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FINAL REPORT

TECHNOLOGICAL REQUIREMENTS COMMON TO MANNED PLANETARY MISSIONS

(Contract NAS2-3918)

Summary Report

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

SD 67-1086

TECHNOLOGICAL REQUIREMENTS COMMON TO MANNED PLANETARY MISSIONS
NAS2-3918

Summary Report

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FOREWORD

This report summarizes the results of the studies conducted under Contract NAS2-3918, Technological Requirements Common to Manned Planetary Missions. The study was conducted by the Space Division of the North American Rockwell Corporation for the Mission Analysis Division of the National Aeronautics and Space Administration. The detailed descriptions of the study are presented in SD 67-621 which consists of the following five volumes:

Technical Summary	(SD 67-621-1)
Appendix A - Mission Requirements	(SD 67-621-2)
Appendix B - Environments	(SD 67-621-3)
Appendix C - Subsystem Synthesis and Parametric Analysis	(SD 67-621-4)
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INTRODUCTION

Several recent studies, together with studies currently in progress, have examined the requirements of manned Mars and Venus stopover missions during the early to mid-1980's. Only a limited number of studies have included a simultaneous evaluation of either the performance requirements or the system requirements of both Mars and Venus missions and more advanced manned planetary missions. A simultaneous evaluation of both the performance and system requirements is appropriate to ensure the efficient application of national resources to any manned planetary exploration program which might transpire. The objective of such an evaluation would be to determine if common requirements exist for the diverse mission objectives which might be considered during the remainder of this century. The evaluation of common requirements must include the total system requirements, the subsystem requirements, and the technology requirements of the missions.

The purpose of the study summarized herein was to perform such an evaluation and to establish potential areas of common requirements. The requirements of potential manned planetary missions are examined and potential areas of common requirements are established in order to assist in the determination of the most rewarding areas of future technological development.

Inherent in such an evaluation is the establishment of reasonable mission objectives, mission modes, and mission opportunities for future manned planetary exploration. The mission objectives which were considered during this study were Mercury, Venus, Mars, and Jupiter, the asteroids Vesta and Ceres, and Ganymede, the third Galilean satellite of Jupiter. Direct , Venus swingby, and flyby mission modes were investigated as appropriate. However, flyby missions to Mars and Venus were not considered under the assumption that these missions can be performed on the basis of near-term advances in technology. The ability to satisfy the requirements of Mars and/or Venus stopover missions using either retrobraking or aerobraking planetary capture was presupposed as a minimum capability.

The characteristics of missions which are representative of opportunities having minimum, average, and maximum performance requirements during a synodic cycle of opportunities were established for each mission objective. To ensure that such a spectrum of performance requirements was obtained, a 20-year time span was considered. The time period considered was 1980 to 2000, although the results obtained can be applied to any other period of interest.

The basic technical study was of nine months' duration and, insofar as establishing performance requirements was concerned, was restricted to the examination of circular planetary parking orbits. The circular orbit restriction was originally imposed because it was felt that elliptical capture orbits would inordinately complicate rendezvous operations and significantly increase launch-window requirements. Analyses conducted within NASA and the industry after the initiation of the study indicated, however, that only modest performance penalties are associated with such factors when elliptical planetary parking orbits are considered. Since the use of elliptical planetary parking orbits can result in significant reductions in the performance requirements, the effects of using elliptical planetary parking orbits were investigated during a three-month amendment to the basic contract.

The examination of the system requirements included the establishment of the characteristics of the modules and subsystem technologies required for all mission objectives and mission modes considered in the study. Subsystem and module weight scaling equations were developed and, together with the performance requirements, were incorporated in the overall weight synthesis analyses. To the maximum extent possible, parametric analyses were conducted to establish the most appropriate subsystems and modules for the complete family of missions. The primary evaluation criterion was initial mass in Earth orbit, although other considerations (e.g., system integration and reliability) were included qualitatively as appropriate.

To establish common requirements for the family of manned planetary missions, the total system requirements were first established assuming that the individual modudes were designed by the individual mission requirements. Common manned modules were then selected, and the effects of utilizing these modules were investigated by determining the attendant increase in the propulsion module mass requirements. Common propulsion modules were investigated by assuming fixed module characteristics and off-loading propellant as required by the particular mission. The final investigations of the use of common modules were based on the use of both common manned modules and common propulsion modules.

Because of the broad scope of this study it was necessary that certain constraints be proposed at its outset. Among the more significant are the following:

Only high-thrust propulsion systems are considered within this category: however, the applicability of both chemical (space-storable and cryogenic and nuclear solid- and gaseous-core) systems are evaluated.

The scientific objectives, associated equipment, and crew functions are not considered, although weight allocations for probes and onboard experiments are made. In addition, characteristics of all crew-related system elements include a parametric variation in crew size from 3 to 20 men.

No explicit analysis of the compatibility between the interplanetary spacecraft system and the Earth-launch vehicle is made.

Neither abort requirements nor launch-window effects are considered.

No development plans, mission plans, or cost analyses are included.

Throughout the subsequent discussion of the technology requirements, allusions have been made as to the possible implications of certain of these analyses on each of the above areas.

MISSION REQUIREMENTS

PERFORMANCE REQUIREMENTS

Basepoint missions were established for mission opportunities representative of minimum, average, and maximum total-velocity requirements for each mission objective and mission mode considered during the study. For all cases except Mercury, stay times of 0, 30, and 60 days were considered. The characteristics of the resultant basepoint missions for circular planetary parking orbits are summarized in Tables 1 through 3.

Table 1. Flyby Basepoint Mission Characteristics

Mission Objective	Mission Opportunity	Mission Duration (days)	Earth-Entry Speed (km/sec)	Total Mission ΔV (km/sec)
Vesta	1991	730	12.4	4.57
Vesta	1993	1096	14.1	6.32
Ceres	1993	1094	13.2	5.37
Ceres	1992	1096	16.0	8.30
Jupiter	1991	1380	17.7	6.80
Jupiter	1985	1035	15.2	7.10

Table 2. Direct Basepoint Mission Characteristics (Circular Planetary Parking Orbits)

Mission Objective	Mission Mode	Mission Opportunities	Mission Duration (days)	Earth-Entry Speed (km/sec)	Total Mission ΔV (km/sec)
Mercury	Retro	1988, 1990, 1992	311 to 369	15.0 to 17.0	19.7 to 24.1
Venus	Aero	1988, 1990, 1991	380 to 535	13.5 to 14.9	7.44 to 8.03
Venus	Retro	1988, 1990, 1991	380 to 585	13.7 to 15.1	10.8 to 12.0
Mars	Aero	1986, 1988, 1993	370 to 445	14.0 to 19.8	7.58 to 11.2
Mars	Retro	1986, 1988, 1993	440 to 546	15.7 to 19.8	10.7 to 15.5
Vesta	Retro	1985, 1987, 1991	720 to 755	12.3 to 15.9	14.4 to 17.8
Ceres	Retro	1980, 1989, 1992	745 to 800	19.1 to 19.8	18.7 to 22.4
Jupiter	Retro	1985, 1987, 1990	1415 to 1424	14.4 to 14.8	18.7 to 20.2
Ganymede	Retro	1985, 1987, 1990	1415 to 1424	14.4 to 14.8	16.4 to 17.8

Table 3. Venus Swingby Mission Characteristics
(Circular Planetary Parking Orbits)

Mission Objective	Mission Mode	Mission Opportunities	Mission Duration (days)	Earth-Entry Speed (km/sec)	Total Mission ΔV (km/sec)
Mercury	Retro (OS)	1985, 1986, 1988	361 to 445	13.9 to 16.4	20.0 to 23.2
Mercury	Retro (IS)	1981, 1987, 1992	380 to 422	11.6 to 12.4	22.5 to 26.4
Mars	Aero (OS)	1986, 1993, 1999	545 to 663	11.4 to 14.1	5.87 to 6.83
Mars	Aero (IS)	1982, 1988, 1995	551 to 619	12.0 to 12.2	7.28 to 8.10
Mars	Retro (OS)	1986, 1993, 1999	563 to 692	11.4 to 15.0	9.73 to 11.5
Mars	Retro (IS)	1984, 1988, 1995	555 to 635	12.0 to 12.6	9.62 to 13.6

OS = Outbound swingby.
IS = Inbound swingby.

These missions are representative of those for which initial mass in Earth orbit is minimized. It has been determined that the minimization of the total incremental velocity requirements yields an excellent approximation to such a mission-selection criterion.

AEROBRAKING TECHNOLOGY REQUIREMENTS

Aerodynamic braking to orbit about Mars and Venus is an attractive mode of decelerating the spacecraft from hyperbolic approach velocities when compared to retrobraking deceleration. The system mass-in-Earth-orbit requirements are lower, but a more complex system is required which is very sensitive to the environment, vehicle characteristics, and trajectory parameters. Additional constraints are imposed on the aerobraking vehicle by packaging, tolerable deceleration levels, and achievable navigation accuracy.

Past studies have considered some of the complex interactions between the environment, vehicle, and trajectory parameters. A promising configuration developed from these studies was employed in the present study as a baseline for parametric analyses. The results of the analyses included the aerobraking entry corridors at Mars and Venus as a function of velocity, vehicle m/C_DA, and various cut-off criteria. Heating rates and total heat loads to the vehicle were determined for the critical entry trajectories, and estimates of the required heatshield weights were made. The effects of atmospheric composition were included in the analyses.

PLANETARY EXCURSION MODULE REQUIREMENTS

The mass requirements of the planetary excursion modules are dependent upon the descent and ascent characteristic velocity requirements. The characteristic velocity requirements were determined for landings on Mercury, Mars, Vesta, Ceres, and Ganymede. The resultant requirements are summarized in Table 4 for the limiting planetary parking orbit eccentricities considered in the study. The total descent characteristic velocity requirements include the incremental velocity requirements for the initial deorbit maneuver, the powered descent, and the additional requirements for hover and translation. The total ascent requirements include the initial ascent requirements, the requirements for transfer from the burnout conditions to the parking orbit, and the final parking orbit insertion. For the elliptic orbit cases, the descent is initiated at apocenter of the parking orbit. The ascent profile consists of an initial ascent to a low-altitude circular parking orbit followed by a Hohmann transfer and tangential injection at pericenter of the parking orbit.

Table 4. Planetary Excursion Module Characteristic Velocity Requirements

Mission Objective	Descent ΔV (m/sec)		Ascent ΔV (m/sec)	
	e = 0	e = 0.7	e = 0	e = 0.7
Mercury	3830	4640	4000	4850
Mars	1220	1080	4880	5915
Vesta	328		328	
Ceres	556		465	
Ganymede	2470	3020	2700	3270

GUIDANCE AND NAVIGATION REQUIREMENTS

The investigations of the guidance and navigation requirements consisted of an examination of the requirements for injection into orbit about Ganymede. The incremental velocity requirements for injecting into orbit about Ganymede are minimized by a direct orbit-injection mission profile. The alternative is to initially inject into a phasing orbit about Jupiter and then perform an orbit transfer and Ganymede orbit-insertion maneuver. The objective of this guidance and navigation analysis was to determine if the differences in the midcourse guidance requirements for the two modes would influence the selection of the mission mode. It was determined that the midcourse-correction requirements did not affect the selection of the Ganymede orbit-insertion mode. Thus, the direct-injection profile appears to be promising and was taken as the nominal profile in the establishment of the baseline missions.

ENVIRONMENTS

During planetary missions, the space environment can have a significant effect on the spacecraft design or mission operation. The environmental factors which were investigated in the present study were the meteoroid environment, thermal environment, and radiation environment. Meteoroid protection must be provided for all modules and components which will be damaged by either the erosion, perforations, or penetrations which result from the impact of meteoritic particles. Thermal protection of the mission module is required in order to maintain a habitable environment for the spacecraft crew and equipment. The propulsion modules will also require thermal protection to either limit propellant boil-off or, in some cases, to prevent propellant freezing. Protection against natural radiation applies primarily to the spacecraft crew, and is required to keep the total mission dose below acceptable limits.

METEOROID ENVIRONMENT

The meteoroid-protection requirements were expressed as a set of scaling equations which define the optimum shield weight for each module as a function of mission objective, mode, duration, and module vulnerable area. The meteoroid models included cometary flux and two levels of asteroidal flux. One potential ramification of the meteoroid environment is shown in Table 5 for a mission to Jupiter and Ganymede. By comparing the initial-mass-in-Earth-orbit requirements, it is seen that the current uncertainty in the environment can result in a three-to-one variation in these requirements. By carrying out a two-plane transfer to avoid the asteroidal belt, however, the effect of this uncertainty can be effectively eliminated.

Table 5. Jupiter Out-of-Ecliptic Mission
(1990 Ganymede Mission)

Mission Mode	Meteoroid Environment	Mass in Earth Orbit (kg)
Direct	Nominal	1, 950, 000
Direct	Maximum	6, 120, 000
Out-of-the-ecliptic	(Cometary Only)	2, 120, 000

THERMAL ENVIRONMENT

The mission module insulation system requirements were established on the basis of minimizing the effects of external heat sources and heat sinks on the thermal balance within the module. In this manner, the environmental control subsystem (ECS) radiators, required to reject all the internal heat dissipation, could be sized for all missions at one time, with only a moderate safety factor on area to account for external heat balance factors. It was found that a single insulation thickness could be applied to the mission modules employed in all missions considered while maintaining the external heat gain or heat loss to less than 10 percent of the internal heat dissipation. It was also determined that spacecraft attitude control is not very critical for thermal-control purposes for missions to Jupiter. For missions to Mercury, either solar orientation will be required or it will be necessary to provide shadow shielding of the ECS radiator to prevent direct solar heating.

The propulsion module thermal protection requirements were determined by optimizing the trade-off between the mass requirements of the insulation system and the boil-off propellant requirements. A set of weight-scaling equations was developed which defined the optimum insulation thickness and boil-off propellant as a function of the thermal properties of the propellant and the mission characteristics. It appears that passive thermal control of the tanks will be adequate.

RADIATION ENVIRONMENT

Two separate analyses were performed to determine the effects of the radiation environment on the spacecraft design. The first investigation considered the space radiation environment which must be considered for all missions. The second investigation considered the effects of the Jupiter trapped radiation which is of concern for missions to Jupiter and its satellites.

Because of the large number of mission objectives and mission opportunities which were considered, the analysis of space radiation was carried out by developing analytical (rather than statistical) relationships between solar and mission parameters to yield mission doses. The resultant mission module shielding requirements are shown in Figure 1. Since the inherent shielding is on the order of 3 to 5 grams/centimeter^2, additional shielding will be required only for missions that occur during periods of maximum solar activity.

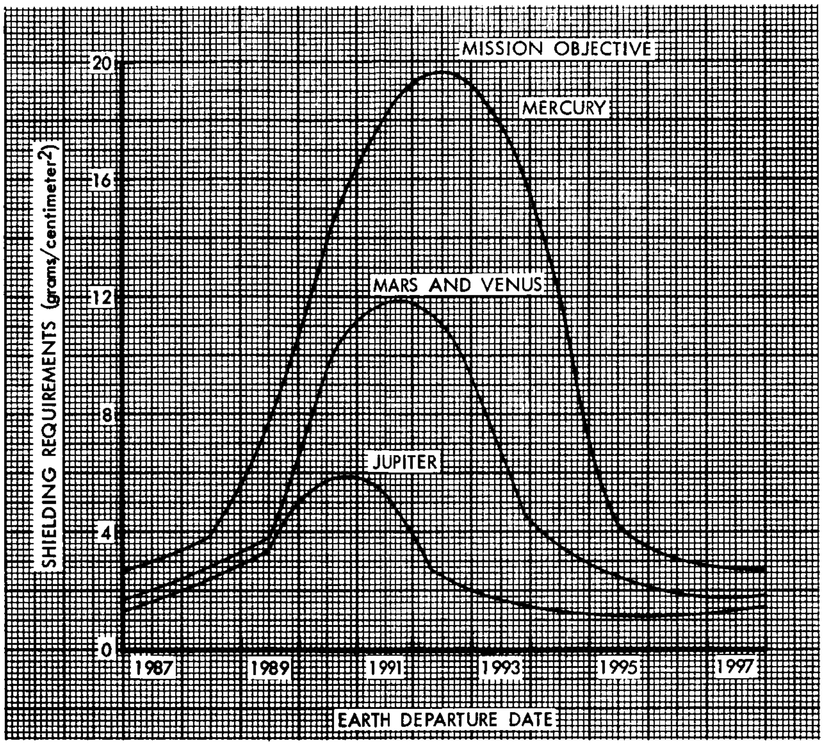

Figure 1. Mission Module Radiation Shielding Requirements

The investigations of the effects of the Jupiter trapped radiation resulted in the development of a new trapped-radiation model. The decimetric and decametric radiation make possible approximate calculations of the flux and spatial extent of trapped electrons; the corresponding quantitities for any trapped protons are matters for conjecture. Calculations were carried out which probably bracket the shielding requirements for Jupiter missions. The fluxes and dose rates associated with the mean model are such that a stopover at Ganymede appears possible, but is not clearly a desirable part (from a radiation-shielding standpoint) of a manned mission to Jupiter. As an alternative, Callisto could be considered as the target body, since the shield thickness required will be approximately a factor of two less. The results of this analyses can also be applied to the study of unmanned orbiting missions to Jupiter.

SUBSYSTEM CHARACTERISTICS

The requirements of the major spacecraft subsystems were evaluated, and the types of subsystems most appropriate for the mission objectives being considered were established. The environmental control and life support subsystem, communications subsystem, and electrical power subsystem will each have a significant influence on the mission module design. Other subsystems which are required were considered only to the extent that they contributed a constant mass to the mission module.

LIFE SUPPORT SUBSYSTEM

The weight, volume, and power requirements of three environmental control and life support subsystems, representing three degrees of closure, were established. The degrees of closure considered were: open, water recovery only, and water and oxygen recovery. The characteristics of the subsystem were represented by scaling equations, and separate equations were established for each principal element of the subsystem.

The mass requirements of the three subsystems considered are compared in Figure 2. As seen, the mass requirements for the open system are excessive. It is also seen that the mass requirements of the oxygen-only recovery system are at least 50 percent greater than the requirements of the more fully closed system. This penalty is considered to be excessive. Thus, in order to utilize a system compatible with all missions considered, the water-and-oxygen recovery system was employed during the subsequent synthesis analyses.

A limited study was also made of food-producing systems to determine their utility for the family of missions. When an allowance of stored food was provided for, it was felt that the resultant mass savings were not sufficient to justify further consideration of such systems in this study.

Due to the short occupancy times, the open system was assumed for use in the Earth reentry module and in the planetary excursion module ascent and descent stages. Although a mass advantage would accrue if a partially closed system were used in the planetary excursion module for the longer occupancy times, the magnitude of the savings does not appear to warrant the additional system complexity.

COMMUNICATIONS SUBSYSTEM

Four subsystems which span the frequency range of 2.3 gigahertz through 357,000 gigahertz were compared; namely, S-band, millimeter, CO_2 laser, and GaAs laser.

The critical parameter in the comparison of the candidate communications subsystems was considered to be the power requirements. The differences in the performance, integration, and the weight of the transmitter, receiver, and antenna will be small compared to the differences in the weight of the electrical power subsystem due to the differences in the input-power requirements. The candidate subsystems are compared in Figure 3 which shows the transmitter output power as a function of transmitting capability and antenna (aperture) diameter. Only two systems have lower power requirement than the two S-band systems. The gallium arsenide noncoherent laser and the carbon dioxide laser with one-meter apertures have the lowest power requirements; but both of these systems have extremely narrow beam widths

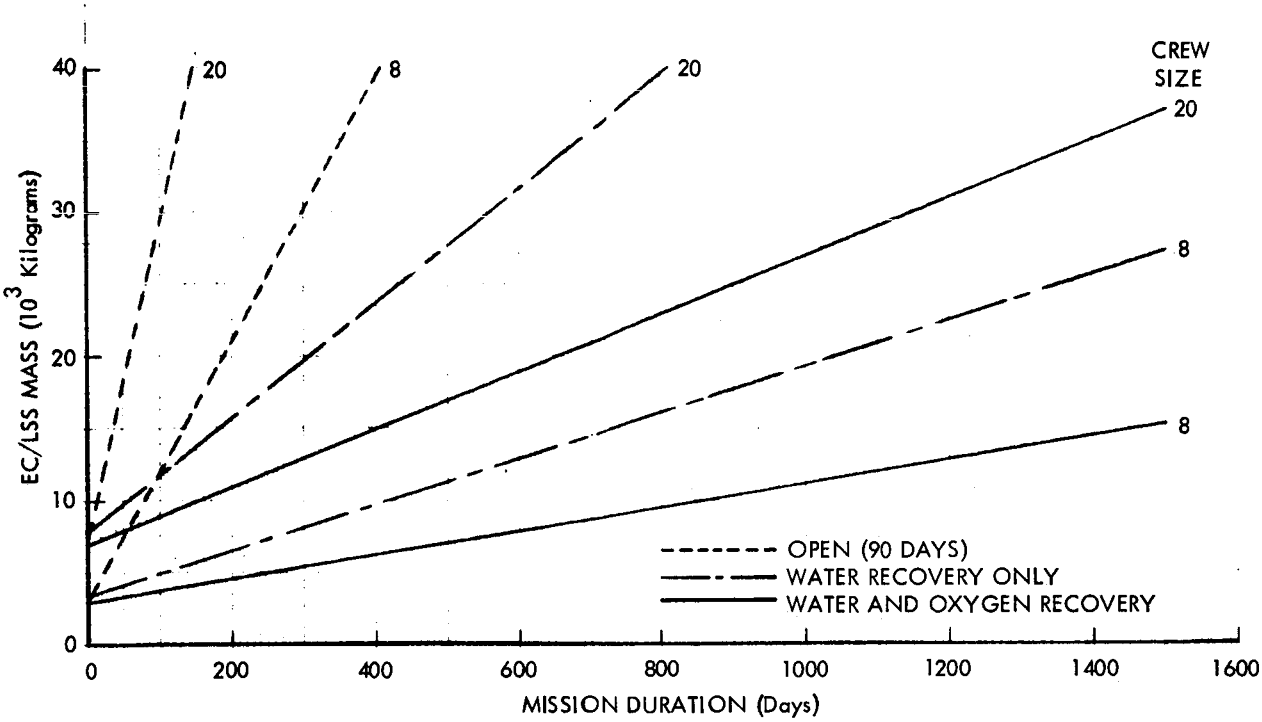

Figure 2. Mission Module EC/LSS Mass Comparison

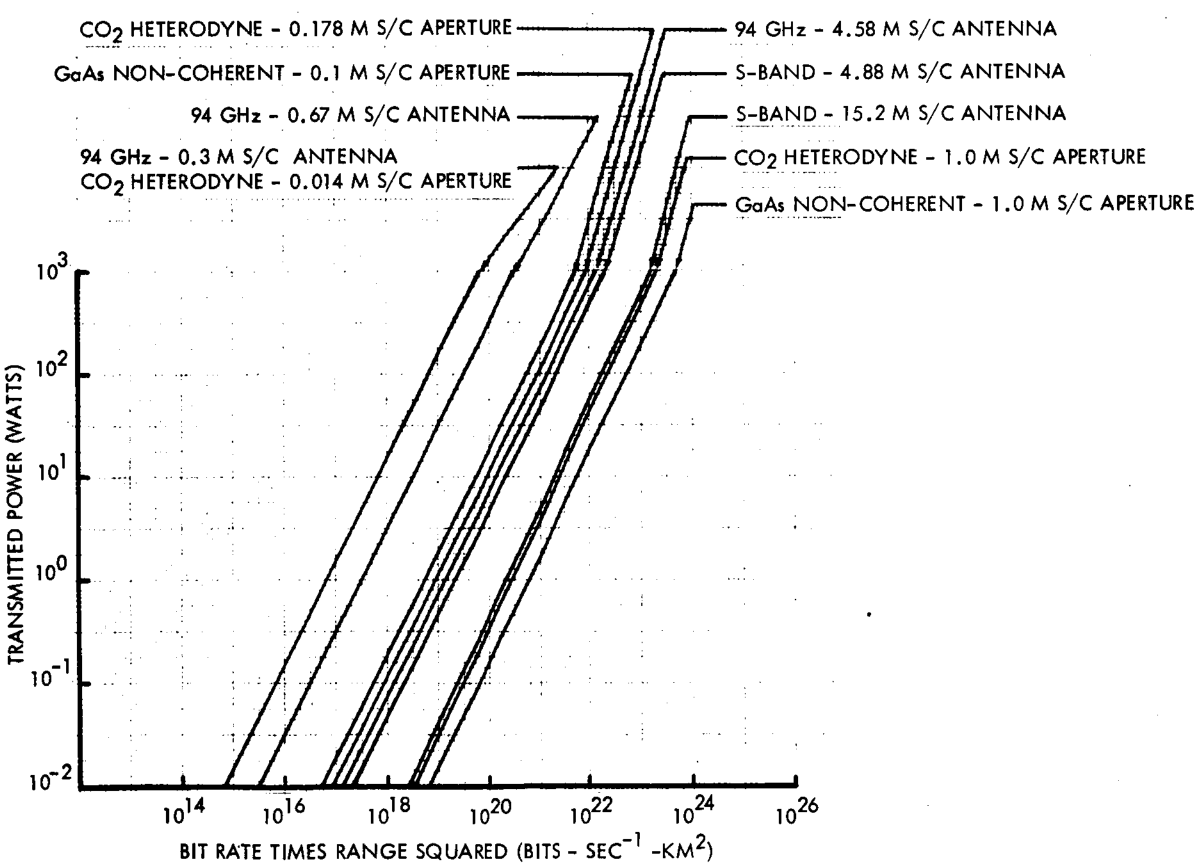

Figure 3. Interplanetary Communications Subsystem Comparison

which is believed to be a serious point-and-tracking problem. The beam width can be increased by decreasing the aperture diameter, but the power requirements are also increased. An order-of-magnitude decrease in the aperture diameter will increase the beam width by the same factor, but it will require an increase in the power requirements of approximately two orders of magnitude.

PROPULSION SUBSYSTEM

The propulsion subsystem analysis was concerned with the establishment of weight-scaling equations and to the selection of candidate propellants for use during the weight-synthesis analysis. Scaling equations are available which can be applied to nuclear engines and to pump-fed, pressure-fed, and toroidal aerospike chemical engines. The propellants selected for inclusion in the subsequent system synthesis were LO_2/LH_2 as representative of high-energy and FLOX/MMH as representative of the class of space-storable systems.

ELECTRICAL POWER SUBSYSTEM

The spectrum of candidate electrical power subsystems which must be considered for application during the post-1980 era is quite broad because of the many combinations of power sources and converters which must be considered. Many combinations were evaluated, and the most promising candidates were identified for use with the mission module and the planetary excursion module. The identification of the most suitable combinations was based on the demonstrated capability of developed systems or systems in the process of development and on improvement projections. To obtain realistic projections, the power-source and conversion-system combinations were analyzed on an equal basis by establishing weight penalties and credits to compensate for inherent differences in the various systems.

A brief investigation of the mission module electrical power requirements was conducted to determine the approximate level of the power requirements. The results of the investigation are shown in Table 6. The electrical power load elements considered were the environmental control and life support subsystem, the communication subsystem, and the requirements for illumination, instrumentation, housekeeping, etc. Based on the assumed power loads, the total power requirements are less than 15 kWe, even when crew sizes of 20 men are considered.

The power-source and conversion-system combinations which are considered to be the most competitive for use with the mission module are presented in Table 7. Nuclear reactors are not considered to be competitive for power levels below 15 kWe. When compared with the radioisotope source at this level, the reactors are heavier and more complex because of higher levels of radiation. Conversely, the radioisotope power source is less attractive at power levels above 15 kWe because of other considerations (e.g., cost), even though it is lighter than the reactor systems.

SYSTEM CHARACTERISTICS

The total system requirements of all potential missions must be examined in order to ensure that maximum utility of new system developments is realized. In this manner, modules which are developed for the nearer-term missions will, to the maximum extent possible, satisfy the requirements of the more advanced missions. The

Table 6. Mission Module Electrical Load Analysis**
(Power Requirement - Watts)

Load Element	Crew Size			
	4	6	10	20
EC/LSS*	2,500	3,500	5,000	9,000
Communications	2,000	2,000	2,000	2,000
Illumination	250	350	500	1,000
Instrumentation	150	225	350	450
Housekeeping and miscellaneous	500	600	750	1,000
Subtotal	5,400	6,675	8,600	13,450
Losses (line) 3%	150	200	250	400
Total	5,550	6,875	8,850	13,850

*With H_2O and O_2 recovery.
**No emergency.

Table 7. Competitive Auxiliary Power Systems for Mission Module

Nominal Power Level (kWe)	Mission Duration (years)			
	≤ 2.5		≈ 4*	
15 to 30	Isotope	Rankine Brayton Thermoelectric	Isotope	Rankine Brayton Thermoelectric
	Reactor	Rankine Brayton Thermoelectric	Reactor	Rankine Brayton Thermoelectric
	Solar photovoltaic			
< 15	Isotope	Rankine Brayton Thermoelectric	Isotope	Rankine Brayton Thermoelectric
	Solar photovoltaic			

*Solar photovoltaic systems omitted since longer missions are consistent with heliocentric radius >2.5 to 3 A.U.

study approach employed to establish common module characteristics was to first establish the requirements of all potential missions, assuming the individual modules were designed for specific mission objectives and mission opportunities. The resultant family of modules was then examined,and modules were defined which satisfied the requirements of the maximum number of missions. Implicit in such an approach was the elimination of either mission objectives, mission opportunities, or mission modes which presented either unique or excessive requirements.

The system mass requirements were determined for representative mission opportunities for each of the mission objectives and mission modes considered in the study. The principle modules which were considered were the manned modules (Earth reentry module, mission module, and planetary excursion module) and the propulsion modules.

MANNED MODULES

The Earth reentry module mass is dependent upon the module configuration, crew size, and Earth-reentry speed. During the present study, three configurations were considered: low L/D (Apollo), biconic, and segmented conic. The effects of reentry speed on the mass requirements of the three configurations are shown in Figure 4 for crew sizes of eight and twenty men. The Apollo configuration, which is the most sensitive to reentry speed, is the lightest configuration for reentry speeds below 14.7 kilometers/second. The conic configuration is the lightest for reentry speeds above 17.5 kilometers/second, while the biconic configuration is the lightest for the intermediate reentry speeds. The relative mass advantages are approximately the same for the entire range of crew sizes considered. The Earth-reentry speeds are less than 15 kilometers/second for the majority of missions considered, indicating that the Apollo configuration is desirable on the basis of mass considerations.

The crew size, mission duration, and selection of the types of subsystems have the predominant effect on the mission module mass, while the free volume per man and the number of floors have an almost negligible effect. The mission module mass is increased by only one percent when the number of floors is decreased from four to three. The above variation is based on a nominal free volume per man of 750 cubic feet/man, the largest crew size considered (20 men), and a mission duration which exceeds the upper limit for the mission considered (1500 days). Therefore, the number of floors can be selected on the basis of considerations other than mass, e.g., diameter, length-to-diameter ratio, etc. The effects of free volume/man vary from 7 to 17 percent for the range of free volumes considered. The lower variation corresponds to a crew size of 20 men and a mission duration of 1500 days, while the upper variation corresponds to a crew of 4 men and a duration of 300 days. Therefore, the free volume/man can be increased for the longer mission durations without a major impact on the mission module mass requirements. These results are displayed in Figure 5 which shows variation in module mass based on the use of the oxygen-and-water EC/LSS subsystem and an isotope-and-mercury Rankine electrical power subsystem.

Two basic types of planetary excursion modules were considered: pure retrobraking, and aerobraking and retrobraking modules. The pure retrobraking modules are required for landings on Mercury, Ceres, Vesta, and Ganymede, while the aerobraking and retrobraking modules are utilized for Mars landings. The pure retrobraking modules are similar in concept to the Apollo Lunar Module with separate stages for descent and ascent. Both ballistic (Apollo) and lifting-body configurations were

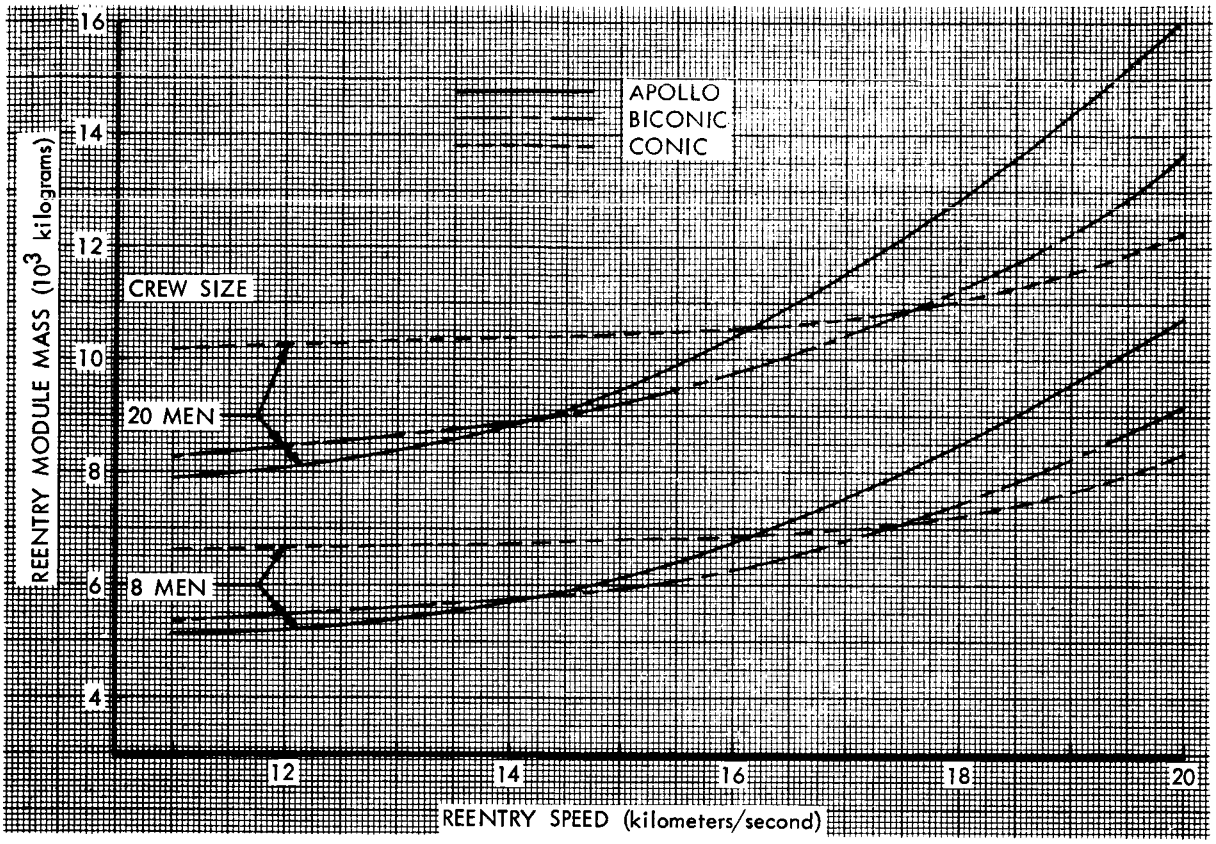

Figure 4. Reentry Module Mass Comparison

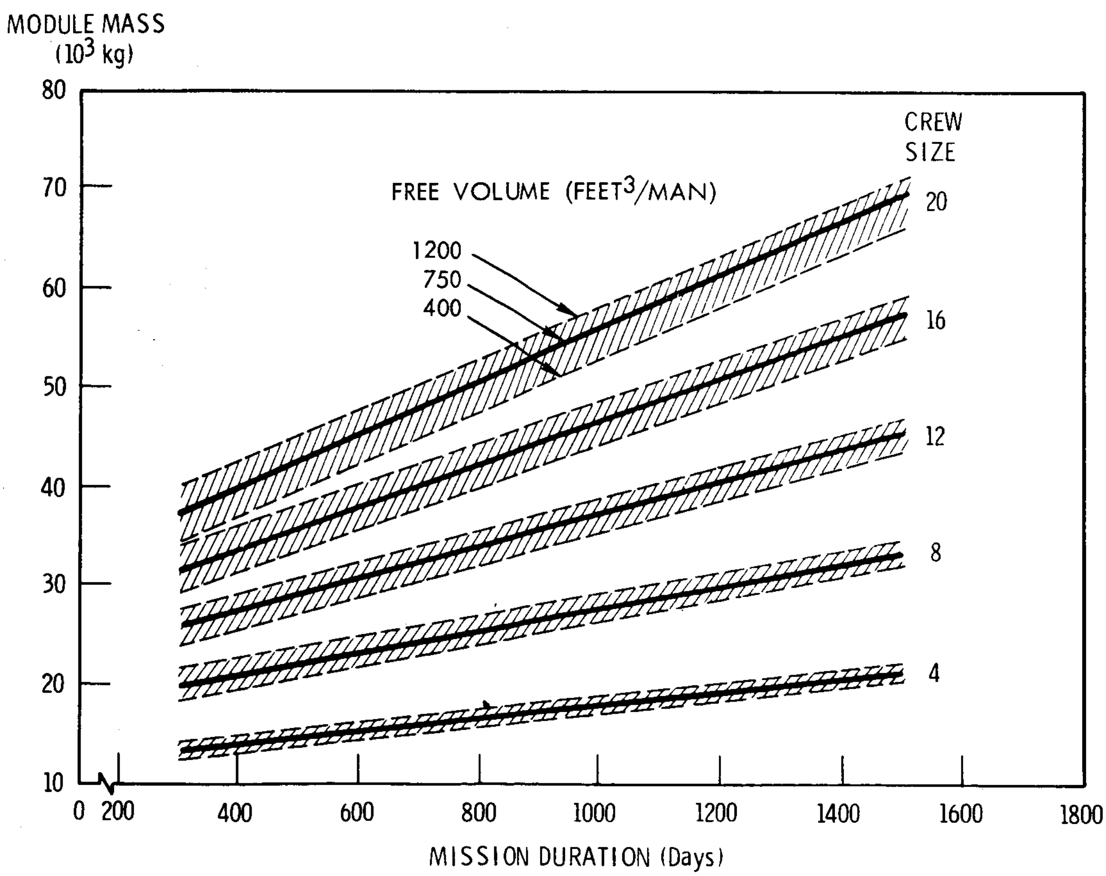

Figure 5. Mission Module Mass Comparison

evaluated for Mars landings. The ballistic configuration proved to be the lightest and was used during all system-mass synthesis analyses. The module-mass requirements are summarized in Table 8.

PROPULSION MODULES

The propulsion modules, which constitute the majority of the mass of the total system, are defined by the propulsion system type, payload, and characteristic velocity requirements. The propulsion systems which were considered were cryogeni and space-storable chemical systems, and solid-core and gaseous-core nuclear systems. The chemical systems were considered for all mission maneuvers for Mars and Venus missions and for planetary-orbit insertion and escape maneuvers for Jupiter missions with elliptical planetary parking orbits. The mass requirements of solid-core nuclear propulsion modules were determined for all maneuvers for all mission objectives. The evaluations of the requirements of gaseous-core nuclear modules were limited to the advanced missions employing circular planetary parking orbits, although the influence of such systems on the mission duration and payload capability of Mars and Venus missions was investigated.

Table 8. Planetary Excursion Module Mass Requirements

Mission Objective	Planetary Excursion Module Mass (kilograms)			
	Four-Man Crew		Ten-Man Crew	
	Circular Orbit (e = 0)	Elliptical Orbit (e = 0.7)	Circular Orbit (e = 0)	Elliptical Orbit (e = 0.7)
Mercury	61,900	103,200	112,100	181,100
Mars	40,400	60,600	70,600	105,200
Vesta	11,000	-	20,000	-
Ceres	12,000	-	23,000	-
Ganymede	27,800	36,400	50,500	65,200
Note: Occupancy Time = 28 days				

COMMON SYSTEM CHARACTERISTICS

The initial examinations of common modules were based on the utilization of a common Earth-reentry module and a common mission module. The modules which were selected satisfied the requirements of the majority of the missions. During the analyses of common manned modules, the propulsion modules were sized by the particular requirements of the missions.

The investigations of common propulsion modules were performed using fixed module characteristics (structure and engines) and off-loading propellant as required by the particular mission and propulsion module payload. During the analyses of common propulsion modules, the manned modules and the environmental protection requirements of all modules were sized by the mission. During the analysis of propulsion module mass requirements associated with circular capture orbits, the propagation of off-loading (i.e., overdesigning) the upper stages to the mass requirements of the lower stages was included. This procedure was not carried out during the analysis of elliptical capture orbits since any such mass penalties can be overcome by a slight increase in eccentricity.

The final investigations of the use of common modules were based on the use of both common manned modules and common propulsion modules. These final analyses were conducted only in the case of circular planetary parking orbits.

On the basis of the parameters which were considered in the study, the Apollo Earth-reentry module configuration will satisfy the requirements of future manned planetary missions. Other considerations which may make the development of a second configuration desirable, however, (e.g. abort), were not considered.

Two distinct approaches to the synthesis of common mission modules were considered. In the first approach, the modules were assumed to be developed in a modular manner in which the number of floors in increased as the crew size is increased. The second approach assumed that a single module was designed for the maximum mission duration and crew size with the crew and consumables off-loaded as required for mission which impose lesser requirements. Regardless of which approach is used, it was assumed that the meteoroid and radiation protection would be sized for the particular mission.

Within the constraint of employing circular capture orbits, the establishment of common propulsion modules was relatively straightforward. Regions of common propulsion module requirements could be defined by limiting the mission opportunities and the crew sizes considered. However, when elliptical planetary orbits are considered, regions of common requirements are not as apparent because of the extreme variations in the propulsion module mass requirements.

The various propulsion module combinations which are considered to be particularly attractive are shown in Table 9 with the applicability of the various modules to the family of missions considered shown in Table 10. Several interesting conclusions are apparent from the Table 9, e.g., (1) a 75,000-kilogram nuclear module is appropriate for all missions except Ganymede; (2) a 150,000-kilogram nuclear module is appropriate for all missions except the asteroids. Moreover, such a module seems appropriate for Venus and Mars missions if chemical stages are employed at the planet or if aerobraking is employed; (3) complete propulsion system commonality exists between Mars and Venus missions; (4) to achieve all mission objectives, a nuclear module of at least 600,000 kilograms will be necessary; and (5) missions to Mars and Venus can be carried out with chemical propulsion modules which do not exceed 300,000 kilograms in size.

Table 9. Candidate Common Propulsion Modules

Propulsion Module Combinations	Propulsion Module Mass (10^3 kg)					
	75	100	150	300	600	1200
	N			N		
	N			N		N
NNN	N				N	
	N					N
			N		N	
			N			N
				N		N
		C	N			
NCC		C		N		
		C			N	
CCC		C			C	
		C		C		
EOE Aerobraker			N			
				N		
					N/C	
EOE	N					
Flyby		C	N			

N = Nuclear propulsion
C = Chemical propulsion (cryogenic or space storable)
F = Flyby mission

Table 10. Applicability of Common Propulsion Modules

Mission Objective	Propulsion Module Mass (10^3 kg)							
	Nuclear					Chemical		
	75	150	300	600	1200	100	300	600
Mercury		X		X				
	X		X		X			
Venus		A	A	A				A
	X		X					
		X		X				
		C	C	C		X		
						X	X	
Mars		A	A	A				A
	X		X					
		X		X				
		C	C	C		X		
						X	X	
Ceres	X		X		X			
	F							
Vesta	X		X		X			
					F			
Jupiter	X				X			
	X			X				
		X		X				
		X			X			
					X	X		
		F						
Ganymede		X		X				
		X			X			
	X		X		X			

X = Propulsion system of specified type
C = Chemical propulsion systems at planet arrival/departure
A = Aerobraking capture
F = Flyby

CONCLUSIONS

It has been determined that several areas of common technological requirements exist when the requirements of both the near-term and advanced manned planetary exploration missions are considered. Common requirements exist at both the module level and the subsystem level; common modules and subsystems can be developed for the near-term missions which will be compatible with the requirements of the advanced missions. Weight and performance penalties are of course incurred, but in many cases are quite small. When the cost and development time of optimized systems developed independently for each specific application are considered, these penalties may well be acceptable.

Of the modules which are required the commonality potential is the greatest for the Earth reentry module (ERM). Only the low L/D (Apollo) configuration need be developed for the entire spectrum of missions, provided the Mars missions are limited to the Venus swingby mode. This configuration will probably require the least development effort. Since the total mass of the Earth reentry module is relatively small, the penalties associated with using an ERM which is designed to meet the highest Earth-entry speed will also be small.

Common mission modules can be achieved in one of two ways. One method would be to utilize a modular approach whereby a basic module is developed and additional floors are added as required to accommodate larger crew sizes. The alternate approach would be to design a module which is compatible with the requirements of the largest crew size and longest mission duration. Crew and consumables would be off-loaded as required for missions with lesser requirements though in extreme cases crew off-loading results in significant weight penalties. The design requirements of the mission module subsystems could also be based on either of the approaches. Regardless of which approach is used, the initial design of both the basic module and the module subsystems must be based on the maximum requirements in order to ensure adequate module growth capability.

The greatest degree of commonality among the planetary excursion modules (PEM) lies, of course, with those required for Ceres and Vesta. A certain degree of commonality exists among the PEM's required at Mercury and Ganymede, although each commonality would likely be limited to elements of the system, e.g., descent stage or crew quarters. Because of its aerodynamic descent requirements, the Mars PEM represents a unique configuration.

The mission-performance requirements, and thus the propulsion-module mass requirements, fall into two basic families. One family includes all the propulsion modules required for the Mars and Venus missions and the planetary-orbit insertion and escape propulsion modules required for the advanced missions. The second family consists of the large propulsion modules required for Earth-orbit escape for the advanced missions. A second conclusion concerning the performance requirements—a conclusion which will benefit future mission studies—is that appropriate trajectories can be established on the basis of velocity requirements alone without recourse to lengthly mass calculations.

An approach to propulsion-module selection which appears to be particularly attractive would be the development of a single nuclear propulsion module which has a restart capability. A single module could be used to perform both the planetary-orbit insertion and escape maneuvers for the Mars and Venus missions, and the same module, without a restart requirement, could be used in multiples to perform the Earth-orbit escape maneuver for these missions. Multiples of the same module could then be used to perform the planetary-orbit insertion and escape maneuvers for Mercury, Ceres, Vesta, and Jupiter and/or Ganymede missions. An alternative to the restartable stage would be the development of a relatively small module which could be used singly for the planetary-orbit escape maneuvers and in multiples for the planetary-orbit insertion maneuvers for the Mars and Venus missions. The same module could be used either singly or in multiples for the planetary-orbit escape maneuvers for Mercury, Ceres, Vesta, Jupiter, and Ganymede missions. An intermediate size module would then be required to perform the orbit-insertion maneuvers for the advanced missions but with this same module used for Earth-orbit escape for the Mars and Venus missions. Regardless of which alternative might be adopted, a large propulsion module would ultimately have to be developed for Earth-orbit escape for the advanced missions.

Due to the short occupancy times, an open environmental control and life support subsystem is the most attractive system for use in the Earth reentry module and the planetary excursion module ascent and descent stages. Although a mass advantage would accrue if a partially closed system were used in the PEM descent stage, the magnitude of the savings does not warrant the additional system complexity. A water-and-oxygen recovery system appears to be the most attractive system for use in the mission module for the family of missions considered in this study. Such a system will not necessitate major technological advancements and could be readily available for all missions during the time period being considered.

Further analyses are required of the psychological and physiological effects of fully closed environmental control and life support subsystems and the mass requirements of such systems. On the basis of the data available for the present study, it appears that food-producing systems will not be required. This conclusion, however, is sensitive to the assumptions made concerning the amount of stored food which must be provided.

A parallel approach appears to be necessary in the area of communications subsystems. S-band should be developed to its full capability. It probably will fulfill many interplanetary requirements for the next 20 to 30 years, provided adequate data-management and data-compaction techniques are developed by the time the advanced missions are considered. On the other hand, the limitations with S-band are clearly evident. Thus, smaller, lighter, and higher data-rate systems will be required eventually and research must be continually applied. A smooth transition from S-band to either millimeter or optical systems should be applied to take advantage of the favorable system characteristics of these latter systems.

If applied to Mars and Venus stopover missions and to flyby missions to the remaining target bodies, chemical propulsion systems can play a significant part in manned planetary exploration systems. Within this propulsion category, both space-storable and cryogenic propellants are useful. To perform the entire family of missions (with high-thrust systems) nuclear rockets are mandatory. The mass-in-Earth-orbit requirements are such that adequate Earth-launch vehicle capability can probably be developed while limiting the spacecraft propulsion systems to solid-core reactors. If gaseous-core reactors were employed instead, the initial mass requirements for the more advanced missions could be reduced by an order of magnitude.

Candidate electrical power subsystems for use with the mission module (for power ranges of 2 to 15 kWe) can be limited to solar cells and to radioisotopes combined with dynamic (Rankine and Brayton cycle) or thermoelectric conversion. At the power levels felt to be necessary, nuclear reactors prove to be heavier and more complex and to impose operational constraints when compared to radioisotopes. Solar concentrators do not appear to be particularly attractive because of high orientation-accuracy requirements when compared to solar cells.

Protection against the space environment can in many cases be accomplished by modifications to the mission operations rather than by major increase in the system design requirements. For instance missions beyond the asteroid belt could become prohibitive due to excessive meteoroid shielding requirements. Employing a two-plane transfer over the asteroid belt, however, maintains the shield weights at reasonable values.

Passive thermal control of the propulsion modules appears feasible for all mission objectives and propulsion systems although the entire concept of propellant storability is based on the ability to limit heat leaks into the propellant. An active thermal control system based on current technology seems appropriate for the mission module. A major problem will be protection of the ECS radiators for missions to Mercury.

Space radiation protection requirements can possibly be met by the inherent spacecraft shielding with additional shielding required only during the years near maximum solar activity. The intensity of the trapped radiation at Jupiter can be such that either the stopover times would be seriously limited or high (>15 radii) orbit altitudes would be required.

The foregoing conclusions must be tempered in view of the uncertainties inherent in their development. Foremost among these uncertainties is the projection of technology into the post-1980 era. Unquestionably, the values quoted herein are subject to refinement. In some instances, gross revisions may be necessary. Nevertheless a fundamental conclusion has been reached; namely, that the concept of commonality can be applied at several module, system, and subsystem levels to a broad spectrum of manned interplanetary missions.

TECHNICAL REPORT INDEX/ABSTRACT

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ABSTRACT

SEVERAL RECENT STUDIES, INCLUDING STUDIES CURRENTLY IN PROGRESS, HAVE EXAMINED THE TECHNICAL REQUIREMENTS OF POTENTIAL NEAR-TERM MANNED PLANETARY EXPLORATION MISSIONS. ONLY A LIMITED NUMBER OF STUDIES HAVE BEEN CONDUCTED WHICH INCLUDE A SIMULTANEOUS EXAMINATION OF BOTH NEAR-TERM AND ADVANCED MISSIONS. TO ENSURE AN EFFICIENT APPLICATION OF NATIONAL RESOURCES FOR MANNED PLANETARY EXPLORATION, THE REQUIREMENTS OF BOTH THE NEAR-TERM AND ADVANCED MISSIONS MUST BE EVALUATED SIMULTANEOUSLY. THE OBJECTIVE OF SUCH AN EVALUATION WOULD BE TO ESTABLISH THE EXISTENCE OF COMMON REQUIREMENTS FOR THE DIVERSE MISSION OBJECTIVES WHICH MIGHT BE CONSIDERED. THE EVALUATION OF COMMON REQUIREMENTS MUST INCLUDE THE MODULES, SUBSYSTEMS, AND TECHNOLOGIES REQUIRED FOR THE MISSIONS. THE PURPOSE OF THIS STUDY WAS TO PERFORM SUCH AN EVALUATION AND TO ESTABLISH POTENTIAL AREAS OF COMMON REQUIREMENTS. THE REQUIREMENTS OF A FAMILY OF MANNED PLANETARY MISSIONS WERE EXAMINED AND POTENTIAL AREAS OF COMMON REQUIREMENTS ESTABLISHED IN ORDER TO ASSIST IN THE DETERMINATION OF THE MOST REWARDING AREAS OF FUTURE TECHNOLOGICAL DEVELOPMENT. INHERENT IN SUCH AN EVALUATION IS THE ESTABLISHMENT OF REASONABLE MISSION OBJECTIVES AND MISSION MODES FOR A MANNED PLANETARY EXPLORATION PROGRAM.

ERRATA

Technological Requirements Common to Manned Planetary Missions

Summary Report, SD67-1086

1. Page 2 - first sentence describing study constraints should read as follows: "Only high-thrust propulsion systems are considered. Within this category, however, . . ."
2. Page 5 - Table 4: Ascent ΔV for Ceres = 565 km/sec.
3. Page 15 - second sentence, last paragraph: Delete exclusion of Ganymede.
4. Page 16 - Table 9: The entries for EOE Flyby requirements should be on three separate lines.
5. Page 17 - Table 10: Enter "F" under 100,000 kg chemical propulsion module for Vesta, Ceres, Jupiter flybys; for nuclear systems module mass for Vesta flybys is 75,000 kg - not 1,200,000 kg.