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FINAL REPORT

TECHNOLOGICAL REQUIREMENTS COMMON TO MANNED PLANETARY MISSIONS

(Contract NAS2-3918)

Technical Summary

SPACE DIVISION

NORTH AMERICAN ROCKWELL CORPORATION

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.

SD 67-621-1

TECHNOLOGICAL REQUIREMENTS COMMON TO MANNED PLANETARY MISSIONS
NAS2-3918

Technical Summary

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FOREWORD

This report contains the final results of the studies conducted under Contract NAS2-3918, Technological Requirements Common to Manned Planetary Missions. This report consists of five volumes. This first volume (SD 67-621-1) summarizes the study results. The detailed descriptions of the study are presented in the following volumes:

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)
Appendix D - System Synthesis and Parametric Analysis	(SD 67-621-5)

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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 the individual modules 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 attendent 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

The total system requirements for each of the mission objectives are defined by the characteristics of the mission, mission mode, and mission opportunity. The mission objectives which were considered as being representative of the objectives for manned planetary exploration during the remainder of this century were Mercury, Venus, Mars, Jupiter and its satellite Ganymede, and the asteroids Ceres and Vesta. Mission performance requirements for representative mission opportunities during the 1980-to-2000 time period were determined for each of the mission objectives. The investigations of the mission requirements also included a limited examination of the aerobraking technology requirements for Mars and Venus aerobraker missions. To define the total system requirements for manned landing missions, the characteristic velocity requirements for landing and ascent to orbit were determined for Mercury, Mars, Vesta, Ceres, and Ganymede. The guidance and navigation requirements for injection into orbit about Ganymede were also examined. The mission requirements analyses which were conducted are summarized in the following paragraphs, and the details of the analyses are contained in Appendix A.

PERFORMANCE REQUIREMENTS

The mission performance requirements are determined by the mission objective, mission opportunity, and mission mode being considered. In order to establish basic performance requirements for manned planetary missions during the post-1980 era, performance analyses were conducted for each of the mission objectives considered in the study. Mission modes considered were direct, Venus swingby, and flyby. Both aerobraker and retrobraker missions were considered for Venus and Mars. Performance requirements data were generated to the level of detail required to define the variations in the performance requirements over a complete cycle of opportunities for each of the mission objectives and mission modes considered. Baseline missions were selected for each mission objective and mission mode which are representative of the minimum, maximum, and average performance requirements which would be required over a complete cycle of opportunities.

Mission Opportunity Selection

The selection of mission opportunities was based on data provided by the NASA/OART Mission Analysis Division, on gross performance scans, and on the relative positions of Earth and target body at the time of Earth departure and target body arrival. Opportunities for direct missions will occur once during each synodic period of the target body; the performance requirements will roughly repeat each synodic cycle. Only a limited number of mission opportunities were investigated in detail, since the objective of the mission analysis study was to determine characteristics of mission opportunities representative of minimum, average, and maximum performance requirements.

Mission Selection

Basepoint missions were selected for each of the mission objectives on the basis of performance requirements. Such an approach neglects the effects of mission duration of the mass requirements of the manned modules, and thus, the total system mass. For the flyby missions, the missions were selected on the basis of minimizing the Earth-departure incremental velocity requirements. However, two methods of evaluating the performance requirements were considered for direct and swingby missions.

The first method consisted of selecting the combination of Earth-departure date, target-body arrival date, target-body departure date (for a given stay time), and Earth-arrival date, which minimized the summation of the mission incremental velocity requirements. This approach assumes the effects of staging and propellant selection will not affect the mission selection. The mission opportunities were determined from plots of the incremental velocity requirements similar to Figures 1 through 5 which define the incremental velocity requirements as a function of the arrival/departure date and trip time for the 1990 Mars opportunity. Figure 1 shows the Earth-departure (transplanet) velocity requirements as a function of Mars-arrival date for a range of transfer times. For heliocentric transfer angles near 180 degrees, two-plane transfers are evaluated, and, if beneficial, the velocity increment is included in the Earth-departure velocity requirement. These data are used to determine the velocity requirements for the Earth-to-Mars phase of an aerobraking mission. Figure 2 shows the total transplanet velocity requirements using a retrobraking maneuver at Mars. The corresponding Mars-to-Earth velocity requirements are shown in Figure 3. The data shown in Figure 3 are based only on the planetary-orbit-escape incremental velocity requirements, since direct reentry was assumed at Earth. Retrobraking maneuvers prior to reentry which reduce the Earth-reentry speed were not considered. The performance requirements are also plotted as velocity contours as shown in Figures 4 and 5 for the Earth-to-Mars and Mars-to-Earth phases, respectively.

Desirable mission opportunities were located by overlaying transparencies of the contour plots. By overlaying Figures 4 and 5, it can be seen that two families of solution exist which have low total velocity requirements. One family of missions, the conjunction-class missions, have mission durations of approximately 1,000 days. The second family of missions, the opposition-class missions, have higher velocity requirements but shorter

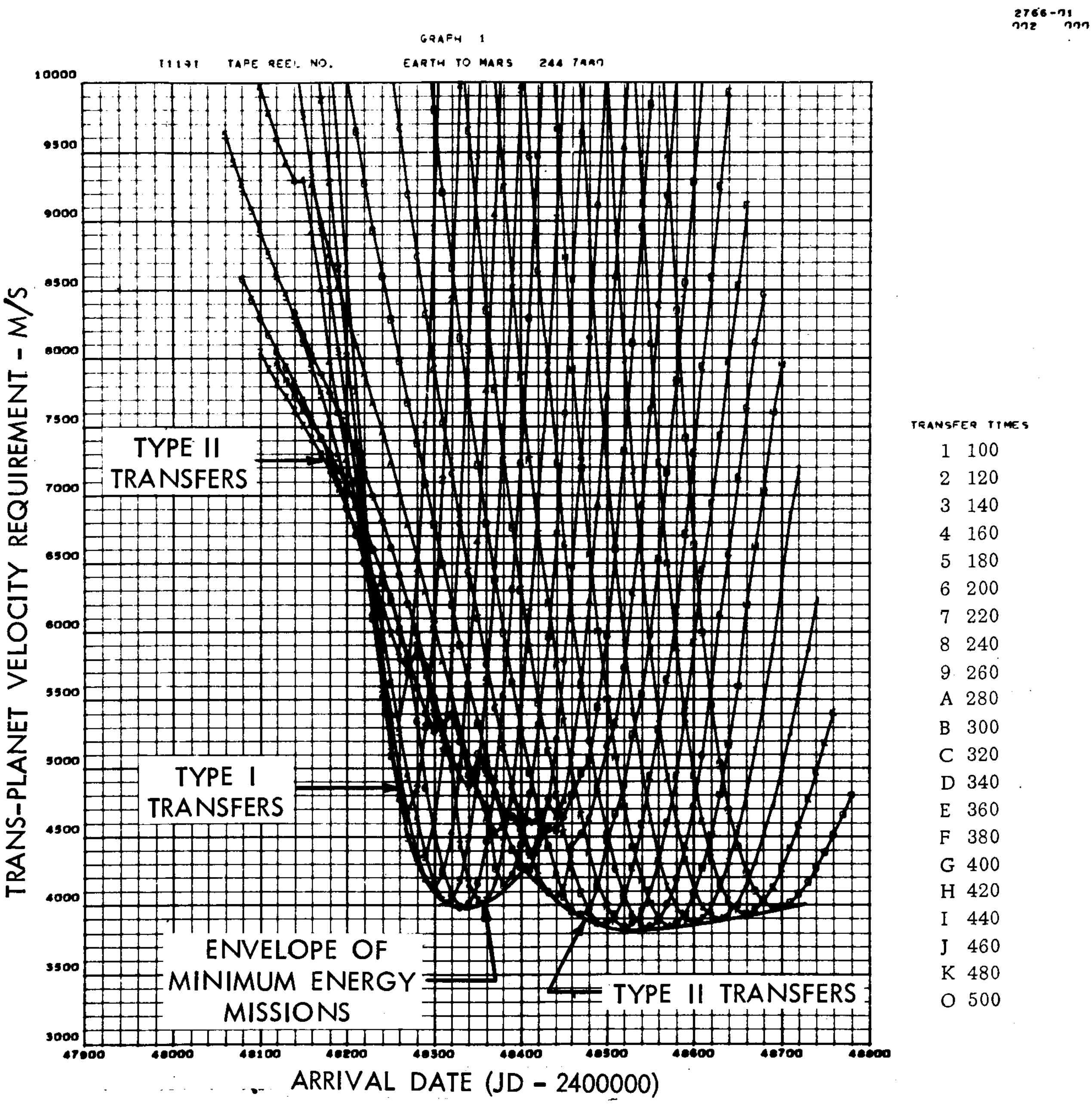

Figure 1. Trans-Planet Velocity Requirements (1990 Mars Opposition)

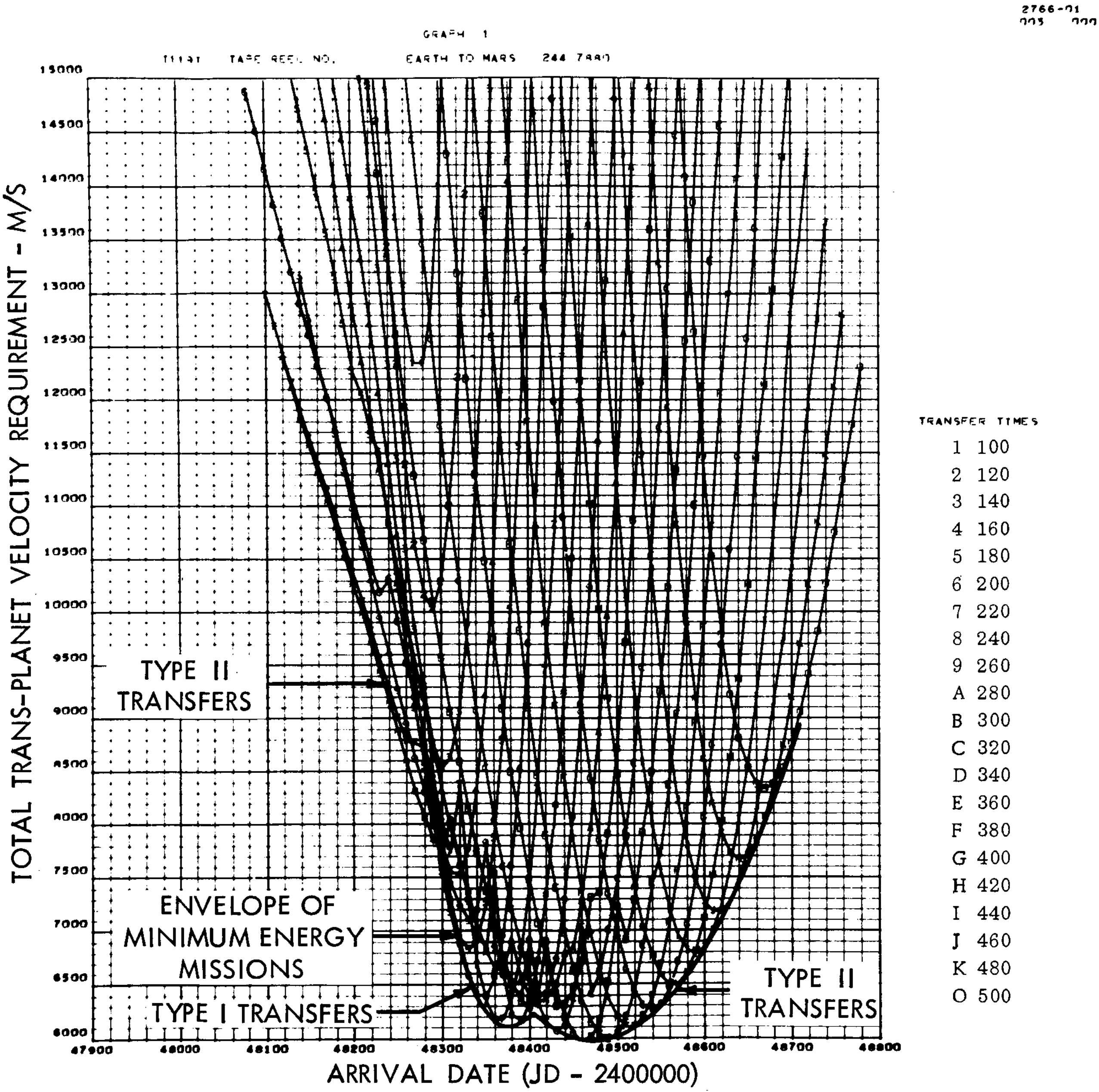

Figure 2. Total Trans-Planet Velocity Requirements
(1990 Mars Opposition)

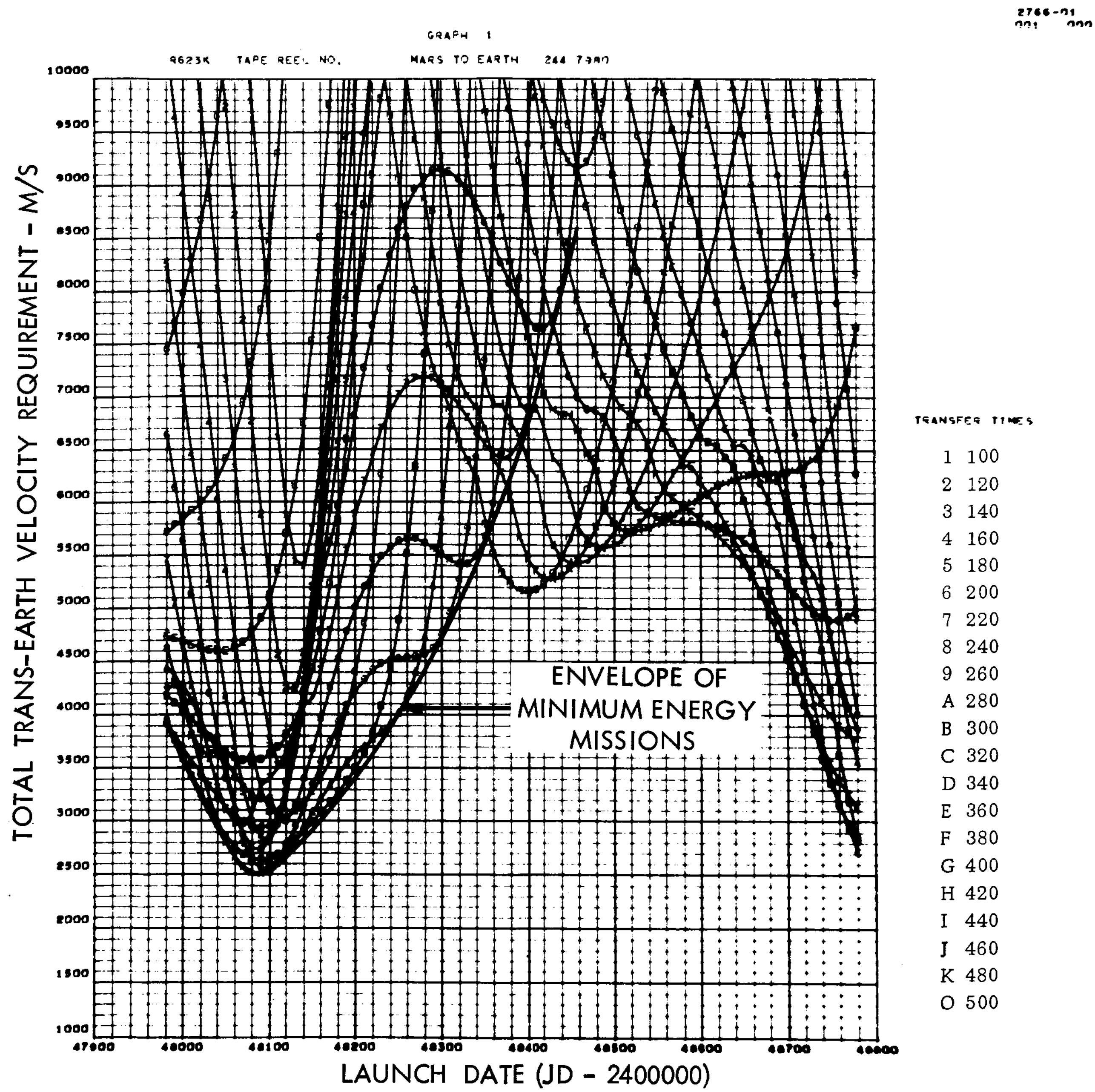

Figure 3. Total Trans-Earth Velocity Requirements (1990 Mars Opposition)

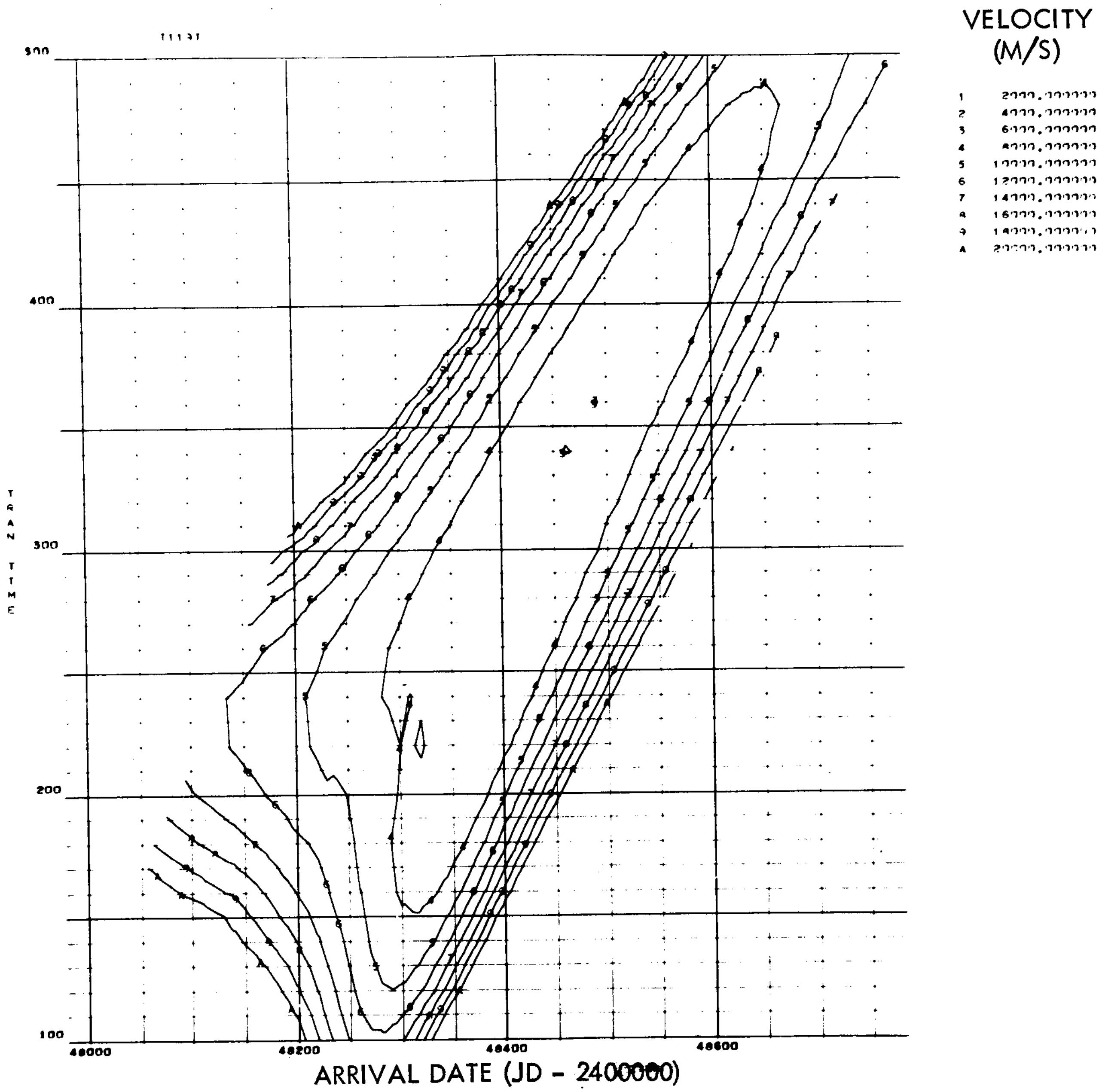

Figure 4. Total Trans-Planet Velocity Contours (1990 Mars Opposition)

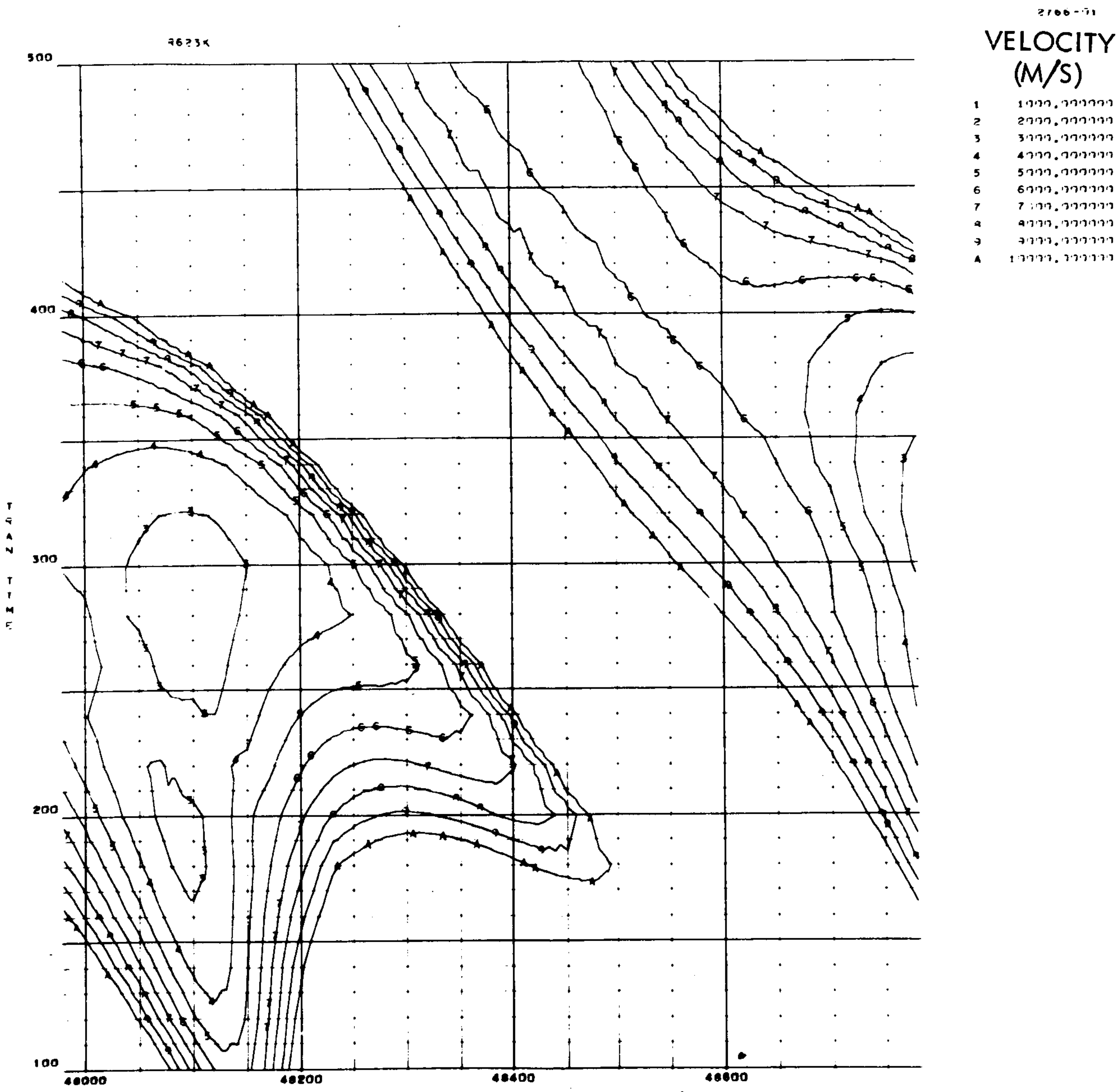

Figure 5. Total Trans-Earth Velocity Contours (1990 Mars Opposition)

mission durations. Only the opposition-class missions were considered in the present study. The effects of stay time at Mars can be seen by translating the overlays horizontally, while trip time effects can be seen by a vertical translation. Once desirable regions of mission opportunities are located, the basic plots of the velocity requirements (e.g., Figures 2 and 3) are used to define the mission. The mission is determined by examining the envelopes of the velocity requirements to determine the arrival date (and thus, the departure date for a given stay time) which minimizes the total velocity requirements. Again, transparencies of the plots can be used to facilitate the analyses. The mission selection is performed by translating the overlays vertically (with an initial horizontal translation to account for stay time) and evaluating the requirements at the intersection of the envelopes. The trip times and the individual incremental velocity requirements can then be determined.

The second method of establishing performance requirements is based on the use of propulsion factors (initial gross mass/payload mass) to obtain an initial mass ratio. For landing missions, the ratio of the initial mass in Earth orbit (W_O) and the trans-Earth payload ($W_{ERM} + W_{MM}$) is approximated by

$$\frac{W_O}{W_{ERM} + W_{MM}} = \left[P_{TEI} + \left(\frac{W_{PEM}}{W_{ERM} + W_{MM}} \right) \right] P_{POI} P_{TPI}$$

where

W_{ERM} = Earth reentry module mass

W_{MM} = mission module mass

W_{PEM} = planetary excursion module mass

P_{TEI} = trans-Earth injection propulsion factor

P_{POI} = planetary orbit insertion propulsion factor

P_{TPI} = trans-planet injection propulsion factor

For simplicity, only the major propulsive maneuvers are shown. Trans-planet and trans-Earth midcourse correction velocity requirements (P_{TPMCC} and P_{TEMCC}) were considered in the actual evaluations.

The mass ratios were evaluated for all mission objectives and mission opportunities considered in the study. In all cases, it was found that essentailly the same missions were defined by minimizing either the total velocity requirements or the mass ratio requirements for the broad spectrum of missions and propulsion systems considered in the study. Therefore, it is concluded that a mission defined by the simple minimization of the total incremental velocity requirements will also approximate the minimum mass in Earth-orbit mission for a given mission duration. This qualification is reasonable since, in general, the duration of the minimum mass mission is slightly less than the minimum total incremental velocity mission. Although the total velocity requirements would increase, the total mass in Earth orbit would decrease due to a reduction in the time-dependent mass requirements (mission module mass, boil-off propellant, etc.). Of significance, however, is that once the mission duration and stay time are prescribed by the mission objective, the best particular trajectory (i.e., proper Earth-departure and planet-arrival dates) can be selected from trajectory considerations only, without recourse to more time-consuming mass calculations.

Mission-Performance Requirements

The characteristics of the missions which must be defined in order to determine the total system requirements are the incremental velocity requirements, atmospheric entry speeds, and trip times for each of the mission legs. The incremental velocity requirements define the propulsion module mass requirements for a given payload. The atmospheric-entry speeds must be defined in order to determine the mass requirements of the Earth reentry module and the aerobraker-heatshield mass for Mars and Venus aerobraker missions. The trip times define the mass characteristics of the time-dependent subsystems such as the environmental control and life support subsystem and define the environmental protection requirements.

Summaries of the characteristics of the basepoint missions from which missions were selected for weight synthesis analyses are presented in Tables 1 through 16. The tables define the dates at which major mission events occur, the durations of the mission phases, the major incremental velocity requirements, and the Earth-reentry speed. Table 1 defines the characteristics of the Vesta, Ceres, and Jupiter flyby missions. The characteristics of the direct missions for all mission objectives are defined in Tables 2 through 10. The Mercury and Mars missions that employ Venus swingbys are shown in Tables 11 through 16.

The mission opportunities selected are representative of opportunities with minimum, maximum, and average total velocity requirements for the post-1980 era. For all cases except Mercury, stay times of zero, thirty, and sixty days have been considered. An Earth-reentry speed of 19.8 kilometers/second (65,000 feet/second) was the only constraint imposed on the mission selection. The Jupiter and Ganymede missions were selected to minimize the reentry speed rather than the sum of the incremental velocity

Table 1. Flyby Baseline Missions

Target	Depart Earth		First Leg Trip Time (days)	Arrive Target (JD*)	Second Leg Trip Time (days)	Entry Speed (kps)	Total Mission Time (days)	Total Mission ΔV (kps)
	Year	JD*						
Vesta	1991	8590	430	9020	300	12.43	730	4.57
	1993	9138	245	9383	851	14.14	1096	6.32
Ceres	1993	9175	315	9490	779	13.22	1094	5.37
	1992	8745	354	9099	742	16.00	1096	8.30
Jupiter	1991	8570	708	9278	672	17.72	1380	6.80
	1985	6170	520	6690	515	15.21	1035	7.10

*Julian date - 244 0000

Table 2. Mercury Baseline Missions (Direct)

Depart Earth Year	Depart Earth JD*	Trip Time (days)	ΔV_1 (kps)	ΔV_2 (kps)	ΔV_3 (kps)	ΔV_{TP} (kps)	Arrive Mercury (JD*)	Stay Time (days)	Depart Mercury (JD*)	Trip Time (days)	ΔV_{TE} (kps)	Entry Speed (kps)	Total Mission Time (days)	Total Mission ΔV (kps)
1988	7344	125	7.32	0.0	9.81	17.13	7469	63	7532	123	6.96	15.02	311	24.09
1990	8022	135	8.57	0.0	6.54	15.10	8157	77	8234	157	7.83	16.96	369	22.94
1992	8752	105	6.83	0.0	6.36	13.18	8857	179	9036	80	6.51	15.59	364	19.69

ΔV_1 = Earth departure incremental velocity
ΔV_2 = Plane change incremental velocity
ΔV_3 = Planetary orbit insertion incremental velocity
ΔV_{TP} = Total transplanetary incremental velocity
ΔV_{TE} = Planetary departure incremental velocity

*Julian date -244 0000

Note: Circular parking orbit
Parking orbit altitude = one planetary radius

Table 3. Venus Aerobraker Baseline Missions

Depart Earth Year	Depart Earth JD*	Trip Time (days)	ΔV_1 (kps)	ΔV_2 (kps)	ΔV_{TP} (kps)	Venus Entry Speed (kps)	Arrive Venus (JD*)	Stay Time (days)	Depart Venus (JD*)	Trip Time (days)	ΔV_{TE} (kps)	Entry Speed (kps)	Total Mission Time (days)	Total Mission ΔV (kps)
1988	7240	120	3.57	0.0	3.57	11.6	7360	0	7360	275	3.87	13.7	395	7.44
	7240	110	3.75	0.0	3.75	12.2	7350	30	7380	285	3.86	14.0	425	7.61
	7240	110	3.75	0.0	3.75	12.2	7350	60	7410	310	4.08	14.3	480	7.83
1990	7840	100	3.81	0.0	3.81	11.4	7940	0	7940	280	3.74	13.6	380	7.55
	7835	100	3.81	0.0	3.81	11.8	7935	30	7965	290	3.87	14.0	420	7.68
	7830	100	3.86	0.0	3.86	12.2	7990	60	7990	310	4.17	14.5	470	8.03
1991	8415	115	3.68	0.0	3.68	10.9	8530	0	8530	270	3.76	13.5	385	7.44
	8405	110	3.78	0.0	3.78	12.0	8515	30	8545	285	3.96	13.7	425	7.74
	8425	130	3.62	0.0	3.62	10.6	8555	60	8615	345	4.19	14.9	535	7.81

ΔV_1 = Earth departure incremental velocity
ΔV_2 = Plane change incremental velocity
ΔV_{TP} = Total transplanetary incremental velocity
ΔV_{TE} = Planetary departure incremental velocity

*Julian date -244 0000

Note: Circular parking orbit
Parking orbit altitude = 1000 km

Table 4. Venus Retrobraker Baseline Missions

Depart Earth		Trip Time (days)	ΔV_1 (kps)	ΔV_2 (kps)	ΔV_3 (kps)	ΔV_{TP} (kps)	Arrive Venus (JD*)	Stay Time (days)	Depart Venus (JD*)	Trip Time (days)	ΔV_{TE} (kps)	Entry Speed (kps)	Total Mission Time (days)	Total Mission ΔV (kps)
Year	JD*													
1988	7180	200	4.15	0.0	3.48	7.63	7380	0	7380	285	3.86	14.01	485	11.49
	7170	200	4.26	0.0	3.39	7.66	7370	30	7400	300	3.97	14.19	530	11.63
	7165	200	4.32	0.0	3.37	7.70	7365	60	7425	325	4.27	14.40	585	11.97
1990	7850	100	3.90	0.0	3.57	7.47	7950	0	7950	280	3.75	13.72	380	11.22
	7850	100	3.90	0.0	3.57	7.47	7950	30	7980	300	4.06	14.27	430	11.53
	7850	115	3.97	0.0	3.30	7.27	7965	60	8025	335	4.32	15.05	510	11.58
1991	8415	130	3.64	0.0	3.23	6.87	8545	0	8545	285	3.96	13.66	415	10.83
	8415	135	3.64	0.0	3.20	6.84	8555	30	8585	325	4.20	14.36	490	11.04
	8400	135	3.64	0.0	3.20	6.84	8550	60	8610	345	4.19	14.89	540	11.02

ΔV_1 = Earth departure incremental velocity
ΔV_2 = Plane change incremental velocity
ΔV_3 = Planetary orbit insertion incremental velocity
ΔV_{TP} = Total transplanetary incremental velocity
ΔV_{TE} = Planetary departure incremental velocity

Note: Circular parking orbit
Parking orbit altitude = 1000 km

*Julian date -244 0000

Table 5. Mars Aerobraker Baseline Missions (Direct)

Depart Earth		Trip Time (days)	ΔV_1 (kps)	ΔV_2 (kps)	ΔV_{TP} (kps)	Mars Entry Speed (kps)	Arrive Mars (JD*)	Stay Time (days)	Depart Mars (JD*)	Trip Time (days)	ΔV_{TE} (kps)	Entry Speed (kps)	Total Mission Time (days)	Total Mission ΔV (kps)
Year	JD*													
1986	6540	120	3.92	0.0	3.92	9.5	6660	0	6660	250	3.66	14.0	370	7.58
	6540	125	3.84	0.0	3.84	8.9	6665	30	6695	255	4.14	14.7	410	7.98
	6540	120	3.92	0.0	3.92	9.5	6660	60	6720	260	4.54	15.5	440	8.46
1988	7350	125	4.06	0.0	4.06	7.7	7475	0	7475	280	4.17	15.9	405	8.28
	7350	120	4.14	0.0	4.14	8.1	7470	30	7500	270	4.46	16.8	420	8.60
	7350	125	4.06	0.0	4.06	7.7	7475	60	7535	260	4.98	18.6	445	9.03
1993	8900	160	4.39	0.0	4.39	9.1	9060	0	9060	257	4.45	19.8	417	8.84
	8900	140	4.89	0.0	4.89	10.9	9040	30	9070	246	4.73	19.8	416	9.62
	8895	120	6.16	0.0	6.16	13.8	9015	60	9075	239	5.00	19.8	419	11.16

ΔV_I = Earth departure incremental velocity
ΔV_2 = Plane change incremental velocity
ΔV_{TP} = Total transplanetary incremental velocity
ΔV_{TE} = Planetary departure incremental velocity

Note: Circular parking orbit
Parking orbit altitude = 800 km

*Julian date -244 0000

Table 6. Mars Retrobraker Baseline Missions (Direct)

Depart Earth		Trip Time (days)	ΔV_1 (kps)	ΔV_2 (kps)	ΔV_3 (kps)	ΔV_{TP} (kps)	Arrive Mars (JD*)	Stay Time (days)	Depart Mars (JD*)	Trip Time (days)	ΔV_{TE} (kps)	Entry Speed (kps)	Total Mission Time (days)	Total Mission ΔV (kps)
Year	JD*													
1986	6545	180	3.57	0.0	2.49	6.10	6725	0	6725	260	4.62	15.7	440	10.71
	6540	180	3.56	0.0	2.57	6.17	6720	30	6750	260	5.06	16.8	470	11.23
	6545	180	3.57	0.0	2.49	6.10	6725	60	6785	250	5.55	18.2	490	11.64
1988	7335	190	3.75	0.0	2.23	6.02	7525	0	7525	265	4.83	18.2	455	10.85
	7330	190	3.79	0.0	2.28	6.10	7520	30	7550	255	5.23	19.4	475	11.33
	7330	180	3.81	0.0	2.47	6.28	7510	60	7570	240	5.72	19.8	480	12.00
1993	8810	260	5.84	0.0	2.79	8.63	9070	0	9070	246	4.73	19.8	506	13.36
	8780	260	6.86	0.0	2.78	9.64	9040	30	9070	246	4.73	19.8	536	14.37
	8770	240	7.42	0.0	3.33	10.76	9010	60	9070	246	4.73	19.8	546	15.49

ΔV_1 = Earth departure incremental velocity
ΔV_2 = Plane change incremental velocity
ΔV_3 = Planetary orbit insertion incremental velocity
ΔV_{TP} = Total transplanetary incremental velocity
ΔV_{TE} = Planetary departure incremental velocity

Note: Circular parking orbit
Parking orbit altitude = 800 km

*Julian date -244 0000

Table 7. Vesta Baseline Missions

Depart Earth		Trip Time (days)	ΔV_1 (kps)	ΔV_2 (kps)	ΔV_3 (kps)	ΔV_{TP} (kps)	Arrive Vesta (JD*)	Stay Time (days)	Depart Vesta (JD*)	Trip Time (days)	ΔV_{TE} (kps)	Entry Speed (kps)	Total Mission Time (days)	Total Mission ΔV (kps)
Year	JD*													
1985	6070	330	4.35	0.15	5.42	9.82	6410	0	6410	390	5.05	12.34	720	14.87
	6080	320	4.32	0.0	5.25	9.58	6400	30	6430	370	4.87	12.78	720	14.44
	6080	310	4.40	0.0	5.27	9.67	6390	60	6450	350	5.01	12.89	720	14.67
1987	7035	435	6.66	0.0	5.38	12.04	7470	0	7470	290	5.32	14.62	725	17.36
	7035	425	6.74	0.0	5.42	12.15	7460	30	7490	270	5.48	14.63	725	17.63
	7035	435	6.66	0.0	5.38	12.04	7470	60	7530	345	5.72	15.91	840	17.77
1991	8605	380	4.39	1.22	5.53	11.14	8985	0	8985	375	5.11	13.378	755	16.25
	8620	345	5.96	0.0	5.18	11.14	8965	30	8995	365	5.14	13.35	740	16.28
	8620	320	5.96	0.0	5.20	11.16	8940	60	9000	365	5.24	13.54	745	16.41

ΔV_1 = Earth departure incremental velocity
ΔV_2 = Plane change incremental velocity
ΔV_3 = Planetary orbit insertion incremental velocity
ΔV_{TP} = Total transplanetary incremental velocity
ΔV_{TE} = Planetary departure incremental velocity

*Julian date -244 0000

Note: Circular parking orbit
Parking orbit altitude = one planetary radius

Table 8. Ceres Baseline Missions

Depart Earth		Trip Time (days)	ΔV_1 (kps)	ΔV_2 (kps)	ΔV_3 (kps)	ΔV_{TP} (kps)	Arrive Ceres (JD*)	Stay Time (days)	Depart Ceres (JD*)	Trip Time (days)	ΔV_{TE} (kps)	Entry Speed (kps)	Total Mission Time (days)	Total Mission ΔV (kps)
Year	JD*													
	4550	300	8.59	0.0	6.16	14.75	4880	0	4880	415	6.53	19.05	745	21.26
1980	4545	315	8.34	0.0	6.62	15.00	4860	30	4890	410	6.65	19.41	755	21.65
	4540	300	8.09	0.0	7.25	15.34	4840	60	4900	405	6.80	19.80	765	21.14
	7775	385	4.79	4.53	5.29	14.61	8160	0	8160	405	6.87	19.8	800	21.47
1989	7770	380	4.78	4.54	5.43	14.75	8150	30	8180	385	7.17	19.8	795	21.92
	7805	310	7.83	0.0	7.53	15.37	8140	60	8175	390	7.07	19.8	760	22.44
	8260	350	5.10	0.0	6.33	11.43	8610	0	8610	425	7.24	19.8	775	18.67
1991	8255	340	5.26	0.0	6.57	11.83	8595	30	8625	410	7.61	19.8	780	19.44
	8250	330	5.61	0.0	6.88	12.49	8580	60	8640	395	8.07	19.8	785	20.56

ΔV_1 = Earth departure incremental velocity
ΔV_2 = Plane change incremental velocity
ΔV_3 = Planetary orbit insertion incremental velocity
ΔV_{TP} = Total transplanetary incremental velocity
ΔV_{TE} = Planetary departure incremental velocity

*Julian date -244 0000

Note: Circular parking orbit
Parking orbit altitude = one planetary radius

Table 9. Jupiter Baseline Missions

Depart Earth Year	Depart Earth JD*	Trip Time (days)	ΔV_1 (kps)	ΔV_2 (kps)	ΔV_3 (kps)	ΔV_{TP} (kps)	Arrive Jupiter (JD*)	Stay Time (days)	Depart Jupiter (JD*)	Trip Time (days)	ΔV_{TE} (kps)	Entry Speed (kps)	Total Mission Time (days)	Total Mission ΔV (kps)
1985	6173	717	6.55	0.0	6.05	12.60	6890	0	6890	705	6.10	14.4	1422	18.70
	6173	702	6.55	0.0	6.13	12.68	6875	30	6905	690	6.18	14.4	1422	18.86
	6173	687	6.57	0.0	6.22	12.79	6860	60	6920	675	6.27	14.4	1422	19.06
1987	6974	711	6.54	0.0	5.99	12.53	7685	0	7685	712	6.28	14.4	1423	18.81
	6974	696	6.56	0.0	6.07	12.63	7670	30	7700	697	6.39	14.4	1423	19.02
	6974	681	6.59	0.0	6.16	12.75	7655	60	7715	683	6.51	14.4	1424	19.26
1990	8174	716	6.83	0.0	6.29	13.12	8890	0	8890	699	6.56	14.7	1415	19.68
	8174	706	6.84	0.0	6.36	13.20	8880	30	8910	680	6.74	14.7	1416	19.94
	8174	696	6.85	0.0	6.44	13.29	8870	60	8930	660	6.94	14.8	1416	20.23

ΔV_1 = Earth departure incremental velocity
ΔV_2 = Plane change incremental velocity
ΔV_3 = Planetary orbit insertion incremental velocity
ΔV_{TP} = Total transplanetary incremental velocity
ΔV_{TE} = Planetary departure incremental velocity

*Julian date -244 0000

Note: Circular parking orbit
Parking orbit altitude
= 14 Jupiter radii

Table 10. Ganymede Baseline Missions

Depart Earth Year	Depart Earth JD*	Trip Time (days)	ΔV_1 (kps)	ΔV_2 (kps)	ΔV_3 (kps)	ΔV_{TP} (kps)	Arrive Ganymede (JD*)	Stay Time (days)	Depart Ganymede (JD*)	Trip Time (days)	ΔV_{TE} (kps)	Entry Speed (kps)	Total Mission Time (days)	Total Mission ΔV (kps)
1985	6173	717	6.55	0.0	4.89	11.44	6890	0	6890	705	4.94	14.4	1422	16.38
	6173	702	6.55	0.0	4.97	11.52	6875	30	6905	690	5.02	14.4	1422	16.54
	6173	687	6.57	0.0	5.06	11.63	6860	60	6920	675	5.10	14.4	1422	16.73
1987	6974	711	6.54	0.0	4.84	11.38	7685	0	7685	712	5.11	14.4	1423	16.49
	6974	696	6.56	0.0	4.91	11.47	7670	30	7700	697	5.21	14.4	1423	16.68
	6974	681	6.59	0.0	5.00	11.59	7655	60	7715	683	5.32	14.4	1424	16.91
1990	8174	716	6.83	0.0	5.12	11.95	8890	0	8890	699	5.36	14.7	1415	17.31
	8174	706	6.84	0.0	5.18	12.02	8880	30	8910	680	5.52	14.7	1416	17.54
	8174	696	6.85	0.0	5.25	12.10	8870	60	8930	660	5.69	14.8	1416	17.79

ΔV_1 = Earth departure incremental velocity

ΔV_2 = Plane change incremental velocity

ΔV_3 = Planetary orbit insertion incremental velocity

ΔV_{TP} = Total transplanetary incremental velocity

ΔV_{TE} = Planetary departure incremental velocity

*Julian date - 244 0000

Note: Circular parking orbit
Parking orbit altitude = one Ganymede radius

Table 11. Outbound Swingby - Mercury Missions

Earth Departure (Year)	Earth Departure (JD)*	ΔV_1 (kps)	Earth-Venus Trip Time (days)	ΔV_2 (kps)	Arrive Venus (JD)*	ΔV_{swing} (kps)	Venus-Mercury Trip Time (days)	ΔV_3 (kps)	Arrive Mercury (JD)*	ΔV_4 (kps)	ΔV_{TTP} (kps)
1985	6200	4.58	160	0.0	6360	0.45	108	1.93	6468	6.46	13.42
1986	6650	3.87	190	0.0	6840	3.72	74	0.0	6914	4.44	12.03
1988	7210	3.96	180	0.0	7390	2.32	100	0.0	7490	7.27	14.50

Earth Departure (Year)	Arrive Mercury (JD)*	Stay Time (days)	Depart Mercury (JD)*	ΔV_{TE} (kps)	Trans-Earth Transit Time (days)	ΔV_5 (kps)	ΔV_{TTE} (kps)	Arrive Earth (JD)*	ΔV_{entry} (kps)	Mission Duration (days)	ΔV_{total} (kps)
1985	6468	18	6486	6.55	75	0.0	6.55	6561	16.37	361	19.97
1986	6914	42	6956	11.21	85	0.0	11.21	7041	13.91	391	23.24
1988	7490	42	7532	6.96	123	0.0	6.96	7655	15.02	445	21.46

Notes:

ΔV_1 = Trans-planetary injection velocity increment

ΔV_2 = First trans-planet leg plane-change velocity increment

ΔV_{swing} = Velocity increment added during Venus swingby passage

ΔV_3 = Second trans-planet leg plane-change velocity increment

ΔV_4 = Mercury orbit-insertion velocity increment

ΔV_{TTP} = Total trans-planet incremental velocity requirement

ΔV_{TE} = Trans-Earth injection velocity increment

ΔV_5 = Trans-Earth leg plane-change velocity increment

ΔV_{TTE} = Total trans-Earth velocity increment requirement

ΔV_{entry} = Earth entry speed

ΔV_{total} = Total mission incremental velocity requirement

*Julian date — 244 0000

Table 12. Inbound Swingby - Mercury Missions

Earth Departure (Year)	Earth Departure (JD)*	ΔV_1 (kps)	Earth-Mercury Trip Time (days)	ΔV_2 (kps)	Arrive Mercury (JD)*	ΔV_3 (kps)	ΔV_{TTP} (kps)	Stay Time (days)	Depart Mercury (JD)*
1981	4710	6.75	90	0.0	4800	8.19	14.94	54	4854
1987	7008	6.36	115	0.0	7123	10.95	17.31	37	7160
1992	8752	6.83	105	0.0	8857	6.35	13.18	35	8892

Earth Departure (Year)	Depart Mercury (JD)*	ΔV_4 (kps)	Mercury-Venus Trip Time (days)	ΔV_5 (kps)	Arrive Venus (JD)*	ΔV_{swing} (kps)	Venus-Earth Trip Time (days)	ΔV_6 (kps)	ΔV_{TTE} (kps)	Arrive Earth (JD)*	V_{entry} (kps)	Mission Duration (days)	ΔV_{total} (kps)
1981	4854	7.65	80	0.0	4934	1.63	160	0.0	9.28	5094	12.39	384	24.22
1987	7160	7.71	100	0.0	7260	1.43	170	0.0	9.13	7430	12.00	422	26.44
1992	8892	7.65	80	0.0	8972	1.66	160	0.0	9.31	9132	11.63	380	22.49

Notes:

ΔV_1 = Trans-planetary injection velocity increment

ΔV_2 = Trans-planet plane-change velocity increment

ΔV_3 = Mercury orbit-insertion velocity increment

ΔV_{TTP} = Total trans-planet incremental velocity requirement

ΔV_4 = Trans-Earth injection velocity increment

ΔV_5 = First trans-Earth leg plane-change velocity increment

ΔV_{swing} = Velocity increment added during Venus swingby passage

ΔV_6 = Second trans-Earth leg plane-change velocity increment

ΔV_{TTE} = Total trans-Earth incremental velocity requirement

ΔV_{total} = Total mission incremental velocity requirement

*Julian date — 244 0000

Table 13. Outbound Swingby - Mars Aerobraker Missions

Mission Year	Depart Earth JD	Depart Earth ΔV_1 (km/s)	Δt_1 (days)	Venus Swingby JD	Venus Swingby h_p (VR)	Δt_2 (days)	Arrive Mars JD	Arrive Mars V_{EN} (km/s)	Stay Time (days)	Depart Mars JD	Depart Mars ΔV_3 (km/s)	Δt_3 (days)	Arrive Earth JD	Arrive Earth V_{EN} (km/s)	Total Mission Time (days)	Total Mission ΔV (km/s)
1986	6143	3.93	172	6315	0.97	160	6475	9.15	0	6475	1.94	213	6688	11.68	545	5.87
	6141	3.93	174	6315	0.82	150	6465	10.06	30	6495	1.96	202	6697	11.59	556	5.89
	6141	3.92	173	6314	0.73	146	6460	10.59	60	6520	2.12	194	6714	11.58	573	6.04
1993	8509	4.18	161	8670	1.49	155	8825	7.89	0	8825	2.24	290	9115	11.41	606	6.42
	8493	4.28	175	8668	1.45	132	8800	8.17	30	8830	2.25	287	9117	11.41	624	6.53
	8493	4.28	175	8668	1.45	132	8800	8.17	60	8860	2.55	267	9127	11.46	634	6.83
1999	10835	4.16	168	11003	0.63	157	11160	8.51	0	11160	1.98	298	11458	12.84	623	6.14
	10835	4.16	168	11003	0.63	157	11160	8.51	30	11190	2.10	288	11478	13.44	643	6.26
	10835	4.16	168	11003	0.63	157	11160	8.51	60	11220	2.29	278	11498	14.07	663	6.45

JD = Julian date - 244 0000
ΔV_1 = Earth departure incremental velocity
ΔV_2 = Planetary departure incremental velocity
V_{EN} = Entry speed
h_p = Venus swingby altitude in Venus radii

Note: Circular parking orbit
Parking orbit altitude = 800 km

Table 14. Outbound Swingby - Mars Retrobraker

Mission Year	Depart Earth		Δt_1 (days)	Venus Swingby		Δt_2 (days)	Arrive Mars		Stay Time (days)	Depart Mars		Δt_3 (days)	Arrive Earth		Total Mission Time (days)	Total Mission ΔV (km/s)
	JD	ΔV_1 (km/s)		JD	h_p (VR)		JD	ΔV_2 (km/s)		JD	ΔV_3 (km/s)		JD	V_{EN} (km/s)		
1986	6155	4.01	163	6318	1.32	207	6525	3.66	0	6525	2.17	193	6718	11.60	563	9.84
	6153	3.99	164	6317	1.31	203	6520	3.72	30	6550	2.49	190	6740	11.87	587	10.20
	6153	3.99	164	6317	1.31	203	6520	3.72	60	6580	2.72	240	6820	12.94	667	10.43
1993	8512	4.20	157	8669	1.53	151	8820	4.48	0	8820	2.26	294	9114	11.41	602	10.92
	8510	4.18	158	8668	1.54	147	8815	4.49	30	8845	2.33	277	9122	11.42	612	10.98
	8509	4.17	159	8668	1.55	142	8810	4.52	60	8870	2.77	260	9130	11.50	621	11.46
1999	10840	4.18	165	11005	0.81	210	11215	3.31	0	11215	2.25	280	11495	13.98	655	9.73
	10838	4.18	167	11005	0.81	205	11210	3.35	30	11240	2.44	272	11512	14.50	674	9.95
	10838	4.18	167	11005	0.81	205	11210	3.35	60	11270	2.72	260	11530	15.00	692	10.23

JD = Julian date - 244 0000
ΔV_1 = Earth departure incremental velocity
ΔV_2 = Planetary orbit insertion incremental velocity
ΔV_3 = Planetary departure incremental velocity
V_{EN} = Entry speed
h_p = Venus swingby altitude in Venus radii

Note: Circular parking orbit
Parking orbit altitude = 800 km

Table 15. Inbound Swingby - Mars Aerobraker Missions

Mission Year	Depart Earth		Δt_1 (days)	Arrive Mars		Stay Time (days)	Depart Mars		Δt_2 (days)	Venus Swingby		Δt_3 (days)	Arrive Earth		Total Mission Time (days)	Total Mission ΔV (km/s)
	JD	ΔV_1 (km/s)		JD	V_{EN} (km/s)		JD	ΔV_2 (km/s)		JD	h_p (VR)		JD	V_{EN} (km/s)		
1982	4940	3.60	300	5240	5.73	0	5240	4.48	149	5389	1.24	160	5549	12.15	609	8.06
	4931	3.64	284	5215	5.78	30	5245	4.48	145	5390	1.26	160	5550	12.14	619	8.10
	4955	3.62	225	5180	6.78	60	5240	4.48	149	5389	1.24	160	5549	12.15	594	8.08
1988	7354	3.73	181	7535	5.61	0	7535	3.81	210	7745	0.93	160	7905	11.95	551	7.52
	7342	3.76	168	7510	6.00	30	7540	3.82	205	7745	0.97	160	7905	11.95	563	7.58
	7340	3.98	140	7480	7.16	60	7540	3.82	205	7745	0.97	160	7905	11.95	565	7.78
1995	9660	3.85	215	9875	6.95	0	9875	3.44	181	10056	0.74	158	10214	12.24	554	7.28
	9660	3.95	195	9855	7.91	30	9885	3.49	172	10057	0.75	158	10215	12.23	555	7.44
	9660	4.17	175	9835	9.21	60	9895	3.61	163	10058	0.74	159	10217	12.21	557	7.77

JD = Julian date - 244 0000
ΔV_1 = Earth departure incremental velocity
ΔV_2 = Planetary orbit insertion incremental velocity
V_{EN} = Entry speed
h_p = Venus swingby altitude in Venus radii

Note: Circular parking orbit
Parking orbit altitude = 800 km

Table 16. Inbound Swingby - Mars Retrobraker Missions

Mission Year	Depart Earth			Arrive Mars		Stay Time	Depart Mars			Venus Swingby			Arrive Earth		Total Mission Time	Total Mission Total
	JD	ΔV_1 (km/s)	Δt_1 (days)	JD	ΔV_2 (km/s)	(days)	JD	ΔV_3 (km/s)	Δt_2 (days)	JD	h_p (VR)	Δt_3 (days)	JD	V_{EN} (km/s)	(days)	(km/s)
1984	5669	4.02	266	5935	2.85	0	5935	4.25	157	6092	0.75	138	6230	12.58	561	11.12
	5643	4.57	262	5905	3.43	30	5935	4.25	157	6092	0.75	138	6230	12.58	587	12.33
	5622	5.18	263	5885	3.78	60	5945	4.64	158	6103	0.11	141	6244	12.86	622	13.58
1988	7350	3.71	190	7540	2.09	0	7540	3.82	205	7745	1.00	160	7905	11.95	555	9.62
	7344	3.75	171	7515	2.45	30	7545	3.87	200	7745	1.02	160	7905	11.95	561	10.07
	7340	3.81	160	7500	2.84	60	7560	4.20	186	7746	1.04	159	7905	11.95	565	10.85
1995	9660	3.82	240	9900	2.72	0	9900	3.71	159	10059	0.74	159	10218	12.21	558	10.22
	9660	3.82	225	9885	3.16	30	9915	4.17	145	10060	0.70	160	10220	12.19	560	11.13
	9585	5.16	270	9855	3.16	60	9915	4.17	145	10060	0.70	160	10220	12.19	635	12.48

JD = Julian date - 244 0000
ΔV_1 = Earth departure incremental velocity
ΔV_2 = Planetary orbit insertion incremental velocity
ΔV_3 = Planetary departure incremental velocity
V_{EN} = Entry speed
h_p = Venus swingby altitude in Venus radii

Note: Circular parking orbit
Parking orbit altitude = 800 km

requirements. This procedure has a negligible effect on velocity requirements but reduces entry speed by at least 2 kilometers/second. All requirements are based on the parking-orbit altitudes shown on the tables.

The initial analyses of the performance requirements were based on circular planetary parking orbits only. The use of elliptical planetary parking orbits can, however, result in significant reductions in the planetary-orbit insertion and planetary-orbit-escape incremental velocity requirements. The magnitude of the reduction is dependent upon the mass of the central body and the pericenter radius. The most significant reductions will occur when considering orbits of low pericenter altitudes about Jupiter. The effects of eccentricity will be the least significant for Vesta and Ceres because of the low mass of the asteroids.

Under an amendment to the basic contract, the effects of planetary orbit eccentricity on the incremental velocity requirements were investigated for Mercury, Venus, Mars, Jupiter, and Ganymede. The planetary-orbit-insertion and -escape incremental velocity requirements were determined for the baseline missions defined in Tables 2 through 4, 9, 10, and 13 through 16, assuming a thirty-day planetary stay time. Only the Venus swingby mission mode was investigated for Mars missions. Both aerobraking and retrobraking mission modes were investigated for Venus and Mars. For the retrobraking missions, the planetary orbit insertion assumes a cotangential incremental velocity at pericenter of the approach hyperbola. Therefore, the approach hyperbola and the resultant elliptical parking orbit are coplanar with a common pericenter radius. In all cases, a cotangential maneuver is assumed for trans-Earth injection. While it is realized that such maneuvers are not possible in practice, the velocity requirements will be optimistic by a small amount (e.g., 0.5 kilometer/second).

The resultant requirements are shown in Figures 6 through 13 for pericenter altitudes of 300 kilometers for Mercury, Mars, and Ganymede; 500 kilometers for Venus; and 0, 5, 10, and 15 Jupiter radii for Jupiter-orbiter missions. It should be noted that the pericenter altitudes differ from the circular-orbit altitudes used in the generation of the incremental velocity requirements shown in the tables for the circular-orbit missions.

Orbit Stability

A limited orbit-stability study was conducted for orbits about Mercury, Ceres, Vesta, and Ganymede. The purpose of the study was to establish the stability of orbits about these mission objectives prior to the generation of extensive mission-performance data. The perturbations due to Sun, Earth, and Jupiter were considered for orbits about Mercury, Ceres, and Vesta. The Sun was found to be the predominant disturbing body for these

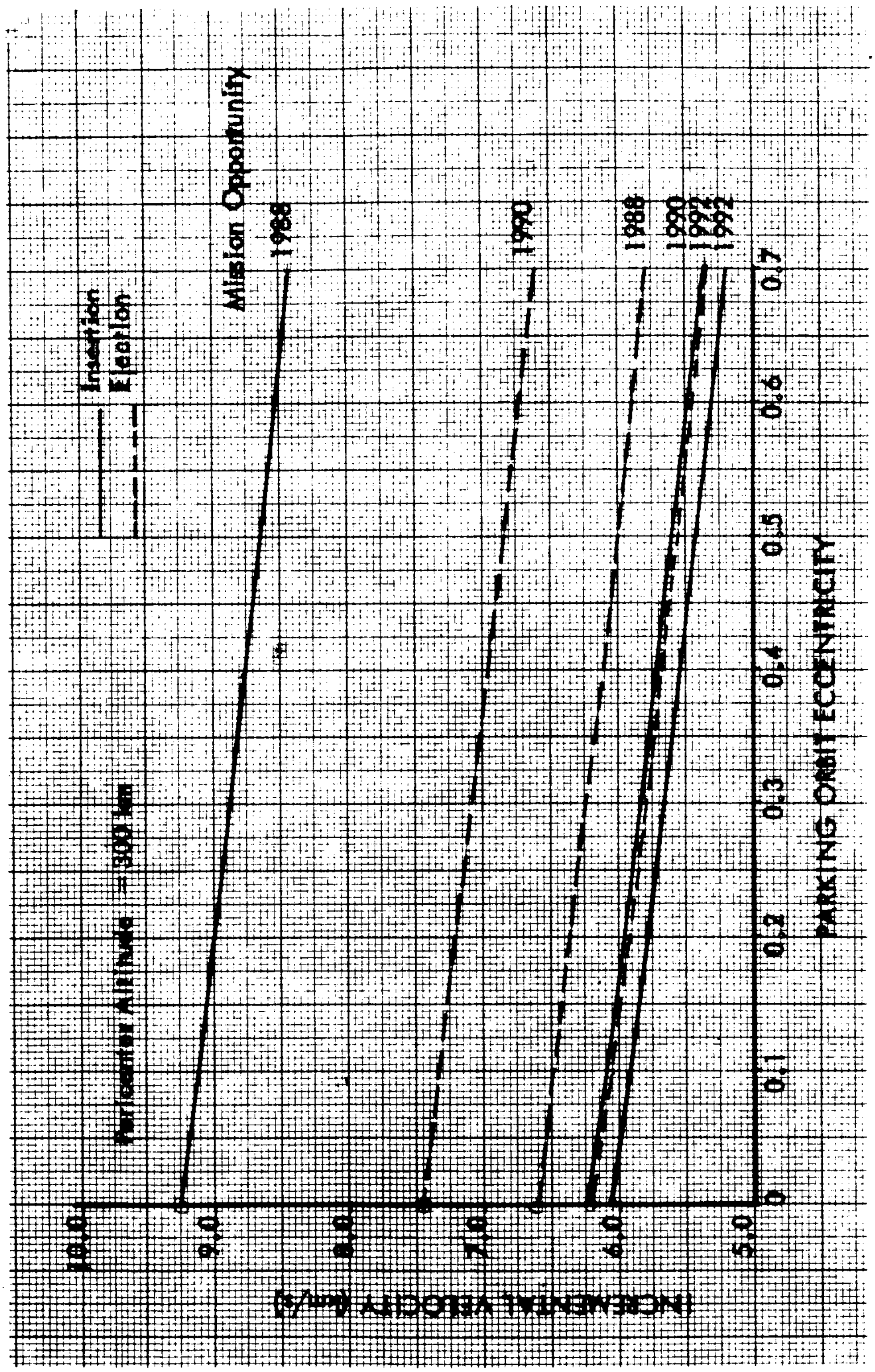

Figure 6. Incremental Velocity Requirements (Mercury Retrobraker)

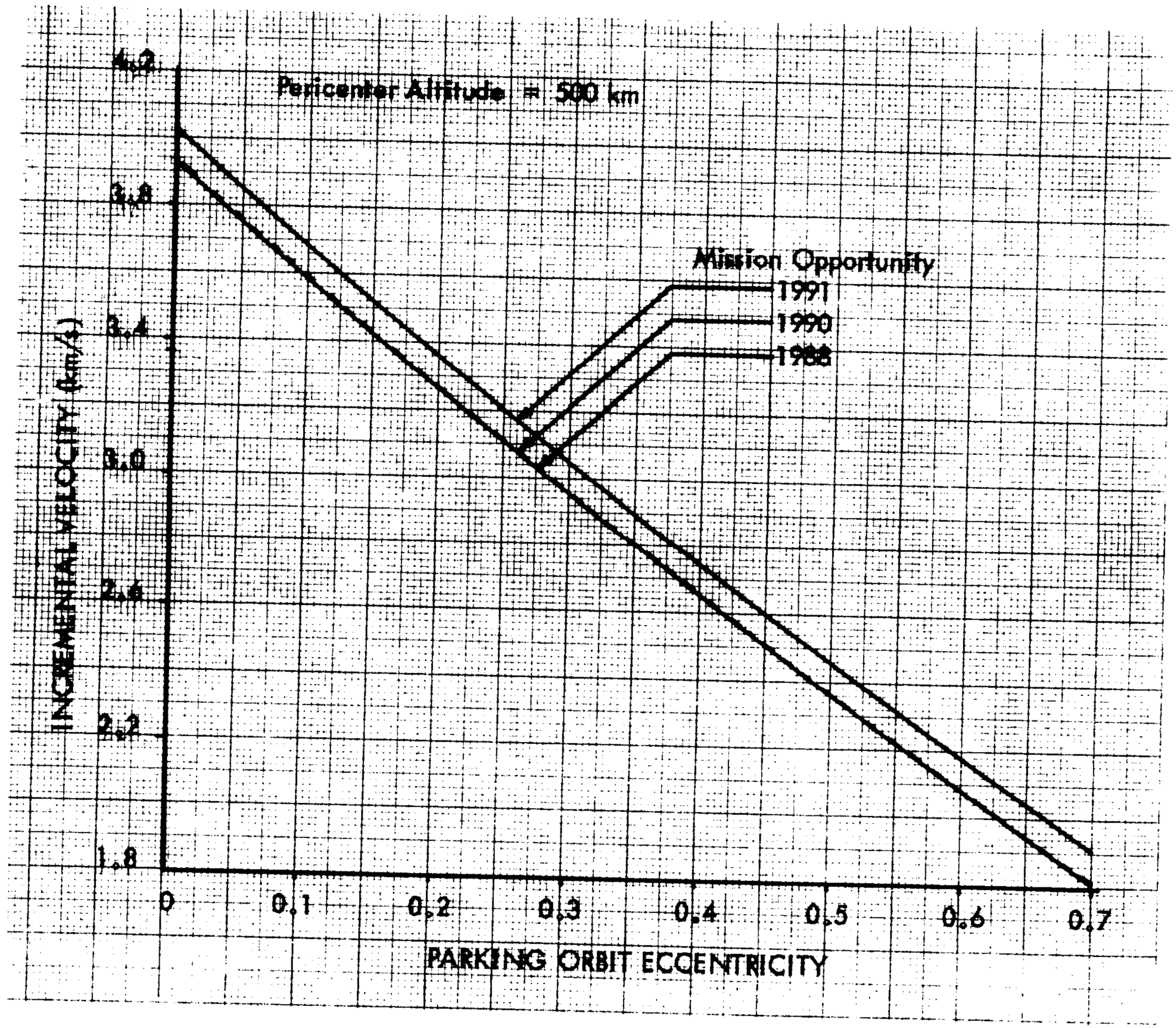

Figure 7. Planetary Orbit Escape Requirements (Venus Aerobraker)

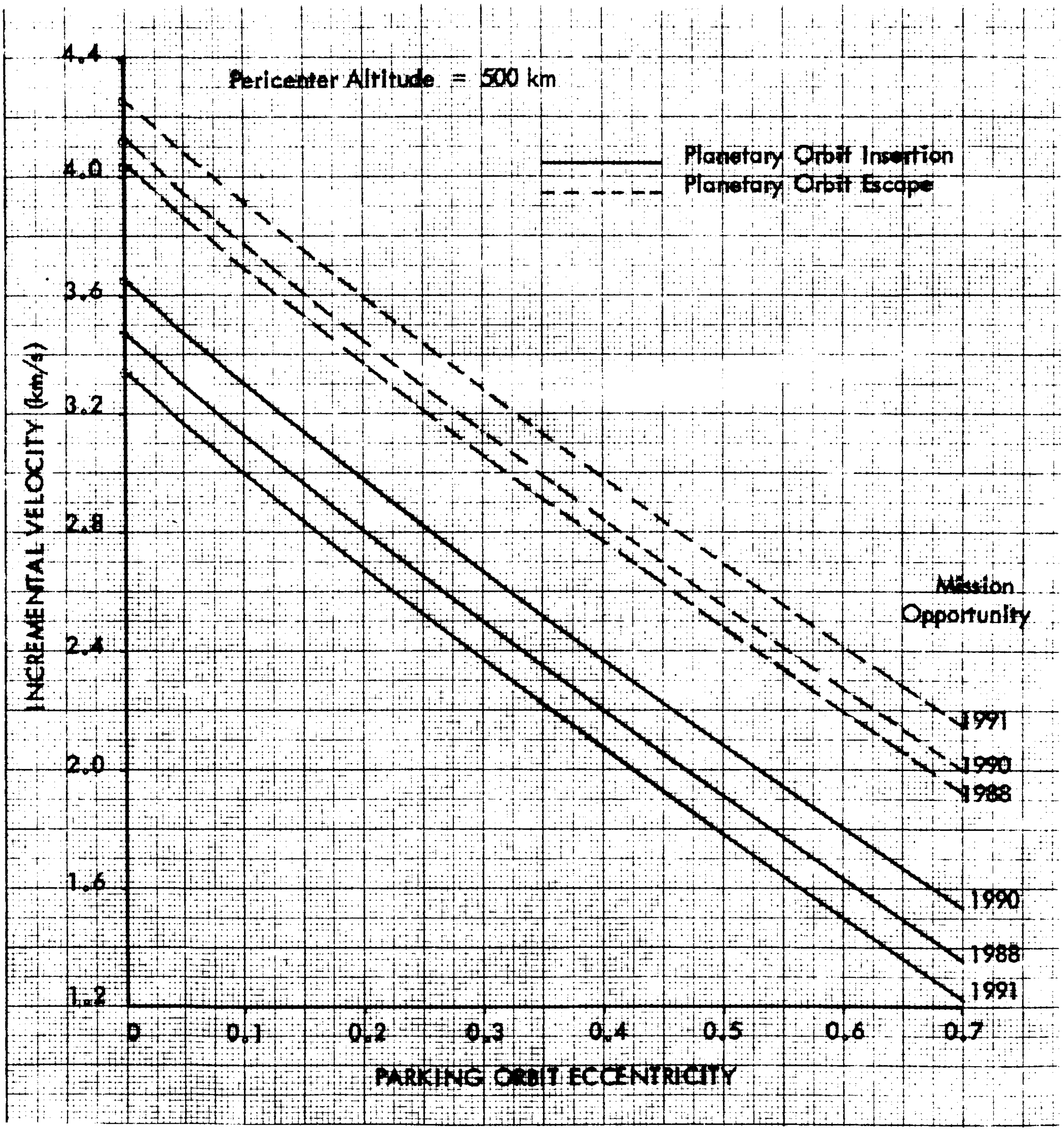

Figure 8. Incremental Velocity Requirements (Venus Retrobraker)

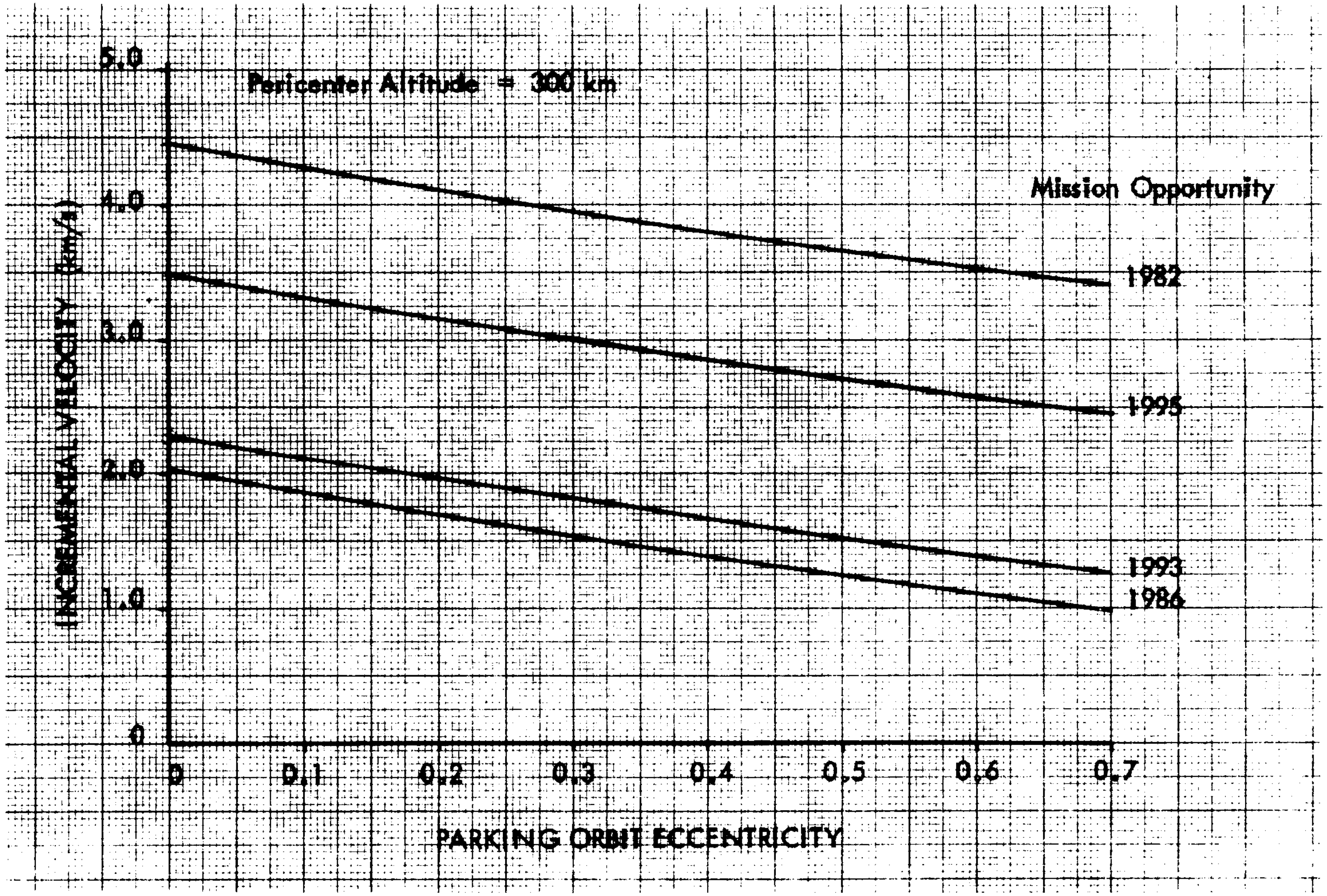

Figure 9. Planetary Orbit Escape Requirements (Mars Aerobraker)

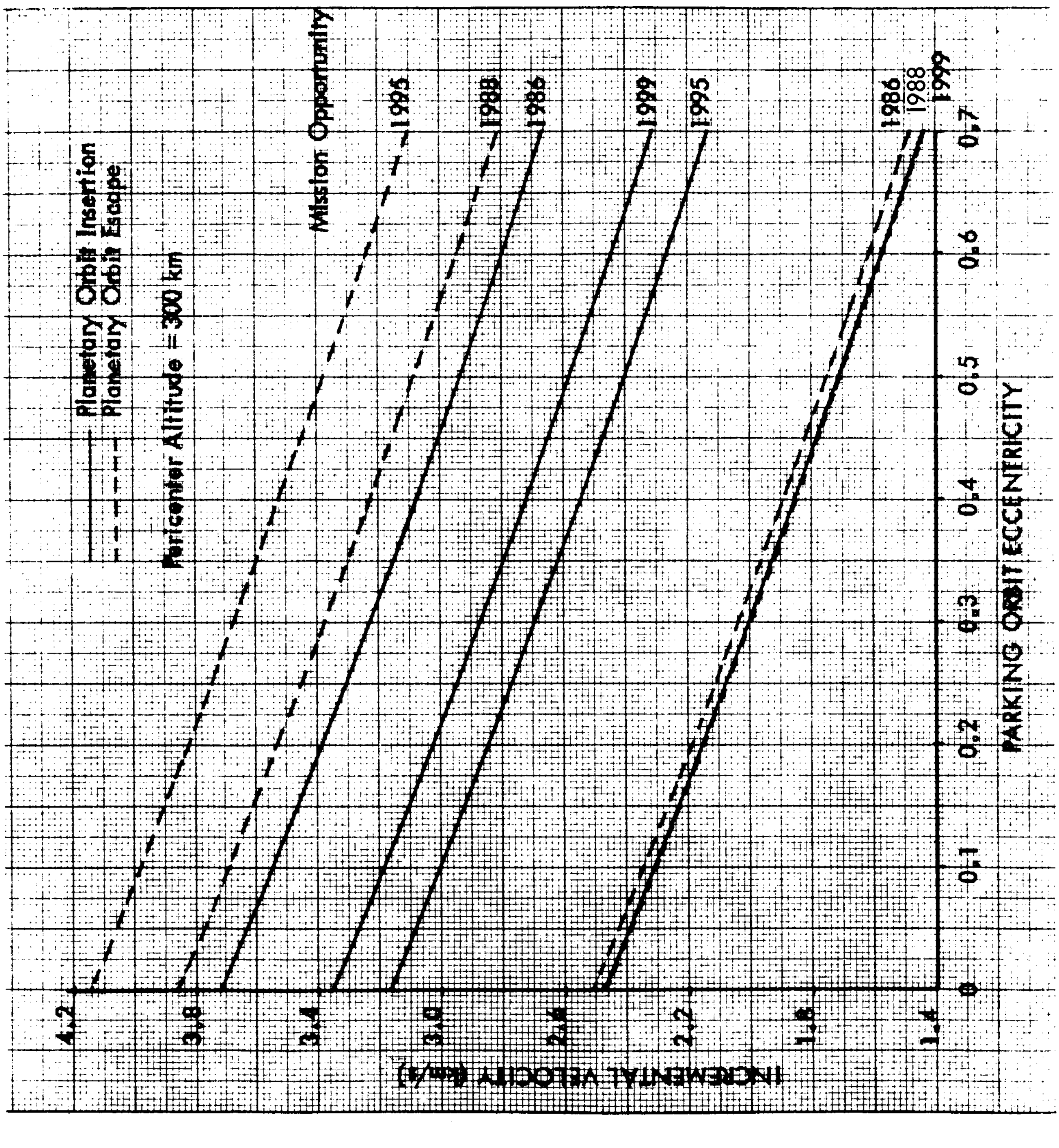

Figure 10. Incremental Velocity Requirements (Mars Retrobraker)

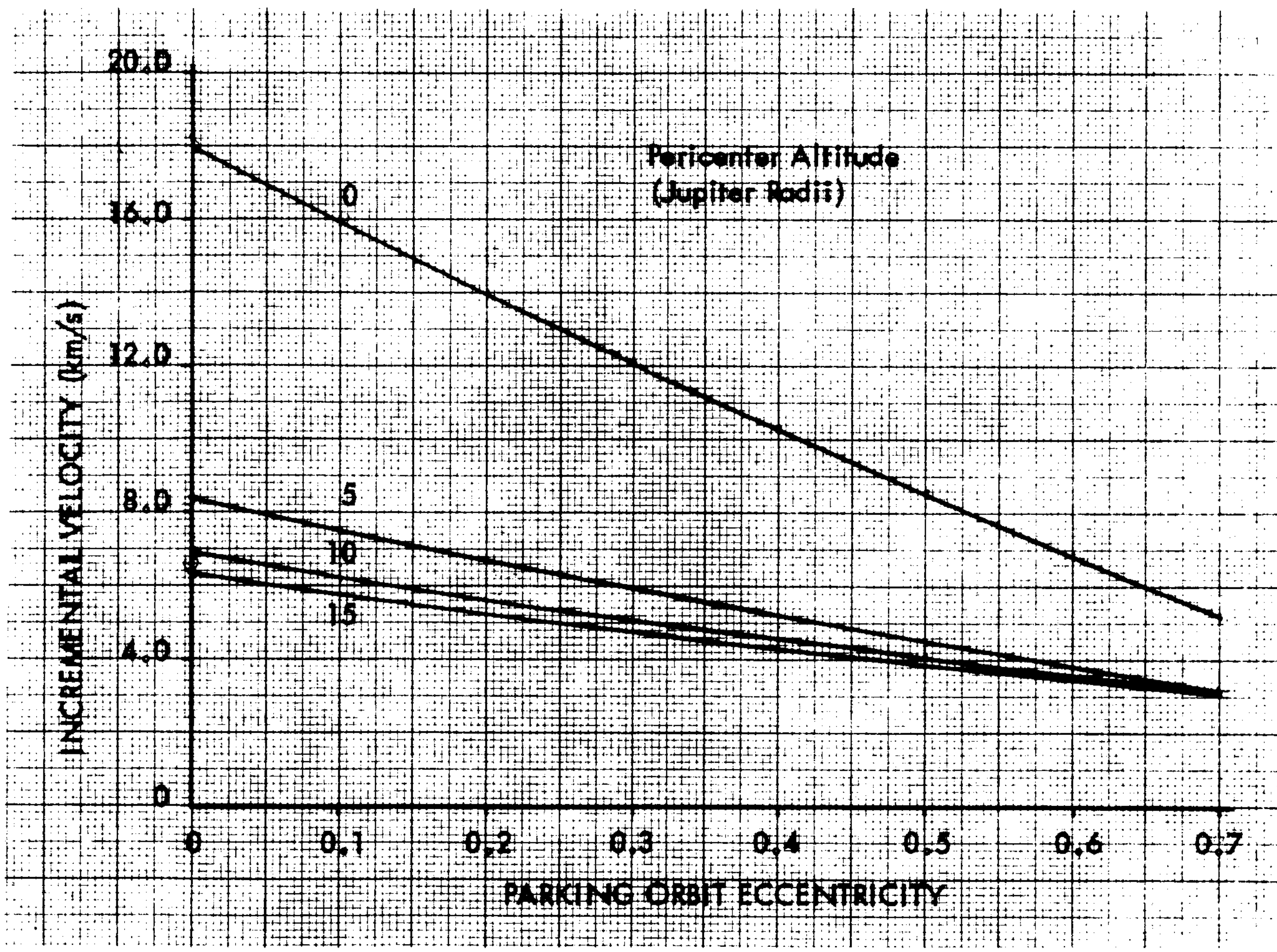

Figure 11. Jupiter Orbit Insertion Requirements (1990 Mission)

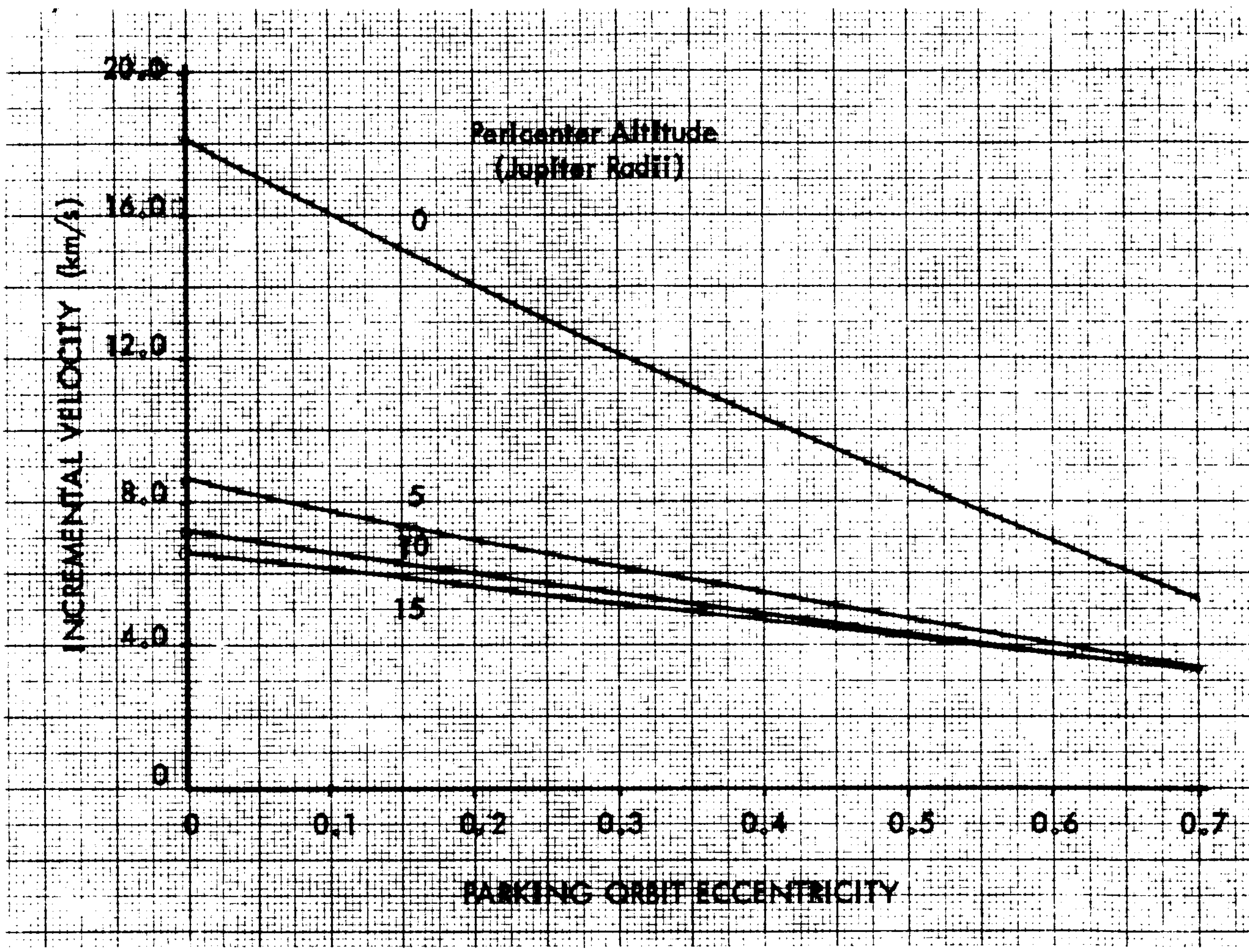

Figure 12. Jupiter Orbit Escape Requirements (1990 Mission)

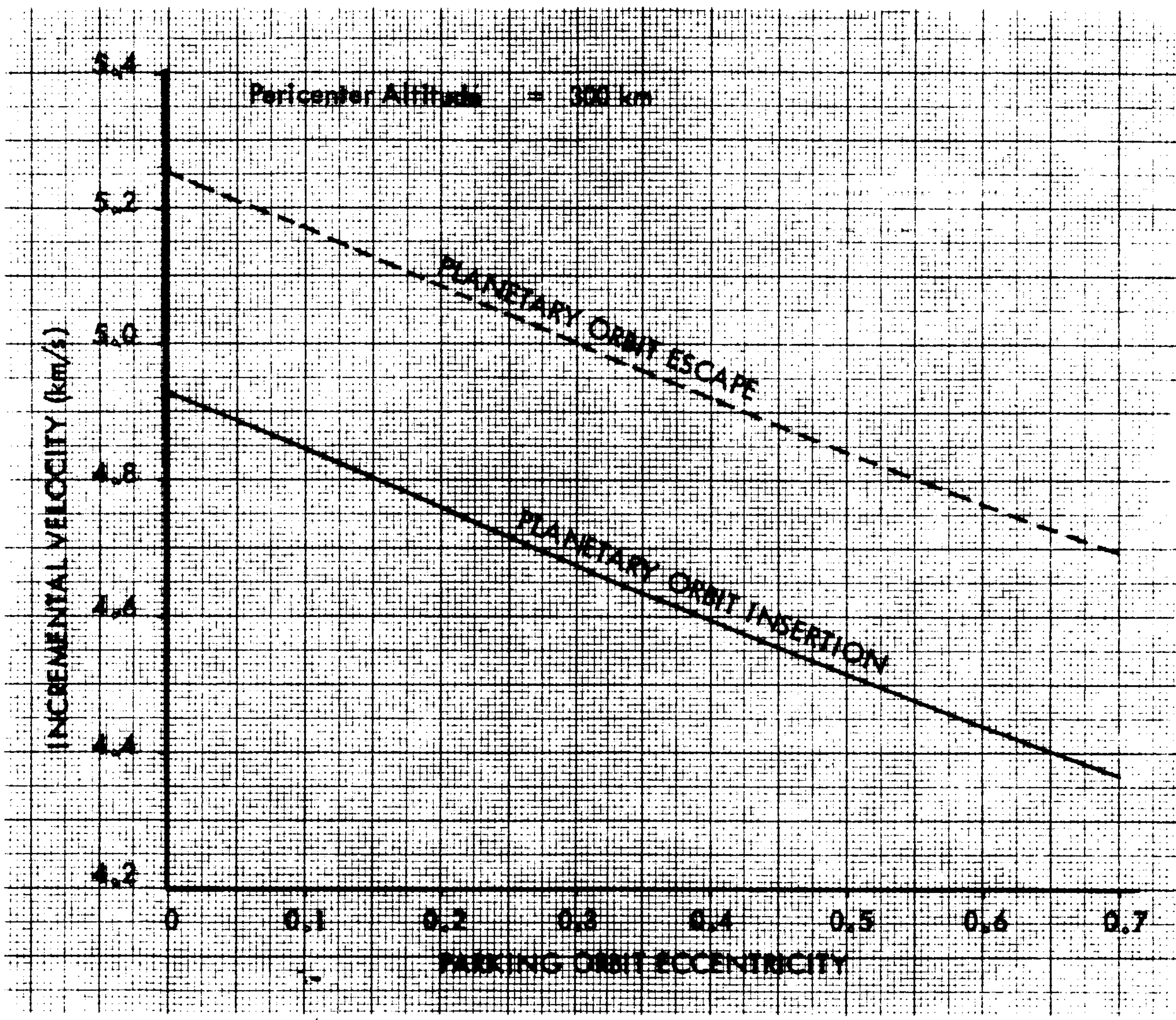

Figure 13. Incremental Velocity Requirements (1990 Ganymede Retrobraker)

cases, and accounted for more than 99 percent of the variations in the orbital elements. The disturbing bodies considered for orbits about Ganymede were Jupiter, Europa, Callisto, and Sun. The effect of Jupiter was predominant, with Europa and Callisto having about equal, but negligible, effect. It was found that the orbits of extended duration about all of the mission objectives considered in the analysis present no significant stability problems for semimajor axes up to two planet or asteroid radii. Variations in orbit shape for orbits about Mercury, Vesta, and Ceres are negligible. Although the disturbing body effects are more significant for orbits about Ganymede, the magnitude of the variations are such that no significant stability problems are apparent.

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 vehicle configuration developed from these studies, was employed in the present study as a baseline for parametric analyses. The configuration was assumed to develop an L/D of 1.0 at a value of C_D of 0.25. Ballistic coefficients ranging from 2400 to 12,200 kilogram/meter^2 were selected for the parametric studies. Entry velocities ranging from 6 to 12 kilometers/second for Mars and 9 to 15 kilometers/second for Venus were chosen as representative.

The results of the study include the aerobraking entry corridors at Mars and Venus as functions of velocity, vehicle M/C_DA and various cut-off criteria, such as maximum deceleration or minimum pull-up altitudes. 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.

Figures 14 and 15 present the heatshield weight fraction as a function of entry velocity for Mars and Venus aerobraking missions. For the Mars entries, the heatshield weight fraction is observed to vary from approximately 6.8 percent to 14.6 percent of the vehicle gross weight at entry for the entry velocity range of 6.1 kilometers/second to 9.2 kilometers/second. The weight fraction for Venus varies from 12.7 percent to 40 percent for

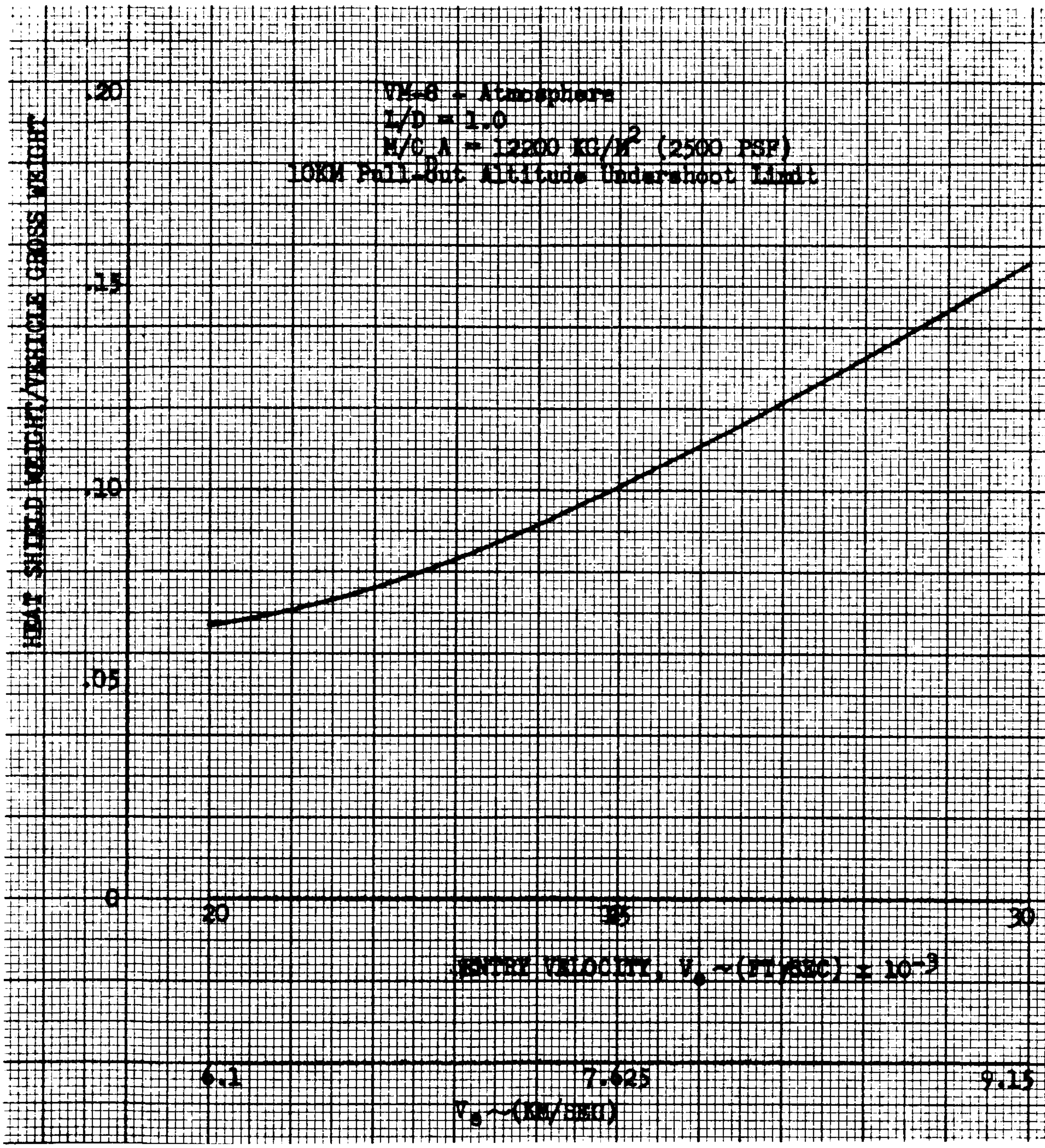

Figure 14. Mars Aerodynamic Braking Heatshield Weight Fraction Variation

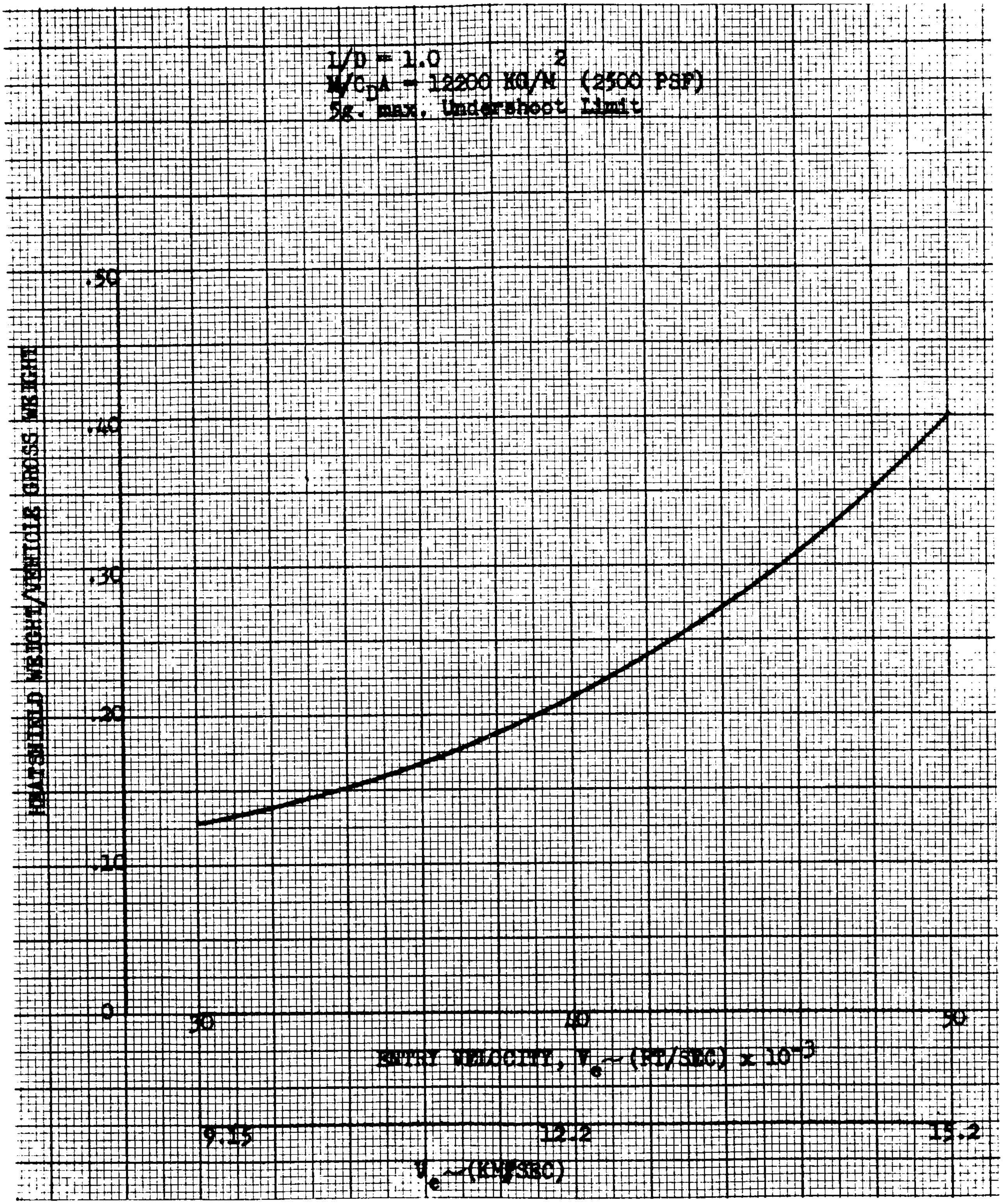

Figure 15. Venus Aerodynamic Braking Heatshield Weight Fraction Variation

the entry velocity range of 9.2 kilometers/second to 15.2 kilometers/second. Measureable design penalties are indicated if for each planet the vehicle is designed for the highest entry velocity. If an aerobraking vehicle with a design entry velocity of 15.2 kilometers/second is applied to a Venus mission requiring an entry velocity of 12.2 kilometers/second, approximately 20 percent of the vehicle weight consists of excess ablative material. Similar weight penalties are indicated for a vehicle design for a Venus aerobraking mission applied to a lower-velocity Mars mission.

A more detailed investigation of the aerobraking technology requirements is currently in progress. The study, "Technology Requirements for Atmosphere Braking to Orbit About Mars and Venus," is being conducted for NASA under Contract NAS2-4135 and is scheduled for completion in January, 1968.

PLANETARY EXCURSION MODULE REQUIREMENTS

The descent and ascent characteristic-velocity requirements were determined for landings on Mercury, Mars, Vesta, Ceres, and Ganymede. The ascent characteristic-velocity requirements for Mercury, Vesta, Ceres, and Ganymede were determined by using calculus-of-variations steering, except for a five-second initial launch phase which utilized an arbitrarily chosen pitchover rate of 2.5 degrees per second. Descent characteristic velocity requirements for these mission objectives were determined assuming a touchdown acceleration of 2.5 times the local acceleration due to gravity. The characteristics of the Mars ascent trajectories were initially determined on the basis of vacuum trajectories utilizing calculus-of-variations steering, except for a five-second vertical boost. The steering profile resulting from the vacuum-trajectory simulation was then used in the atmospheric trajectory simulation. The steering profile was modified, as required, to effect the same ending conditions that were obtained in the vacuum-trajectory simulation. The basic shape was retained, thereby assuring a near-optimum ascent under the influence of drag forces. The atmosphere model used was VM-7.

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 characteristic-velocity requirements include the initial ascent requirements, the requirements for transfer from the burnout conditions to the parking orbit, and the final parking-orbit insertion. The total incremental velocity requirements for descent and ascent, assuming circular planetary parking orbits, are summarized in Table 17. The circular parking orbit altitude was assumed to be one planetary radius in all cases except for Mars, in which case the altitude was 800 kilometers.

Table 17. Planetary Excursion Module Characteristic-Velocity Requirements (Circular Parking Orbits)

Mission Objective	Descent ΔV (meters/second)	Ascent ΔV (meters/second)
Mercury	3830	4000
Mars	1220	4880
Vesta	328	328
Ceres	556	565
Ganymede	2470	2700

The effects of planetary-parking-orbit eccentricity on the planetary excursion module characteristic-velocity requirements were also established. The resultant requirements are presented in Figures 16 through 18 for Mercury, Mars, and Ganymede missions, respectively. The descent profile for Mercury and Ganymede consists of a Hohmann transfer from apocenter of the elliptical parking orbit to circular orbit speed at the altitude at which the terminal descent is initiated. The ascent profile consists of an initial ascent to circular orbit followed by a Hohmann transfer and tangential injection at pericenter of the elliptical parking orbit. A pericenter altitude of 300 kilometers was used in all cases which, it should be noted, differs from the altitudes used during the circular-orbit analyses. In general, both the ascent and descent velocity requirements, and thus the planetary excursion module mass requirements, increase with increasing eccentricity. The only exception is the descent requirement for Mars missions which decrease with increasing eccentricity because atmospheric braking is used.

GUIDANCE AND NAVIGATION REQUIREMENTS

The incremental velocity requirements for establishing a circular orbit about Ganymede were determined for two possible mission profiles. The first profile assumes the spacecraft is injected into an orbit about Jupiter and, after a coast period to attain the proper phase angle, is injected into a transfer orbit that results in a Ganymede-centered orbit with the required perifocal radius. At Ganymede perifocus, a third propulsive maneuver is required for injection into orbit about Ganymede. The second profile, which requires only one propulsive maneuver, is a direct injection into orbit about Ganymede from the Jupiter and Ganymede approach hyperbola. It was determined that the direct injection profile results in an

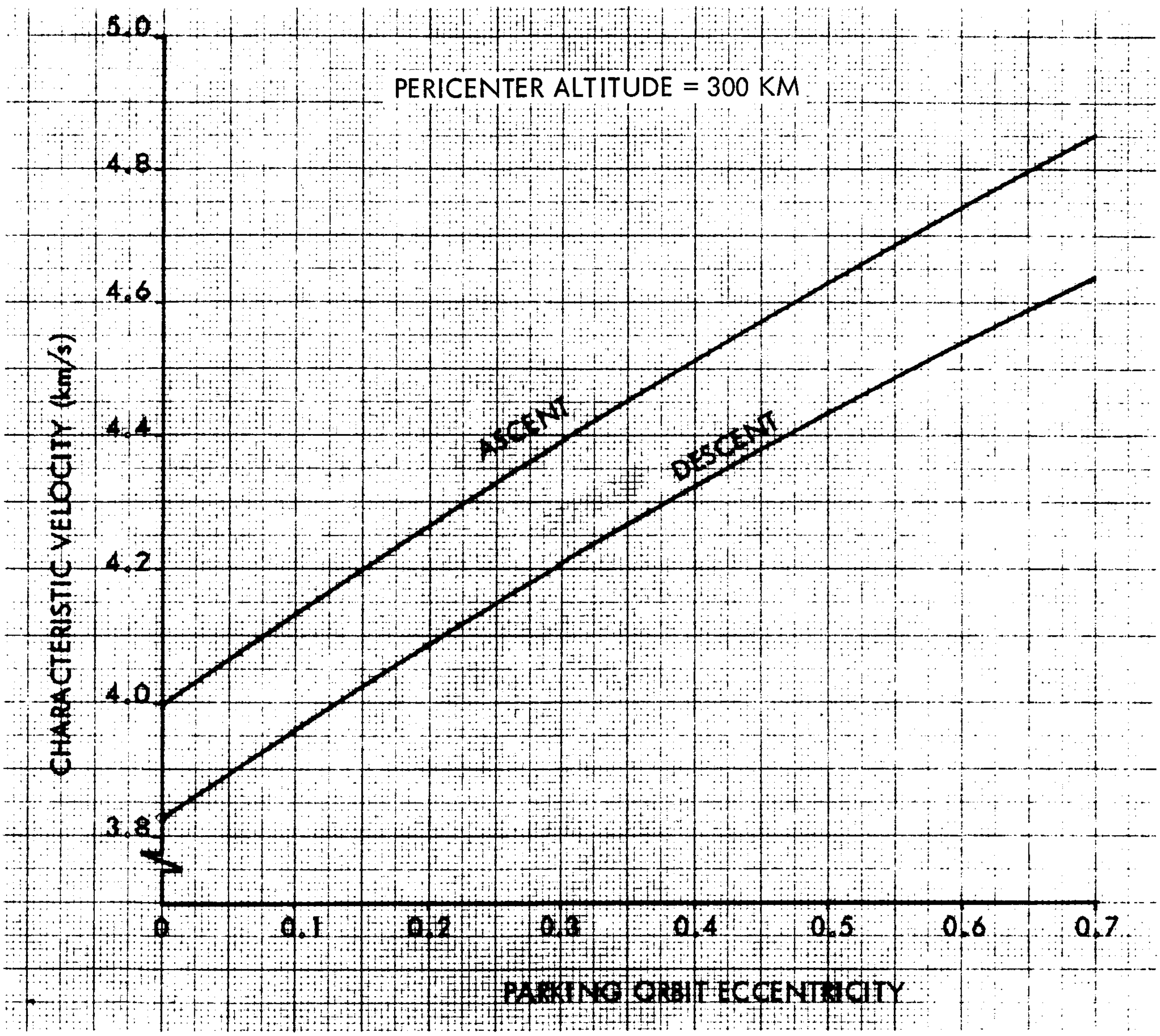

Figure 16. PEM Characteristic Velocity Requirements, Mercury

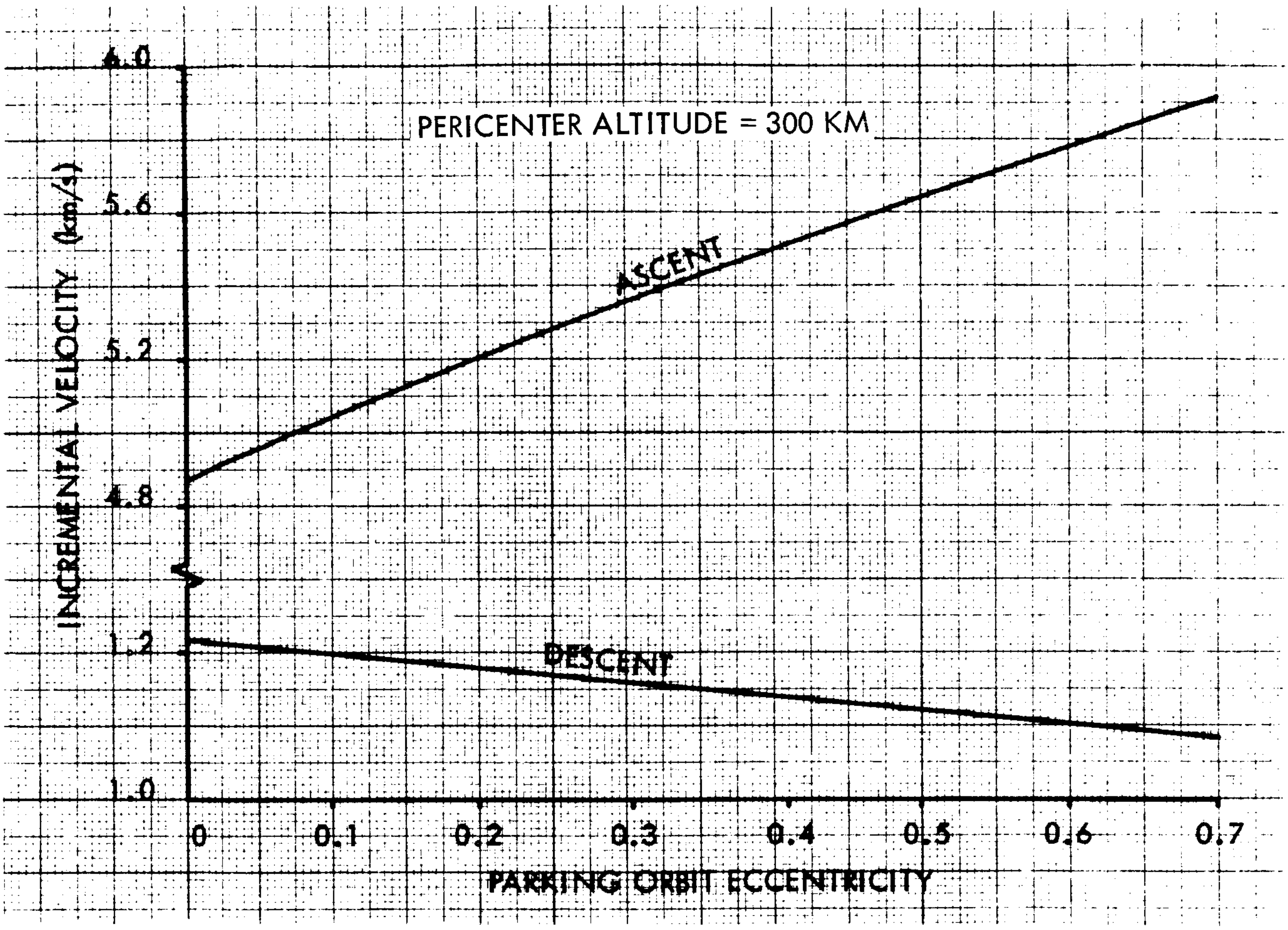

Figure 17. PEM Characteristic Velocity Requirements, Mars

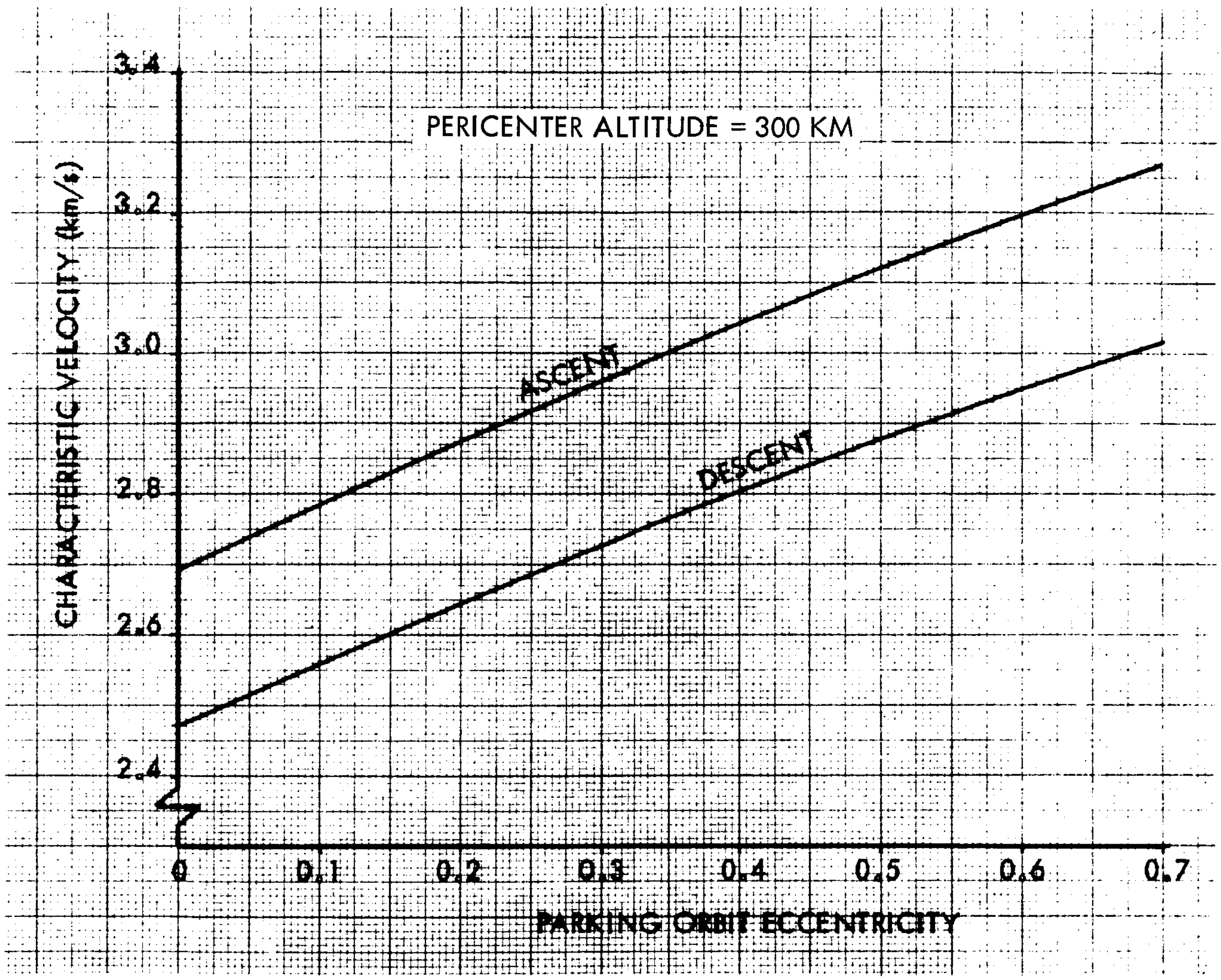

Figure 18. PEM Characteristic Velocity Requirements, Ganymede

incremental velocity savings of approximately 1.56 kilometer/second when compared with the indirect profile. The velocity savings is based on a Jupiter circular phasing orbit radius of 15 Jupiter radii (radius of Ganymede) and a Ganymede circular parking orbit radius of two Ganymede radii.

A limited study was conducted to determine the effects of the midcourse guidance requirements on the selection of the mission profile. The objective of the investigation was to determine whether or not the midcourse correction requirements for the direct-orbit-insertion profile would exceed the savings in the nominal mission performance requirements and thus invalidate the premise that the direct-insertion profile minimizes the total mission incremental velocity requirements. It was determined that the savings associated with the nominal requirements for the direct-insertion profile exceed the additional midcourse correction requirements. Thus, the direct-injection profile appears to be a promising mission concept and was used during subsequent analyses.

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. The environmental models and scaling equations used in the determination of the environmental protection requirements are summarized in the following paragraphs. Detailed discussions are contained in Appendix B.

METEOROID ENVIRONMENT AND PROTECTION

Meteoroid protection will be required for all manned modules and propulsion modules. The extent of the protection for each module will be dependent upon the mission objective, mission mode, and the assumptions made regarding the meteoroid environment, penetration mechanics, and damage criteria.

Meteoroid Environment

Cometary, nominal asteroidal, and maximum asteroidal meteoroid flux models were provided by the Mission Analysis Division (MAD). The characteristics of these models are given in Table 18.

Penetration Mechanics

The penetration mechanics for quasiinfinite and single-sheet structures are discussed in Appendix B. The equation defining the penetration depth (p) in centimeters is given by

$$p = \frac{1.38 d_p^{1.1} \rho_p^{0.5} V_p^{2/3}}{\rho_t^{1/6} H_t^{1/4}} \text{ centimeters}$$

where

d_p = particle diameter (centimeters)

ρ_p = particle density (grams/centimeter^3)

V_p = particle impact velocity (kilometers/second)

ρ_t = target density (grams/centimeter^3)

H_t = target Brinell hardness number (kilograms/millimeter^2)

Table 18. Meteoroid Flux Models

Environment	Flux	Particle Density (gm/cc)	Particle Impact Velocity (km/sec)
Cometary	$\text{Log}\,\phi_c = -14.44 - 1.34\text{Log}\,m$	0.5	$30r^{-1/2}$
Nominal asteroidal	$\text{Log}\,\phi_a = -17.27 + 3.44r$ $-0.61r^2$ $-0.73\text{Log}\,m$	3.5	$15r^{-1/2}$
Maximum asteroidal	$\text{Log}\,\phi_a = -21.00 + 8r$ $-1.43r^2$ $-0.91\text{Log}\,m$	3.5	$15r^{-1/2}$

Notes:

ϕ = number of particles/meters^2-second

m = particle mass (grams)

r = heliocentric radius (A.U.)

Meteoroid Shielding and Damage Criteria

Structural models, damage criteria, and the placement of the meteoroid shielding were adopted for each of the modules in order to define the shielding requirements. The damage criteria and the placement of the meteoroid shielding are summarized in Table 19.

Table 19. Meteoroid Damage Criteria

Module	Damage Criteria	Placement of Shielding
Earth reentry module	No perforations of module shroud to prevent loss of pressurization[1]	Increase thickness of module shroud
Mission module	No perforations of cabin wall to prevent loss of pressurization	Increase thickness of cabin wall
Planetary excursion module	No perforations of module wall to prevent loss of pressure-vessel integrity	Increase thickness of module shroud[2]
Propulsion modules	No perforations of load carrying wall to prevent high-energy impact on propellant tank	Increase thickness of meteoroid protection[3]
Aerobraker heatshield	Limit penetration into ablator to full depth of ablator	Increase thickness of heatshield

Notes:

[1] Earth reentry module is assumed to be housed in a pressurized shroud to prevent ablator outgassing.

[2] Planetary excursion module is assumed to be housed in a load-carrying shroud.

[3] Propulsion module meteoroid protection is assumed to be provided by a separate structure which is jettisoned prior to ignition.

Meteoroid Protection Scaling Equation

The scaling equation, which was used to determine the total structural unit mass required for meteoroid shielding, is given by

$$W_{ti} = C_1 + C_2 \, (A \cdot T)^{\alpha}$$

where A is the vunerable area, T is the exposure time and the parameters C_1, C_2, and α are dependent upon the module, mission objective, mission mode, and environment (nominal or maximum). The values of these parameters were determined by optimizing the allocation of the shielding to the modules by use of Lagrange's method of the undetermined multiplier. The resultant values of C_1, C_2, and α are given in Appendix B for all mission objectives, modules, and flux models.

THERMAL ENVIRONMENT AND PROTECTION

A thermal protection and environmental control system is required for the mission module in order to maintain a balance between the heat loads (both internal and external) and heat losses. Thermal protection is also required for the propulsion modules in order to permit long-term storage of propellants. The analyses which were conducted to establish the thermal protection requirements for these modules are summarized in the following paragraphs.

Mission Module

The required weight for thermal-insulation and heat-rejection systems for the mission module were determined for missions to Mercury and Jupiter, considering crew sizes from three to twenty men. The internal heat sources considered in the heat balances were the crew metabolic heating, life support and environmental control subsystem, and electrical loads. The electrical power subsystem was considered to be independent of the mission module and was not involved in the heat balances except for the energy dissipation within the module, e.g., illumination. The only external heat load considered was that of direct radiation from the sun.

The effect of the optical-thermal properties of surface coatings on the surface temperatures of the mission module was examined as a function of heliocentric distance. It was determined that the preferred surface coating would be one which provides the lowest solar-absorptivity to thermal-emissivity ratio within practical constraints. In this manner, it is possible to isolate the effects of internal and external heat sources.

The insulation system requirements were established on the basis of minimizing the effects of external heat sources and sinks on the thermal balance within the mission 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 in this study while maintaining the external heat gain or heat loss to less than 10 percent of the internal heat dissipation. While no attempt has been made to optimize the weight penalties involved, one benefit from this concept is that a water-glycol ECS is not likely to freeze (load stays relatively constant) and another is that the selected insulation thickness would prevent water-vapor deposition on the module surfaces. Ideally, about ten layers of multilayer superinsulation will be sufficient, which corresponds to an insulation mass of approximately 0.49 kilograms/meter^2 of module surface area (0.1 pounds/feet^2). The resultant radiator area would be, for example, approximately 31.5 square meters (about 15 percent of the total module area) for a crew size of twenty men, assuming an internal heat load of 10 kilowatts. The corresponding radiator weight would represent only one percent of the total module mass.

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 an absolute necessity or it will be necessary to provide shadow shielding of the ECS radiator to prevent direct solar heating. The radiator will be unable to reject heat if exposed either continuously or cyclically to direct solar heating.

Propulsion Modules

A study of the propellant storability and thermal protection requirements was conducted to develop propellant-boiloff and insulation-thickness weight scaling equations and to examine typical insulation-mass requirements for long-term propellant storage. Both the factors of heat transfer into the tank and heat storage within the propellant were examined. The examination included both no-loss type of storage and evaporative-storage techniques. For the no-loss storage of cryogens, pressure rises of 14.7 to 90 psia and 50 percent slush to 90 psia were used to establish the allowable heat budget. In addition, the insulation requirements for total evaporation of 5, 10, and 20 percent of the total propellant were examined.

An initial examination was conducted to determine where potential propellant boiling and freezing will occur for the propellants considered during the study. The propellants considered were monomethylhyrazine (MMH), methane (CH_4), diborane (B_2H_6), hydrogen (H_2), oxygen difluoride (OF_2), FLOX, and oxygen.

At Jupiter, MMH is well below its freezing condition. Insulation must be added to prevent heat from being lost or it may be necessary to add heat to the system. The B_2H_6 and CH_4 may possibly freeze if left at Jupiter for a long time, so insulation would also be required. Oxygen, FLOX, and OF_2 are all storable propellants. Liquid hydrogen will boil off, but the temperature differential is small.

At 3 A.U., or approximately the asteroid belt, OF_2 and CH_4 are storable, oxygen and FLOX will boil-off slowly even at 90 psia, liquid hydrogen is only slightly changed from Jupiter, and B_2H_6 is near its freezing point. For B_2H_6, this is approximately the limit of storability, and MMH is still likely to freeze.

Between Earth and Mars, all of the oxidizers and CH_4 are well above their boiling points at 90 psia; therefore, insulation is always required for closer approach to the Sun. The B_2H_6 is perhaps storable in this region, but MMH still requires insulation to prevent freezing. At Mercury, all of the fuels and oxidizers are cryogens with the exception of MMH which is storable at pressure slightly above the normal boiling point and below 90 psia.

Weight scaling equations were developed for the optimization of the propulsion-module insulation thickness and boil-off propellant requirements. The basic assumptions required for the development of the scaling equations were: the thermal conductivity could be expressed analytically as a function of temperature, the interior surface temperature of the propellant tanks is equal to the fuel or oxidizer bulk temperature at the boil-off temperature, and the propulsion module surface temperature is equal to the equilibrium wall temperature. The resultant scaling equations define the optimum insulation thickness and boil-off propellant as a function of the mission, propellant, and insulation characteristics. The optimum insulation thicknesses (d_{opt}) for a two-stage monopropellant system are given by

$$d_{1_{opt}} = \sqrt{\frac{K_1}{\mu_1 \rho_{INS} L}}$$

and

$$d_{2_{opt}} = \sqrt{\frac{K_1 + \mu_1 K_2}{\mu_1 \mu_2 \rho_{INS} L}}$$

where

K_i = a function of the mission characteristics and the propellant boil-off temperature.

μ_i = mass ratio ($e^{\Delta V_i / Ig}$)

ρ_{INS} = insulation density

L = heat of vaporization

The optimum boil-off propellant requirements ($W_{B_{opt}}$) are given by

$$W_{B_{1_{opt}}} = A_1 \sqrt{\frac{\mu_1 K_1 \rho_{INS}}{L}}$$

and

$$W_{B_{2_{opt}}} = A_2 (K_1 + K_2) \sqrt{\frac{\mu_1 \mu_2 \rho_{INS}}{(K_1 + \mu_1 K_2) L}}$$

where A_i is the insulated area.

The above scaling equations can be extended for any number of stages and to include the case of bipropellant propulsion stages. The extension of the above equations was employed in the computation of insulation thickness and propellant boil-off requirements during the weight-synthesis analyses.

Data were prepared to illustrate the propellant tank insulation requirements for missions to Mercury and Jupiter. The results include an additional 50-percent heat transfer as an estimate of the effects of structural supports attaching the insulation to the module structure. Also, an absorptivity-to-emissivity ratio of 0.2 was assumed. In general, the insulation requirement is no greater than 2.5 centimeters (1 inch) even for the most cryogenic application of superinsulations. It is significant that similar amounts of insulation on a weight-per-unit area basis are required to keep MMH from freezing during transfers to Jupiter (Ganymede PEM) as are required for keeping liquid hydrogen from boiling on a mission to Mercury.

RADIATION ENVIRONMENT AND PROTECTION

Two separate analyses were performed to determine the effects of the radiation environment on the spacecraft design. The first investigation concerned the space radiation environment which must be considered for all missions. The investigations resulted in analytical expressions which define the shielding requirements in terms of the environmental and mission characteristics. The second investigation considered the effects of the Jupiter-trapped radiation, which is of concern for missions to Jupiter and its satellites.

Space Radiation

The analysis of the space radiation can be carried out by two different methods. One is to calculate the expected solar environment for each mission being considered from statistical correlations obtained from past solar events. This technique provides the most accurate expected particle fluxes and doses possible from the available data. The other method is to obtain analytical representations between solar and mission parameters which can be combined to yield mission fluxes and/or doses. The second approach was used in the present study because of the large number of mission objectives and mission opportunities which were considered.

The development of an analytical representation of the space radiation environment incorporated those factors known to have major effects upon the mission dose while neglecting factors considered to be less important. The factors considered were solar radiation, Van Allen radiation, and galactic radiation and are summarized in the following paragraphs. The detailed development, required approximations, and the effects of the approximations are discussed in Appendix B.

Solar-Flare Radiation

Solar-flare radiation is usually treated statistically, since our knowledge of the physical mechanisms involved does not currently permit a deterministic treatment. However, it is possible to approximate the proton-flux probability (P) as a function of the mission flux and mission-time period relative to the 11-year solar cycle. It is assumed that the probability of receiving a given solar particle flux at solar minimum is 0.1 the corresponding probability at solar maximum, with an approximate sinusoidal behavior in between. The expressions which were obtained are based on the past solar cycle (Cycle 19). Therefore, they can be applied to future solar cycles only if it is assumed that future cycles will be like the last one. Since the last cycle was the most active ever observed, the assumption is believed to be conservative when applied to future solar cycles.

The intensity of solar-flare radiation must decrease as some function of the heliocentric distance (r). The analyses conducted during the present study assumed that the event probabilities are independent of heliocentric distance, but that the particle fluxes decrease as r^{-2}.

Analytical expressions were obtained for the solar-flare radiation dose (including proton and alpha-particle effects) as a function of shield thickness. Combining these with the assumed r^{-2} dependence on heliocentric distance and the statistical flare-probability function the mission biological dose was determined as a function of these parameters. The resultant expressions neglect secondary radiations produced by nuclear interactions initiated by the solar-flare protons and alpha particles.

Van Allen Radiation

The spatial, temporal, and energy distributions of the geomagnetically trapped radiation (Van Allen belts) have been investigated extensively. As a result it is possible to calculate rather accurately the particle (electron and proton) fluxes and doses expected along any trajectory. For the high-thrust propulsion systems considered in this study, the point rad dose is fairly small unless the shield thickness is less than 1 gram/centimeter^2. For deep-space missions undertaken at solar maximum, the trapped radiation contributes less than three percent of the mission dose, and it is often less than one percent. For missions at solar minimum, the contribution may increase to approximately five percent, with three percent being far more common. It was thus assumed that the geomagnetically trapped radiation dose amounted to three percent of the solar radiation dose for all missions.

Galactic (Cosmic) Radiation

Many uncertainties exist concerning the characteristics of the galactic radiation. Near the Earth, flux and dose measurements have been made which extrapolate to a deep-space value of approximately fifty millirads per day (quiet Sun). The extrapolated deep-space value was used as a conservative value, since any perturbing influences will decrease this value.

Space-Radiation Shielding Requirements

Mission dose limits are usually specified for one or more of the critical human organs such as the eyes, skin, bone marrow, central nervous system, or reproductive organs. Skin and bone marrow doses (for solar and geomagnetically trapped radiation) are related to point doses by

$$(\text{Skin Dose})\ (X) = (0.5)\ (\text{Point Dose})\ (X)$$

and

$$(\text{Bone Marrow Dose})\ (X) = (0.5)\ (\text{Point Dose})\ (X + 5)$$

where X is the shield thickness (weight per unit area).

The equivalent doses for other critical organs were not considered since the organs are more localized and can be protected by special shielding without appreciable weight penalty. For shield thickness (X) less than approximately 5 grams/centimeter^2, the skin dose is usually the determining factor, but the bone-marrow dose becomes dominant if the thickness is greater than approximately 5 grams/centimeter^2. The resultant equations which define the aluminum equivalent shield thickness (X) in terms of the skin and bone marrow dose limits (D) are given by

$$\left.\begin{aligned} X &= 26 \left[\left(\frac{1.03t}{2D_{skin} - 0.35t}\right)\left(\frac{\overline{1}}{r^2}\right)\right]^{0.77} A^{1.54} \\ X + 5 &= 26 \left[\left(\frac{1.03t}{2D_{marrow} - 0.35t}\right)\left(\frac{\overline{1}}{r^2}\right)\right]^{0.77} A^{1.54} \end{aligned}\right\} \text{D in rad}$$

$$\left.\begin{aligned} X &= 59 \left[\left(\frac{1.03t}{2D_{skin} - 0.35t}\right)\left(\frac{\overline{1}}{r^2}\right)\right]^{0.625} A^{1.25} \\ X + 5 &= 59 \left[\left(\frac{1.03t}{2D_{marrow} - 0.35t}\right)\left(\frac{\overline{1}}{r^2}\right)\right]^{0.625} A^{1.25} \end{aligned}\right\} \text{D in rem}$$

where

$\left(\frac{\overline{1}}{r^2}\right)$ = the time-averaged value of $1/r^2$ for all mission phases

t = mission duration

$$A = \left\{ \frac{-\ell n\, P}{5.5 - \frac{33}{2\pi}\left[\frac{\sin\left[\frac{2\pi}{11}(Y_2 - 1965)\right] - \sin\left[\frac{2\pi}{11}(Y_1 - 1965)\right]}{Y_2 - Y_1}\right]} \right\}$$

P = the probability of not exceeding the mission dose limit

Y_1 and Y_2 = the years at which the mission begins and ends, respectively

The resultant mission module radiation-shielding requirements are shown in Figure 19 as a function of the year the mission is initiated for missions to Mercury, Venus, Mars, and Jupiter. Since the inherent spacecraft 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. The required additional shielding could conceivably be achieved by judiciously locating equipment and supplies (food and water) housed within the mission module or by providing a storm cellar, which could be occupied for short periods during solar flares. At present it appears that radiation protection can be provided by judicious design of the mission module and by providing a minimum of additional protection.

Jupiter-Trapped Radiation

Various studies of the characteristics of Jupiter's trapped electron belt have been carried out. These studies, which usually assume that the non-thermal decimeter-wavelength radiation is synchrotron emission from electrons trapped in a magnetic dipole field, have not previously been carried to the point necessary for numerical evaluation of electron dose rates in the vicinity of Jupiter. Such dose rates are necessary for the analysis of missions to that planet and were carried out during this study. Three Jupiter electron flux models were developed, and the resultant additional shielding requirements are shown in Figure 20. The shielding requirements shown are for the 1985-to-1989 Jupiter-and-Ganymede mission which occurs during a period of minimum solar activity. Similar analyses were conducted for missions which occur during periods of average and maximum solar activity.

Order-of-magnitude uncertainties are associated with the trapped radiation about Jupiter. The decimetric and decametric radiation make possible approximate calculations of the flux and spatial extent of trapped electrons; the corresponding quantities for any trapped protons are matters for conjecture. Until the source mechanisms for the Earth's Van Allen belts' protons are better understood, it is not possible to estimate parameters associated with protons in the Jovian trapped radiation.

It is felt that the calculations performed during this study bracket the actual situation on Jupiter. The values calculated for an equatorial field (B_o) of 2 gauss represent a lower limit, which will most probably be exceeded. On the other hand, the values calculated for B_o = 15 gauss are probably too high. Therefore, for planning purposes, the values associated with B_o = 5 gauss are recommended.

The fluxes and dose rates associated with a 5-gauss field are such that stopovers at Ganymede appear possible, but are clearly not a desirable part (from a radiation-shielding standpoint) of manned missions to Jupiter. For missions undertaken during the active portion of the solar cycle, a small

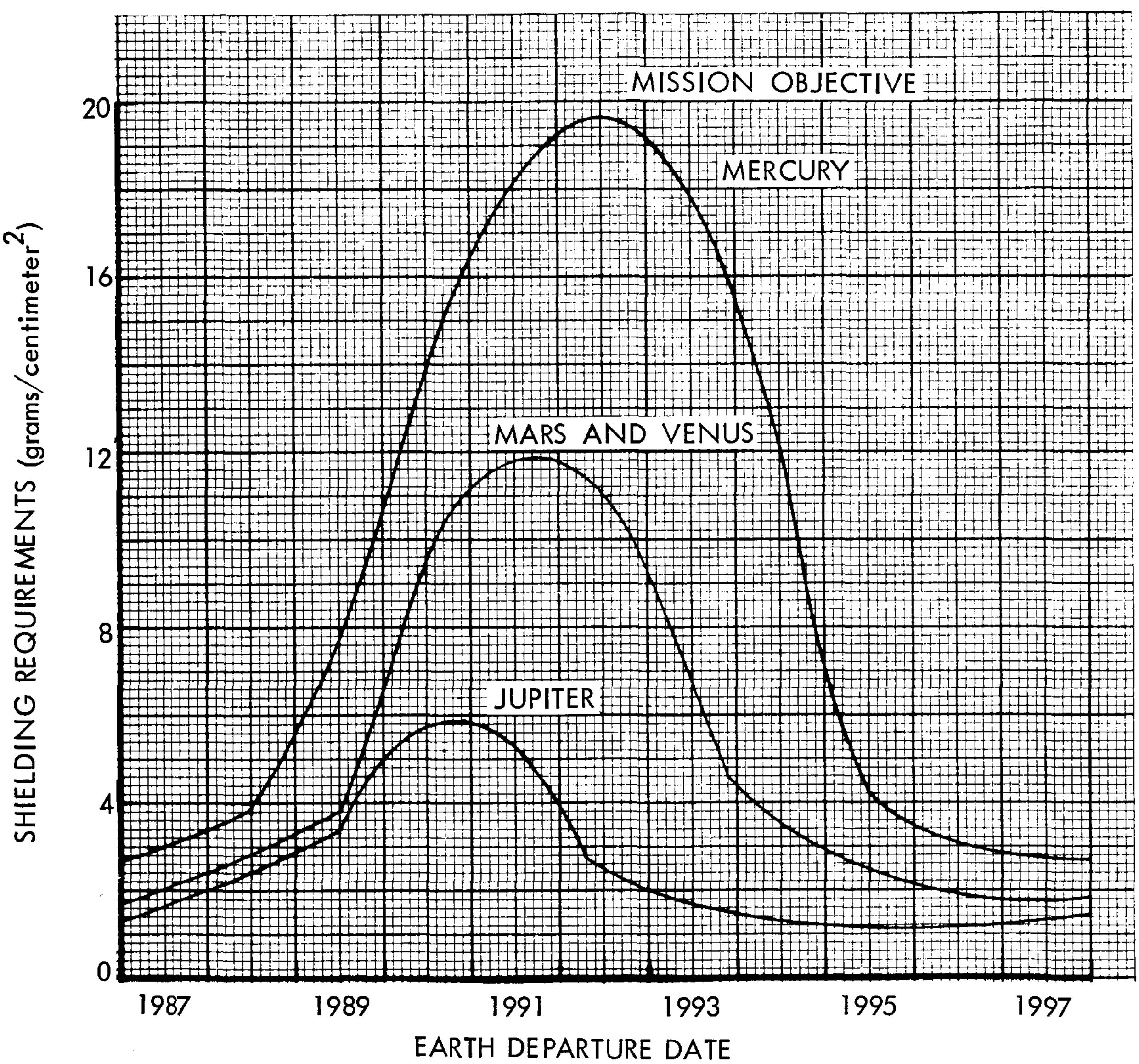

Figure 19. Mission Module Radiation Shielding Requirements

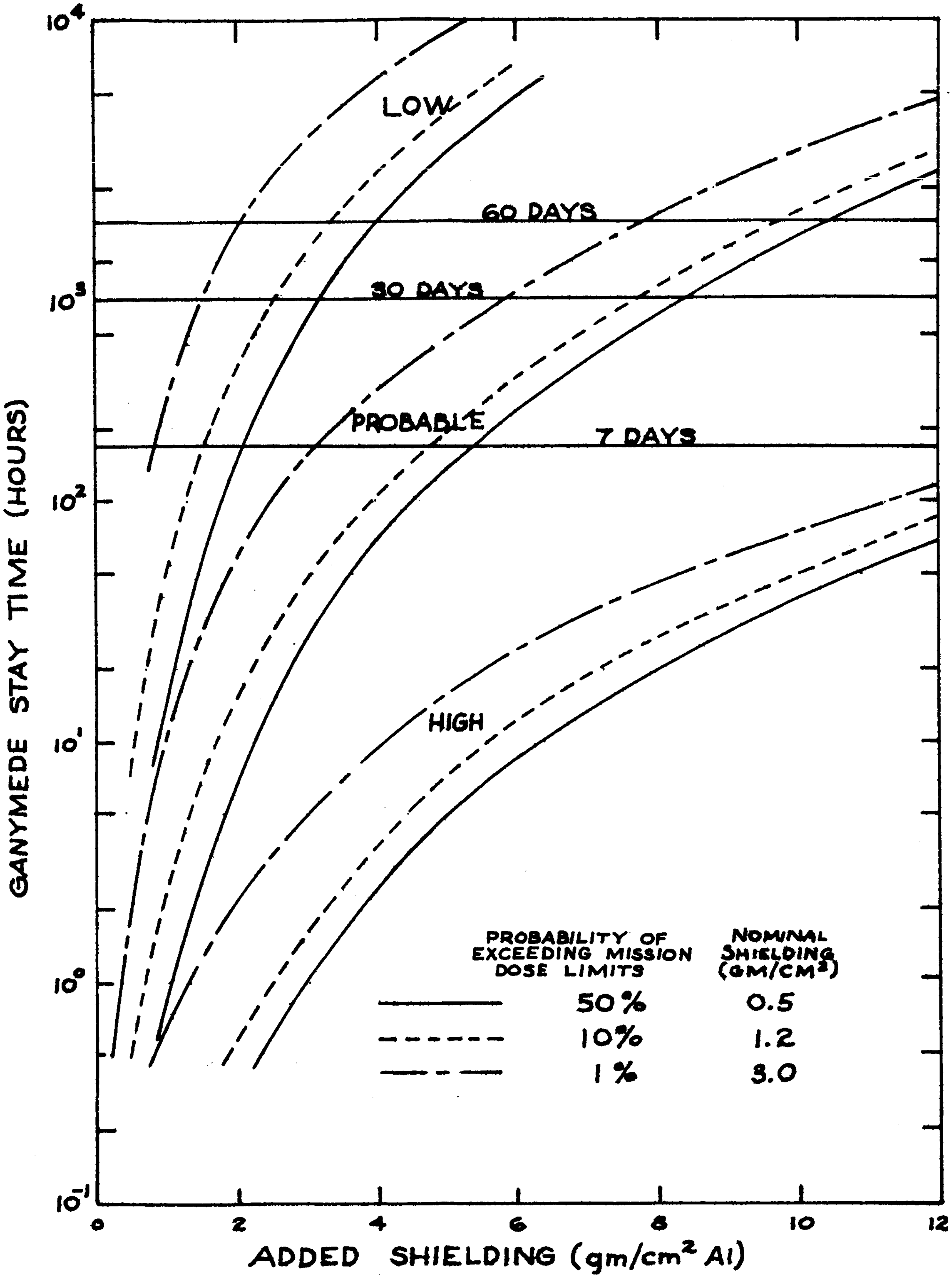

Figure 20. Added Shielding (1985 to 1989 Mission)

amount of extra shielding ($\lesssim 3$ grams/centimeter^2) will most probably suffice for 60 days at Ganymede, while for missions undertaken when the Sun is quiet $\gtrsim 6$ grams/centimeter^2 extra shielding will be required. Total shield thicknesses of $\gtrsim 10$ grams/centimeter^2 appear necessary in any event if a 60-day stay at Ganymede is contemplated. Reducing the stay time to 30 days only decreases this approximately 2 grams/centimeter^2 at the most. As an alternative, Callisto could be considered as the target body, since the shield thickness required will be approximately a factor of two less.

SUBSYSTEM SYNTHESIS AND PARAMETRIC ANALYSIS

Requirements of the major spacecraft subsystems were evaluated by first establishing the types of subsystems most appropriate for the mission objectives being considered in this study. The subsystems considered here are the environmental control and life support subsystem, communications subsystem, propulsion subsystem, and the electrical power subsystem. Each will have a significant influence on the total system design. Other subsystems included in the spacecraft weight synthesis analyses are the guidance and navigation, reaction control, and the scientific instrumentation and control subsystems. The parametric analyses conducted for the environmental control and life support subsystem, communications subsystem, propulsion subsystem, and electrical power subsystem are summarized in the following paragraphs. The detailed discussions of these subsystems are contained in Appendix C.

LIFE SUPPORT SUBSYSTEM PARAMETRIC ANALYSIS

An environmental control and life support subsystem (EC/LSS) is required in the Earth reentry module, mission module, and planetary excursion module. Since the EC/LSS is a major contributor to the mass of the manned modules, it is appropriate to examine the weight, volume, and power requirements of the subsystem to determine the degree of closure required for the family of missions being considered.

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 are represented by scaling equations, and separate equations were established for each principal element of the subsystem. The basic equations for the Earth reentry module, and the planetary excursion module ascent stages; the open system for the mission modules (MM) and planetary excursion module descent stages (PEM/DS); and the water and oxygen recovery system for the MM and PEM/DS were provided by MAD. These scaling equations were either modified by mutual agreement or corroborated by parametric data used by the Space Division for other studies. Scaling equations for the long-duration (over 90 days) open system and a water-recovery-only system were developed during this study.

The principal elements considered in defining the total subsystem characteristics are shown in Table 20. The table also defines the components assumed to be included in each principal element. The resultant weight, volume, and power scaling equations for each of the systems considered are summarized in Table 21. The scaling equations at the element level, the ground rules and assumptions concerning the operational duration, man's daily balance, emergency supply requirements, leakage, feces storage or disposal, atmospheric supply storage, and electrical power requirements are discussed in detail in Appendix C.

A study was also made of food-producing systems to determine their utility for the family of missions. The primary evaluation criterion was subsystem weight but potential reliability was also considered qualitatively. It was determined that the food producing systems did not warrant further consideration in the weight synthesis analysis.

Because of the short occupancy times (no more than 24 hours), the open system was assumed for use in the ERM and the PEM ascent stage during subsequent analyses. The open system was also used in the PEM descent stages. Although a mass advantage would accrue if a partially closed system were used for the longer occupancy times, the magnitude of the savings does not seem to warrant the additional system complexity.

The mass requirements of the three subsystems considered in detail for use in the mission module are shown in Figure 21 as a function of mission duration for crew sizes of 8 and 20 men. As can be seen from the figure, the mass requirements of the open system are excessive for the mission durations required -- 300 to 1500 days. Therefore, this system was not considered further. The mass requirements of the system with oxygen recovery only are approximately 50 percent heavier than the system with both water and oxygen recovery for a mission duration of 300 days. As the mission duration increases to 1500 days, the system with water recovery only is approximately 80 percent heavier than the more fully closed system. This mass penalty was considered to be excessive for these missions. In order to utilize a system which is compatible with the requirements of all of the missions considered in this study, the water and oxygen recovery system was employed during the module and system synthesis analyses. Such a system will not necessitate major technological advancements and could be readily available for all missions during the time period being considered.

COMMUNICATIONS SUBSYSTEM PARAMETRIC ANALYSIS

The economics of planetary exploration missions dictate that a maximum amount of data be obtained and transmitted back to Earth. In particular, some form of color-television or color-picture transmission

Table 20. Environmental Control and Life Support Subsystem Components

Principal Element	Components
1. Crew and crew support	95-percentile man (> 197 pounds) Shoes Undergarments Coveralls Bedding Personal property Personal hygiene kit Space suit Space helmet Space boots and gloves Space back pack Space suit 14-day O_2 supply Fire fighting equipment Medical equipment and supplies Puncture sealant
2. Furniture and housekeeping	Two airlocks Sleeping compartment Furniture Clothes laundry Janitorial equipment Cleaning and janitor supplies
3. Food management	Kitchenette Culinary equipment Water heater and stove Initial water supply Food Meal containers Refrigeration Repackaging supplies
4. Water supply	Drinking water Cooking water Wash water Containers
5. Waste management	Toilet room Feces collector - commode, dehydrator, and supplies Urine collector - adapter, pump, holding tank, and water in system Wash water collector - filter unit, pump, filter supplies, holding tanks, and water in system Personal hygiene - filter unit, suction pump, and supplies
6. Temperature and humidity control	Main condensing coil Spare condensing coil Heating coils Spare heating coils Fan Controls Ducting Coolant in system Coolant pump Electronic heat conduction plates Condensed water separator Condensed water pump Condensed water tank Plumbing

Principal Element	Components
7. Atmospheric purification	Charcoal filters Fiberglass filters Diverter valves Heater Cooling coils Ducting Trap Ultra-violet lamp Silica-Gel Zeolite Blowers Chromatograph Catalytic burner
8. Atmospheric supply	Oxygen Oxygen tankage Emergency oxygen supply Emergency oxygen tankage Pressure control Valves and piping
9. Instruments and controls	Panel board Instruments Controls Digitizing equipment

Note: If the oxygen is recovered by the Bosch process, then the above subsystem functions of numbers 7 and 8 are combined into the following subsystem:

Atmospheric purification and supply	Charcoal filters Fiberglass filters Ducting Diverter valves Heater Cooling coils Trap Ultra-violet lamp Chromatograph Silica-gel Zeolite Blower Valving Bosch process unit Electrolysis unit Oxygen pumps Hydrogen pumps Tankage Catalytic burner

Table 21. Environmental Control and Life Support Subsystem Scaling Equations

Degree of Closure	Scaling Equations
Open $\leq$ 90 days	$M = 408 + 330\ N_c + 0.09\ \Delta t + 6.204\ N_c\ \Delta t$ $V = 7.6 + 3.93\ N_c + 0.001\ \Delta t + 0.0197\ N_c\ \Delta t$ $P = 835 + 105\ N_c$
Open > 90 days	$M = 408 + 330\ N_c + 0.09\ \Delta t + 11.317\ N_c\ \Delta t$ $V = 7.6 + 3.93\ N_c + 0.001\ \Delta t + 0.02597\ N_c\ \Delta t$ $P = 835 + 105\ N_c$
Water Recovery Only	$M = 468 + 367\ N_c + 0.09\ \Delta t + 1.981\ N_c\ \Delta t$ $V = 7.7 + 3.95\ N_c + 0.001\ \Delta t + 0.01466\ N_c\ \Delta t$ $P = 985 + 205\ N_c$
Water and Oxygen Recovery	$M = 471 + 323\ N_c + 0.09\ \Delta t + 0.997\ N_c\ \Delta t$ $V = 7.7 + 3.39\ N_c + 0.0007\ \Delta t + 0.0066\ N_c\ \Delta t$ $P = 860 + 400\ N_c$

M = Mass, excluding leakage and repressurization (kilograms)
V = Volume (cubic meters)
P = Electrical power (watts
N_c = Crew size
Δt = Time (days)

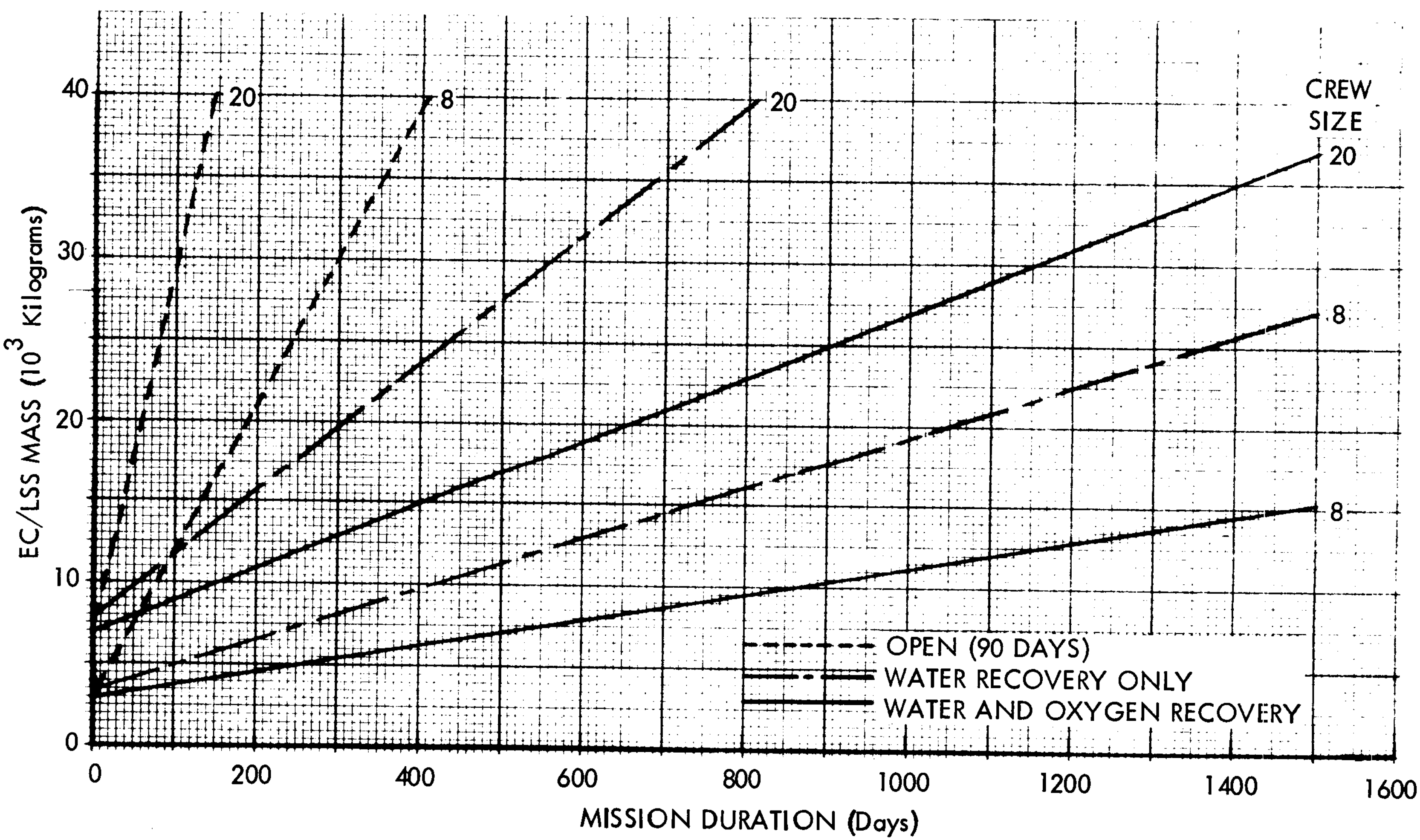

Figure 21. Mission Module EC/LSS Mass Comparison

would be desirable. Due to the extremely high number of data bits in a high resolution picture or TV frame, a high bit rate is required to transmit in a reasonable time, even with low frame rates and compaction. This problem is aggravated by the extremely long communicating distances. Thus the spacecraft-Earth communications subsystem is a critical element of interplanetary spacecraft design. It will represent compromises and/or penalties in the areas of power requirements, antenna sizes, pointing and tracking requirements, transmission duty cycle, data rate, etc. Much work must be done to develop communication technology and spacecraft hardware, and perhaps even the replacement of the existing ground communications network, to be compatible with the new spacecraft equipment.

Four subsystems, which span the frequency range of 2.3 gigahertz through 357,000 gigahertz were compared, namely S-band, millimeter, carbon dioxide laser and gallium arsenide laser. Although only four subsystems were investigated in depth, these represent the inherent advantages and problems of many such subsystems and are considered to be those most likely to be considered for future applications. A fifth subsystem, helium neon laser, was considered but not used in the comparison because it has very low efficiency and is limited to an output of about 0.1 watt.

Many parameters effect the capability of any specific communication subsystem. To simplify this study and still obtain meaningful results, certain assumptions were made regarding the performance parameters of the ground receiving station, modulation efficiency, performance margin, type of modulation, efficiencies, etc., expected in the 1980-to-2000 time period. The values of these parameters used in the comparisons of the selected systems are given in Table 22; the rationale for their selection is discussed in Appendix C.

The critical parameter in the comparison of the candidate communication systems was considered to be the power requirements. The differences in the performance, integration, and the weight of the transmitter, receiver, and antenna (aperture) will be small compared to the differences in the weight of the electrical power subsystem because of differences in the input power requirements. The candidate systems are compared in Figure 22 which shows the transmitter output power as a function of bit-rate-range squared product (BR^2) and antenna (aperture) diameter. Since the system efficiencies are essentially equal (either 40 or 50 percent), comparing the transmitted power is analogous to comparing the input power. The parameter BR^2 was used in the comparison since bit rate can be traded off equally with the square of communication range.

The comparison of the candidate systems shows that only two systems have lower power requirements than the two S-band systems. The galium arsenide non-coherent laser and the carbon dioxide laser with one-meter

Table 22. Fixed Parameters for Comparison Optical and Radio Space Communication

Parameter	S-Band	Millimeter	CO_2 (heterodyne)	GaAs (non-coherent)
Frequency	2.3 GHz	94 GHz	28,300 GHz	357,000 GHz
Wavelength	13.05 cm	3.19 mm	10.6 microns	0.8400 microns
Spacecraft antenna diameter (m)	4.88 15.2	0.304 0.67 4.58	0.014 0.178 1	0.10 1 m
Spacecraft antenna gain (db) ([A])	38.8 48.7	47 53.8 70	73.22 94.8 109.6	111.5 131.5
Beam angle (arc-sec) ([B])	6750 2150	2640 1200 165	180 15 2.67	2.12 0.21
Ground antenna diameter, D_R (m)	64 (210 ft)	4.58 (15 ft)	2	10
Ground antenna gain (db)	61	70	NA ([F])	NA ([F])
Ground antenna area (effective) A_R (-dbm^2)	32.5	9.56	4.96	18.9
Modulation efficiency, ξ - (db) ([C])	10	10	10	10
Performance margin, M (db) ([D])	10	10	20	20
Receiver system noise temperature (°K)	35	400	NA	NA
Noise spectral density				
ψ = KT (radio spectrum)	4.83×10^{-22}	5.52×10^{-21}	NA	NA
ψ = KT(db)	-213	-203	NA	NA
ψ = hf (optical spectrum)	NA	NA	1.88×10^{-20}	
ψ = hf (db)	NA	NA	-197.3 dbw-cps^{-1}	
Detector responsivity, ρ	NA	NA		0.002 amp-watt^{-1}
Quantum efficiency, η	NA	NA	0.20	
Modulation	PCM/PSK/PM	PCM/PSK/PM	PCM/PL	PCM/PL
Range equation ([E])	$P_T G_T = \frac{BR^2(\xi)M\,4\pi\psi}{A_R}$	$P_T G_T = \frac{BR^2(\xi)M\,4\pi\psi}{A_R}$	$P_T G_T = \frac{R^2 B\,hf(16)\,M\,(\xi)}{D_R^2\,\eta}$	$P_T G_T = \frac{BR^2\,32M\,(\xi)\,e}{D_R^2\,\eta}$
Efficiency (percent)	50	40	40	50

Footnotes to Table of Fixed Parameters

([A]) Gain of optics assumes uniform density within limits of beam

([B]) Diffraction limit assumed for optical beams

$\theta = 1.22 \frac{\lambda}{D}$

Beam angle for radio frequency based on 3-decibel points

$\theta = \frac{252{,}000}{D}$ arc-seconds

([C]) Modulation efficiency for digital systems:

$(\xi) = \frac{S\,T}{N/B}$ = 10 decibels for all systems

where:

ξ = modulation efficiency in cycles per sec per bit per second
S = signal power in watts
T = time in seconds per bit
N = Noise power in watts
B = bit rate in bits per second

Example of bit error rate (BER):
For 10 cps/bit sec.
S-band BER 5×10^{-6}
Optical BER 4×10^{-3}

([D]) Performance margin

For S-band and 3 mm, includes transmission line and atmospheric losses

For Optics, includes following transmissivities:

	GaAs		CO_2	
Transmitter optics	T_T = 0.50	-3 db	T_t = 0.50	-3 db
Atmospheric	T_a = 0.80	-1 db	T_a = 0.36	-4.5 db
Filter	T_f = 0.20	-7 db	T_f = 0.90	-0.5 db
Diffraction (farfield)	T_d = 0.50	-3 db	T_d = 0.50	-3 db
Receiving optics	T_r = 0.50	-3 db	T_r = 0.50	-3 db
Modulation	T_m = 0.50	NA	T_m = 0.50	-3 db
Subtotal		-17 db		-17 db
Tolerance		-3 db		-3 db
Total		-20 db		-20 db

([E]) The terms in the range equations are as follows:

B = bits per second
R = range (meters)
ξ = modulation efficiency
M = performance margin
ψ = noise spectral density
A_R = antenna area (effective)
D_R = diameter of optical aperture
e = electronic charge
η = quantum efficiency
ρ = detector responsivity

([F]) Not applicable because ground aperture used as collector of photons only.

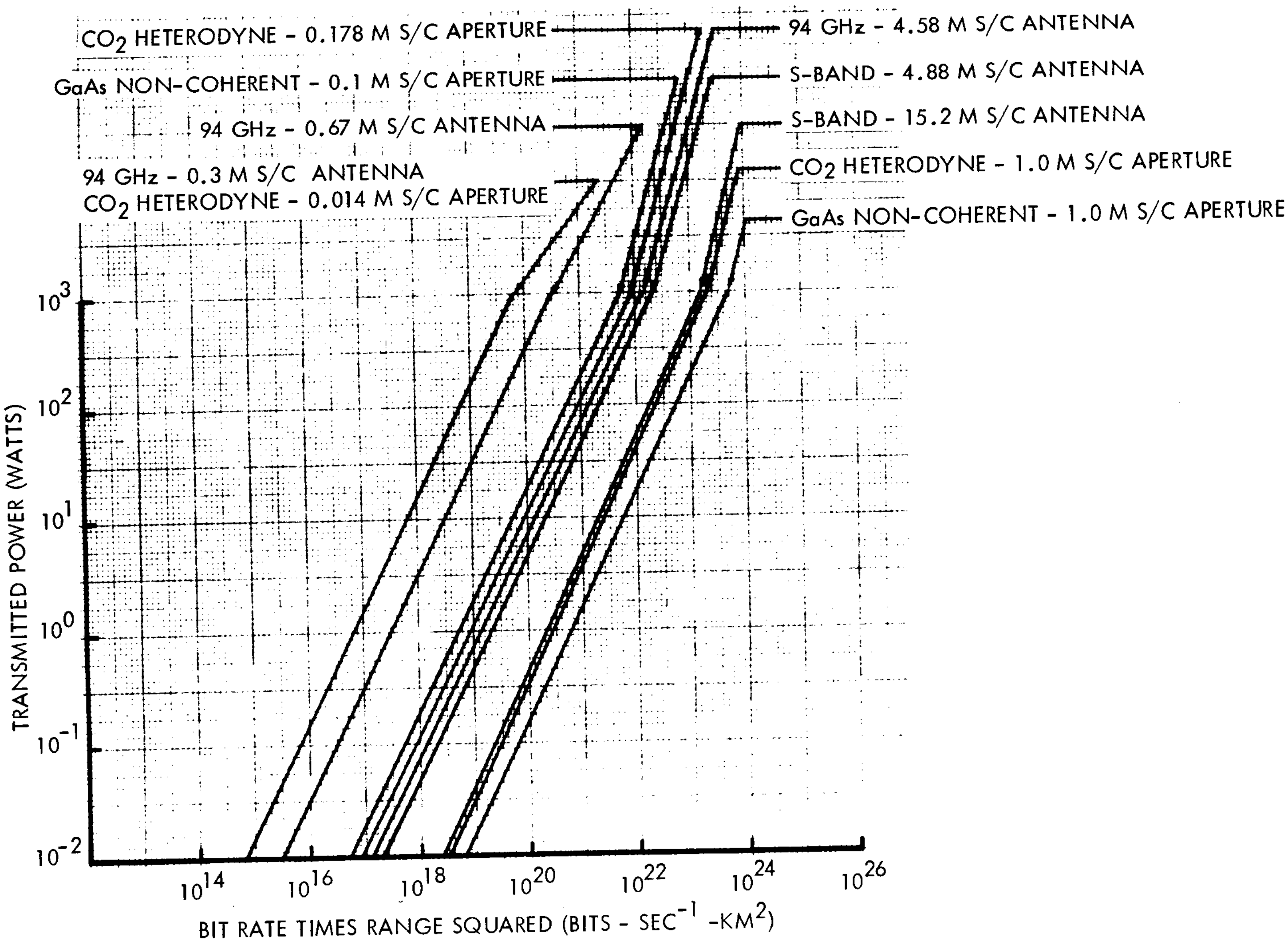

Figure 22. Interplanetary Communications Subsystem Comparsion

apertures have the lowest power requirements, but both of these systems have extremely narrow beam widths (0.21 arc-seconds and 2.67 arc-seconds) which is believed to be a serious pointing 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.

The S-band systems appear to be very attractive if a 15.2-meter (50-foot) antenna can be provided. Decreasing the antenna diameter to 4.88 meters (16 feet) increases the power requirements by an order of magnitude, but the system will still require less power than any of the millimeter systems or the laser systems with the smaller-aperature diameters.

Figure 22 also illustrates the sensitivity of the power requirements to the communciations range and data rate requirements. Although the power requirements vary with the square of the range, the difference between the inner planets and the outer planets is only 15 decibels. Therefore, the range problem can be solved, in part, by increasing the antenna (aperture) gain. Data management and data compaction also will reduce the power requirements since the requirements are directly proportional to the data rate. Data compaction appears to be particularly attractive and compaction ratios of 30 have been postulated. (Compaction ratios of 4:1 to 6:1 are within the current state of the art.)

It appears that S-band will hold a significant position in post-1980 communications. The assumed 15.2 meter (48.7 decibel gain) parabolic anatenna is considered to be about the upper diameter limit for an unfurlable antenna which can be deployed and retracted, and thus better antenna efficiency is desirable. The only significant drawback of S-band is the limited bandwidth. If compaction ratios of 10:1 or more are not achieved, the higher frequency systems may be selected over S-band.

Millimeter waves have power requirements which are an order of magnitude higher than required for S-band, but the antennas and other equipment required are much smaller. The millimeter systems would be more competitive if antenna arrays could be developed that would greatly exceed the 70-decibel figure which was assumed for both the spacecraft and ground terminals. Also, system noise temperature may be decreased, although only a 3-decibel gain can be achieved in this area. The millimeter system is an attractive successor to S-band because the equipment can be co-located at the S-band stations and much of the existing electronics and physical facilities can be shared to reduce costs.

Due to high antenna gains possible with lasers, wide band, high data-rate communications can be achieved with significantly smaller power requirements. The transition from today's components and devices to space-qualified hardware, however, will require significant breakthroughs in many areas and a heavy expenditure in research and development dollars.

Ultimately, any system becomes limited by power and data rates. Optical systems are inherently capable of transmitting wide bandwidths due to the high frequency of the light source. They can also transmit high data rates for less power, provided tracking/pointing problems are solved. Therefore, optical systems must ultimately be developed if high resolution, live motion, real time color television becomes a requirement. The state of the art is such that only the feasibility of using optics for such purposes can be visualized. There is much research and development to be done in basic components, system techniques, and supporting hardware before a highly reliable, workable system can replace the present microwave spacecraft and ground terminals.

A parallel research and development approach appears desirable for the continued development of communication subsystems. S-band should be developed to its full capability, since it probably will fulfill many interplanetary requirements for the next 20 to 30 years. On the other hand, smaller, lighter, and higher data-rate systems will be required eventually and research must be continually applied. A gradual transition from S-band to either millimeter or optical systems should be developed to take advantage of their favorable system characteristics.

PROPULSION SUBSYSTEM PARAMETRIC ANALYSIS

The propulsion subsystem analysis was concerned with the establishment of weight scaling equations and the selection of candidate propellants for use during the weight synthesis analyses. The basic scaling equations defining the mass of chemical and solid core nuclear engines were provided by MAD.

Engine Mass

The scaling equation defining the mass of chemical engines was given as

$$W_c = \left(\frac{T}{\tau} + Z\right) n$$

where

W_c = weight of chemical engine cluster (kilograms)

T = thrust of each engine (kilograms)

τ = engine thrust-to-weight ratio

Z = constant (nominal value = 45)

n = number of engines in cluster

The engine thrust-to-weight ratio was examined as a function of engine thrust for various types of engine designs. These data were correlated with the above scaling equation, and appropriate coefficients were derived which define the engine thrust-to-weight ratio and mass as a function of the engine thrust level. The resultant scaling equation is given by

$$W_c = K\left(\frac{T}{\tau} + Z\right) n$$

where the scaling coefficient K is a function of the engine thrust and engine type. The engine thrust-to-weight ratio and the scaling coefficient are presented in Appendix C for representative pump-fed, pressure-fed, high-chamber-pressure, and torodial-aerospike engines.

The nuclear engine mass equation is given by

$$W_n = (\alpha T + \beta)\ n$$

where

W_n = weight of nuclear engine cluster including radiation shield (kilograms)

T = thrust of each engine (kilograms)

n = number of engines in cluster

α = constant (nominal value = 0.129)

β = constant (nominal value = 3310)

This equation was compared with data derived from several sources and it was determined that the proposed equation agrees reasonably well with the most recent estimates of nuclear engine weights.

Candidate Propellant Combinations

Potential chemical propellants were examined and the characteristics of all propellants considered in the study are presented in Table 23. The table lists the appropriate performance levels, physical characteristics, and thermal properties. Also shown in the table is a criterion which has been developed to provide an approximate measure of the in-space storage capability of the various propellant combinations. It is, in effect, the ability of the propellant to absorb heat through bulk temperature increases and evaporative cooling (through venting) divided by the potential heat-absorption rate. The heat absorption rate is proportional to the bulk liquid temperature less the environmental temperature. Those combinations which exhibit the higher values have the greater degree of storability.

A criterion for determining the relative cooling capability of these propellant combinations in regenerative rocket engines is also presented in Table 23. This is of particular importance to large propulsive stages, where it may become impractical to design and develop ablative-cooled engines at the required thrust level because of the excessive weight penalties incurred. Those combinations which exhibit the higher values have the greater cooling capability.

To provide the basis for the propulsion subsystem design data, the following propellant combinations were selected as representative of the chemical systems applicable to the missions considered during the study:

LO_2/LH_2
OF_2/B_2H_6
OF_2/MMH
87.5%FLOX/MMH
82%FLOX/CH_4

These combinations were selected, in part, on the basis of performance and storage considerations. Liquid oxygen and liquid hydrogen was selected as the propellant combination representative of the high performance levels and storability characteristics which are consistent with large, orbital-launch vehicles. The remaining propellant combinations exhibit a reasonable degree of in-space storability when both boiling and freezing characteristics are considered.

Table 23. Comparison of Candidate Liquid Propellants

Propellants	Opt. Mixture Ratio	Bulk Density (lb/ft^3)	Vacuum Specific Impulse (sec)	Vacuum Density Impulse 1000 lb-sec ft^3	Chamber Temp (°F)	Fuel Temp (°F)			Oxidizer Temp (°F)			Chamber Regenerative Cooling Merit Rating	Storability Criterion Temperature at Environmental Temperature of		
						Freezing Point	Boiling Point		Freezing Point	Boiling Point					
							14.7 psia	90 psia		14.7 psia	90 psia		150°F	0°F	-150°F
F_2/N_2H_4	2.27	81.7	430	35.1	7140	35	236	355	-363	-307	-266	1.48	0.0202	0.0126	0.0072
N_2F_4/N_2H_4	3.24	89.2	390	34.8	6950	35	236	355	-264	- 99	-23	1.56	0.0218	0.0318	0.0184
OF_2/MMH	2.42	78.3	416	32.6	6620	-62	189	305	-371	-230	-177	1.11	0.0222	0.0237	0.0255
N_2F_4/MMH	3.38	85.5	380	32.5	6600	-62	189	305	-264	-99	-23	1.13	0.0228	0.0398	0.0264
87.5% FLOX/MMH	2.75	77.0	421	32.4	7080	-62	189	305	-363	-306	-264	0.85	0.0202	0.0210	0.0169
ClF_5/MHF-3[1]	2.60	90.4	358	32.4	6350	-65	194	310	-153	8	104	1.32	0.0302	0.0377	0.0170
N_2F_4/NH_3	4.50	78.3	381	29.8	6660	-62	189	305	-153	8	104	1.26	0.0305	0.0380	0.0172
82% FLOX/CH_4	5.75	66.1	424	28.0	7090	-296	-259	-197	-363	-305	-262	0.15	0.0049	0.0079	0.0207
OF_2/B_2H_6	3.75	62.4	444	27.7	6990	-266	-135	-61	-371	-230	-177	0.18	0.0079	0.0183	0.0410
O_2/CH_4	3.32	51.2	379	19.4	5370	-296	-259	-197	-362	-297	-258	0.28	0.0068	0.0110	0.0296
F_2/H_2	11.0	35.0	479	16.8	6660	-435	-423	-408	-363	-307	-266	6.29	0.0040	0.0057	0.0105
O_2/H_2	4.80	19.8	461	9.1	5070	-435	-423	-408	-362	-297	-258	15.4	0.0067	0.0096	0.0171

NOTE:

1. MHF-3 = 86 percent MMH + 14 percent N_2H_4
2. 100 psia to vacuum, 100 percent theoretical shifting equilibrium, ϵ = 60

During subsequent propulsion module and system analyses, LO_2/LH_2 and 87.5%FLOX/MMH were assumed as the nominal cryogenic and space-storable propellants, respectively. A bulk density of 317 kilograms per cubic meter (19.8 pounds per cubic foot), a mixture ratio of 4.80:1, and a specific impulse of 450 seconds were used during the sizing of LO_2/LH_2 propulsion modules. The corresponding values for 87.5%FLOX/MMH were 1233 kilograms per cubic meter (77.0 pounds per cubic foot), 2.75:1 mixture ratio, and 387 seconds, respectively.

ELECTRICAL POWER SUBSYSTEM PARAMETRIC ANALYSIS

The purpose of the electrical power subsystem analysis was to develop, for a spectrum of candidate systems, relationships between operational power levels, subsystem weight and dimensional requirements, and the heliocentric radius at which a system might be used. These relationships were based on the estimated technology in the 1980-to-2000 time period. Several subsystems appropriate to mission modules (with mission durations between one and four years) and planetary excursion module descent stages (with occupancy times not greater than 90 days) were considered. The Earth reentry module and planetary excursion module ascent stages were assumed to be occupied for no more than 24 hours. Therefore, only batteries were considered for use in these modules during subsequent analyses.

The spectrum of candidate electrical power subsystems for 1980-to-2000 application is quite broad when consideration is given to the many possible combinations of power sources and converters. Identification of the most suitable combinations in this study is based on demonstrated capability of developed systems, systems in the process of development, and on projected improvements. The electrical power subsystems which are expected to be available through the remainder of this century and the applicable power levels are shown in Figure 23. Also shown in the figure is the expected mission module power requirements. (It is reiterated that electric propulsion systems were not considered during this study).

In order to compare candidate subsystems on a realistic basis, promising combinations of energy sources and power-conversion systems were analyzed on an equal basis such that appropriate weight variations were included to compensate for inherent differences in the various combinations. Also, the most advantageous utilization of the candidate subsystems was identified.

In general, the approach taken was to obtain a system weight from reference reports describing systems applicable to the 1975-to-1985 time period. Many of these data were readily available for nuclear and solar

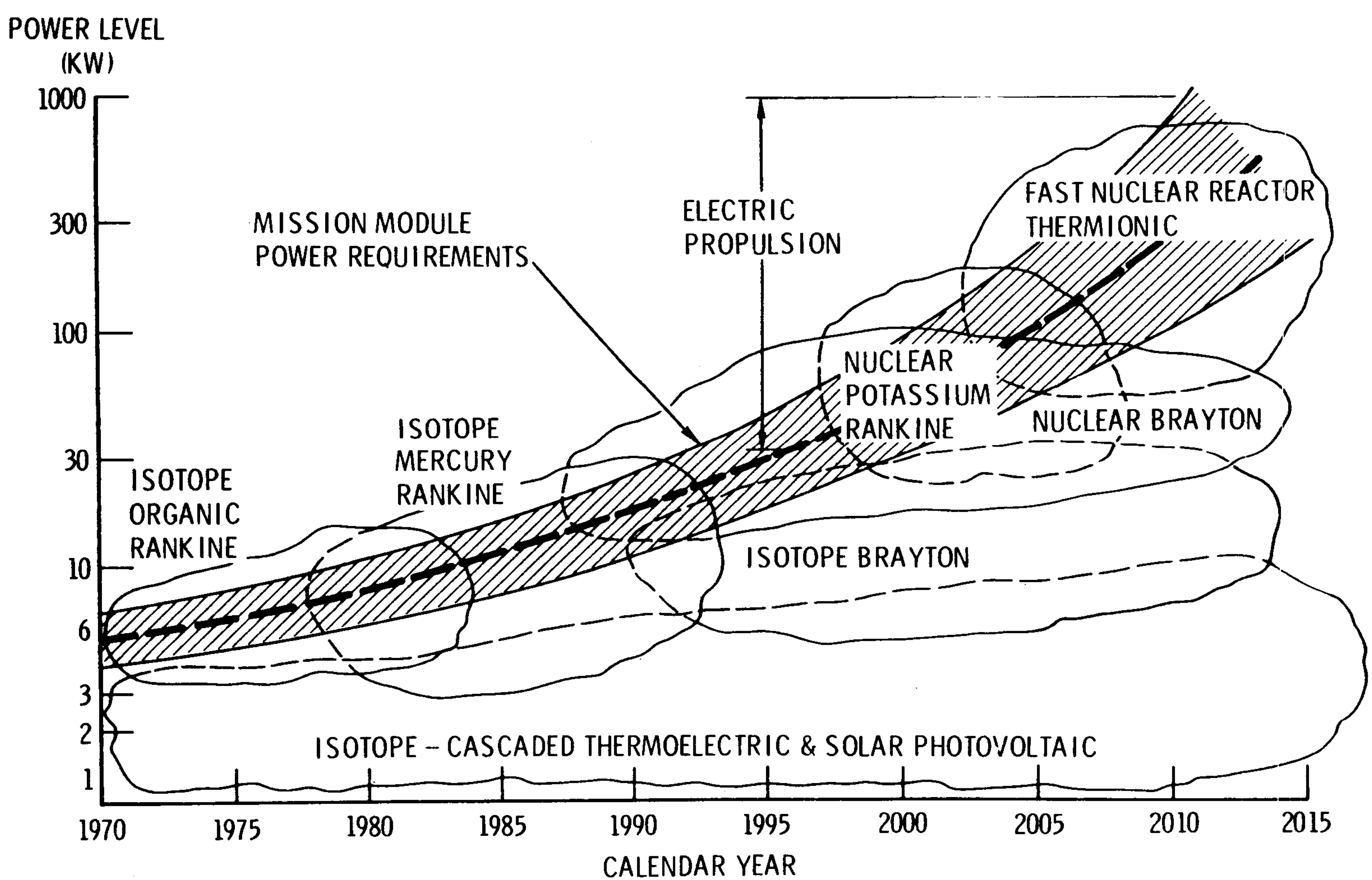

Figure 23. Candidate Electrical Power Subsystem

photovoltaic systems from NR Space Division and Atomics International studies. Solar dynamic systems data were prepared by assuming the same conversion design as for applicable isotope systems with only the heat source, i. e., solar concentrator-absorber, being different. Chemical systems data were available from the Apollo and Apollo Applications Programs. The accumulated data were examined and an adjustment made for expected system improvements by the 1980-to-2000 time period. Detailed weights were tabulated and a comparison was made between extrapolated systems and reference design weights.

Competitive power subsystems for the mission module and the planetary excursion module descent stage are shown in Tables 24 and 25, respectively. The mass requirements of the combinations that could be used with the mission module are presented in Figure 24 as a function of delivered power for mission durations of one year. Similar data were generated for mission durations of 2, 3, 4, and 5 years. Although the mass requirements are higher for longer-duration missions, the relative comparisons of the mass requirements are essentially the same. The mass data are based on the expected post-1980 values and include the subsystem redundancy required to meet a projected reliability goal of 0.999. Similar data are presented in Figure 25 for subsystems applicable to the planetary excursion module descent stage for a mission duration of 30 days. Power systems for mission durations of 2, 10, and 60 days were also evaluated.

Selection of candidate subsystems cannot be based on mass alone; other factors must be considered such as radiator requirements, integration and operational constraints, heliocentric radius sensitivity, shock sensitivity, etc. The radiator area requirements for the conversion systems considered in the study are shown in Figure 26. The variations in the Brayton cycle radiator requirements are due to different lower-cycle temperatures for a given upper-cycle temperature. The optimum lower-cycle temperature for a given upper-cycle temperature is largely a function of design criteria and vehicle constraints. If weight is the primary factor, one optimum lower temperature exists; minimum radiator area yields another optimum value (these two may be the same for missions requiring heavy meteoroid protection); maximum cycle efficiency (minimum isotope inventory) gives another value.

The electrical power subsystems which were used in the manned modules during subsequent module and system synthesis analyses (Appendix D) are shown in Table 26. Reactor systems were not selected for use in the mission module since they are heavier than the isotope systems, could require shutdown and retraction during propulsive (or aerobraking) maneuvers, and present potential operational constraints (e. g., during rendezvous). Solar systems were not assumed since they are not generally

Table 24. Competitive Auxiliary Power Subsystems for Mission Module

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

*Solar photovoltaic systems omitted since longer missions are consistent with heliocentric radius 2.5 to 3 AU

Table 25. Competitive Auxiliary Power Subsystems for Planetary Excursion Module

Nominal Power Level (kWe)	Operating Time (days)			
	2	10	30	60
20	Fuel cells Solar photovoltaic	Fuel cells Solar photovoltaic Isotope thermoelectric	Solar photovoltaic Isotope thermoelectric	Solar photovoltaic Isotope thermoelectric
10	Fuel cells Solar photovoltaic Chemical dynamic Primary batteries	Fuel cells Solar photovoltaic Isotope thermoelectric	Solar photovoltaic Isotope thermoelectric	Solar photovoltaic Isotope thermoelectric
5	Fuel cells Solar photovoltaic Chemical dynamic Primary batteries	Fuel cells Solar photovoltaic Isotope thermoelectric	Fuel cells Solar photovoltaic Isotope thermoelectric	Fuel cells Solar photovoltaic Isotope thermoelectric
2	Fuel cells Solar photovoltaic Chemical dynamic Primary batteries	Fuel cells Solar photovoltaic Isotope thermoelectric	Fuel cells Solar photovoltaic Isotope thermoelectric	Fuel cells Solar photovoltaic Isotope thermoelectric

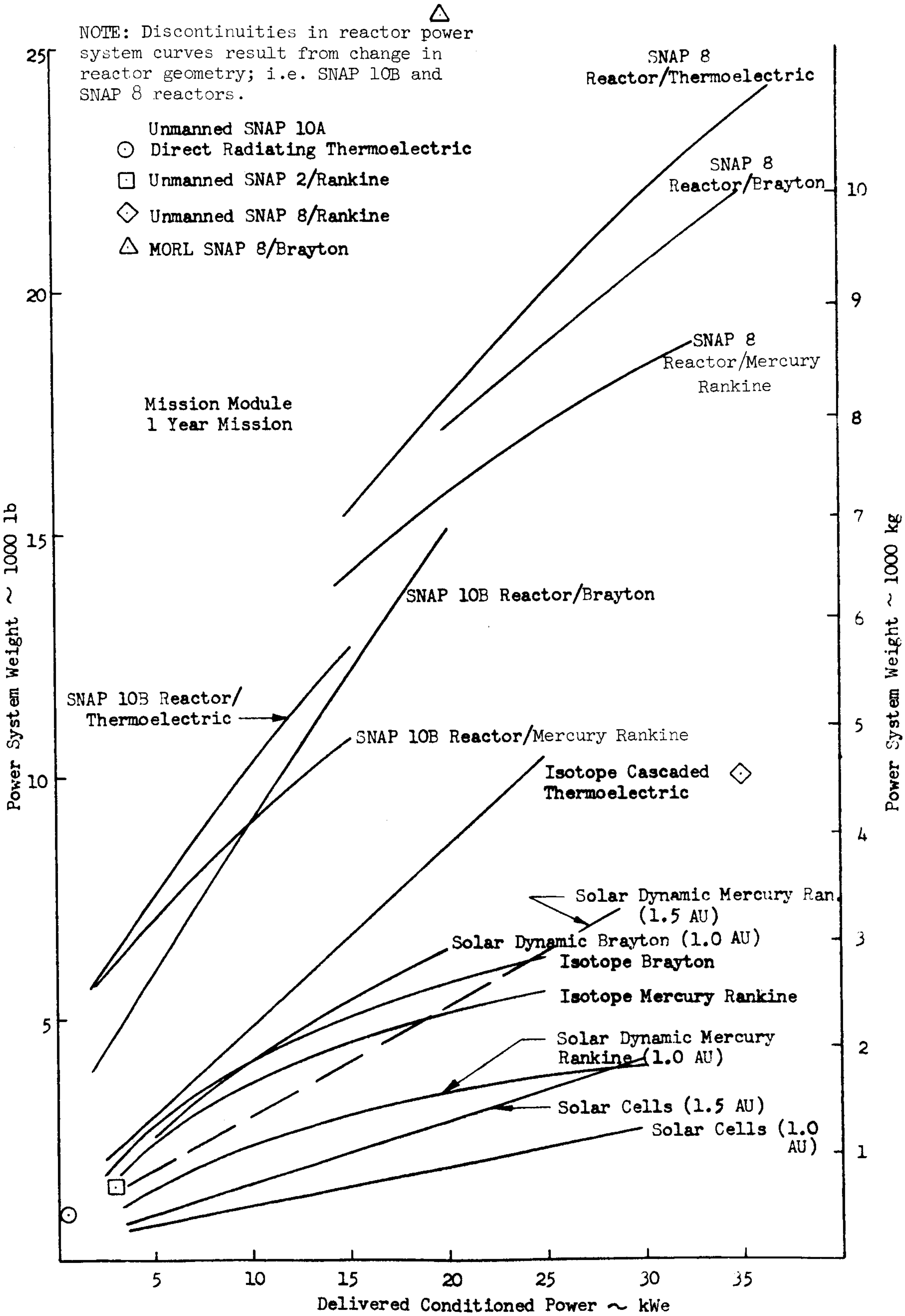

Figure 24. Power System Weight for Mission Module, 1-Year Mission

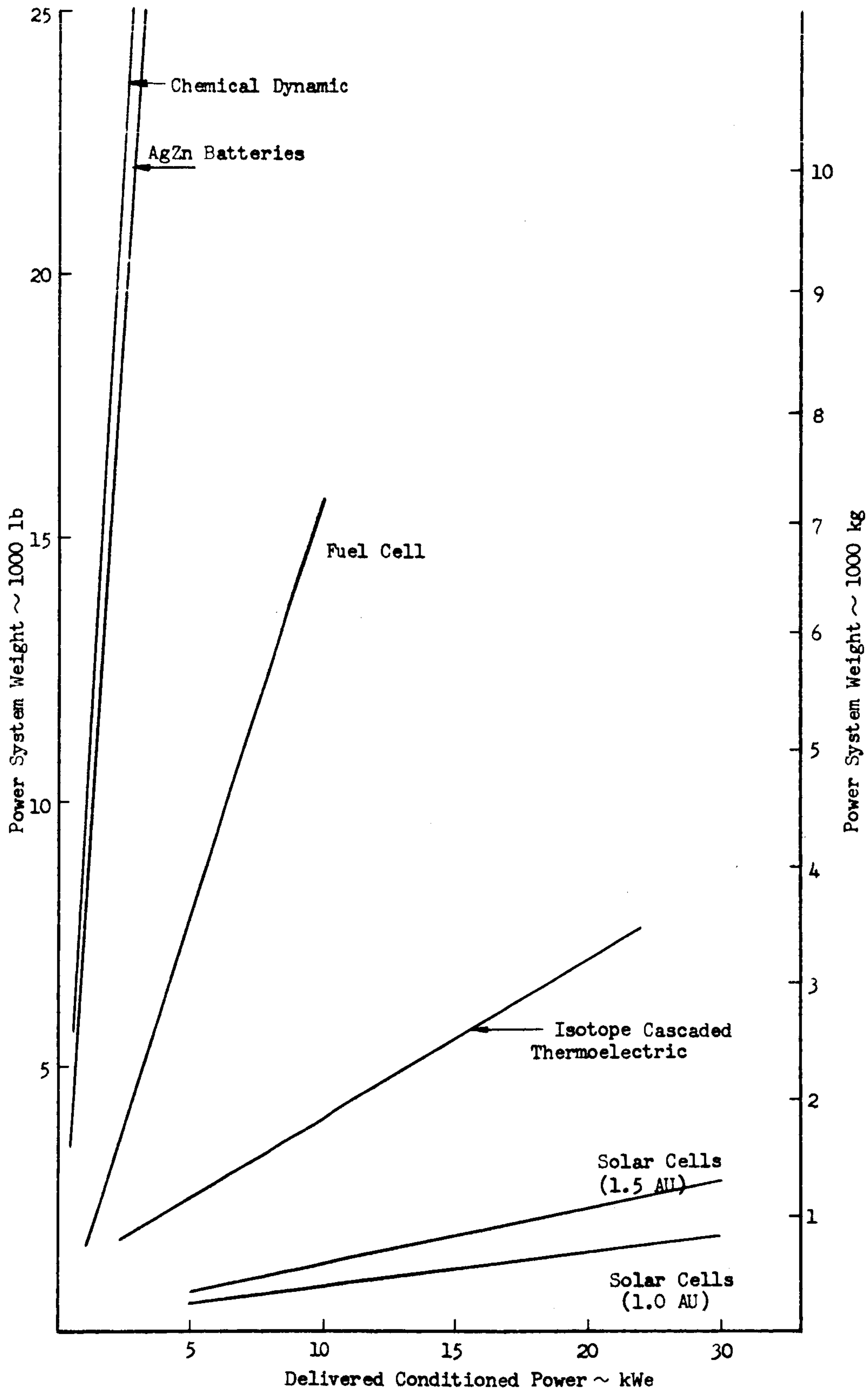

Figure 25. Power System Weight for Planetary Excursion Module, 30-Day Operation

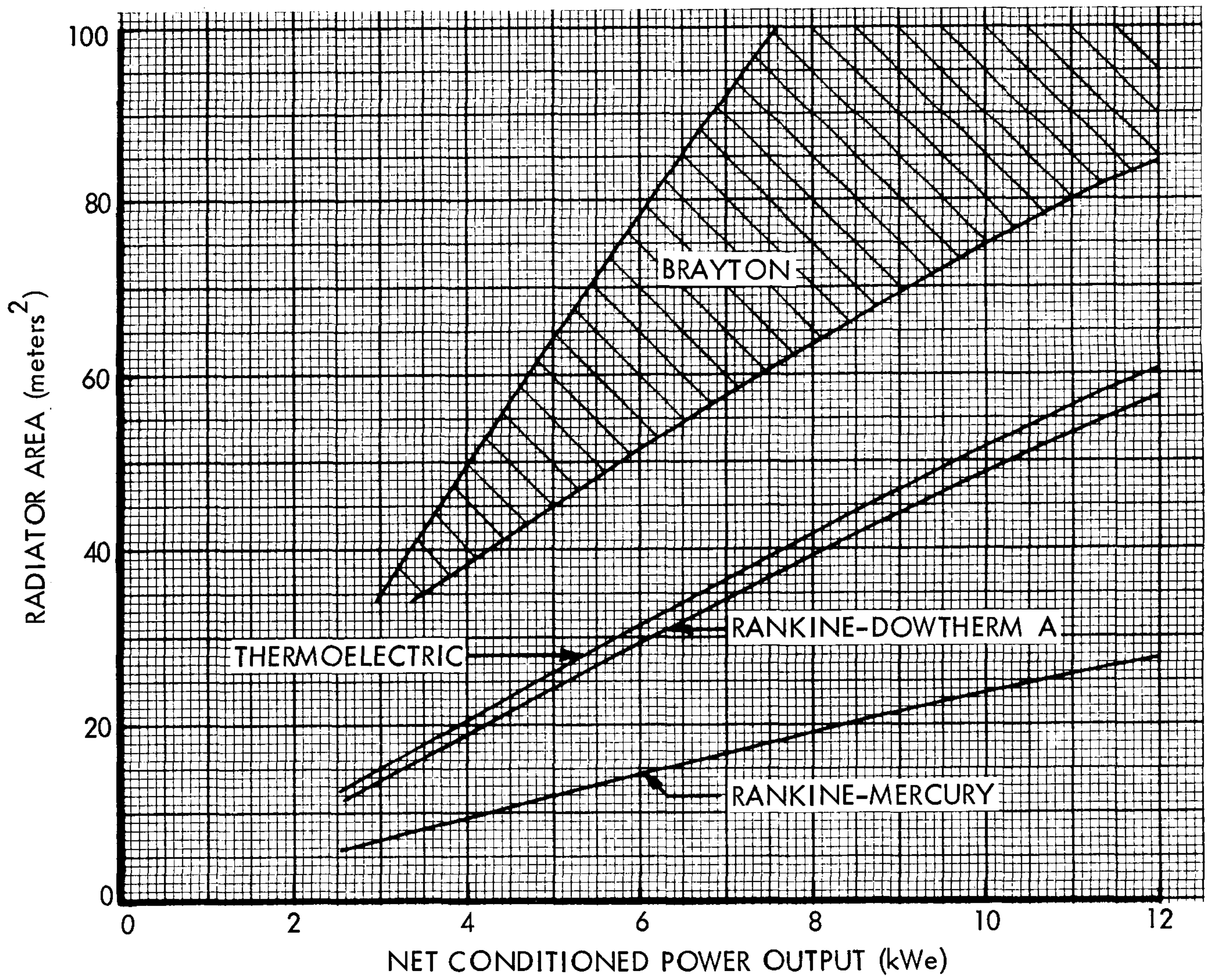

Figure 26. Radiator Area Requirements

applicable to all missions considered in this study. Although solar systems are appropriate for some of the missions, large arrays (on the order of 170 square meters) would be required.

The isotope cascaded thermoelectric system was selected for use in the planetary excursion module descent stage since it is the most appropriate system for the range of stay times considered (0 to 60 days). Chemical-dynamic systems, fuel cells, and batteries would result in an excessive weight penalty for the longer stay times. Solar cells, although the lightest system, would impose operational constraints (e. g. , landing site location), and are not generally applicable to all mission objectives.

Only batteries were considered for use in the Earth reentry module and the planetary excursion module ascent stage. The short occupancy times (up to 24-hours) precluded the necessity of considering more exotic systems.

Table 26. Selected Electrical Power Subsystems

Module	Subsystem Type
Mission module	Isotope/mercury rankine
Planetary excursion module, descent stage	Isotope cascaded thermoelectric
Planetary excursion module, ascent stage	Batteries
Earth reentry module	Batteries

SYSTEM SYNTHESIS AND PARAMETRIC ANALYSIS

Configuration studies were carried out for the modules which form the total systems required to accomplish the missions. These designs were generated in order to assure the development of realistic module weight-scaling equations for incorporation in the weight synthesis analysis which established the system-design requirements for the basepoint missions. The design studies, weight-scaling equations, and the results of the weight-synthesis analyses are summarized in the following paragraphs. The details of the analyses are contained in Appendix D.

CONFIGURATION DESIGN

Limited conceptual design studies were conducted in order to assure that the module weight scaling equations would be valid for the range of module design parameters which were considered in the study. The conceptual design studies included the Earth reentry module, mission module, planetary excursion module, and the aerobraker spacecraft. Since several configuration designs were already available from previous Space Division studies, the conceptual designs which were generated for the present study were limited to the extension of these studies to include the range of parameters applicable to this study. The designs which were available and the new designs are indicated in Table 27. As can be seen from the table, the majority of the conceptual design studies were devoted to the extension of past studies to include the larger crew sizes which were considered in the present study. The exceptions are the retrobraking planetary excursion modules for which no configurations were available.

WEIGHT-SCALING EQUATIONS

Modular weight-scaling equations were developed and incorporated into the Weight Synthesis computer program. The modules which were considered were the Earth reentry module (ERM), mission module (MM), planetary excursion module (PEM), propulsion modules and aerobrakers. In addition, scaling equations were developed for synthesizing the aerobraker spacecraft. The scaling equations were generated utilizing data provided by MAD, from the results of the conceptual design studies, and from the results of the subsystem synthesis studies.

Table 27. Conceptual Design Study Summary

Module	Type	Crew Size 2-6	8-10	14-16	20
Mission module	—	E	E	C	C
Earth reentry module	Apollo	E	E	E	E
	Biconic	E	E	C	C
Aerobraker	Cryogenic propellant	-	E	C	-
	Storable propellant	E	E	-	-
	Nuclear	-	E	-	-
Planetary excursion module	Apollo	E	C	-	-
	Lifting body	E	C	C	-
	Ceres/Vesta retro	C	C	-	-
	Ganymede/Mercury retro	C	C	-	-

E = Applicable existing designs
C = Conceptual designs developed

Weight-scaling equations were developed for the following three types of Earth reentry modules: biconic, segmented conic, and Apollo. The parameters which define the ERM characteristics are the Earth reentry speed and crew size.

The mission module sizing is based on volumetric requirements of the crew, subsystems, number of floors, and bulkhead aspect ratio.

Weight scaling equations were developed for both retrobraking and aerobraking planetary excursion modules. All planetary excursion modules are assumed to be two-stage vehicles. The ascent stage is composed of the crew and one-day life support and electrical power subsystems. The descent stage is composed of the subsystems required to land on the planet or asteroid surface and the subsystems necessary to support the crew during the surface stay.

The propulsion module scaling equations were provided by MAD and modified by SD to account for installation of the meteoroid and thermal protection systems. The effects of finite burning were accounted for in the sizing of propulsion modules by utilizing a MAD-supplied computer routine which emperically determines the velocity losses due to finite burning.

In order to determine the structural, meteoroid protection and heatshield weights of the aerobraker shroud, an iterative method was used for the volumetric scaling of the propulsion modules and the fixed volumes of the MM, PEM, and ERM enclosed within the shroud. In addition the effects of staging portions of the unused shroud were considered in the sizing of the planetary-departure propulsion modules. The detailed scaling equations and the assumptions which were required to develop all equations are discussed in Appendix D.

WEIGHT-SYNTHESIS METHODOLOGY

The total system mass requirements for a given mission objective, mission mode, and mission opportunity are computed using the Weight Synthesis computer program. Weight synthesis is accomplished by selecting the basic mission parameters of mission objective, mission purpose (i. e., orbiter or lander), mission mode, mission opportunity, and orbital stay time. The necessary input parameters are then determined for each of the basic modular routines as defined by the scaling equations. The weight-computing process is developed in reverse to that of the mission sequence, i. e., the Earth reentry module is sized first, and the Earth orbit escape propulsion module last.

WEIGHT-SYNTHESIS PARAMETRIC ANALYSIS

Generalized weight synthesis data were generated for each of the system modules in order to establish the sensitivity of the mass of the modules to the fundamental design parameters. The data can also be used to approximate the total system mass for specific missions although the total system mass values obtained in this manner would be somewhat in error, since environmental effects (which are mission dependent) are not included. These generalized data are presented in Appendix D for the Earth reentry modules, mission module, planetary excursion module, propulsion module, and the aerobraker spacecraft.

Earth Reentry Module

The effects of Earth-reentry speed on the mass requirements of the three configurations considered in the study are shown in Figure 27 for crew sizes of 8 and 20 men. The Apollo configuration is the lightest for reentry speeds below about 14.7 kilometers per second while the conic configuration is the lightest for reentry speeds above 17.5 kilometers per second. 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 per second for the majority of the missions considered, indicating that the Apollo configuration is desirable on the basis of mass considerations.

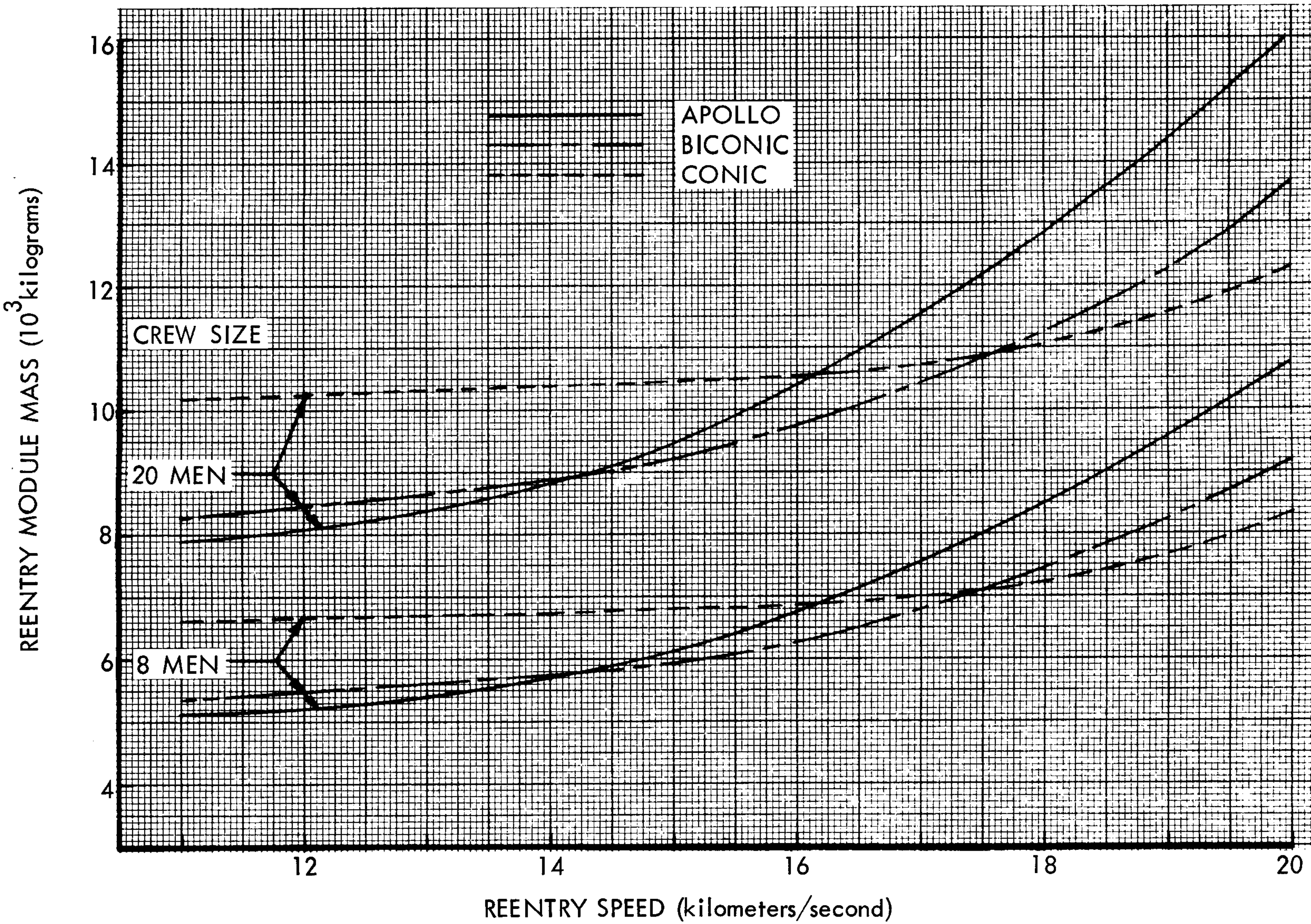

Figure 27. Reentry Module Mass Comparison

Mission Module

The crew size, mission duration, and selection of the types of subsystems have the predominant effect on the 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 (800 kilograms) 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 per man, the largest crew size considered (20 men), and a mission duration which exceeds the upper limit for the missions 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 crew size, mission duration, and free volume per man on the mission module mass are shown in Figure 28. The data are based on the oxygen-and-water environmental control and life support subsystem and an isotope and mercury-Rankine electrical power subsystem. The effect of free volume per man (from 400 to 1200 cubic feet per man) varies from 7 percent to 17 percent. The lower variation corresponds to a crew size of twenty men and a mission duration of 1500 days, while the upper variation corresponds to a crew of four men and a duration of 300 days. For all mission objectives except Jupiter and Ganymede, the mission durations are less than 800 days. For a nominal crew size of eight men and a mission duration of 700 days, the module mass increases from 22, 730 to 25, 575 kilograms (12. 5 percent) for the same variation in the free volume. For a nominal free volume of 750 cubic feet, the module mass is 24, 070 kilograms.

The degree of closure of the mission module environmental control and life support subsystem has been shown (Figure 21) to have a major effect on module mass. The open system is 25, 600 kilograms heavier than the system with water and oxygen recovery for a mission duration of 300 days and a crew size of eight men. This is an increase of more than 100 percent in the module mass. The system with water recovery only would be about 10 percent (26, 960 kilograms) heavier than the system with both water and oxygen recovery. The effect of trip time is also significant. For a 1400-day mission, a water-recovery system would be about 34 percent heavier than the more fully closed system compared to only a 10-percent increase for a 300-day mission.

Planetary Excursion Module

The planetary excursion module mass depends primarily on the mission objective, configuration, and occupancy time. The mass requirements for landing on Mars, Mercury, Ganymede, Vesta, and Ceres are shown in Figures 29 through 31. The data are based on circular planetary

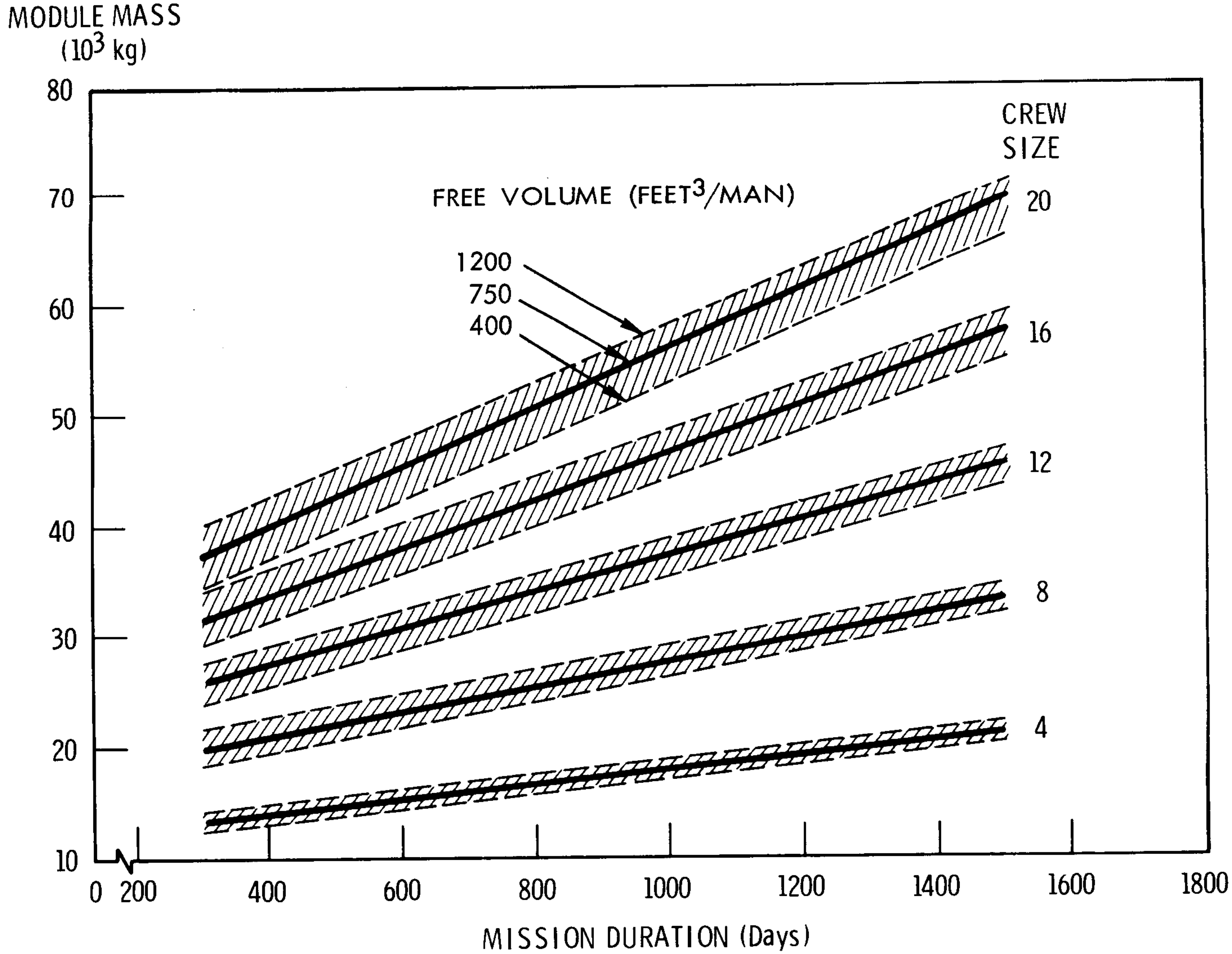

Figure 28. Mission Module Mass Comparsion

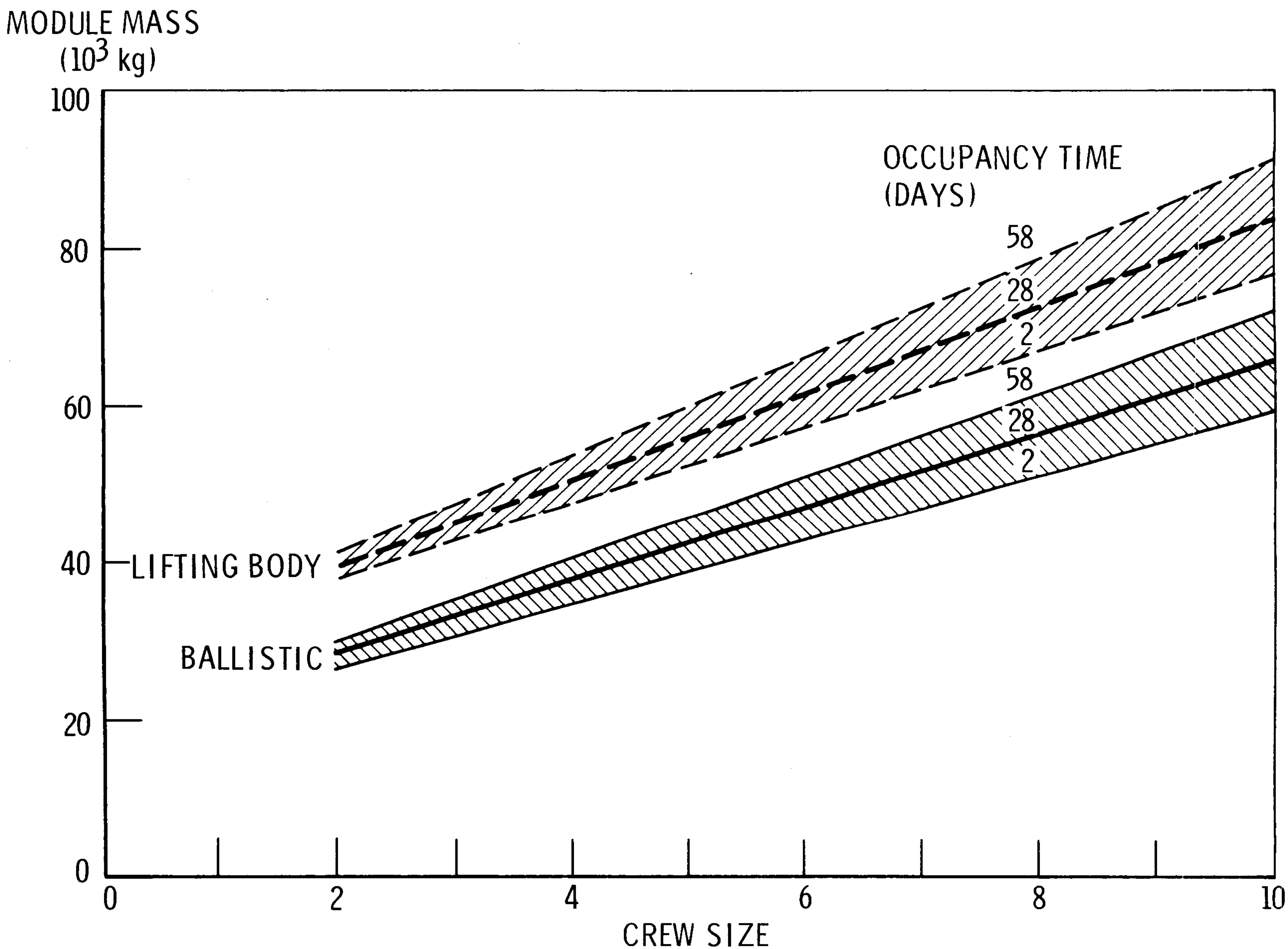

Figure 29. Mars Excursion Module Mass

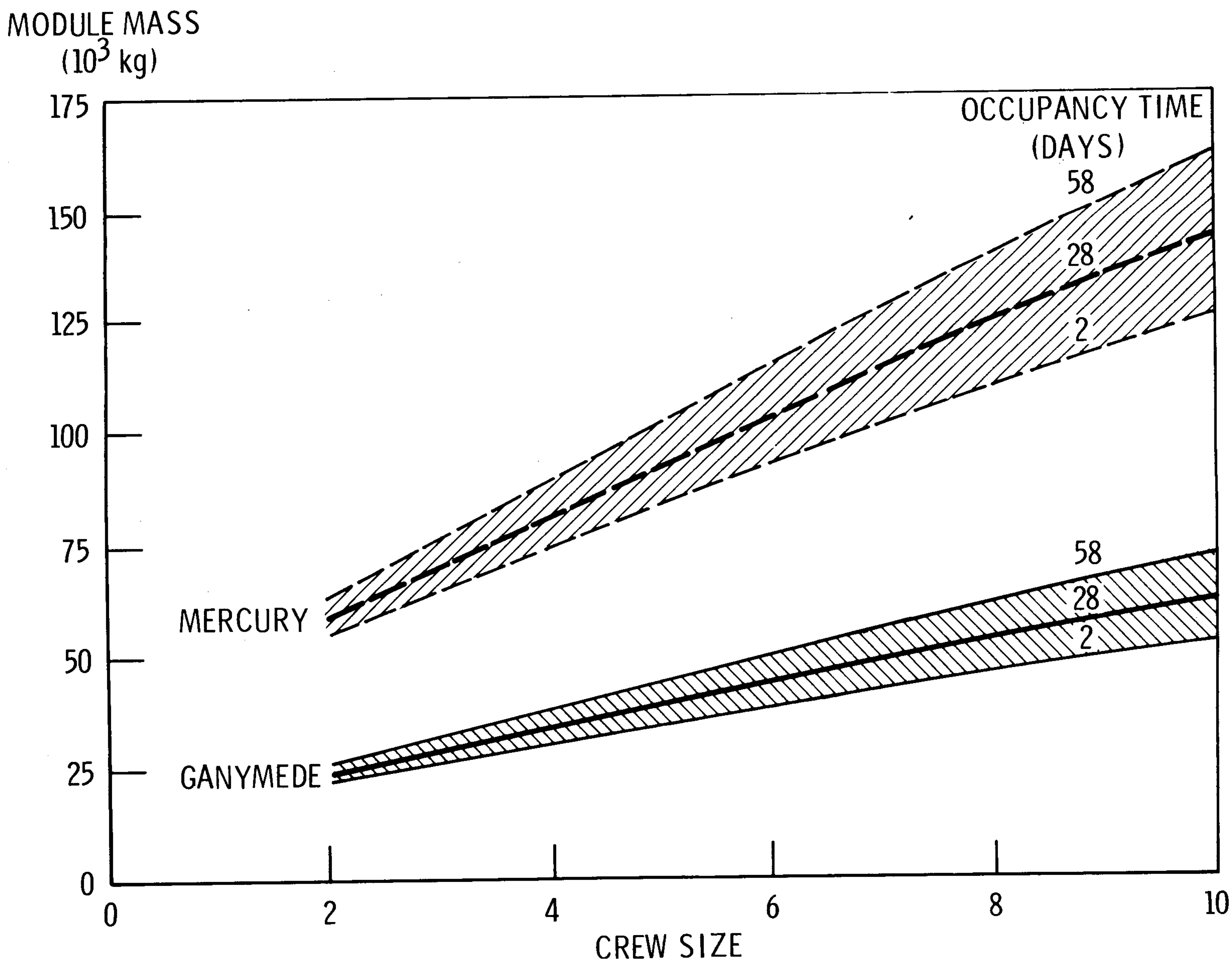

Figure 30. Planetary Excursion Module Mass (Mercury and Ganymede)

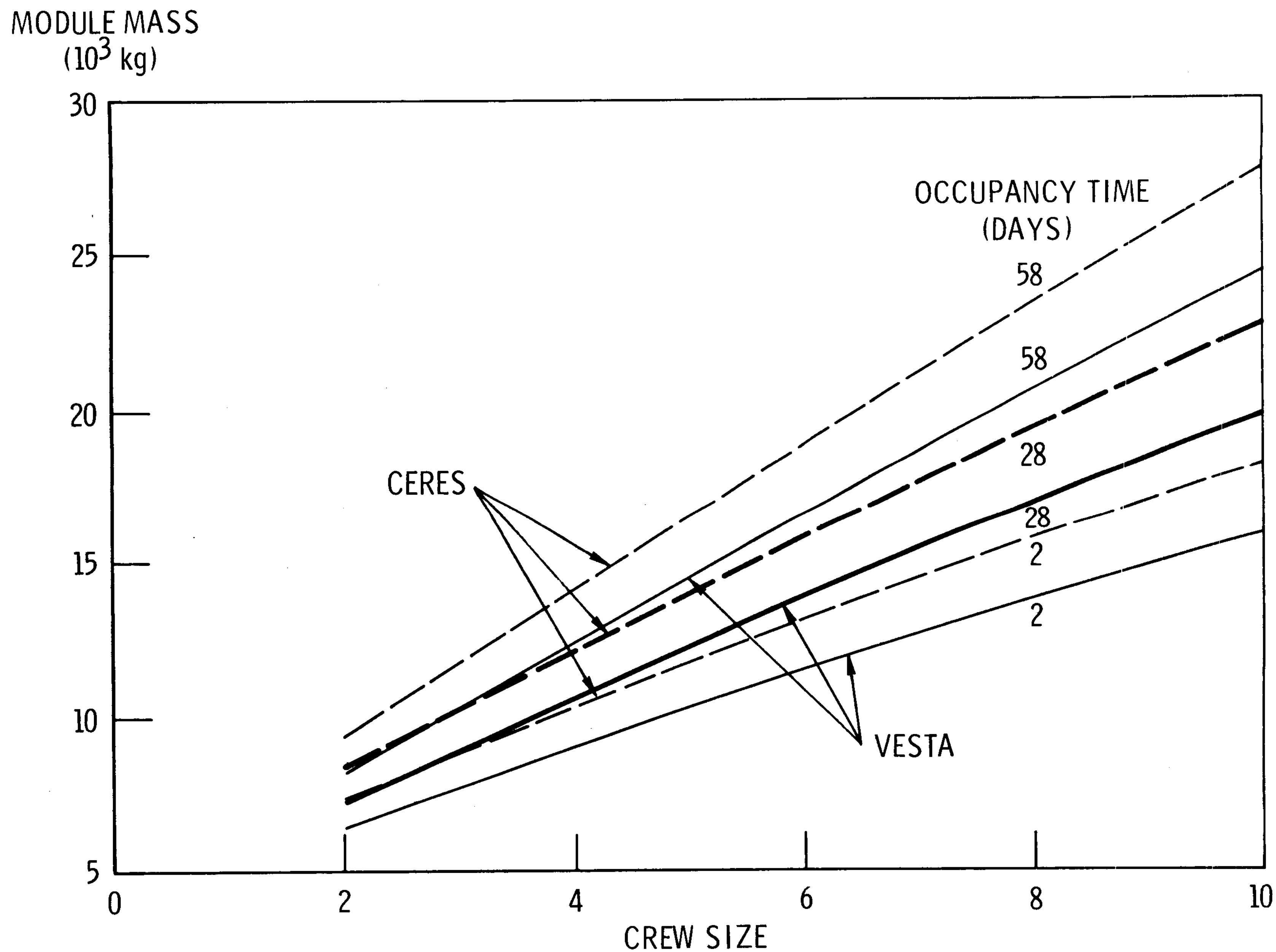

Figure 31. Planetary Excursion Module Mass (Ceres and Vesta)

parking orbits and exclude the effects of environmental considerations. Two configurations (lifting body and ballistic) were investigated for Mars landings. The configurations used for landing on Mercury, Ganymede, Vesta, and Ceres were similar to the current Lunar Module.

The effects of planetary parking orbit eccentricity on the planetary excursion module mass requirements are shown in Figure 32. As the parking orbit eccentricity increases, the characteristic velocity requirements also increase, resulting in an increase in the planetary excursion module mass requirements. The data are based on a crew size of four men and a planetary stay time of 28 days. The mass requirements include the effects of environmental considerations and the mass of the planetary excursion module shroud required for meteoroid protection during the transplanet mission phase. (It should be noted that the pericenter altitudes of the elliptical parking orbits are, in general, lower than the circular parking orbit altitudes.)

MISSION/SYSTEM DESIGN

To establish the common module requirements for future manned planetary exploration missions, the particular requirements of all potential missions must first be evaluated simultaneously, assuming the individual modules are designed for the specific mission objective and mission opportunity. The resultant family of modules can then be examined and module designs selected which satisfy the requirements of the maximum number of mission objectives, modes, and opportunities. The total system mass requirements were determined for representative mission opportunities for each of the mission objectives. Candidate common modules were then selected and the effects of using the common modules were evaluated in terms of the increased propulsion module mass requirements and the increased mass in Earth orbit. The results of these analyses are summarized in the following paragraphs. The details of the analyses are presented in Appendix D.

Optimized System Characteristics

The basic system-synthesis analyses were performed assuming that all modules were sized by the requirements of the mission objective and mission opportunity. These data provided the basic mass requirements for establishing common module requirements and for evaluating the penalties and advantages which result from the use of common modules. The initial analyses were performed assuming circular planetary parking orbits only. The circular-orbit restriction was imposed at the onset of the study because it was felt that elliptical orbits would inordinately complicate rendezvous operations and significantly increase launch-window requirements. Analyses conducted after the initiation of the study, however, have shown that only

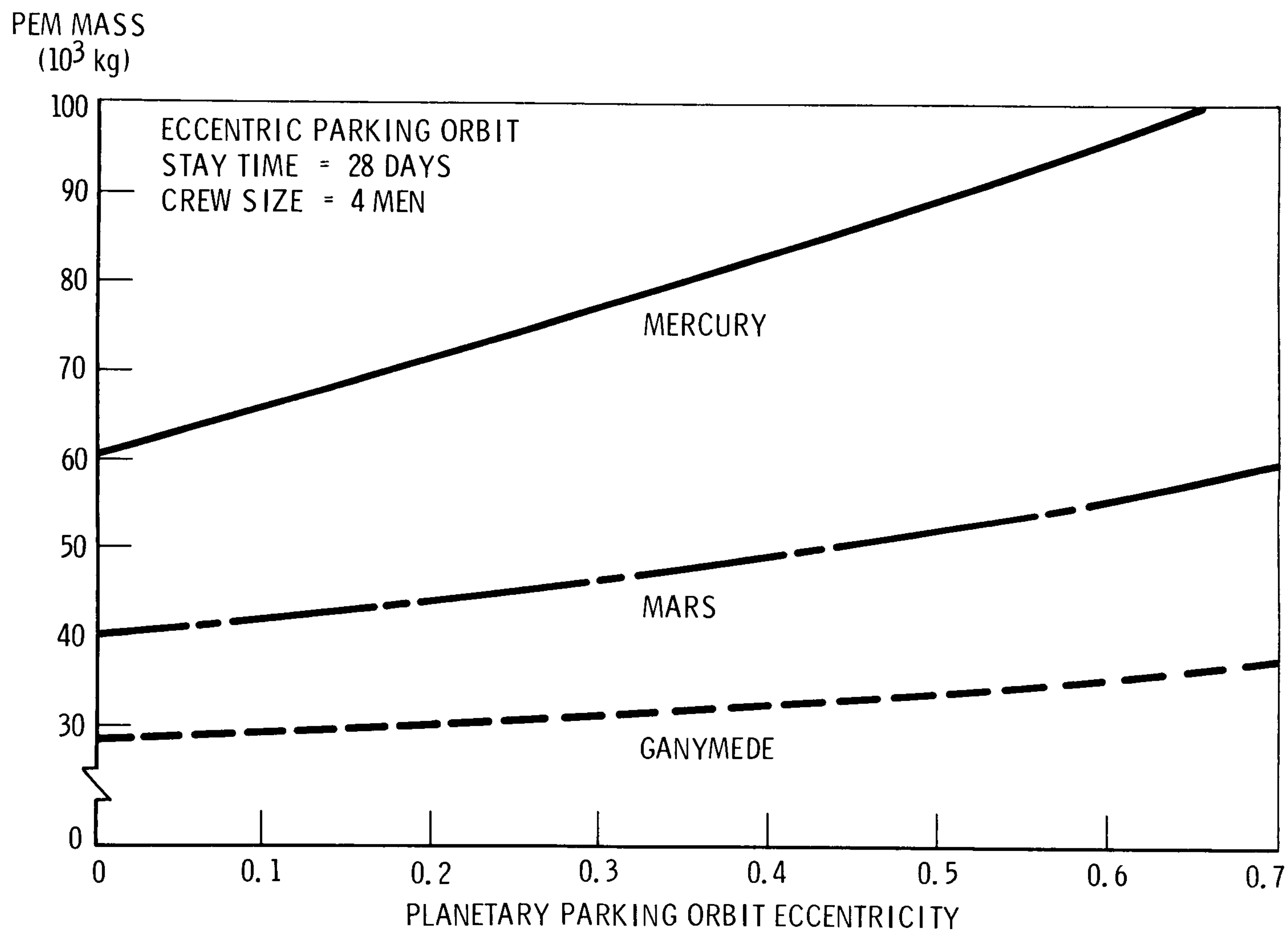

Figure 32. **Planetary Excursion Module Mass (Elliptical Planetary Orbits)**

modest performance penalties are incurred for performing off-pericenter planetary-orbit insertion and escape maneuvers. Therefore, the effects of using elliptical planetary parking orbits were investigated for missions to Mercury, Venus, Mars, Jupiter, and Ganymede under an amendment to the basic contract. Elliptical orbits were not considered for Ceres and Vesta since the use of elliptical parking orbits would not result in significant performance advantages because of the small mass of the asteroids.

Manned Modules

The total mass of the manned modules includes the mass of the basic module plus the additional mass requirements for environmental protection (thermal, meteoroid, and radiation). The masses (measured at the beginning of the mission) of the system configuration at the beginning of the trans-Earth mission phase are shown in Figure 33 for crew sizes of 8 and 20 men. The system at this point in the mission consists of the Earth reentry module, mission module and the trans-Earth midcourse correction propulsion module with sufficient propellant to perform midcourse correction maneuvers totaling 60 meters/second for each return mission leg, i.e., 60 meters/second for direct returns and 120 meters/second for swingby returns. The variations in the mass requirements for a given crew size are due to different Earth-reentry speeds, mission durations, and environmental-protection requirements. The module mass requirements for intermediate crew sizes can be approximated quite accurately by linear interpolation.

The planetary excursion module (PEM) mass requirements are dependent upon the eccentricity of the planetary parking orbit. For all mission objectives except Mars, both the ascent and descent characteristic velocity requirements increase as the planetary parking orbit eccentricity increases; this results in an increase in the PEM mass requirements. The PEM mass requirements are summarized in Table 28 for the limiting eccentricities which were considered in the study. The data are based on a PEM occupancy time of 28 days and include the mass of the interstage and the meteoroid protection required during the trans-planet mission phase. The mass requirements for intermediate eccentricities can be obtained quite accurately by linear interpolation.

Note that, from Figure 33 and Table 28, the weight of what might be termed "mission payload" (i.e., trans-Earth mass plus PEM mass) seldom exceeds 100,000 kilograms for the smaller crew sizes. This suggests the possibility of employing a Saturn V to place these systems in Earth orbit, regardless of the total mass-in-Earth-orbit requirements.

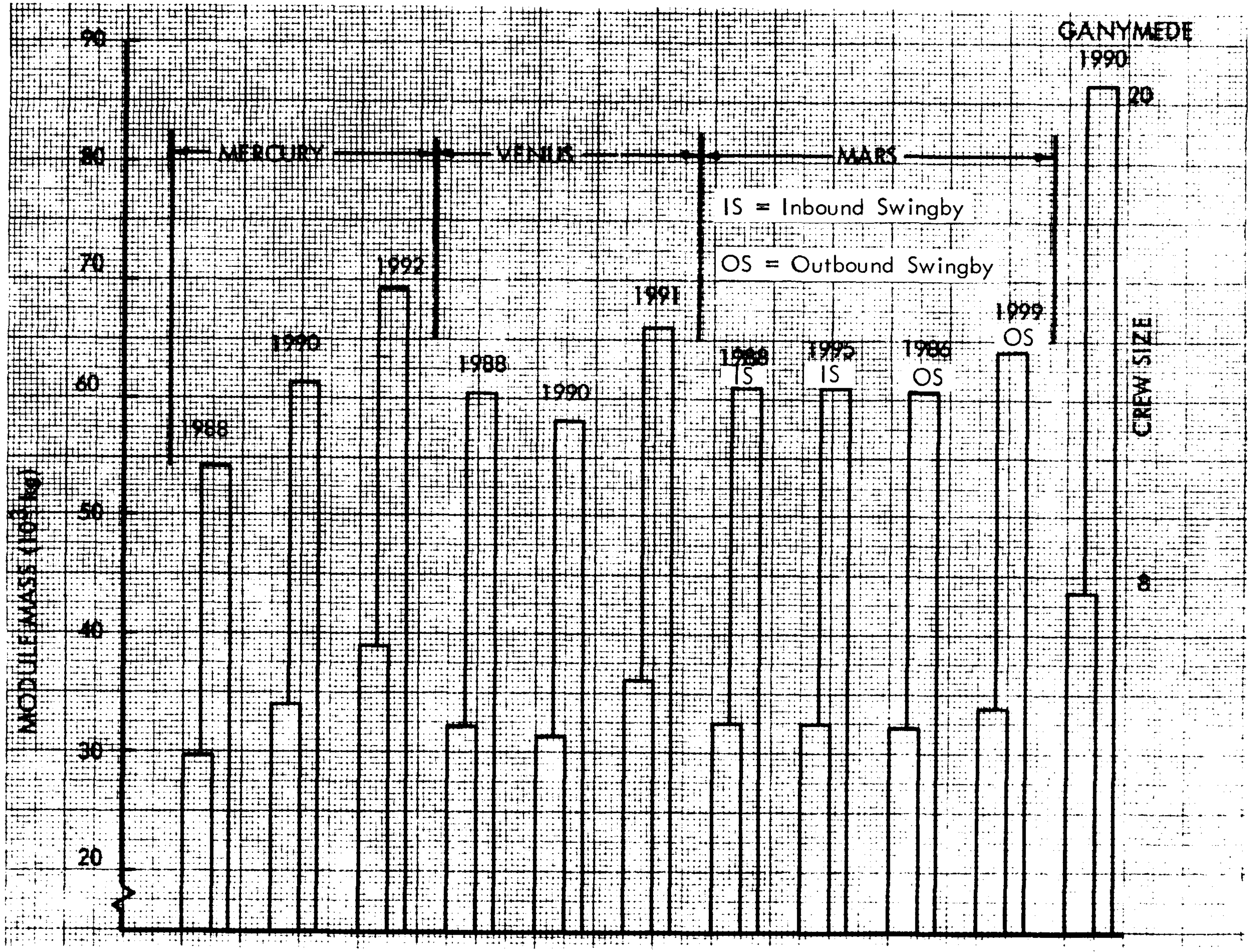

Figure 33. Transearth Mass Requirements

Table 28. Planetary Excursion Module Mass Requirements

	Planetary Excursion Module Mass (kg)			
	Four-Man Crew		Ten-Man Crew	
Mission Objective	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				

Propulsion Modules

The propulsion module mass requirements for a given propellant type are dependent upon the module payload, characteristic velocity requirements, and environmental protection requirements (thermal and meteoroid). The total propulsion module mass consists of the basic shell (tankage, accessories, etc.), engine, propellant (including boil-off propellant), meteoroid protection system, insulation system, and interstage structure. The engine mass was determined by optimizing the initial thrust-to-weight ratio. The insulation and boil-off propellant requirements for each module were optimized by minimizing the total system mass in Earth orbit. The meteoroid protection requirements were determined for each mission objective, and it was assumed that the protection was provided by a separate structure. The meteoroid protection shroud and the interstage were jettisoned prior to stage ignition.

Propulsion Module Mass - Circular Planetary Parking Orbits

The examinations of the chemical propulsion module mass requirements for circular planetary parking orbits were limited to Mars and Venus missions. The mass requirements were determined for all mission maneuvers for representative mission opportunities. The analyses included the determination of the mass requirements for planetary orbit insertion and escape for retrobraker missions and for Earth orbit escape for both retrobraker and aerobraker missions. The planetary orbit escape propulsion modules for aerobraker missions were determined as part of the aerobraker synthesis, since the escape module is an integral part of the aerobraker spacecraft at planet encounter.

The mass requirements of solid-core nuclear propulsion modules were determined for all maneuvers for all mission objectives. The resultant mass requirements for representative mission opportunities are shown in Figure 34 for circular planetary parking orbits. Included in the figure are the nuclear Earth-orbit escape propulsion module mass requirements for Mars and Venus missions which use cryogenic propulsion modules for planetary orbit insertion and escape. Also shown are the module mass requirements for Earth orbit escape for Mars aerobraker missions.

One significant result of the study can be seen from Figure 34 in that the propulsion module mass requirements are essentially continuous if all mission objectives and if both chemical and nuclear upper stages are considered for Mars and Venus missions. There are no natural divisions in the mass requirements which make the selection of common modules obvious. Even if some mission opportunities are eliminated without eliminating mission objectives, the mass requirements are still continuous in the lower range (600, 000 kilograms) of requirements.

Certain similarities in the propulsion module mass requirements can also be observed from Figure 34. The planetary orbit escape requirements for Vesta and Ganymede missions are comparable to the nuclear propulsion modules required for planetary orbit insertion for Mars and Venus missions. The planetary orbit insertion requirements for Mercury and Ceres missions are comparable to the requirements for either the planetary orbit insertion or the Earth orbit escape maneuver for Mars and Venus missions, depending upon the mission opportunity considered. Vesta planetary-orbit insertion requirements are similar to the Mars and Venus Earth-orbit escape requirements using nuclear upper stages, while Ganymede missions and the low energy Mercury and Ceres missions have requirements similar to the Earth-orbit escape requirements for Mars and Venus missions which use cryogenic upper stages.

An investigation was conducted to determine the effects of the mission profile and the meteoroid environment on the mass requirements for Ganymede missions. The nominal mission profile consists of a single plane transfer from Earth to Jupiter and Ganymede and from Jupiter and Ganymede to Earth. The alternate mission profile consists of a two-plane transfer for each mission phase such that the heliocentric conic is approximately 0. 5 A. U. out of the plane of the ecliptic at the radius of the center of the asteroid belt (2. 8 A. U.). The mass in Earth orbit requirements associated with the out-of-the ecliptic mission profile are only 9 percent greater than the requirements for the nominal profile with a nominal

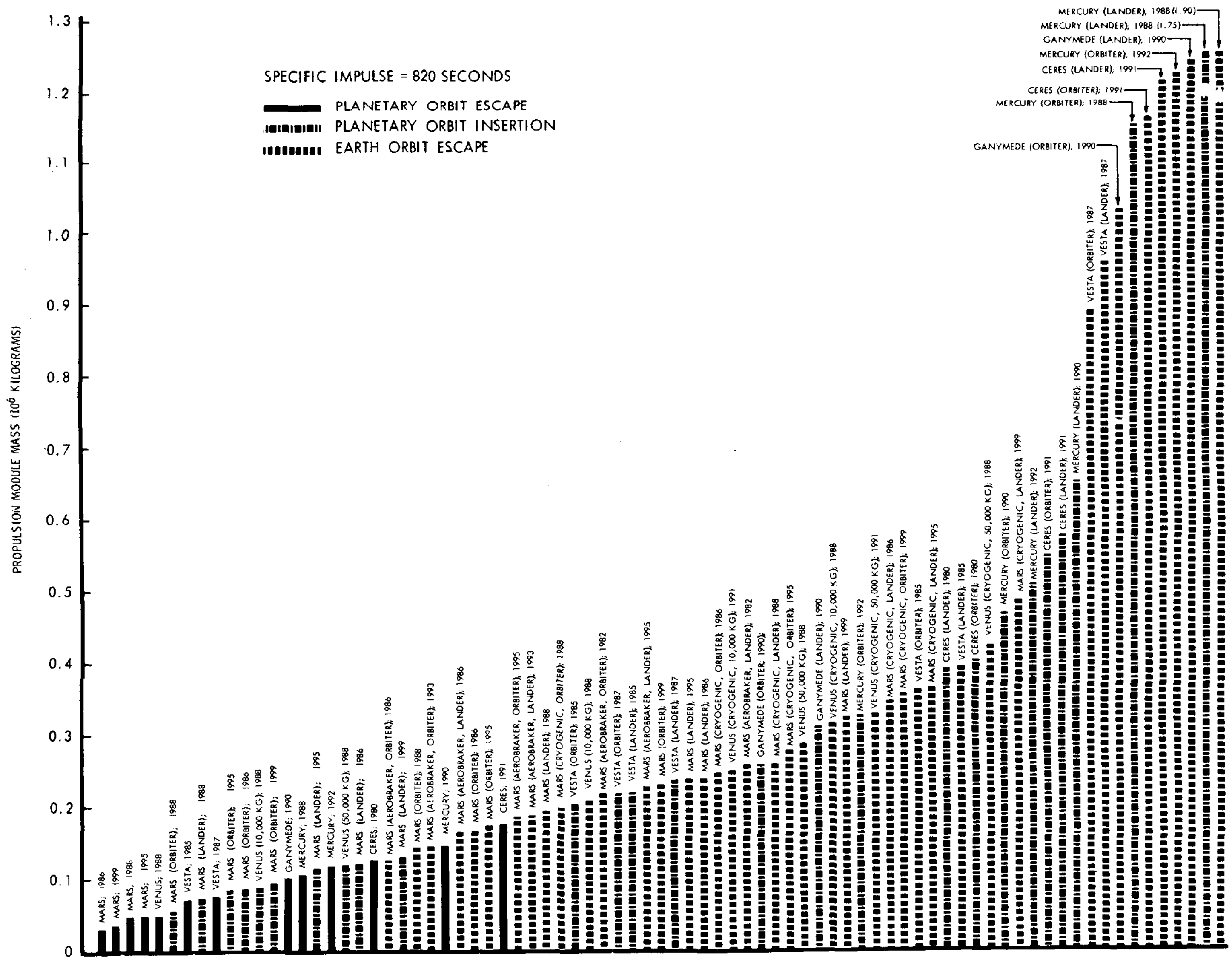

Figure 34. Nuclear Propulsion Mass Module Requirements (8-Man Crew)

meteoroid environment. If the maximum environment is considered with the nominal mission profile, additional shielding is required for all modules, which increases the mass in Earth orbit by more than a factor of three. The relatively small increase of the mass-in-Earth-orbit requirements associated with the out-of-the-ecliptic profile and the uncertainty in the asteroidal environment makes the out-of-the-ecliptic profile particularly attractive. It appears that this mission mode should be given serious consideration during the definition of the mission and system requirements for all (manned and unmanned) missions to Jupiter.

The propulsion module mass requirements were also determined for gaseous core nuclear propulsion modules. The analyses were based on a specific impulse of 2500 seconds and an engine thrust-to-weight ratio of eight. The mass of the largest single module does not exceed 300, 000 kilograms; the majority lie below 100, 000 kilograms.

Propulsion Module Mass - Elliptical Planetary Parking Orbits

The mass requirements of chemical propulsion modules were determined as a function of parking-orbit eccentricity for representative Mars and Venus mission opportunities. The results are summarized in Figure 35 which shows the mass requirements as a function of eccentricity for crew sizes of 8 and 20 men for Mars and Venus retrobraker missions and Mars aerobraker missions. For the Mars missions, the range of requirements throughout a cycle of launch opportunities are indicated by the two families of curves (solid and dashed curves). In all cases, the lower curve of each set corresponds to a crew size of 8 men while the upper curve corresponds to a crew size of 20 men. The mass requirements for intermediate crew sizes can be estimated quite accurately by linear interpolation.

The significance of the planetary parking orbit eccentricity is quite apparent from Figure 35, particularly for Venus missions. The planetary orbit escape propulsion module mass requirements for Venus missions can be decreased by over 50 percent by increasing the eccentricity from zero (circular orbit) to 0. 7. The planetary-orbit-insertion requirements can be decreased by over a factor of four, while the Earth-orbit-escape requirements can be decreased by a factor of approximately three. Also of significance for Venus missions is the comparison between the planetary-orbit-insertion and planetary-orbit-escape module mass requirements at the higher eccentricities. Although the planetary-orbit-insertion module payload is greater, the insertion incremental velocity requirements are between 55 and 75 percent of the planetary-orbit-escape requirements, resulting in nearly identical propulsion module mass requirements.

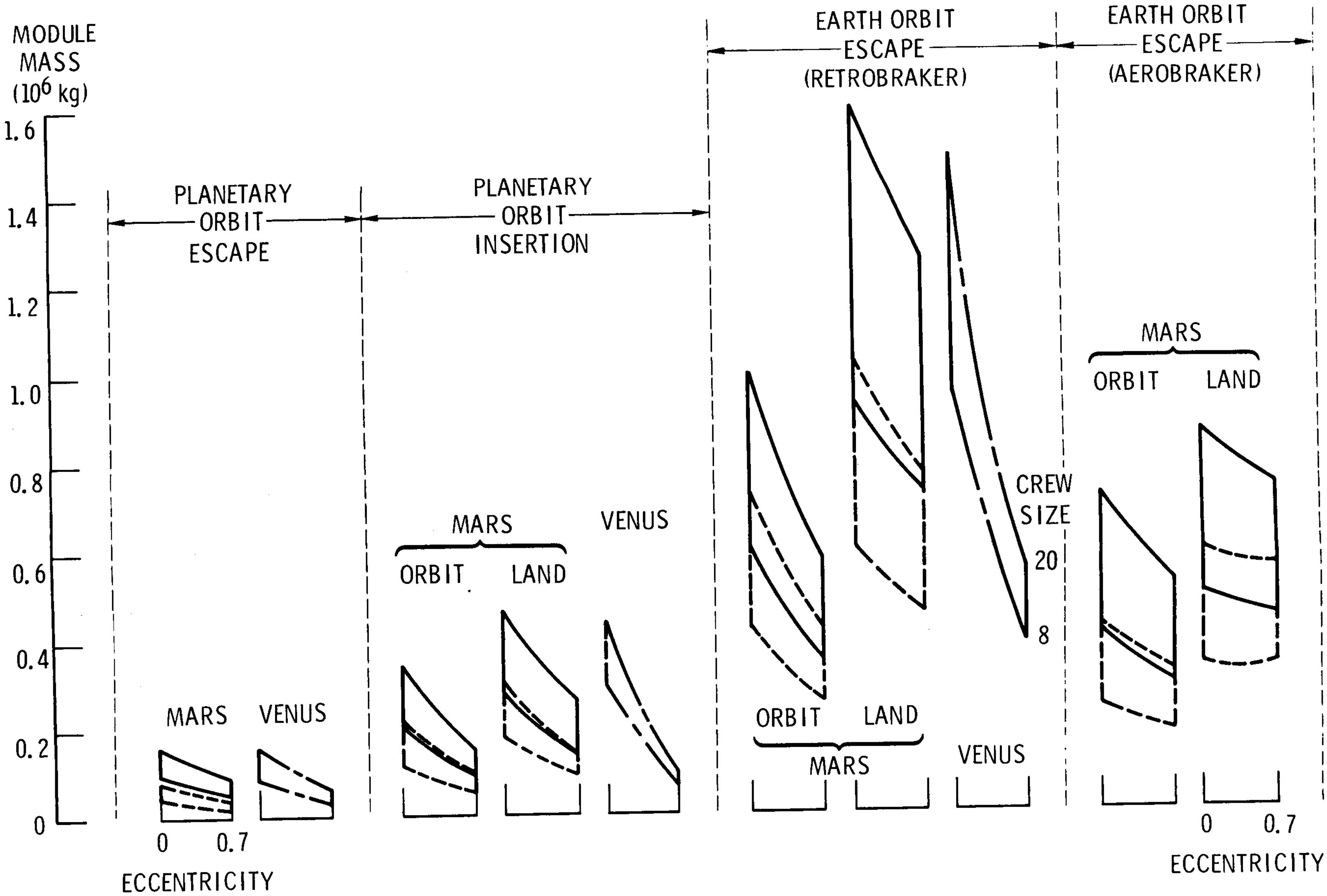

Figure 35. Chemical Propulsion Module Mass

The mass requirements of solid core nuclear propulsion modules are summarized in Figures 36 and 37 for Mercury, Venus, Mars, Jupiter (h_e = 10 radii), and Ganymede missions. The range of requirements is shown for the Mars and Mercury missions only since the variations in the requirements for Venus, Jupiter, and Ganymede missions over a cycle of opportunities are relatively small.

The significance of planetary-parking-orbit eccentricity on the mass requirements for Venus missions is again apparent. Of even more significance is the effect of eccentricity on the mass requirements for Jupiter missions. It can be seen that, if highly eccentric orbits about Jupiter are considered, the mass requirements of the planetary-orbit-insertion and planetary-orbit-escape propulsion modules are comparable to the mass requirements for insertion and escape for Mars and Venus missions. Also, the mass requirements of all propulsion modules for the Jupiter orbiter missions with high parking orbit eccentricities are less than the requirements for either the Ganymede orbiter or lander missions at all eccentricities. Therefore, the desirability (on the basis of mass requirements alone) of either a Jupiter orbiter mission or a Ganymede orbiter or lander mission is dependent upon the parking orbit eccentricities considered.

Common System Characteristics

The results which are summarized in the previous paragraphs are based on the assumption that all modules are sized by the requirements of each particular mission objective and mission opportunity. The results of the analyses to establish the feasibility of utilizing common manned and propulsion modules are summarized in the following paragraphs.

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., over-designing) the upper stages to the mass requirements of the lower stages was included. This rather time-consuming 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.

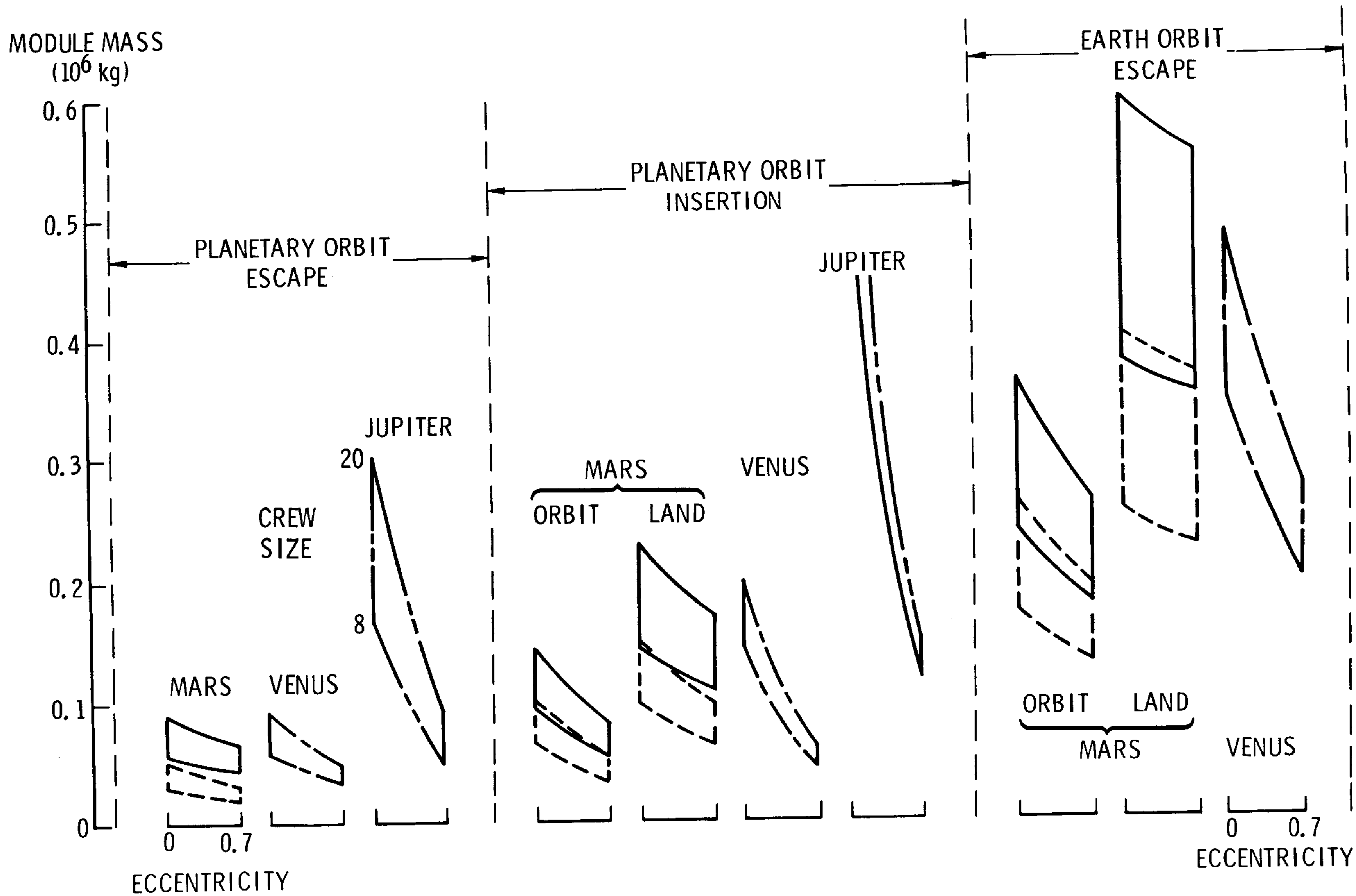

Figure 36. Nuclear Propulsion Module Mass (Mars and Vensus)

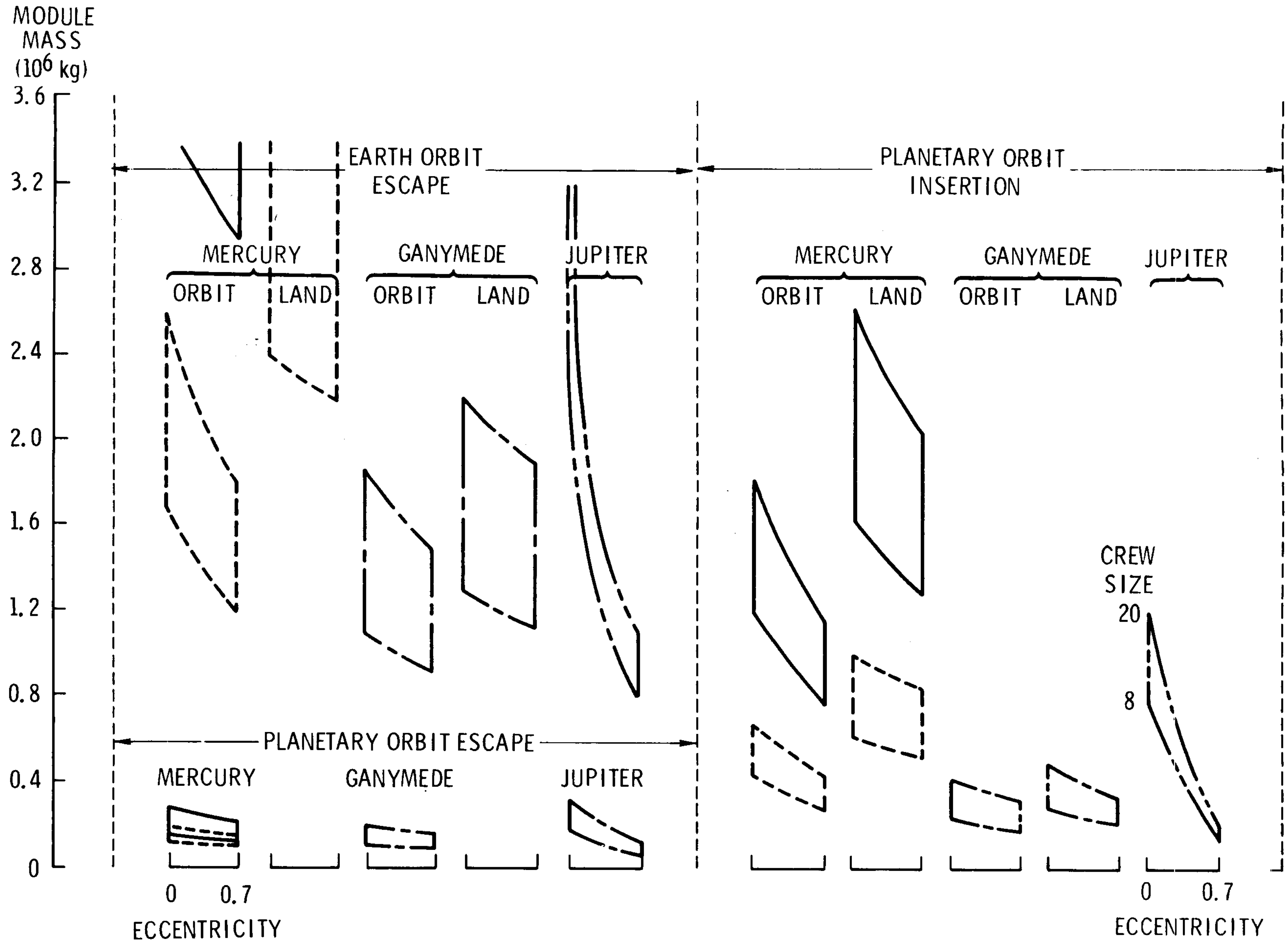

Figure 37. Nuclear Propulsion Module Mass (Mercury, Jupiter, and Ganymede)

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.

The investigations of common modules were based on mass requirements alone. Other factors will also effect the selection of future modules—for example, the development cost and development time. Operational factors must also be considered. These include the compatibility of the modules with the launch vehicle(s), the compatibility of the launch vehicle(s) with the launch site facilities, the number and frequency of launches, in-orbit assembly time, more precise definition of the module weights, and scientific mission objectives insofar as they influence spacecraft weight.

Common Manned Modules

An examination of the Earth-reentry speeds (Tables 1 through 16) shows that the reentry speeds are generally less than 15 kilometers/second. The major exceptions are the Ceres and Mercury missions and the direct Mars missions. The Earth-reentry speeds for the Ceres missions can be reduced only by significantly increasing the incremental velocity requirements. Omission of missions to Ceres would seem rather unimportant, particularly since entry speeds for Vesta missions lie within a 15-kilometer/second limit. The high reentry speeds associated with the direct Mars missions can be avoided by considering only the Venus swingby mission mode. This is also the more attractive mission mode when propulsive requirements are considered. The reentry speeds for Mercury missions can be reduced by limiting the mission opportunities which are considered. Limiting the missions on the basis of reentry speed is also compatible with the elimination of mission opportunities on the basis of excessive performance requirements.

The Earth reentry module mass requirements were shown in Figure 27. As can be seen from the figure, the biconic configuration has a slight mass advantage for reentry speeds between 14.2 and 15.0 kilometers/second. The slight mass advantage does not appear to warrant the development of a new generation of reentry modules, however. It is therefore concluded that, on the basis of the parameters which have been considered in the present study, the Apollo configuration will satisfy the requirements of future manned planetary missions. Other considerations which may make the development of a second configuration desirable, e.g., abort, have not been considered.

Common mission modules could be achieved by two methods. First, the mission modules can be developed in a modular manner in which the number of floors are increased as the crew size is increased. As an example, a single module could be developed which could be used for crew sizes from eight to twelve men, with the consumables added as required by the mission duration. As the crew size increases, additional floors could be added and the additional consumables provided. An alternate approach would be to develop a single module which is designed for some maximum mission duration and crew size and to then off-load crew and consumables as required for missions which impose lesser requirements. This latter approach becomes unattractive if major crew off-loading is attempted. For example, if a module designed to accommodate 20 men were employed in missions which carried only 8 men, mass-in-Earth-orbit penalties of 20 to 30 percent would result. Regardless of which approach is used, it is assumed that the meteoroid and radiation protection would be sized for the particular mission. This assumption seems reasonable since the environmental protection requirements would probably consist of an incremental structure which is added to the basic structure and could be conveniently sized for a given mission objective and mission opportunity.

The only feasible areas for designing common planetary excursion modules would be among the retrobraking PEM's. For a given crew size, the only differences in the ascent stages of the PEM's would be in the amount of propellant provided for ascent and in the ascent-stage engine thrust. Thus, a common ascent stage could be developed which provides the basic structure and equipment for the crew, but which has different propellant tanks and engines for a given mission objective. As an alternative, common propellant tanks could be used and off-loaded as required. The descent stages fall into two basic categories: a relatively large module for landings on Mercury and Ganymede, and a relatively small module for landings on Ceres and Vesta. Thus, two common descent stages could be developed which are sized on the basis of the requirements for the Mercury and Ceres missions.

Common Propulsion Modules - Circular Planetary Parking Orbits

The examinations of common chemical propulsion modules were limited to the establishment of potential common modules which could be used to satisfy the requirements of all maneuvers for the majority of the Mars and Venus missions. The evaluations were performed on the basis of an eight-man crew under the assumption that larger crew sizes could be used during missions which have more modest performance requirements.

It was determined that one module with a mass of approximately 100,000 kilograms could be used for planetary-orbit insertion and escape. This module could not be used for Earth-orbit escape, however, without excessive clustering so that a second module would have to be developed.

The chemical Earth-orbit escape module could be on the order of 500,000 kilograms and could be used either singly, in pairs, or in combination with the 100,000-kilogram module to accomplish the Earth-orbit escape maneuver for all missions considered. An alternative would be to develop either a 300,000-kilogram module or a 600,000-kilogram module. Of the modules considered, the 100,000-kilogram and 300,000-kilogram combination appears to be the most attractive.

Extensive analyses were conducted to establish common nuclear propulsion modules since they are the only high-thrust modules which can be sensibly applied to the entire spectrum of missions considered. The analyses were limited to the examination of common solid-core propulsion modules since their application is considered to be, at this time, less speculative than the use of gaseous-core systems.

As noted previously, the solid-core nuclear propulsion module mass requirements are essentially continuous when all mission opportunities are considered for all mission objectives. A limited number of discrete bands of requirements can be obtained by limiting the crew size to a given value and the mission opportunities to those opportunities which have the more modest energy requirements. Even after imposing the above restrictions, the mass requirements, assuming an eight-man crew, are still essentially continuous up to approximately 600,000 kilograms. The feasibility of selecting discrete propulsion modules within this band was investigated in detail, assuming only two propulsion module sizes were to be developed. It was determined that a 75,000-kilogram module could be used singly for planetary-orbit escape, and that either one or two of the modules would suffice for planetary-orbit insertion for all Mars and Venus missions. Two of these modules could be used for planetary-orbit escape for Mercury and Ganymede missions.

Additional propulsion modules would be required to perform the remaining manuevers. After examining the effects of using a common 75,000-kilogram module where applicable, a second module was selected which has a mass of 300,000 kilograms. The module could be used either singly, in pairs, or in combination with the 75,000-kilogram module to satisfy the propulsion module requirements for all remaining maneuvers except the Earth-orbit escape requirements for the Mercury, Ceres and Vesta, and Ganymede missions.

Common Propulsion Modules - Elliptical Planetary Parking Orbits

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. Regions of common requirements are not as apparent when elliptical planetary orbits are considered because of the extreme variations in the propulsion module mass requirements. By examining the various propulsion module mass requirements, from data such as shown in Figures 35 through 37, it was possible to identify several propulsion module combinations that seem appropriate.

Summary of Common Propulsion Modules

The results of the propulsion module commonality evaluation are summarized in Table 29. Representative module sizes are shown for each propulsion-system combination considered. The values shown reflect the best compromises that could be made between the variation in requirements brought about by the large eccentricity and crew-size variation. To interpret the format of the table consider the all-nuclear (NNN) system. The first option is to develop two modules (a 75, 000-kilogram module and a 300, 000-kilogram module); the second option is to develop three modules (75, 000 kilograms, 300, 000 kilograms, and 1, 200, 000 kilograms); and so forth. Note that the location in the mission at which a module of a given size might be used is of no concern at this point.

The applicability of these various modules to the family of missions considered in this study are shown in Table 30. Several interesting conclusions are apparent from the table: 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 29. Candidate Common Propulsion Modules

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

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

Table 30. 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 such 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 lengthy 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.

ERRATA

Technical Requirements Common to Manned Planetary Missions

Technical Summary, SD67-621-1

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 56 - t = mission duration (weeks)

3. Page 107 - second sentence, second paragraph: Delete exclusion of Ganymede.

4. Page 108 - Table 29: The entries for EDE Flyby requirements should be on three separate lines.

5. Page 108 - Table 30: 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.