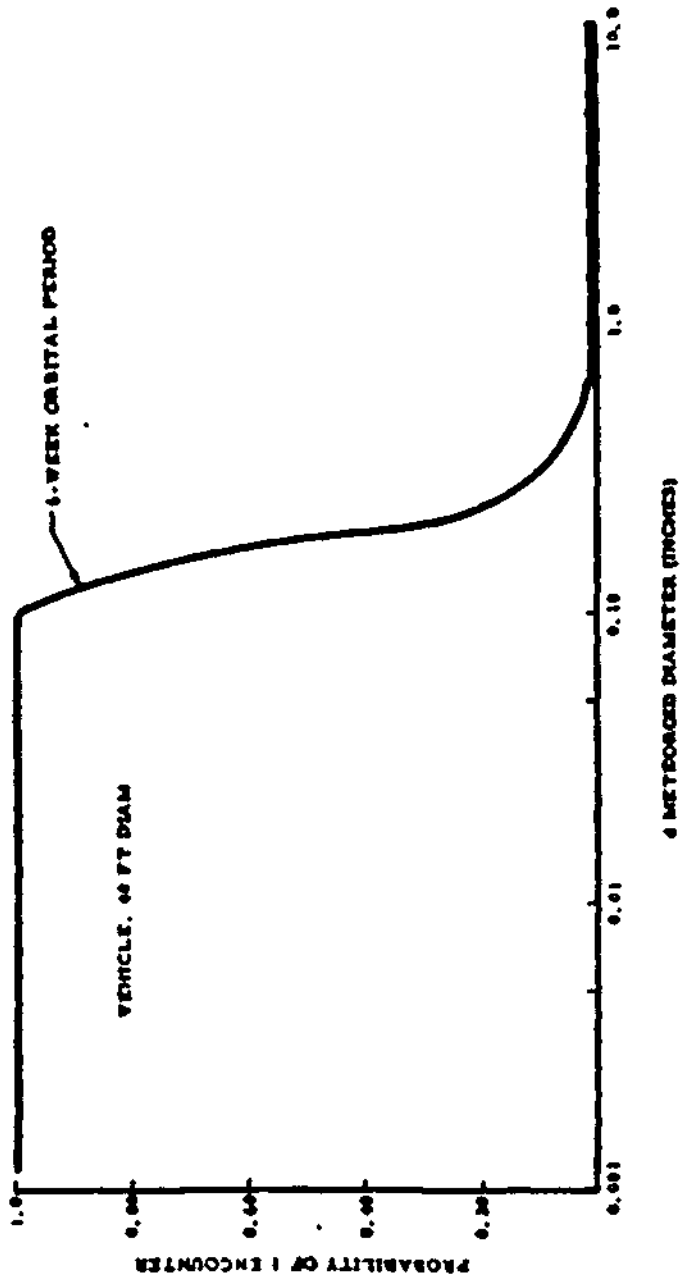Figure 31. Temperature Versus Insulation Thickness, Vented Walls

Figure 32. Probability of Meteoroid Encounter Versus Meteoroid Size

Figure 33 shows the required shield thickness needed to ensure zero puncture at various probability levels. It can be concluded that for a cabin area wall thickness of 0.020 aluminum, which is otherwise a structural requirement, a 0.96 probability of not experiencing a puncture can be achieved.

These two figures are representative of an average integrated area of 1260 square feet, representative of the selected vehicle design, exposed to meteoroidal material for a period of 6 weeks.
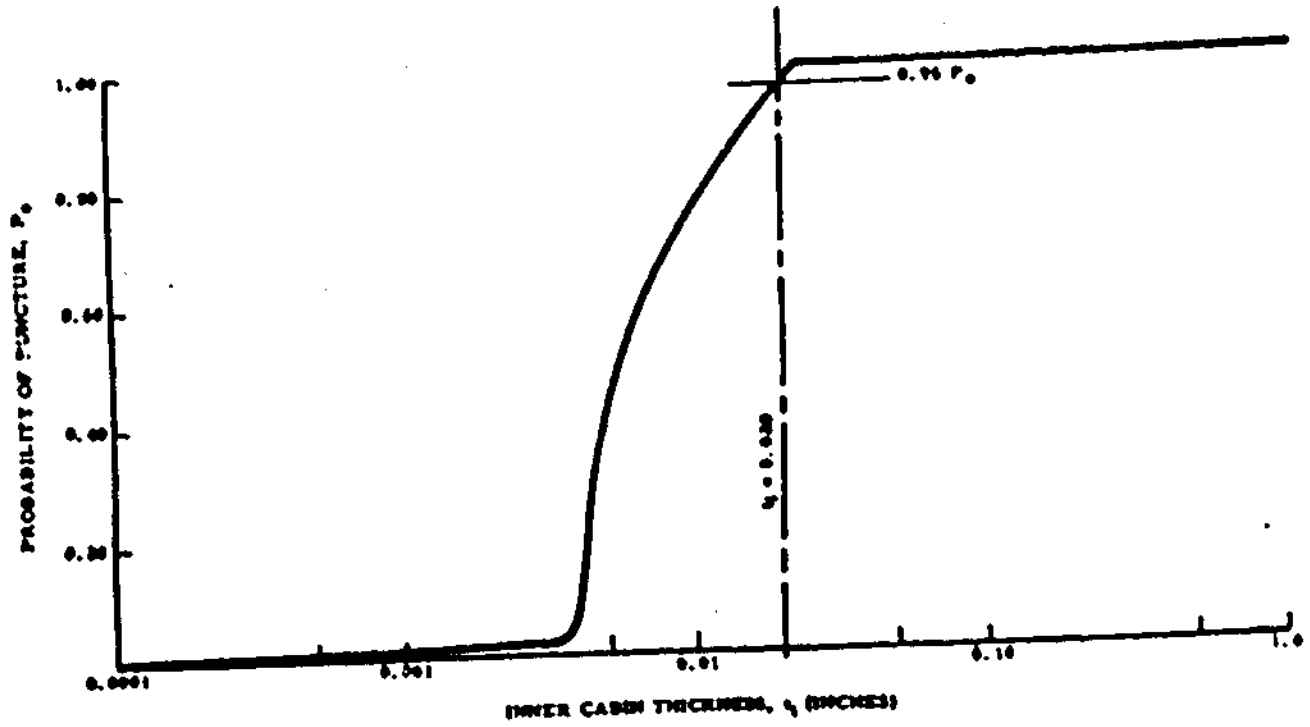
Figure 33. Probability of Survival Versus Required Inner cabin Shell Thickness

REFERENCES

1. General Reference: "NASA Joint Conference on Lifting Manned Hypervelocity and Reentry Vehicles", Parts I and II, April 11-14, 1960.

2. "NASA Congressional Hearings for Fiscal Year 1961", Parts I and II, March and June 1960, respectively.

3. ASD TR-61-240, Part I, Volume II, "Environmental Control Systems Selection for Manned Space Vehicles", December 1961.

4. SID 61-280-I, Section 1, Technical Proposal, "NASA Project Apollo Spacecraft", October 1961.

5. ASD TR-61-20, Volume III, SR 178, "Primary Vehicle Global Surveillance System", May 1961.

6. WADD TR-61-7, Volume I (SR-79821), "Earth Satellite Weapon System", February 1961.

7. ASD TR-61-20, Volume III (SR 178), "Primary Vehicle Global Surveillance System", May 1961.