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ASD-TR-61-240
PART I, VOLUME II

(UNCLASSIFIED TITLE)

Environmental Control Systems Selection for Manned Space Vehicles

Appendix L

Missions, Vehicles, and Equipment

TECHNICAL REPORT NO. ASD-TR-61-240, PT I, VOL II
May 1962

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

Project No. 6146, Task No. 61118

(Prepared under Contract No. AF 33(616)-7635
by Los Angeles Division, North American Aviation, Inc., Los Angeles, California.
Authors: R A. Paselk and A. C. Martin.)

62ASRM-1047-/7

SECRET

NOTICES

FOREWORD

This appendix to "Environmental Control Systems Selection for Manned Space Vehicles" has been separated from Volume I (unclassified). Certain pages of this appendix have been classified because of the possibility of suggesting 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 Abstract is classified SECRET

ABSTRACT

Determination of thermal and atmospheric control requirements necessitate examination of realistic manned vehicles. Three versions of a manned, orbital, reentry, base-point vehicle are developed for the purpose of providing tangible reference points for determination of the thermal and atmospheric control requirements of realistic vehicles. Preliminary concepts of a manned orbital base and a manned lunar vehicle are also outlined. More complete development of the latter two concepts is planned for a later phase of the study.

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

The publication of this report does not constitute approval by the Air Force of the findings or conclusions contained herein. It is published only for the exchange and stimulation of ideas.

FOR THE COMMANDER:

William C. Savage
WILLIAM C. SAVAGE
Chief, Environmental Branch
Flight Accessories Laboratory

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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-off data useful in selecting and sizing thermal and atmospheric control systems.

The reasons for selection of a manned orbital reentry vehicle, manned orbital base, and manned lunar vehicle for representative investigation of environmental requirements for thermal and atmospheric control systems are outlined in Volume I. (See Figure 1.) The mission of the manned orbital reentry vehicle has been further specified as a global surveillance because of the military value of reconnaissance relative to other possible missions and vehicles and because of the likelihood that reconnaissance represents the first employment of manned military space vehicles (Reference 30).

Although it is common practice to develop a few specific vehicles as a practical means of obtaining generalized data, care must be taken to limit not only the number of specific vehicles but also the detail to which these selected vehicles are developed. Thus, the end goal of the present study is to produce realistic data pertinent to the design of thermal and atmospheric control systems for manned space vehicles. Figure 92 is a preliminary check list for possible interactions between vehicles and vehicle environment. This figure indicates that an elaborate mission and vehicle optimization effort is not justified, since many factors which might be optimized have little significance to the study. For example, minimizing the weight of a heat exchanger is of great significance to the study, whereas minimizing the weight of the vehicle itself is significant only insofar as it is related to thermal radiation and insulation, pressure hull integrity, meteoric penetration, etc.

The scope of this portion of the study is further 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 31 and 32).

Manuscript released by the author on 9 May 1961 for publication as an ASD Technical Report.

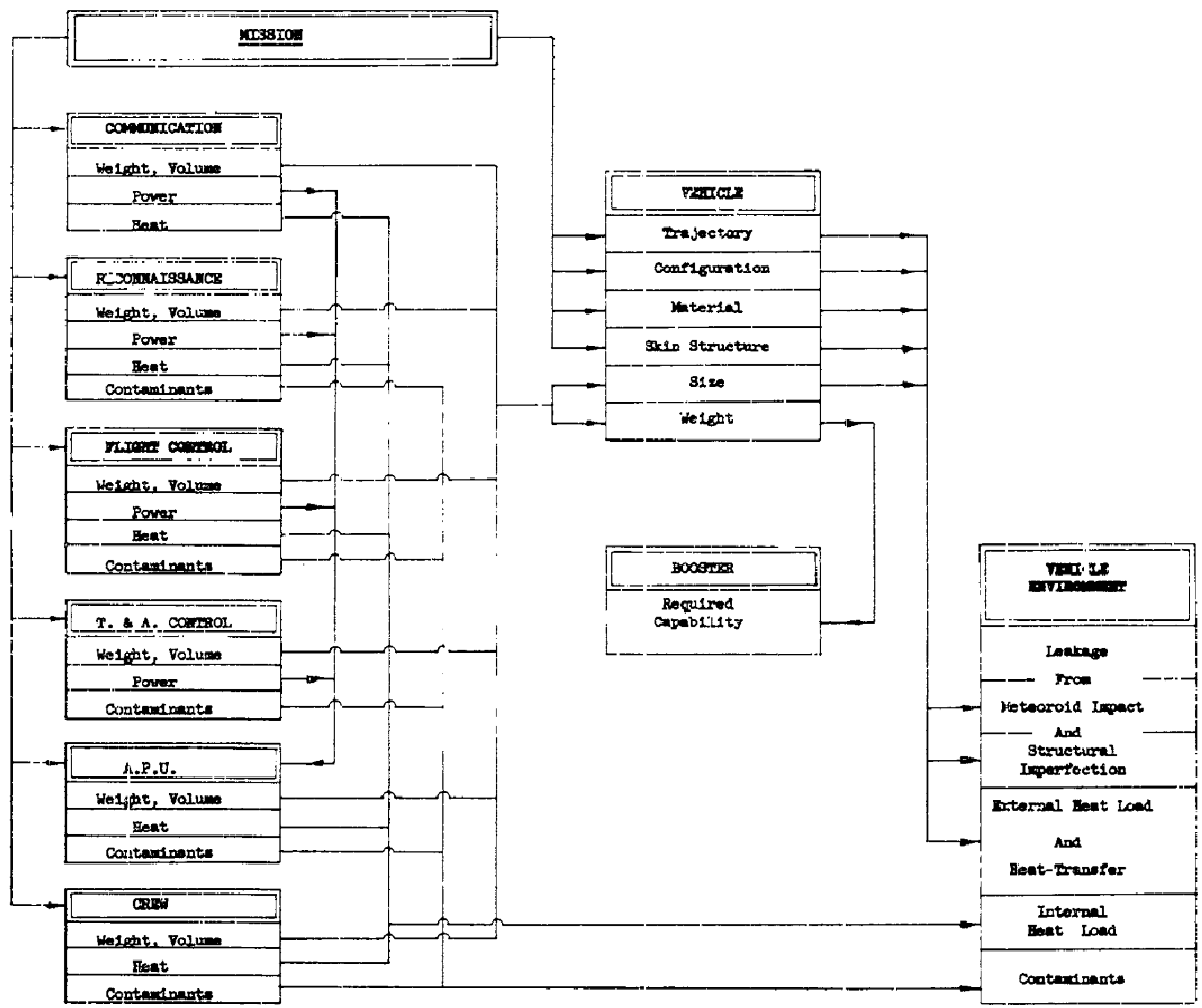

Figure 92. Manned Space Vehicle Environment Flow Chart

With these limitations in mind, this appendix contains the following: (1) detailed development of a global surveillance manned orbital reentry vehicle; (2) development of likely variations of the orbital reentry vehicle configuration in order to establish the influence of crew size, mission equipment, and mission duration on thermal and atmospheric control system requirements; (3) preliminary development of the manned orbital base and lunar landing vehicle; and (4) compilation of general data on the more important aspects of manned space vehicle design (i.e., flight vehicle power, structures, effect of meteoroids, mission equipment, and examination of these general data for environmental requirements).

Table 23 indicates the status of selected vehicles, estimates the earliest availability of the required boosters, and summarizes the environmental factors associated with each vehicle.

Primary emphasis to date has been placed on the manned orbital reentry vehicle whose mission would be global surveillance. 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. Three 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 subclasses 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 1C — Two-man, 12- or 36-hour, partial-surveillance version

All three versions are boosted, winged, orbital configurations with pilot-controlled reentry trajectory similar in concept to the Dyna-Soar. The Dyna-Soar-type configuration was selected because it is believed to be representative of likely future military systems. Vehicles 1A, 1B, and 1C are discussed in detail including analyses of missions, equipment, power load requirements, abort/maneuvering/retro capabilities, reentries, and weights.

TABLE 23

SELECTED VEHICLE TYPES AND ASSOCIATED ENVIRONMENTAL FACTORS

	MANNED ORBITAL REENTRY FOR GLOBAL SURVEILLANCE			MANNED ORBITAL BASE	MANNED LUNAR LANDING VEHICLE
	Vehicle 1-A: 5-man, 6-week full-equipment version	Vehicle 1-B: 2-man, 1-week, full equipment version	Vehicle 1-C: 2-man, 12 or 36 hour, limited equipment version		
Status of selection and development of realistic concept	Initial base point design.	Variation of base point design.	Variation of base-point design.	Selection and preliminary review.	Selection and preliminary review.
T/O weight, required booster; earliest availability.	57,800 lb; Saturn C-2', C-3, or D-1; 1966, 1967 or 1969 (?)	46,400 lb; Saturn C-2; late 1965.	26,000 Saturn C-1; mid-1964	Multiple launch; 1970 (?)	--20,000 lb; 12,000,000 lb. NOVA; refueled Saturn; launch from orbiting base; 1970-75 (?)
On-board heat sources	20KW from equipment including life support for 5 men. Nuclear or solar flight vehicle power unit has separate radiator for non-useful energy.	20 KW from equipment including life support for 2 men. Nuclear or solar flight vehicle power unit has separate radiator for non-useful energy.	6 KW from equipment including life support for 2 men. 1.5 KW non-useful heat from chemical power supply excluding 14.5 KW in exhaust gases.	Power dissipation to be determined. Crew size of 40-60 men being considered. Power supply likely to be an independent nuclear system.	6 KW from equipment including life support for 1 or 2 men.
Solar Influence on Vehicle Radiator	Orbital Inclination: 70° (60°-90°) Altitude: 300 n. mi. (100-400) Eccentricity: 0 (0-0.035)	Same	Inclination: 75°(60°-90°) Altitude: 100 n. mi. (100-400) Eccentricity: 0(0-0.035)	Three altitudes being considered: 300 n. mi. 22,000 n. mi. (radiation)? 100,000 n. mi.	100% sunlight during most of mission; trajectories to be determined.
Meteoric damage to radiators	Requires special design or repair concepts to minimize effect of punctures. Erosion of radiating surfaces probably negligible.	Same	Same	Penetration and erosion time dependent--requires further study.	Requires design attention to possible puncture. Lunar dust may have to be removed.
Reentry Heating	Trajectories determined; Leading edge, skin and cabin temperatures appear reasonable.	Same	Same	Non-reentry, except for emergency and logistic vehicles (attached)	Escape-velocity trajectories to be determined.
Pressure losses	Goal is 5 lb/day loss; requires considerable improvement over current structural practices. Meteoric damage to pressure vessel probably negligible for bumper designs.	Same	Same	Airlocks, external ports, skin penetration, accumulated effects of meteoric puncture are problems for future investigation.	Airlocks, lunar landing shock damage and other sources of leakage are problems for future investigation.
Atmospheric contamination	5 men plus possible equipment contaminants. Contamination rates from sealed off equipment are expected to be low, except perhaps for possible "catastrophic" failure.	2 Men Same	2 Men Same	40-60 men (?) Even extremely low equipment contamination rates represent a cumulative problem.	1 or 2 men. No new problems anticipated.

Vehicle 1A: Five Men, 6 Weeks

Description

As previously noted, Vehicle 1A is a five-man, 6-week version having a complete complement of equipment for global surveillance. The configuration and many of the design details of Vehicle 1A are shown in Figure 93. The basic characteristics of this version are as follows:

Crew size	5
Gross launch weight	57,825 lb
Reentry weight	52,000 lb
Wing area	1325 ft^2
Reentry wing loading	39.2 lb/ft^2
Fuselage overall length (w/o flight vehicle power unit)	86-1/2 ft
Fuselage diameter (maximum inside)	8-1/4 ft
Total volume (separate on-duty, off-duty, and equipment compartments)	1500 ft^3
Equipment and crew heat rejection	20 kw (continuous and nearly steady)
Flight vehicle power	Nuclear or solar turboelectric
Sweep	73 deg
Leading-edge radius	6 in.
Nose radius	12 in.

Although stored heat sinks are indicated during reentry, sustained orbital operation requires space radiators for temperature control. The lower surface of the wing appears to be the most convenient location for the space radiators, as the photographic, infrared, and radar equipment require the vehicle to orbit in the inverted position (considering the landed attitude as normal position). The location of the radar antenna and photographic and IR

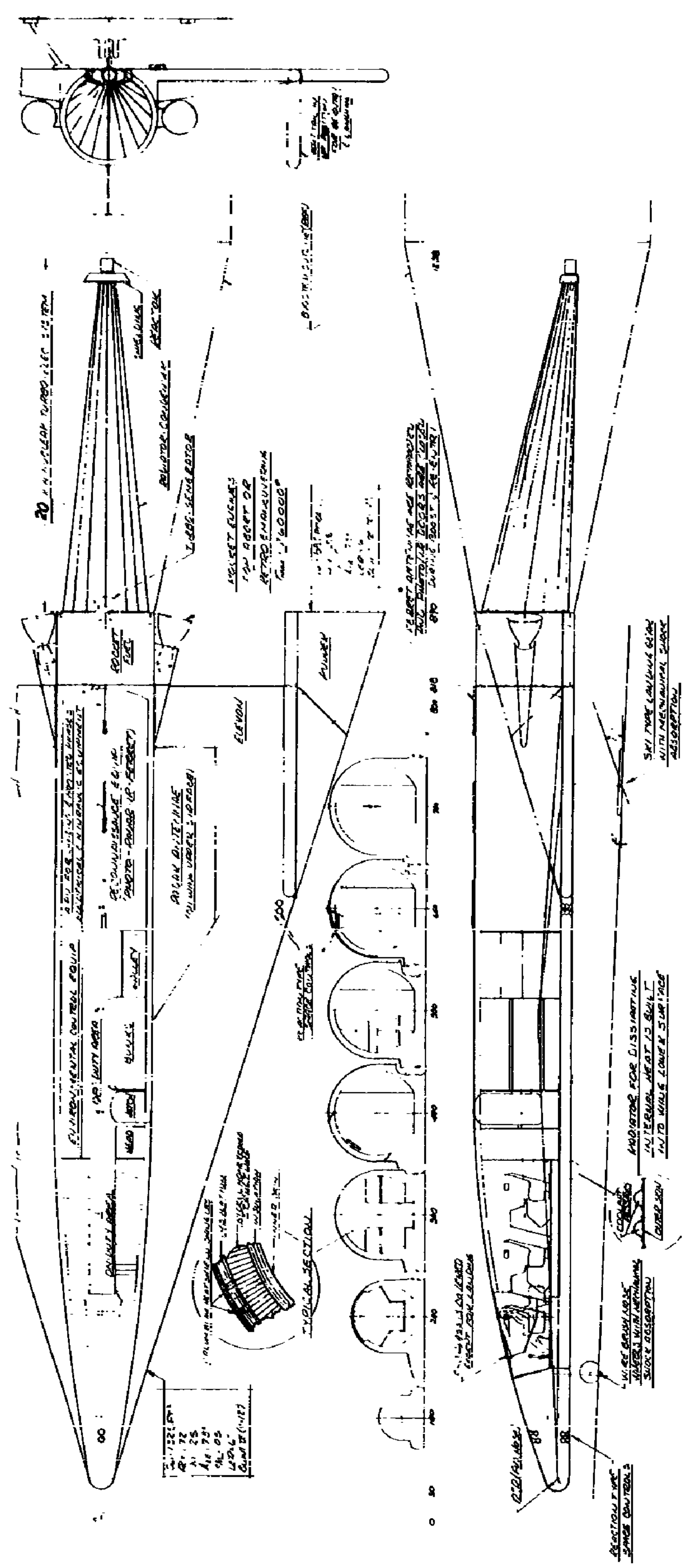

Figure 93. Vehicle 1A

windows on the "top" surfaces of the vehicle is indicated by reentry heating considerations. As presently designed, Vehicle 1A has a separate, high-temperature radiator for the nuclear (or solar) flight vehicle power unit. The power supply radiator located behind the vehicle proper is covered during launch and jettisoned before reentry; accordingly, aluminum fins with steel tube coolant passages are used for this radiator.

Radiator damage by meteoroids is treated later in this appendix. This analysis (Figure 111) indicates that the pressure side of the wing, made of 0.035-inch-thick columbium, will provide an 82-percent probability of no penetration of the radiator coolant passages during a 6-week mission. This probability can be increased to about 99 percent by trebling the thickness of the columbium or, more economically, by providing a larger sectionalized radiator. Loss of one set of coolant passages will have a smaller effect than might at first be supposed, as the inoperative areas will serve as increased fin area for adjacent operating passages. It should be noted that the estimated effect of meteoroids is based upon relatively meager data with respect to both impingement rate and effect. Furthermore, design difficulties in installing the radiator tubing inboard on the wing skin may require installation of, say, aluminum tubing on the outside surface of the columbium wing.

The cabin wall has particular significance to thermal and atmospheric control. The construction of the wall is indicated in a detailed blowup of a typical section in Figure 93. This construction is basically similar to cases IV and V considered later in the general section on structures. The outermost portion of the wall is a columbium heat shield. This heat shield is mounted on, and insulated from, a nickel honeycomb sandwich structure. Additional insulation is placed between the sandwich structure and the cabin's inner surface, which must not exceed 130°F during reentry unless ventilated suits are to be used (Appendix C).

Both the inner surface of the cabin wall and the inner surface of the sandwich structure are pressure hulls independently capable of withstanding a pressure differential of 20 psi (ultimate). Not shown in Figure 93 are the frame supports between the two pressure hulls. Principal potential sources of leakage are (1) imperfect structure, and (2) puncture of the pressure hull by meteoroids.

The practical leakage of cabin atmosphere due to imperfect structure must be held to something like 5 lb/day for missions lasting for 6 weeks. The magnitude of the problem may be judged from the present performance of the X-15 cabin, which leaks 1440 lb/day at 3.5 psi. Poor hatch sealing is responsible for a loss of 540 lb/day. It should be noted here that the X-15 and some concepts of Dyna-Soar can tolerate considerable leakage because of the short duration of the design mission. The design leakage rate of the Mercury capsule indicates that a loss of 5 lb/day at 15 psi should be achievable (Appendix A).

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vehicle, if taken in two increments, might permit a change from a low circular orbit to a somewhat higher circular orbit. Similarly, the retro capability might be used to lower the altitude in increments prior to reentry (with additional fuel required). The eccentricity will normally approximate zero to alleviate stabilization and image motion compensation problems. However, an orbit with a 100-mile perigee and a 400-mile apogee over the northernmost latitudes would require less launch and retro energy than a 400-mile circular orbit and would increase slightly the time over the USSR and China. An inclination near 63-1/2 degrees would keep apogee in the same latitude, since the major axis does not rotate at this inclination.

As far as environmental factors are concerned, the Van Allen radiation belt imposes constraints, especially in view of Explorer VII evidence, for relationship between the radiation belt and auroral activity. Otherwise, changes in the orbit appear to affect only the relative orientation of the vehicle radiator relative to the sun, the time in sunlight, and terrestrial reflection and radiation.

Accordingly, no elaborate orbital analysis has been made, but only the range of orbits is indicated. A typical orbit is considered to be at an altitude of 300 nautical miles, circular, and at an inclination of 70 degrees. Typical variations are inclinations of from 60 to 90 degrees, altitudes from 100 to 400 nautical miles (either circular or eccentric), and wintertime and summertime missions.

An optimum crew size cannot be selected at present. Crews of from 2 to 5 men have been studied. A wide variety of crew sizes and work cycles appears reasonable. In addition to obvious demands upon the environmental control system, the crew size has a bearing on the type and amount of equipment and the degree of automation. The work cycle affects equipment utilization and heat rejection by both the equipment and crew. For the sake of being specific, a 24-hour day, with three men on duty during the busiest 8- to 10-hour period, has tentatively been selected as the work cycle for this hypothetical mission. Although total sleeping time would not be reduced, it is assumed here that it would be more efficient, and psychologically better to work more than 1/3 of the time in orbit, considering that no time is consumed in going to and from duty stations and that the crew would be well occupied with intelligence processing as well as intelligence collection.

Reconnaissance Equipment

The design intention is to provide a full complement of equipment for global surveillance. The following reconnaissance equipment is included in Vehicles 1A and 1B: (1) side-looking radar, ELINT, infrared scanner, high-resolution cameras, low-resolution cameras, photographic processing equipment, and data processing equipment. Provision is also made for communications, guidance, and sensing equipment (Table 24).

In general, electronic equipment operates well in the temperature range from 0°F to 160°F, except that transistors lose reliability above 100°F. Proper cooling of transistors, diodes, resistors, and capacitors may become a large problem for densely packaged electronics, as equipment becomes increasingly more miniaturized. The thermal aspects of electronic equipment are treated more thoroughly in the general equipment section later in this appendix.

Photographic surveillance is effected by means of a 48-inch focal length for mapping and a 96-inch focal length for detailed investigation (7 feet static and perhaps 40 feet practical resolution at 300 nautical miles). If we assume a 9-inch film format along the direction of travel, then an allotment of 300 negatives per day would allow an average of 15 minutes of 100 percent overlap mapping per day. The detail camera has been allotted 200 9 x 9-inch negatives per day or 50 18 x 18-inch negatives per day. For high-resolution reconnaissance information, on the order of 5 feet, the temperature variation must be maintained to less than approximately 1°F. The temperature gradient is critical and temperature variations across the lens must be held to a fraction of a degree (Reference 34).

The temperature regulation for infrared equipment must be maintained within approximately 1 degree.

The amount of film or other recording media required for the electronic equipment has not been determined, although it is estimated that it will be comparable to the film used for photographic surveillance. It might also be well to investigate reusable film for possible savings in weight and for the consequent change in the environment required for storing, using, and processing such film.

The general environmental control aspects of electronic and photographic equipment are considered later in this appendix.

Mission Power Requirements and Flight Vehicle Power

A preliminary operations analysis indicates the following equipment power requirements, computed on the basis of the peak and standby power values found in Table 24.

TABLE 24
RECONNAISSANCE, COMMUNICATIONS, AND GUIDANCE EQUIPMENT

	Weight (Lbs.)	Volume (Ft.^{3})	Power (KW) Peak	Stby.	Avg.	Possible Contaminants*	Temperature Limits (°F)	Performance Criteria	Display Area Ft^2
Side-Looking Radar	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	Insulation breakdown due to high heat loads. Ionization from arcing. Dielectric leakage from waveguides	Equip: [illegible] to [illegible] Film: [illegible] to 120	Accepts $10^{?}$ bits/day Transmits: 10^7 bits/day Stores: $10^{?}$ bits/day at $10^{?}$ bits/lb.	2
ELINT	[illegible]	37 (inc. antennas)	8.[illegible]	[illegible]	6.[illegible]		Equip: 0° to 160° Film: 0° to 120°	Accepts 10^7 bits/day Transmits: 10^3 bits/day Stores: 10^4 bits/day	2
INFRA-RED	250	[illegible]2	[illegible]	[illegible]	[illegible]	Insulation breakdown in electronic circuits.	Equip: 0° to 160° Film: 0° to 120°	Accepts: $10^{1?}$ bits/day Transmits: 10^7 bits/day Stores: $10^{?}$ bits/day at 10^9 bits/day	2
Photo Surveil — Cameras	1000	[illegible]	[illegible].3	[illegible]		Variety of processing solutions-uncertainty as to long term toxicity will dictate the use of sealed chambers.	Equip: 0° to 160° Lens gradient may be critical Film: 0° to 120° Process: ±1°F of nominal (approx 90°) Solutions: ±1°F of nominal (50 to 70°F) (camera)	Accepts 4 x $10^{?}$ bits/day or 400 negatives per day (9" x 18") Transmits: 10^8 bits/day Recall Storage: 10^{10} bits/day Other Storage: 4 x 10^{11} bits/day at 2 x 10^{10} bits/lb.	2 (Shared)
Photo Surveil — Processor	200	2[illegible]	0.3	0.2					
Photo Surveil — Film	700	1[illegible]	-	-	0.3				
Photo Surveil — Solutions	700	1[illegible]	-	-					
Data Processing	[illegible]000	3[illegible]	4.0	[illegible]	3.7	Component and insulation breakdown. Film and tape	Equip: 0° to 160° Transistors: 5 cu. ft., [illegible] KW (lose reliability above 100°F)	Inputs from above; part of output to communications	8
Communications — Transmitter	150	2.4	0.5	0.05	0.1	Insulation breakdown (Transmitters) Ionization (arcing) Waveguide leakage	Equip: 0° to 200° Transistors: 0° to 100°F	Transmits: 10^8 bits/day or 250 watts peak Receives: 10^6 bits/day	Minor
Communications — Receiver	30	0.5	[illegible]	[illegible]	0.[illegible]				
Guidance & Sensors — Stellar Monitored Inertial Platform	40	0.8	0.3	-	[illegible].3	Component and insulation breakdown.	Inertial Platform: Independently controlled, 100° max	Attitude drift rate 0° .0[illegible] degrees per hour without stellar monitoring; 15 secs of arc with stellar monitoring.	1
Guidance & Sensors — Guidance computers	100	1.0	[illegible]	0.2	[illegible]		Computer: 0° to 160° (mostly transistors)	Computer has 10,000 words storage and 500 kc synchronization frequency	
Guidance & Sensors — Horizon seeker	15	0.3	0.015	0.015	0.015		Seeker: 0° to 160° IR Detector: 50°	Horizon seeker pointing accuracy is about 2 min. of arc, rms.	
TOTALS	6400 (8000)[a]	200[b]	17.[illegible] (14.[illegible])[c]		1[illegible][d]	* also: Misc. Sealants, plastics, lubricants, refrigerants (when used)			

Notes: (a) This weight includes allowance for mounting and for installation.
(b) This volume does not include provisions for adaptation to cooling systems.
(c) This figure represents actual system peak power resulting from duty cycle.
(d) This figure represents the average for a typical day; since (c) and (d) are nearly equal, additional power storage is not required.

TABLE 24 (CONT)

		Window Area Ft^2	Manual Attention	Maintenance Concept	Special Requirements	Range Vs. Weight and Power
Side-Looking Radar		(Pair of long antennas)	Continuous, at least one man, shared with other sensors.	Replacement of items with inherently low mean time to failure (Transmitter tubes, rectifiers). Otherwise replacement of small modules.	Large antennas externally mounted. Attitude stabilization to close tolerances (fractions of beam width). Low illumination near displays.	Increases exponentially with range and resolution.
ELINT		Many directional antennas.	Mostly automatic. Attention needed when unusual signals are received.	Replacement of faulty modules.	Directional external antennas, size dependent on wavelengths covered.	Depends upon number of frequency bands to be covered.
INFRA-RED		4	Continuous, at least one man.	Low wear bearings for rotating parts. Replacement of sensors and electronics (modules).	Probably need rotating scanners. Prefer open window to be pointed in desired direction. Cryostat for supercooled sensors. Some sensors subject to radiation damage, small shields probably necessary.	Passive and generally low power consumption. Weight and size increase with focal length and aperture.
Photo Surveil	Cameras - Processor - Film - Solutions	5 (Minimum for cameras)	Automatic, infrequent inspection.	Access to cameras necessary. Test and replace electronic modules. Spare units for processor.	Separate environment for cameras preferred. Cameras are mounted to vehicle, which must be stabilized to $5\text{-}10^{-5}$ radians/sec. Film must be protected from radiation, humidity, temperature during storage and usage. Provision for viewing and analyzing negatives (wet). Operating pressure for processor must be high enough to prevent boiling of 90°F solution.	Weight of camera increases roughly with cube of focal length. Also increases with degree of automation.
Data Processing		None			Protection of film against radiation, humidity, etc.	
Communications	Transmitter - Receiver	None (Antennas)	Very little	Complete set of spare units.	Transmitter and receiving antennas directional and controllable.	Transmitter weight approximately proportional to data rate and range squared, and inversely proportional to antenna area. Receiver weight approximately proportional to data rate.
Guidance & Sensors	Stellar Monitored Inertial Platform - Guidance computers - Horizon seeker	Sperhical coverage for star tracker. -- Horizon seeker window	Periodic manual data inputs to computer. Occasional inspection of trackers.	Redundant equipment plus simplified alternates. Periodic tests of hardy components. -- Computer-self-testing circuits, credibility indication, fault location, replacement modules.	Inertial platform mounted accurately on structure. Cooling and heating provisions for sealed chambers. Computer requires careful cooling of transistors and accurately regulated secondary power. One type of horizon seeker scans entire horizon, while another scans in discrete planes. Window for star trackers usually a cone with a 90° apex.	Platform will vary from 30 to 200 lbs. Weight may vary inversely with the monitored drift rate, with a minimum of 10 lbs. for the star tracker. -- Computer weight may vary from 50 to 800 lbs., depending upon packaging.

Intelligence gathering -- 14.1 kw

On: Radar, IR, ELINT, photo, guidance and attitude sensing, communications receiver

Standby: Data processing, transmitter

Nighttime operation would be perhaps 0.3 kw less with photo off.

Intelligence processing -- 13.3 kw

On: Data processing, guidance and attitude sensing, partial ELINT, communications receiver

Standby: Radar, IR, photo, transmitter

Intelligence dissemination -- 13.8 kw

Adds to intelligence processing requirements 0.5 kw for transmitter

Since the power requirements are nearly equal for the foregoing three functions, it would probably not be worthwhile to provide a power storage capability for the purpose of coping with unusual demands upon the power generator or the environmental control system. A more thorough operations analysis might indicate a greater variation of peak loads, in which case the duration of each requirement would be important. A preliminary analysis of a 300-mile, 70-degree inclination orbit indicates that 20 percent of the time is spent collecting data, 70 percent of the time is spent processing only, and 10 percent of the time is spent processing and disseminating. In order to illustrate the thermal and atmospheric control problems which might arise if the power dissipation duty cycle involved larger fluctuations, large fluctuations might be arbitrarily assumed even though they are not indicated by the operations analysis.

The total power requirement during the orbital phase of this specific mission is as follows:

Peak equipment requirement:	14.1 kw
Crew requirements (100-200 watts/man):	1.0 kw
Refrigeration:	5.0 kw
	20.1 kw

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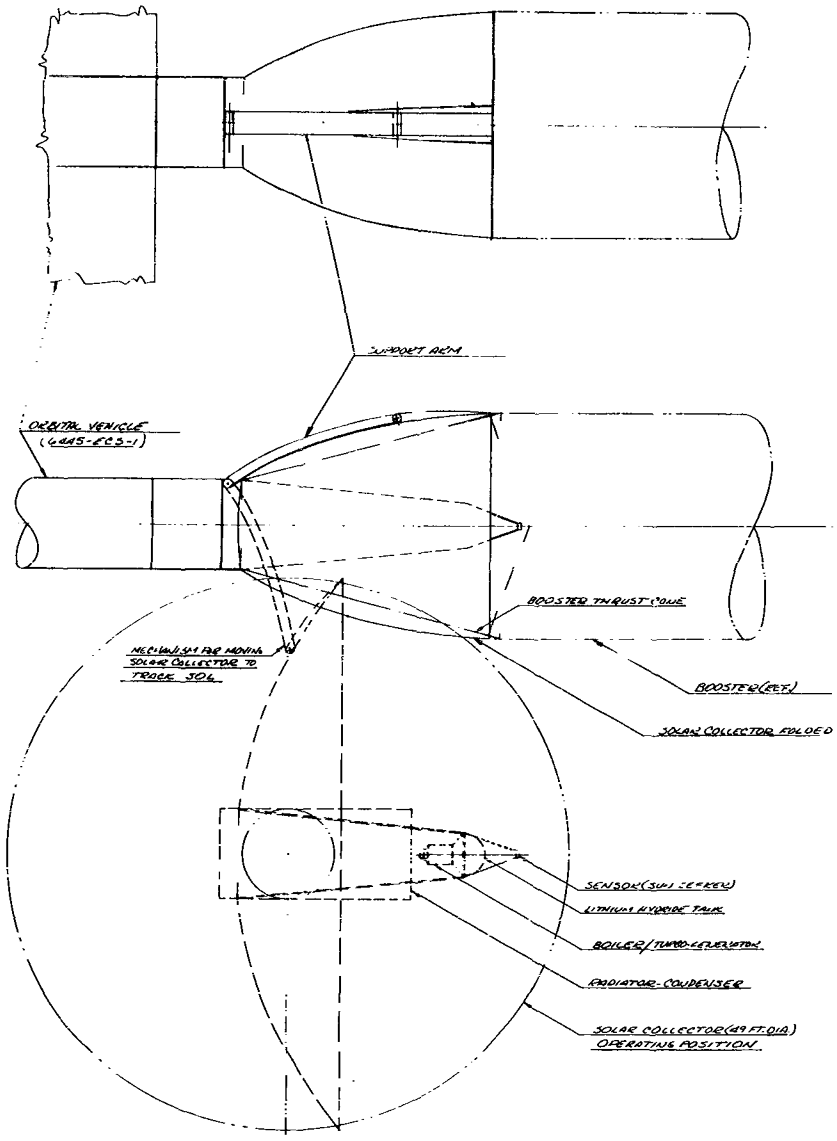

Figure 94. **Alternate Flight Vehicle Power Systems for Vehicle 1A**

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Hardware weight	60 lb	
Fuel	220	
Tank	---	(charged to on-board propulsion)
Misc	50	
	330 lb	

TABLE 25

REENTRY POWER REQUIREMENTS FOR VEHICLE 1A

Function	Duration (sec)	Power (kw)	Energy (kw-hr)
Comm & nav) Environmental control) Refrigeration for above)	0-5150	2.7	3.8
Aerodynamic control	700-5100	40 (normal peaks)	24.3
		74 (unusual peaks)	22.6
		11 (minimum)	3.5
Approach and landing	5100-5150	58	0.8
Total			55.0

If the solar power unit is used for orbital power, then the LiH heat storage power supply can be used to increase further the reliability of essential power supply during reentry. Also, the total reentry power demands on the chemical unit might be reduced by perhaps 20 percent. However, this arrangement would require housing the LiH unit within the reentry part of the vehicle and would increase the reentry weight by perhaps 1000 pounds.

The auxiliary power requirements during launch can be quite high due to large aerodynamic control forces. However, it is assumed that the power is available from the booster during this period. Launch phase power for the environmental control system, flight instruments, and other systems which are essential immediately after launch can be supplied by the reentry phase flight vehicle power unit.

Reentry

Reentry is initiated by a velocity decrement of 500 ft/sec. Before retro firing, the vehicle is traveling about 24,900 ft/sec at 300 nautical miles. After retro thrust, the vehicle travels another 2800 nautical miles (47.3 degrees range angle) in about 700 seconds, at which time aerodynamic forces become appreciable. The initial conditions for atmospheric reentry are velocity, 26,150 ft/sec; angle to the horizontal, 1.75 degrees; and altitude, 300,000 ft.

Two trajectories have been selected for the analysis of the external heat load and heat transfer. Each of these trajectories has been computed for wing loading (W/S) of 30 and 40 lb/ft^2.

The first trajectory assumes constant maximum lift coefficient, $C_L = 0.7$, at a constant angle of attack of 51 degrees. Reentry time and distance are relatively short at the expense of rather high skin temperatures. (See Figures 95 and 96.) Preliminary single-point skin and leading-edge temperatures for the 40 lb/ft^2 wing loading trajectory are only about 2 to 3 percent higher than those corresponding to a wing loading of 30.

The second trajectory assumes a constant maximum lift coefficient, $C_L = 0.7$, until the temperature has passed its first maximum. At that time, the trajectory assumes a $C_L = 0.15$ at a constant angle of attack of 14 degrees, corresponding to maximum L/D. (See Figures 97 and 98.) This reentry operation will keep skin temperatures low at the expense of a longer duration reentry and rather high leading-edge temperatures of about 4100°F. Recent jet plasma tests indicate that a combination beryllium oxide-graphite leading edge will endure such temperature without distortion through ablation or rupture because of thermal strain.

The accelerations encountered at speeds greater than about Mach 1 are low — the maximum drag acceleration being about 1 g, and the maximum accelerations normal to the flight path being of the order of 0.8 g.

Using the two trajectories corresponding to a wing loading of 30 lb/ft^2, an approximate integration of the velocity time history of the $C_L = 0.7$ trajectory yields a range of about 5000 nautical miles for the aerodynamic portion, while the combination $C_L = 0.7$ and $C_L = 0.15$ trajectory yields a range of about 9800 nautical miles. If the vehicle is kept at maximum L/D from the start, it is likely to skip out of the atmosphere. The indicated range maneuverability of 4800 nautical miles along the flight path can be further increased by using negative lift at appropriate times (References 35, 36, 37).

Weight Summary

Table 26 presents a preliminary weight summary which should be regarded as a weight allotment, rather than a refined estimate.

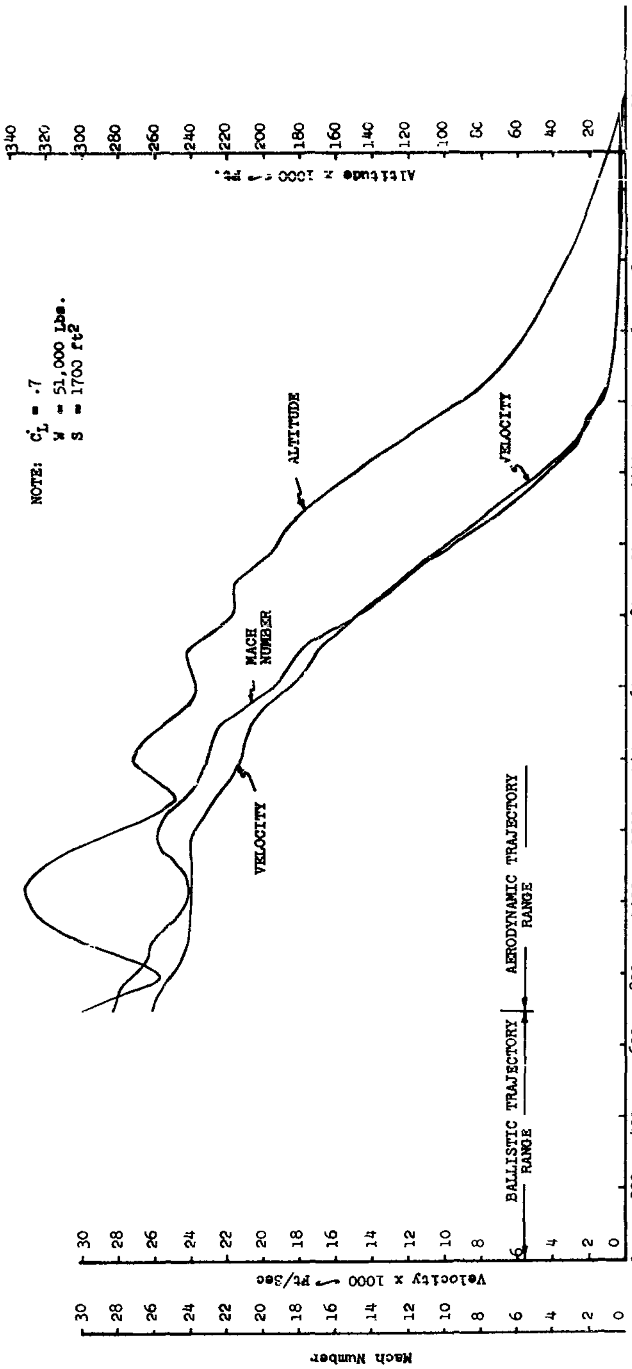

Figure 95. Reentry Trajectory of the Manned Orbital Reentry Vehicle (C_L = 0.7; W/S = 30)

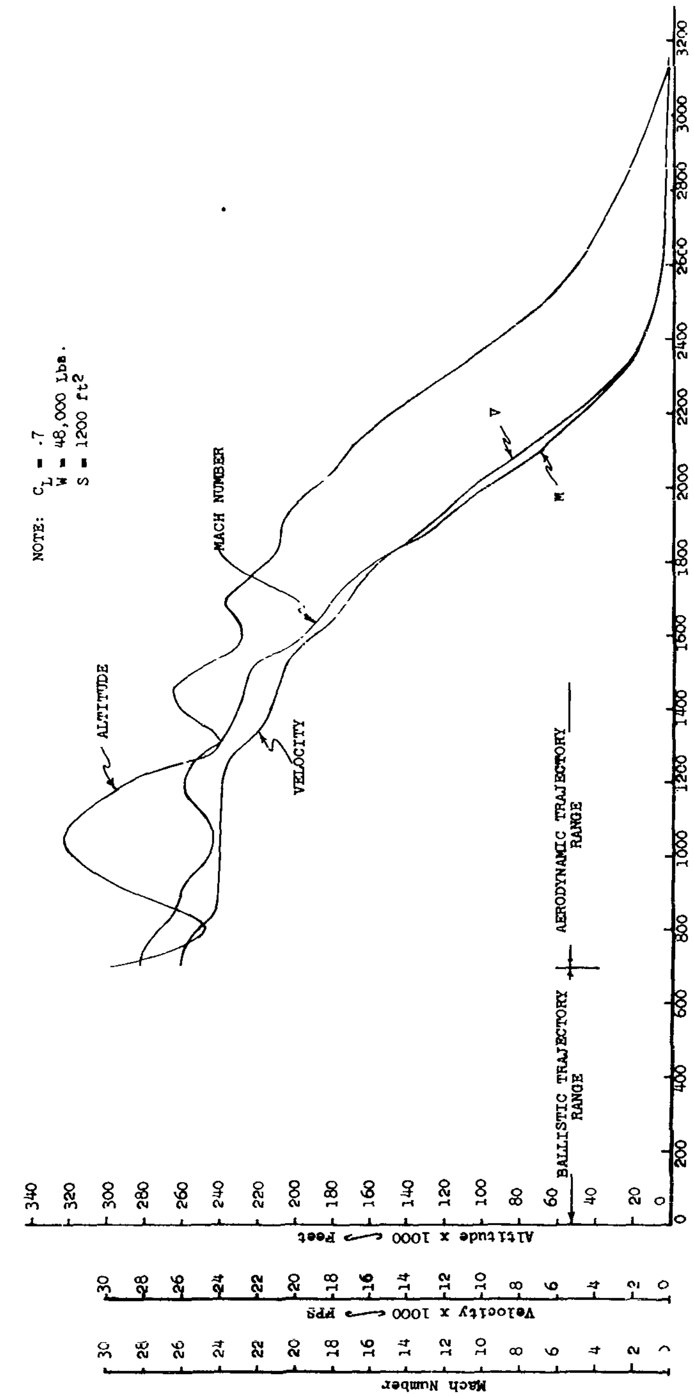

Figure 96. Reentry Trajectory of the Manned Orbital Reentry Vehicle

(C_L = 0.7; W/S = 40)

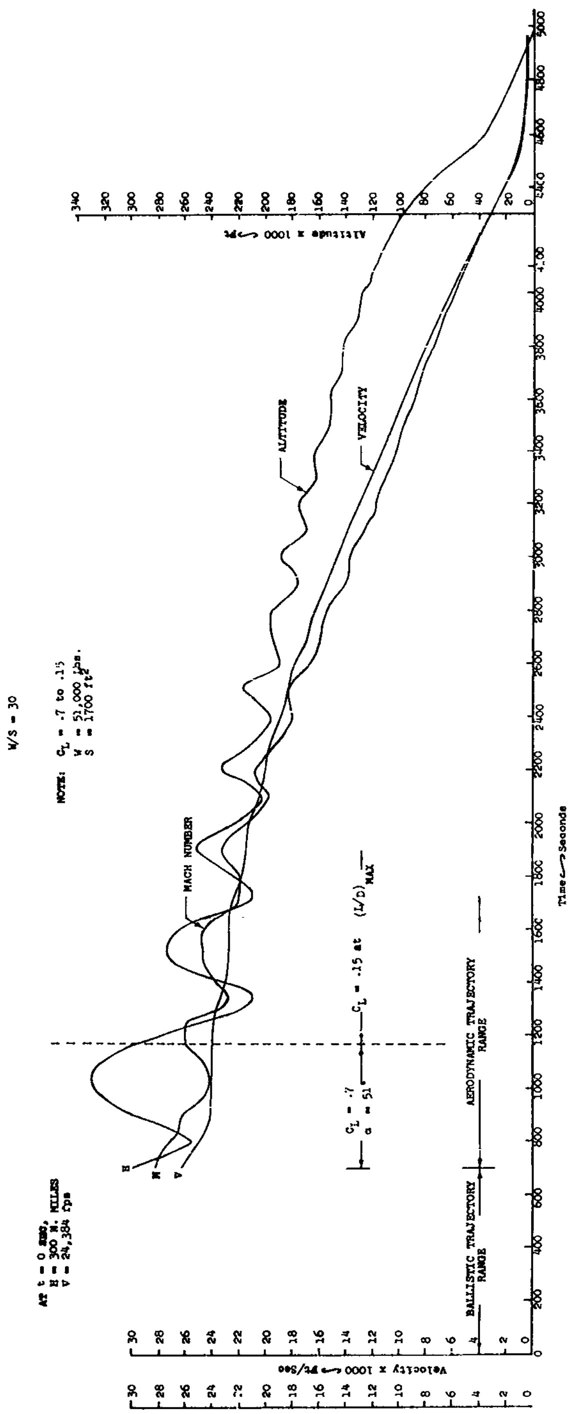

Figure 97. Reentry Trajectory of the Manned Orbital Reentry Vehicle (C_L = 0.7 and 0.15; W/S = 30)

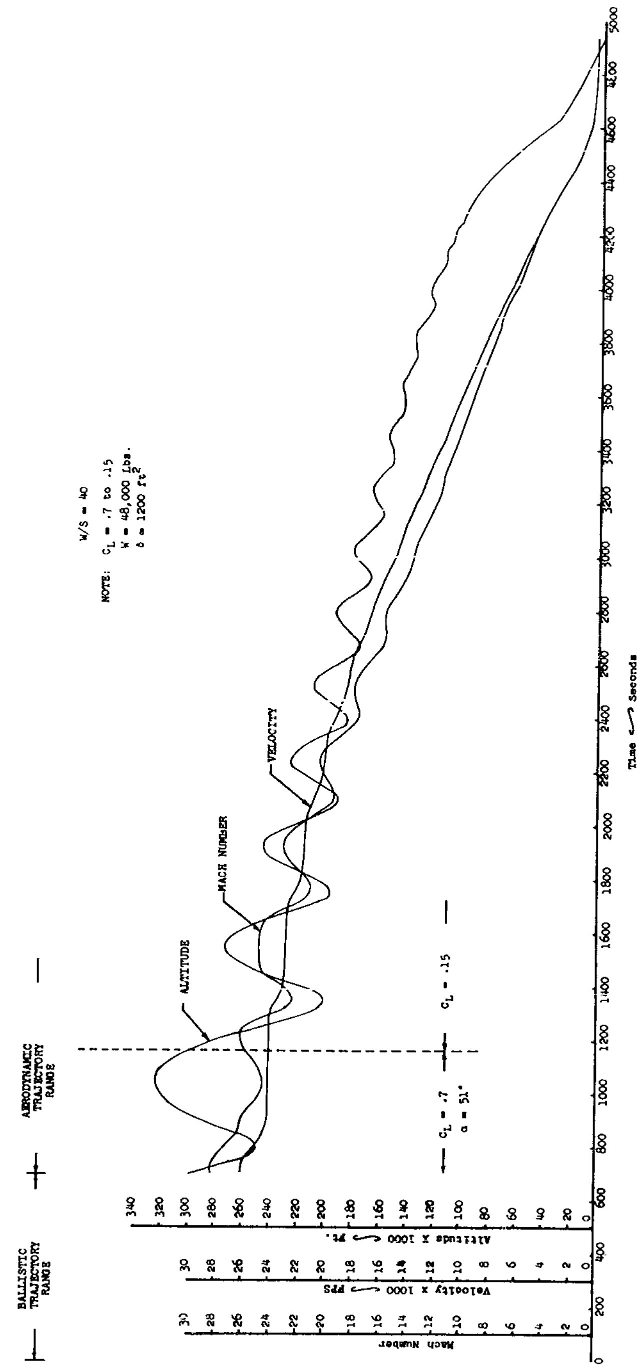

Figure 98. Reentry Trajectory of the Manned Orbital Reentry Vehicle (C_L = 0.7 and 0.15; W/S = 40)

TABLE 26

WEIGHT SUMMARY FOR VEHICLE 1A

Structure:		26,300 lb
Wing	11,500	
Vertical tail	2,700	
Fuselage	7,950	
Landing gear	1,500	
Engine section	150	
Surface controls	2,500	
Power plant:		6,380 lb
Rockets	2,750	
Rocket controls	100	
Fuel system	[illegible],000	
Flight vehicle power System (nuclear)	2,200	
Reentry APU	330	
Fixed equipment:		16,750 lb
Instruments	500	
Hydraulics	500	
Electrical	750	
Electronics (6400 plus structure)	8,000	
Furnishings	1,750	
T & A system	4,550	
Food and H_2O		
Data return capsules	600	
Auxiliary gear	100	
Weight empty		49,430 lb
Crew, fuel, and other:		8,395 lb
Crew (5 men)	1,500	
Fuel (abort/maneuver & retro)	5,470	
Trapped fuel	400	
Fuel (attitude control)	500	
Oil	25	
Miscellaneous	500	
Launch gross weight		57,825 lb
Orbital weight (2770 for maneuver)		55,050 (if all maneuvering fuel used)
Reentry weight (2700 for retro, 350 for attitude control)		52,000 lb

Vehicle 1B: Two men, One Week

The size of the basic vehicle and the duration of the mission have been varied in order to illustrate possible crossover points for changes in flight vehicle power systems, environmental control systems, and other subsystems. The different versions have purposely been kept similar in configuration in order to facilitate comparison of any critical parameters. Accordingly, in what follows only the differences between the smaller vehicles and the basic five-man, 6-week version will be mentioned.

Description

The configuration of the two-man, 1-week version of the global surveillance vehicle is shown in Figure 99. A full complement of reconnaissance equipment is included. Characteristics of this version are as follows:

Crew size	2
Gross launch weight	46,430 lb
Reentry weight	42,000 lb
Wing area	915 ft
Reentry wing loading	46 lb/ft^2
Fuselage length (W/O flight vehicle power unit)	61-3/4 ft
Fuselage width (maximum inside)	8-1/4 ft
Fuselage height (maximum inside)	7-1/2 ft
Total volume (separable crew and equipment compartments)	1200 ft^3
Equipment and crew heat rejection	20 kw (continuous and nearly steady)
Flight vehicle power	Nuclear or solar turbo-electric

In the two-man version, there are only two separable compartments:

The flight deck and the equipment area

An advanced version of Saturn is required to boost 46,000 pounds into orbit. Such a weight could be handled by the C-2 configuration, the first copy of which is planned for late 1965 (Reference 33).

Orbital Mission Analysis and Selected Orbit

Orbital considerations remain unchanged.

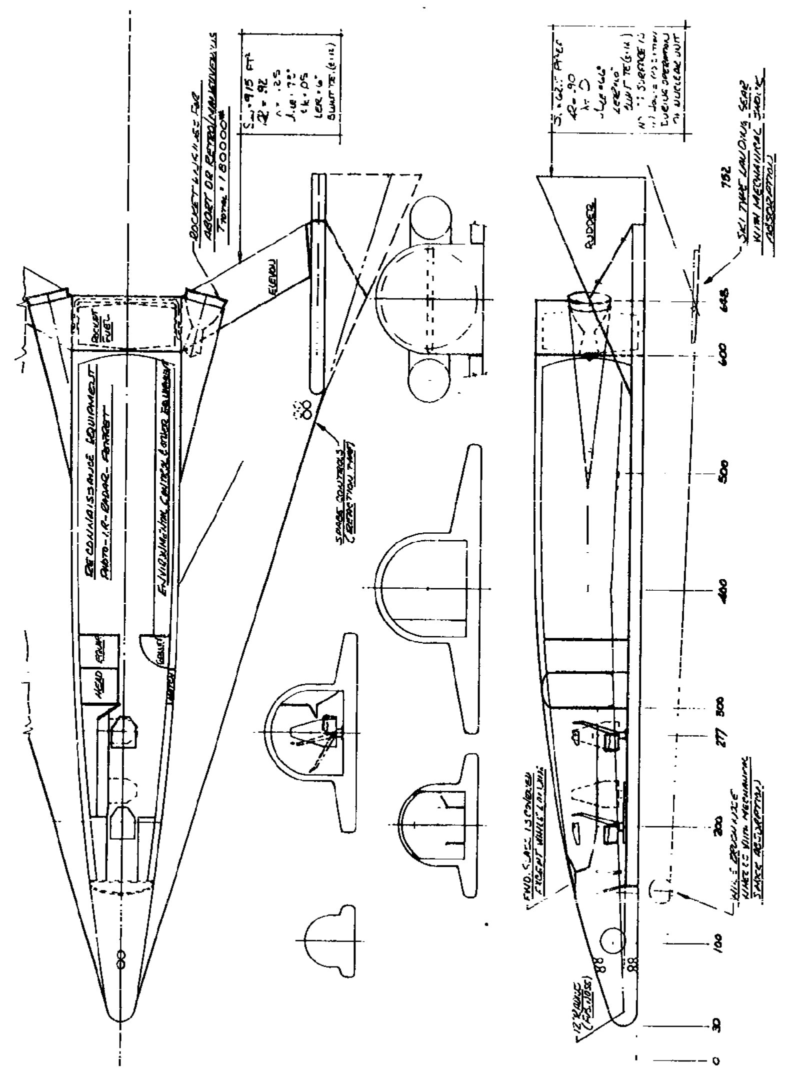

Figure 99. Vehicle 1B

The reduction of the crew from five to two necessitates increased automation, which is reflected in the slightly higher weight allowance for equipment. Because of the automation, the two-man work cycle will probably have no great effect on equipment utilization and power requirements.

Reconnaissance Equipment

Except for increased automation, the equipment remains basically similar to that outlined in Table 24.

Mission Power Requirements and Flight Vehicle Power

Since the equipment remains unchanged, only about 400 to 600 watts are saved by the reduction in crew size. Moreover, at least 600 watts are required for the increased automation.

For a 1-week mission, either the nuclear SNAP-8 unit or the "sunflower" solar collector unit is appropriate. If the duration of the mission is reduced to about 4 days, an LH_2-LOX turboelectric system becomes competitive for flight vehicle power. In either instance, an LH_2-LOX turboelectric unit is a possibility for flight vehicle power during reentry. This unit must supply about 69 kilowatt-hours of energy during reentry.

The weight of this version of the global surveillance vehicle has been computed on the basis of the heavier SNAP-8 unit for orbital power supply; and the heavier hydrazine turboelectric unit for reentry power supply.

Abort, Maneuvering, and Retro Capability

The total fuel requirements for this version are as follows:

Fuel Consumption -- 185,000-lb thrust at I_{sp} = 315 sec --- 585 lb/sec

Abort or Maneuver -- ΔV = 500 ft/sec at 4.1 g for 3.8 sec -- 2220 lb

Retro Thrust -- ΔV = 500 ft/sec at 4.2 g for 3.7 sec -- 2160 lb

Reentry

Trajectories have been computed for wing loadings of 30 and 40 (Figures 95 through 98). Although the wing loading of the two-man, 1-week version is 46 lb/ft^2, preliminary analysis indicated that the skin and leading-edge temperatures were not sensitive to a change of wing loading from 30 to 40. It has been assumed that a wing loading of 46 will also result in reasonable temperatures; additional trajectories have not been computed.

Weight Summary

The weight summary for Vehicle 1B is given in Table 27.

TABLE 27

WEIGHT SUMMARY FOR VEHICLE 1B

Structure		20,050 lb
Wing	8000	
Vertical Tail	2000	
Fuselage	6500	
Landing Gear	1200	
Engine Section	150	
Surface Controls	2200	
Power Plant		5,600 lb
Engine	2000	
Engine Controls	100	
Fuel System	1000	
Flight Vehicle Power System (Nuclear)	2200	
Reentry APU	300	
Fixed Equipment		15,050 lb
Instruments	400	
Hydraulics	400	
Electrical	700	
Electronics (6400 plus automation plus structure	8500	
Furnishings	1000	
T & A System, Food & H_2O	3550	
Data Return Capsules	400	
Auxiliary Gear	100	
Weight Empty		40,700 lb

TABLE 27 (Con't)

WEIGHT SUMMARY FOR VEHICLE 1B

Crew, Fuel and Other		5,730 lb
Crew (2 Men)	600	
Fuel (Abort/Maneuver & Retro)	4380	
Trapped Fuel	350	
Fuel (Attitude Control)	175	
Oil	25	
Miscellaneous	200	
Launch Gross Weight		46,430 lb
Orbital Weight		44,200 lb
Reentry Weight		42,000 lb

Vehicle 1C: Two Men, 12 or 36 Hours

Vehicle 1C is a two-man, 12- or 36-hour version of the manned orbital reentry vehicle. This version contains only limited equipment for global surveillance. In the following paragraphs, only the difference between Vehicle 1C and the basic five-man, 6-week, full surveillance version are described.

Description

The configuration of the two-man, limited-surveillance vehicle can be surmised from the drawings of the other two versions. Characteristics of this configuration are as follows:

Gross launch weight	26,075 lb
Reentry weight	24,175 lb
Wing area	750 ft^2
Reentry wing loading	32.2 lb/ft^2
Total volume (separable crew and equipment compartments)	1080 ft^3
Equipment and crew heat rejection	6 kw
Flight vehicle power	Chemical turboelectric

Although the maximum inside fuselage height is not specified, it is assumed that it is approximately 8 feet high to accommodate the crew and the nonfolded optics camera.

A gross take-off weight of about 25,000 pounds can be accommodated by the Saturn C-1 configuration, the first operational model of which is due in mid-1964 (Reference 33).

Orbital Mission Analysis and Selected Orbit

A useful mission can be executed by launching at the range safety limiting azimuth of 165 degrees from the Pacific Missile Range and recovering on a north-northeasterly heading some 12 or 36 hours later. Inclination would be about 75 degrees. The 12-hour mission should give approximately seven useful transits over the USSR or China, while the 36-hour mission should yield 14 useful transits. The 36-hour mission would allow considerable time for data processing and revised targeting, with one man sleeping and one man processing during the middle 14-hour period.

Reconnaissance Equipment

The design intention is to provide only limited surveillance in order to conform to boosters available at an earlier date. Full photographic surveillance was specifically considered. An alternate of full ELINT would demand perhaps 12 kw for flight vehicle power as compared to a total of 6 kw for the photo vehicle.

Mission Power Requirements and Flight Vehicle Power

A chemical flight vehicle power system is feasible, since a total of only 250 kw-hr is required for the 36-hour photographic surveillance mission. Cryogenic fuels are also feasible. This offers the possibility of using LH_2-LOX for abort, maneuvering, retro, orbital-phase flight vehicle power, reentry-phase flight vehicle power, and heat sink. Even though the specific fuel consumption of an LH_2-LOX fuel cell may eventually approach 1 lb/kw-hr as compared to 2 lb/kw-hr for a LH_2-LOX turboelectric system, the latter system was selected since the hardware weighs only 50 pounds, and because the heat sink capacity available with higher LH_2 flow is necessary to balance the integrated cooling load requirements. The fuel cell becomes attractive for lower power levels and perhaps somewhat longer endurance.

If ELINT, rather than photo surveillance, is selected as the mission, then 12 kw is demanded of the flight vehicle power unit during the orbital phase.

Abort, Maneuvering, and Retro Capability

The total fuel requirements for this version are as follows:

Fuel consumption – 104,000-lb thrust at I_{sp} = 415 sec or 260 lb/sec

Abort or maneuver – Δv = 500 ft/sec at 4.1 g for 3.8 sec	– 980 lb
Retro thrust – Δv = 500 ft/sec at 4.2 g for 3.7 sec	– 960 lb
Total	1940 lb

Reentry

See Figures 95 through 98.

Weight Summary

The weight summary for Vehicle 1C is given in Table 28.

TABLE 28

WEIGHT SUMMARY FOR VEHICLE 1C

Structure		16,425 lb
Wing	6500 lb	
Vertical Tail	1500 lb	
Fuselage	5500 lb	
Landing Gear	1000 lb	
Engine Section	125 lb	
Surface Controls	1800 lb	
Powerplant		2,150 lb
Engine	1200 lb	
Engine Controls	100 lb	
Fuel System	800 lb	
Flight Vehicle Power System (6 Kw LH_2-LOX Turboelectric)	50 lb	
Fixed Equipment		4,300 lb
Instruments	250 lb	
Hydraulics	300 lb	
Electrical	500 lb	
Photo Surveillance, Communications, and Navigation	2000 lb	
Furnishings	700 lb	
T&A System, Food, and H_2O (Stored gas atmospheric control)	300 lb	
Data Return Capsules	200 lb	
Auxiliary Gear	50 lb	
Weight Empty		22,875 lb
Crew, Fuel and Other		3,200 lb
Crew (Two men)	600 lb	
Fuel (Abort/Maneuver and Retro)	1870 lb	
Trapped Fuel	60 lb	
Fuel (Attitude Control)	50 lb	
Fuel (APU for 36 hr)	500 lb	
Oil	20 lb	
Miscellaneous	100 lb	
Launch Gross Weight		26,075 lb
Orbital Weight		25,100 lb
Reentry Weight		24,175 lb

Manned Orbital Base

To complete the category of environmental problems associated with orbital space missions, a system is required to introduce environmental problems common to high altitude permanently manned orbiting bases. This particular class of vehicle is believed to be of considerable military importance. Without elaborating the point, it is evident that a nation utilizing earth-orbital systems of this type during the period 1965-1975 would possess both unique defense and retaliatory capability (Reference 38).

The vehicle selected for the general mission suggested here may be identified with orbital base concepts developed under SR 181 (Reference 38). Briefly, the orbital base vehicle design entails three sets of rotating, eccentric, compartmented capsules remotely situated from an axially located power supply and control center. (See Figure 100.) The vehicle is designed to provide almost continual shirt-sleeve comfort and an artificial g environment for a crew of 40 to 60 men. Periodic maintenance of power equipment, the occasional change of duty stations, and departure on remote missions are the general nature of tasks requiring exposure of the crew to reduced atmospheric conditions common to companion ways and access tunnels. The logistics and reconnaissance vehicles indicated in the figure are not immediately important to the study. They are presented principally as subjects for later period equipment optimization discussions and system trades. Items to keep in mind here are (1) a possible reduction in meteoroidal and radiation shielding requirements for support craft beyond the 12-hour global surveillance vehicle requirements, and (2) the probable utilization of cryogenic chemicals aboard logistics craft.

Aside from the consideration of air locks, the internal environmental conditions the men will be subjected to will be quite similar to those already discussed for lesser duration orbital missions. The permanent nature of the base may indicate more sophisticated atmosphere regeneration techniques, and the accumulated effects of trace contaminants may be a problem.

External environmental problems deserving special attention are those associated with exposure of the vehicle, its equipment, and the crew to long periods of meteor bombardment, the Van Allen belt, and/or other radiation. Immediately obvious design requirements arising from exposure of the vehicle and crew to increased meteoroidal and radiation hazards are (1) heavier cabin wall structure and a better state of preparedness for serious meteoroidal punctures, and (2) duty assignment rotation for the crew commensurate with job locality, radiation exposure, and individual radiation exposure tolerances. Other items worthy of investigation are (1) effects of erosion on physical properties of radar dishes, optical equipment and radiators from prolonged meteor bombardment, and (2) new environmental aspects introduced by order of magnitude changes and differences in vehicle geometry. The fact that the base does not reenter may lead to different solutions to various environmental problems.

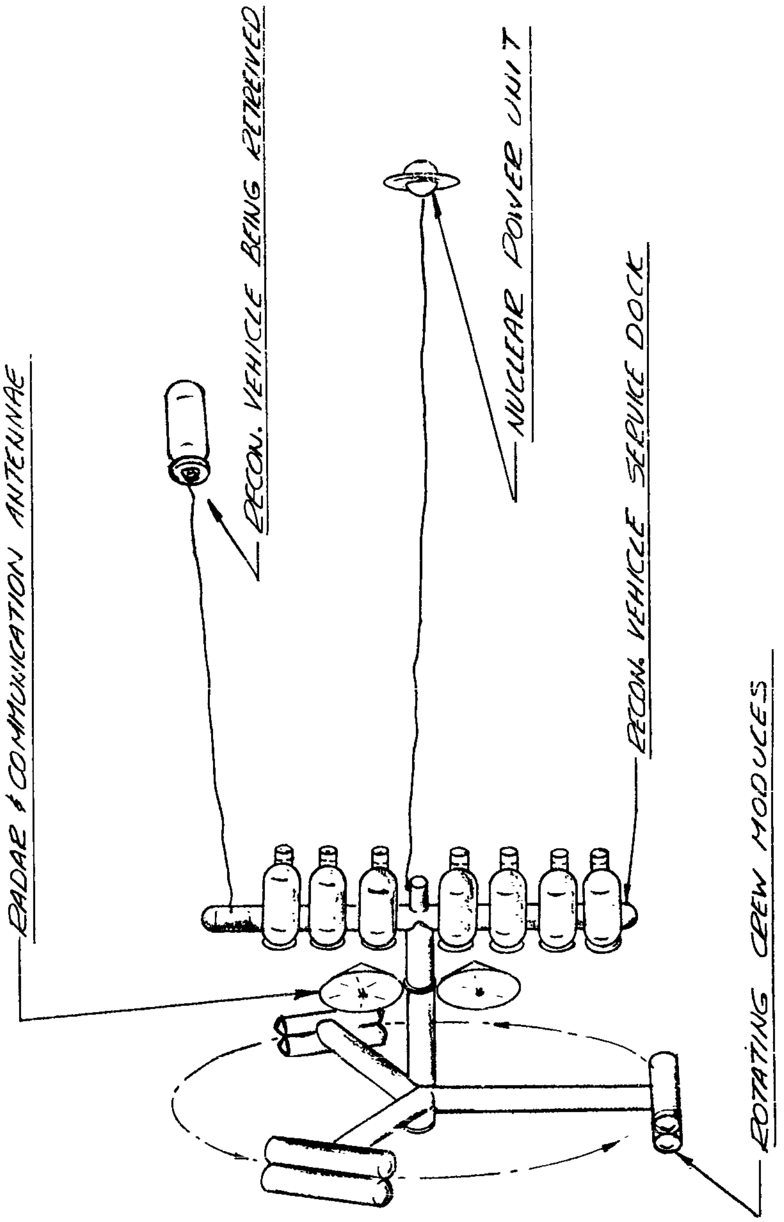

Figure 100A. Tentative Configuration of a Manned Orbital Base

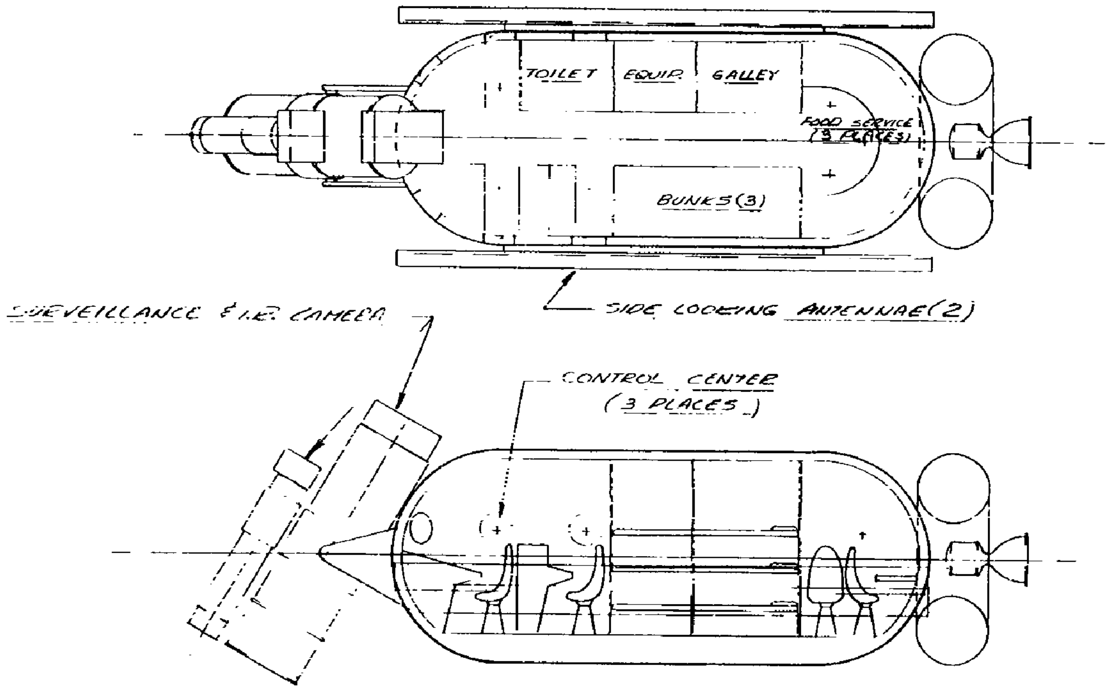

Figure 100B. Orbital Reconnaissance Vehicle for Orbital Base

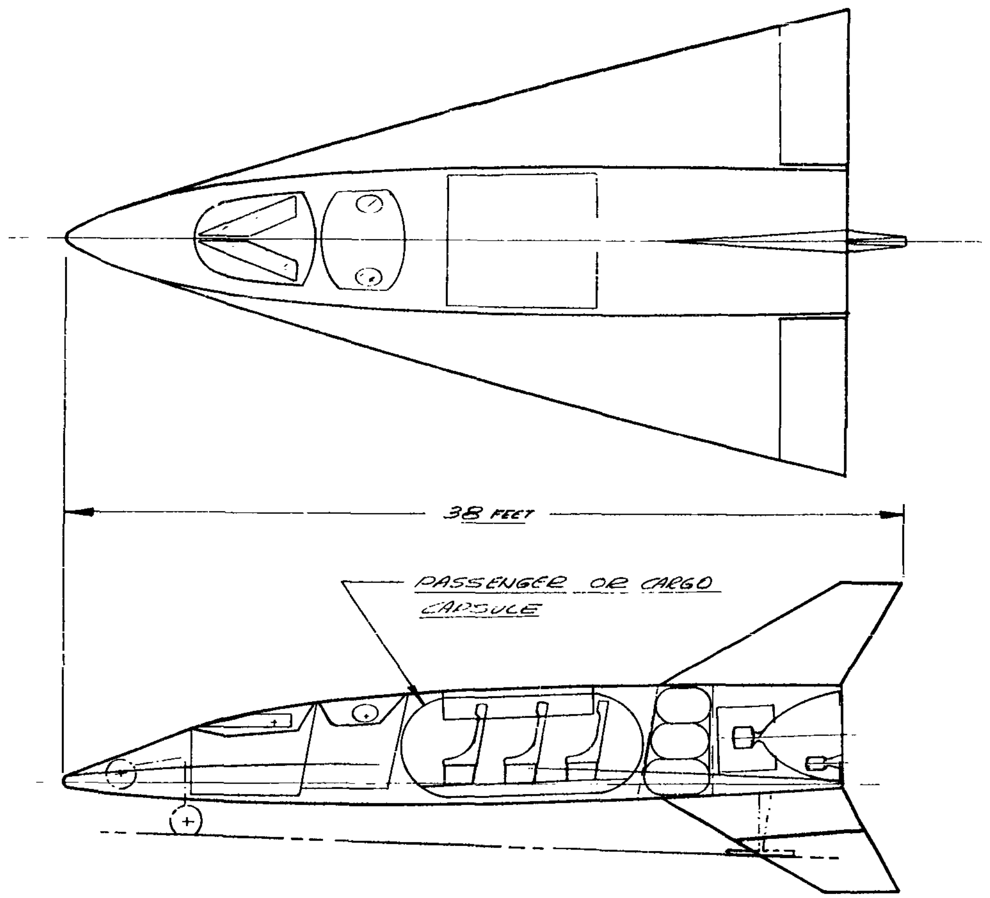

Figure 100C. Reentry Shuttle Vehicle for Orbital Base

Manned Lunar Landing Vehicle

Previously described space missions specify orbital systems that provide the basis for investigating a majority of environmental problems associated with space flight. The mission selected to represent the balance of environmental problems likely to be encountered in executing manned space flights during the period 1965 to 1975 is the manned lunar landing vehicle.

The general characteristics of a lunar mission are familiar (References 39 and 40). Lunar flights may be considered to originate either from a fixed earth launch site or from an orbiting base, depending on the time period in question. Early time phasing, such as might be possible with the NOVA concept, calls for earth launching. Later period flights most likely will originate from orbiting bases. The timing of this particular study (1965 to 1975) would appear to exclude consideration of environmental problems associated with orbital-base moon launches and recoveries, except perhaps for orbital refueling concepts. New environmental problem areas introduced by considering early period manned lunar flights are effects arising from exposure of the vehicle, its occupants, and equipment to high-landing shocks, possible effects of lunar dust on radiating surfaces, long periods of 100-percent sunlight, passage through the radiation belt, and hyperbolic reentry velocities.

A major portion of environmental problems associated with early moon flights may occur during the reentry phase of the mission at hyperbolic reentry velocities. In fact, the accumulative environmental problems encountered here actually define the reentry path of the vehicle. Here we find limiting environmental values of heat, deceleration, and radiation exposure defining what is known as the reentry corridor.

Typical moon-flight entry corridors are shown in Figure 101. It may be observed that two different techniques for reentry are indicated here: the single-pass and the multiple-pass reentry techniques. At its conception, the multiple-pass technique was considered the more practical due to reduced guidance requirements and stage-wise energy decay considerations. Later findings of high-intensity radiation belts surrounding the earth make the single-pass reentry more likely (Reference 37).

The depth of the entry corridor is generally small. Correspondingly, for ballistic-type vehicles, guidance accuracy required to effect reliable single-pass reentry is high. Lifting-type vehicles are known to increase the dimensions of the entry corridor and ease single-pass guidance requirements (References 35, 36, and 37). The essential point of the referenced reports is that entry corridor dimensions may be effectively increased by discrete modulation of lift and drag during reentry. Beneficial entry corridor effects achievable here would be (1) a general smoothing-out of thermal and deceleration peaks, (2) a reduction in skip-out tendency, and (3) steeper permissible reentry trajectories. Although ballistic vehicles are structurally more

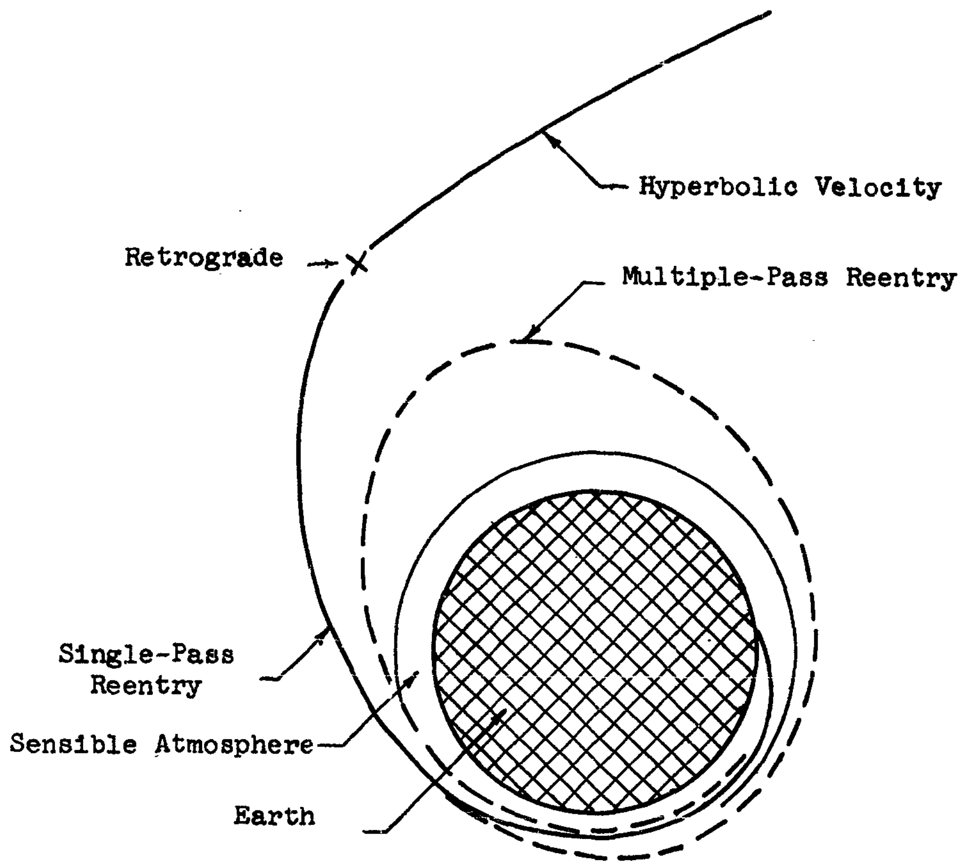

Figure 101. Single- and Multiple-Pass Reentry Paths

efficient than lifting vehicles, evidence suggests that moon vehicle design will rapidly depart from such early ballistic concepts as indicated in Reference 39 (ballistic design was favored here because of booster limitations). Relaxed guidance requirements and less severe exposure to environmental hazards are afforded by lifting vehicles.

Flight Vehicle Power

The primary considerations for the selection of a flight vehicle power system are the mission duration, the magnitude of power required, and the load duty cycle. Once these factors have been determined, data, such as are contained in Figures 102 and 103, are helpful in selecting the proper flight vehicle power system. These figures indicate specific weight versus power output and system weight versus duration, respectively. A detailed discussion of these two figures is given later in this section.

Typically, a manned vehicle requires a load duty cycle that fits the mission requirements, controls environmental conditions, and supplies the stored reserve of additional power for peak loads and emergency needs. In addition to satisfying this load duty cycle, the auxiliary power system must be reliable. System reliability is materially increased by using a basic power supply augmented with subsystems as backup for any special requirements. Generally speaking, one basic power supply will not completely satisfy all mission requirements. Integration of the several power supplies will result in a minimum weight/volume system, in addition to one that is flexible enough to allow readily for expansion of the system or to accommodate changes with modifications of the mission. A tie-in of a chemical flight vehicle power system with the on-board propulsion fuel can also effect a substantial weight saving if the systems are compatible.

By definition, the flight vehicle power systems considered for the time period in question exclude exotic energy sources such as heat from thermonuclear fusion and the more advanced heat conversion schemes such as magnetohydrodynamic techniques.

The primary sources of energy available for power conversion are (1) nuclear sources, (2) solar sources, and (3) chemical sources. Application of these sources cannot fit logically into integrated systems unless the advantages and disadvantages of each candidate system are taken into account.

Nuclear Sources (SNAP 1, 2, 3, 8, and 10)

The term SNAP has been coined from the words "systems for nuclear auxiliary power". Power levels up to 1 kw can be supplied by radioisotope sources such as SNAP 1, 1A, and 3. At higher outputs, they are not economically competitive because of the very limited supply of radioactive material adaptable to this type of heat source. The SNAP 10 thermoelectric system can supply 300 watts, and, with development, will produce 600 watts. For power levels of 3 to 6 kw, the SNAP 2 turboelectric system is appropriate. At 35 kw and over, the SNAP 8-type turboelectric power source can be utilized. Nonradioisotope-type systems can be extended up to the megawatt level without serious

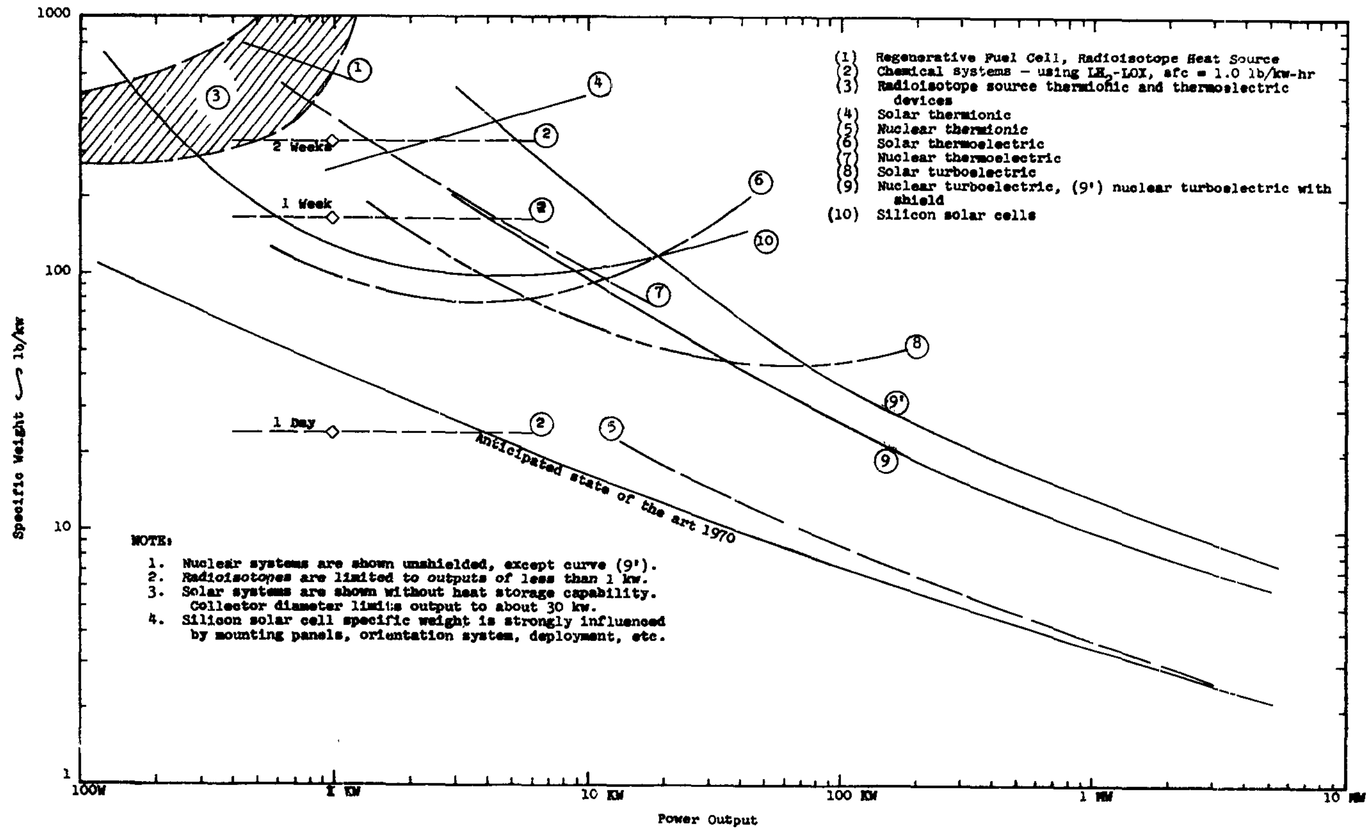

Figure 102. Flight Vehicle Power Systems, Specific Weight Versus Power Output

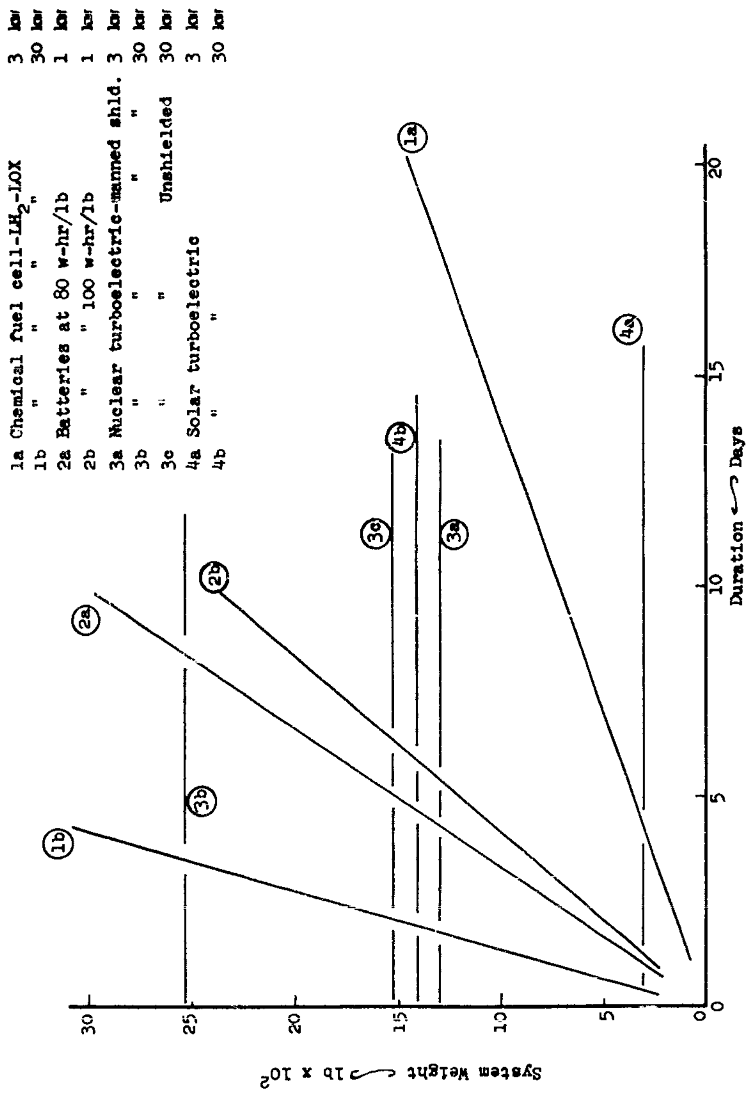

Figure 103. Flight Vehicle Power Systems, System Weight Versus Duration

problems. Advancements in thermionic technology will substantially reduce the specific weight of nuclear systems.

Nuclear sources are desirable for long duration applications since, within limits, weight is independent of mission duration. Compactness is another advantage. Nuclear sources possess a good growth potential since output can be increased without substantial increase in weight and short-duration overloading can be tolerated. Only low-to-moderate development costs would be required for early delivery. The thermionic and thermoelectric types have no principal moving parts.

On the other hand, shielding of nuclear sources may increase system weight by 50 to 200 percent, depending upon allowable dose level, separation distance, reactor output, etc. Power output is fixed and heat dissipation is necessary for off-design operation. Start-up problems and shutdown problems associated with reentry of nuclear systems are not well defined. At the lower power levels (less than 10 to 20-kw) the specific weight (pounds per kilowatt) is higher than for the nonnuclear systems.

Solar Sources (Sunflower, Solar Cells)

For each kilowatt of output, solar cell panels require about 100 square feet of area and weigh about 150 pounds, depending on installation. Surface area limitations impose restrictions on the magnitude of power generated. These quoted values can be reduced by using reflectors and by operating at higher temperatures. The Sunflower-type system is a turboelectric system whose heat source is located at the focus of a large parabolic mirror. These systems can be used, up to 30-kw output, with collector diameters of about 60 feet, depending on component efficiencies. Increasing the output will aggravate problems associated with collector deployment and orientation.

In general, solar sources are advantageous for long-duration applications, weight being independent of mission duration. Up to 30-kw power output, solar sources have low specific weight (pounds per kilowatt); they are the lightest of energy sources if duration is more than 10 to 20 days.

Disadvantages of solar sources include the necessity for large collecting surfaces, the damaging effects of irradiation and meteoroids because of these large surfaces, the necessity of energy storage during earth shadow operation, packaging and deployment problems because of great bulk at 20 to 30-kw output, and orientation problems. The orientation problems are primarily indirect; pointing accuracies (± 10 degrees for solar cells ± 1 degree for collectors) are easily attained, but the perturbing torques can be a serious problem for pointing reconnaissance cameras, for example.

Chemical Sources (Fuel Cell, Batteries, Turbomechanical)

In general, for a given output, the system weight is proportional to duration. Chemical systems are best suited to low output (3 to 5 kw) short-duration (50 to 100 hour) applications. Super insulators are making cryogenic fuels more practical. With cryogenic fuels, such as LH_2-LOX, the system is readily integrated with environmental temperature control. Very low fuel consumption using liquid hydrogen makes the system attractive for short-duration, light-load missions, and emergencies. A weight saving can sometimes be made by utilizing the on-board propulsion fuel and tanks. A storable hot-gas system is ideally suited for the short-duration, high-power reentry and landing loads. Batteries are generally limited to low-energy requirements because of weight. However, storage batteries are useful in supplementing other systems during short-duration peak loads or periods of nonoperation.

One important advantage of chemical systems stems from the fact that many of these systems have been developed for some time and reliability has been proven. Desirable possibilities for integration, short-duration economy, and energy storage were outlined above. Another advantage is the adaptability of chemical power units to vehicles which operate independently of the "mother-ship".

The principal drawback of chemical systems is the fuel weight, which becomes prohibitive for high-power, long-duration applications. In many instances, the specific fuel consumption (pounds per kilowatt hour) is particularly poor at off-design (part load) operation. Space and weight required for tankage are other disadvantages. Special problems include possible irradiation effects and zero-g handling.

Comparison of Various Flight Vehicle Power Systems

The curves of specific weight (pounds per kilowatt) versus power output in Figure 102 show the relative position of all the power sources and conversion techniques that were considered for this study. Curve (1) represents a regenerative fuel cell with a radioisotope heat source. This curve stops at 1 kw since high power output information was not available. Total system weight includes inverter equipment, supplementary batteries, miscellaneous structure, and approximately 190 pounds of shielding at 0.55-kw output. Further information may be found in Reference 41.

Curves labeled (2) give data for a LH_2-LOX chemical system for durations of 2 weeks, 1 week, and 1 day (Reference 42). For fuel cells, a specific fuel consumption (SFC) of 1.0 pound per kilowatt-hour is estimated as the best possible for LH_2-LOX. Fuel cell hardware is estimated to be about 20 to 100 pounds per kilowatt depending on the output power level and duration of operation. The turboelectric systems may use an SFC of 1.5 to 2.0 pounds per kilowatt-hour for short durations when it is integrated with the thermal control system as the vehicle heat sink. For longer durations (days instead of hours), when a radiator heat sink is advantageous from the weight standpoint, an LH_2-LOX turboelectric system having a high efficiency is necessary to realize a minimum total system weight. The corresponding SFC may be as low as 1.1 or 1.2 pounds per kilowatt-hour.

The area labeled (3) in Figure 102 represents radioisotope heat sources with currently achievable thermionic and thermoelectric conversion techniques. Radioisotope systems have very limited application, power output being limited to less than 1 kilowatt. Shielding requirements strongly influence specific weight.

Solar thermionic systems expected around 1970 are indicated on curve (4) (Reference 43). The fact that power output is limited by collector size suggests the use of multiple units. System weight estimates may vary as much as ± 50 percent.

Curve (5), showing nuclear thermionic systems assumed for 1970, is based upon a 300-kw system and recent technical reports (References 44 and 45). Not included in the system weight is the shielding weight, which is a function of the mission, the conversion weight, which is a function of power output and duration, and miscellaneous structure and components.

Solar thermoelectric systems, curve (6), are not competitive at high outputs because of material temperature limits and to solar collector diameter. Research and development is required for thermoelectric materials. There is a possibility that thermoelectric elements can be utilized in a binary-type cycle using the waste heat as input to the high-temperature side. System weight does not include heat storage or sun orientation equipment. Calculations are based on the following approximate weight estimates: 8.5 percent for the thermoelectric element, 62 percent for the solar collector, 21 percent for the radiator, and 8.5 percent for miscellaneous.

Nuclear thermoelectric systems are indicated by curve (7). A SNAP 10A unit is currently being fabricated for flight test in 1963 (Reference 46).

Development trends indicate that 3 to 15 kw solar turboelectric system, curve (8), may be feasible by 1965. (See Reference 47).

Curve (9) is for a nuclear turboelectric system and Curve (9) is for the same nuclear turboelectric system with shielding added. Shield weight is based upon a dose rate of 0.03 rem/hr at 50-ft separation. SNAP 2 will be flight tested in 1964. A 300-kilowatt version will be tested in 1968, and a 3000-kilowatt version by 1970. (See Reference 48.)

Silicon solar cells are considered in Curve (10). (See Reference 49.) Total system weight is strongly affected by panel and structural support, orientation, development, and other factors. Current use is for power levels less than 1 kilowatt. With development, 3- to 5-kilowatt units may be feasible by 1965.

Another interpretation of power system weight is shown in Figure 103, where the nuclear and solar power sources are seen to be independent of duration.

General Aspects of Structures for Manned Space Vehicles

The general investigation of structures for manned space vehicles reported in this section has centered around cabin walls for reentry vehicles. A maximum skin temperature of 2600°F during reentry was assumed. Simultaneous stress loads corresponding to an ultimate cabin pressure loading of 20 psi and panel shear of 300 lb/in. were considered to be realistic mechanical loading criteria. Under actual conditions, the heat and shear loads would occur only during exit and reentry. As is indicated in the general section on meteoroids, pressure hull damage by meteoroids during a 6-week period is insignificant for cabin wall designs which meet weight, temperature, pressure, and shear stress requirements. In addition, the cabin wall must provide sufficient protection from noise, gamma rays, cosmic rays, high-speed neutrons, protons, and electrons.

Within this section, basic types of cabin wall are identified and analyzed, and the basic properties of candidate materials are shown in graphical form.

Cabin Wall Structures for Manned Reentry Vehicles

Three basic types of cabin wall structure are considered: (1) a conventional skin, stringer, and C-frame type structure; (2) honeycomb sandwich panels and honeycomb sandwich frames with insulation on both sides of the panels, and corrugated sandwich insulation retainer on the outside of the panels; and (3) a structure similar to (2) with the addition of an interior pressure membrane (cabin pressure retainer) supported by the frames. These three types of wall are illustrated in Figure 104. Case I corresponds to the conventional skin and C-frame structure made of columbium. Cases II and III illustrate the honeycomb sandwich structure without the inner pressure hull, with a nickel alloy (René 41) assumed for Case II and titanium alloy assumed for Case III. Cases IV and V incorporate the inner pressure hull, with René 41 used for Case IV and the titanium alloy used for Case V. In all cases the external skin is columbium, which retains much of its strength at 2600°F. In Cases II and IV a temperature drop through the outer insulation is assumed, so that the honeycomb panels and frames are at about 1400°F. René 41 is still suitable at this temperature. Similarly, in Cases III and V, a temperature drop to about 800°F is assumed, at which temperature the titanium alloy is suitable. In Cases IV and V, aluminum is assumed for the pressure membrane. Figure 104 shows the pressure membrane in detail. Room temperature was assumed in analyzing the stresses on the pressure membrane.

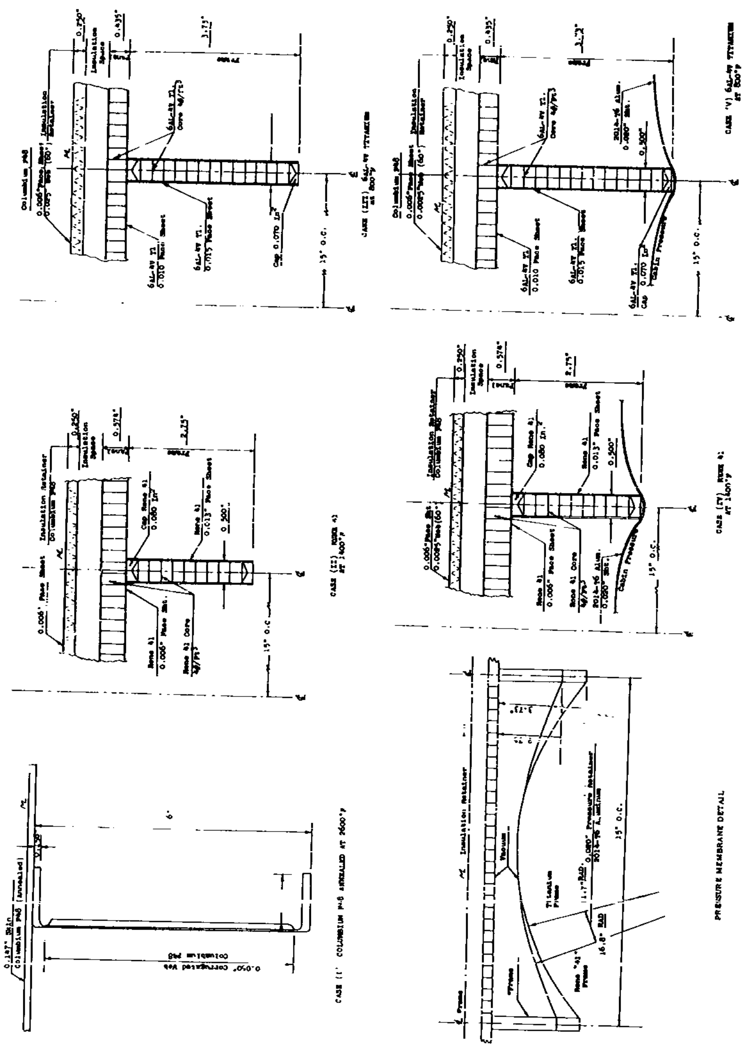

Figure 104. Types of Cabin Walls Considered for Manned Reentry Vehicles

Analysis of Appropriate Structures

The results of the analysis of Cases I to V are shown graphically in Figure 105. Note that results obtained to date do not include insulation weight, and thus, the optimum weight points shown in Figure 105 will have to be modified when the insulation weight is integrated into the total cabin wall weight. To facilitate the integration of insulation weight, a temperature-versus-weight correction was included on the graphs for each type of structure studied.

The bar chart in Figure 105 indicates that Case I (conventional skin and C-frame structure) is not suitable when compared with Cases II and III (honeycomb sandwich structure). Optimum weights are 9 lb/ft^2, 2.25 lb/ft^2, and 1.93 lb/ft^2 for Cases I, II, and III, respectively. Note also that meteoroid damage becomes a significant factor at skin gages corresponding to frame spacing of less than about 5-1/2 inches for the conventional skin and C-frame structure.

Addition of the cabin pressure retainer increases the weight by about 14 percent (Case IV compared to Case II, and Case V compared to Case III). The cabin pressure retainer has a safety advantage, in that cabin pressure can be contained by the honeycomb sandwich panel should the membrane cabin pressure retainer fail. Moreover, a vacuum between the cabin pressure retainer and the sandwich panel increases the insulation effectiveness; thus, Cases IV and V may be competitive with Cases II and III when insulation weight is considered.

For optimum weight considerations, minimum-gage face sheets are assumed for the honeycomb panels: 0.006 inch for René 41 and 0.010 inch for 6AL-4V titanium. A honeycomb panel core density of 4 lb/cu ft was set as a representative value.

Figure 105 shows that for Cases III and V, a ±13-percent weight change (±0.25 lb/ft^2) is indicated for titanium in the temperature range of 500°F to 900°F; for Cases II and IV, a reduction from 1400°F to 1200°F represents a 10-percent weight saving (0.15 lb/ft^2) for René 41, no further weight reduction below 1200°F being indicated.

In using the data contained in Figure 105, several things should be kept in mind. First, panel depth and frame depth are a function of frame spacing. Second, a correction must be made for thermal characteristics if the René 41 is to be used at other than 1400°F, or if the titanium alloy is to be used at other than 800°F. Third, allowance must be made for the insulation retainer and cabin pressure retainer, according to the instructions contained in the figure. Fourth, allowance must be made for insulation weight. Finally, it should be remembered that inclusion of insulation may modify the optimization of the cabin wall with respect to weight.

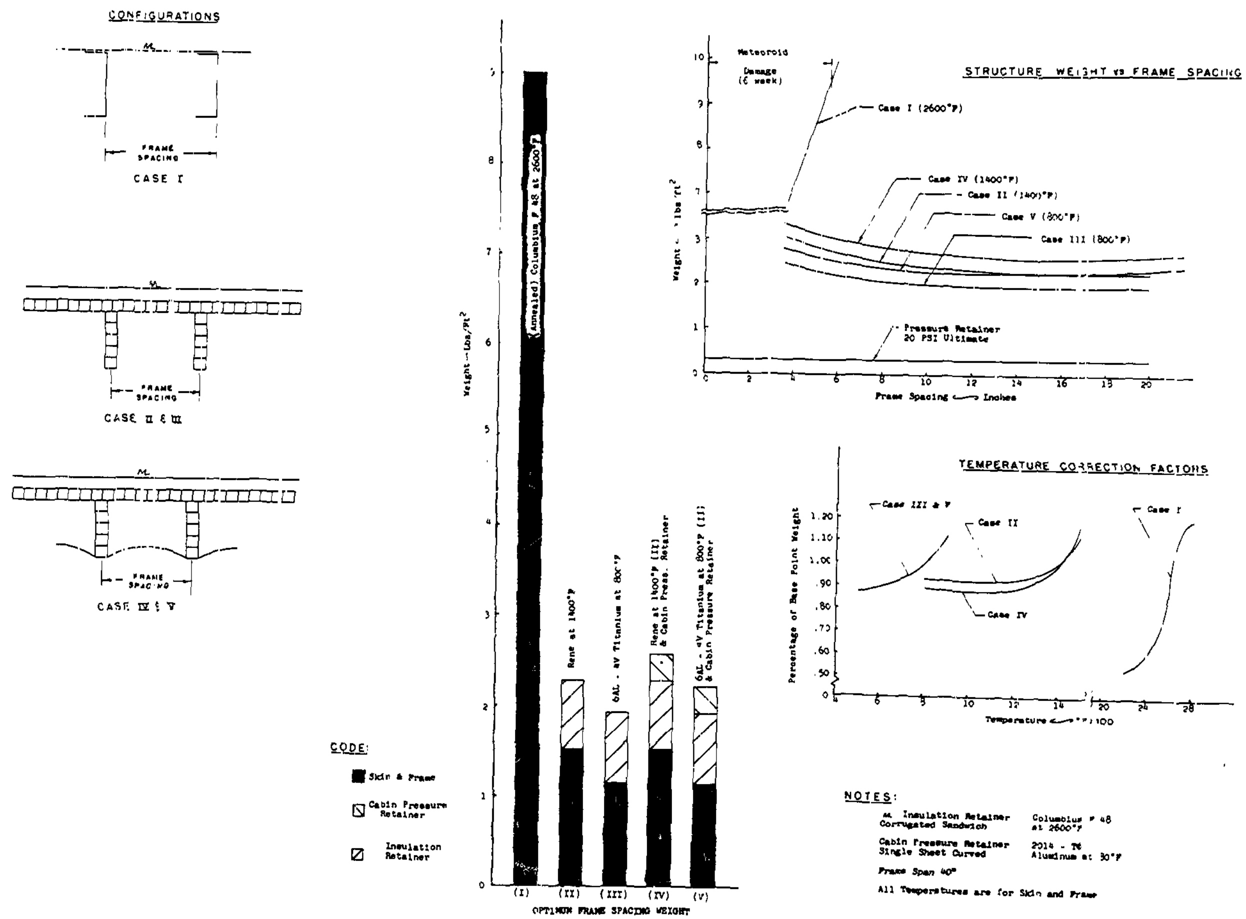

Figure 105. Analysis and Optimization of Cabin Walls

Applying the formulas and data contained in the general section on damage by meteoroids, it is found that Cases II and III might involve a small weight penalty to insure better than a 99-percent probability of no penetrations (less than one penetration in 100 6-week missions). It should be noted that this estimate tends to be conservative and that a single penetration of the size being considered would result in a negligible pressure loss. The pressure hull integrity corresponding to Cases IV and V is even greater because of the added cabin pressure retainer.

Materials

Various possible materials are considered in Figure 106, where tensile stress/unit weight, compressive yield/unit weight and modulus of elasticity/unit weight are all plotted as a function of temperature. Of the three properties, compressive yield/unit weight is most often the determining property because of panel buckling problems.

Examination of these material property curves indicates that for honeycomb structure in the 1400°F temperature range, chromium- or vanadium-base alloys could be used with little change in weight. Therefore, if thermal properties favor a change in material, weight would not be an objectionable factor. For the 800°F range, honeycomb PH-15-17 Mo could be used with a weight penalty of 10 percent; at 600°F, the weight penalty would be 5 percent. Other materials appearing more favorable on a weight basis are not practical for manufacturing reasons (minimum gage problems).

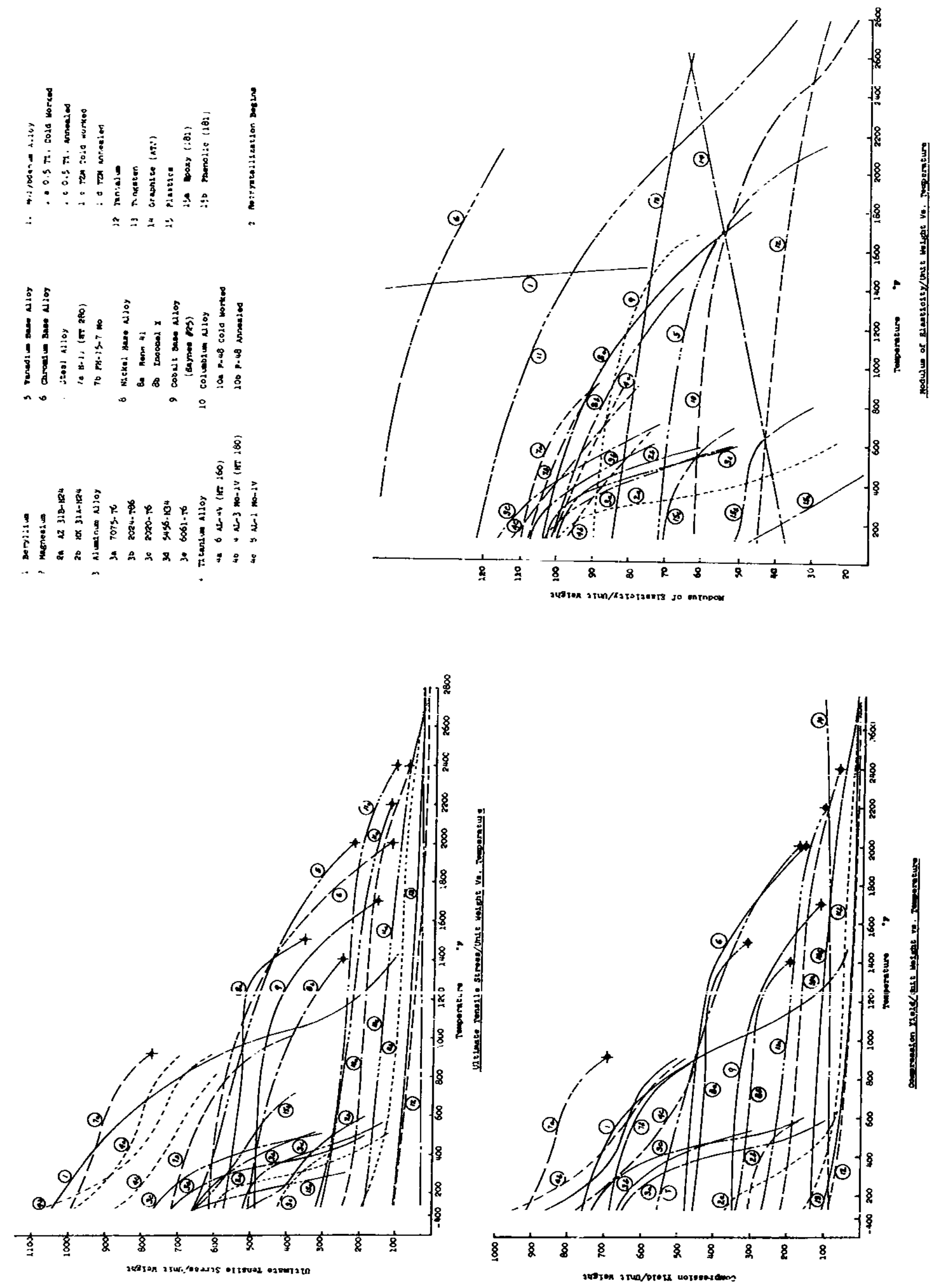

Figure 106. Material Properties Versus Temperature

Effects of Meteoroids

Meteoroids represent a potential hazard to pressure hulls and radiators through puncture, and to radiating surfaces through erosion. The potential hazard to the pressure hull appears to be no problem for missions lasting 6 weeks or less for cabin wall structure whose minimum weight design is determined by thermal and mechanical stresses. Damage to radiators during a 6-week period can be significant enough to require division of the radiator into sections and/or use of heavier gage outside surfaces. There appear to be no significant erosion effects indicated for a 6-week mission. Pressure hull and radiator damage are considered in detail in the following. Detailed analysis of erosion effects has been deferred, since long-duration missions are not part of the present study.

In all of what follows, it should be noted that there is a lack of empirical data on the flux of meteoroids and the effect of meteoroids on materials.

Damage to Pressure Hulls

The estimated pressure hull integrity of the cabin wall design selected for Vehicles 1A, 1B, and 1C rests upon the meteoroid bumper effect of the insulation retainer and the multisandwich-type construction. The outer layers of material act as a meteoroid bumper by breaking up impinging particles and spraying them over a large area.

The problem of meteoroid encounter and penetration of a space vehicle structure involves four basic parameters: (1) probability of no puncture of a given structural configuration of some exposed area for a given time period, (2) the shield thickness or thickness combinations (bumper concept) required to insure no puncture, (3) probability of at least one encounter during a given time by particles of various size, and (4) the size of the impacting particle for at least one encounter.

For Vehicle 1A, which is to have a 6-week mission duration, the structural configuration assumed is shown in Figure 107. The multisandwich-type construction consists of an outer insulation retainer face of columbium sheets, an intermediate wall of Rene′41 honeycomb, and an inner wall of aluminum. Based on a mean fuselage diameter of 100 inches, a length of 668 inches, and the various shielding effects, the cabin average mean surface area is estimated to be 1192 ft^2. Included in the average mean surface area is the protected area afforded by the wing.

Reference 78 contains the development of the following general penetration equation:

$$t_s = \alpha d \left(\frac{\rho_P}{\rho_T}\right) \quad \phi \left(\frac{V}{C}\right) \quad \Theta \tag{L-1}$$

where (t_s) is the thickness of the meteoroid shield; (d) is the diameter of the striking particle that is just capable of penetrating the shield; (ρ_P) is the density of the particle; (ρ_T) is the density of the target material; (C), is the velocity of sound in the undisturbed target material; (α), (ϕ), and Θ are empirical parameters; and (V) is the velocity of the particle. Based upon Equation (L-1), Herndon has derived the following meteoroid puncture equation (Reference 79):

$$t_s = \frac{2.48 \times 10^{-3.6}\left[V^{1/3}\right]K_B\left[(A\tau)^{3/10}\right]}{\left[-\mathrm{Log}_e P(0)\right]^{3/10}} \tag{L-2}$$

where (K_B) is an empirical constant relating penetration depth to critical physical properties of the target material, (A) is the surface area, (τ) is time of mission and P(0) is the probability of zero penetration. This equation does not include the bumper effect to evaluate the effective shield thickness.

The "meteoroid bumper" has been suggested by Whipple (Reference 80). The beneficial effect of bumper surfaces external to the pressurized cabin wall is highly theoretical. However, Whipple states that if the bumper thickness is 10 percent of the primary wall thickness, the number of punctures should be reduced by a factor of approximately ten to a hundred.

Using Equation (L-2) and the more conservative Whipple factor of 1/10th for relieving nature of a secondary surface, Herndon has shown the following (Reference 79):

$$\frac{t_B}{t_s} = \left(\frac{1}{10^n}\right)^{3/10} \tag{L-3}$$

where (t_B) is the bumper shield thickness, (t_s) is the single shield thickness, and (n) is the number of bumper shields. For the structural configuration given in Figure 107, and assuming $t_1 = t_2 = t_3 = t_4 \geq \frac{1}{10}\ t_i$, the effective energy relieving value is

$$\frac{t_B}{t_s} = \left(\frac{1}{10^4}\right)^{3/10} \approx \frac{1}{16} = 0.0625 \tag{L-4}$$

This indicates that the required thickness of the inner cabin wall (t_i) is 1/16th of that required if no bumper were present.

Applying the bumper concept of Equation (L-4) to the structural configuration of Vehicle 1A, the effective shield thickness is

$$t_s = \frac{t_B}{0.0625} = \frac{0.02}{0.0625} = 0.32 \text{ in.}$$

The calculation of the particle mass that would have the required energy level to penetrate this shield thickness is as follows:

$$M = \left(\frac{t}{K}\right)^3 V^{-1} \qquad \text{(L-5)}$$

$$M = \frac{(0.32)\ (2.54)}{0.496}^{3} \left(\frac{1}{30}\right)$$

$$M = 3.94\ 10^{-2}\ \text{gm}$$

where

M = Mass of particles, gm

t = Thickness of target, cm

V = Velocity of particle, km/sec

K = Constant from Bjork correlation relating particle density to target density (Reference 81)

The diameter of the particle based on a spherical mass and a density of 0.05 gm/cm^3 would be 0.69 inches. Therefore, this is the critical particle size involved in this study.

Based on Whipple's meteoroid flux data (Reference 80), Figure 108 indicates the probability of at least one encounter for various particle sizes during the mission of Vehicle 1A.

From Equation (L-2) and Figure 108, the probability (P_D) of at least one encounter by a particle equal to, or greater than, the critical size is as follows:

$$t_s = \frac{\left[2.48 \times 10^{-3.6}\right] K_B \left[V^{1/3}\right]\left[(A\tau)^{3/10}\right]}{\left[-\text{Log}_e P(0)\right]^{3/10}} \qquad \text{(L-6)}$$

$$-\text{Log}_e P(0) = \frac{\left[10^{-\frac{40}{3}}\right]\left[K_B^{\frac{10}{3}}\right]\left[V^{\frac{10}{9}}\right](A\tau)}{t_s^{\frac{10}{3}}}$$

$$-\text{Log}_e P(0) = \frac{2.154 \times 10^{-13}\left[(0.496)^{\frac{10}{3}}\right]\left[(30)^{\frac{10}{9}}\right](5 \times 10^4)}{(0.32)^{\frac{10}{3}}}$$

$$-\text{Log}_e P(0) = 1.975 \times 10^{-6}$$

$$P(0) = 0.9999$$

$$\text{or } P_D = 1 - P(0)$$

Therefore, there is no tangible probability of at least one encounter of a particle of this size. For the assumed exposed area and duration, the structural configuration of Vehicle 1A is most satisfactory.

From Equation (L-2), Herndon (Reference 79) developed a parametric design curve of P(0) vs $AT/\tau_s{}^3$ for aluminum. Using this curve, different values of P(0) are assumed in order to solve for corresponding values of effective shield thickness. Figure 109 is a plot of the probability of zero penetration versus cabin pressure retainer thickness for the exposed area and mission duration corresponding to Vehicle 1A. For the configuration given ($t_i = 0.02$ in.), the probability of zero puncture is 0.99^+, since the probability of encounter is $0 < P_D < 0.01$. Therefore, the possibility of a puncture during the mission of Vehicle 1A is very remote.

Damage to Radiators

The space radiator has critical surfaces exposed directly to meteoroid flux during orbit. The following two design approaches to this problem are considered: (1) a configuration similar to the first one shown in Figure 110

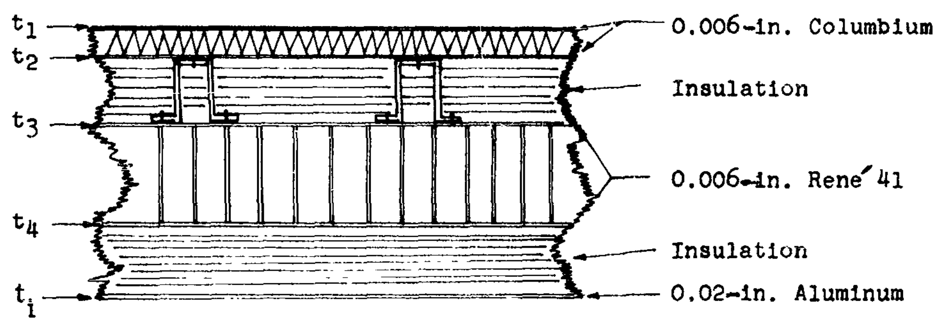

Figure 107. Vehicle 1A Structural Configuration

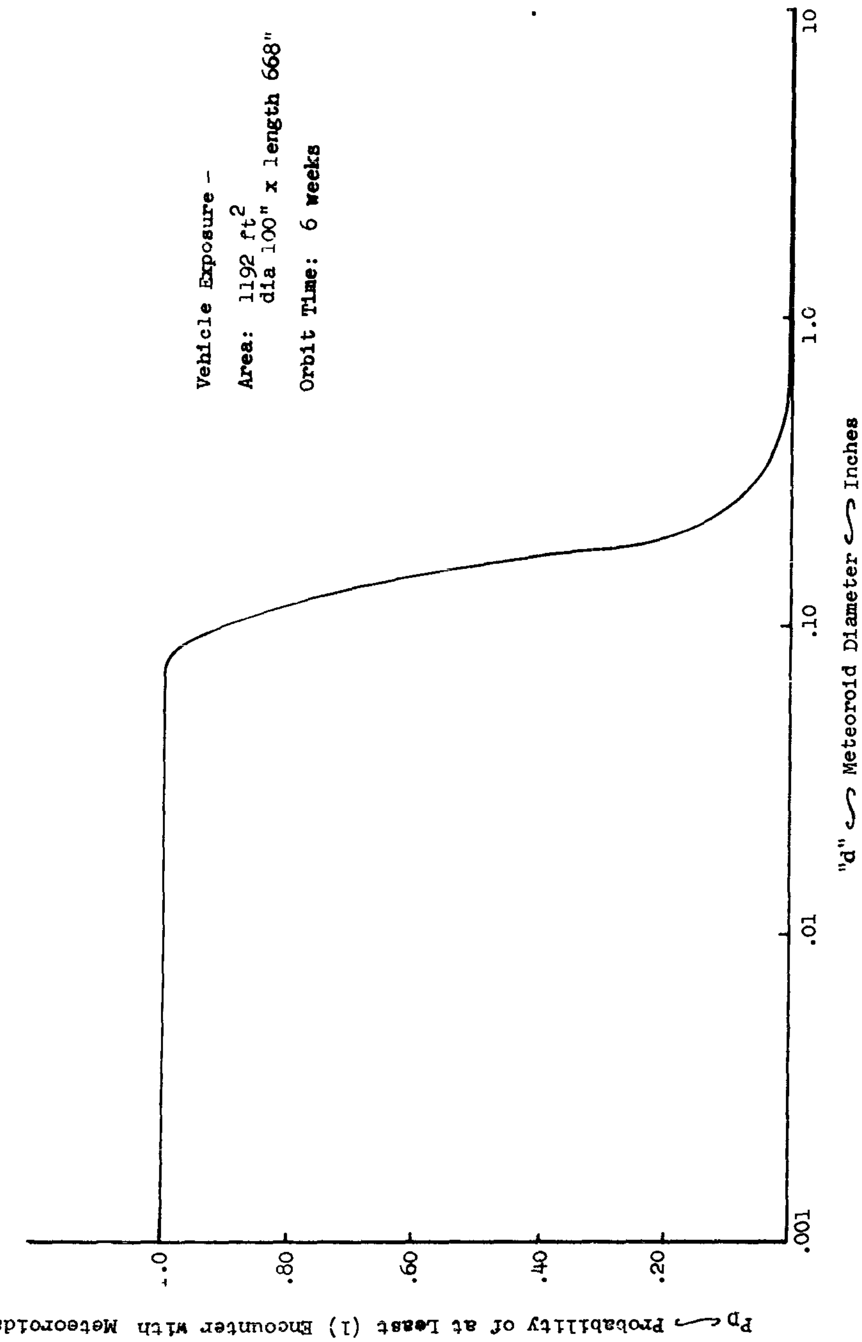

Figure 108. Probability of Meteoroid Encounter Versus Size of Meteoroid

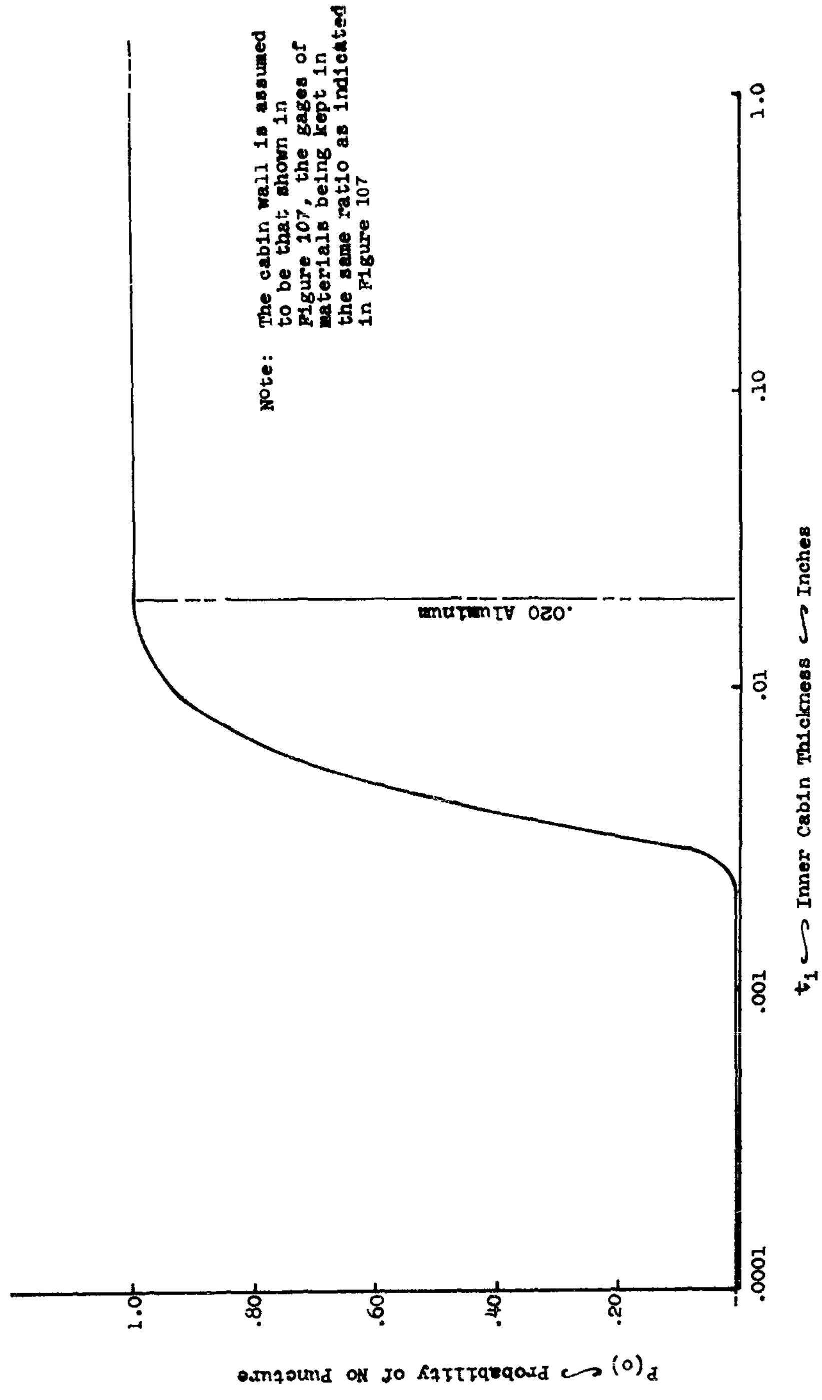

Figure 109. Probability of no Puncture of Cabin Pressure Retainers of Various Thicknesses

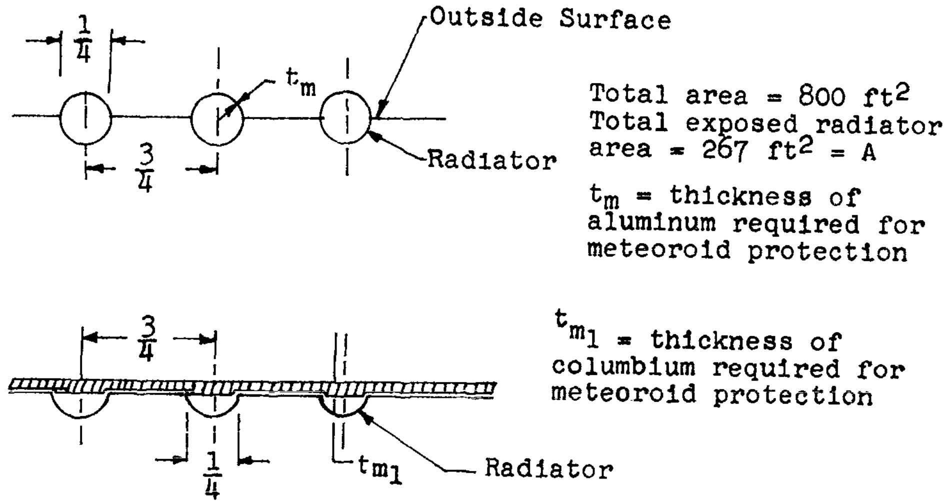

Figure 110. Radiator Designs Analyzed for Damage by Meteoroids

and made of aluminum, and (2) a configuration similar to the second one shown in Figure 110 and made of.a refractory metal such as columbium.

Based on the current most reliable meteoroidal flux data, Whipple theoretical data and Explorer Series empirical extrapolations (Reference 80) and the Bjork penetration equations (Reference 81), the two curves shown in Figure 111 can be drawn.

The Bjork equation is:

$$t = \frac{2.5 \times 10^{-4} K_T \; \bar{V}^{1/3} \; (A\tau)^{3/10}}{\left[-\mathrm{Log}_e P(0) \right]^{3/10}} \qquad \text{(L-7)}$$

where

t = Thickness required

K_T = Penetration constant

V = Average particle velocity

= Time in orbit

P(0) = Probability of no penetrations

Figure 111 shows that the required thickness of columbium is only about one half the required aluminum thickness. It is also evident that the minimum thicknesses required to insure a high probability of survival are 0.026 and 0.013 for aluminum and columbium, respectively. Since aluminum is roughly one third as heavy as columbium but only twice as critical, relative to penetration by meteoroids, the employment of aluminum where temperature conditions permit provides better protection from meteoroids for a given weight. Although the radiators located on the undersurface of the wings of Vehicles 1A, 1B, and 1C must be made of a refractory material, the resistance of various materials to meteoroidal penetration should be one of the considerations for the design of the power supply radiator for these vehicles and for any radiator contemplated for the orbital base.

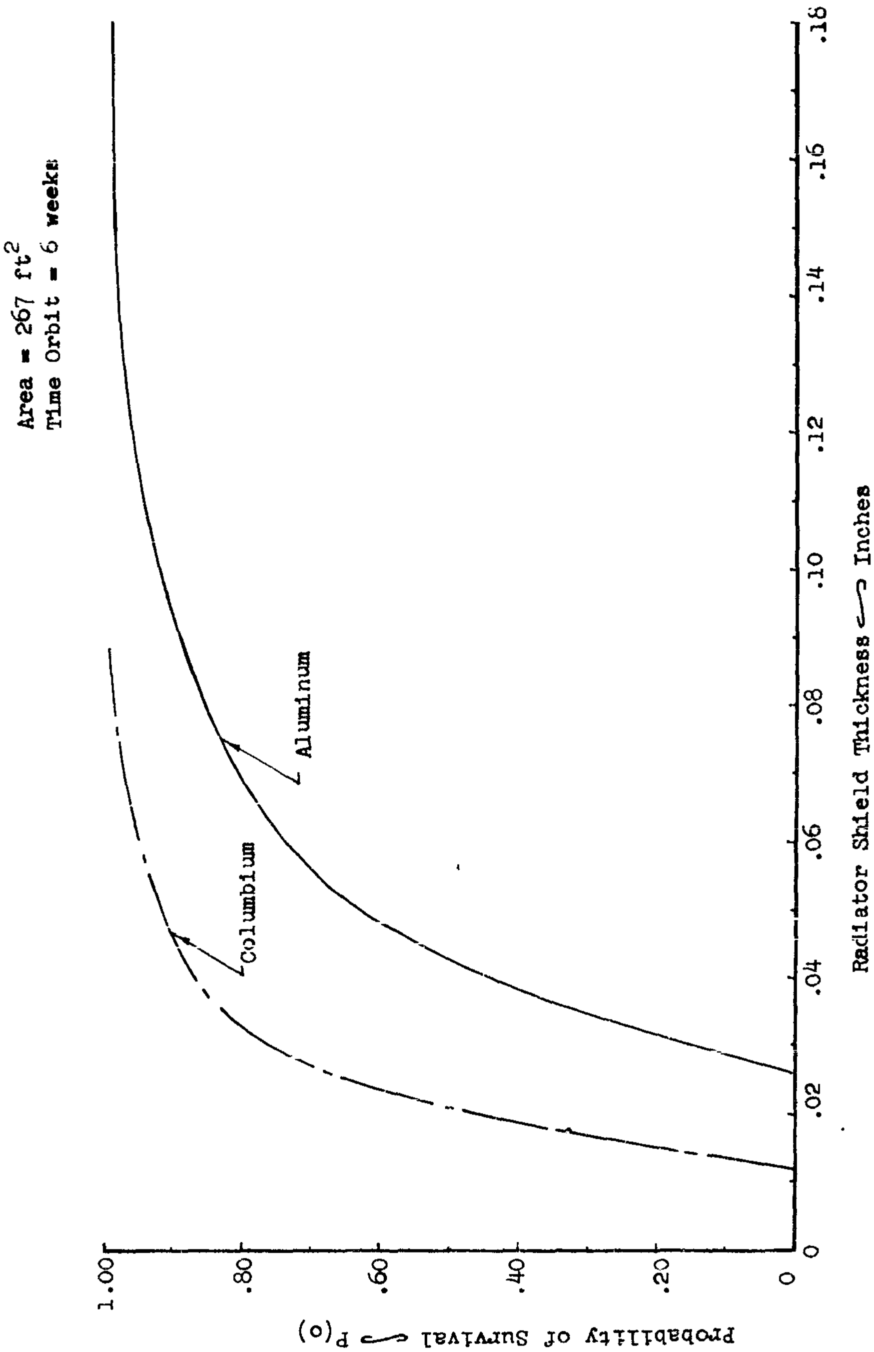

Figure 111. Probability of no Puncture of Radiator Shields of Various Thicknesses

General Aspects of Mission Equipment

This section discusses subsystem equipment for communications, guidance, reconnaissance, and data processing, which could appear in a reconnaissance vehicle operating in the span between 1965 and 1975. The equipment itself is discussed initially. A study of a similar system (SR-178) is cited as background information (Reference 50).

Communications Subsystem

The communications subsystem will contain equipment necessary for command and control, and for the transmission and reception of data. Although communication links to other satellites, as well as to ground stations, will be required, the subject of communication with other space vehicles will be deferred.

A few general comments will be made about equipment for space-ground communications. First of all, the frequencies used probably will be in the range from 1 to 10 kmc. The basic components needed for transmission and reception in this band certainly exist now. Some improvements can be made in receivers by employing parametric amplifiers for low-noise figures, and the new ceramic tubes for temperature tolerance. The size and power required of transmitters depend on a number of factors: (1) bandwidth of transmission, and (2) the combined gain of the transmitting and receiving antennas. When the orbital altitude gets in the vicinity of 20, 000 nautical miles and greater, the power dissipated during transmission at data rates of 10 megabits per second could create a thermal problem, even at assumed frequencies of 30 kmc. (See Figure 112.) Such high frequencies are probably not attainable during the early part of the time under study. Consequently, the transmitter assumed for Vehicles 1A, 1B, and 1C will probably operate at a frequency an order of magnitude less than 30 kmc. At this lower frequency, more power is required at a given altitude because of the decrease in transmitter/receiver gain and the higher bandwidth to frequency ratio. Moreover, the difference between slant range and satellite altitude is more significant at lower altitudes.

Guidance Subsystem

The guidance subsystem will contain all the necessary equipment for guidance during boost, attitude sensing, and position calculations during orbital motion, and guidance during return to a specified base back on earth. The subsystem will contain the basic portion which is normally used (an inertial platform and associated electronics, plus a digital computer for carrying out the calculations necessary) and an emergency portion which might include a horizon seeker, a periscope for sighting, and other aids.

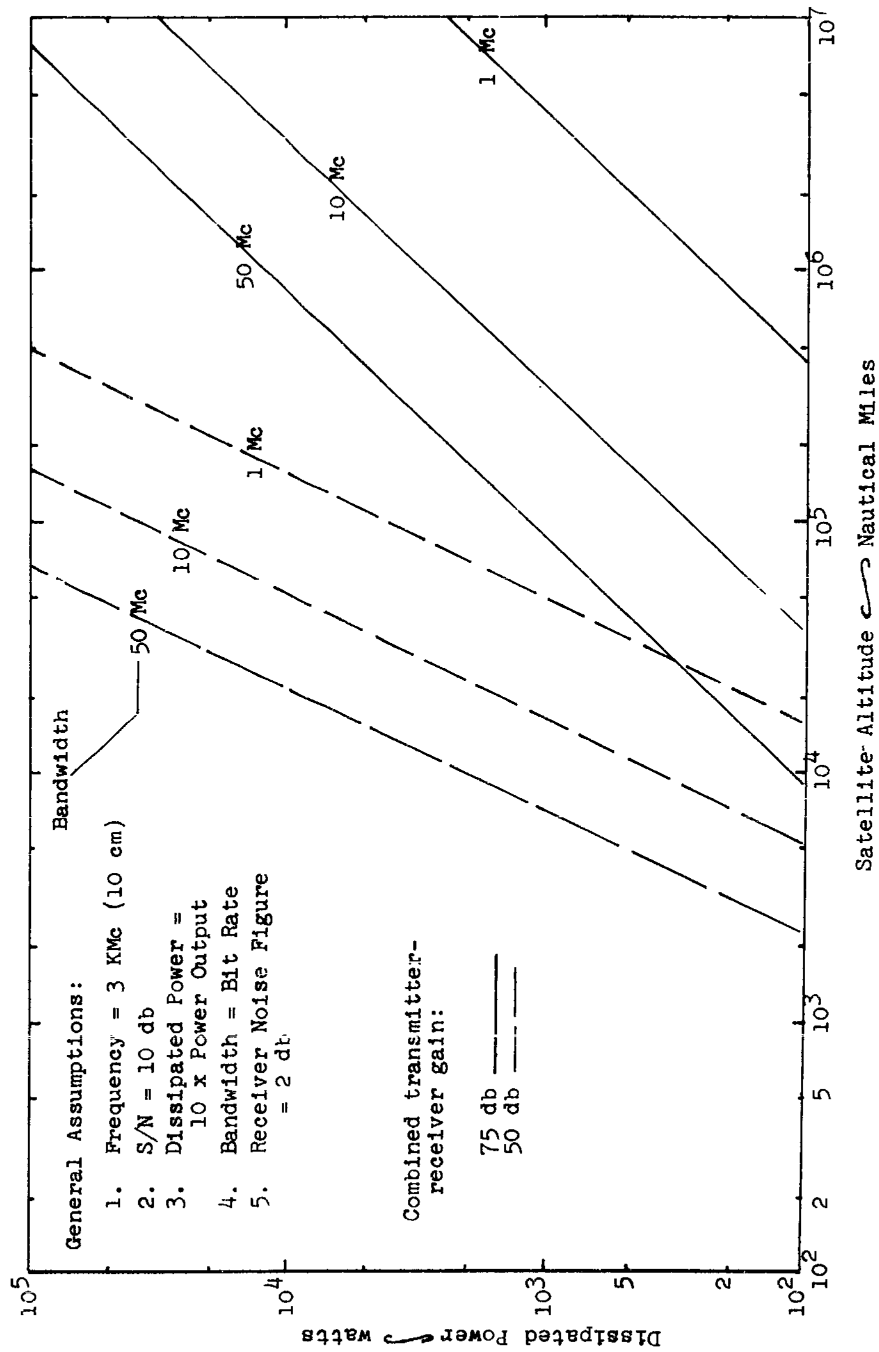

Figure 112. Dissipated Power Versus Altitude for Communications Transmitters

The term "inertial equipment" means the set consisting of the sensor, usually an inertial platform, and the control electronics. Data from the inertial equipment are transmitted to a computer which calculates the position and velocity with respect to a specified set of coordinates. The computer also sometimes calculates certain correction terms which are fed back to the inertial equipment.

The critical elements in inertial equipment are the gyroscopes and the accelerometers, and a considerable effort is being expended by both the Government and the electronics industry to improve them. There has been a continued effort to reduce the drift rate of gyros, principally by lessening the friction about the various axes about which motion takes place. Friction about the axis defined by the rotor was first reduced by air lubrication, which replaced the conventional ball bearings. Another scheme is electrostatic suspension of the rotor in a vacuum. This gyro has proven successful in the laboratory, and will first find application in the guidance of submarines. The large control equipment associated with this gyro presently precludes its use in spacecraft. Another method for suspending the rotor in a gyro is by levitation obtained by a magnetic field and the Meissner effect. This effect can only operate when the surface of the rotor is brought to cryogenic temperatures, usually involving liquid helium, the exact temperature depending upon the material of the rotor surface. Experiments with this device are in progress. A contract now exists to build a gyro of this kind with a random drift rate not to exceed 0.0001 degrees per hour, averaged in the sense of the root-mean-square (RMS).

Current inertial equipment take the form of stabilized platforms. There has been a sustained effort to reduce their weight and volume. A platform (30 pounds) representative of the current state of technology is the P-200, being produced by Litton Industries. Such an inertial platform, being a precise piece of equipment, must be placed in a closely controlled environment. In fact, the sensors are housed in a sealed case, filled usually with helium, and the internal temperature is controlled within 2 degrees of the nominal temperature.

Accuracy in determining attitude of a space vehicle operating over a long period of time presents a problem; therefore, periodic corrections are made with star trackers. Trackers currently available introduce an error of about 15 seconds of arc RMS. They need not be used during exit and reentry when there will be serious problems with aerodynamic heating and other effects due to the earth's atmosphere.

A new kind of inertial equipment, in which the inertial components are mounted on an unstabilized base, is being developed. This approach offers some advantages, especially in maintainability, and it may make it possible to incorporate the new gyros more easily. However, it is too early to predict their applicability in space vehicles.

In trying to predict the characteristics of a digital computer which might be available in the period concerned, the Verdan computer will be first examined. The Verdan is in production with a number of applications; namely, the Hound Dog missile, and the AN/ASB-12 bomb director set for the A3J, among others. It combines the advantages of a digital differential analyzer for speed and the general-purpose type for flexibility in programing. The memory, of the disc type, is somewhat limited in capacity. Because the transistors are made of germanium, the temperature limit is 160°F. Standard parts are employed and mounted on cards. Cooling is implemented by forced air. Its main characteristics are summarized in Table 29 (obtained from Reference 51).

A number of improvements in airborne digital computers can be expected within the near future, capitalizing on the improvements in components and packaging methods. A digital computer about which some data exist is the Miniver by Autonetics. Miniver is still in a very early stage of design(Reference 52). The method of packaging will depend on the amount of lead time given. If only a year or so is available, then the modules must make use of standard parts, similar to Dense, which is described later. If 3 years or more are available, then, integrated circuits will replace these welded modules. Some preliminary characteristics of Miniver, using this preferred method of packaging, are shown in Table 29.

A digital computer built around the micromodule is in the prototype stage (Reference 53). All available data are given in Table 29. This computer is a part of an inertial guidance package, the whole set weighing 45 pounds, occupying 0.82 cubic foot, and consuming 225 watts of electrical power. Displays and controls are included in the package.

Reconnaissance Subsystem

The reconnaissance subsystem will have within it photographic, infrared scanning, side-looking radar, and ELINT equipment. This subsystem will be closely integrated with the displays, the data processing subsystem, and the communications subsystem. The power of the crew members to edit and evaluate information will be used to reduce information transmitted out by several orders of magnitude.

Discussion of the infrared scanning equipment, side-looking radar equipment, and ELINT equipment will be deferred. A preliminary estimate of weight versus altitude for this type of equipment is given in Figure 113.

Photographic equipment includes not only the cameras, but also the film processing equipment, and other associated control equipment as well. A gradual evolution has resulted in lenses with long focal lengths and high resolving power, fast films, film processing techniques which can be

TABLE 29
CHARACTERISTICS OF VERDAN, MINIVER, AND RCA MICROMODULE COMPUTERS

	Verdan	Miniver	RCA Micromodule Computers
Type	Serial, general-purpose digital computer combined with a serial digital differential analyzer.	Serial, general-purpose digital computer combined with a serial digital differential analyzer.	Digital differential analyzer.
Word Length	26 bits, including sign, spare, and synchronizing bit.	27 bits, including sign, space, and synchronizing bit.	Probably 14 bits.
Storage Medium	Magnetic drum rotating at 6,000 rpm. Holds 1664 words; 1024 words for the general purpose section, and remainder for the analyzer portion.	Magnetic disc rotating at 6,000 rpm. Holds 3,737 words of which 2048 is for use by the general purpose section.	Ferrite core type. Capacity unknown, but sufficient for navigation or ICBM guidance.
Pulse Repetition Frequency	332.8 kilopulses per second.	345.6 kilopulses per second.	Unknown.
Weight	82 pounds	Approximately 25 pounds, including power supply.	4 pounds, exclusive of power supply.
Volume	1.39 cubic foot, 15-3/8 x 8 x 19-9/16 inches	Approximately 0.3 cubic foot	0.05 cubic foot
Power	70 watts, 115 volts, 3 phase, 400 cps, regulated within 0.01 percent	Roughly 80 watts, 28 volts DC	12.5 watts
	250 watts, 208 volts, 3 phase, 400 cps regulated within 5 percent.		
Components	1,500 transistors (germanium) 10,000 diodes, (germanium) 670 capacitors 4500 resistors	Integrated	600 micromodule units containing 1500 transistors, 600 diodes.
Cooling	Forced Air	Undetermined at present	Each module is surrounded by a corrugated metal sleeve which served both as a shock absorber and a means of transferring heat to a heavier metal frame.

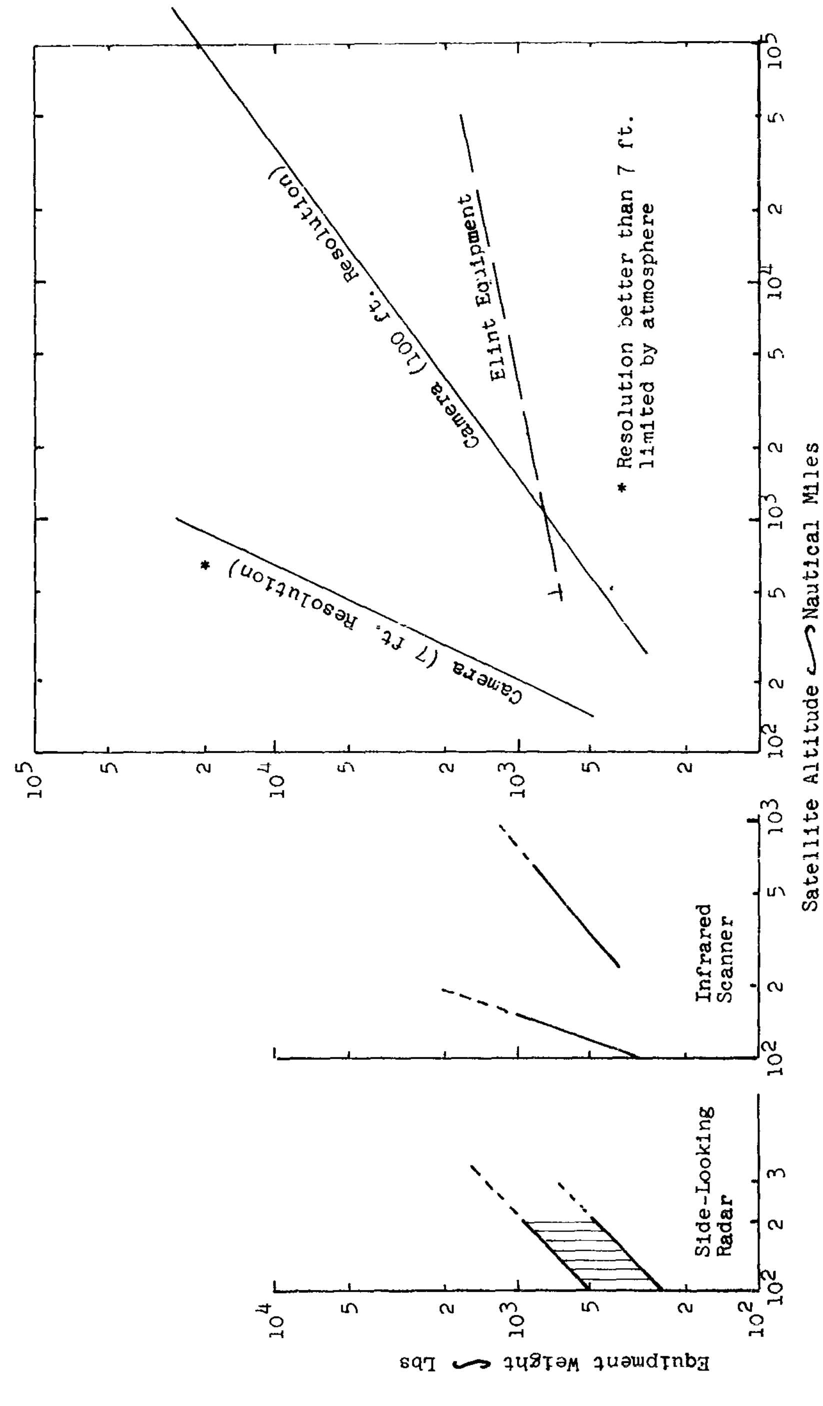

Figure 113. Weights of Mission Equipment Versus Altitude

completed in a few seconds, and many other refinements. More development in cameras will be necessary before reconnaissance from satellites at high altitudes will be practical. Figure 113 indicates the weight of the camera required for resolutions of 7 feet and 100 feet from various altitudes.

The photographic equipment considered here consists of a panoramic camera with a focal length of 96 inches, a framing camera with a focal length of 48 inches, and rapid processing equipment. Stabilization is achieved by controlling the attitude of the vehicle. Signals for image motion compensation will be computed from navigational data.

A controlled environment must be provided. Ambient temperature around the camera must be in the range from 50°F to 70°F, and controlled within 1°F; ambient pressure must be in the range from 2 to 14 psi, and controlled within 4 mm Hg; and humidity must also be controlled. Light transmitting surfaces must be kept free of fogging. The temperature gradient across the lens is critical, and temperature extremes over the lens must not exceed a fraction of a degree if high resolution is to be obtained (Reference 34). The total heat dissipated will not exceed 700 watts or so, and no difficulty is expected in disposing of it. Film processors may introduce small quantities of acidic fumes and water vapor to complicate the control problem. Unexposed film can be stored in sealed cans in an environment not exceeding 120°F, preferably at about 70°F. Higher temperatures will cause a partial change of state. The minimal temperature at which film should be stored is -20°F, at which point the base becomes brittle. The relative humidity should range between roughly 40 and 60 percent. If the humidity falls too low, then there is a problem with the generation of static electricity. If it gets too high, the film may become sticky.

Data Processing Subsystem

The broad, general, function of the data processing subsystem is to assist the crew in interpreting the tremendous rate of information (in the order of 10^{11} bits per day, mostly from the photographs) coming in through the sensors. From this, it is clear that depending on the size of the crew or number of interpreters on board, automatic aids will be necessary.

As considered here, the on-board data processing subsystem consists of three classes of equipment: digital computers, the information library with a means of making corrections and additions, and the display and control equipment. No data processing equipment of any consequence is known to be in an aircraft, and certainly not in a satellite. Hence, there is no experience from which to extrapolate. However, there are under development several ground-based equipments from which much can be learned; there is the data processing equipment for SAMOS (subsystem "I"), and similar equipment is being developed for air traffic control or for data retrieval. On this basis, it is possible to say that a digital computer having a very large memory is a necessity, and that the library of information probably will consist of a large number of indexed photographic film records, rapid processing equipment for modifying and viewing films or negatives, etc. An analysis has been made of this problem (Reference 54).

General Thermal Considerations of Electronic Equipment

The fact that heat is removed by means of space radiators in satellites, and that these radiators have an efficiency proportional to the fourth power of the absolute temperature, has brought about a desire that equipment aboard operate at as high a temperature as practical. However, the design of electronic equipment, which dissipates a large fraction of the heat, to withstand high temperatures depends to a large degree on the components used, the method of packaging them, and the technique used in cooling assembled circuits; their interrelations are complex.

It is becoming increasingly more obvious that, as time passes, it will become impossible to consider components, packaging, and cooling as three distinct, separate problems. Although, for equipment currently under design, the packaging method will accommodate standard parts, future highly dense methods will demand parts designed especially for this construction. In a few years, parts will be so integrated with circuits that they will be difficult to identify as separate entities. And, molecular electronics will make such thinking almost meaningless. As this trend to denser packaging progresses, the problem of cooling will become more acute, necessitating, perhaps, new developments in parts which operate with only microwatts of power.

While the concept of discrete components or detail parts is obsolescent, nevertheless, a discussion is warranted for at least two reasons: (1) some form of these parts will be employed in circuits and modules for equipment being designed within the next few years; and (2) the basic technology and skills used in fabrication of some parts, notably those of the solid-state variety, will be passed onto the building of future circuits.

Standard parts of today, even though they have been constantly improved over the years, are mostly unsatisfactory, both from the standpoint of reliability or failure rate and tolerance to high temperatures. The variation of failure rate as a function of temperature of operation, with the ratio of the actual to the rated watts or voltage as a parameter, is indicated in charts found in Figure 114 for transistors, diodes, resistors, and capacitors. The high failure rates of standard parts became intolerable in the development of guidance equipment for ICBM's, and a very substantial effort is being made to reduce this rate by a factor of roughly 100, at least at room temperature. There is a sporadic effort, it seems, by the industry to build parts capable of operating at elevated temperatures, comparable to those which silicon

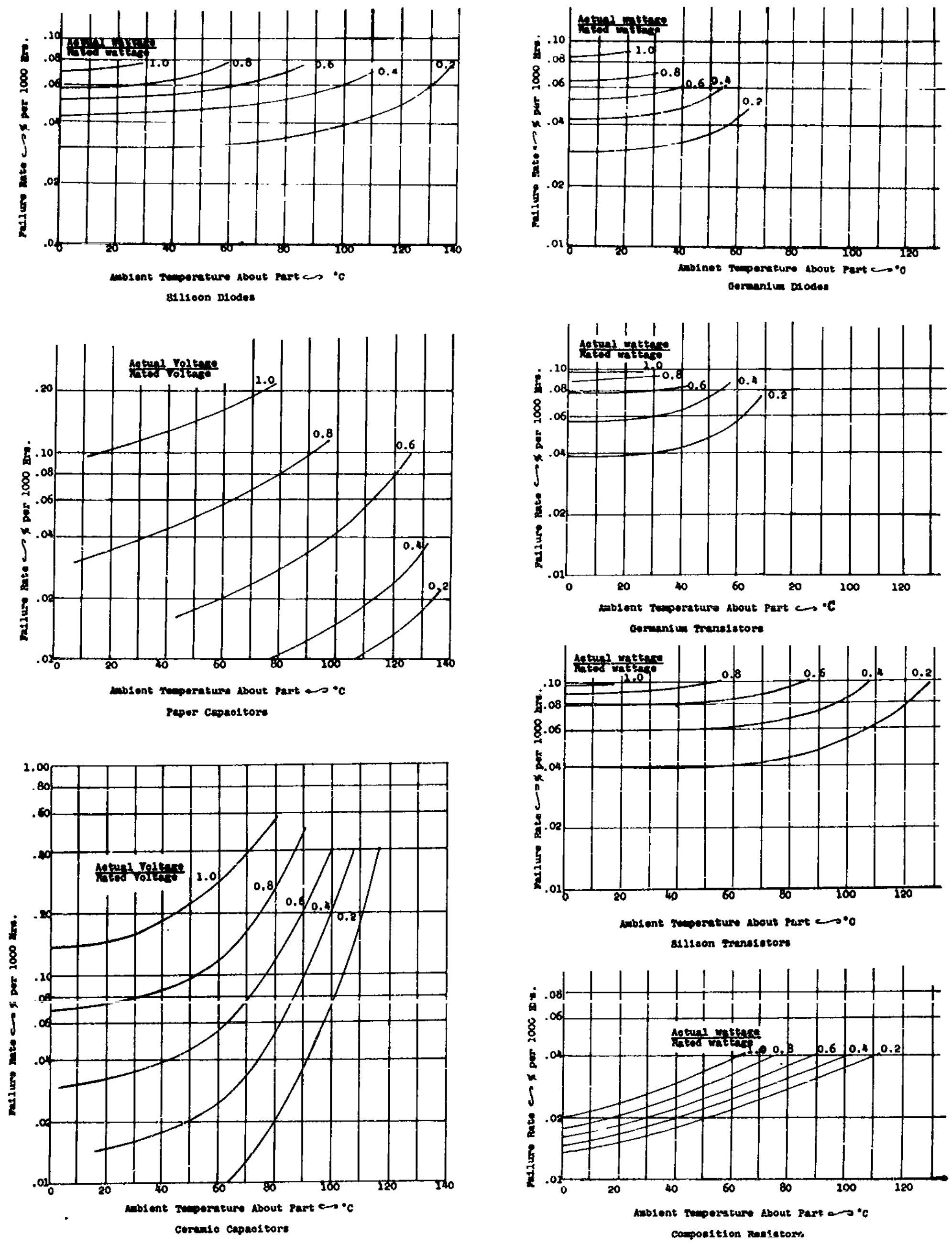

Figure 114. Failure Rates Versus Ambient Temperature for Various Electronic Parts

transistors can withstand. In particular, effort is being expended to build parts working at temperatures on the order of 752°F (400°C) for special applications.

Several new lines of standard parts are being developed so that they fit, in terms of their configuration, with new packaging schemes. Data on the performance of such parts are scanty, and nothing has been reported about their failure rates.

A ceramic vacuum tube of interest in certain applications is being produced in limited quantities. It is built up from layers of ceramic (alumina or beryllia) rings and planar control elements bonded together. These tubes have many desirable features: reasonably small size, resistance to radiation, wide frequency response, resistance to vibration and shock, and tolerance to high temperatures. Their life is becoming comparable to solid-state devices, so that the tubes can be permanently wired into circuits. One of their strongest points is their ability to operate in a stable manner with bulb temperatures in the range from 600°F to 900°F. Inductances in the leads and transit time of electrons become problems in the vicinity of 5000 mc. The main difficulty in fabrication is the seal between the ceramic and metal to hold a high vacuum over the wide temperature range. General Electric has designed integrated circuits around a "filamentless" version of these tubes, called TIMM (for thermionic integrated micromodule). The circuits must be controlled at about 1076°F.

Transistors are the key element in all other methods of packaging, with the exclusion of molecular circuits, and so a review of their status is in order.

Most of the current intensive work in the development in the field of transistors is in the direction of carefully improving standard types. Some improvements being made are (1) to increase their power handling capability at high frequencies; (2) to decrease the failure rate; (3) to fabricate them with greater uniformity and lower cost; (4) to reduce their operating power, in some instances below one milliwatt; (5) to design them so that dissipated heat can be transferred more easily; and (6) to increase their temperature tolerance. Although germanium units have the edge in frequency response, the trend is to silicon for several reasons. In the first place, silicon is one of the most abundant chemical elements on the crust of the earth, and thus, cheaper in the long run. Another is that silicon is better suited for use in integrated circuits, the oxide of silicon being stable and impervious. Finally, the temperature limit of silicon transistors is significantly higher.

Much confusion exists about the maximal temperatures at which transistors will operate. First of all, there is a maximal theoretical value, the temperature of the junction. For germanium and silicon units, this temperature is evidently above 100°C (212°F) and about 300°C (572°F), respectively. But, in practice, there are other limitations, particularly with silicon. For example, solders used on a silicon base impose a limit of about 225°C. The ratio between the actual and the rated power dissipation is another practical limitation.

The minimal transistor temperature stated in the literature is -65°C.

New materials are being discovered for the manufacture of transistors having a still greater tolerance to temperature. Gallium arsenide permits a maximum junction temperature of 400°C (752°F), which could yield a welcome margin of safety in densely packaged equipment. Further, transistors made of this material have a desirable feature of having a high-frequency response similar to that found in germanium units. Silicon carbide transistors withstand at least 600°C (1112°F), but they operate poorly at ordinary temperatures (Reference 55). The most remarkable material is boron phosphide. According to Reference 56, it is a hard, refractory substance, not attacked by corrosive agents. Transistors made of it will operate at 1800°F in an inert or reducing atmosphere, and at 1480°F in air. Transistors made of any of these materials are not yet on the market, but gallium arsenide units may be available within 2 years or so, in limited quantities. However, they will be expensive, and reliability data probably will not be found for an additional year or more.

The set of parameters associated with a transistor change with temperature, sometimes causing complications. In some applications, as in digital circuits, these variations may not be very important; but, in amplifiers such variations are important and necessitate compensating networks. Noise generated by transistors also varies as a function of temperature. This does not appear to be a serious problem with germanium transistors (Reference 57). No data on the noise generated by silicon transistors are available.

There is currently great interest in the tunnel diode, as it has the desirable features of having wide frequency response, resistance to nuclear radiation, and an ability to operate satisfactorily at high temperatures. Diodes made of germanium now operate at frequencies as high as 4000 mc. Diodes made from a given semiconductor operate at higher temperatures than transistors made of the same material. For example, silicon tunnel diodes operate at 650°F (Reference 58). However, the development of this element is still in the experimental stage, and circuits must be developed. Unless there is a requirement for extremely fast data processing in the era near 1975, no wide application is seen for the tunnel diode.

Two substantial efforts in building miniature electronic modules were started by the U. S. Army just prior to 1958. The Signal Corps contracted with industry to manufacture the "micromodule" with applications in mind for infantry communications. The Ordnance Corps, started work at about the same time in the so-called "2-D" approach. The USAF has more recently become interested in "molectronics", and has funded industry to perform research in this field. Other less advanced concepts utilizing proven components originated with specific programs for the USAF. One example is a program in connection with the guidance equipment for ATLAS. In addition to these efforts supported by the government, the electronics industry has expended much energy in arriving at their own methods of building smaller equipment. But, all too often in this race for miniaturization, there has been an apparent neglect of the cooling problem which naturally arises with the increase in density of heat sources. Also in this competition, much confusion has been introduced about the claimed density of parts for each new method of packaging.

These various methods of packaging - which can be classified as high-density packaging, micromodules, 2-D circuits, integrated circuits, and molecular electronics - are expected to be available as indicated in Figure 115. It must be remembered, however, that the method applicable will lag roughly 4 years behind the operational date of the particular weapon system. Since these methods have been described adequately and compared in the literature of their field (References 59 and 60), only a brief summary and comments will be given here.

High-density packaging can be characterized by modules which use small standard parts of uniform length and wafers on which the wiring is printed. The ends of the parts are attached to the printed wiring either by soldering or by welding. The result is a unit which resembles a sandwich.

A particular manufacturer is manufacturing soldered modules in production quantities for circuits in a digital computer. The claim is made that the density of parts is approximately 350,000 per cubic foot, and that the modules can be operated at 120°C. Few details are currently available.

The Autonetics Division of North American Aviation, Inc., has fabricated some similar modules of sandwich construction, but with welded connections. These have been called Dense for densely packaged encased standard elements. Encapsulation is not necessary. Density is about 200,000 parts per cubic foot. Production of such modules could be initiated probably within a few months.

Welded modules are attractive because thev could largely eliminate unreliable connections. But, there are some problems still extant at present. The fact is, that there is no standard for leads on parts. It is hoped that the informal committee established to resolve such problems will be able to make some definite headway with the member companies.

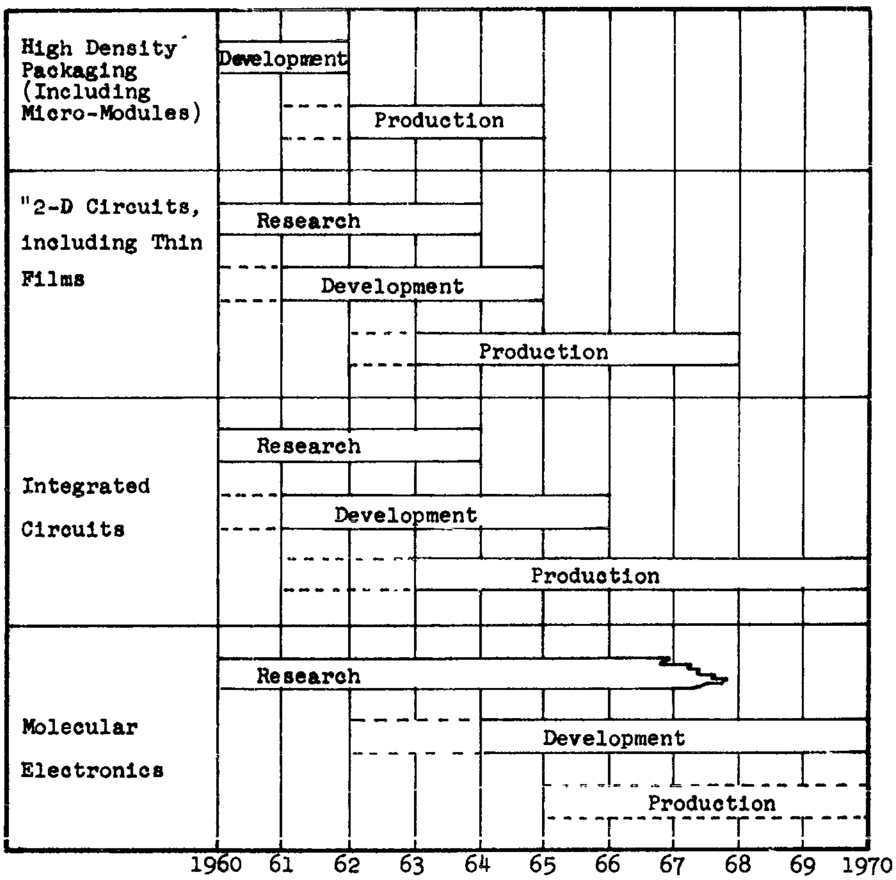

Figure 115. Anticipated Schedule for Availability of Microminiaturization Techniques

Under contract to the Army Signal Corps, work started early in 1958 to design and to fabricate the basic wafer-like elements and modules constructed from these "micro-elements". Refer to References 61 through 68 for progress reports. Although the initial application was for a helmet radio, other modules were produced for other equipment, the "Micropac" digital computer is an example.

A brief description of the modules follows, and numerous sources of further data exist in the published literature. The ceramic wafer, on, or in which, the elements are placed, is a square, nominally 0.31 inch on a side and 0.1 inch in thickness. Special parts having the desired shape and electrical properties were developed during the program, with the aid of a large segment of the electronic parts industry. A group of these micro-elements are stacked and combined into the desired circuit by soldering appropriate connections on the sides. The stack is then encapsulated, and the resulting cubic unit is called a micromodule. Such construction leads to densities of approximately 250,000 parts per cubic foot. One of the problems with this approach is how to build special transistors which involve hermetic seals to ceramic. The modules can be operated in an environment ranging from -67°F to 180°F.

Although a few micromodules for application in communication gear and digital computers are available, a more general usage is not expected to take place for about 3 years.

The Ordnance Corps took another direction to that taken by the Signal Corps. Both organizations started their development at about the same time. Instead of stacking elements, a number of them are placed on a single wafer, and interconnected. This results in so-called 2-D circuits. Some of the parts are deposited, but others, such as transistors, diodes, inductors, and large capacitors and resistors, are of the standard variety. An interesting note is found in the report of the symposium held on microminiaturization at Diamond Ordnance Fuse Laboratories (DOFL). According to the report, work was begun on the development of germanium transistors without cases, but it was found that their characteristics would change with exposures of only a few seconds to nearly pure oxygen, a small percent of ozone, a high relative humidity, or a slight amount of chemical vapors. The principal advantage of DOFL's approach seems to be that standard reliable parts can be used.

Several companies are now working on thin film 2-D circuits. A new "flip-flop" exists which is about 0.5 by 0.5 inch and only a few milli-inches thick. It is formed by successive vacuum deposition of conductive, resistance, and dielectric material, through masks, onto a glass base. Two uncased transistors are then soldered into the wiring. A flip-flop operating at 100 kc is currently available, and one operating at 10 mc has been built. A process

The limitation in temperature is imposed by the transistors. Circuits of this kind can be expected to supplant the earlier 2-D type, but at present, only a few circuits are available.

The term "integrated circuit" means electronic units, which perform certain functions, built by starting with a common base from which passive and active elements are constructed by employing various tricks. Resistances are simple to construct, since the bulk properties of the basic semiconductor can be used directly. Capacitors can be formed in various ways - for example, by oxidizing a part of the area of the base and plating the same area with metal. Inductances, especially those of large value, are a serious problem. Transistors and diodes can be fabricated on the base by any of the available techniques, such as alloying or diffusion. These elements are then interconnected, often through the semiconductor itself, or by plating metal. Contacts for external connections are made by first plating on the semiconductor and then forming an alloy by heat treatment. Leads are then attached to the contacts by thermal compression bonding. The unit is hermetically sealed in much the same manner that transistors are now packaged. A discussion by Lathrop and others can be found in Reference 69.

The ultimate temperature that these units can withstand is determined by the bonded connections made at 572°F. No standard method seems to have been established for cooling large numbers of these circuits, although some idea of the magnitude of the problem is indicated in the reference cited above. The thermal problem probably will determine the maximum density of detail parts.

Only a few circuits have been fabricated, but there is a program to develop a complete set for a digital computer. Eventually, these circuits will replace the high-density modules.

Molectronics is a radically new approach to the design of circuits, which exploits phenomena occurring within or between domains of molecules in the solid state. This work is still in the research stage, although a few devices, such as amplifiers, have been built to indicate its feasibility. It is far too early to predict the applicability of these techniques to equipment which could appear in vehicles within the time span concerned.

Cooling electronic equipment has become a very serious problem for a number of reasons. First, there is a trend to a high density of parts, and thus of heat sources. To illustrate this point, assume that 300,000 parts, half of which dissipate 10 mw, are packed in a cubic foot. Then the heat generated amounts to a total of 1500 watts. Temperatures will reach high values inside quickly unless efficient cooling is provided. Second, the maximum temperatures at which parts will operate with required reliability are [illegible]ated earlier, high-temperature components are in development, but much testing is necessary before they can be used with confidence. Third, the thermal conductivity of insulators and encapsulating compounds is low.

Cooling by forced convection - a scheme compatible with high-density packaging - does not appear to be satisfactory, because the heat generated in causing the air flow is a significant fraction of the heat to be transferred. A far more attractive method is to transfer heat by circulating a liquid between an exchanger in the equipment and the sink. This is basically compatible with micromodules, and perhaps with the still more dense methods. The penalties of increased equipment weight should be examined.

Atmospheric Contamination by Equipment

Some possible sources of contaminants associated with equipment selected for Vehicle 1A were itemized earlier in Table 25. Equipment contaminants are discussed here in a general way

Investigation of equipment contamination of the atmosphere of manned space vehicles suffers from lack of empirical data. Submarine experience may not be as applicable as it might be suspected, since submarines and space vehicles involve different types of equipment and different methods of atmospheric control. Nevertheless, submarine experience may be indicative of the probable magnitude of the problem. Tables 30 and 31 indicate a large group of substances which have been identified as part of submarine atmospheres (Reference 70). Table 32 may serve as a reference for indicating the maximum allowable concentrations of some common substances (Reference 71).

In what follows, it will be assumed that all equipment which makes use of toxic or corrosive gases and solutions will be designed to be normally free of leaks, as far as is practical. Even though these leakage rates are very low in terms of customary design criteria used in aircraft, the accumulation during long missions may cause trouble. Pollutants arising from faults in electronic equipment were considered only in a preliminary manner.

Photographic Processing Fumes

One of the more troublesome problems may arise from the fact that a large amount of photographic equipment is carried aboard. The exposed film must be processed rapidly for inspection analysis. Also, some of the other sensors, for example, the infrared scanner, side-looking radar, and the ELINT equipment may use photographic recording with rapid processing.

The processing of film by standard methods usually involves the following steps: (1) developing, (2) arresting of development, (3) washing, (4) fixing, and (5) drying. Sometimes additional steps to obtain color reversal, not necessary here, are included. Steps (3) and (4) can be eliminated; the film can be fixed on the ground after the mission, if permanent storage is required. Acidic fumes can originate from the fixer.

The only process about which detailed information was received is called Rapidol-10 (Reference 72), marketed by Micro-Copy, Inc., of Los Angeles. This process normally requires four steps, taking a total of 15 seconds, 5 for developing and 10 for fixing, all performed at room temperature. The final step can be deleted on board, and the exposed film can be examined and stored in a stabilized condition for a period of 3 or 4 months without deterioration of the image. The only volatile substance used in the stabilizer is acetic acid.

Films with a thin emulsion, to expedite drying, exist. They have three stages: developing, clearing and fixing, and washing with liquid solutions. Development occurs at 125°F.

A number of other companies are working on various aspects of rapid processing. Some comments on the recent symposium on this subject sponsored by the Society of Photographic Scientists and Engineers are found in a recent issue of Photographic Methods for Industry (Reference 73).

Gaseous Dielectrics

Gaseous dielectrics are employed to increase the ability of wave guides to transfer power from one place to another. When high-powered microwave generators are involved, there is a problem in getting the power from the final amplifier to the antenna, because of breakdowns in the dielectric contained within the wave guide. This problem increases as the frequency is increased and as the physical internal dimensions decrease. Tables indicating the limits of power as a function of wave length for standard wave guides filled with air are given in Reference 74.

Various dielectrics, including sulfur hexafluoride and other inert gases, can be introduced in lieu of air. Because of the interest exhibited by radar manufacturers in sulphur hexafluoride, some of its characteristics will be presented. It is an inert gas at ordinary temperatures, with a sublimation point at -63.8°C (-19.3°F), and density of 0.387 pound per cubic foot under the standard conditions of 1 atmosphere pressure and 70°F. Care must be taken in its handling, since it decomposes slowly under severe stresses of high temperature and electrical discharges. Also certain metals tend to act as a catalyst at about 200°C (392°F). Activated alumina and potassium hydroxide are effective absorbants of any decomposition products; leaks can be detected by a halide torch or an existing commercial instrument.

As far as its electrical properties are concerned, the dielectric constant is 1.00191, independent of frequency, at 27.5°C and pressure corresponding to 708 millimeters of mercury. Its dielectric strength is more complicated to indicate, since the breakdown voltage is a function of the shape of the electrodes and frequency of the voltage. At any rate, it is more than twice that of air or nitrogen at 1 atmosphere pressure, and this ratio increases with pressure. The power handling capacity of a wave guide is increased by a factor of 7.5:1 when air is replaced by this material. Further details may be found in Reference 75.

Cryogenic Coolants

Cryogenic coolants are being used frequently for refrigerating sensors such as infrared cells, masers, and even some computer elements and special gyros. Some of these applications are reviewed in Reference 76.

Many coolants are used, the more common being liquid nitrogen, neon, hydrogen, and helium with boiling points at 77.3, 27.2, 20.4 and 4.2°K, respectively, pressure being 1 atmosphere. Liquid nitrogen is often used to cool infrared cells, one of the most likely applications. More recent wide-band infrared detectors such as the ZIP (zinc impurity photodetector) unit, require cooling by liquid helium. The purpose in cooling sensors to such low temperatures is to improve their sensitivity by reducing internal noise. Some information about these detectors and cryostats may be found in a series of articles in Reference 77.

Although leakage of these refrigerating agents may cause a problem of replenishment, fortunately there is no known damage which could be inflicted on the crew from these, with the exception of hydrogen, which could burn. Possible leakage of cabin refrigerants could be a more serious problem if a substance such as freon or ammonia is used.

Cryostats, especially for helium, are inefficient and could pose a cooling problem.

Other Equipment Sources

Inert gases are utilized, at times, to provide a very special environment for critical equipment. For example, helium is placed in the environmental housing for a stellar-inertial platform.

Other gases could be created by the action of certain equipment. For example, ozone created by arcs from electrical equipment or the possible influence of natural radiations on the internal atmospheric gases can have deleterious effects on both equipment and crew.

Production of ozone and nitrogen oxides by direct or secondary radiation is probably not a problem at the radiation levels which are tolerable for human beings. However, these toxic compounds can be formed with photons of an energy level otherwise not dangerous to humans. Preliminary investigation indicates that there are reactions which could produce a troublesome equilibrium level, if there is sufficient secondary radiation energy flux. Although it appears unlikely that radiation production of ozone or nitrogen oxides is a problem, further investigation is necessary before the problem can be dismissed completely.

In addition to the obvious contaminants introduced by the crew, food, waste products, and equipment, all supplies brought on board will have to be screened carefully. Volatile compounds such as lubricating oil, glue, cleaning fluids, lighter fluid, shaving lotion, and other substances could contribute contaminants which could be hard to eliminate in a regenerative atmospheric control system.

TABLE 30

COMPOUNDS QUALITATIVELY IDENTIFIED IN TRACE AMOUNTS IN SUBMARINE ATMOSPHERES

Material	Chemical Formula	Type Submarine (F=Fleet; N=Nuclear)	Maximum Acceptable Concentrations* in ppm (ACGIH)
Arsine	AsH_3	F	--
Benzene	C_6H_6	N	25
1-3 Dimethyl-5-ethylbenzene	$1,3\text{-}(CH_3)_2\text{-}5\text{-}C_2H_5C_6H_3$	N	--
Ethylene	C_2H_4	N	--
p-Ethyl Toluene	$1,4\text{-}CH_3C_6H_4C_2H_5$	N	--
Freon-114	CF_2ClCF_2Cl	N	1000
"Gasoline vapors"	---------------	F	500
Hydrogen Chloride	HCl	N	5
Mesitylene	$1,3,5\text{-}(CH_3)_3C_6H_3$	N	--
Propane	C_3H_8	N	--
Pseudocumene	$1,2,4\text{-}(CH_3)_3C_6H_3$	N	--
Sulfur Dioxide	SO_2	F, N	5
Toluene	$C_6H_5CH_3$	N	200
o-xylene	$1,2,\text{-}(CH_3)_2C_6H_4$	N	200
m-xylene	$1,3\text{-}(CH_3)_2C_6H_4$	N	200
p-xylene	$1,4\text{-}(CH_3)_2C_6H_4$	N	200

*M. A. C. either not applicable or not established for materials listed with a dash in the column.

TABLE 31

COMPOUNDS QUANTITATIVELY IDENTIFIED IN SUBMARINE ATMOSPHERES

Material	Chemical Formula	Type of Submarine (F=Fleet; N=Nuclear)	Highest Concentration Normally Found	Maximum Acceptable Concentration* in ppm (ACGIH)
Acetylene	C_2H_2	N	0.5 ppm	--
Ammonia	NH_3	N	>1 ppm	100
Carbon Dioxide	CO_2	F, N	1.1% By Volume	5000
Carbon Monoxide	CO	F, N	38 ppm	100
Chlorine	Cl_2	F, N	1 ppm	1
Freon-12	CCl_2F_2	F, N	70 ppm	1000
"Hydrocarbons" (other than CH_4)	"HC"	F, N	25 ppm	--
Hydrogen Fluoride	HF	N	0.3 ppm	3
Hydrogen	H_2	F, N	1.75% By Volume	--
Methane	CH_4	F, N	118 ppm	--
Methyl Alcohol	CH_3OH	N	6ppm	200
Monoethanolamine	$HOCH_2CH_2NH_2$	N	<1 ppm	--
Nitrogen	N_2	F, N	80% By Volume	--
Nitrogen Dioxide	NO_2	N	0.1 ppm	5
Nitrous Oxide	N_2O	N	27 ppm	--
Oxygen	O_2	F, N	20% By Volume	--
Stibine	SbH_3	F, N	1 ppm	0.1
Water Vapor	H_2O	F, N	60% R. H.	--
"Cigarette Smoke"	--	F, N	0.4 mg/liter	--

*M. A. C. either not applicable or not established for materials listed with a dash in this column

+ American Conference of Government Industrial Hygenists

TABLE 32

MAXIMUM ALLOWABLE CONCENTRATIONS OF COMMON INDUSTRIAL SUBSTANCES

SUBSTANCE	MAC	SUBSTANCE	MAC
Dusts:			
Asbestos (0.5 to 10.0 μ)	5	Hydrogen cyanide	20
Cement	15	Hydrochloric acid	10
Organic	50	Hydrogen fluoride	3
Pottery	4	Hydrogen sulfide	20
Silica (25-35% SiO_2)	10	Methanol	200
(0.5 to 5.0 μ)		Methyl bromide	30
Silica (75% SiO_2)	5	Methyl chloride	500
		Monochlorbenzine	75
(0.5 to 5.0 μ)		Naptha (petroleum)	1000
Slate	15	Nitrobenzene	5
Talc	15	Nitrogen oxides	10
Nuisance dusts	50	Ozone	1
		Phosgene	1
Gases and Vapors:		Phosphine	2
		Phosphorus trichloride	1
Acetone	200	Sulfur dioxide	10
Ammonia	400	Tetrachlorethane	10
Amyl acetate	200	Tetrachlorethylene	200
Analine	5	Toluol	200
Arsine	1	Trichlorethylene	200
Benzol	100	Turpentine	200
Butenol	50	Xylol (coal tar naptha)	100
Butyl acetate	20		
Carbon disulfide	20		
Carbon dioxide	5000		
Carbon monoxide	100	Metallic dusts and fumes:	
Carbon tetrachloride	100	Cadmium	0.1
Chlorine	1	Chromic acid	0.1
Chloroform	100	Lead	0.15
Dichlorbenzene	75	Manganese	6.0
Dichlorethyl ether	15	Mercury	0.1
Ether (ethyl)	400	Zinc oxide	15
Ethyl alcohol	250	Chlorodiphenyl	1.0
Ethyl bromide	1700		
Ethyl chloride	70		
Ethylene dichloride	100		
Formaldehyde	20		
Gasoline	1000		

Dusts: Million particles/ft^3
Gases and vapors: Parts/million
Metallic dusts and fumes: milligrams/m^3

Discussion of Assumptions

1. Cabin structural design was based upon an assumed maximum reentry skin temperature of 2600°F occurring simultaneously with stress loads corresponding to ultimate cabin pressure loading of 20 psi and panel shear of 300 lb/in.

2. Allowable radiation dosage rates of 0.005 REM/hr was assumed for the purpose of determining shielding requirements.

3. The power requirements during reentry and high altitude trajectory were based upon weight to power requirements of the X-15. These power requirements are based on direct hydraulic coupling to the load and thereafter neglect electrical conversion losses.

4. Choice of abort thrust level is based on the assumption that initial booster thrust is about 2g or less, and that the pilot has fail-safe cut-off control over booster thrust.

5. It is assumed that during reentry, a wing loading of 46 lb/ft^2 (for Vehicle 1B) will not result in unreasonable temperatures. Computations were made for wing loadings of 30 lb/ft^2 and 40 lb/ft^2.

6. Considerations relating to incidence of meteoroids are, in general, based on the assumption that observed incidence data are accurate and can be extrapolated to apply to future space flights. A specific theory of interaction of meteoroids and matter was assumed, although empirical data are lacking. Confidence levels are not attached to the stated probabilities of meteoroid impact.

7. Equipment employing toxic substances such as wave guide dielectrics were assumed to be normally free of leaks.

8. Refined optimization of the vehicle was not attempted, and was considered beyond the scope of this study.

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Determination of thermal and atmospheric control requirements necessitate examination of realistic manned vehicles. Three versions of a manned, orbital, reentry, base-point vehicle are developed for the purpose of providing tangible reference

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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).

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2. Spaceships
3. Integration optimization
4. Environmental control system requirements

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II. Contract AF33(616)-7635
III. North American Aviation, Inc., Los Angeles, Calif.
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DEPARTMENT OF THE AIR FORCE
HEADQUARTERS AIR FORCE MATERIEL COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

Recd 4/16/2001

APR 12 2001

MEMORANDUM FOR DTIC/OCQ (ZENA ROGERS)
8725 JOHN J. KINGMAN ROAD, SUITE 0944
FORT BELVOIR VA 22060-6218

FROM: AFMC CSO/SCOC
4225 Logistics Avenue, Room S132
Wright-Patterson AFB OH 45433-5714

SUBJECT: Technical Reports Cleared for Public Release (Case AFMC 00-265)

1. The following reports listed in the attached HQ AFMC/PAX Memo, 28 Dec 00, para 1.a., b., and c. were reviewed and cleared for public release in accordance with AFI 35-101, 1 Dec 99, *Public Affairs Policies and Procedures*, Chapter 15.

- AD 330051
- AD 333266
- AD B972544

2. Please direct further questions to Lezora U. Nobles, AFMC CSO/SCOC, DSN 787-8583.

Lezora U. Nobles

LEZORA U. NOBLES
AFMC STINFO Assistant
Directorate of Communications and Information

Attachment:
HQ AFMC/PAX Memo, 28 Dec 00, w/1 Atch