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SD 67-621-5

FINAL REPORT

TECHNOLOGICAL REQUIREMENTS COMMON TO MANNED PLANETARY MISSIONS

(Contract NAS2-3918)

APPENDIX D

System Synthesis and Parametric Analysis

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

SD 67-621-5

TECHNOLOGICAL REQUIREMENTS COMMON TO MANNED PLANETARY MISSIONS

NAS2-3918

Appendix D - System Synthesis and Parametric Analysis

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SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

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. The 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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SYSTEM SYNTHESIS AND PARAMETRIC ANALYSIS

Parametric designs have been made of the modules that form the systems (vehicles) to accomplish the designated missions. Conceptual design drawings have been made as necessary to allow formulation of weight-scaling equations based on realistic configurations. These equations are used in the Weight Synthesis Computer Program to examine the sensitivity of these vehicles to variations in design approach and uncertainties in design requirements. Conceptual designs will be discussed first, followed by the resulting weight-scaling equations, weight synthesis methodology, and parametric analysis.

CONFIGURATION DESIGN

This section summarizes the conceptual design activity performed during the study. The scope of this activity encompasses the development of two Earth reentry modules, six planetary excursion modules, two mission modules, and one aerobraking spacecraft.

Since a basic objective of this phase of the study was the generation of scaling equations for the various spacecraft modules for a broad spectrum of planetary missions, the preparation of conceptual point designs to serve as an anchor point for the scaling equations was considered to be essential. The point designs serve as a check on the weight synthesis. Specifically, the designs would illustrate the packaging requirements and constraints of the various modules, the on-board propellants and propulsion systems, and would delineate general structural requirements and staging concepts. Analyses of the "first cut" concepts generally show that a disparity exists between the concept as it is designed, and the original synthesized weight statement upon which the concept was generated. Normally, refinement of the synthesized preliminary weights would be accomplished by a design iteration to narrow the spread between the two. However, for the purposes of this study, this procedure is not necessary. Sufficient accuracy of the weights can be achieved without revising the drawings. Therefore, the drawings discussed in this section are to be considered only as a guide to the formulation of weight scaling equations and not as optimum designs.

One of the initial planning tasks performed at the beginning of the study was the preparation of a matrix of all of the spacecraft modules and crew sizes that were of potential interest to the study. In an attempt to reduce the number of concepts to be configured during the study, a summary of previous SD studies was conducted to identify existing module concepts that would be directly applicable. Earth reentry module (ERM) concepts for three-, six-, and eight-man crews in the Apollo and biconic shapes were available from previous SD studies of Mars flyby and aerobraking spacecraft. Apollo-shaped ERM's for larger crew sizes were obtained from in-house studies of advanced Apollo mission studies. Planetary excursion modules (PEM) had been designed for Mars for smaller crew sizes (three or four) for other studies, as had mission modules (MM) for crew sizes up to eight. Previous SD studies of a broad spectrum of aerobraking spacecraft were adequate for the crew sizes of three, six, and eight for cryogenic storable propellants and nuclear propulsion systems. Subsequent to a detailed review of the available concepts, it was possible to identify the desired new point designs which would be appropriate to generate in this study.

The matrix of modules and crew sizes is shown on Figure 1. The items designated "E" are applicable existing designs that were considered suitable for this study. The items designated "N" are the point designs which were developed in the course of this study.

EARTH REENTRY MODULE DESIGN

Biconic ERM's were developed for crews of 14 and 20 and are shown respectively in Figures 2 and 3 . The basic biconic shape used for these design features is a right-elliptical cone afterbody and an elliptical fore cone as developed by Lockheed. The crewman used for the internal arrangement is a 90-percentile astronaut in an inflated pressure suit. The 14-man vehicle (Figure 2) has an overall length of 6.65 meters (262 inches), an overall height of 3.27 meters (129 inches), and a gross moldline volume of 29.74 cubic meters (1050 cubic feet). A 14-man crew is accommodated on four levels of seating, with numbers of 3, 3, 4, and 4 in successive levels from the uppermost level down. The crew ingress/egress hatch is located in the upper surface above the second level of crew. Although no attempt was made to provide internal equipment arrangements, past studies have shown that there is generally sufficient volume available, in vehicles with more than three crew members, after the crew seating arrangement has been developed to accommodate the on-board subsystems and position them to provide a satisfactory vehicle, center-of-gravity location.

The 20-man vehicle (Figure 3) has an overall length of 7.49 meters (295 inches) an overall height of 3.7 meters (145.5 inches) and a gross moldline volume of 42.48 meters (1500 cubic feet). The 20-man crew is accommodated in 5 levels of 3, 3, 4, 5, and 5, respectively, from the uppermost level. The ingress/egress hatch is positioned in the upper surface over the middle level of crew.

These two concepts complete the family of point designs needed for the Earth reentry modules.

Module	Type	Crew Size 2	4	6	8	10	12	14	16	18	20
ERM	Apollo		← E →		E	E		E		E (19)	
	Biconic		← E →		E			N			N
PEM	Apollo MEM		← E →			N (2 versions)			N (Deleted)		
	Lifting Body MEM		← E →			N			N		
	(Ceres/Vesta) Retro		N (3)			N					
	(Ganymede/Mercury) Retro		N (3)			N					
	Venus	N									
MM	Elliptical Bulkhead		← E →		E			N			N
Aerobaker	Cryo Propellant				E			N			
	Storable Propellant		← E →		E						
	Nuclear Propellant				E						

Figure 1. Configuration Matrix

PLANETARY EXCURSION MODULE DESIGN

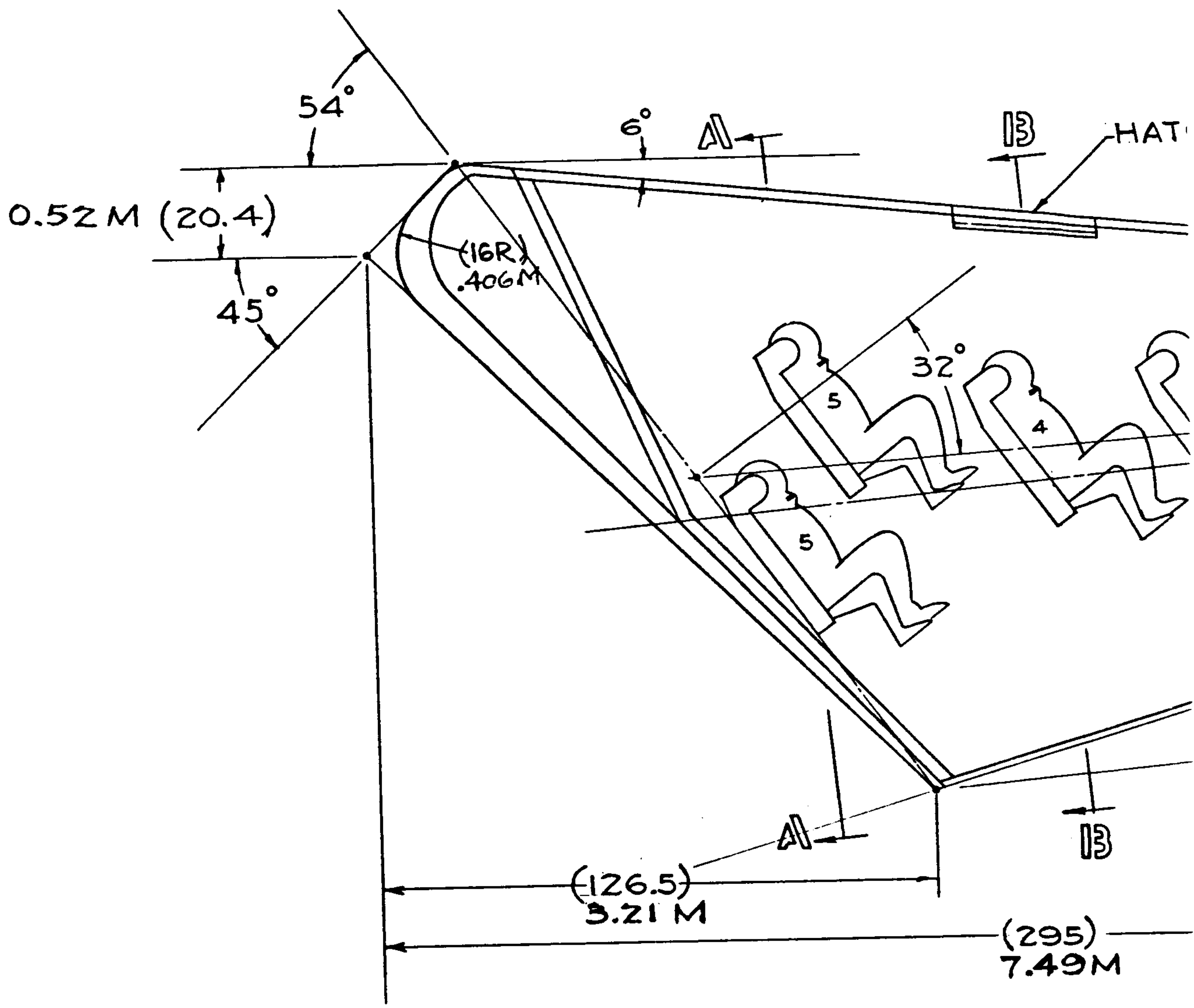
Ballistic Mars Excursion Module (MEM)

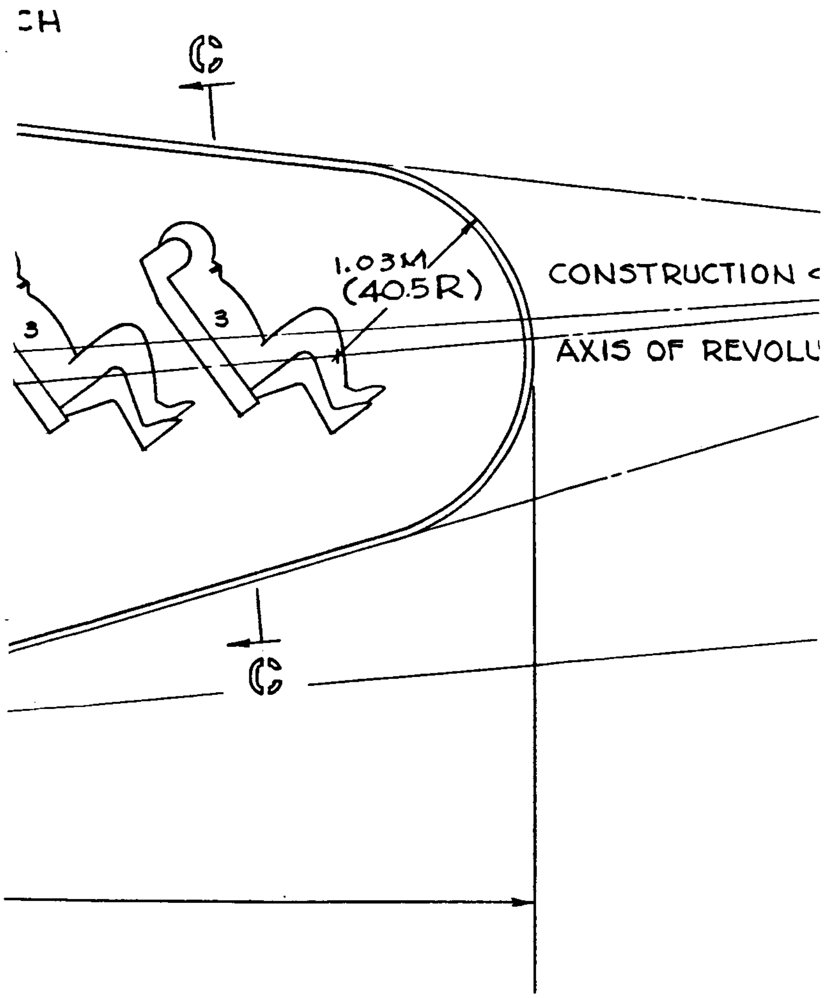
Two point designs for ballistic MEM's were designated on the module concept summary as being of interest. These vehicles were to be sized for crews of 10 and 16. Subsequent to the conceptual design of the 10-man vehicle, which is described in the following paragraph, the requirement to examine the 16-man vehicle was deleted from the study. This decision was a direct consequence of the size trend indicated by the 10-man point design.

Two conceptual designs for the 10-man vehicle have been prepared, the second concept being a variation of the Apollo shape to provide higher volumetric efficiency for larger crew sizes. The first concept shown in Figure 4 , is basically a modified Apollo shape and is based upon recent SD studies of manned planetary excursion modules of three- to four-man crew for Mars. As a direct consequence of the revision of the estimates of the density of the Martian atmosphere, the shape of the ballistic vehicles has been altered from that of the basic Apollo, which has an aft body cone angle of 64 degrees. To provide the significant increase in base area for a lower W/C_DA and without an excessive increase in surface area of the aft body cone, the aft body angle was increased to 90 degrees. The basic vehicle has an overall height of 7.92 meters (312 inches) and a base diameter of 13.56 meters (534 inches). The gross weight is estimated to be 5,300 kilograms (117,000 pounds). The general arrangement consists of a single-level crew compartment that serves as the mission module for the stay time on the surface of the planet. A single propulsion system using OF_2/MMH propellants is used for terminal retro and landing and eventual ascent.[1] Four sections of the heat shield are hinged and deployed to serve as landing shoes. At initiation of ascent, the center cone of the vehicle separates and pulls out of the peripherical structure which contains the landing gear.

At the completion of this first concept, shown in Figure 4 , it was apparent that the concept was extremely inefficient from staging considerations because the entire living quarters were being returned as part of the ascent stage. The alternate design made to improve the staging efficiency is shown in Figure 5 . In the alternate concept, the crew compartment is essentially of a high-density loading configuration, with the crew accommodated in three levels of seats for landing and return only. The two upper crew members have multiposition seats to permit visual assessment of the terrain for final touchdown. The other crew seats are fixed in a single position to accommodate entry deceleration and ascent acceleration "g" loadings. The primary portion of the living quarters and laboratory space is located in the lower landing stage. Access to this pressurized section is through crew compartment floor hatches and descent through a passage-way between the fuel tanks. A total of 29.4 M^3 (1040 ft^3)

[1] FLOX/CH_4 is also considered to be an attractive space storable propellant. Because of similarities in bulk density and specific impulse, the configurations shown are appropriate for this propellant.


54°
6°
A
B
HAT
0.52 M (20.4)
(16R)
.406M
45°
32°
5
4
5
A
B
(126.5)
3.21 M
(295)
7.49M



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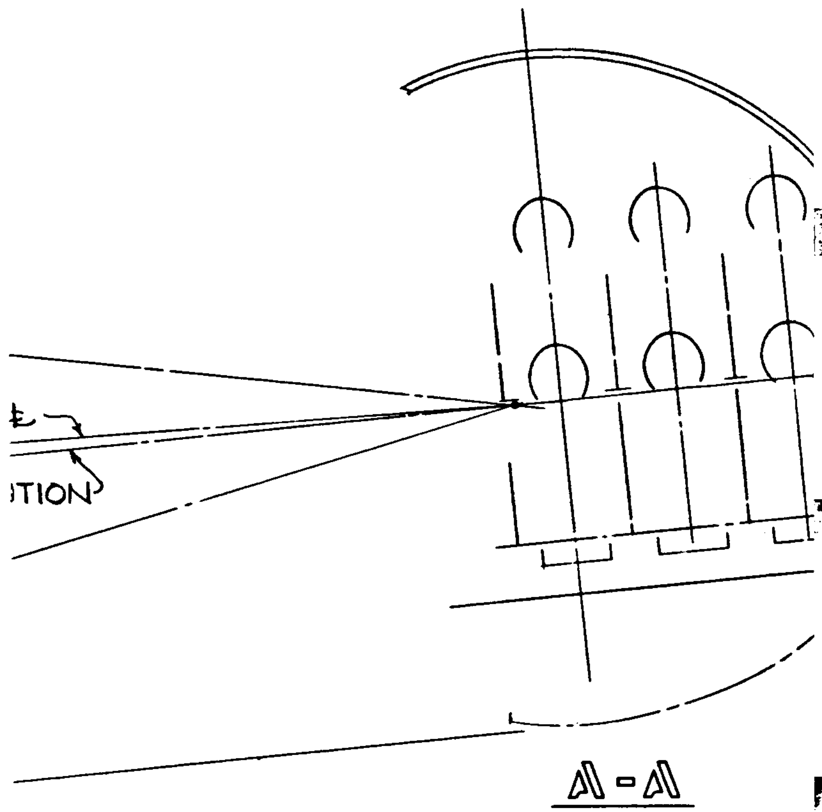

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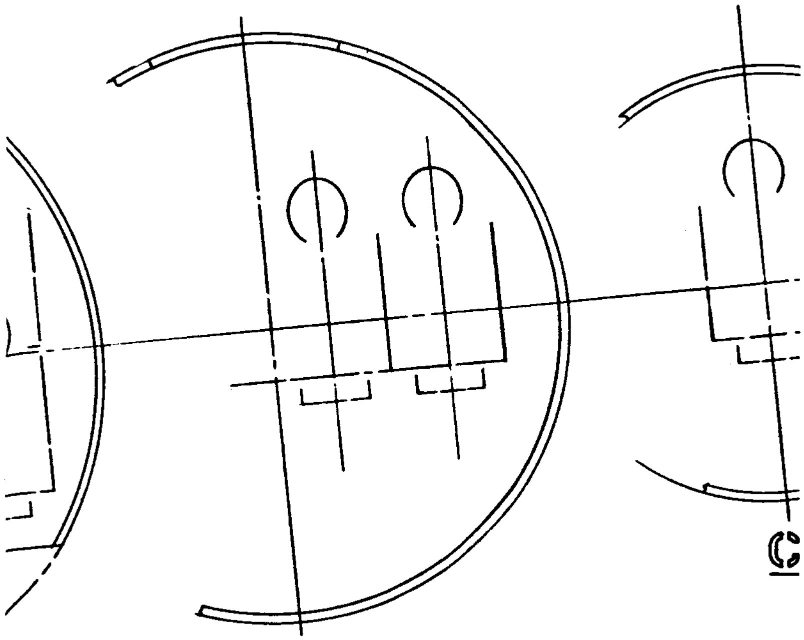
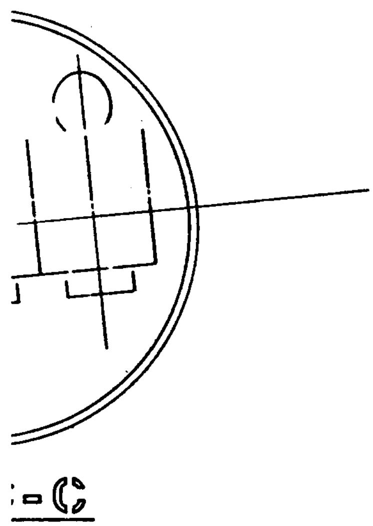


B - B

C

Figure 3. B



VOLUME $42.48\ M^3$ ($1,500\ FT^3$)
WEIGHT

PRELIMINARY CONCEPT
Not Iterated for Weight Scaling

iconic 20-Man ERM

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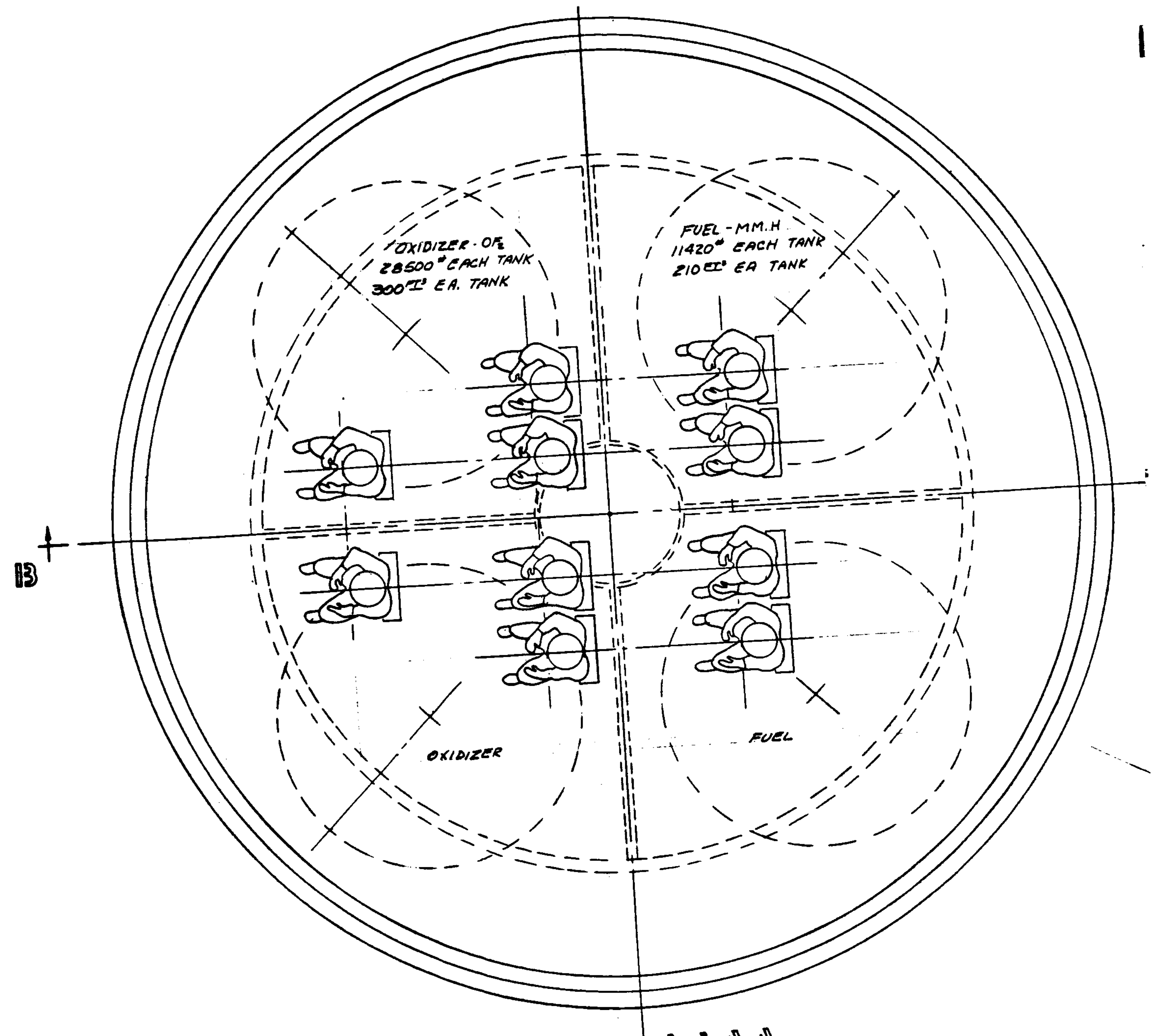

SECTION A-A-A-A

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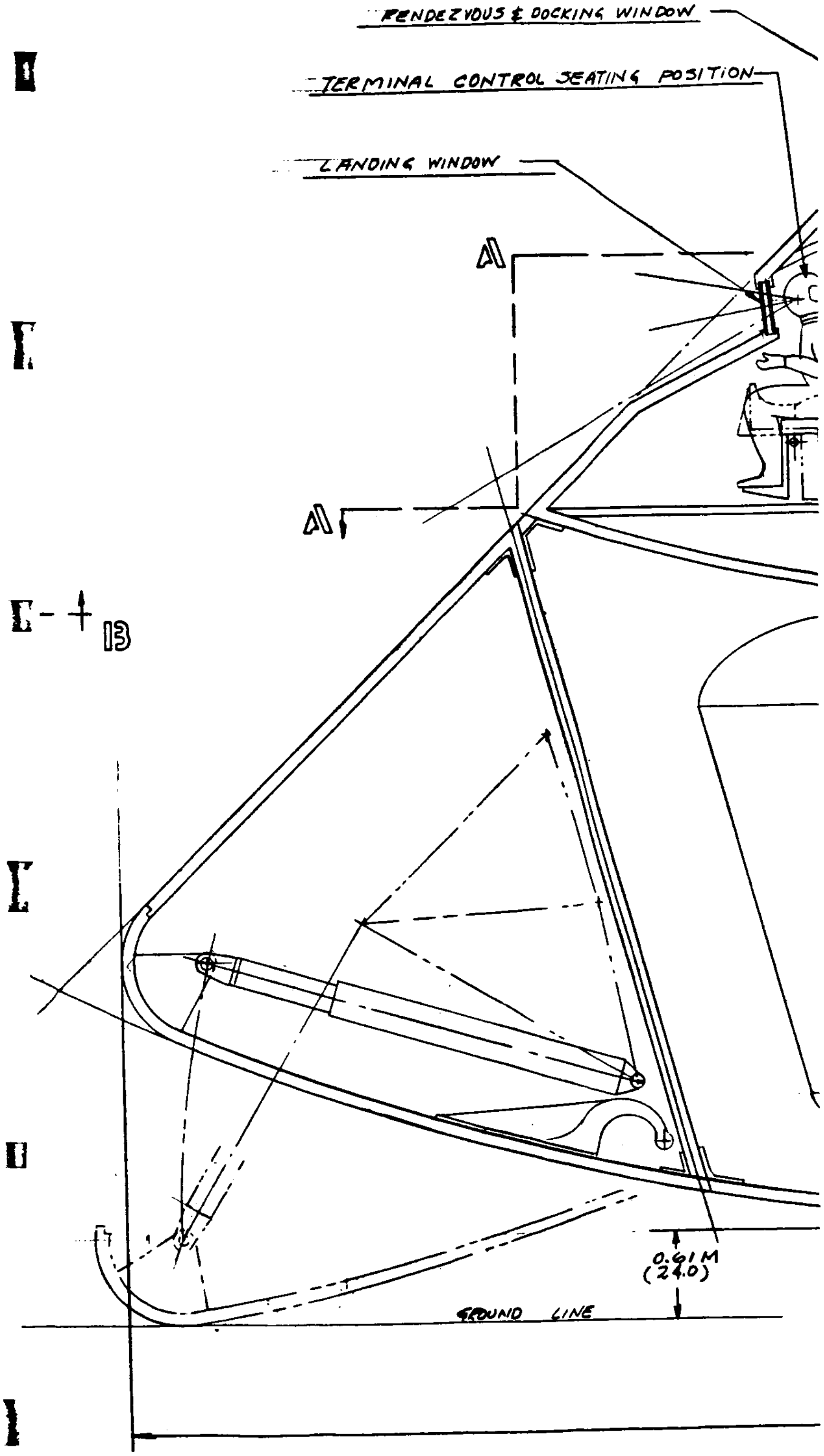
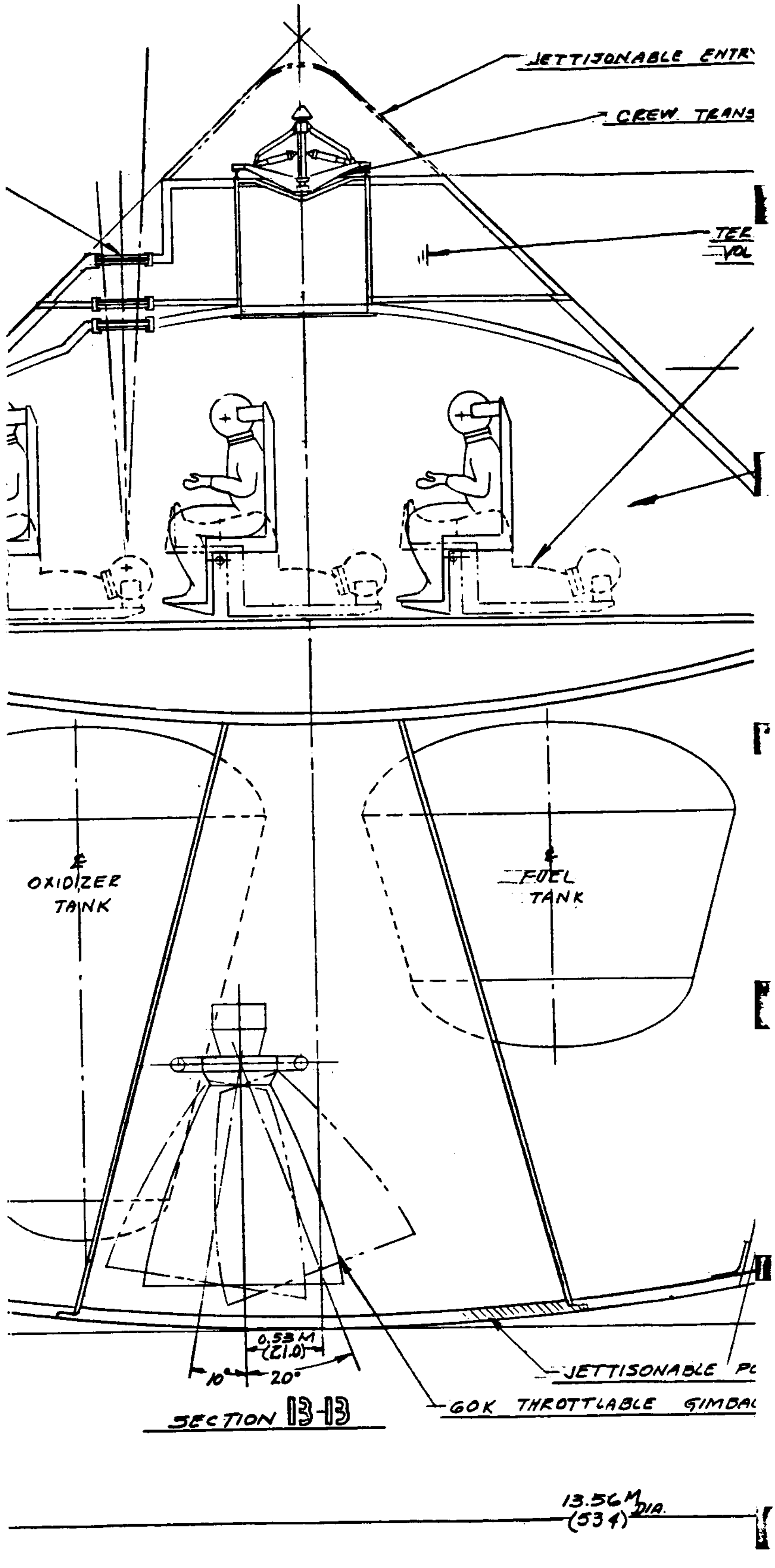


RENDEZVOUS & DOCKING WINDOW
TERMINAL CONTROL SEATING POSITION
LANDING WINDOW
A
A
B
0.61 M
(24.0)
GROUND LINE



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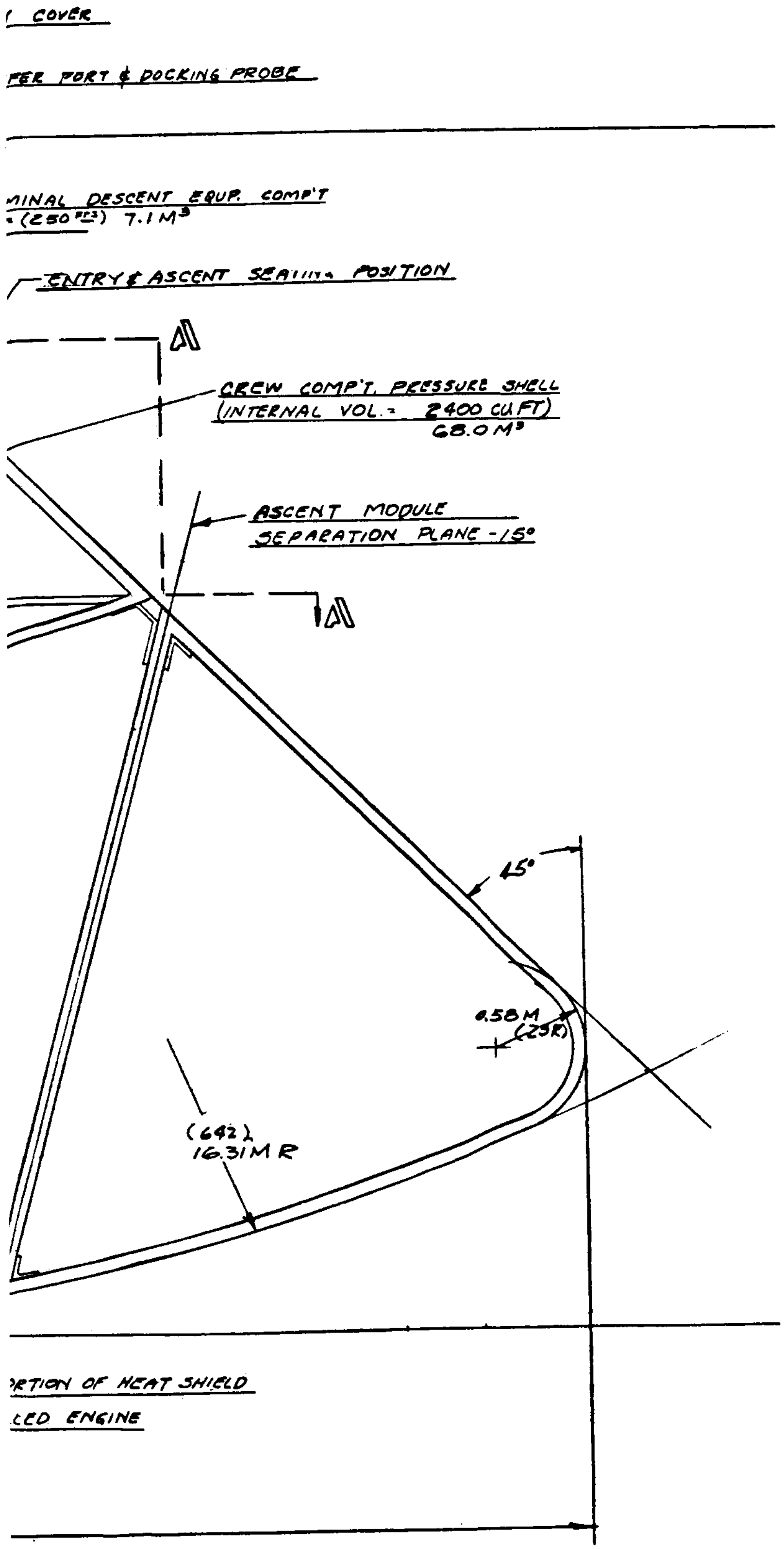
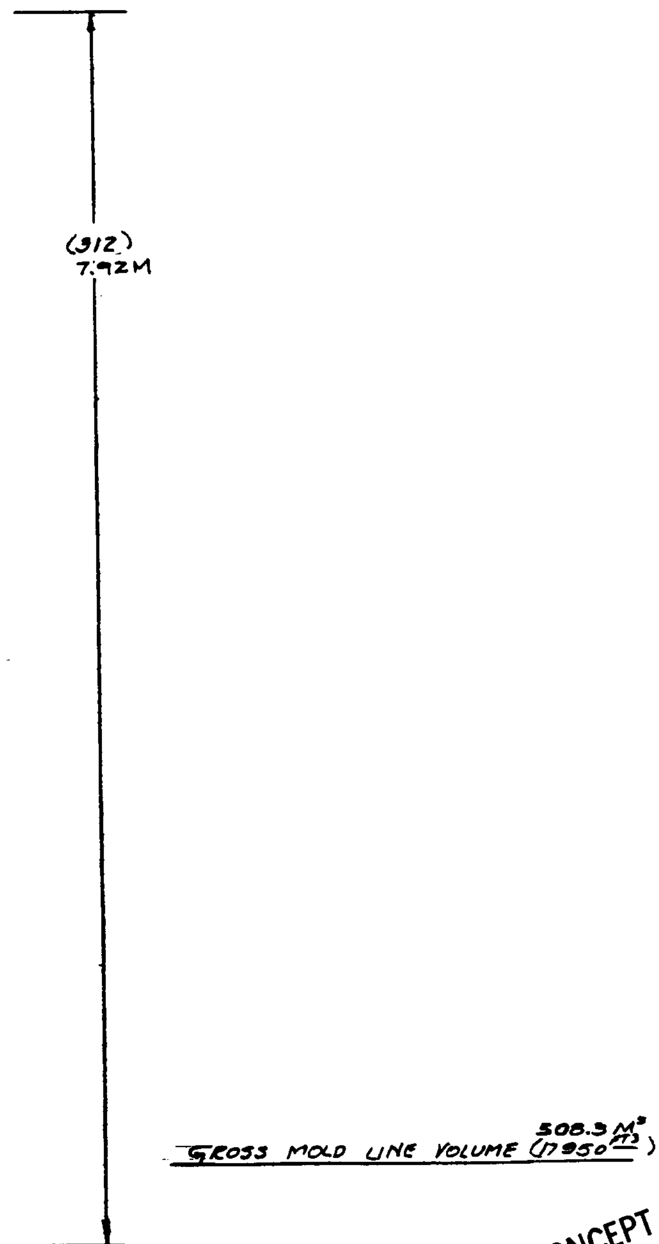


Figure 4. 10-M

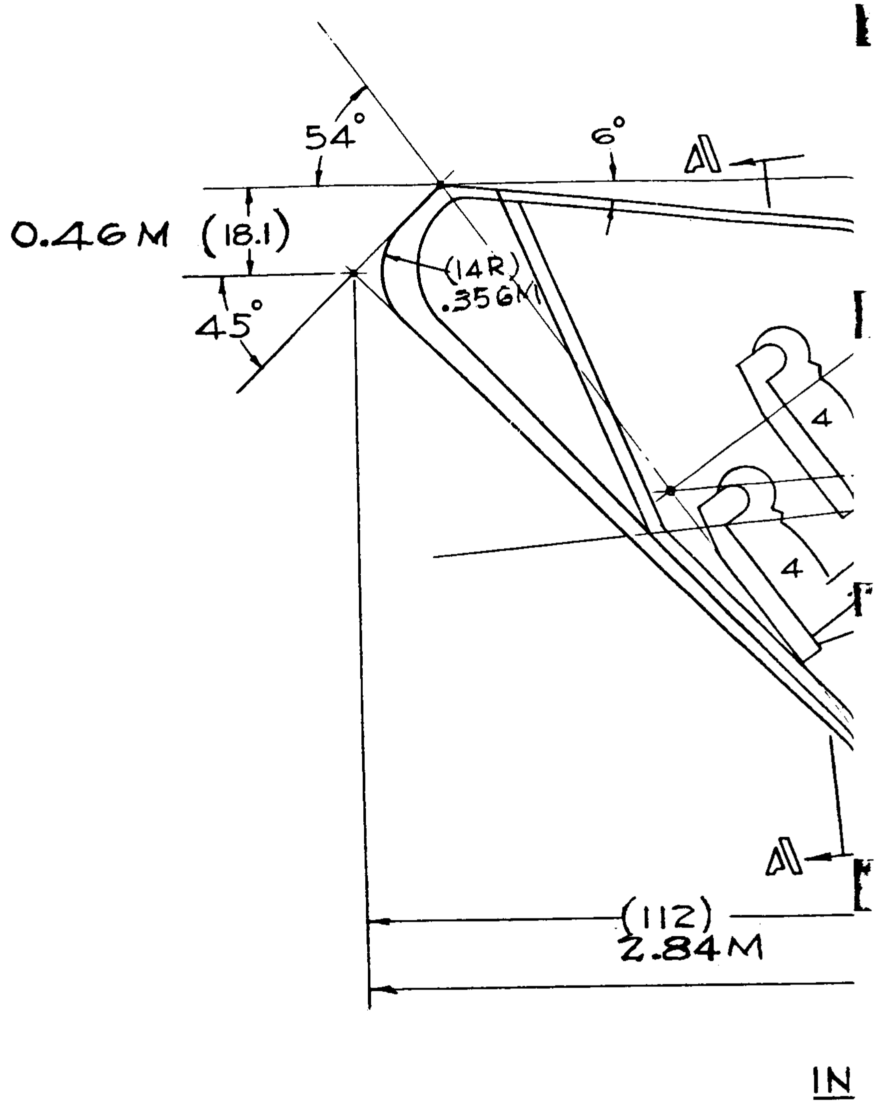
#3



PRELIMINARY CONCEPT
Not Iterated for Weight Scaling

Man Ballistic PEM

; 10 -



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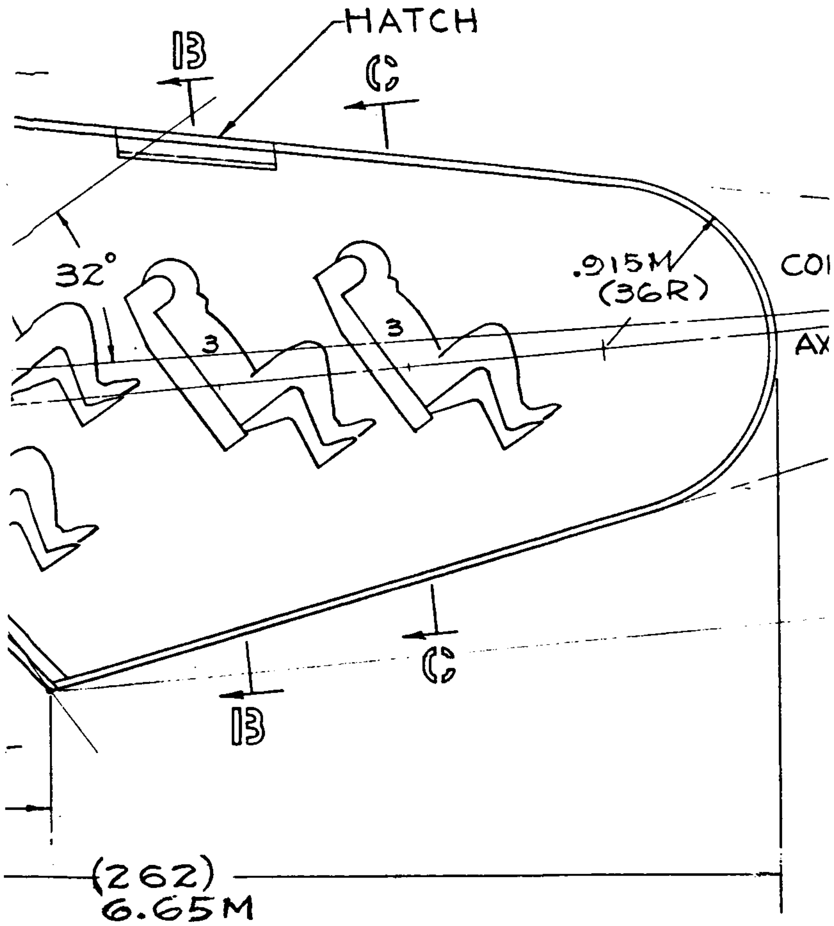
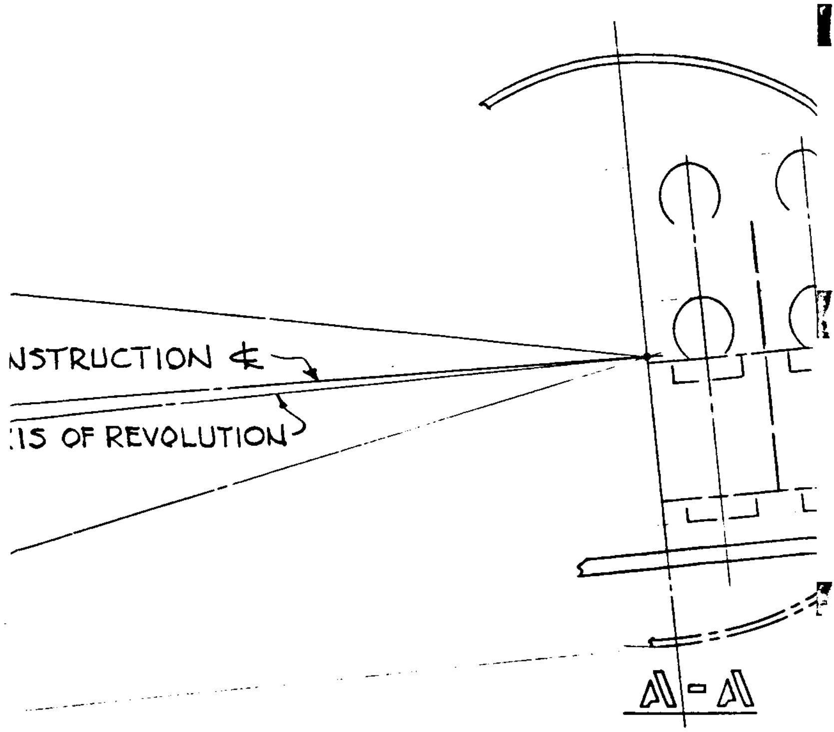


BOARD PROFILE
4-MEN ERM



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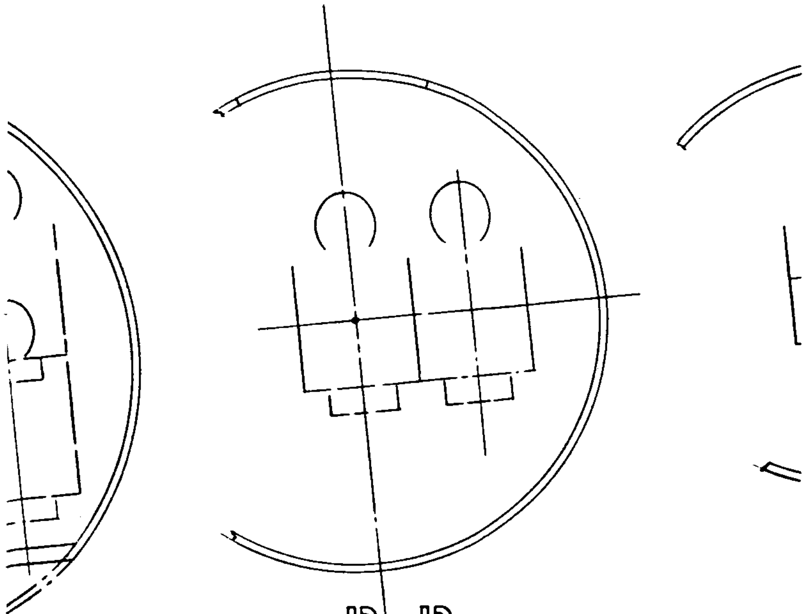
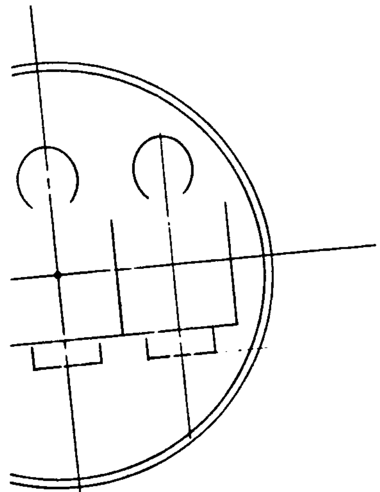


B - B

Figure 2.

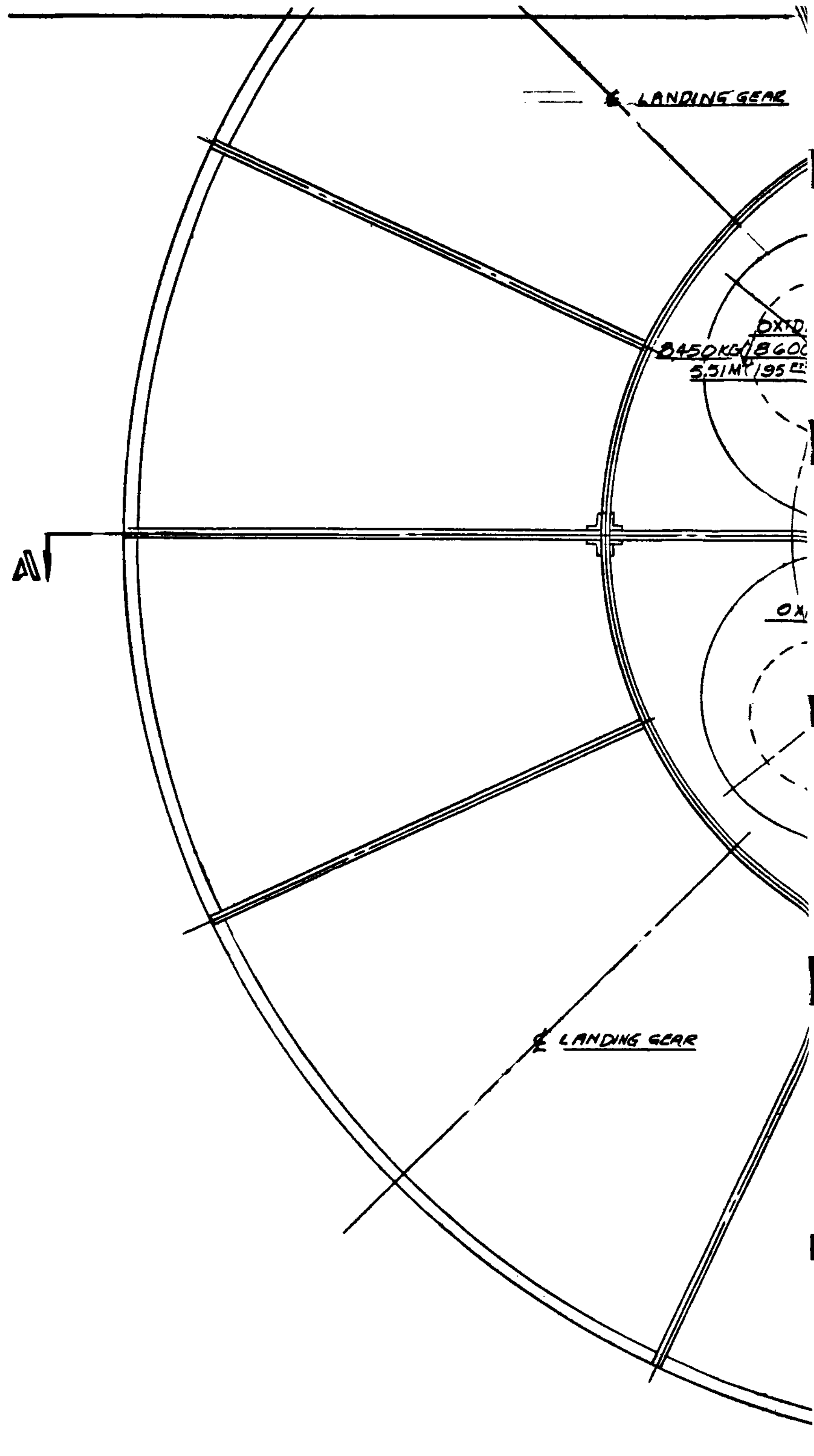


C-C

VOLUME 29.74M^3 (1,050 FT^3)
WEIGHT

PRELIMINARY CONCEPT
Not Iterated for Weight Scaling

Biconic 14-Man ERM



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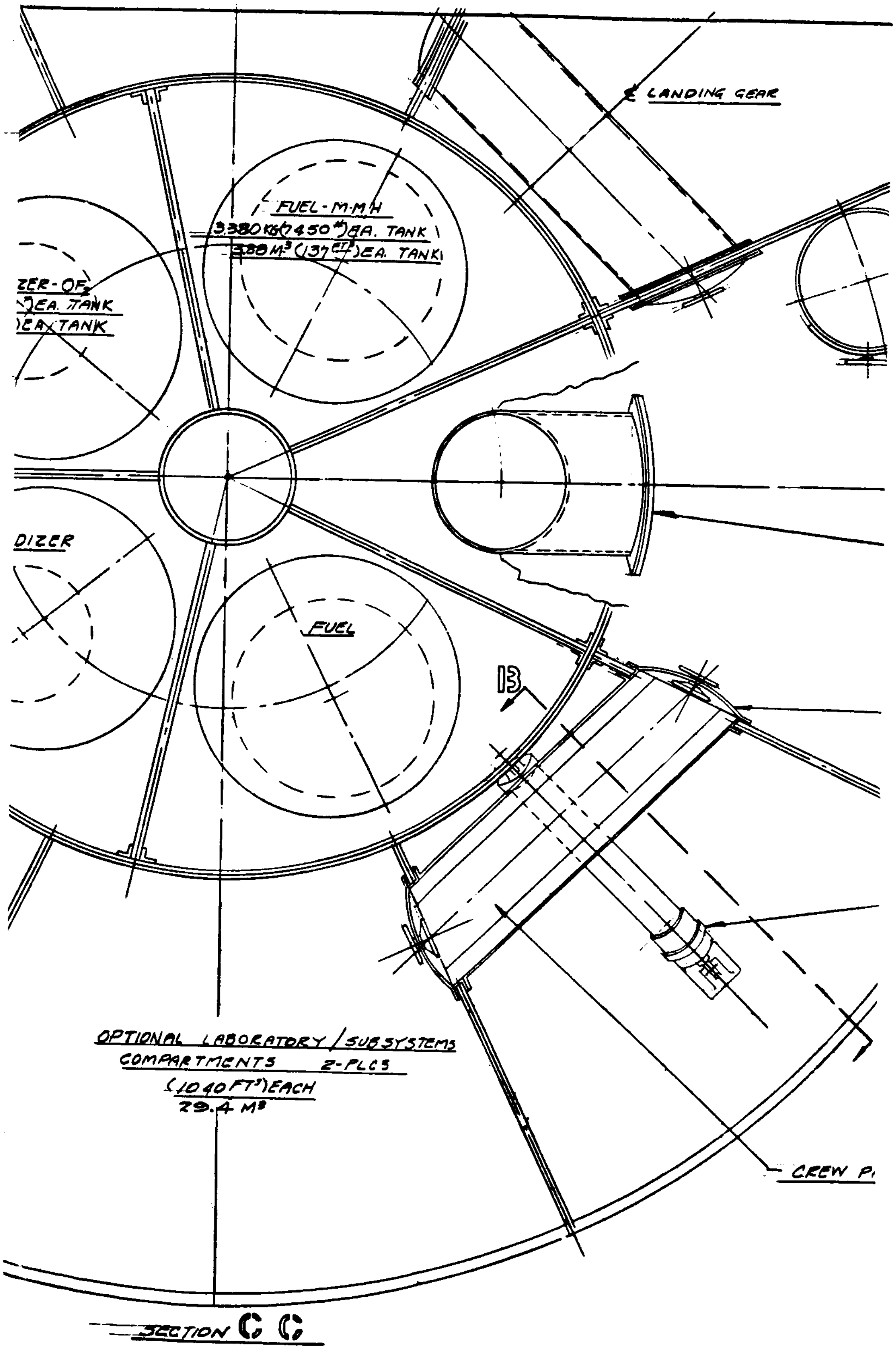

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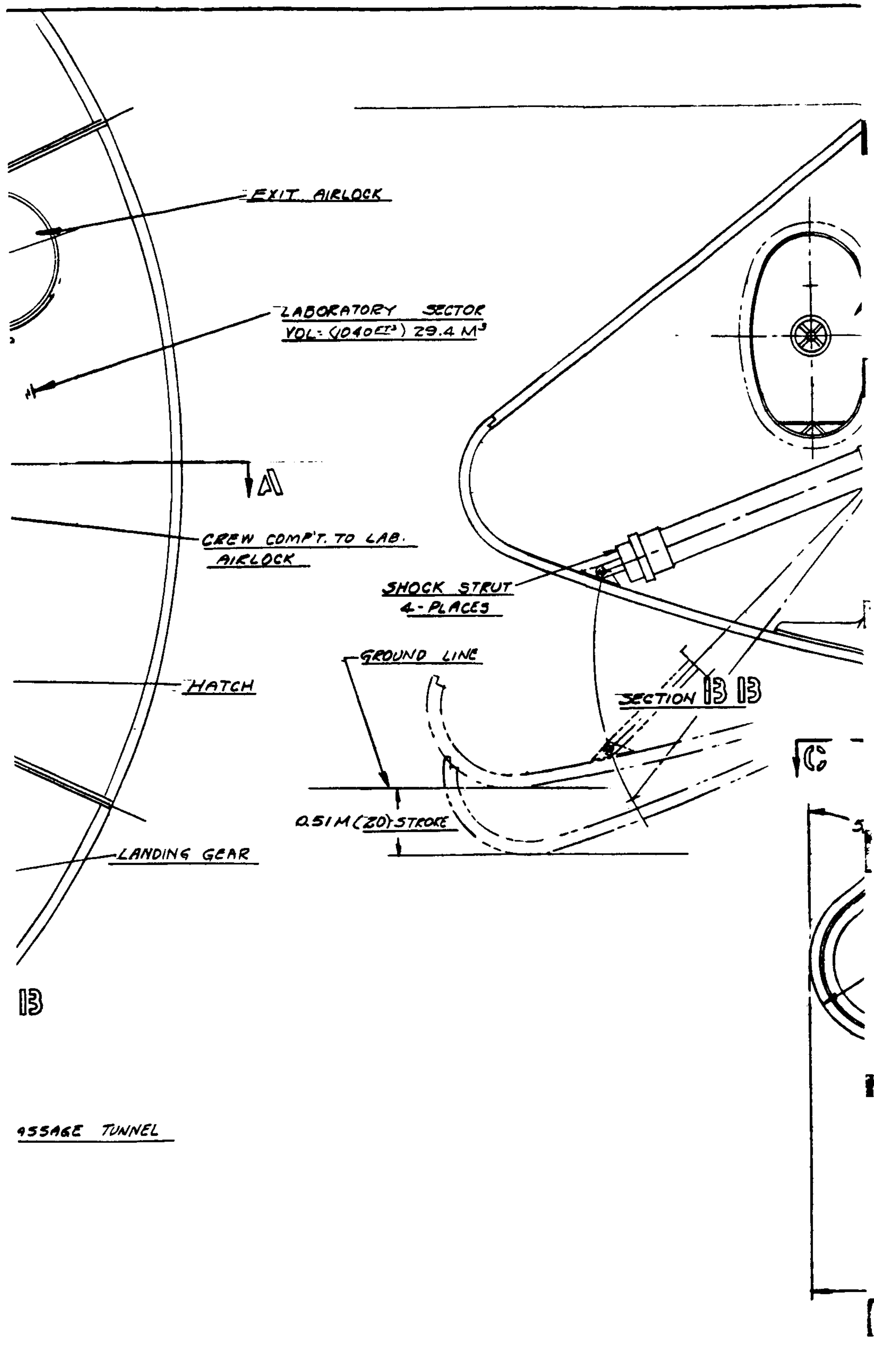

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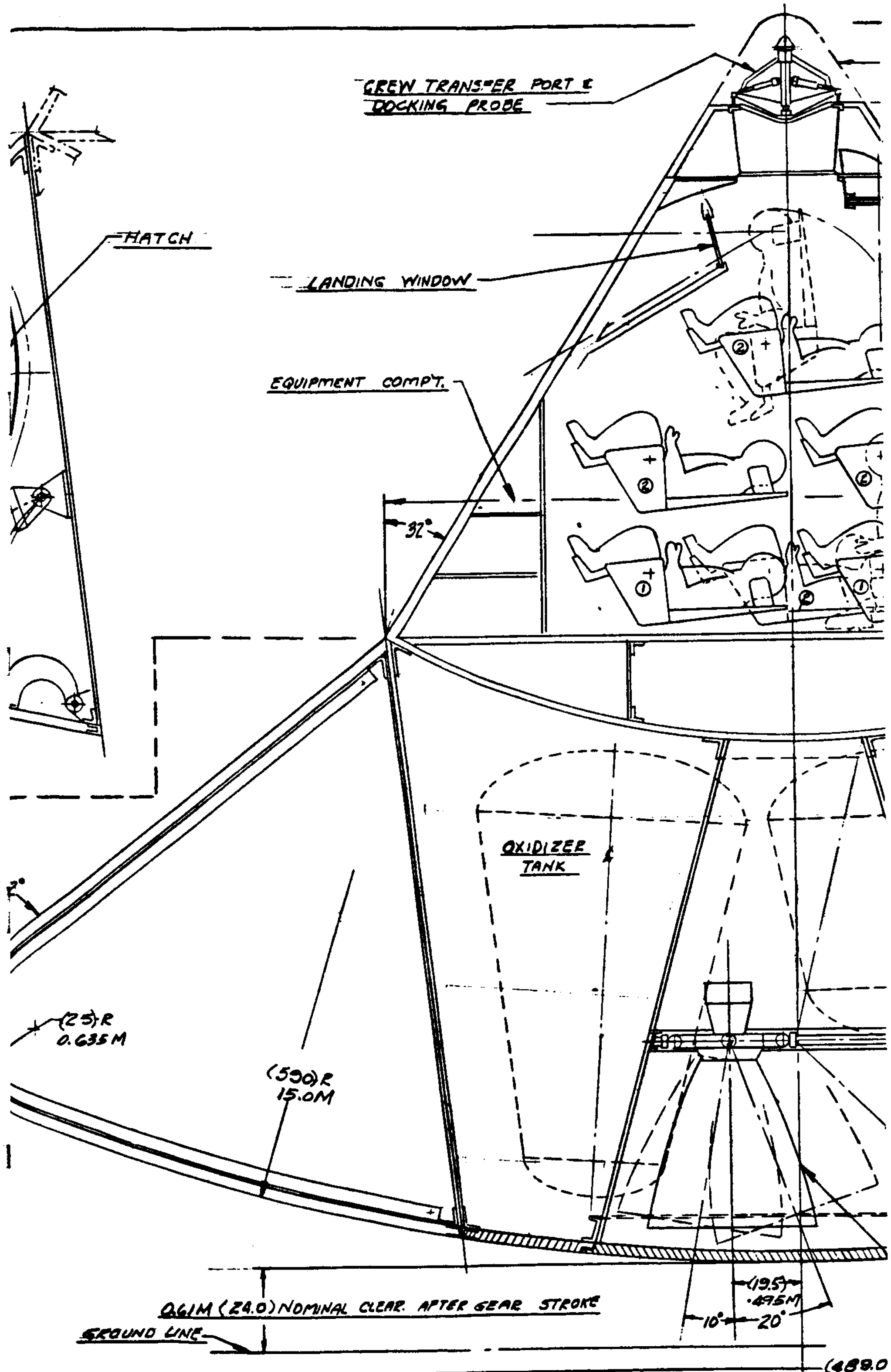
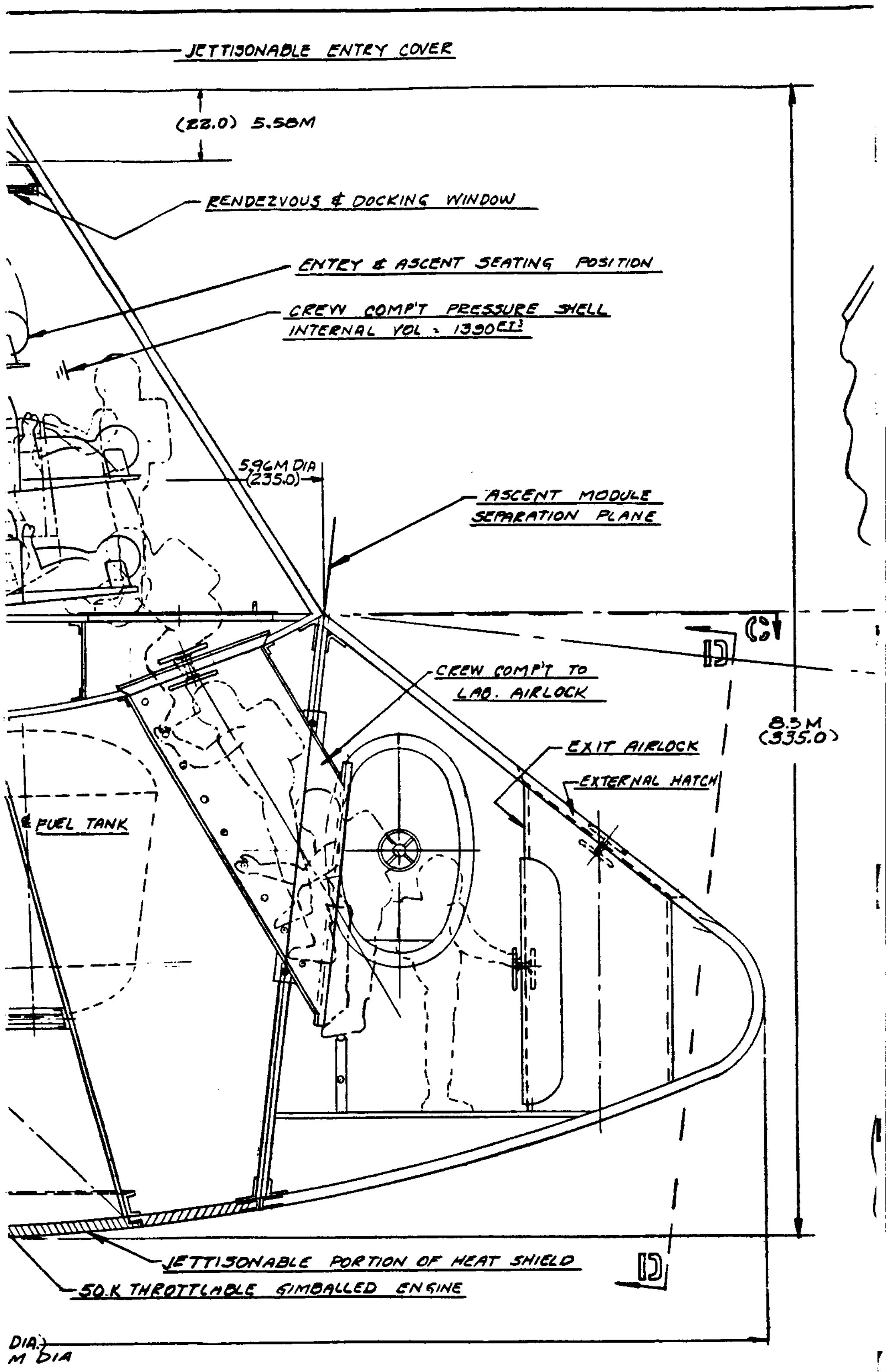

CREW TRANSFER PORT & DOCKING PROBE
HATCH
LANDING WINDOW
EQUIPMENT COMPT.
32°
OXIDIZER TANK
(25)R
0.635M
(590)R
15.0M
0.61M (24.0) NOMINAL CLEAR. AFTER GEAR STROKE
GROUND LINE
(19.5)
.495M
10°
20°

SECTION A A

#3



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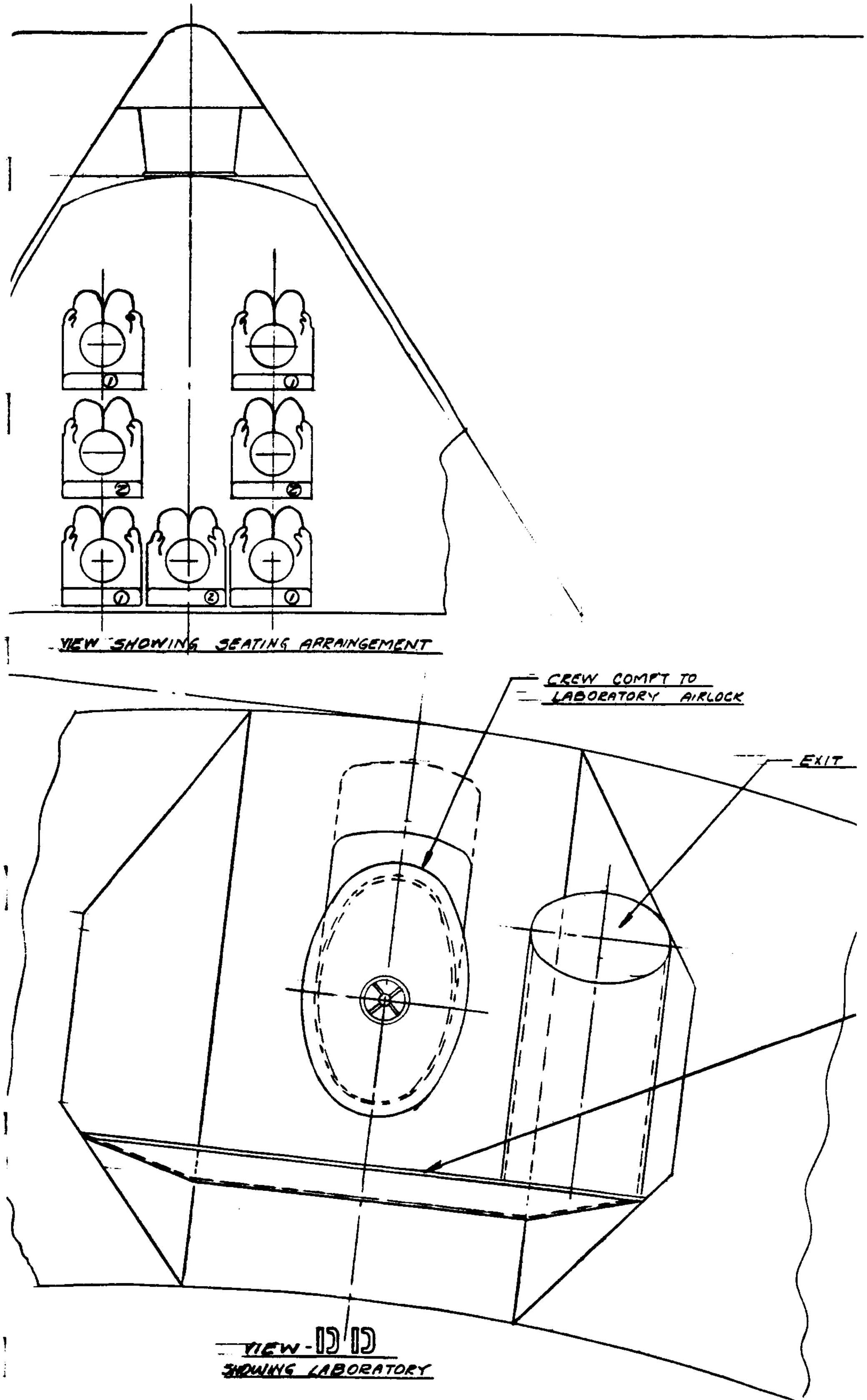


Figure 5. 10

FOLDOUT FRAME 4

AIRLOCK

— LABORATORY FLOOR

GROSS MOLD LINE VOLUME (12730 FT^3) 361.0 M^3
PROPELLANT WEIGHTS
FUEL (14900#) MMH 6750 KG
OXIDIZER (37200#) OF_2 16,900 KG

PRELIMINARY CONCEPT
Not Iterated for Weight Scaling

-Man Ballistic Mars Excursion Module, Biconic Aft Body is available in the one pressurizable compartment located between the aft pair of landing gear heat shield segments. Three other equivalent sized compartments are available, spaced between the sections of the heat shield which are deployed as landing gear pads. Access to two of these compartments, on an "if required" basis, has been provided by elliptical pressurized tunnels which extend from the primary section. These passageways have an internal height of 1.42 meters (56 inches), which would permit a crew member to walk in a slightly stooped attitude. The height of the tunnel is limited by the space available between the outside structure of the vehicle and the landing gear. An airlock has been provided to facilitate ingress and egress to the planet surface from the living quarters compartment. A weight iteration on this concept resulted in a decrease in estimated gross weight and the attendant base diameter as a result of the more efficient ascent staging brought about by minimizing the size of the ascent crew compartment. The modified aft body shape, consisting of two different conic angles, has been evaluated from aerodynamic considerations and is considered to be satisfactory, provided that drogue chutes are used to provide stability after entry and prior to ignition of the terminal retro and landing propulsion system. Because of the low density of the Martian atmosphere, parachutes were not considered for descent and landing.

Lifting Body Mars Excursion Module

Two lifting-body PEM concepts — a 10-man and a 16-man vehicle — were identified on the point design module matrix, (Figure 1), as being necessary. The basic body of these vehicles is a scaled half-conic shape known as D-9, which was developed by Aeronutronics under NASA contract in 1963. This vehicle has subsequently been used as a baseline lifting body configurations in all studies conducted by SD. Two different versions of the basic D9 shape exist, one which assumes a canted angle landing attitude, and the other which lands in a vertical tail-sitting attitude. This latter mode and configuration was used in the point designs that will be described in the following paragraphs. The original lifting body configuration was predicated upon a more dense Martian atmosphere than is currently being used as a model. Consequently, additional planform area must be provided to comply with the lower W/C_LA required to perform a lifting entry into the VM Martian atmosphere currently used as a design ground rule. In the two concepts prepared for this current study, the wing area is obtained by providing two surfaces that are hinged to fold out along the side of the body. This approach was used to provide wings that would have a simple deployment scheme and would have a minimum of effect on stowage and integration of the lifting body vehicle in the aerobraking spacecraft concepts subsequently developed for this study. The projected planform of the basic body of the vehicle and the desire to provide a swept leading edge for the deployed wings does not permit obtaining a total wing area equal to that of the basic body, but generally only about an increase of 55 percent, depending upon the specific vehicle. In all cases, the outer tip of the deployed wings is folded down to provide increased lateral stability and to improve the overall aerodynamic characteristics of the vehicle. During entry, vehicle control will be provided by movable aft body mounted surfaces and a reaction control system mounted on the aft surface.

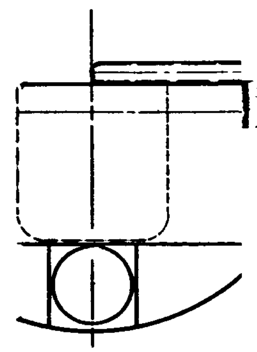
For purposes of the gross conceptual generation of 10- and 14-man lifting body vehicles, the staging approach and general internal arrangement of the vehicle is the same as the Aeronutronics final design, with appropriate changes to accommodate the increased propellant weights and larger mission quarters for the increase in numbers of crew. Each vehicle utilizes a four-legged landing gear, which is stowed at the aft end of the body prior to deployment.

10-Man Lifting Body MEM

The configuration of the 10-man vehicle is shown in Figure 6. This vehicle has an overall length of 12.3 meters (484 inches), a maximum body width of 10.81 meters (426 inches) and an estimated gross weight of 60,500 kilograms (133,400 pounds). The 10-man crew is accommodated in a 5.03 meter (198 inch) -long nose section of the vehicle in a high-density, pressurized compartment during entry, landing, and ascent. A two-man seating arrangement accommodates four in the first row and six in the aft row. The crew compartment section contains all of the flight control systems, displays, and life support systems required for descent and ascent. In addition, it contains the ascent reaction control system and a docking interface and associated systems for docking with the orbiting spacecraft subsequent to ascent from the surface. Access to the mission living quarters is provided by an interconnecting tunnel extending from the pressure bulkhead of the crew compartment to the mission quarters located in the aft end of the vehicle.

Propellants used for the vehicle are OF_2/MMH with 10,600 kilograms (23,400 pounds) provided for descent and 15,500 kilograms (34,020 pounds) for stage 1 ascent and 9,200 kilograms (20,250 pounds) for stage 2 ascent. The propellant tanks for stage 1 ascent are jettisoned after propellant expenditure during ascent. Separate throttleable plug nozzle rocket motors (which are desirable on the basis of volumetric considerations) are used for descent and ascent.

The maximum wing area that is available by folding the two wing sections across the upper surface of the body is 48.3 M^2 (520 ft^2). The resultant total planform area, including that of the body is (1438 square feet) which provides a W/C_LA of 730 kg/M^2 (150 lb/ft^2).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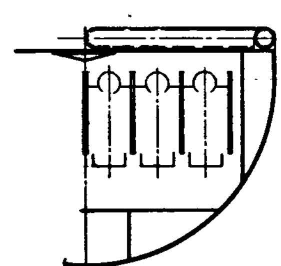
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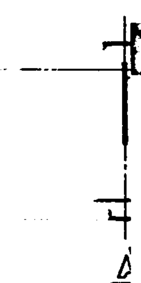

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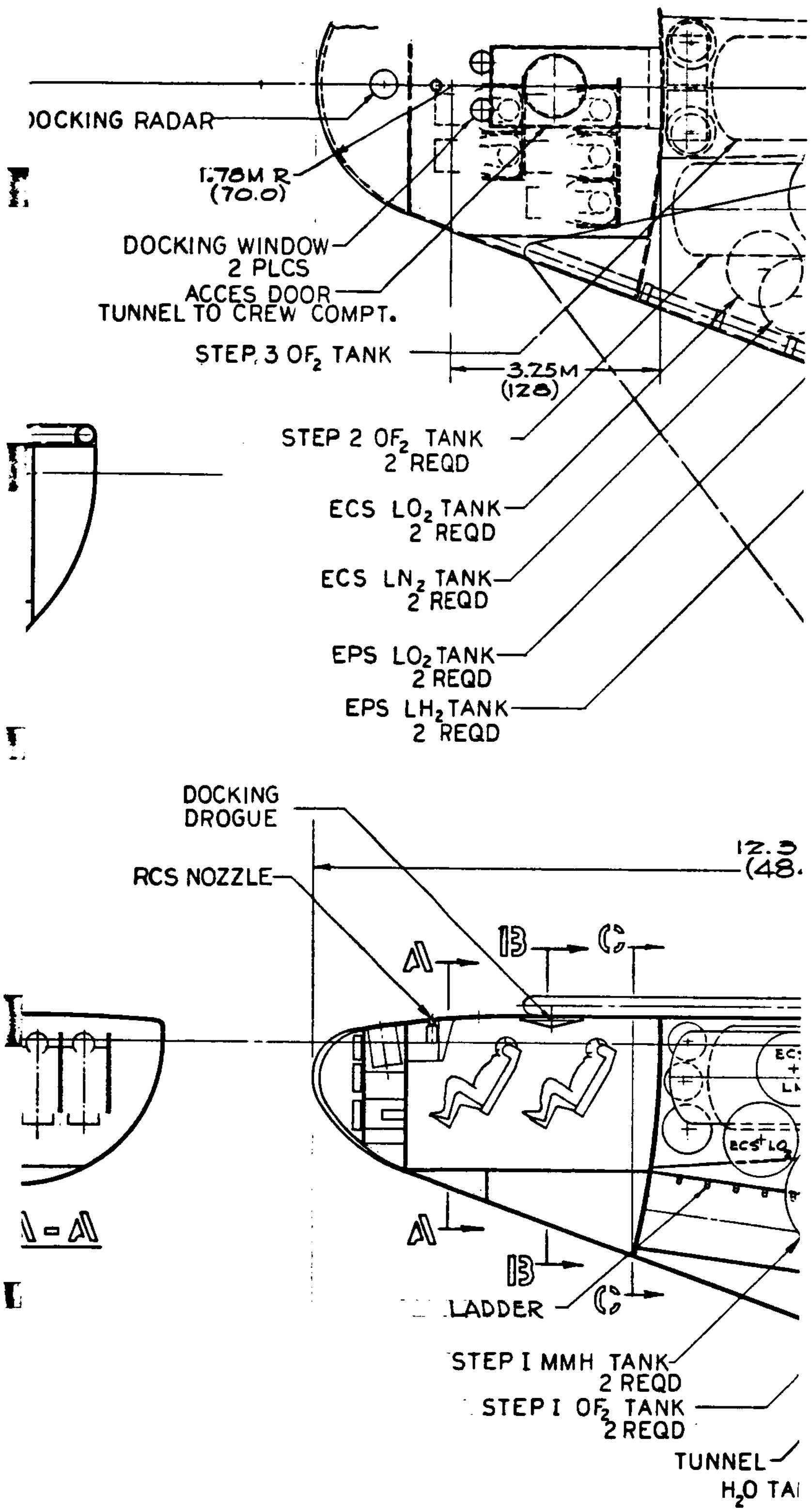
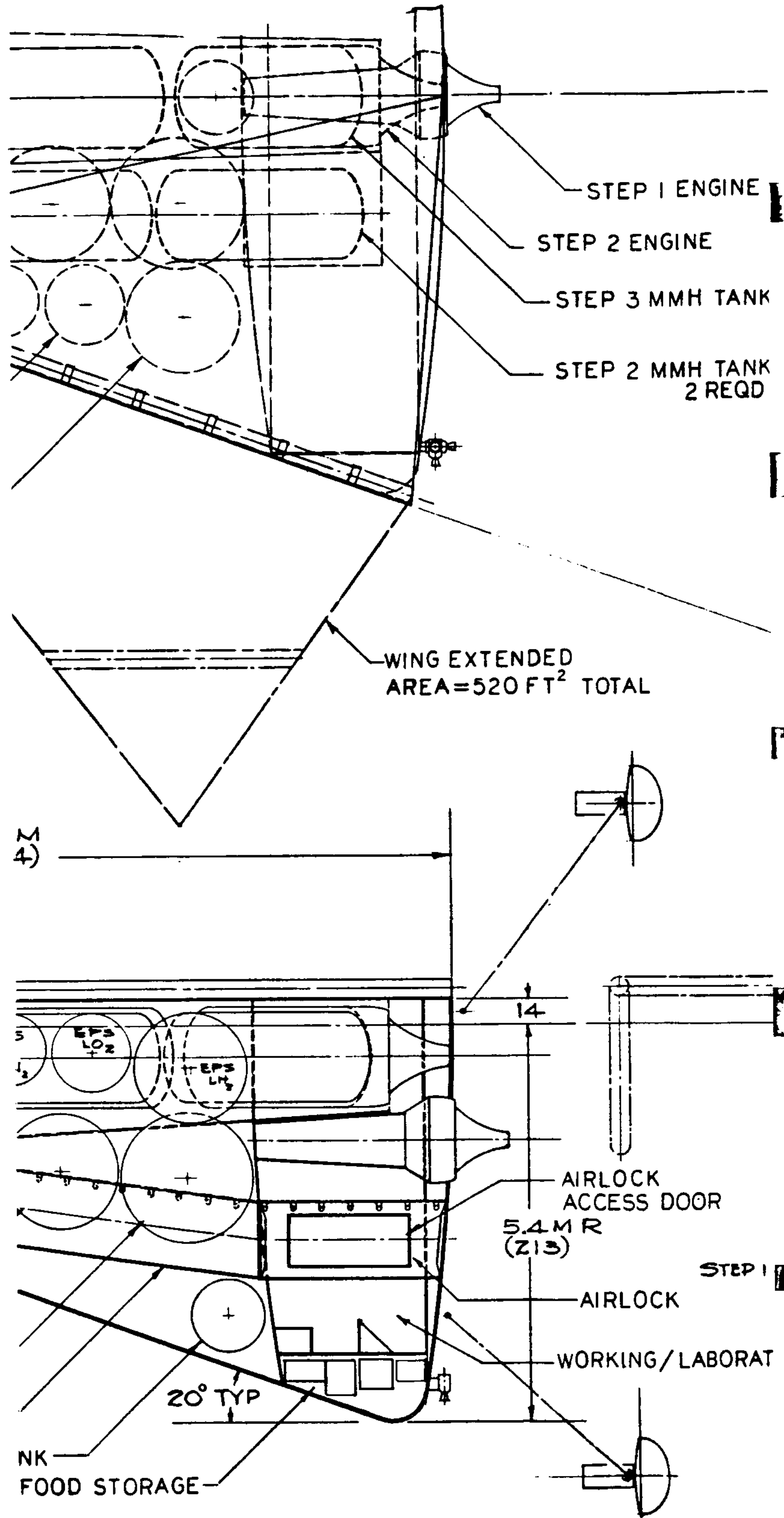

DOCKING RADAR
1.78M R
(70.0)
DOCKING WINDOW
2 PLCS
ACCES DOOR
TUNNEL TO CREW COMPT.
STEP 3 OF2 TANK
3.25M
(128)
STEP 2 OF2 TANK
2 REQD
ECS LO2 TANK
2 REQD
ECS LN2 TANK
2 REQD
EPS LO2 TANK
2 REQD
EPS LH2 TANK
2 REQD
DOCKING
DROGUE
RCS NOZZLE
12.3
(48.
A
B
C
A-A
LADDER
STEP I MMH TANK
2 REQD
STEP I OF2 TANK
2 REQD
TUNNEL
H2O TA



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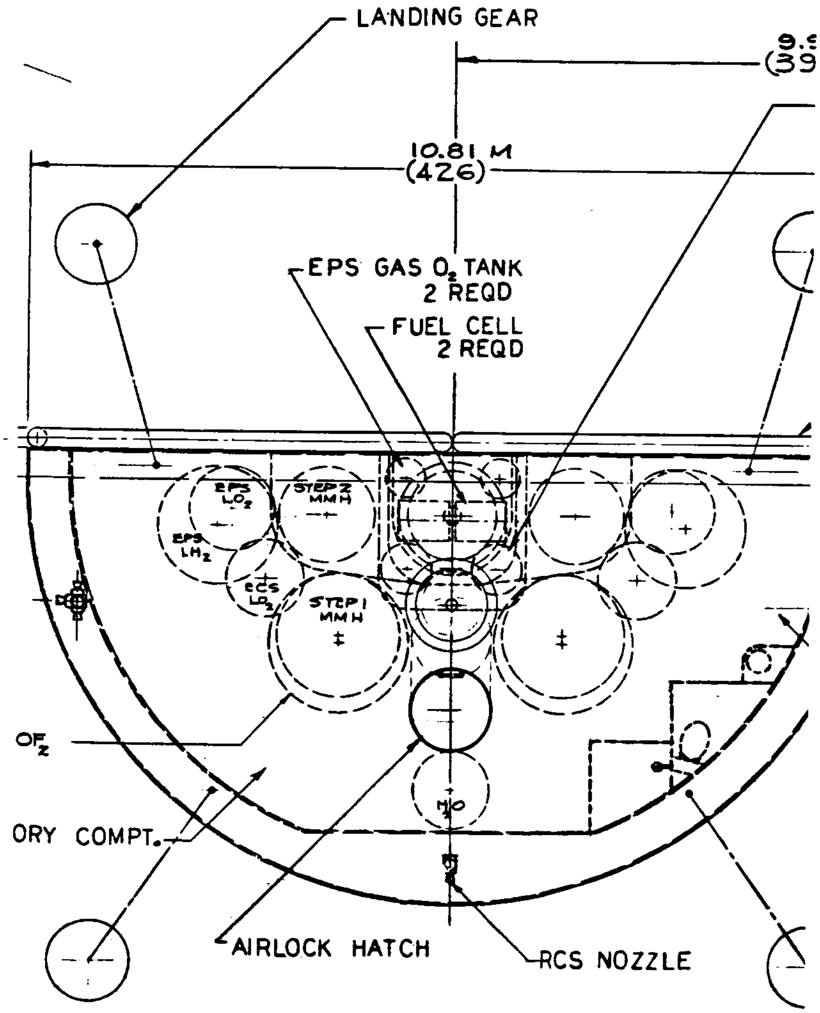
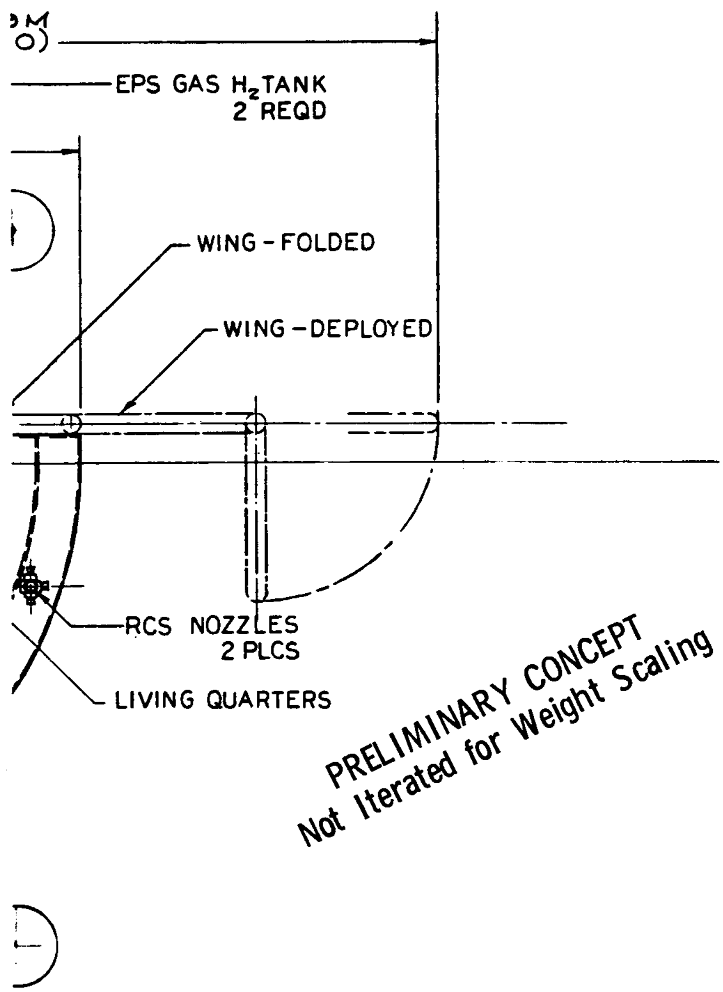


Figure 6. 10-Man Li

ROSS WT	(133,400 LB)	60 500 KG
ROPELLANT WT (OF$_2$/MMH) MR= 2.5 : 1.0		
STEP 1 (DESCENT)	(23,400 LB)	10,600 KG
STEP 2 (1ST ASCENT)	(34,020 LB)	15,430 KG
STEP 3 (2ND ASCENT)	(20,250 LB)	9,200 KG
ROJECTED AREAS		
BODY	(918 FT^2)	85.3 M^2
WING	(520 FT^2)	48.3 M^2
TOTAL	(1,438 FT^2)	133.6 M^2



ifting Body Mars Excursion Module

16-Man Lifting Body MEM

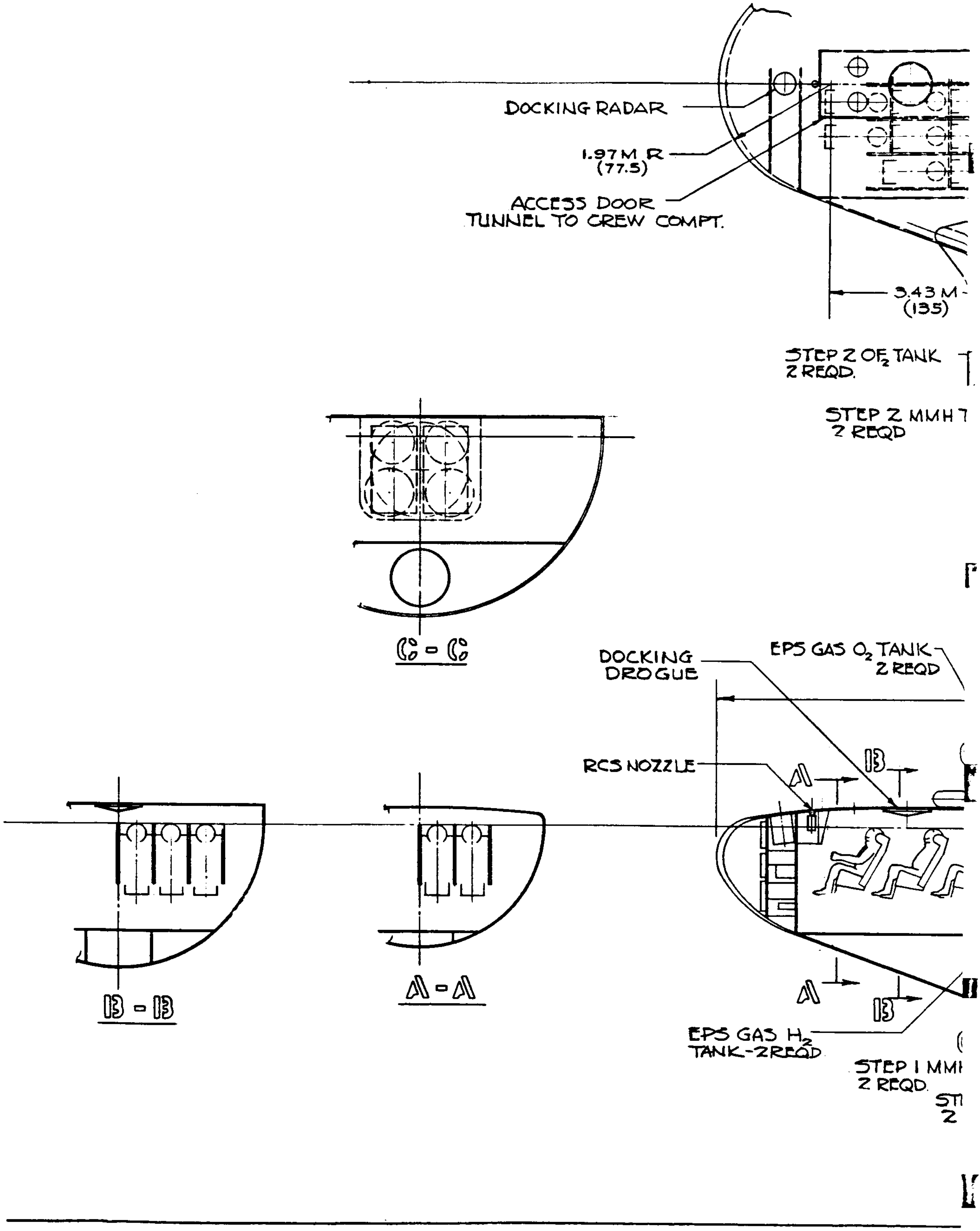
The configuration for this vehicle is shown in Figure 7. The general arrangement and concept design philosophy for the 16-man vehicle is the same as previously discussed for the 10-man vehicle. This vehicle has an overall length of 14.8 meters (584 inches), a body width of 12.8 meters (504 inches), and a gross weight of 84,700 kilograms (186,700 pounds). The 16-man crew is accommodated during descent and ascent in a pressurized nose compartment 3.25 meters (128 inches) long, in a three-row seating arrangement, with two-men abreast in the first row, and six men each in the aft two rows. The propellants are OF_2/MMH with 14,850 kilograms (32,720 pounds) provided for descent, (47,750 pounds) in stage 1 ascent and 12,850 kilograms (28,350 pounds) in stage 2 ascent. The available area for the folded wings results in a deployed wing area of 66.0 M^2 (710 ft^2). The total resultant planform area at entry is 185.1 M^2 (1992 ft^2) with a W/C_LA of (150 lb/ft^2).

The ground rule noted above regarding use of propulsive landing for all PEM's also affects the lifting body PEM configurations. The wings shown on the two configurations of Figures 6 and 7 will not be used. As in the case of the Apollo-shaped PEM's, it was not necessary to modify the drawings; weights were adjusted to compensate for the change in design.

Retrobraking Planetary Excursion Modules

This class of planetary excursion modules, which has been identified for conceptual developement on the module summary matrix (Figure 1), includes 3- and 10-man vehicles for nonatmospheric planetary bodies. The specific planetary bodies for which the vehicles were configured are the asteroids Ceres and Vesta, Jupiter's moon Ganymede, and Mercury. The basic design philosophy used in the conceptual generation of these vehicles reflects that philosophy currently utilized in the Lunar Module, which is part of the Apollo program. Since these vehicles are operating on nonatmospheric bodies*, aerodynamic considerations do not exercise any influence on the general arrangement. All of the vehicles incorporate a high-density crew compartment used for descent and ascent, with separate propulsion and propellant systems used for each. At this point, no attempt has been made to force commonality on the designs, such as using a single stage on one vehicle and subsequently using it as the ascent stage for another vehicle. All of the vehicles utilize FLOX/MMH propellants, and have been configured for a nominal 30-day mission stay time.

*Mercury is assumed to have no significant atmosphere for purposes of this study.



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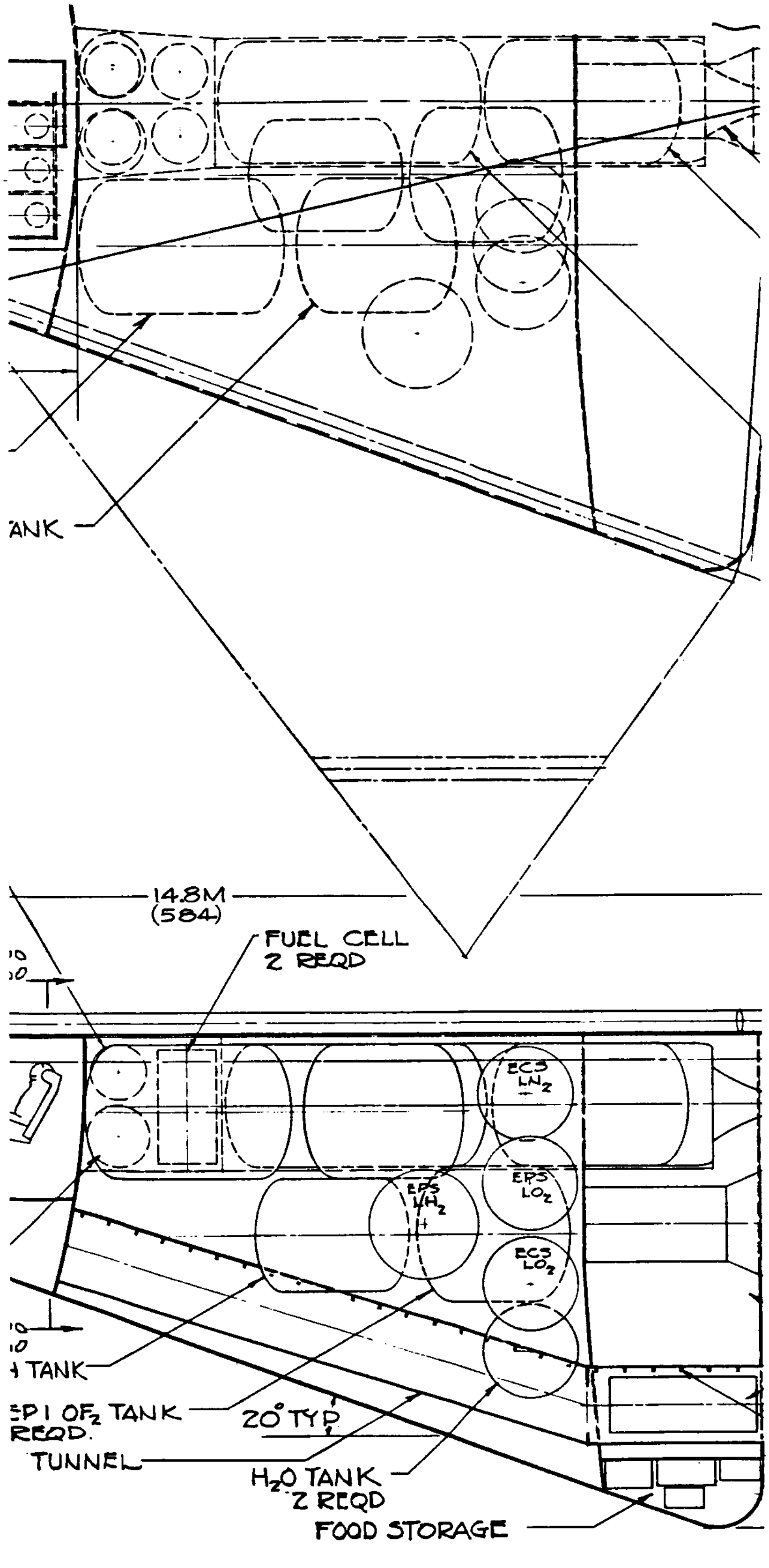

ANK
14.8M
(584)
FUEL CELL
2 REQD
ECS
LN_2
EPS
LO_2
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ECS
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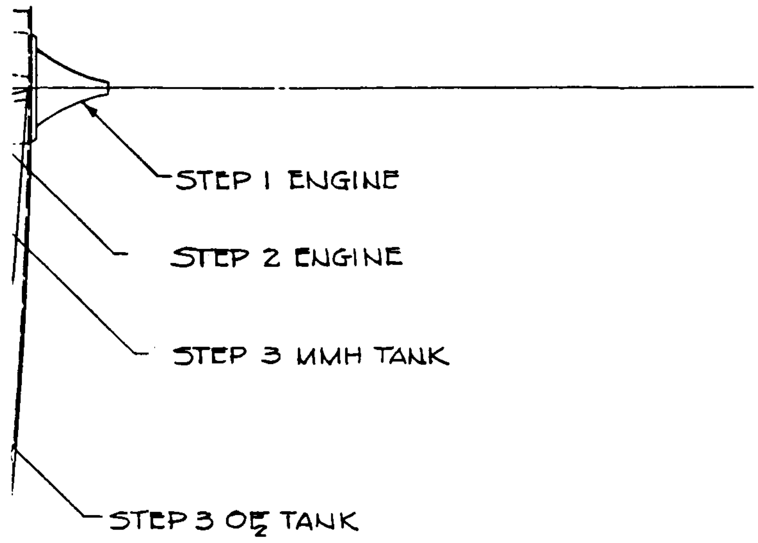
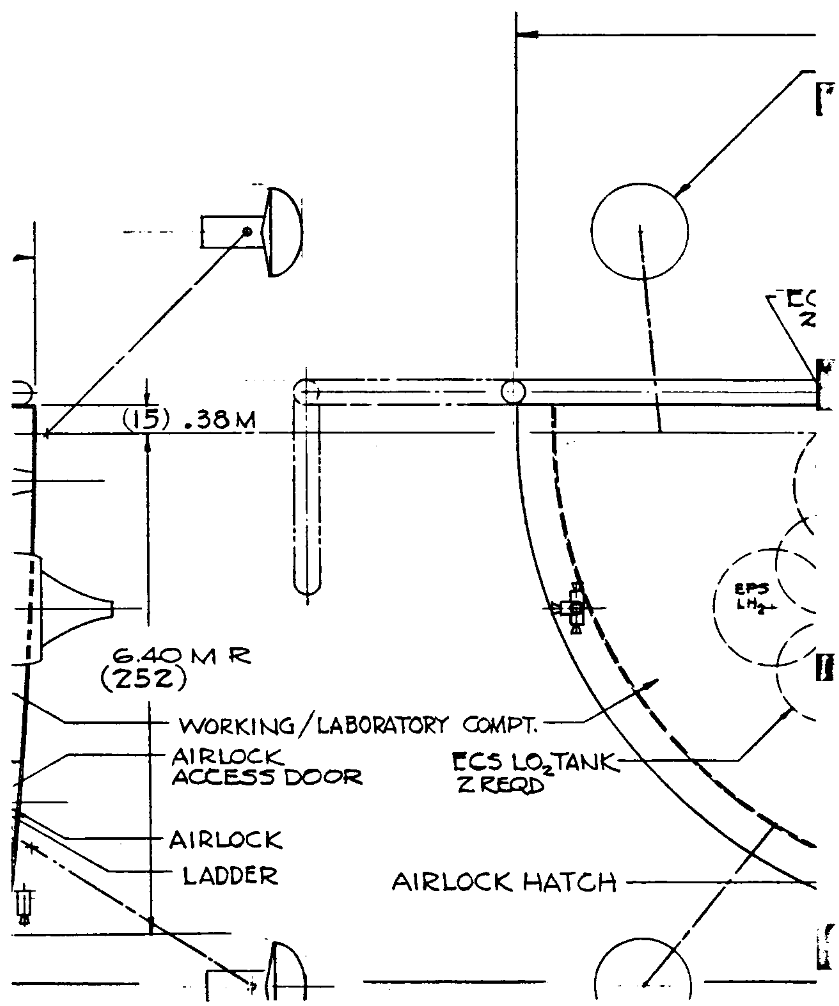
20° TYP
TUNNEL
H_2O TANK
2 REQD
FOOD STORAGE





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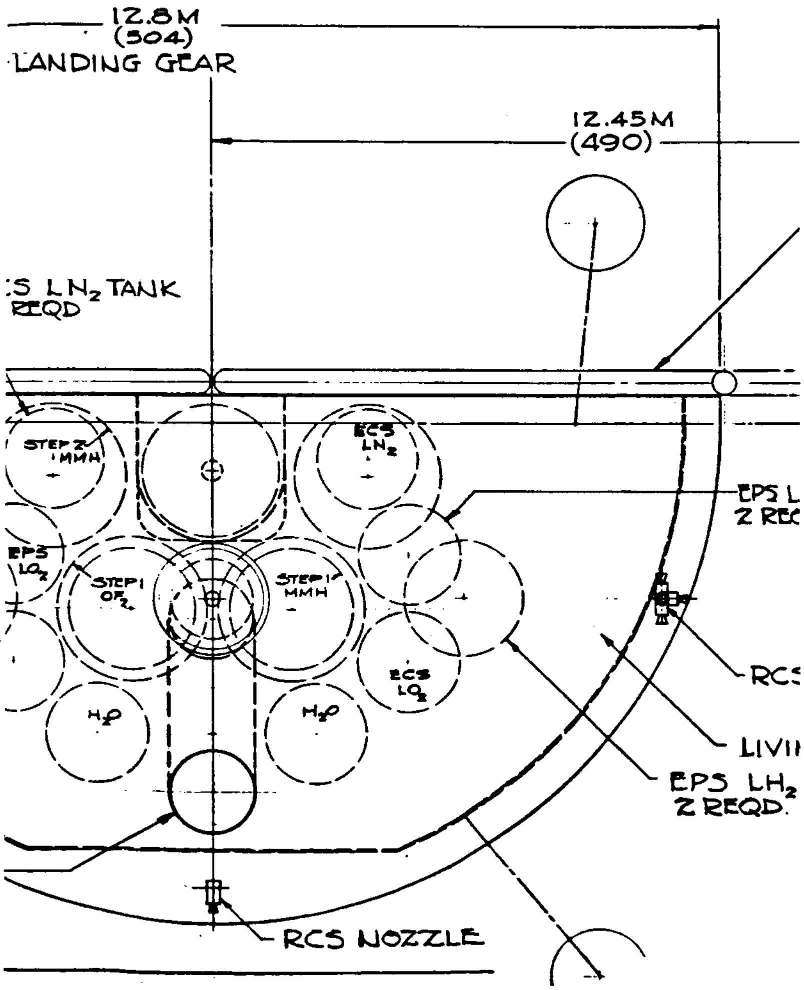
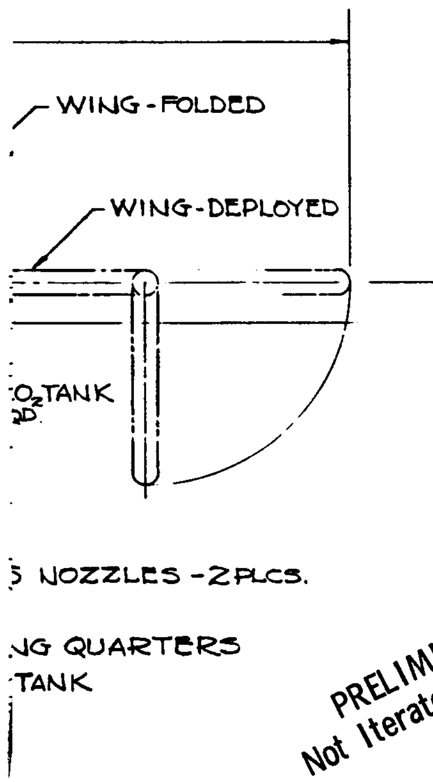


Figure 7. 16-Man Lifti

3SS WT.	84,700 KG	(186,700 LB)
PELLANT WT. (OF_2/MMH) MR = 2.5:1.0		
TEP 1 (DESCENT)	14,850 KG	(32,720 LB)
TEP 2 (1ST ASCENT)	21,700 KG	(47,750 LB)
TEP 3 (2ND ASCENT)	12,850 KG	(28,350 LB)
JECTED AREAS		
ODY	119.1 M^2	(1,282 FT^2)
VING	66.0 M^2	(710 FT^2)
OTAL	185.1 M^2	(1,992 FT^2)



PRELIMINARY CONCEPT
Not Iterated for Weight Scaling

.ng Body Mars Excursion Module

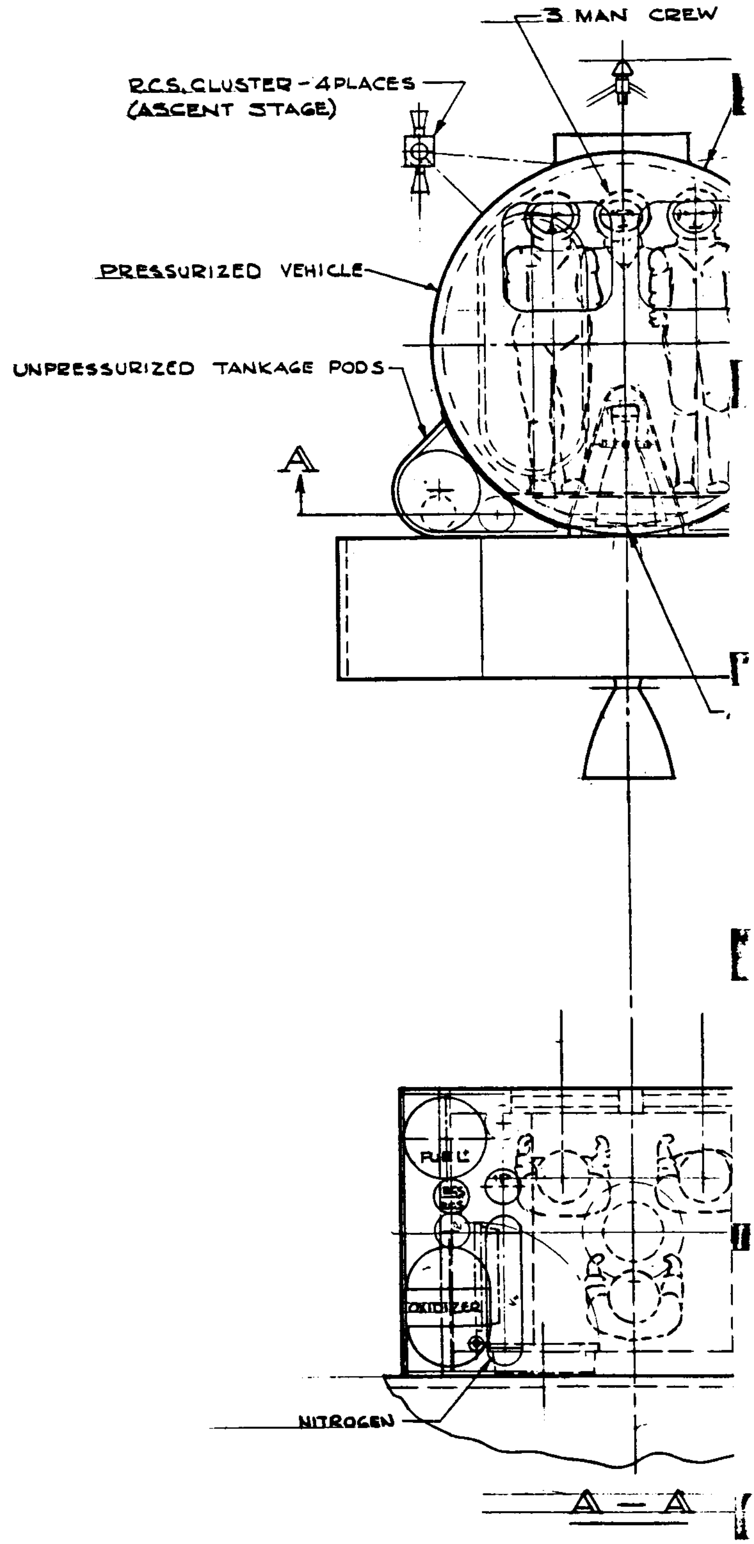
3-Man Retro PEM (Ceres/Vesta)

A 3-man retro PEM for Ceres and Vesta is shown in Figure 8. This vehicle has an estimated gross weight of 14,000 kilograms (31,000 pounds). The general arrangement of the vehicle features a hexagonal-shaped descent stage which incorporates the descent propellant tanks, propulsion system, mission life support consumables, descent reaction control system (RCS) and a four-legged landing gear system. Mounted on the upper surface of the landing stage are the ascent vehicle and a cylindrical-shaped pressurized structure that serves as the crew mission quarters.

The descent stage provides for 2,350 kilograms (5200 pounds) of FLOX/MMH propellants contained in six spherical tanks with the other mission consumables and RCS propellants contained in spherical and cylindrical tanks integrated into suitable spaces in the landing stage. The four-legged landing gear has been designed to fold in under the body to reduce the overall space requirements when the vehicle is integrated into an overall spacecraft. A throttleable throat-gimbaled, pressure-fed engine of 835-kilogram (1800-pound) thrust is used for descent and terminal landing.

The ascent stage is a complete autonomous stage that accommodates the 3-man crew in a standing position during descent and ascent. The pressurized compartment is essentially a cylindrical section 2.44 meters (96 inches) in diameter and 1.83 meters (72 inches) long with a gross volume of 6.3 m^3 (222 ft^3). Ascent propellants, weighing 406 kilograms (900 pounds), are supported off of the basic cylindrical structure in spherical and cylindrical tanks. A single pressure-fed engine of 384-kilogram (850-pound) thrust protrudes through the floor of the crew compartment and is sealed from the compartment by a tapered cylindrical can. A docking interface (and appropriate systems)is integrated into the upper surface of the crew compartment to permit docking with the parent spacecraft after ascent. A separate ascent reaction control system is provided for the ascent stage.

A cylindrical pressurized compartment 2.44 meters (96 inches) in diameter and 3.96 meters (156 inches) long is located adjacent to the aft face of the crew compartement to serve as the mission quarters. Access to this compartment is through a pressurized door integrated into the aft bulkhead of the crew compartment and through a short interconnecting tunnel between the two compartments. Prior to ascent, the tunnel structure would be severed at the crew compartment aft face. The mission quarters has a gross volume of 14.9 m^3 (532 ft^3), which would be supplemented by the 6.3 m^3 (222 ft^3) crew compartment to provide approximately 21.2 m^3 (750 ft^3) gross. The crew compartment could be used as a control center and possibly as sleeping quarters during the stay time. Access to the target body surface is provided through a cylindrical walk-in airlock integrated in the aft side of the mission quarters. The lower end of the airlock would be opened and descent made by a ladder to the surface.



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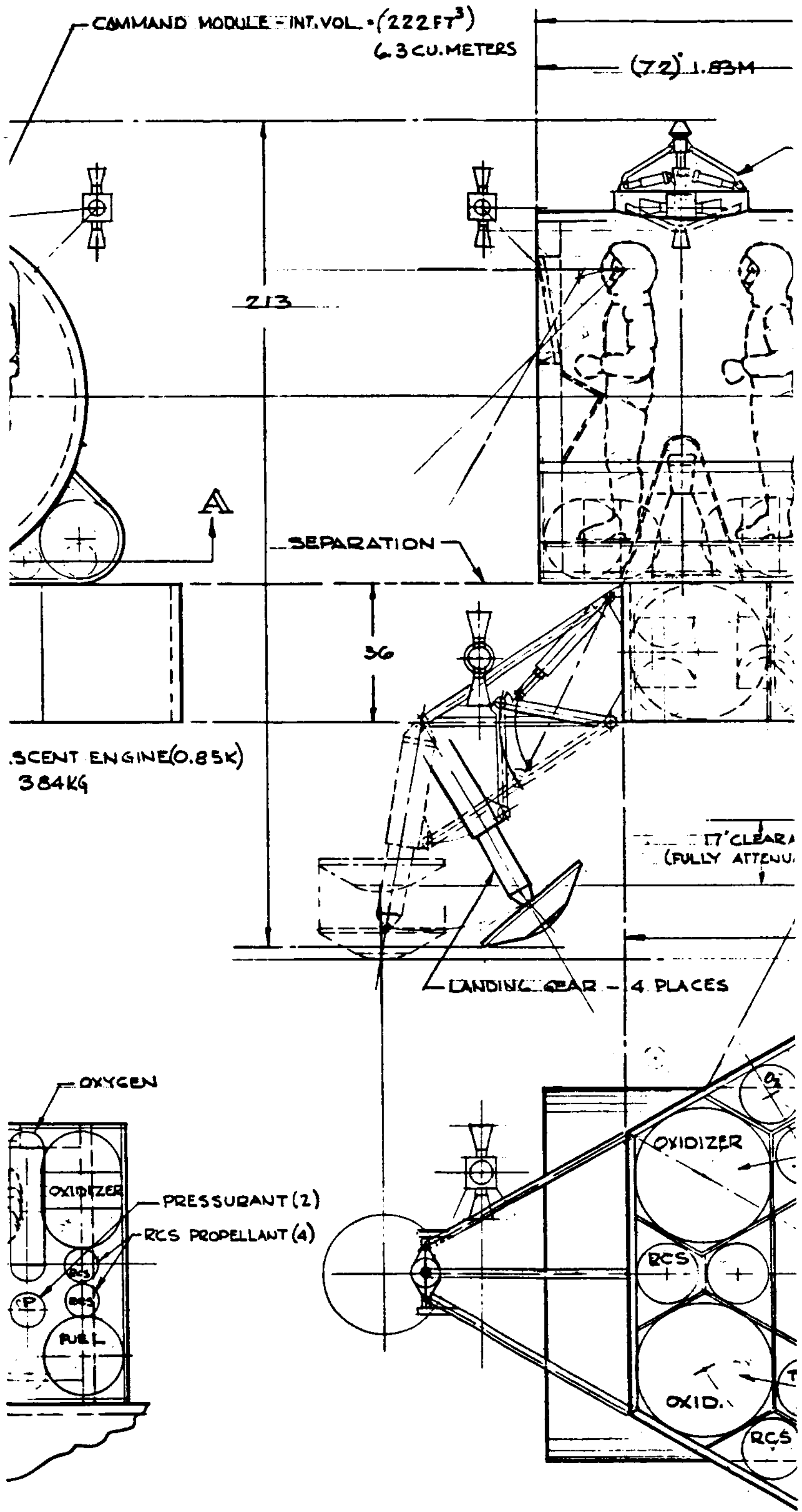

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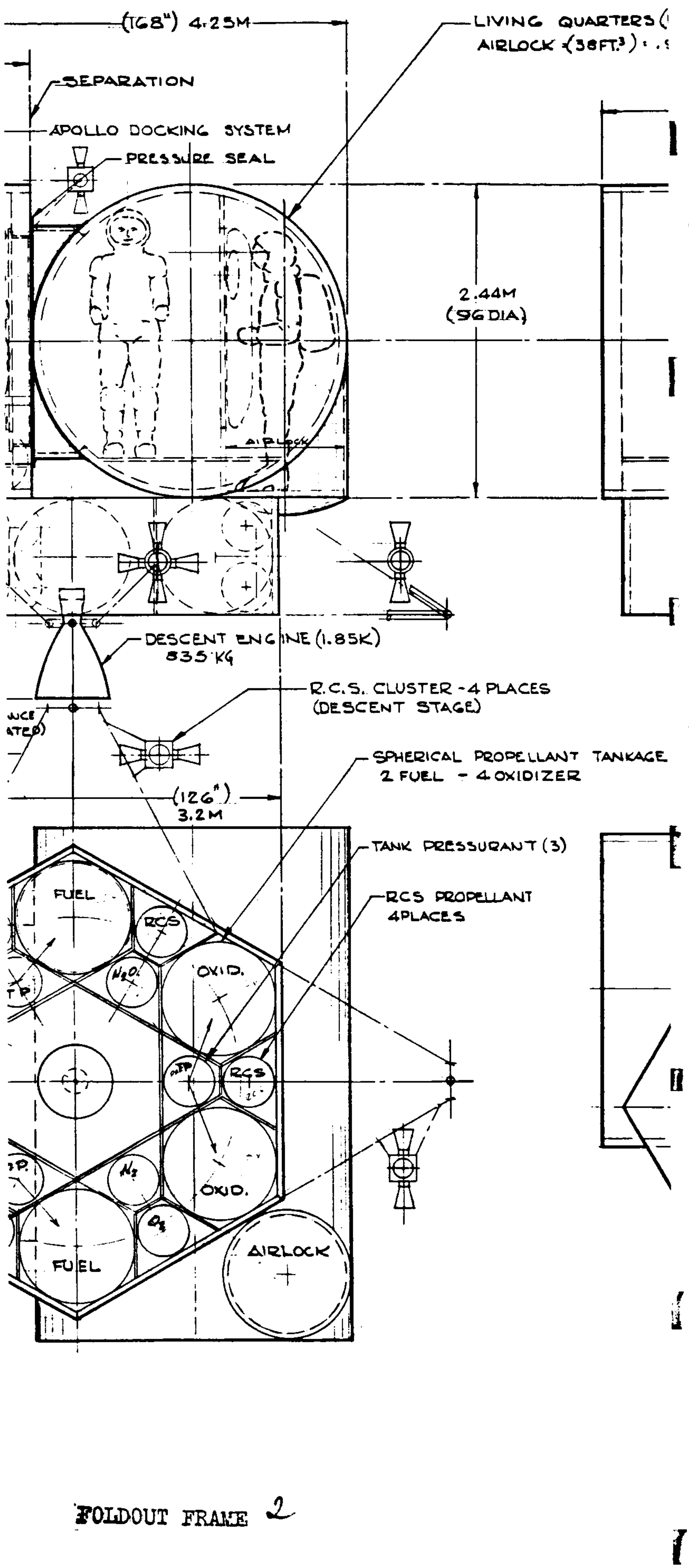

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NCL. AIRLOCK) - (532 FT^3) = 15.1 CU. METERS
365 CU.M.

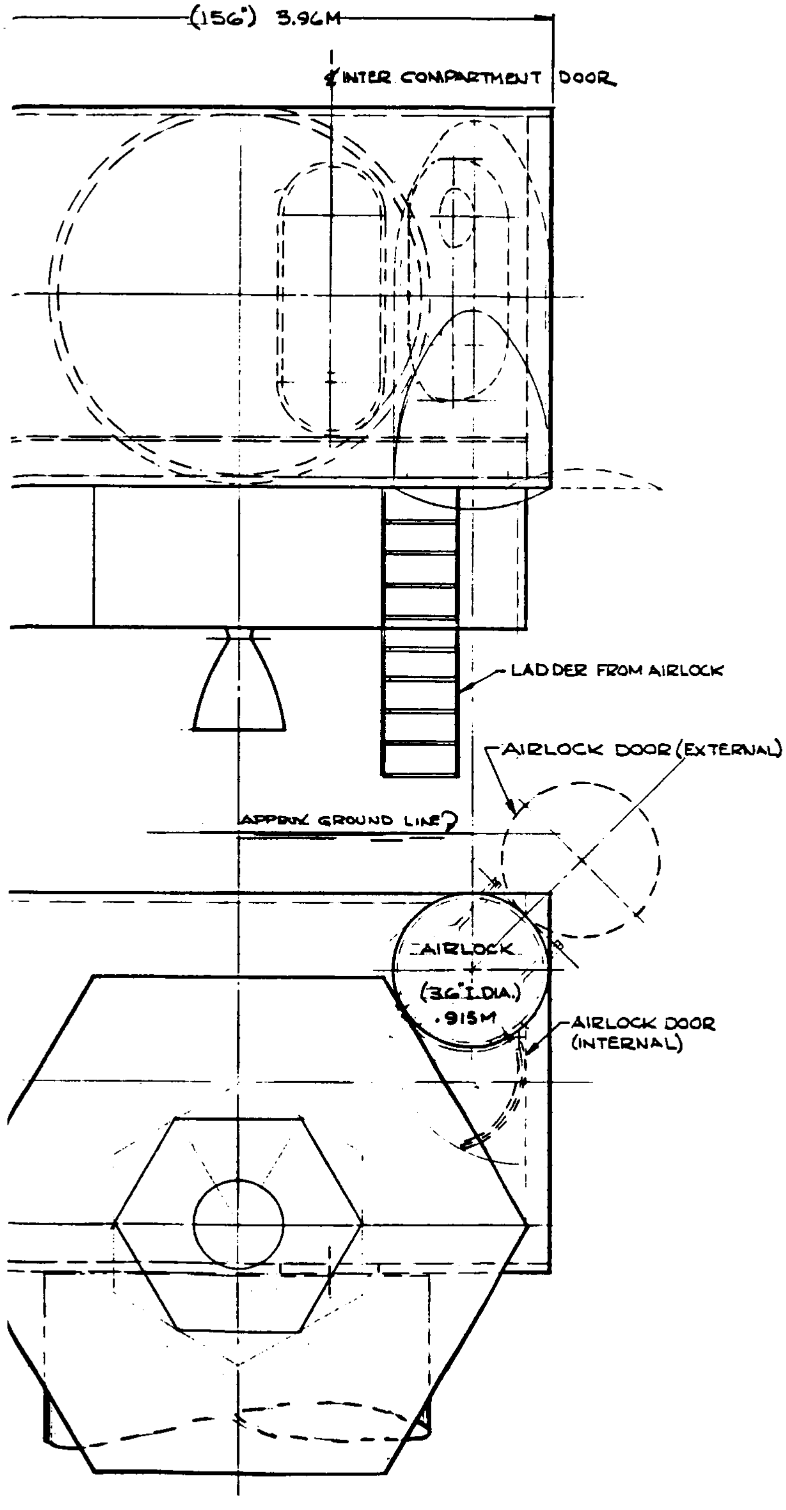


Figure 8. 3-Man R

ASCENT PAYLOAD		5,000#	2,280 KG
"	PROP'L.	900	406 KG
DESCENT PAYLOAD		20,000	9,000 KG
"	PROP'L	5,200	2,350 KG
	TOTAL	31,100#	14,100 KG

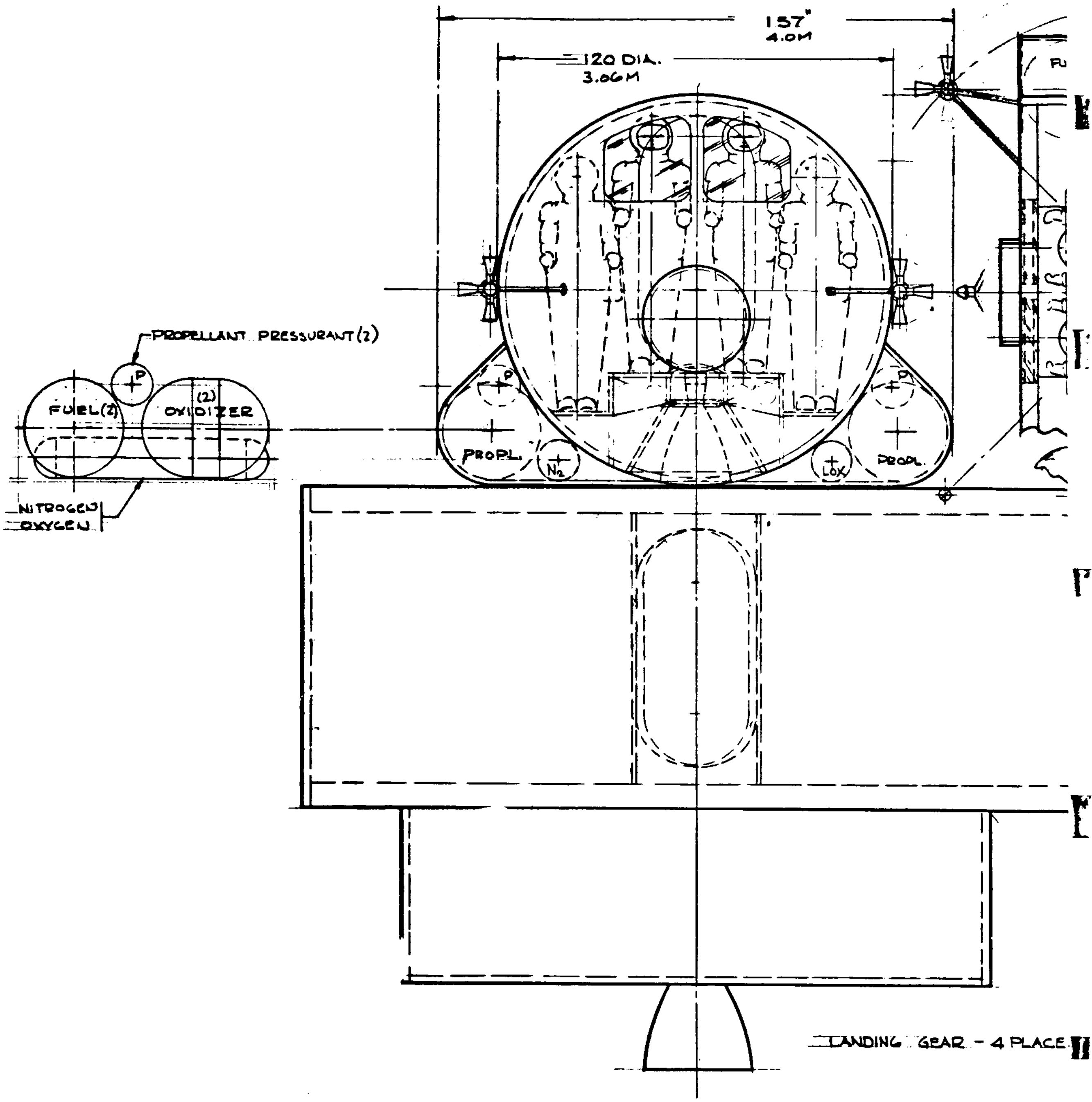
PROPELLANT = FLOX/MMH (MIX R = 2.75/1)
MMH = 54.4#/FT³
FLOX = 90.9#/FT³

30 DAY MISSION (3 MAN)

NITROGEN	12.90 FT³	650#	295 KG
OXYGEN	10.00 FT³	720#	325 KG
WATER	8.86 FT³	540#	244 KG
FOOD	5.63 FT³	393#	175 KG

PRELIMINARY CONCEPT
Not Iterated for Weight Scaling

etro PEM - Ceres and Vesta



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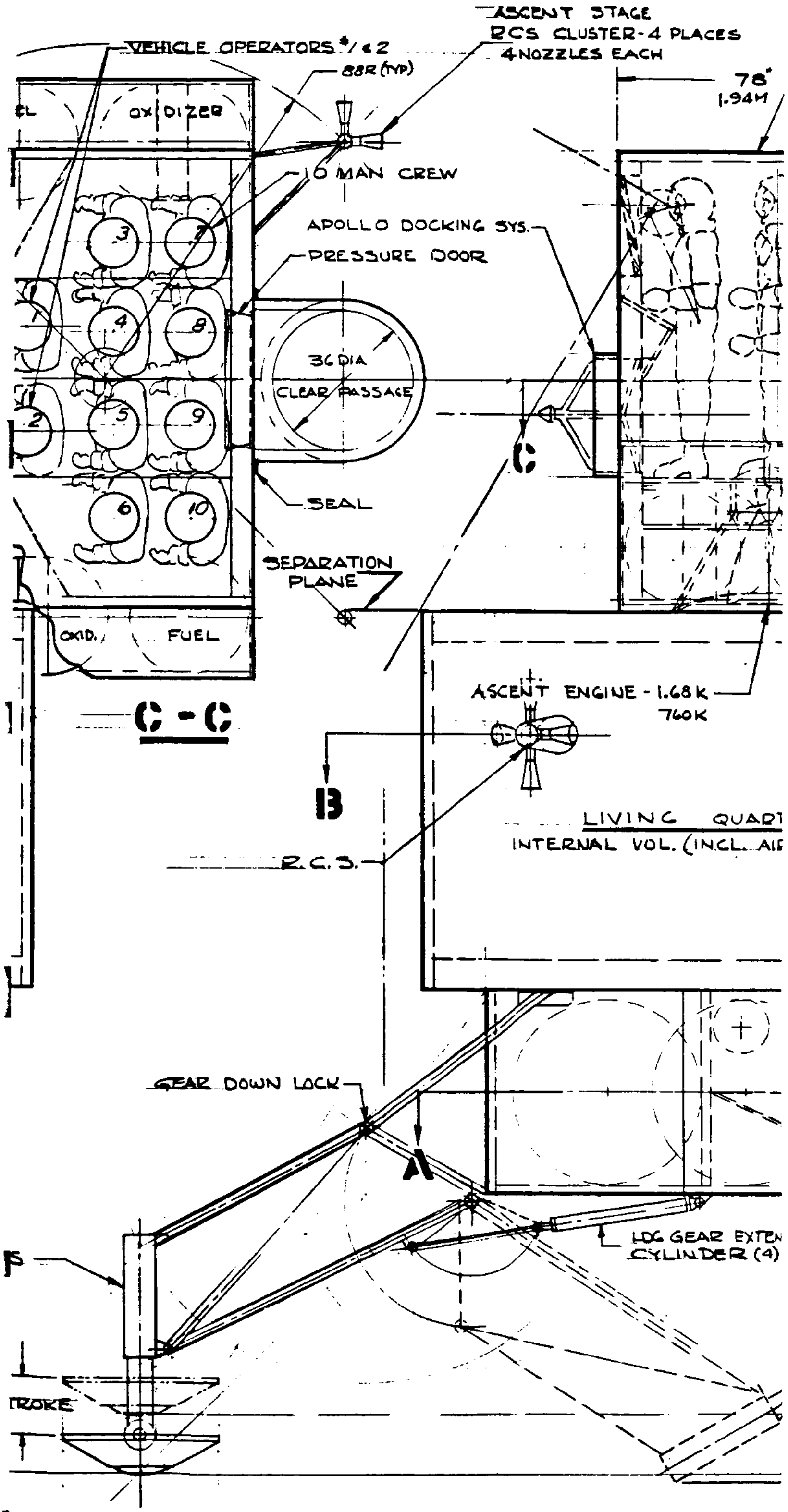
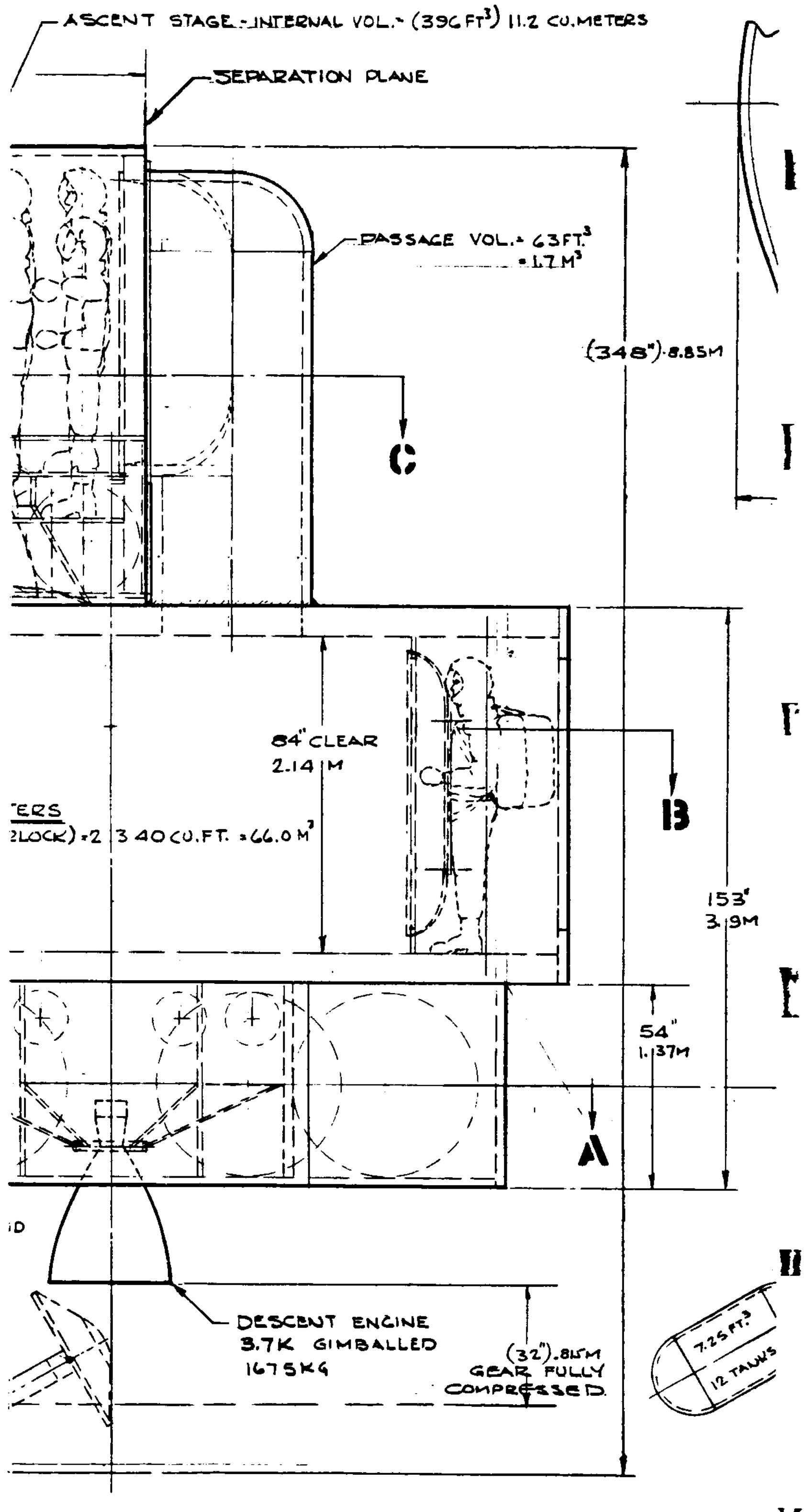


ASCENT STAGE
RCS CLUSTER-4 PLACES
4 NOZZLES EACH
VEHICLE OPERATORS #1 & 2
88R (TYP)
78"
1.94M
OXIDIZER
10 MAN CREW
APOLLO DOCKING SYS.
PRESSURE DOOR
36 DIA
CLEAR PASSAGE
SEAL
SEPARATION PLANE
OXID.
FUEL
C - C
C
ASCENT ENGINE - 1.68 K
760K
B
LIVING QUART
INTERNAL VOL. (INCL. AI
R.C.S.
GEAR DOWN LOCK
A
LDG GEAR EXTEN
CYLINDER (4)
TROKE



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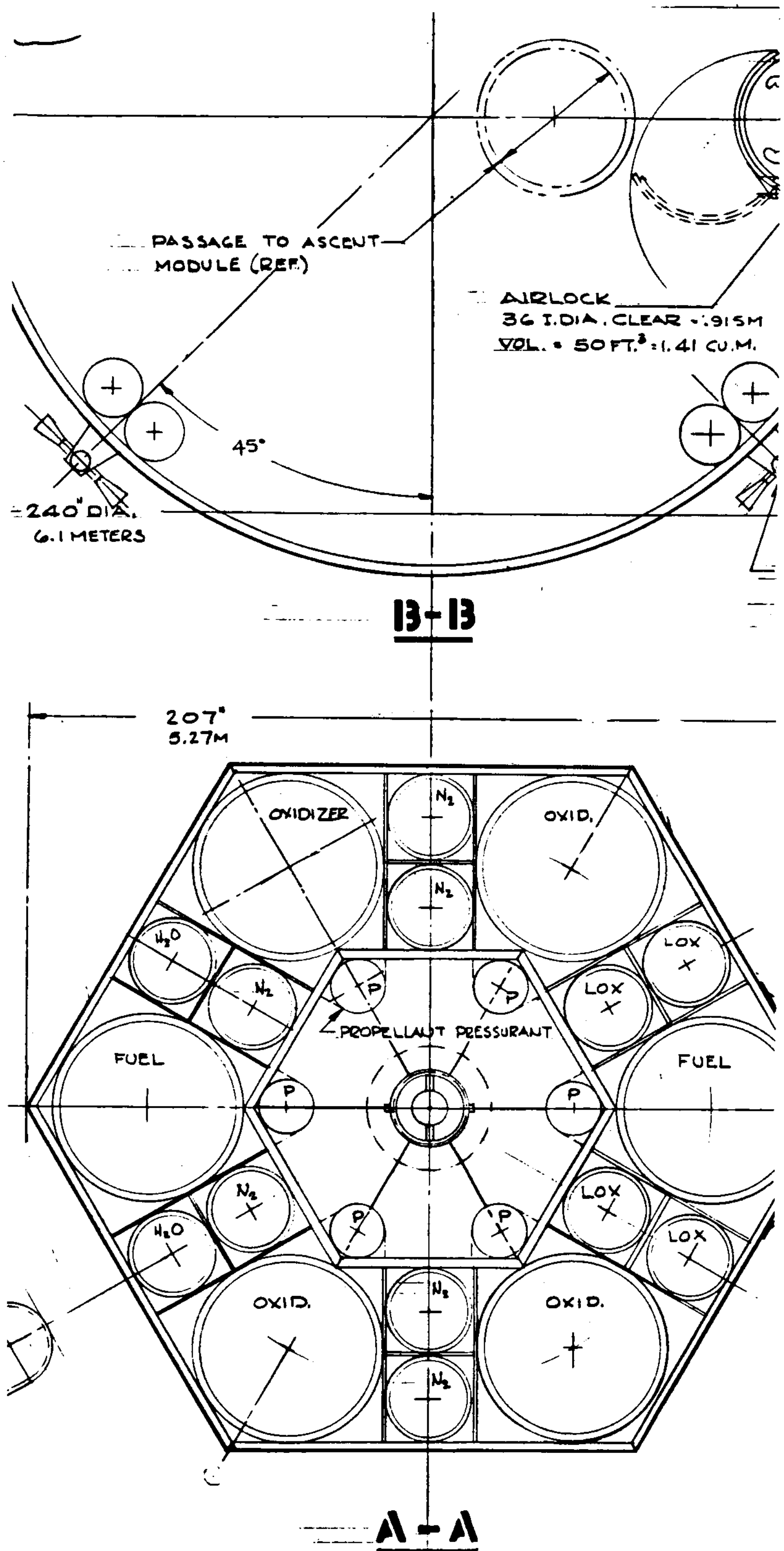
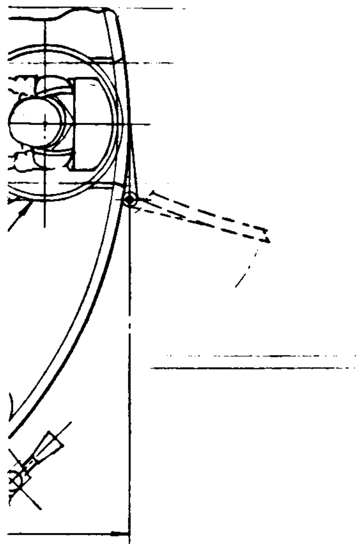


Figure 9. 10-Man R

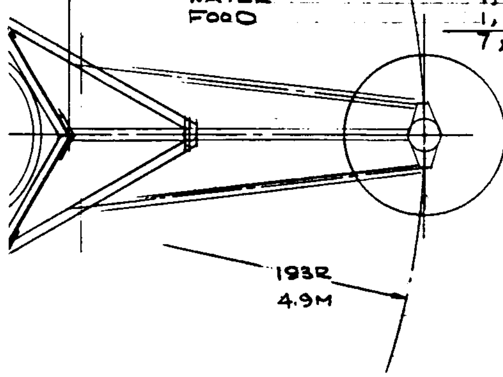


DESCENT STAGE
R.C.S CLUSTER -4 PLACES
-4 NOZZLES EACH

ASCENT PAYLOAD	10,000*	4,530KG
ASCENT PROPELLANT	1,800	815KG
DESCENT PAYLOAD	40,000	18,100 KG
DESC. PROPELLANT	10,400	4,700 KG
TOTAL	62,200*	27,800KG

PROPELLANT FLOX/MMH (MIX 2.75/1)
MMH 54.4#/FT.^{3}
FLOX 90.9#/FT^3

	30DAY MISSION (10MEN)	
NITROGEN	2,170*	980 KG
OXYGEN	2,400	1080
WATER	1,800	815
FOOD	1,300	590
	7,670*	3470 KG.



PRELIMINARY CONCEPT
Not Iterated for Weight Scaling

etro PEM - Ceres and Vesta

FOLDOUT FRAME

In the stowed configuration in the parent spacecraft, the 10-man vehicle would require a clearance envelope of 6.5 meters (260 inches) in diameter and 8.83 meters (348 inches) long.

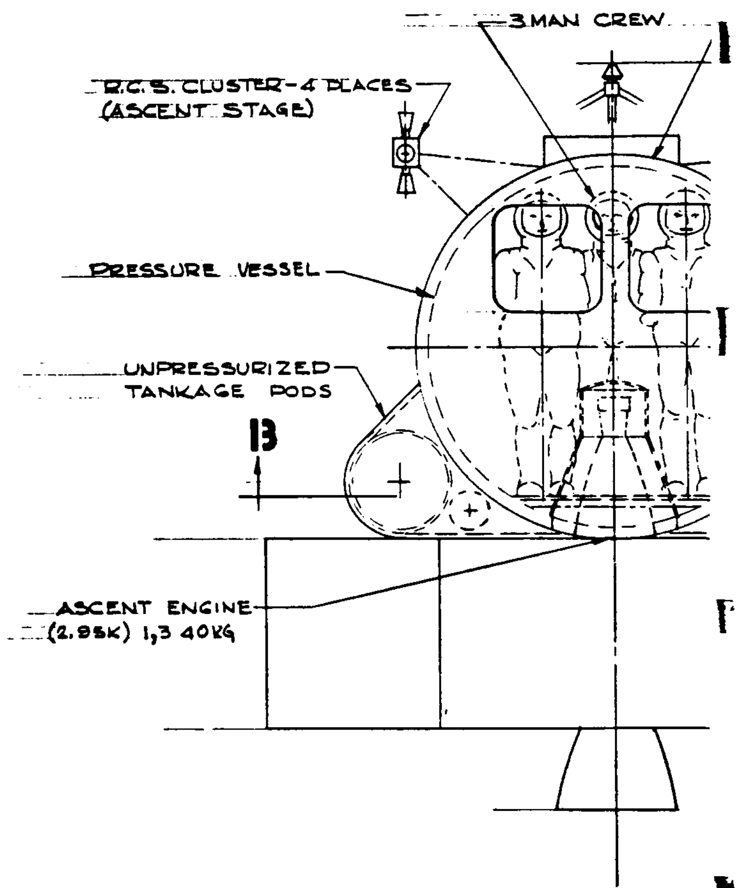
3-Man Retro PEM (Ganymede)

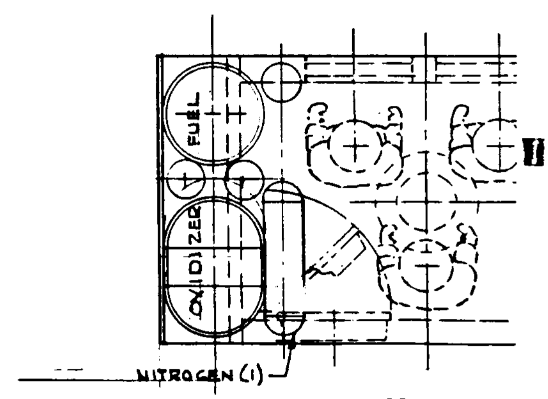
A 3-man retrobraking planetary excursion module for Ganymede is shown in Figure 10. This vehicle has an estimated gross weight of 17,300 kilograms (37,900 pounds). The general arrangement is a logical extension of the basic 3-man vehicle previously configured for Ceres and Vesta, differing basically in the size of the descent and ascent rocket motors and propellant quantities. The descent stage features a hexagonal shaped segmented box structure which incorporates the descent propellant tanks, propulsion system, mission life support consumables, descent attitude control system, and a four legged landing gear system. Mounted on the upper surface of the landing stage are the ascent stage and the cylindrically shaped pressurized compartment which serves as the crew mission quarters.

The descent stage provides for 3,740 kilograms (8,250 pounds) of FLOX/MMH propellants contained in six spherical tanks with the other mission consumables and Attitude Control Subsystem (ACS) propellants contained in cylindrical tanks integrated into available spaces in the landing stage. The four legged landing gear has been designed to fold in under the body to reduce the overall space requirements when the PEM is subsequently integrated into a complete spacecraft. A throttleable throat-gimballed pressure-fed engine of 2,800 kilograms (6,150 pounds) thrust is used for descent and terminal landing.

The ascent stage is a complete autonomous stage that accommodates the 3-man crew in a standing position during ascent and descent. The pressurized compartment is basically a cylindrical section 2.44 meters (96 inches) in diameter and 1.83 meters (72 inches) in length, with a gross volume of 6.3 cubic meters (222 cubic feet). Ascent propellants, weighing 3,010 kilograms (6,650 pounds) are supported off of the basic cylindrical structure in spherical and cylindrical tanks. A single pressure-fed engine of 1,340 kilograms (2,950 pounds) thrust protrudes through the floor of the crew compartment and is sealed from the compartment by a tapered cylindrical can. A docking interface and appropriate systems are integrated into the upper surface of the crew compartment to permit docking after ascent. A separate ascent attitude control system is provided for this stage.

The mission quarters for this vehicle is a cylindrical pressurized compartment 2.44 meters (96 inches) in diameter and 3.96 (156 inches) meters in length positioned adjacent to the aft face of the ascent stage crew compartment. Pressure doors in both compartments and a short interconnecting tunnel provides access between the two compartments. The mission quarters has a gross volume of 14.93 cubic meters (532 cubic feet) which would be





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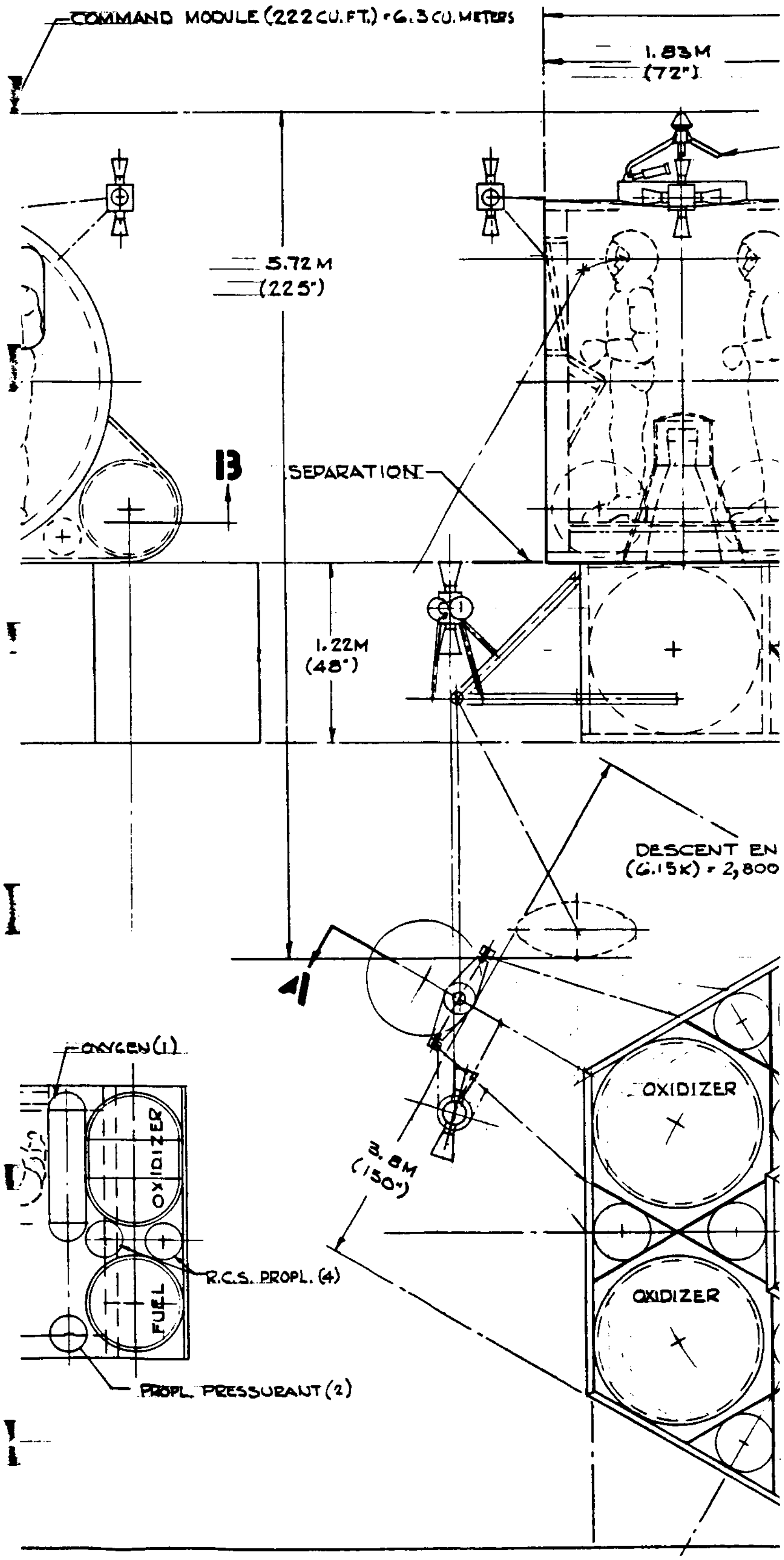
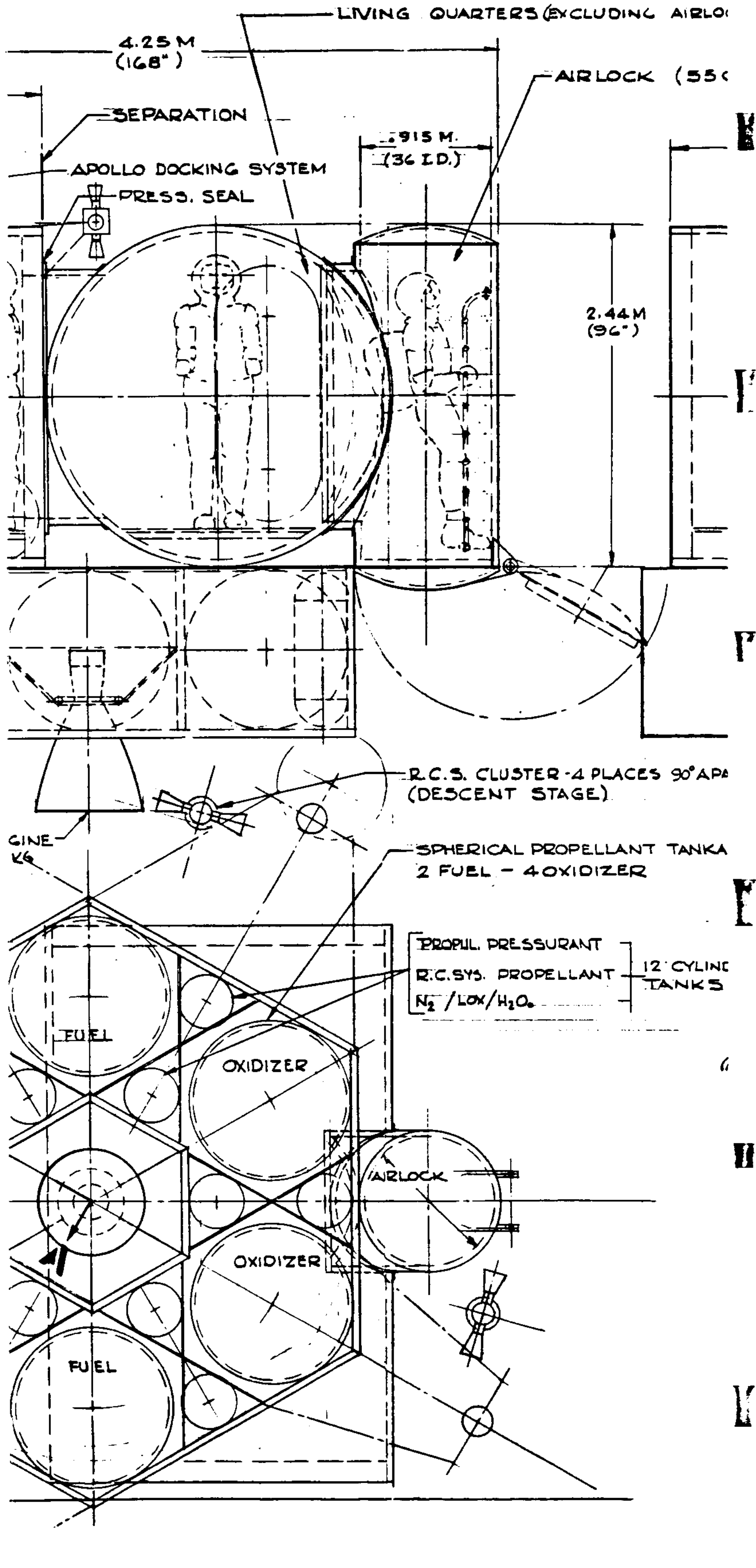

COMMAND MODULE (222 CU.FT.) = 6.3 CU. METERS
1.83M
(72")
5.72 M
(225")
B
SEPARATION
1.22M
(48")
DESCENT EN
(6.15K) = 2,800
A
OXYGEN (1)
OXIDIZER
FUEL
R.C.S. PROPL. (4)
PROPL. PRESSURANT (2)
3.8M
(150")
OXIDIZER
OXIDIZER

MED.



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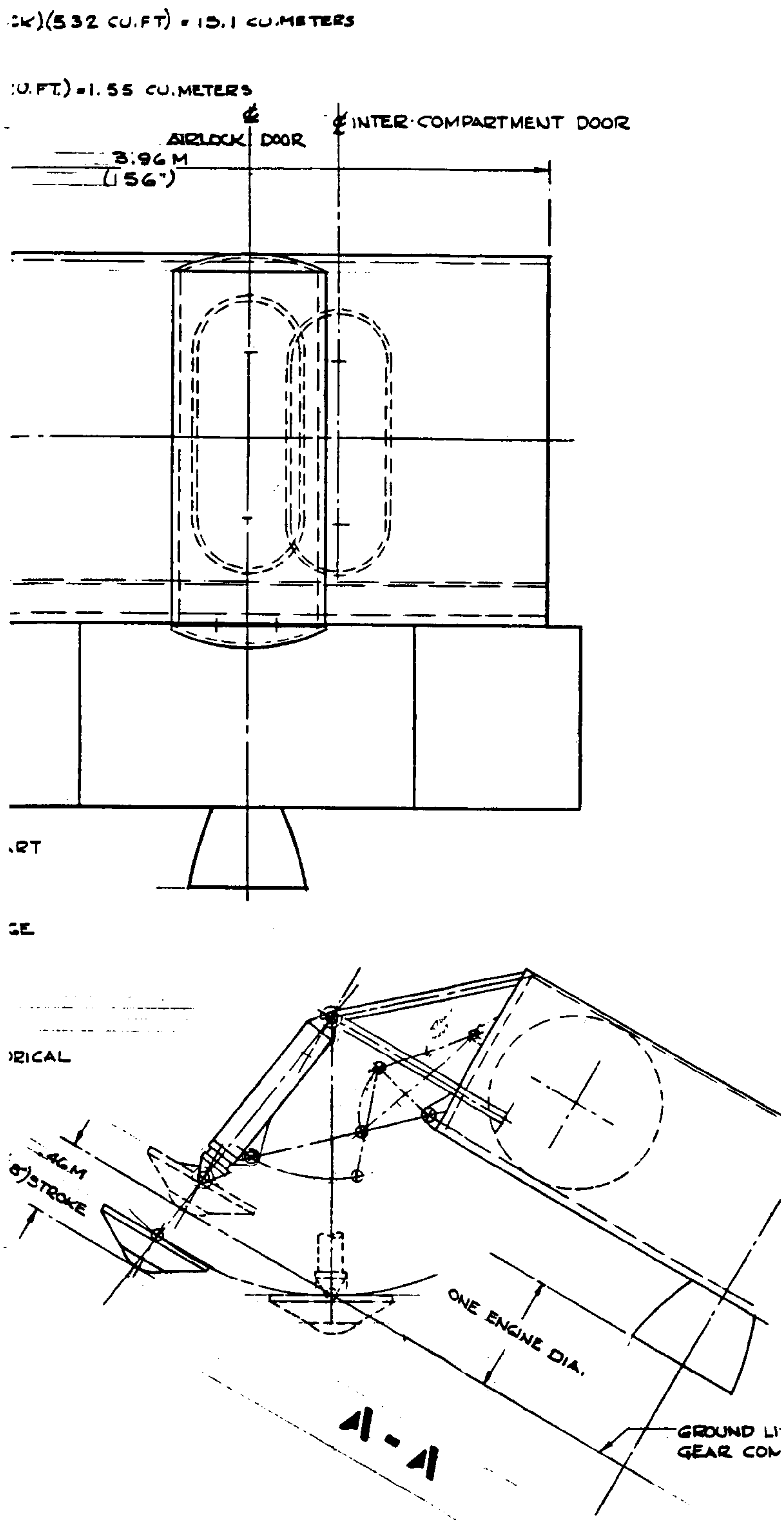

Figure 10. 3-Ma

ASCENT PAYLOAD	5,000 LBS.	2,280 KG
ASCENT PROPELLANT	6,650	3,010 KG
DESCENT PAYLOAD	21,000	9,500 KG
" PROP'L.	8,250	3,740 KG
TOTAL	37,900	17,300 KG

PROPELLANT FLOX/MMH (MIX = 2.71/1)
MMH = 54.4#/FT.³
FLOX = 90.9#/FT.³

30 DAY MISSION (3 MEN)

NITROGEN	12.90 FT³	- 650# -	295 KG
OXYGEN	10.00 "	720# -	325 KG
WATER	8.86 "	540# -	244 KG
FOOD	5.63 FT³	393# -	175 KG

PRELIMINARY CONCEPT
Not Iterated for Weight Scaling

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ın Retro PEM - Ganymede

supplemented by the 6.3 cubic meters (222 cubic feet) crew compartment volume to provide approximately 21.2 cubic meters (750 cubic feet) gross. A cylindrical airlock has been externally mounted to the aft surface of the mission quarters to provide access to the surface of the target body. The lower end closure of airlock serves as a hatch for access to an externally mounted ladder.

In the stowed configuration, this vehicle would require a clearance envelope of 6.5 meters (256 inches) in diameter and a length of 5.72 meters (225 inches) in the parent spacecraft.

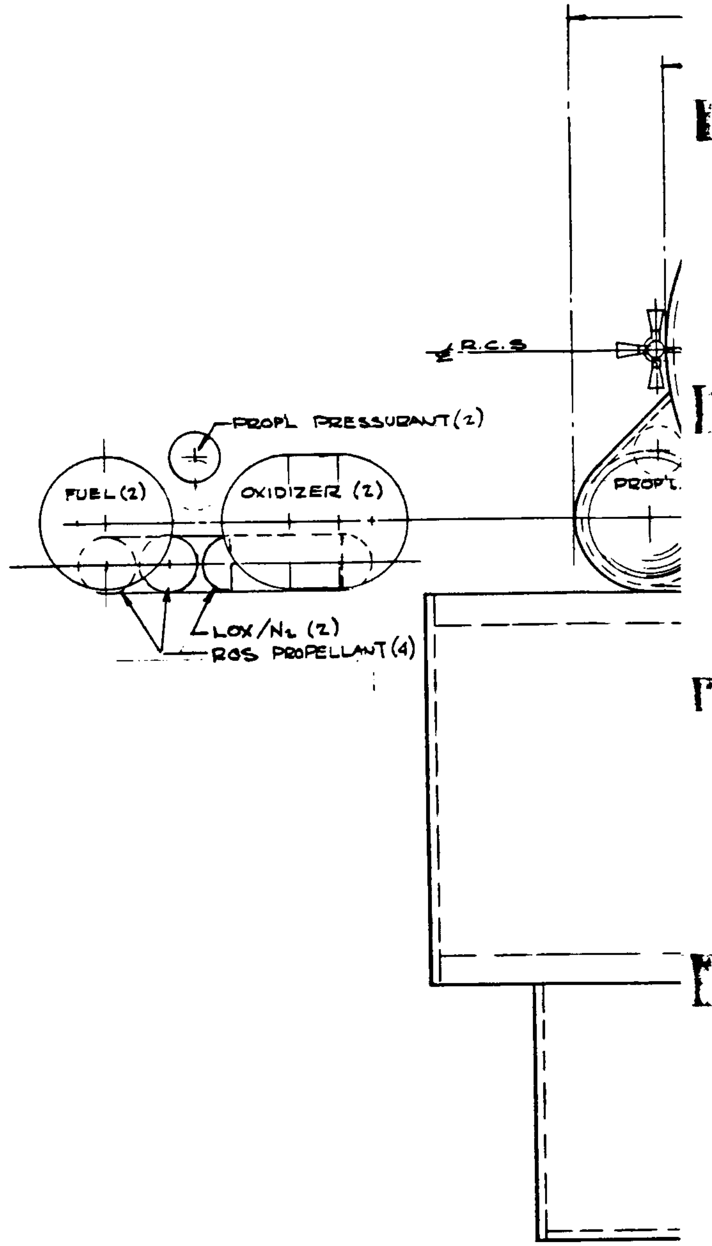
10-Man Retro PEM (Ganymede)

The configuration for this vehicle is shown in Figure 11. It utilizes a separate ascent and descent stage with the mission quarters compartment integrated into the landing stage. This concept is basically identical to the 10-man retro PEM previously developed for Ceres and Vesta. Since the propulsion requirements are greater for Ganymede, larger rocket engines and more propellant have been accommodated in the ascent and descent stages.

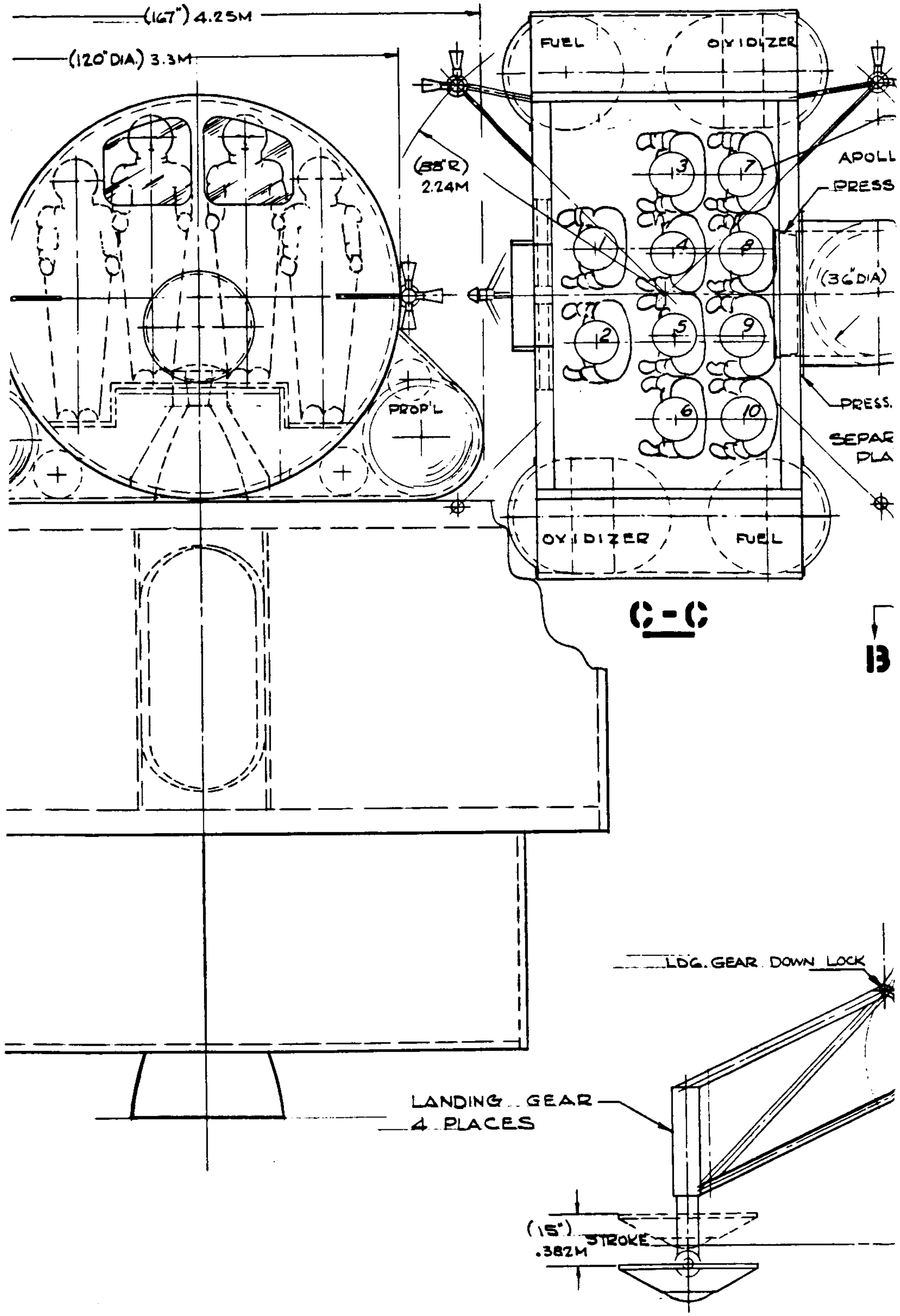
The landing stage consists of a hexagonal shaped box structure with internal sector beams to provide structural support for the six spherical propellant tanks with a capacity of 7,500 kilograms (16,500 pounds), the descent attitude control propellants and the mission life support consumables, and the four-legged deployable landing gear. A throat-gimballed throttleable pressure-fed engine with a thrust level of 5,550 kilograms (12,300 pounds) is provided for descent retro-propulsion and terminal landing.

As previously discussed under the 10-man Ceres and Vesta retro PEM configurations, the mission quarters compartment has been configured to be a cylindrical section 6.1 meters (240 inches) in diameter with flat ends with an internal clear ceiling height of 2.14 meters (84 inches). The gross mold line pressurized volume of the compartment is 66 cubic meters (2,340 cubic feet). The compartment is positioned symmetrically on top of the descent stage and is provided with an internal cylindrical airlock to facilitate access to the target body surface by a ladder.

The ascent stage is located on top of the mission quarters, offset to provide adequate down visibility to the operating crew for landing site assessment prior to touchdown. The ascent propellants weighing 6,050 kilograms (13,300 pounds) are contained in spherical and cylindrical tanks which are structurally supported off of the crew compartment. The 10-man crew is accommodated in a standing position during descent and ascent in a 3.06 meter (120 inch) diameter cylindrical pressurized compartment 1.98 meters (78 inches) long. A pressure-fed engine with a thrust level of 2,670 kilograms (5,900 pounds) protrudes in an enclosed well through the floor of the compartment to provide a clear separation plane for the ascent stage. Separate attitude control nozzles and propellant systems are provided for both the



R.C.S
PROPL PRESSURANT (2)
FUEL (2)
OXIDIZER (2)
PROP'L
LOX/N2 (2)
RCS PROPELLANT (4)



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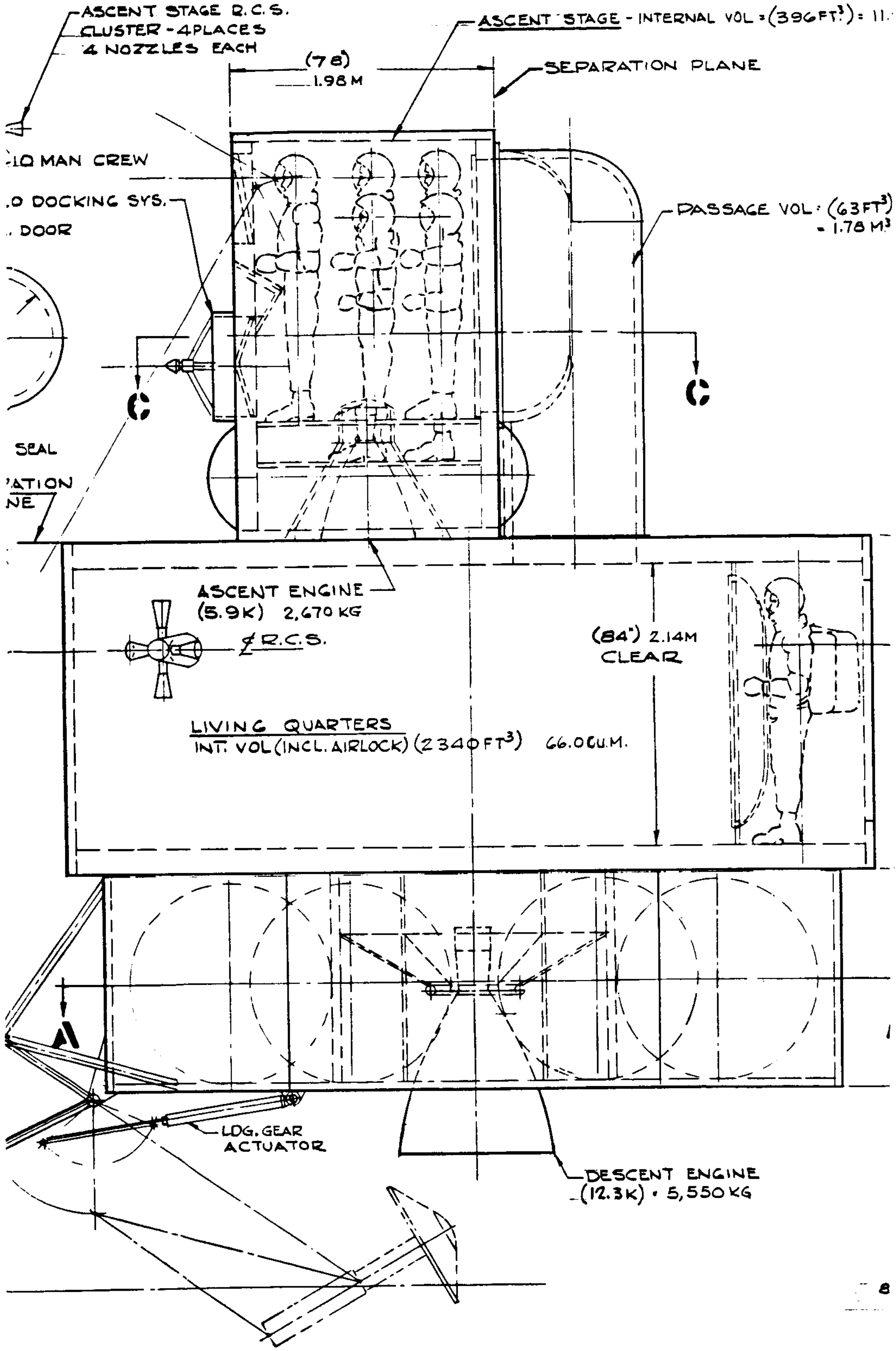
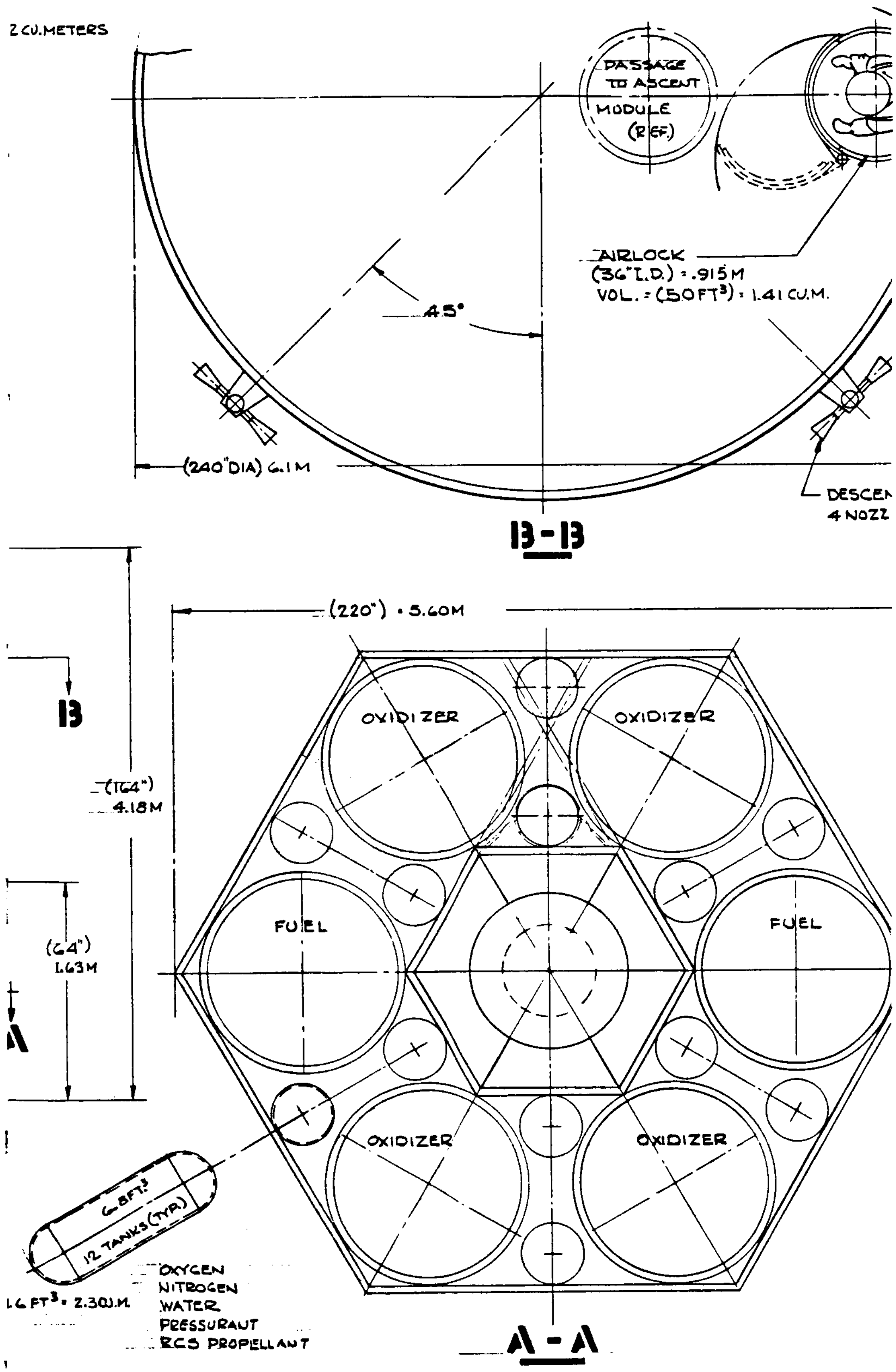
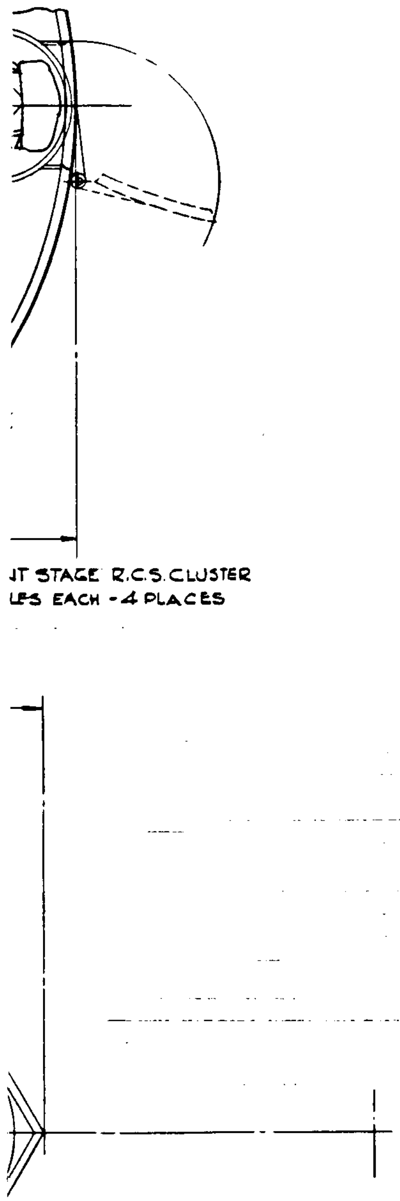


ASCENT STAGE R.C.S. CLUSTER - 4 PLACES 4 NOZZLES EACH
ASCENT STAGE - INTERNAL VOL = (396 FT³) = 11.
(78") 1.98 M
SEPARATION PLANE
10 MAN CREW
DOCKING SYS.
DOOR
PASSAGE VOL: (63 FT³) = 1.78 M³
C
C
SEAL
ASCENT ENGINE (5.9K) 2,670 KG
℄ R.C.S.
(84") 2.14 M CLEAR
LIVING QUARTERS
INT. VOL (INCL. AIRLOCK) (2340 FT³) 66.0 CU.M.
A
LDG. GEAR ACTUATOR
DESCENT ENGINE (12.3K) = 5,550 KG

#2


2 CU.METERS
PASSAGE TO ASCENT MODULE (REF)
AIRLOCK
(36" I.D.) = .915 M
VOL. = (50 FT³) = 1.41 CU.M.
45°
(240" DIA) 6.1 M
DESCEN
4 NOZZ
B-B
(220") = 5.60 M
B
(164")
4.18 M
(64")
1.63 M
A
OXIDIZER
OXIDIZER
FUEL
FUEL
OXIDIZER
OXIDIZER
6.8 FT³
12 TANKS (TYP.)
1.6 FT³ = 2.30 U.M.
OXYGEN
NITROGEN
WATER
PRESSURANT
RCS PROPELLANT
A-A

Figi



ASCENT PAYLOAD	10,000 LBS	4,536. KG
ASCENT PROPELLANT	13,300	6,050
DESCENT PAYLOAD	62,000	28,100
DESCENT PROP'L.	16,500	7,500
TOTAL	101,800	45,800

PROPELLANT FLOX/MMH (MIX 2.75/1)
MMH 54.4#/FT.³
FLOX 90.9#/FT.³

30 DAY MISSION (10 MEN)

NITROGEN	2,170 LBS	985 KG
OXYGEN	2,400	1,090
WATER	1,800	815
FOOD	1,300	590
TOTAL	7,670	3480

PRELIMINARY CONCEPT
Not Iterated for Weight Scaling

ure 11. 10-Man Retro PEM - Ganymede

#3

ascent and descent stages. Access to the mission quarters is provided by a pressure-tight door in the aft face of the crew compartment and an adjacent vertical passageway. The docking interface and associated systems have been integrated into the forward facing surface of the crew compartment to reduce overall stowed height and to provide easy direct visual assessment during the docking maneuver.

In the stowed configuration in the parent spacecraft, the 10-man vehicle would require a clearance envelope 7.27 meters (286 inches) in diameter and 9.06 meters (357 inches) in length.

Mercury Retro PEM

The Mercury PEM is identical in concept to the Ganymede configuration, differing basically in the size of the descent and ascent rocket motors and propellant quantities. Since the propulsion requirements for Mercury landings are greater than the requirements for Ganymede landings, larger rocket engines and more propellant will be required in the ascent and descent stages. Detail configuration drawings were not generated for the Mercury PEM's since it was determined that the configurations developed for Ceres, Vesta, and Ganymede provided adequate data for the development of the PEM weight scaling equations.

Venus Excursion Module

A gross analysis was conducted to establish the order of magnitude of the required mass of a Venus excursion module (VEM). The configuration assumed was an aerobraking vehicle with a propulsive terminal descent. The manned module was assumed to be a 2970 kg (6500 lb) two-man module with a minimum volume and a habitable lifetime of seven days or less. A three stage launch vehicle was assumed with an equal distribution of the total characteristic velocity requirement, a stage mass fraction of 0.90 for all stages, and N_2O_4/MMH propellant. Specific impulse values of 240 and 272 seconds were considered for the first stage with second and third stage values of 318 and 320 seconds, respectively. The specific impulse values were based on the NASA/MSFC low density atmospheric model. Based on the above assumptions, the required VEM lift-off weight is shown in Figure 12 as a function of the total launch vehicle characteristic velocity requirement. It has been estaimated that the ascent characteristic velocity requirement will not be less than 10.67 km/s (35,000 ft/s) which would require a lift-off weight of approximately 363,000 kg (800,000 lb) assuming the lower first stage specific impulse. For comparison, the characteristic velocity requirement for direct ascent into a 300-nautical-mile Earth orbit for the Saturn IB is 10.07 km/s (33,050 ft/s) using optimum steering after launch escape tower jettison. A study reported in Reference 1 probably establishes the upper limit on the characteristic velocity requirements. The characteristic velocity

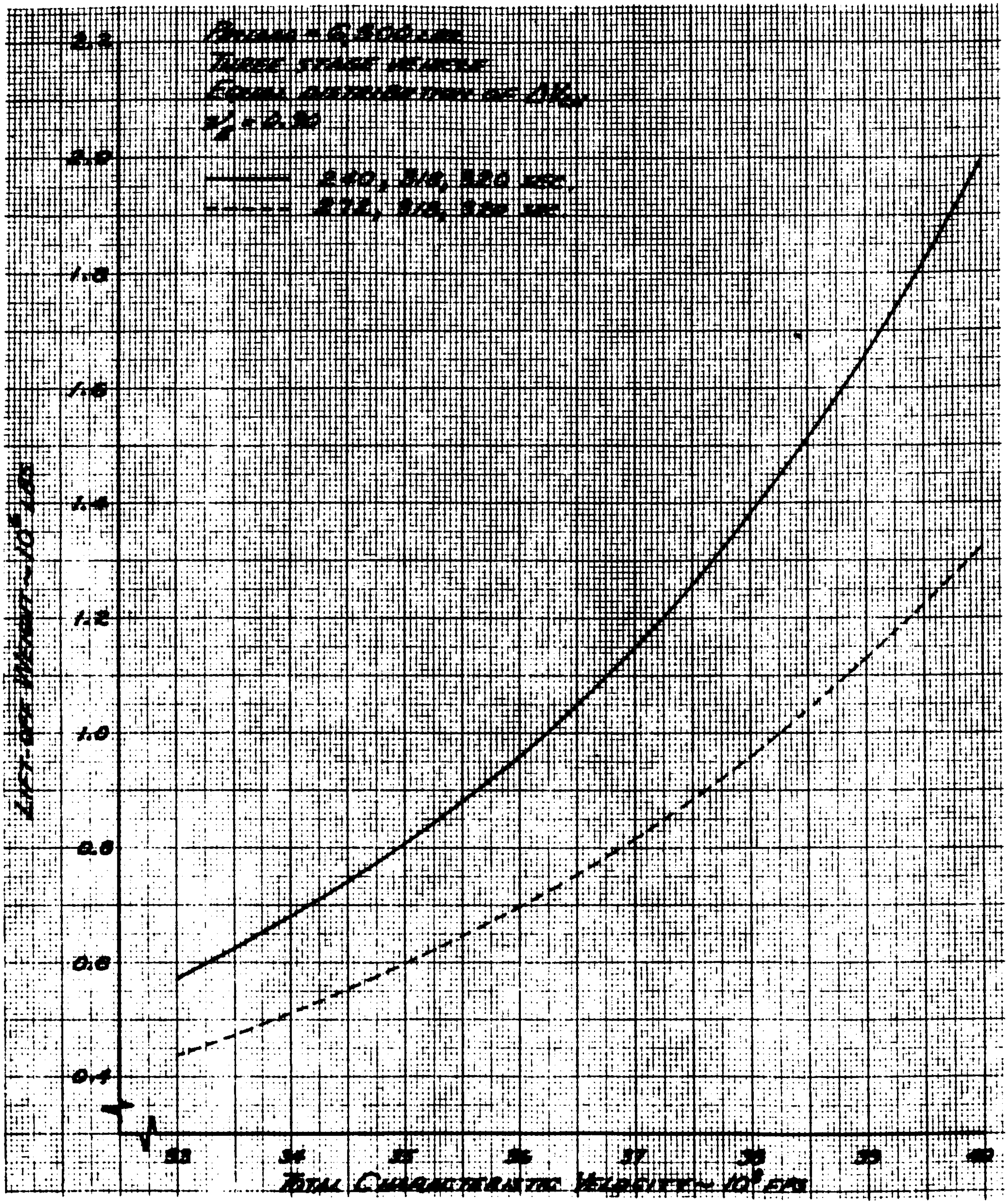


Figure 12. Venus Excursion Module Mass Characteristics

requirements were determined in the reference report for various initial thrust to weight ratios and number of stages. The minimum characteristic velocity requirement for a three stage vehicle was 12.10 km/s (39,700 ft/s) which, for the above assumptions, results in a lift-off weight of 840,000 kg (1,850,000 lb). The characteristic velocity requirements presented in the reference were based on a gravity turn steering mode which would result in a higher characteristic velocity requirement than would result if a more optimum steering mode were employed.

MISSION MODULES

Mission module configurations for crew sizes of 14 and 20 men have been parametrically configured in Figure 13. The ground rules that were established for the mission modules include the following:

1. Mission durations of 500, 1000, and 1500 days.

2. Partially closed ecological system.

3. Provisions for a "storm cellar" to provide radiation protection during peaks of solar activity and serve as a basic command and control center.

4. Provide sufficient volume for on-board personnel centrifuges for periodic gravity conditioning.

The basic elements that were a factor in determining the volumes for the six mission modules are the following:

1. Crew and crew support

2. Furniture, housekeeping

3. Food management

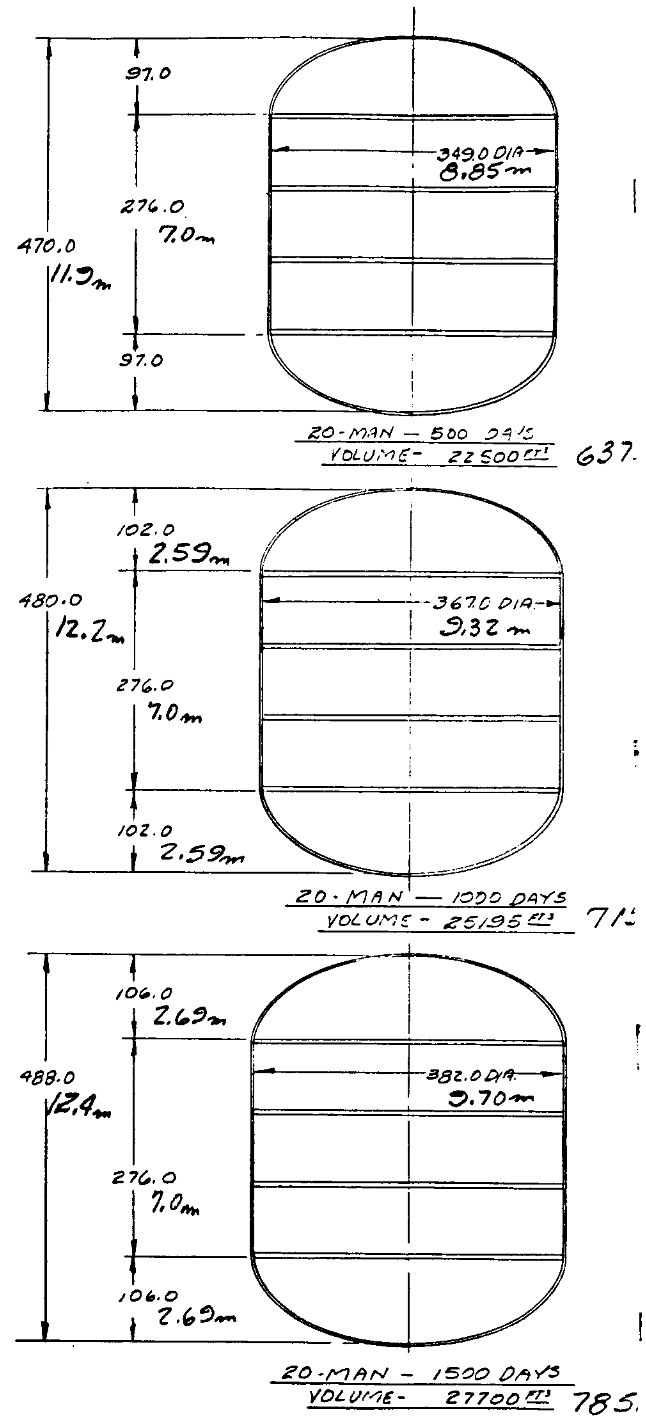
4. Water supply

5. Waste management

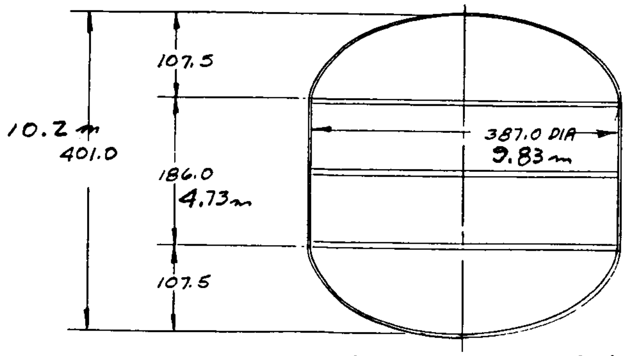
6. Temperature and humidity control

7. Atmosphere

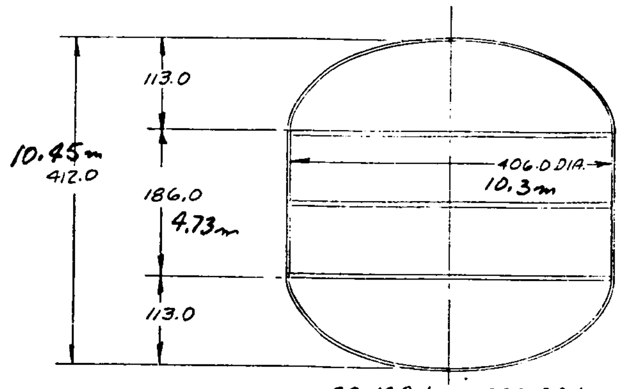
8. Instrument controls



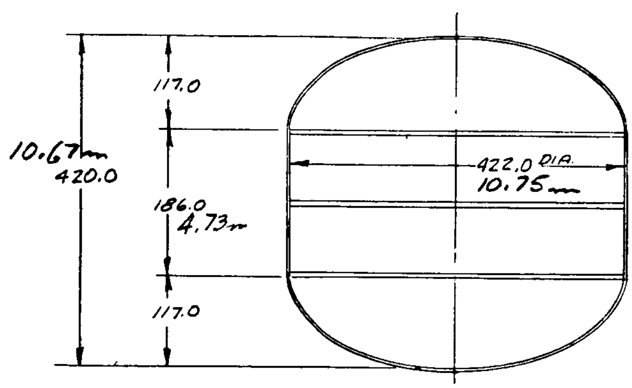
3-DECK MODULE



20-MAN 500 DAYS
VOLUME - 22440 FT3 6

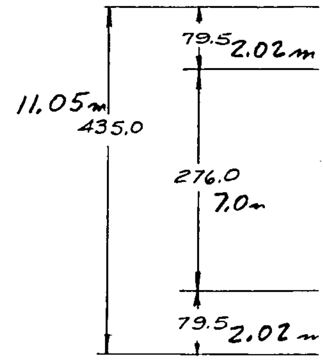


20-MAN 1000 DAYS
VOLUME - 25180 FT3 7

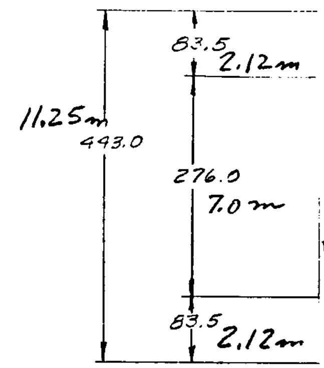


20-MAN 1500 DAYS
VOLUME - 27700 FT3 7E

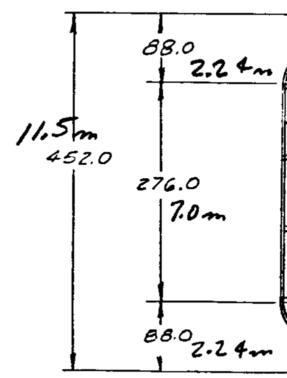
2- DECK MODULE



35.0 m³



20 m³



35 m³

413

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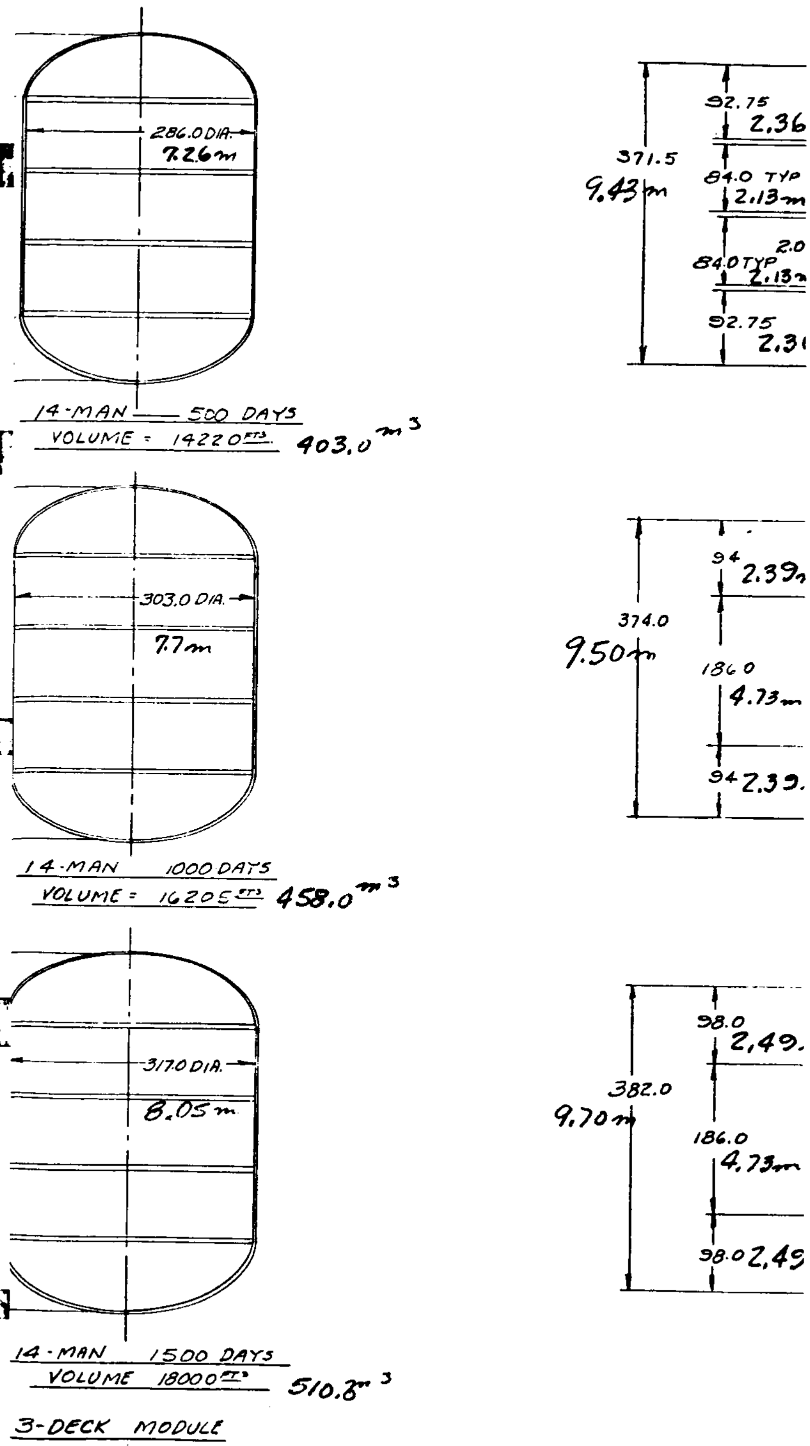
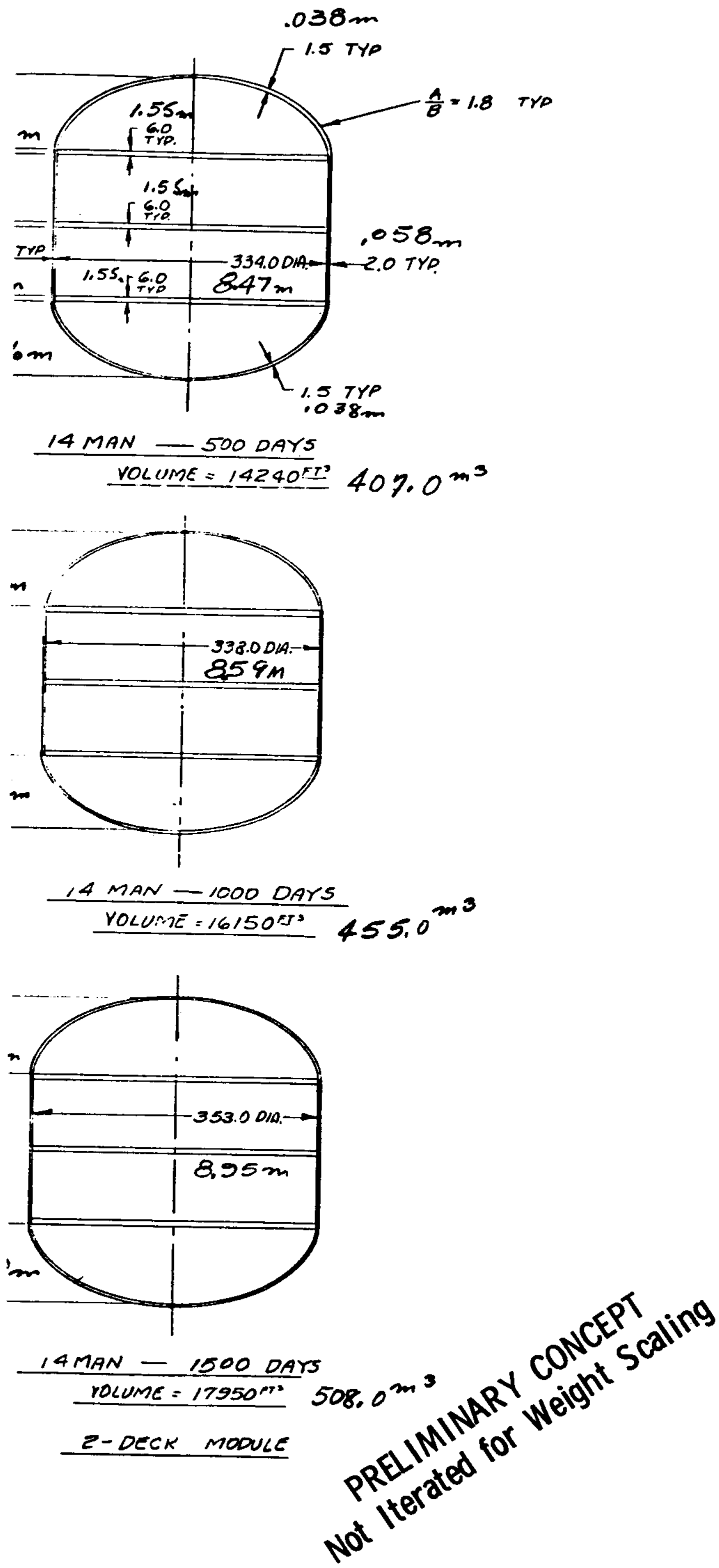


Figure 13. Mission



Module Concept Summary

9. Storm cellar

10. Centrifuge

11. Free volume for crew

12. Life support system gas storage

The scaling equations used to establish the specific volumes for these items were obtained from Reference 2. Table 1 summarizes the volumes calculated for the two different crew sizes and three mission durations. It should be noted that a free volume per man of 600 cubic feet was used during the generation of the conceptual designs. Subsequent to the generation of the conceptual designs, a nominal value of 750 ft^3/man was established as the nominal value.

Since the mission modules will be subsequently integrated into a variety of retrobraking and aerobraking spacecraft concepts, it was decided to configure two different types of mission modules, one with two floors and one with three floors. Elliptical end bulkheads with an a/b ratio of 1.8 were used. The spacing between floors was set at 84 inches for head clearance. This value is a standard value currently in use. During the sizing operation, the diameters of the mission modules was allowed to be the dependent variable. It is assumed that during the integration of the mission modules into the spacecraft at a later date, the concepts that have been generated will provide a spectrum from which a near-optimum choice can be made, with a resultant minimum of iteration for fit with specific spacecraft. It is of significance to note (Figure 13) that, if a 10 meter diameter constraint is imposed, the mission modules will satisfy such a constraint for all crew sizes considered. Although this conclusion is shown for a free volume per man of 600 ft^3/man, it was subsequently determined that this conclusion is valid even if the free volume per man is increased to 750 ft^3/man.

AEROBRAKING SPACECRAFT DESIGN

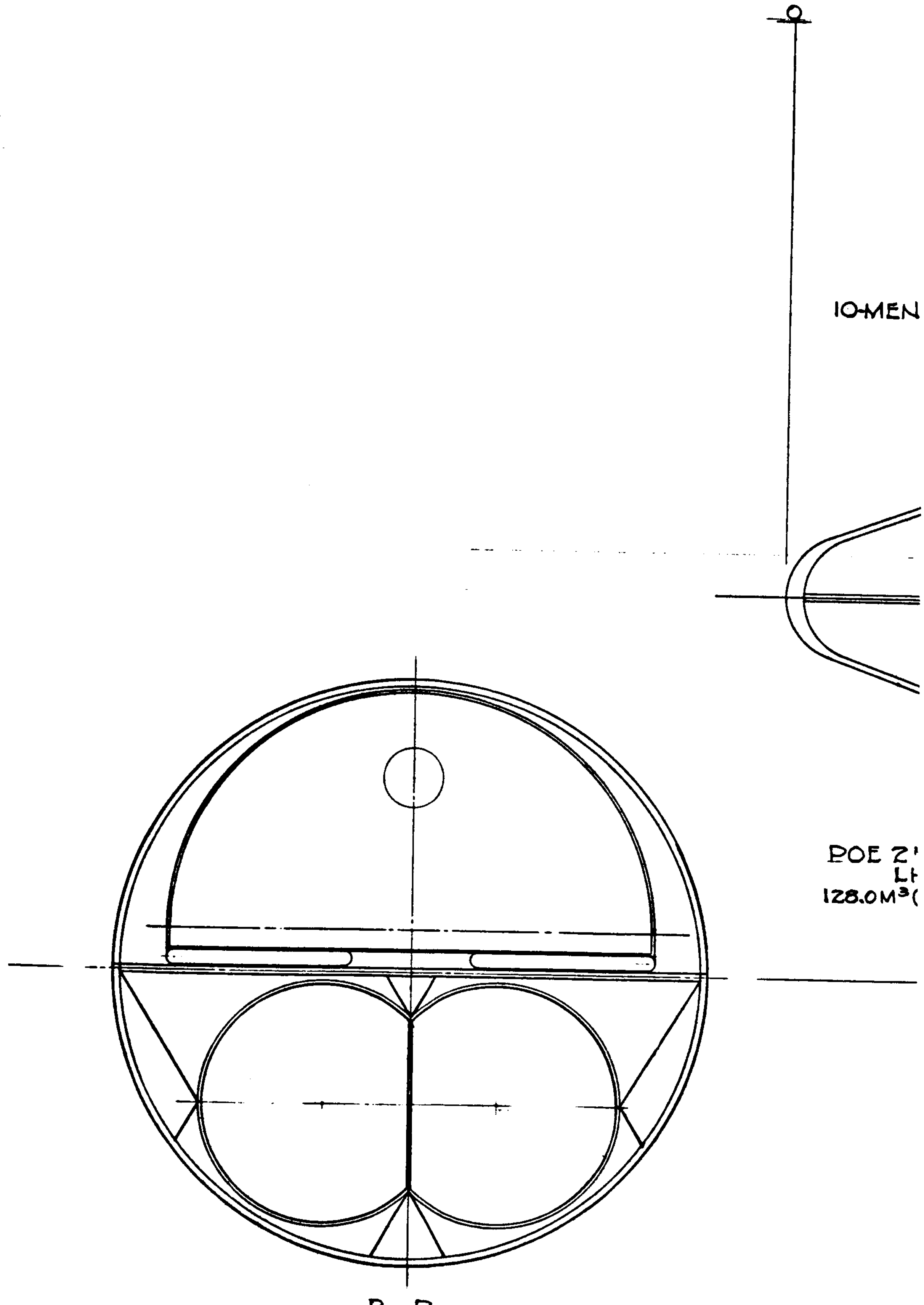
The first aerobraking spacecraft to be configured under this study is shown in Figure 14. This vehicle is a 14-man spacecraft utilizing cryogenic propellants for the major portion of its propulsive requirements. The basic vehicle is of biconic configuration with an overall length of 31.0 meters (1222 inches) and a base diameter of 17.7 meters (696 inches) with an estimated gross weight of 353,360 kilograms (777,000 pounds). The forward conic section has an angle of 20 degrees, which is established by the lifting body PEM integrated into the section. The aft body has an angle of 8 degrees, established by aerodynamic requirements during the aerobraking maneuver.

The aerobraking vehicle utilizes two of the modules that were developed in the early phases of the study. The MEM selected for integration is the

Table 1. Module Volumes

Item \ Mission Duration	14-Man Crew			20-Man Crew		
	500 days	1000 days	1500 days	500 days	1000 days	1500 days
Crew and crew support	135 ft^3	197 ft^3	260 ft^3	192 ft^3	281 ft^3	370 ft^3
Furniture and housekeeping	1,777	1,882	1,987	2,482	2,632	2,782
Food management	1,116	2,173	3,280	1,572	3,082	4,592
Water supply	5.5	5.5	5.5	7	7	7
Waste management	91.5	225	302	183	291	398
Temperature and humidity control	55	55	55	70	70	70
Atmospheric purification	16	21	26	19	24	29
Instrument and controls	3.5	3.5	3.5	4	4	4
Centrifuge	1,980	1,980	1,980	1,980	1,980	1,980
LSS gas storage	917	1,456	1,951	1,443	2,282	3,046
Storm cellar (125 ft^3/man)	1,750	1,750	1,750	2,500	2,500	2,500
Free volume (600 ft^3/man) [1]	8,400	8,400	8,400	12,000	12,000	12,000
Total	14,247 ft^3	16,148 ft^3	17,940 ft^3	22,452 ft^3	25,153 ft^3	27,778 ft^3

[1] Used for conceptual design purposes only.



B - B

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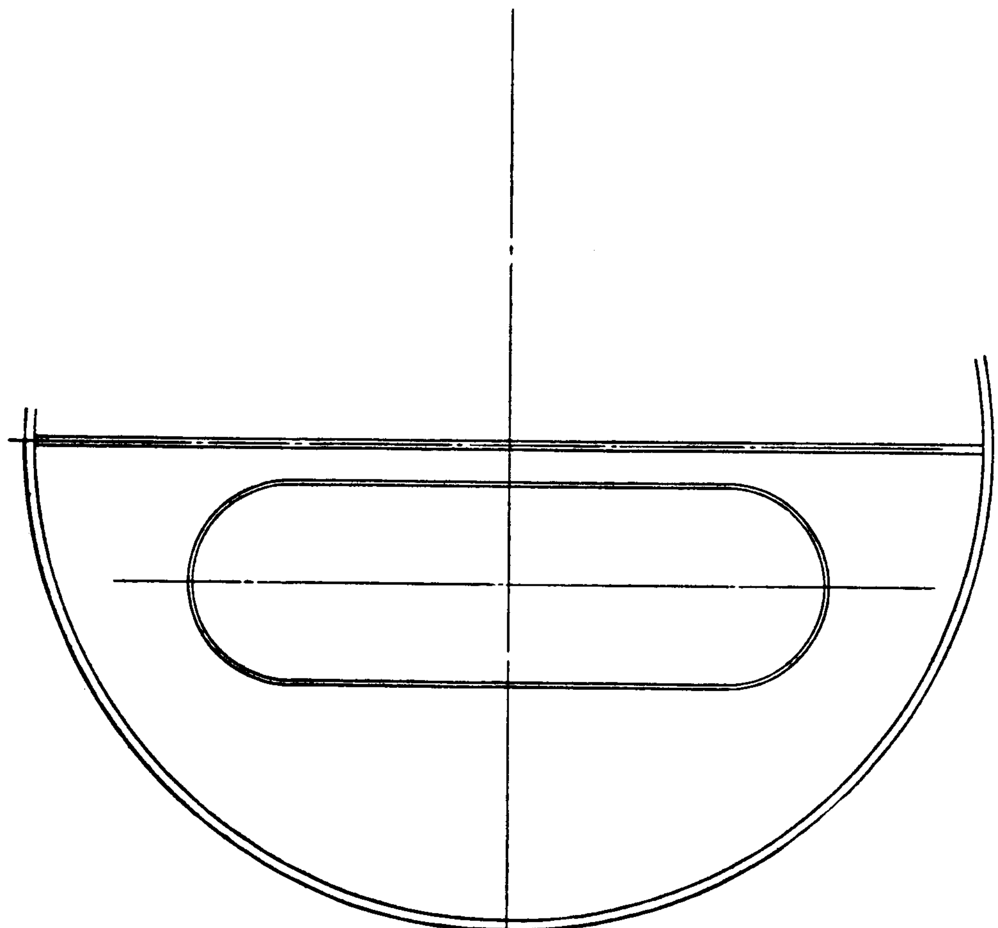

C-C

45P

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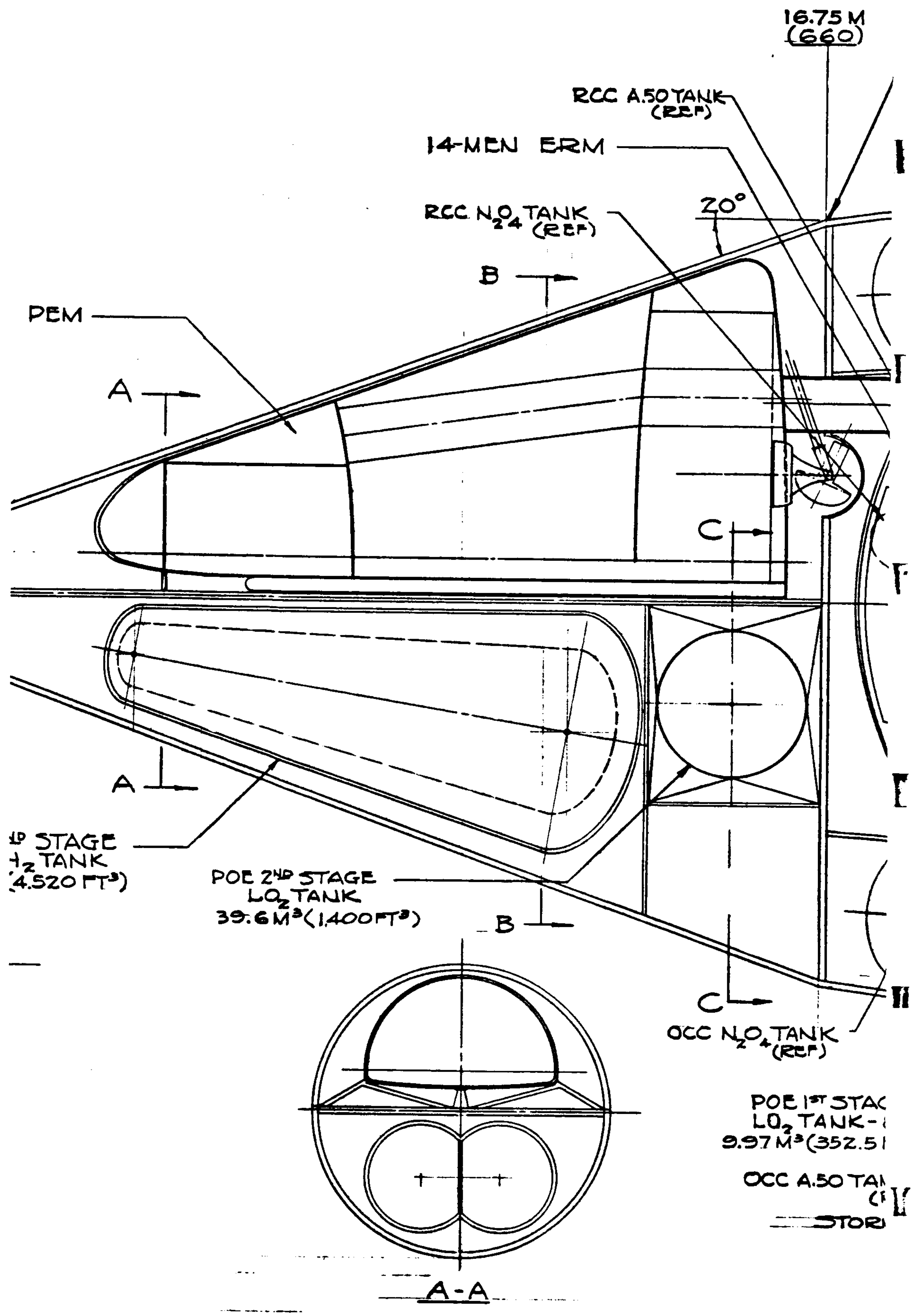

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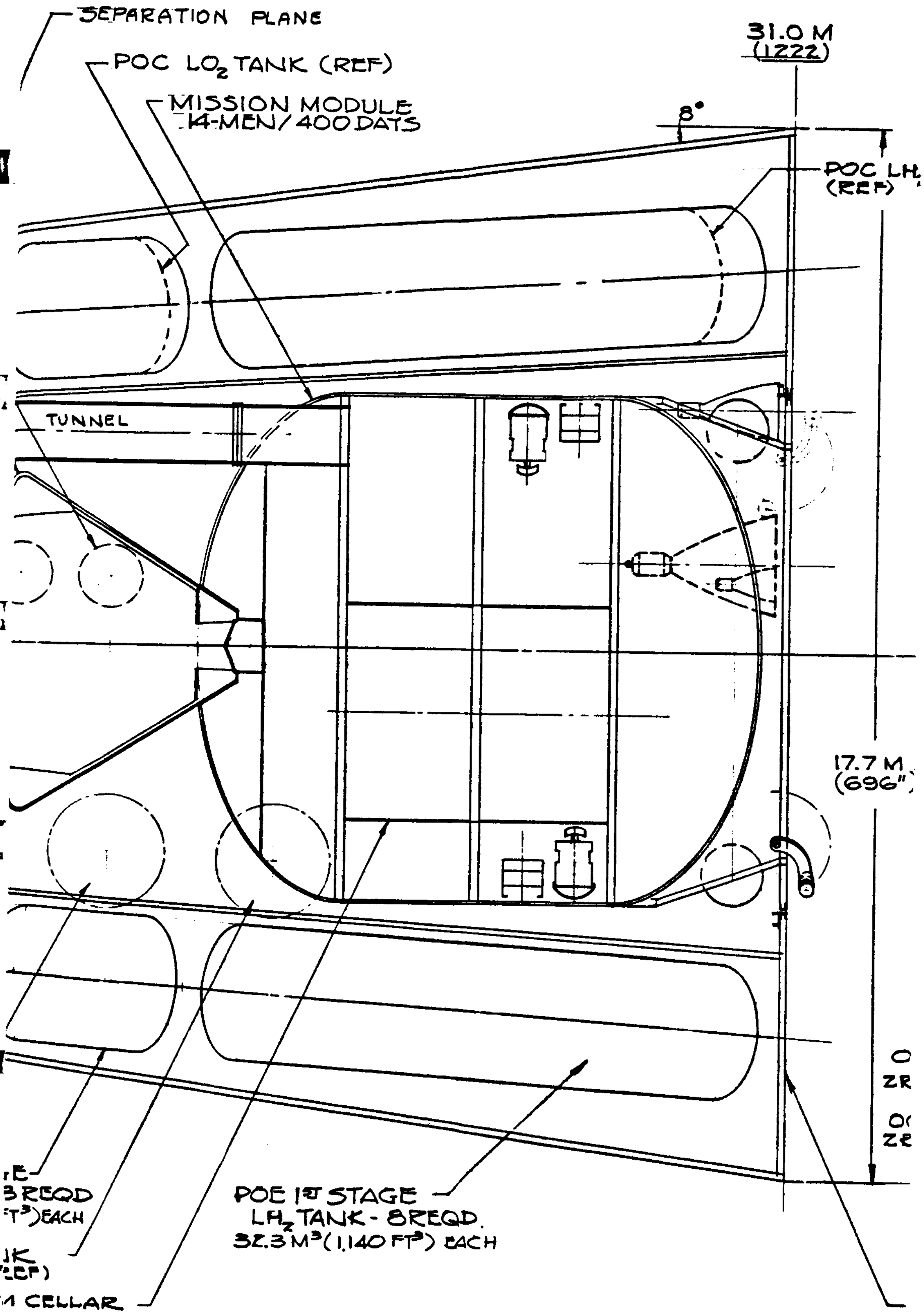
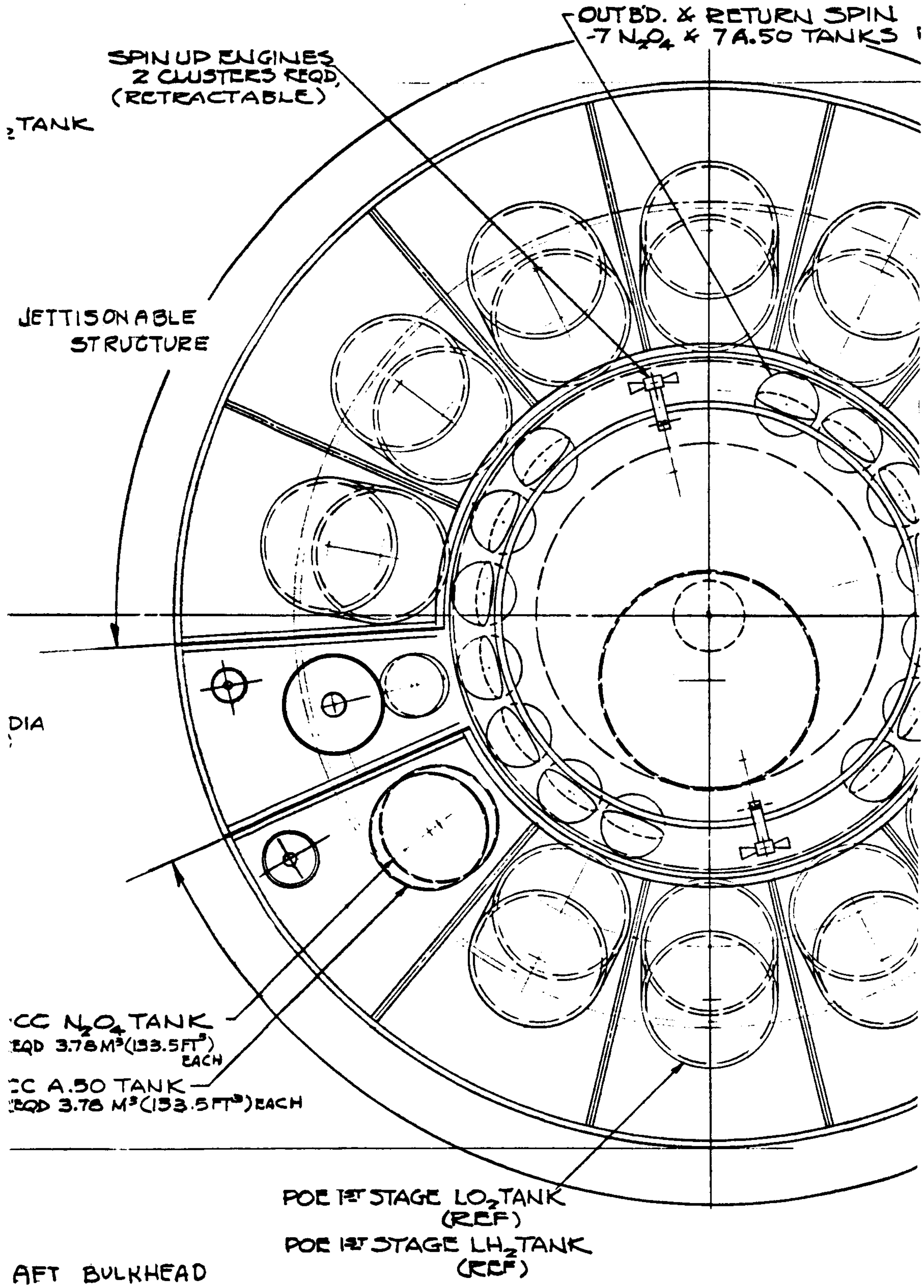


INBOARD PROFILE



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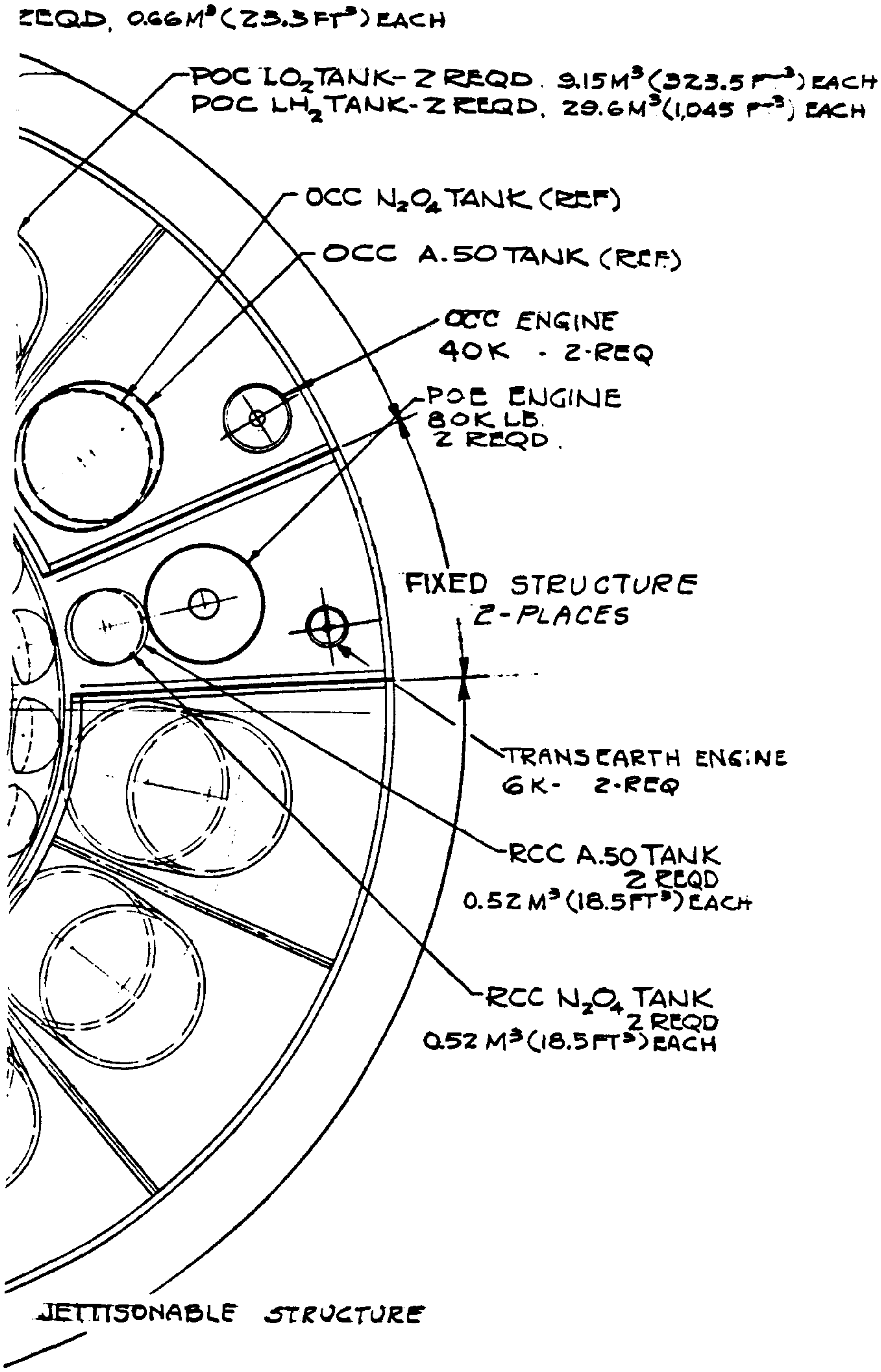

SPACECRAF

EEM ——
MM ——
MEM ——

STRUCTUR
HEAT SHIE

STORABLE

OCC	9.7
O-SPIN	5,1
RCC	1.5
R-SPIN	7

CRYOGEN

POC	18E
POE*1	82.5
POE*2	40.

OUTBOUND

PLANET ORI

TRANS EART

Figure 1-

45 E

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WEIGHT DATA

T GROSS WEIGHT	353,360 KG	777,400 LB
	5,600 KG	12,350 LB
	31,800 KG	70,000 LB
	60,500 KG	133,400 LB
E	16,400 KG	36,150 LB
D	41,000 KG	90,200 LB
(N_2O_4/A.50) PROPELLANT - MR = 1.6:1	27,465 KG	60,500 LB

N_2O_4		A.50		TOTAL
50 KG	21,450 LB	6,100 KG	13,450 LB	15,850 KG
10 KG	11,260 LB	3,190 KG	7,040 LB	8,300 KG
150 KG	2,950 LB	840 KG	1,850 LB	2,180 KG
'00 KG	1,540 LB	435 KG	960 LB	1,135 KG

IC (LO_2/LH_2) PROPELLANT - MR = 5:1	170,595 KG	375,600 LB

LO_2		LH_2		TOTAL
195 KG	41,500 LB	3,760 KG	8,300 LB	22,655 KG
500 KG	181,670 LB	16,500 KG	36,330 LB	99,000 KG
780 KG	89,830 LB	8,160 KG	17,970 LB	48,940 KG

GROSS WT.	353,360 KG	777,400 LB
BIT ESCAPE GROSS WT.	209,000 KG	460,000 LB
H GROSS WT	50,800 KG	112,000 LB

PRELIMINARY CONCEPT
Not Iterated for Weight Scaling

4. 14-Man Cryogenic Mars Aerobraking Spacecraft Concept

10-man lifting body vehicle shown in Figure 6 and previously discussed in this report. The 14-man mission module shown is 14-man, 500-day concept shown in Figure 13. This module is a two-floor level pressurized cylindrical can 8.47 meters (334 inches) in diameter, with an overall length of 9.43 m (372 inches). An Apollo-shaped ERM with a 14-man crew capacity was selected, although the 14-man biconic vehicle of Figure 2 would integrate equally well. The criteria for ERM selection would be primarily Earth entry speed.

In the conceptual development of an aerobraking spacecraft, several major design considerations must be concurrently exercised and evaluated to configure a valid design. Included in these are c.g./c.p. relationships, structural and propulsion system staging approach during the mission, and artificial gravity requirements.

For conceptual design purposes, it was assumed that the aerobraking spacecraft could be separated on a telescoping rail system. Such an approach could be utilized to provide a continuous artificial gravity environment during the mission. The entire vehicle is rotated during those phases of the mission during which the vehicle is not performing a propulsive or planetary aerobraking maneuver. The most efficient spacecraft arrangement which satisfies this requirement is one in which the mission module can be separated from the rest of the vehicle resulting in a minimum extension of the two. Even if an artifical gravity environment is not required, some separation (though of a smaller distance) will probably still be required to provide for the deployment of antennae, radiators, experiment equipments, etc.

The general arrangement of the aerobraking spacecraft features the 10-man MEM positioned in the upper half of the nose section of the spacecraft. The lower section is occupied by 54,300 kilograms (119,800 pounds) of LH_2/LO_2 stage 2 planet orbit escape propellant. Since the MEM weight is greater than the propellant weight in the nose, the aerobraker center of gravity would be offset in that direction and, consequently, that side of the vehicle would be at a positive angle of attack during entry. The 14-man Apollo shaped ERM is positioned midship, with the upper apex entry hatch and passageway nested in the end of the mission module. During the mission when the spacecraft mission module is extended from the spacecraft for artificial gravity spinning, the ERM could remain attached to the mission module or remain attached to the vehicle structure to maximize the mass at that end to reduce the separation distance between the two extended sections.

The mission module is located along the centerline of the spacecraft at the aft end of the aerobraker. In this position, the mission module is not obstructed by any other major elements of the vehicle and can easily be extended for artificial-gravity spinning.

The aft body structure features two fixed box sections that would contain the telescoping rail system used to extend the mission module during the spinning artificial-gravity mode. These two fixed sections also contain the two

36,000 kilogram (80,000 pound) -thrust planet orbit escape rocket motors, the trans-earth course correction propellant and the two 2700-kilogram (6000-pound) -thrust rocket motors provided for that purpose. The remainder of the aft body structure would be jettisoned after the planet orbit escape stage 1. The two jettisonable sections contain the outboard course correction propellants and propulsion systems, and the planet orbit escape stage 1 cryogenic propellants.

Deployment of the MEM is accomplished by jettisoning the section of the aerobraker nose structure that covers the MEM. Docking of the crew compartment of the MEM on return from the planet surface takes place at the interface provided at the end of the crew transfer tunnel which connected the mission module and PEM.

A mix of propellants is used for the cryogenic aerobraking concept. The spinning, outbound course correction, and return course correction propellants are all Aerozine 50/N_2O_4 storables. The main propulsive requirements of the vehicle — planet orbit circularization and the two stages of planet orbit escape — utilize LH_2/LO_2 propellants. To provide an acceptable c.g./c.p. relationship, the oxidizer tanks of the planet orbit escape stage 1 propellants have been placed as far forward as possible in the structural sections allocated for these propellants.

This aerobraking concept represents a "first-cut" configuration. It is recognized that certain improvements could be made to increase overall packaging efficiency if the concept was iterated. However, from size considerations, it is a valid general concept.

WEIGHT SCALING EQUATIONS

Modular weight scaling equations have been incorporated in the SD-developed Weight Synthesis Computer Program. The modules considered are the Earth reentry modules (ERM), mission modules (MM), planetary excursion modules (PEM), and the propulsion modules (PM) and aerobrakers. The methodology employed in the generation of the equations was to combine the data furnished by NASA/MAD and available at NR/SD and extrapolate these data to form the scaling equations.

The basic logic in forming weight scaling equations involves generating a weight statement of all applicable elements and systems as a function of the primary parameters. These elements are then combined to form the weight scaling equations. Assumptions required to form elements of the equations are noted.

EARTH REENTRY MODULE WEIGHT SCALING EQUATIONS

Earth reentry module weight equations have been developed for three configurations: biconic, segmented conic, and Apollo. The systems weight data are based on the data contained in References 3 and 4 and the ERM structural weight data contained in Reference 5. The referenced weight scaling equations reflect the studies based on crew sizes from six to ten men. The parametric considerations utilized in these equations are of sufficient flexibility to make them adequate for crew sizes of three to twenty men. The resultant scaling equations are summarized in Tables 2 through 5.

MISSION MODULE WEIGHT SCALING EQUATIONS

The mission module weight scaling equations are based on NASA/MAD crew support and life support subsystem weight data contained in Reference 2 and NR/SD structure, electrical power, reaction control system, and fixed subsystem weight data. A module subsystem summary is presented in Table 6. The structural weight is based on a cylindrical shape with either flat or elliptical bulkheads. The mission module sizing is based on volumetric requirements of crew, subsystems, number of floors, and bulkhead aspect ratio. Mission module life support subsystem weight, volume and power are shown in Table 7. A ten-percent contingency weight is included in the total module weight. The mission module scaling equations not shown in either Table 6 or 7 are summarized in the following subsections.

Table 2. Earth Reentry Module Subsystems Summary

Code	Systems	Notes	Scaling Equations (pounds)
W1	Crew	Reference 3	227 (crew)
W2	Heatshield	f(surface area, entry velocity, and W/CDA)	See text
W3	Structure	f(surface area and entry velocity)	See text
W4	Airlock	Data input (applicable to Apollo configuration only)	100.0
W5	Recovery system	5.25 percent of weight at entry	0.0525 W_{ent}
W6	Controls	Data input, Reference 3	270.0
W7	Guidance	Data input, Reference 3	300.0
W8	Life support system	Reference 3	85.25 (crew) + 221.5
W9	Electrical power system	Reference 3	11.75 (crew) + 588.5
W10	Communications	Data input, Reference 3	185.0
W11	Attitude control	6.75 percent of weight at entry	0.0675 W_{ent}
W12	Scientific payload	Data input, Reference 3 (used to size vehicle)	800.0
W13	Contingency	10.0 percent of weight at entry (0.100 W_{ent})	0.10 Σ (W 1 through W 13)
	Weight at entry		Σ(W 1 through W 13)

Table 3. Conic Configuration Scaling Equations

Parameter	Scaling Equations (pounds)	Constraints (feet/second)
Surface area	$SA = 6.56\ (VTOT)^{2/3}$	Ve = 35,000 to 50,000
	$SA = (9.226 - 5.33 \times 10^{-5}\ Ve)\ (VTOT)^{2/3}$	Ve = 50,000 to 65,000
Structural weight (Double wall structure)	$WS = 7.97\ (SA)$	Ve = 35,000 to 50,000
	$WS = (7.6433 + 6.533 \times 10^{-6}\ Ve)\ (SA)$	Ve = 50,000 to 65,000
Heat shield weight ($W/C_DA = 100\ lb/ft^2$)	$WHS = \left[3.40\ (1.083)^{\frac{Ve - 35000}{10000}}\right] (SA)$	Ve = 35,000 to 45,000
	$WHS = \left[3.68\ (1.2174)^{\frac{Ve - 45000}{10000}}\right] (SA)$	Ve = 45,000 to 55,000
	$WHS = \left[4.48\ (1.4286)^{\frac{Ve - 55000}{8000}}\right] (SA)$	Ve = 55,000 to 65,000
CDA calculations	$CDA = 1.88\ (VTOT)^{2/3}$	Ve = 35,000 to 50,000
	$CDA = (59.527 - 12.268 \log Ve)\ (VTOT)^{2/3}$	Ve = 50,000 to 54,000
	$CDA = (40.768 - 8.304 \log Ve)\ (VTOT)^{2/3}$	Ve = 54,000 to 60,000
	$CDA = (23.251 - 4.638 \log Ve)\ (VTOT)^{2/3}$	Ve = 60,000 to 65,000
W/CDA calculation	$W/CDA = WENT/CDA$	
Delta heatshield weight	$DHSWT = \left[5.277 \times 10^{-4}\ Ve - 21.371\right] (W/CDA - 100.0)$	

Table 4. Biconic Configuration Scaling Equations

Parameter	Scaling Equations (pounds)	Constraints (feet/second)
Surface Area	$SA = 5.44\ (VTOT)^{2/3}$	Ve = 35,000 to 50,000
	$SA = (4.407 + 2.066 \times 10^{-5}\ Ve)\ (VTOT)^{2/3}$	Ve = 50,000 to 65,000
Structural weight (Double wall structure)	$WS = 8.126\ (SA)$	Ve = 35,000 to 50,000
	$WS = (8.4793 - 7.066 \times 10^{-6}\ Ve)\ (SA)$	Ve = 50,000 to 65,000
Heat shield weight ($W/CDA = 100\ lb/ft^2$)	$WHS = (1.0 \times 10^{-4}\ Ve - 1.1)\ (SA)$	Ve = 35,000 to 40,000
	$WHS = \left[2.9\ (2.069)^{\frac{Ve - 40000}{16000}}\right] (SA)$	Ve = 40,000 to 56,000
	$WHS = \left[6.0\ (1.5167)^{\frac{Ve = 56000}{8000}}\right] (SA)$	Ve = 56,000 to 65,000
CDA calculations	$CDA = 1.71\ (TVOL)^{2/3}$	Ve = 35,000 to 50,000
	$CDA = \dfrac{1.71}{1.276^{\frac{Ve - 50000}{5000}}}\ (VTOT)^{2/3}$	Ve = 50,000 to 55,000
	$CDA = 21.888 - 4.335 \log Ve\ (VTOT)^{2/3}$	Ve = 55,000 to 58,000
	$CDA = 7.016 - 1.2126 \log Ve\ (VTOT)^{2/3}$	Ve = 58,000 to 65,000
W/CDA calculations	$W/CDA = WENT/CDA$	
Delta heatshield weight	$DHSWT = (5.789 \times 10^{-4}\ Ve - 24.05)\ (W/CDA - 100.0)$	

Table 5. Apollo Configuration Scaling Equations

Parameter	Scaling Equations (pounds)	Constraints (feet/second)
Surface area	$SA = 5.11\ (VTOT)^{2/3}$	Ve = 35,000 to 65,000
Structural weight (Double wall structure)	$WS = 8.50\ (SA)$	Ve = 35,000 to 65,000
Heatshield weight ($W/C_DA = 100\ lb/ft^2$)	$WHS = \left[1.8\ (1.50)^{\frac{Ve - 35000}{7000}}\right] (SA)$	Ve = 35,000 to 42,000
	$WHS = \left[2.7\ (3.374)^{\frac{Ve - 42000}{14000}}\right] (SA)$	Ve = 42,000 to 56,000
	$WHS = (7.625 \times 10^{-4}\ Ve - 33.6)\ (SA)$	Ve = 56,000 to 65,000
CDA calculation	$CDA = 1.77\ (VTOT)^{2/3}$	
W/CDA calculation	$W/CDA = WENT/CDA$	
Delta heatshield weight	$DHSWT = (5.433 \times 10^{-4}\ Ve - 19.016)\ (W/CDA - 100.0)$	

Table 6. Mission Module Subsystems Summary

Code	Systems	Notes	Scaling Equations (pounds)
W1	Structure	Reference 6	See text
W2	Meteoroid protection	See Appendix B	See text
W3	Solar radiation protection	See Appendix B	See text
W4	Crew and seats	Reference 2	See text (Table 4)
W5	Life support system	3 types	See text (Table 4)
W6	Electrical power system		See text
W7	Guidance and navigation	*Reference 6 & 7	1385
W8	Communications	*Reference 6 & 7	1200
W9	Scientific instruments	Reference 6 & 7	1580 + 500 (crew)
W10	Reaction control		See text
W11	Contingency	10 percent	$0.1 \sum_{i=1}^{10} W_i$

*Includes backup and emergency

Table 7. Mission Module Life Support Subsystems

Item	Open Life Support Subsystems	Regenerative Life Support Subsystems	Partial Regenerative Life Support (With H_2O Recovery Only) Subsystems
WEIGHT SCALING EQUATIONS, 1 TO 1500 DAYS (POUNDS)			
Crew and crew support	65.0 + 360.0 (crew) + 0.14 (crew)(days)	65.0 + 360.0 (crew) + 0.14 (crew)(days)	65.0 + 360.0 (crew) + 0.14 (crew)(days)
Life support			
Furniture and housekeeping	295.0 + 65.0 (crew) + 0.025 (crew)(days)	295.0 + 65.0 (crew) + 0.025 (crew)(days)	295.0 + 65.0 (crew) + 0.025 (crew)(days)
Leak and cabin pressure	Equation (A)	Equation (A)	Equation (A)
Instruments and controls	100.0	150.0	100.0
Life support requirements	442.0 + 307.0 (crew) + 0.2 (days) + 25.52 (crew)(days)	527.0 + 283.0 (crew) + 0 2 (days) + 2.03 (crew)(days)	522.0 + 387.0 (crew) + 0.2 (days) + 4.18 (crew)(days)
Equation (A)	Equation (A) $(\text{days})\left\{3 \times 10^{-4}\left(2\pi R_{mm}^2 + \frac{2V_{mm}}{R_{mm}}\right) + 2.0 + 2.056 \times 10^{-4} V_{mm}\right\}$		
VOLUME AND POWER CONSIDERATIONS FOR THE ABOVE TABULATED WEIGHTS			
Total volume ($feet^3$)	267.4 + 139.3 (crew) + 0.015 (days) + 0.9169 (crew)(days)	271.3 + 119.8 (crew) + 0.025 (days) + 0.2339 (crew)(days)	270.9 + 139.6 (crew) + 0.015 (days) + 0.518 (crew)(days)
Total power (kilowatts)	$\left\| 835 + 105 \text{ (crew)} \right\|$ /1000	$\left\| 860 + 400 \text{ (crew)} \right\|$ /1000	$\left\| 985 + 205 \text{ (crew)} \right\|$ /1000

Mission Module

Volumetric Requirement

$$V_{mm} = (\text{Crew}) \cdot V_f + V_e + V_{lss} + V_{sc}$$

$$V_{sc} = (\text{Crew}) \cdot V_{scf} \text{ if (crew)} < 4$$

and

$$= 500 + 50(\text{Crew} - 4) \text{ if (crew)} \geq 4$$

where:

V_{mm} = mission module volume ~ (feet^3)

V_{sc} = storm cellar volume ~ (feet^3)

Crew = crew size

V_f = free volume per man

= 750 ft^3/man (nominal) $(400 \leq V_f \leq 1200)$

V_e = equipment volume $= \sum_{i=6}^{9} \frac{W_i}{30}$

V_{lss} = crew and life support system volume (3 types)

(a) open system
(b) water recovery only
(c) water and oxygen recovery

} Discussed in Appendix C

V_{scf} = free volume per man, storm cellar

= 125 ft^3/man (nominal).

Structural Weight Determination (W_i)

Elliptical Bulkhead Geometry

$$R_{mm} = A + B - 5.25 \cdot AR \cdot N_f/3 \quad \text{[for 7-foot high ceiling]}$$

where:

$$A = \left(-\frac{b}{2} + \left(\frac{b^2}{4} + \frac{a^3}{27}\right)^{1/2}\right)^{1/3}$$

$$B = \left(-\frac{b}{2} - \left(\frac{b^2}{4} + \frac{a^3}{27}\right)^{1/2}\right)^{1/3}$$

$$a = -(5.25 \cdot AR \cdot N_f)^2/3$$

$$b = \frac{1}{27}\left(2\,(5.25 \cdot AR \cdot N_f)^3 - 6.44 \cdot AR \cdot V_{mm}\right)$$

AR = bulkhead aspect ratio

$$A_{mm} = \pi R^2 \left[2 + \frac{1}{AR\sqrt{AR^2 - 1}} \ln(2AR^2 + 2AR\sqrt{AR^2 - 1} - 1)\right] + 44RN_f$$

$$\left.\begin{aligned} R_{sc} &= \left(V_{sc}/(7\pi)\right)^{1/2} \\ A_{sc} &= 2\pi\, R_{sc}\,(R_{sc} + 7) \end{aligned}\right\}\ \text{7-foot high ceiling}$$

Flat Bulkhead Geometry

$$R_{mm} = \left(V_{mm}/(7\pi\ N_f)\right)^{1/2}$$

$$A_{mm} = 2\pi\, R_{mm}\,(R_{mm} + 7 \cdot N_f)$$

Structural Weight

$$W_1 = (UW)_{str} \cdot A_{mm} + (UW)_{sc} \cdot A_{sc}$$

where:

A_{mm} = mission modules surface area

A_{sc} = storm cellar surface area

R_{mm} = mission module radius

R_{sc} = storm cellar radius

N_f = number of floors (for 7-foot high ceiling)

$(UW)_{str}$ = mission module unit structural weight

$(UW)_{sc}$ = storm cellar unit structural weight

W_1 = structural weight

Meteoroid Protection System (W_2)

$$W_2 = \left[(UW)_{mp} - (UW)_{str}\right] \cdot A_{mm} \quad \text{if } (UW)_{mp} > (UW)_{str}$$

$$W_2 = 0 \quad \text{if } (UW)_{mp} \leq (UW)_{str}$$

where:

$(UW)_{mp}$ = mission module unit meteoroid protection weight determined from the scaling equations defined in Appendix B

Solar Radiation Protection System (W_3)

$$W_3 = \left[(UW)_{sr} - (UW)_{str} - (UW)_{sc}\right] \cdot A_{sc} \quad \text{if } (UW)_{mp} \leq (UW)_{str}$$

$$W_3 = (UW)_{sr} - (UW)_{mp} - (UW)_{sc} \cdot A_{sc} \quad \text{if } (UW)_{mp} > (UW)_{str}$$

$$W_3 = 0 \quad \text{if } W_3 < 0$$

where:

$(UW)_{sr}$ = storm cellar unit solar radiation protection weight determined from the scaling equations defined in Appendix B.

Electrical Power System (W_6)

$$kw = kw_{fix} + kw_{tc} + kw_{lss}$$

$$W_6 = (UW)_{eps} \cdot kw$$

where:

kw = total mission module electrical power requirement (kilowatts)

kw_{fix} = fixed housekeeping power requirement

kw_{tc} = telemeter/communication power requirement

kw_{lss} = crew and life support system power requirement

$(UW)_{eps}$ = mission module electrical power unit weight

W_6 = electrical power system weight

Reaction Control System (W_{10})

$$W_{10} = \frac{0.02\ W_{10} = 0.02 \sum_{i=1}^{9} W_i}{1 - 0.02 - 4.4\ (10^{-6}) \cdot \text{Days}_r}$$

Gross Mission Module Weight

$$W_{mm} = \sum_{i=1}^{11} W_i$$

PLANETARY EXCURSION MODULE WEIGHT SCALING EQUATIONS

Planetary excursion module weight equations have been developed for two classes of vehicles: aerobraking vehicles for Mars and retrobraking vehicles for Mercury, Ceres, Vesta and Ganymede. Each target has different structural/insulation criteria and, therefore, a different weight due to local environmental characteristics. All other scaling equations for the two classes of modules are assumed to be the same.

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 systems. The equations for the life support and electrical power systems are identical to those used in the earth reentry module. Sufficient equipment and propulsion for ascent, rendezvous, and docking with the parent spacecraft are also provided. The descent stage is composed of the systems required to land on the surface and the systems necessary to support the crew while on the planet or asteroid surface.

Common Systems

All systems components except the structure are considered to be the same for all vehicles. The common ascent stage systems, their primary characteristics and assumptions, and the associated weight equations are presented in Table 8. A contingency of ten percent of the gross manned capsule weight plus five percent for rendezvous and docking with the parent spacecraft are provided. The gross manned capsule is the manned capsule structure and all ascent stage systems except the ascent stage propellant. The common descent stage systems are presented in Tables 9 and 10.

Table 8. PEM Ascent Stage Common Systems

Code	Systems	Notes	Scaling Equations (pounds)
W1	Structure	See Table 11	
W2	Electrical power	ERM systems type	588.5 + 11.75 (crew)
W3	Electronic equipment	G&N, S&C (ascent)	570.0
W4	Reaction control system	3.1 percent of ascent stage weight	0.031 (ASWI)
W5	Crew and suits	173 pounds (95 percentile) + suits	227.0 (crew)
W6	Life support systems	1 day ECS/LSS	221.5 + 85.3 (crew)
W7	Scientific payload	Returned to ERM (surface samples)	800.0
W8	Propulsion system	8.5 percent of ascent W_{PA} + engine at F/WE = 60, F/Wo = 2.5 g's local	$0.085\ \nu_A\ (ASWI) + 0.0417\left(\frac{g_L}{g_\oplus}\right)(ASWI)$
W9	Rendezvous propulsion	5.0 percent of gross manned vehicle ΔV = 400 fps, I_{sp} = 315 and ν_A = 0.50	$0.050\ \Sigma$(W1 through W8)
W10	Ascent propellant (W_{PA})	ν_A (ASWI) where $\nu_A = W_{PA}/ASWI$ $\nu_A = 1.0 - 1.0/\exp\left[(\Delta V)AS/I_{sp(AS)}g\right]$	ν_A (ASWI)
W11	Contingency	10.0 percent of gross manned vehicle	$0.10\ \Sigma$(W1 through W8)
W32	Ascent propellant	ν_A (ASW2)	ν_A (ASW2)
	Ascent stage weight (ASWI)	(ASWI) Used to determine system weights but not W_{PA} in weight statement	Σ(W1 through W11)
	Ascent stage weight (ASW2)	(ASW2) Used to determine MSF weight and W_{PA} in weight statement	Σ(W1 through W9 + W11, W30, W32)

$$ASWI = \frac{1.15\ (W1) + 2387.0 + 372.66\ (crew)}{(1.0 - \ A) - (0.085\ \ A + 0.0356 + 0.0013\ (g_L)}$$

$$ASW2 = \frac{1.15\ (W1) + 2387.0 + 372.66\ (crew)}{(1.0 - \ A) - (0.085\ \ A + 0.0356 + 0.0013\ (g_L)}$$

Table 9. PEM Descent Stage Common Systems

Code	Systems	Notes	Scaling Equations (pounds)
W13	Electrical power	Isotopic thermoelectric	500 (kw) up to 120 days
W14	Electronic controls	Controls	50.0
W15	Instrumentation	Housekeeping and experimental	250.0 + 500.0 (crew)
W16	RCS system	1.5 percent of gross entry weight	0.015 (GEW)
W17	Life support systems	Open life support systems	See equations
W18	Payload	Samples picked up at target	0.0
W19	Descent propulsion	8.5 percent of descent W_P + engine at F/WE = 60, F/Wo = 1.5 g's local	$0.085\ W_{PD} + 0.025 \left(\frac{g_L}{g_\oplus}\right)$ (GEW)
W20	Landing gear	4.0 percent of gross landing weight	$0.040\ (GLW) \left(\frac{g_L}{g_\oplus}\right)$
W21	Contingency	10.0 percent of descent stage	0.10 Σ(W12 through W17)
W22	Descent propellant (W_{PD})	$\nu D = 1.0 - 1.0/\exp\left[(\Delta V)ds/(I_{sp})ds \cdot g_\oplus\right]$	ν_D (GEW)
W23	Stability drogue chute	Mission dependant* 1/4 percent of GEW	0.0025 (GEW)
Descent stage weight		DSW	Σ(W12 through W21 - W5)
Gross landed weight		GLW	ASW + DSW - W7
Gross entry weight		GEW	GLW + W22 + W23*
PEM planetary injection		PEMPI	GEW - W5

*For Mars aerobraking mission only

Table 10. PEM Life Support Subsystems

Item	Open Life Support Systems	Regenerative Life Support Systems	Partial Regenerative Life Support (With H_2O Recovery Only) Systems
WEIGHT SCALING EQUATIONS, 1 TO 90 DAYS (POUNDS)			
Crew and crew support	65.0 + 360.0 (crew) +0.14 (crew)(days)	65.0 + 360.0 (crew) + 0.14 (crew)(days)	65.0 + 360.0 (crew) + 0.14 (crew)(days)
Life support			
Furniture and housekeeping	295.0 + 65.0 (crew) + 0.025 (crew)(days)	295.0 + 65.0 (crew) + 0.025 (crew)(days)	295.0 + 65.0 (crew) + 0.14 (crew)(days)
Leak and cabin pressure	Equation (A)	Equation (A)	Equation (A)
Instruments and controls	100.0	150.0	100.0
Life support requirements	442.0 + 307.0 (crew) + 0.2 (days) + 25.52 (crew)(days)	527.0 + 283.0 (crew) + 0.2 (days) + 2.03 (crew)(days)	522.0 + 387.0 (crew) + 0.2 (days) + 4.18 (crew)(days)
Equation (A)	Equation (A) Aerobraking: $\left[16.5 \times 10^{-4} \left[250\ (\text{crew})\right]^{2/3} + 2.0 + 0.0514\ (\text{crew})\right]$ (days) Retrobraking: $\left[17.4 \times 10^{-4} \left[1.2\ (FV)_{DS}(\text{crew})\right]^{2/3} + 2.0 + 2.466 \times 10^{-4} (FV)_{DS}(\text{crew})\right]$ (days)		
VOLUME CONSIDERATIONS FOR THE ABOVE TABULATED WEIGHTS			
Total volume (feet^3)	267.4 + 139.3 (crew) + 0.015 (days) + 0.9169 (crew)(days)	271.3 + 119.8 (crew) + 0.025 (days) + 0.2339 (crew)(days)	270.9 + 139.6 (crew) + 0.015 (days) + 0.518 (crew)(days)

Structure

A summary of the structural weight equations for the ascent stage (W_1) and the descent stage (W_{12}) is presented in Table 11. The structural weight is added to the common systems weight to obtain the gross planetary excursion module weight. Removal of crew and scientific payload represents the planetary excursion module weight at Earth orbit escape.

PROPULSION STAGE WEIGHT SCALING EQUATIONS

Stage weight scaling equations have been established using the shell weight and engine weight characteristics supplied by NASA and modified by SD. The scaling equations are summarized in Table 12. A variable (K_T) has been included in the shell weight equation to account for differences in the structural model. In the chemical engine weight equation, the engine thrust-to-weight ratio ( ) is obtained from a curve fit of the data contained in Reference 7. Also, a coefficient (K) has been included to reflect the effect of engine type on engine weight. The effects of finite burning have been accounted for in the performance calculations by including emperical equations provided by NASA. These equations yield the gravity loss as a function of initial thrust-to-weight ratio, specific impulse, hyperbolic excess speed, and orbit altitude for each of the target bodies considered in the study.

Shell Weight (W_1)

A propulsion tank and system scaling equation has been established using the shell weight characteristics supplied by NASA and modified by S&ID to include a coefficient to account for the structural model. The propulsion tank and systems equations are summarized as follows:

$$W_1 = 0.11\, K_T \left[W_p^{0.9} / \left(\frac{\rho_p}{62.4}\right)^{0.533} \right] + 5732;\ \text{lb}^1$$

W_p = propellant weight

ρ_p = bulk density

$$K_T = 0.8034 + 0.1184 \times 10^{-5}\, W_p - 0.730 \times 10^{-12}\, (W_p)^2 \qquad \text{For } W_p = 10^4 \text{ to } 10^6 \text{ lb}$$

$$K_T = (W_p)^{0.08312} / 2.5004 \qquad \text{For } W_p = 10^6 \text{ to } 10^8 \text{ lb}$$

where K_T accounts for the propulsion module structural model and is based on previous unpublished NR studies. This model accounts for the installation of thermal and meteoroid protection to the basic structure.

[1] Note that the total weight of a cluster of n tanks becomes $W_{1_n} = W_1 n^{0.1}$

Table 11. Structural Weight—Ascent and Descent

Mode (Target)	Notes	Scaling Equations (pounds)
Aerobraking (Mars)		
Apollo shape		
Ascent stage (W1)	Empirical	$1050.0 + 208\ (\text{crew})^{1.1}$
Descent stage (W12)	(W/C_LA) and (C_L) given	$\left[6.25/(W/C_LA)(C_L)\right]$ (GEW)
Lifting body (Aeronutronics) shape		
Ascent stage (W1)	Empirical	$1750.0 + 140\ (\text{crew})^{1.2}$
Descent stage (W12)	Empirical (W/C_LA) and (C_L) given	$\left[17.36/(W/C_LA)(C_L)\right]$ (GEW)
Retrobraking		
Ascent stage (W1)	Empirical (UW) AS given	$(UW)\ AS\ (5.1 - 0.1\ \text{crew})^{*}$ $\left[(84.3-3.3\ \text{crew})\ \text{crew}\right]^{2/3}$
Descent stage (W12)	Empirical $(UW)_{DS}$ and $(FV)_{DS}$ given	$(UW)_{DS}\ 5.82\ \left[1.2\ (FV)_{DS}\ (\text{crew})\right]^{2/3}$

Table 12. Propulsion Modules Summary

Code	Systems	Notes	Scaling Equations (pounds)
W_1	Propulsion tank and systems	For K_T see text	$0.11\ K_T \left[W_p^{0.9} / (\rho_p/62.4)^{0.533} \right] + 5732$
W_{2j}	Engine selection		
J=1	Chemical	For K and τ see text	$K\left(\frac{T}{\tau} + 99.2\right) N$
J=2	Solid core nuclear		$0.129T + 7298$
J=3	Gaseous core nuclear		$Wo\ (T/Wo)/(T/We)$
W_3	Interstage		$(UW)_{INT}\ (A_{IS})$
W_4	Meteoroid protection		$(UW)_{MP}\ (A_S + A_{IS})$
W_5	Insulation		$(UW)_{INS}\ (A_S)$
W_6	Propellant		νW_o
	Propulsion module weight		$\Sigma\ (W_1,\ W_{2j},\ W_3$ through $W_6)$

Engine Weight (W_{2j})

The engine equations are summarized as follows, together with their prime parameters.

Chemical Engine (J = 1)

$$W_{2J} = K\left(\frac{T}{\tau}\right) + 99.2 \quad N$$

The coefficient (K) is given by the following expressions which are emperical equations of the curves shown in Appendix C.

$$\left.\begin{array}{l} K = (0.3325)\, T^{0.0884} \text{ for } T = 2 \times 10^{4} \text{ to } 1 \times 10^{5} \text{ pounds} \\ K = (0.7250)\, T^{0.0207} \text{ for } T = 1 \times 10^{5} \text{ to } 1 \times 10^{7} \text{ pounds} \end{array}\right\} \begin{array}{l}\text{For all} \\ \text{propellant} \\ \text{types}\end{array}$$

The engine thrust-to-weight ratio (τ) is defined by the following equations for storable propellants. These equations were developed from the curves shown in Appendix C.

$$\tau = (4.302)\, T^{0.262} \text{ for } T = 2.0 \times 10^{4} \text{ to } 5.5 \times 10^{4}$$

$$\tau = (1.929)\, T^{0.335} \text{ for } T = 5.5 \times 10^{4} \text{ to } 1.5 \times 10^{5}$$

$$\tau = (7.660)\, T^{0.2197} \text{ for } T = 1.5 \times 10^{5} \text{ to } 4.4 \times 10^{5}$$

$$\tau = (45.26)\, T^{0.0830} \text{ for } T = 4.4 \times 10^{5} \text{ to } 2.2 \times 10^{6}$$

$$\tau = (101.07)\, T^{0.0279} \text{ for } T = 2.2 \times 10^{6} \text{ to } 1.0 \times 10^{7}$$

In a similar manner, the engine thrust-to-weight ratio is defined by the following equations for LO_2/LH_2 propellant.

$$\tau = (14.63)\, T^{0.1373} \text{ for } T = 2 \times 10^{4} \text{ to } 4 \times 10^{5}$$

$$\tau = (32.074)\, T^{0.0765} \text{ for } T = 4 \times 10^{5} \text{ to } 1 \times 10^{7}$$

Solid Core Nuclear Engine (J = 2)

$$W_{2J} = 0.129\, T + 7298$$

During the examination of common propulsion modules (discussed in the Mission/System section of this Appendix), fixed engines were assumed with thrusts of 75, 000 and 250, 000 pounds. The resultant engine weights are 16, 973 and 39, 548 pounds, respectively.

Gaseous Core Nuclear Engine (J = 3)

$$W_{2J} = W_o\ (T/W_o)/(T/W_e)$$

T/W_e = pounds of thrust per pound of engine

Engine Common Equations

$$T/W_o = I_{sp}\ \nu/t_B$$

$$T = W_o\ (T/W_o)/N$$

$$\nu = 1 - \frac{1}{e^{(\Delta v + dV)}}$$

$W_o = W_{pl}/1 - \nu/\nu_B$ = propulsion module gross weight

$W_p = \nu W_o$ = propellant weight = W_6

ν_B = stage mass fraction = 0. 99 to start iteration loop

dV = velocity loss due to gravitation field at planet location (NASA supplied)

t_B = stage propellant burning time

N = number of engines

Surface Area

The tank surface area is based on the propellant weight (W_p) and bulk density (ρ) with 10 percent added to account for ullage and tank bulkheads. The stage diameter (D_s) or the length to diameter ratio (L/D) is required to compute the surface area (A_s), as noted below in the following equations.

$$V_s = 1.10\ W_p/\rho_p = \text{Stage Tank Volume}$$

$$D_s = \left[V_s/0.7854\ (L/D)\right]^{1/3}$$

$$L/D = \left(V_s/0.7854\ D_s^{\ 3}\right)$$

$$A_s = \pi\ (D_s)^2\ (L/D)$$

Interstage Weight (W_3)

Interstage weight is characterized by the type of engine system employed in the propulsion module. The relationships for the lengths, surface area, and weight for the various interstages are expressed parametrically as follows.

Chemical Engine

$$L_{IS} = (0.94 - 0.07\ N)\ D_s = \text{length of interstage}$$

$$A_{IS} = (L_{IS})\ \pi\ D_s = \text{area of interstage}$$

$$W_3 = A_{IS}\ (UW)_{ce} = \text{weight of interstage}$$

Solid Core Nuclear Engine

$$L_{IS} = 0.648 \times 10^{-5} T + 28.513$$

$$A_{IS} = (L_{IS})\ \pi\ D_s$$

$$W_3 = A_{IS}\ (UW)_{sne}$$

Gaseous Core Nuclear Engine

$$L_{IS} = 0.680 \times 10^{-5} T + 29.938$$

$$A_{IS} = (L_{IS})\ \pi\ D_s$$

$$W_3 = A_{IS}\ (UW)_{gne}$$

(UW) = weight per foot squared for the interstage under consideration

= 2 lb/ft^2 (nominal)

Meteoroid Protection (W_4)

The meteoroid protection system weight is the sum of the tank and interstage surface area times a unit weight [$(UW)_{MP}$] obtained from the meteoroid protection scaling equations (Appendis B). The meteoroid protection weight, if required, represents an incremental weight over the basic structural weight.

$$W_4 = (UW)_{MP}(A_S + A_{IS}) = \text{weight meteoroid protection}$$

Thermal Protection (W_5)

The insulation weight (W_5) is the product of the insulation density, insulation thickness, and insulated area. As discussed in Appendix B, a nominal density of 5 lb/ft^3 was assumed during this study. The insulation thickness is obtained from the optimization discussed in Appendix B. Therefore,

$$W_5 = \left(\rho_{INS}\right)\left(d_{OPT}\right)\left(A_S\right) = \text{weight of insulation}$$

Propulsion Module Gross Weight (W_{STG})

The gross weight is the sum of the preceding elements, i.e.,

$$W_{STG} = W_1 + W_{2J} + \sum_{i=3}^{6} W_i; \; (J = 1,\ 2,\ \text{or}\ 3)$$

AEROBRAKER MODULE WEIGHT SCALING EQUATIONS

The aerobraker shroud weight is determined by first sizing the modules within the shroud using the previously defined scaling equations. The volume of the individual modules and the accumulated total volume within the shrouded configuration is derived. A packaging factor is applied to the total volume to permit for non-ideal configuration arrangement. The surface area of the aerobraking shroud configuration is determined and evaluated as structural and ablator weight. The ablator determination varies with planetary entry velocity.

A portion of the structural shroud, ablator and heat shield is ejected prior to ignition of the planetary orbit escape propulsion stages; whereas, some of the ablator and meteoroid protection is burned off during planetary entry. These losses during the mission made it necessary to modify the propulsion module sizing routine for proper payload weight determination. These effects are represented in the Weight Synthesis computer program as percent of the total weight of the structural shroud, ablator and meteroid protection at the time the loss occurs.

The shroud sizing parameters are summarized in Table 13.

Table 13. Aerobraker Module Summary

Thermal Protection	Thermal Factor = EKHS = $0.0796(1.1809)^{\frac{PEV - 26000}{4000}}$ = EKHS = $0.0940(3.7234)^{\frac{PEV - 30000}{16000}}$ Thermal Weight = EKHS (Wt of modules within shroud)	PEV < 30000 ft/sec PEV > 30000 ft/sec
Structure	Structural Area (SA) If Shroud L/D Given: $D = [V_{Total}/(0.424(L/D) - 0.04)]^{1/3}$ If Shroud D Given: $L/D = [(V_{Total}/D^3) + 0.04]/0.424$ Compute $D = D_{PEM}$ = PEM Diameter + 2.5 $SA = \left\lvert 0.0728 + 0.559\,[3.52(L/D)^2 - 0.625(L/D) + 1.0]^{1/2} + 0.972\,[1.49(L/D)^2 - 0.298(L/D) + 1.0]^{1/2} \right\rvert D^2$ Structure Weight = (SA) (Unit Structural Weight)	Determine D for PEM and compare with computed shroud diameter. Larger of the two is used for surface area determination.
Meteoroid Protection	$W_{MP(Shroud)} = SA(UW_{MP(Shroud)})$ $W_{MP(Bulkhd)} = SA(UW_{MP(Bulkhd)})$	UW_{MP} determined from scaling equations defined in Appendix B.

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WEIGHT SYNTHESIS METHODOLOGY

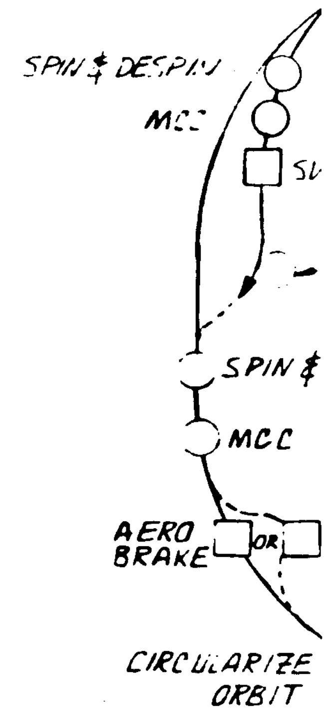
The planetary weight synthesis program is a computer tool, developed by NAA-SID on company funds, to provide a simplified parametric approach for deriving planetary vehicle, module, and subsystem weight data.

The program is designed to compute the weight of any configuration for designated mission modes as a function of available subsystem and modular criteria. Figure 15 presents a flow diagram of the program logic which provides alternate capability of selecting fixted data inputs, weight scaling equations, look-up weight tables, or complete weight synthesis. The mission sequence diagram shown on Figure 15 illustrates the various points in the mission for which weight computations are required, and the alternate modes of computation. Capability is provided in the program for deriving the weight of an Earth reentry module, Earth retro stage, mission module, planetary excursion module, and various propulsion stages for midcourse corrections, spin and despin of the spacecraft (if required), swingby inbound or outbound, planetary orbit escape, planetary orbit insertion and Earth orbit escape.

Weight synthesis is accomplished by selecting the basic mission parameters of target, mission mode, mission class, orbital stay time, and opportunity. The mission mode defines the various modules required to perform a specific mission. The necessary input parameters are then determined for each of the basic modular routines defined by the equations in the previous section . The weight computing process is developed in reverse to that of the mission sequence. Starting with the Earth reentry module computed weight, and following through the selection routine in the flow diagram of Figure 15, each required module is added in turn, and this accumulated total is treated as payload to the next stage for sizing the propulsion systems. The configuration weight in-Earth-orbit is finally determined when the Earth orbit escape stage is computed and added to the previously accumulated total.

Vehicle weights can be derived for various planetary mission modes, including direct lander, direct orbiter, outbound swingby lander or orbiter, inbound swingby lander or orbiter, and planet flyby. The capability of performing parametric weight sensitivities is inherent in the program by varying one or more parameters while holding all others constant.

PLANE
SYNT



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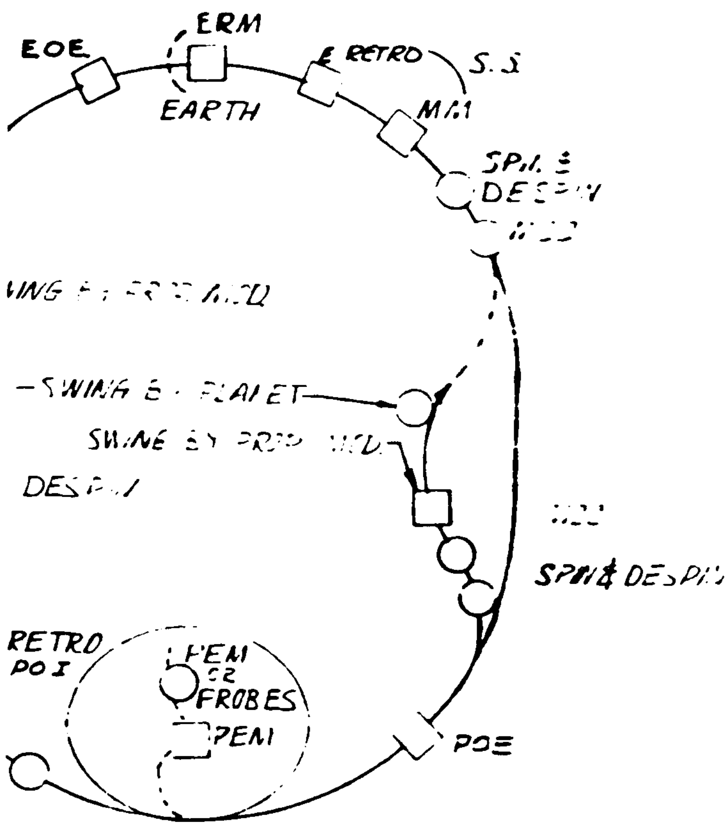
TARY MISSION WEIGHT
HESIS PROGRAM LOGIC



INDICATOR NAME
ERM
ERETRO
EMM
SPIN 4
EMCC 4
PROBE H
EMCC 3
SPIN 3
POE
PEM
PROBEM
BRAKE
EMCC 2
SPIN 2
PROBE O
EMCC 1
SPIN 1
EOE
SBPM 1
SBPM 2

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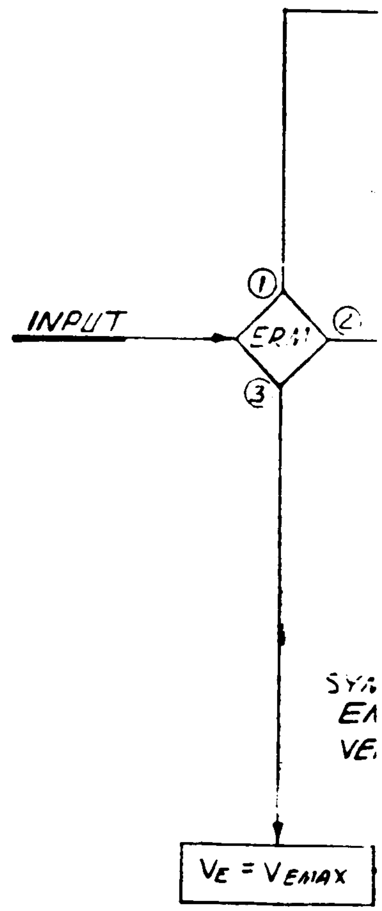
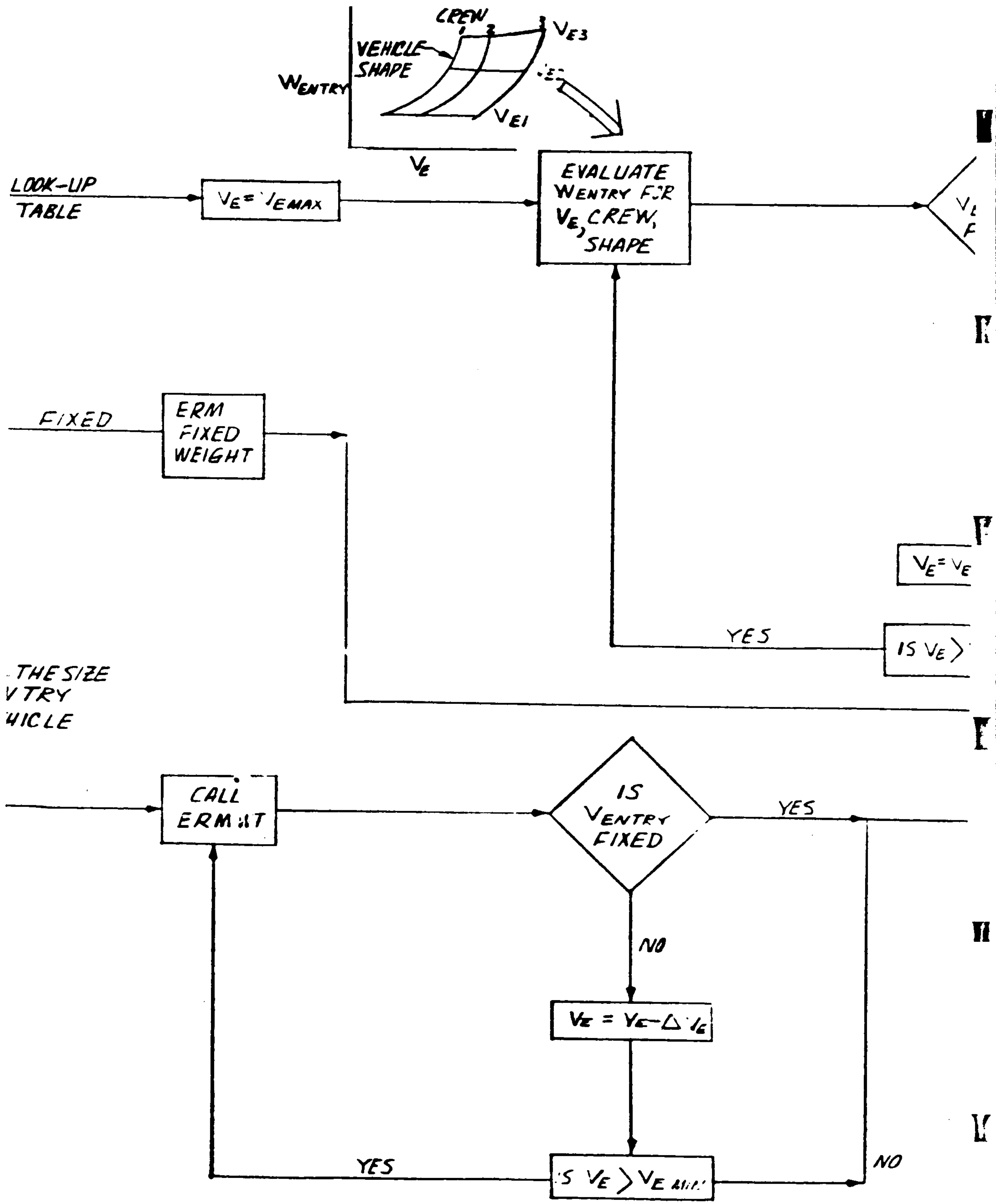
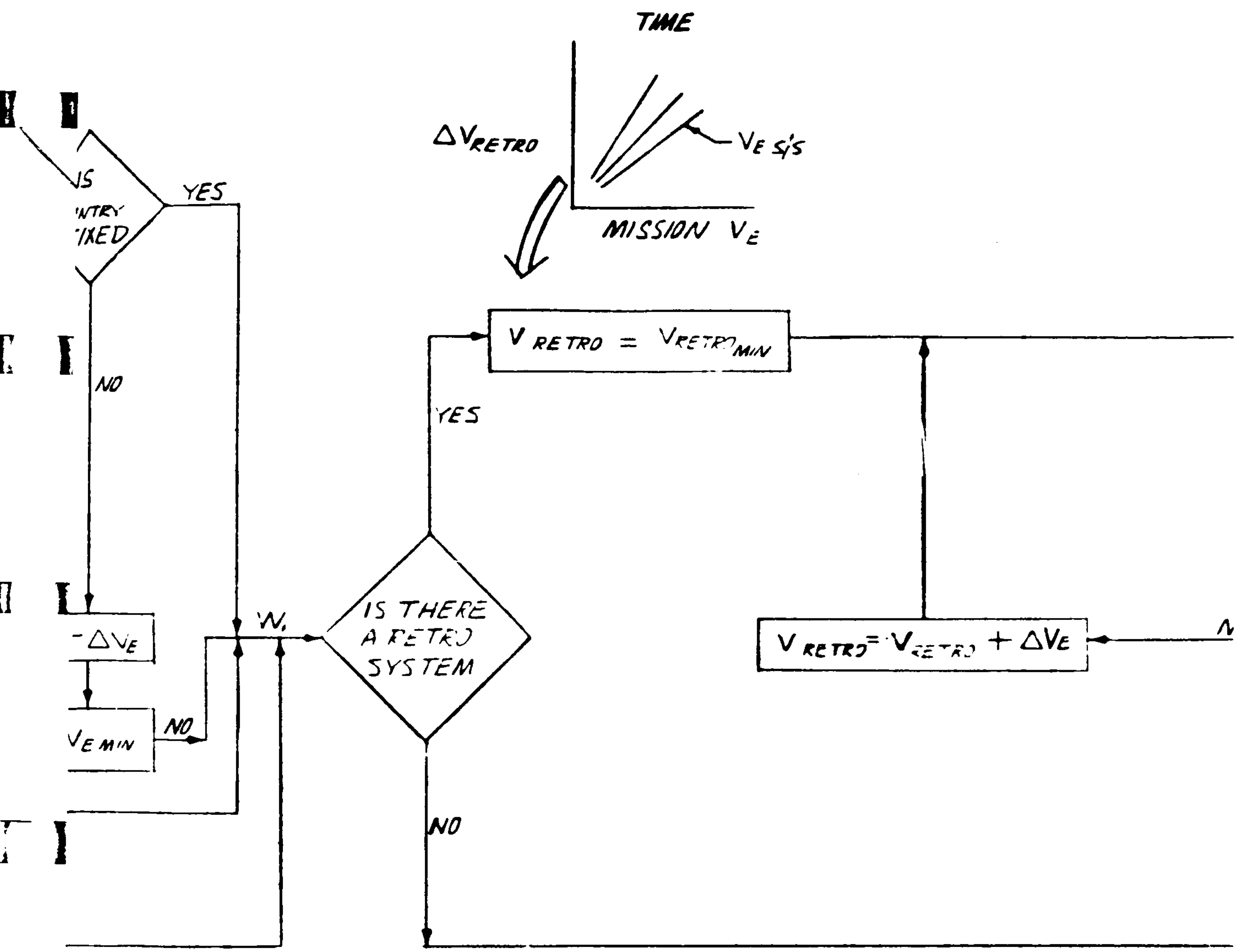
OMENCLATURE

MODULE DESCRIPTION
EARTH ENTRY MODULE
EARTH RETRO STAGE
MISSION MODULE
SPIN & DESPIN FINAL RETURN
MIDCOURSE CORRECTION FINAL RETURN
PROBES RELEASE INBOUND SWING-BY
MIDCOURSE CORRECTION INITIAL RETURN
SPIN & DESPIN INITIAL RETURN
PLANET ORBIT ESCAPE
PLANET EXCURSION MODULE
PROBES RELEASE AT PLANET
PLANET BRAKING STAGE RETRO-AERO
MIDCOURSE CORRECTION FINAL OUTBOUND
SPIN & DESPIN FINAL OUTBOUND
PROBE RELEASE OUTBOUND
MIDCOURSE CORRECTION INITIAL OUTBOUND
SPIN & DESPIN INITIAL OUTBOUND
TRANS PLANET STAGE EARTH ORBIT ESCAPE
SWING BY PROP. MODULE
SWING BY PROP. MODULE




CREW
VEHICLE SHAPE
W ENTRY
V E3
V E1
V E
LOOK-UP TABLE
V E = V E MAX
EVALUATE W ENTRY FOR V E, CREW, SHAPE
FIXED
ERM FIXED WEIGHT
V E = V E
YES
IS V E >
THE SIZE
TRY
ICLE
CALL ERMAT
IS V ENTRY FIXED
YES
NO
V E = V E - Δ V E
YES
IS V E > V E MIN
NO



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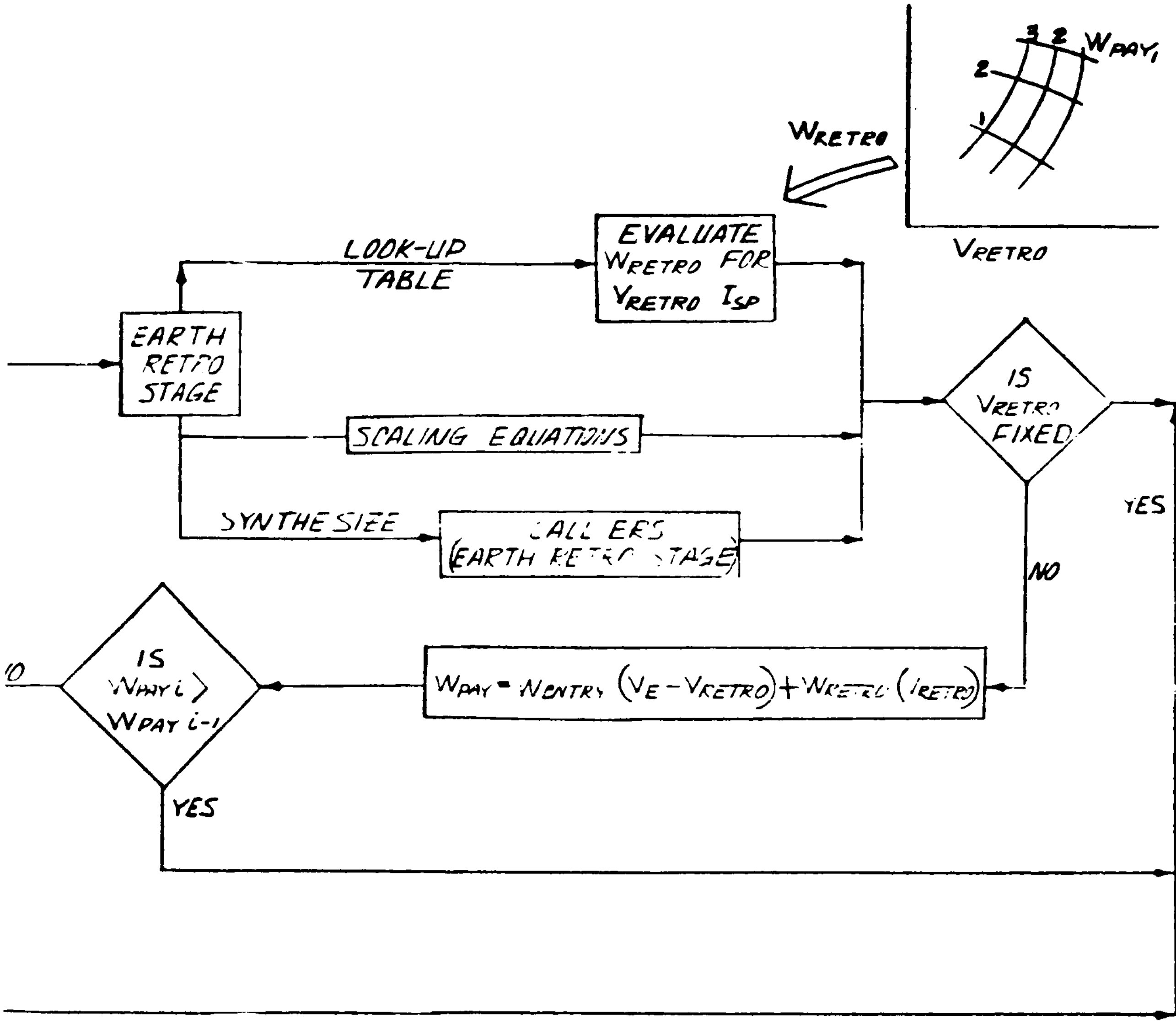

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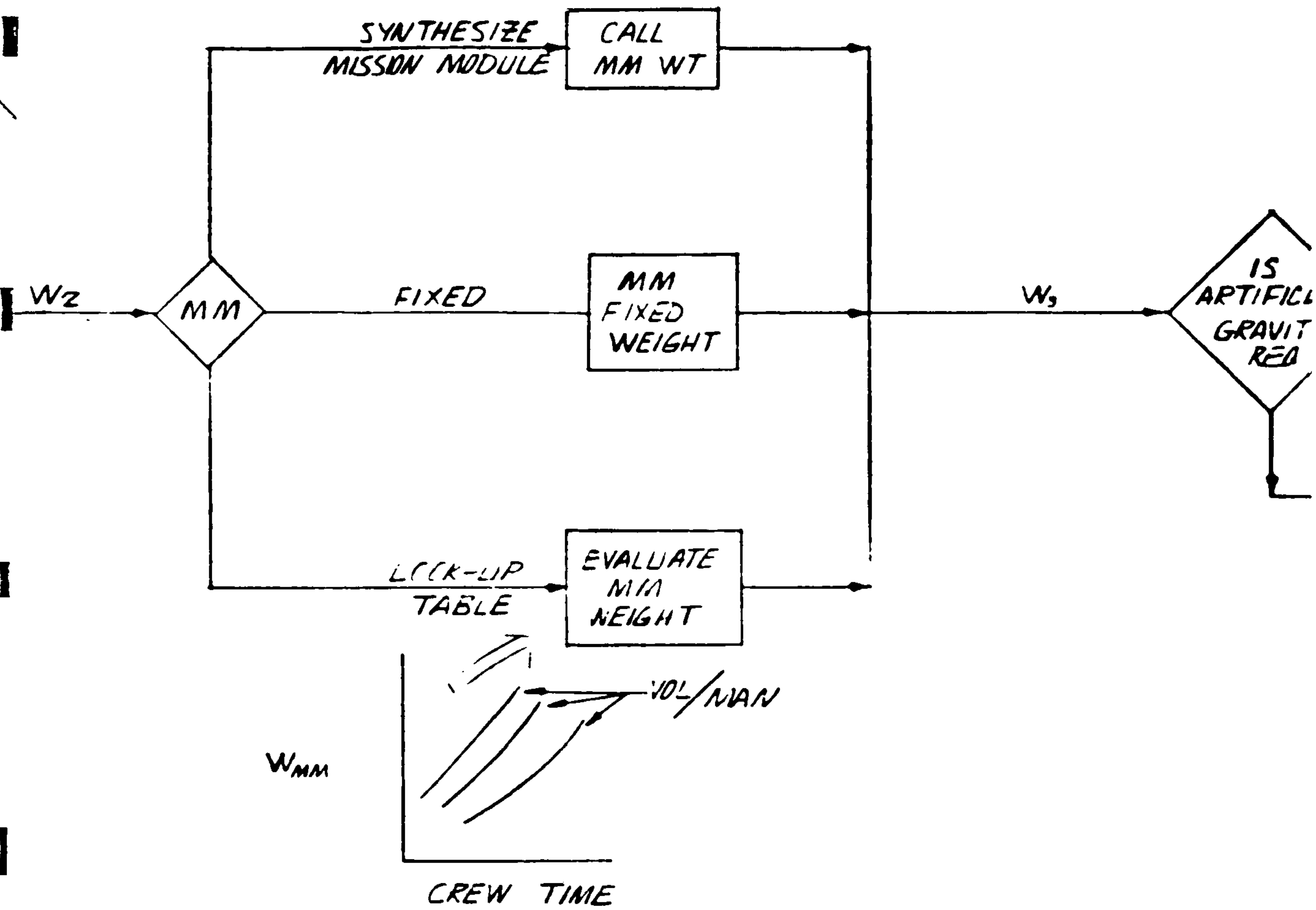

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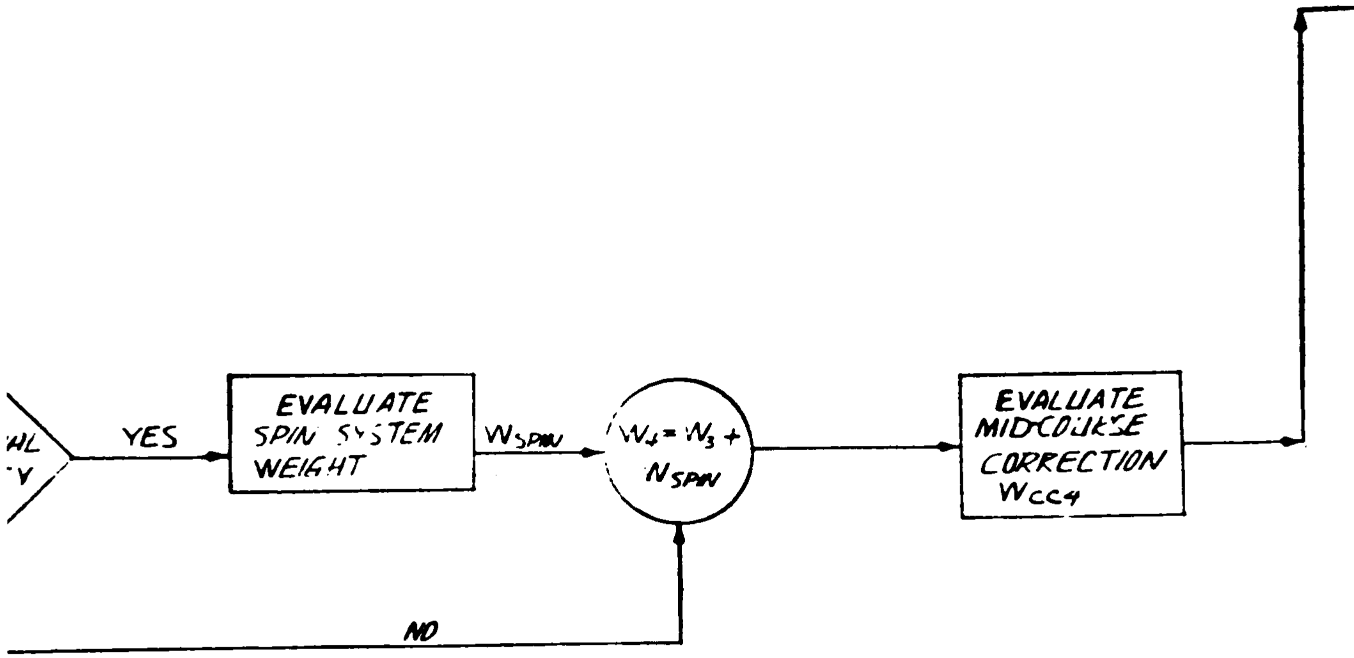

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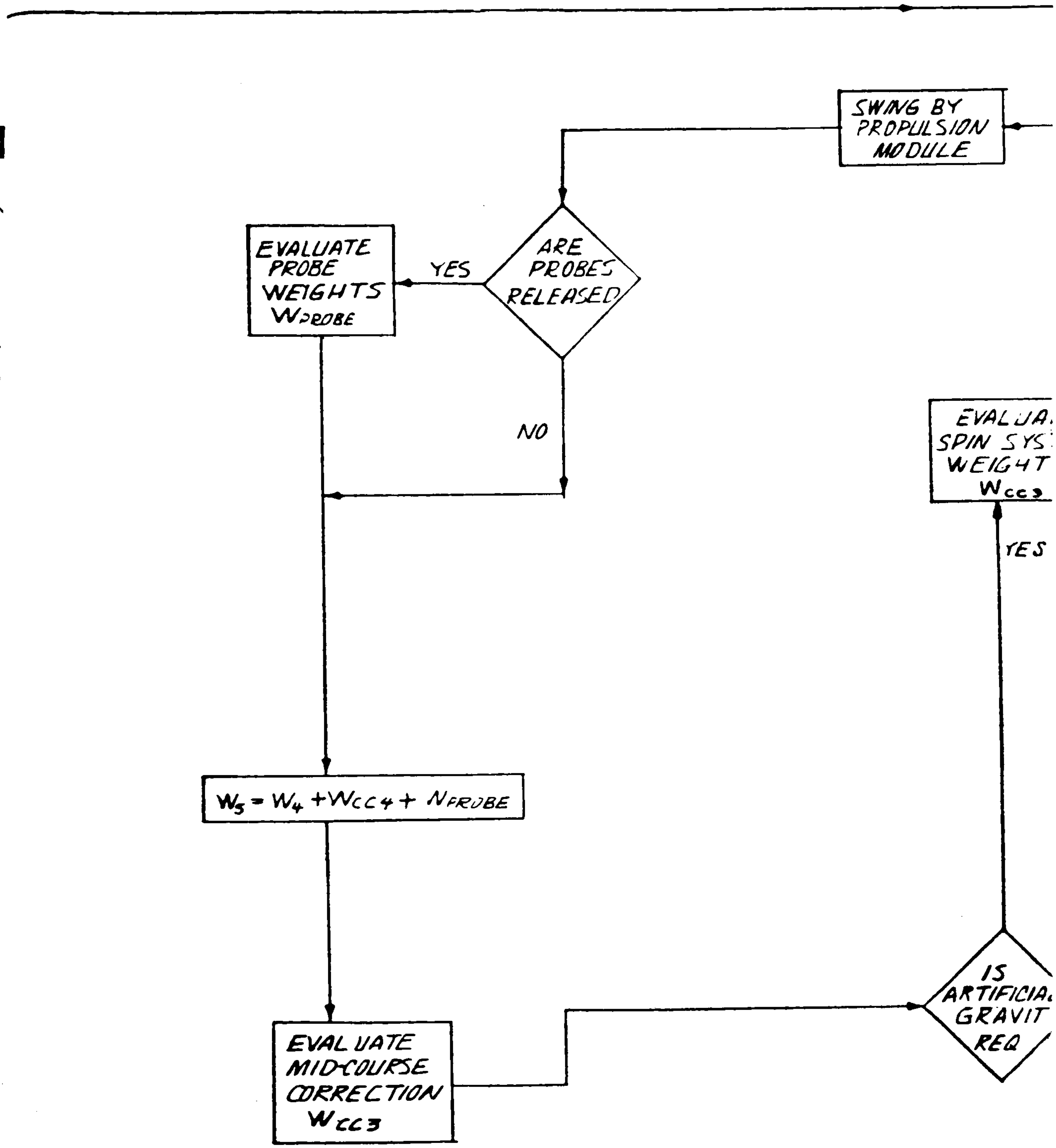

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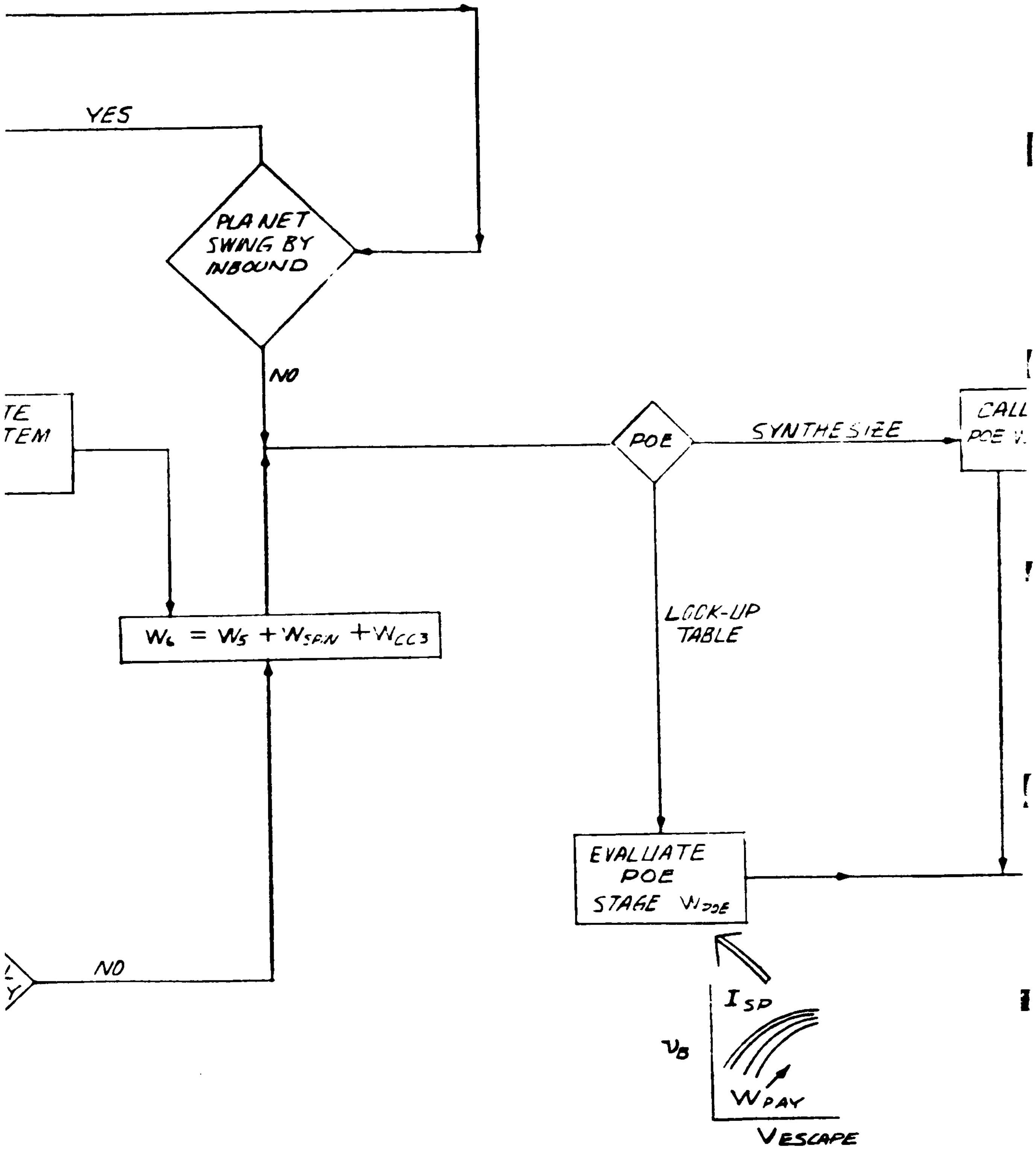

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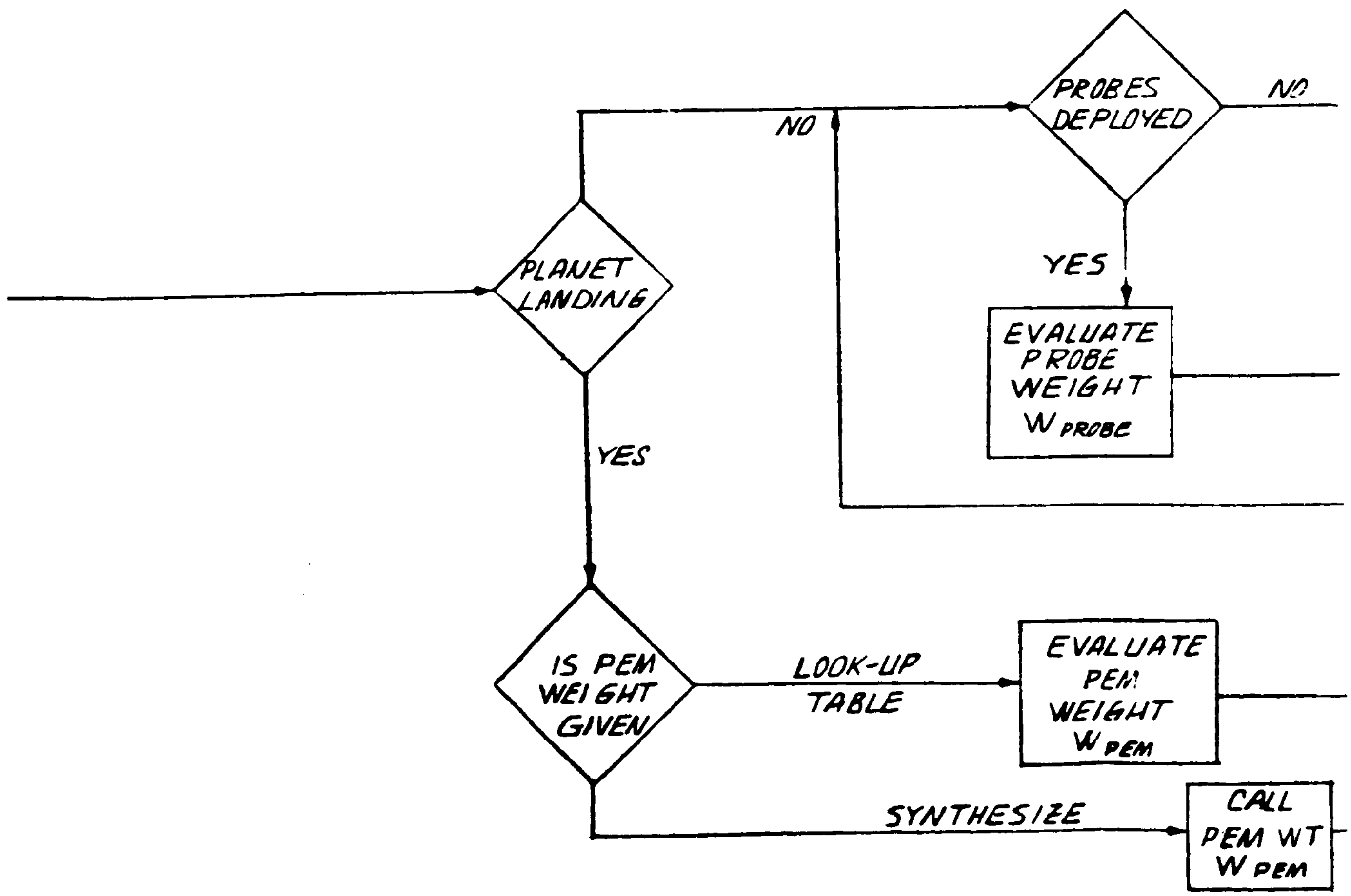

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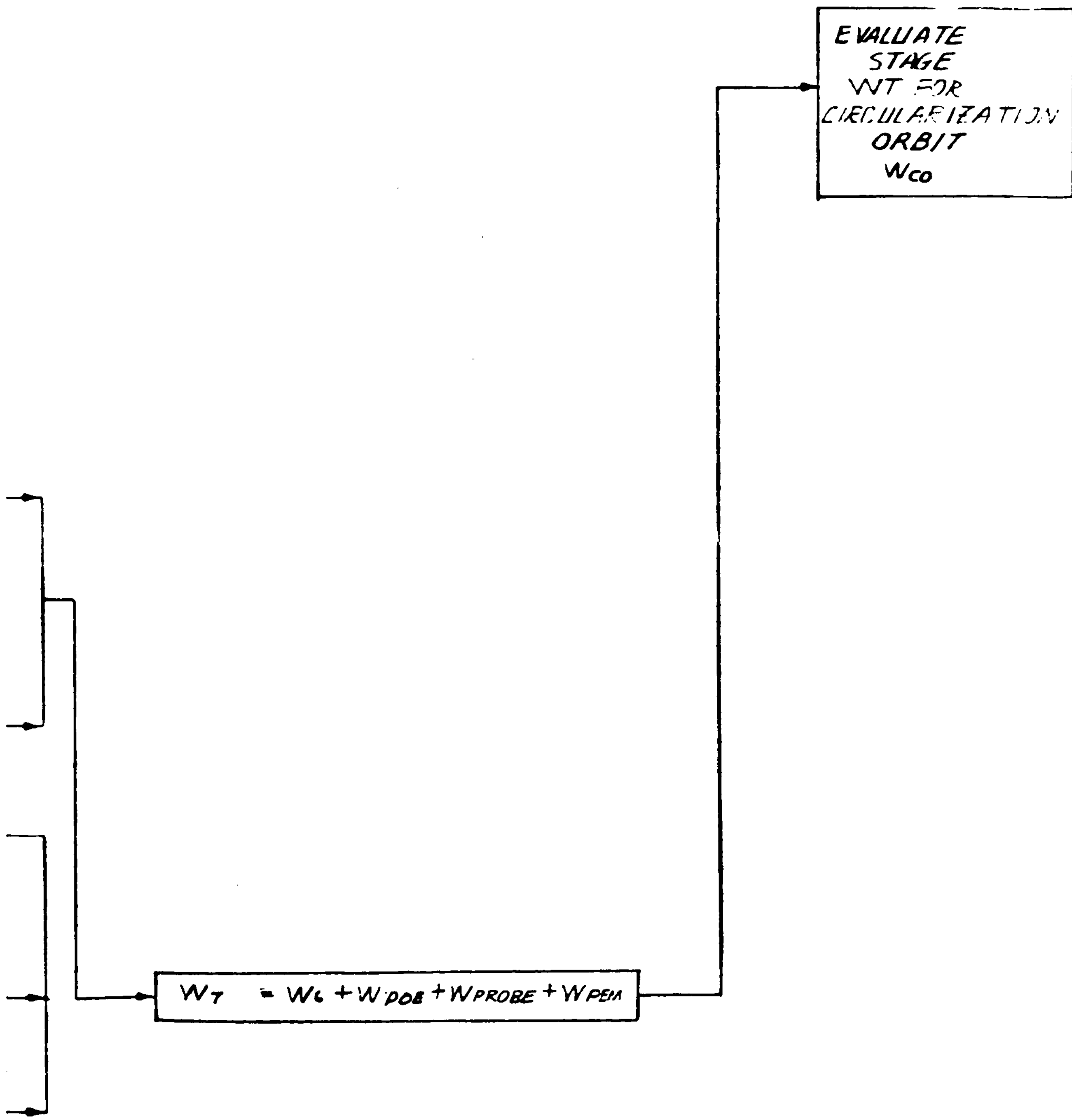

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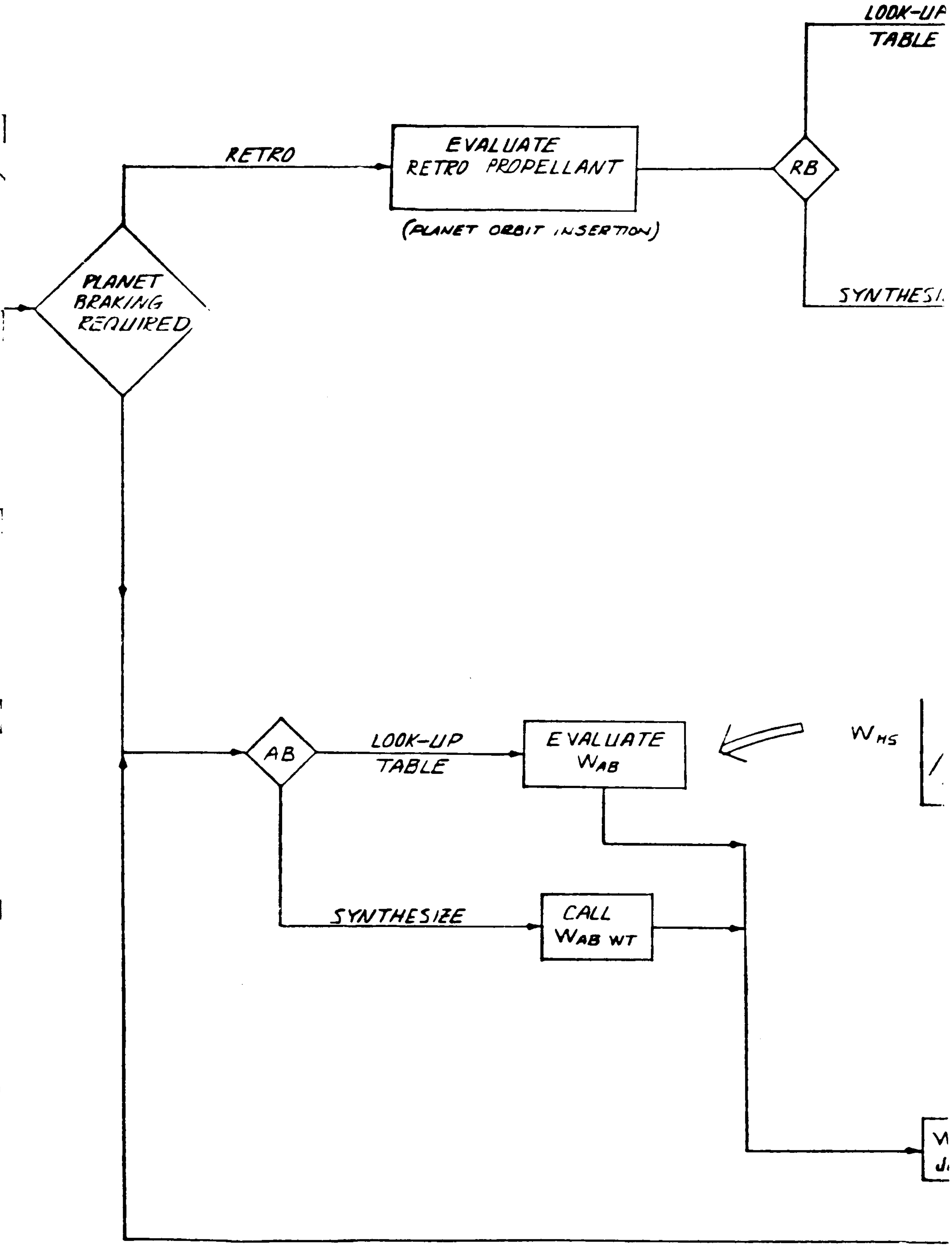

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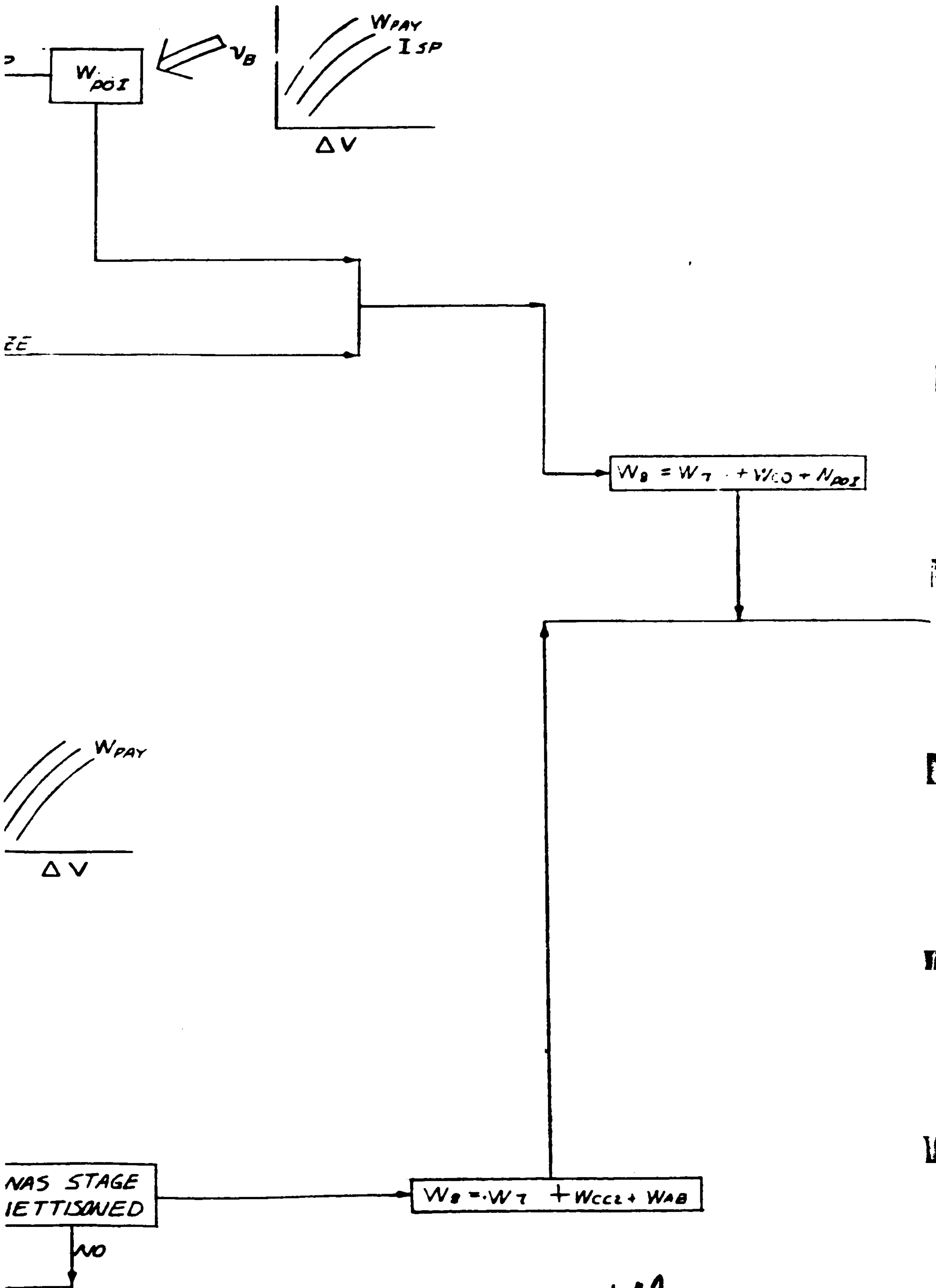

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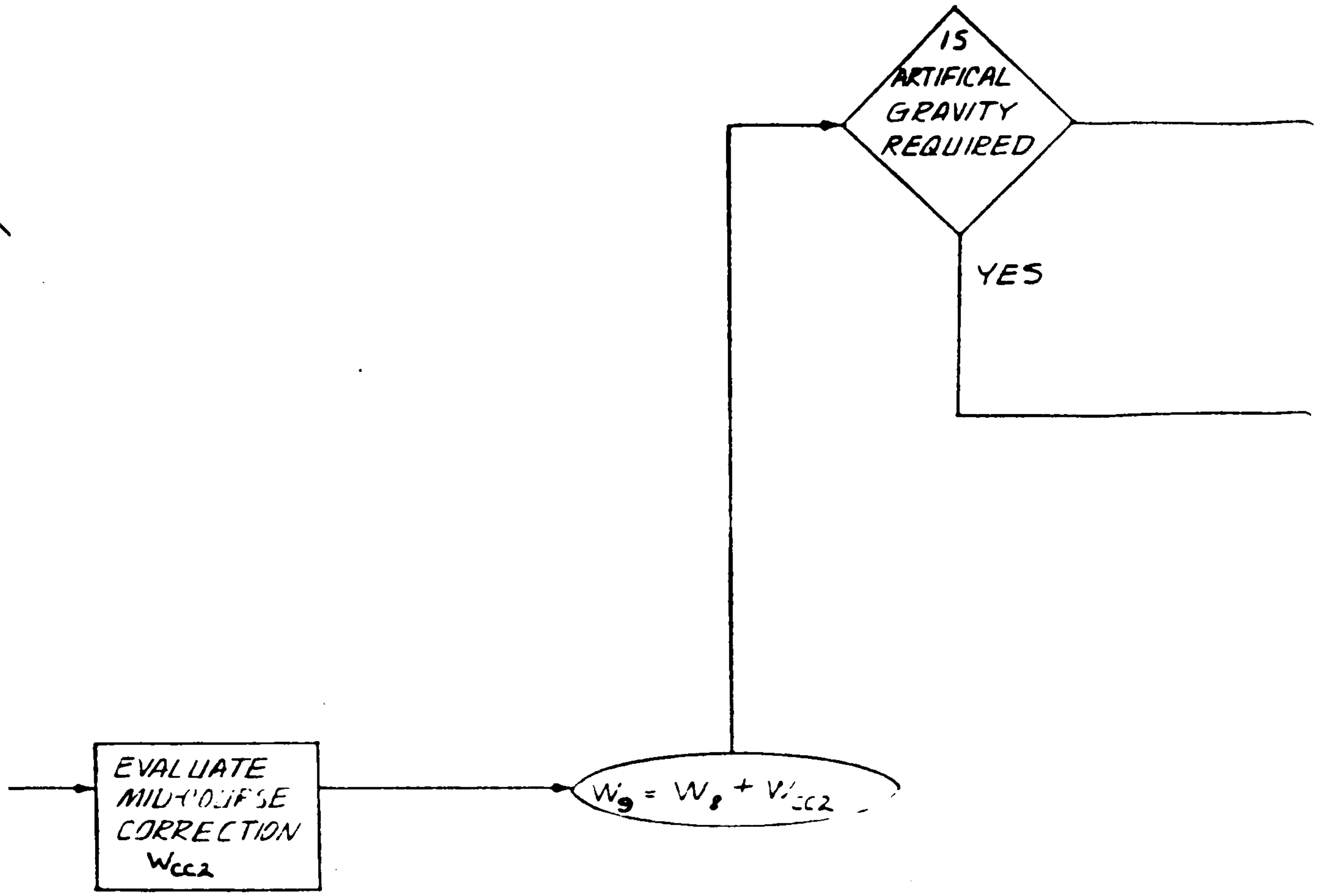

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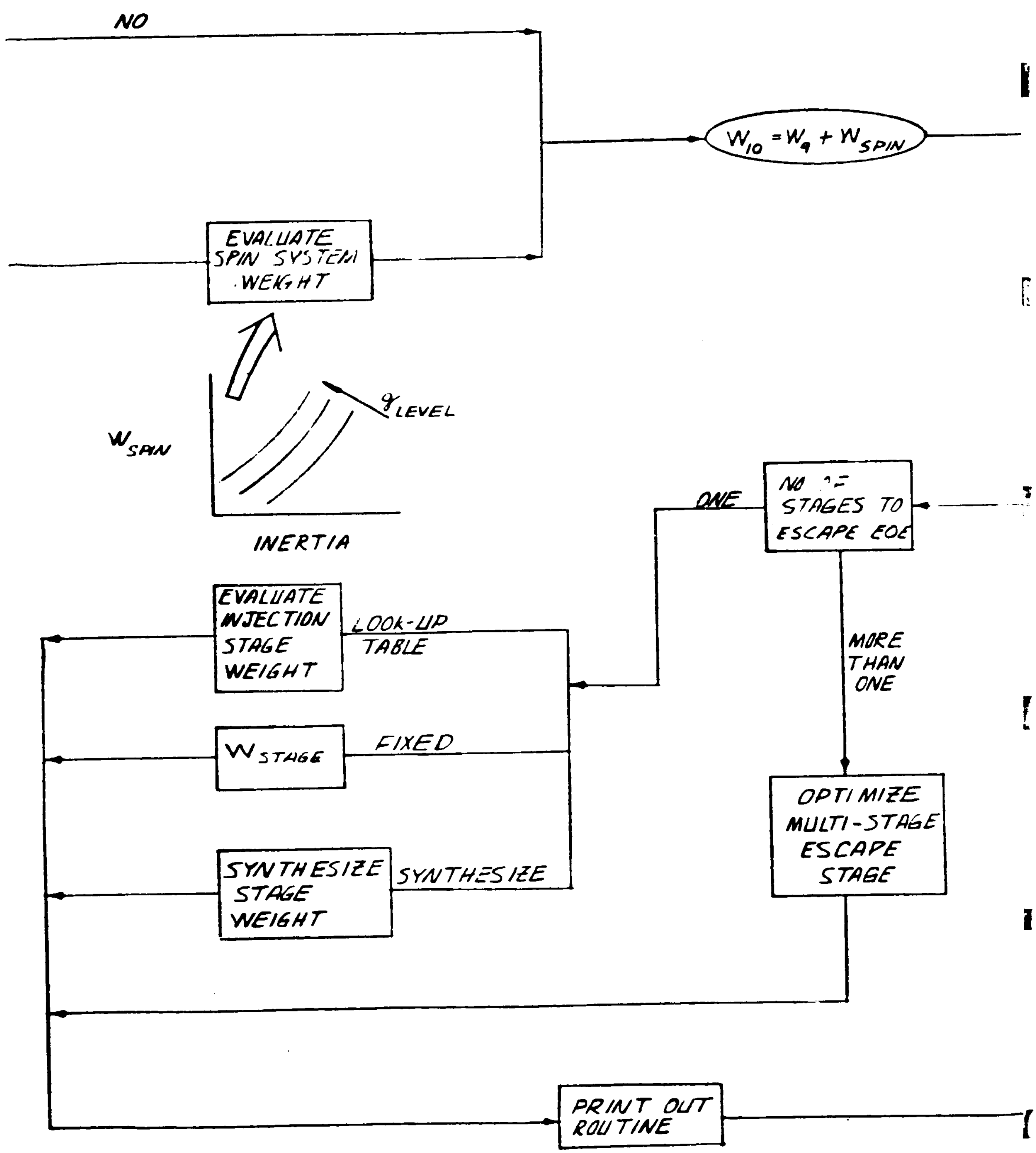

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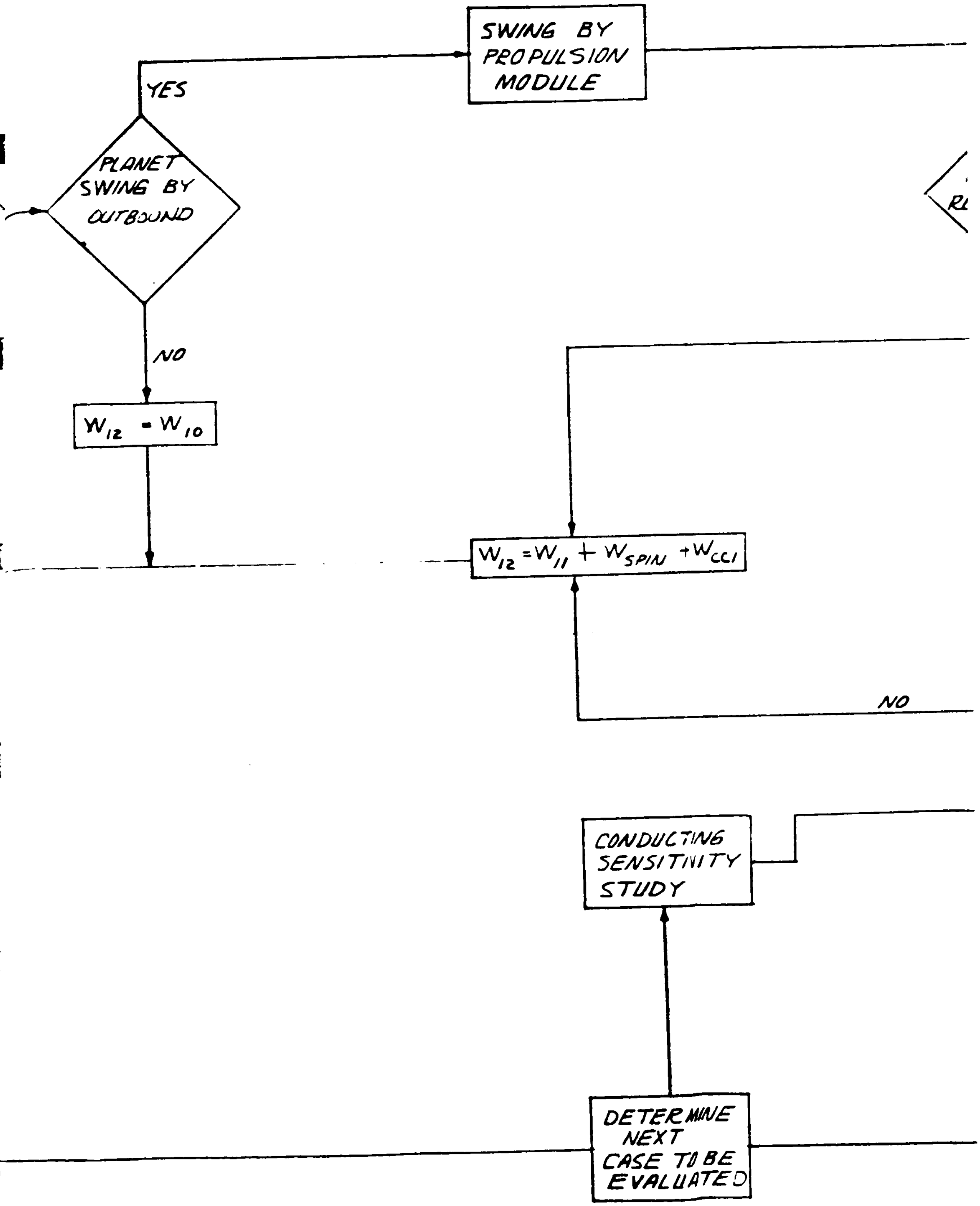

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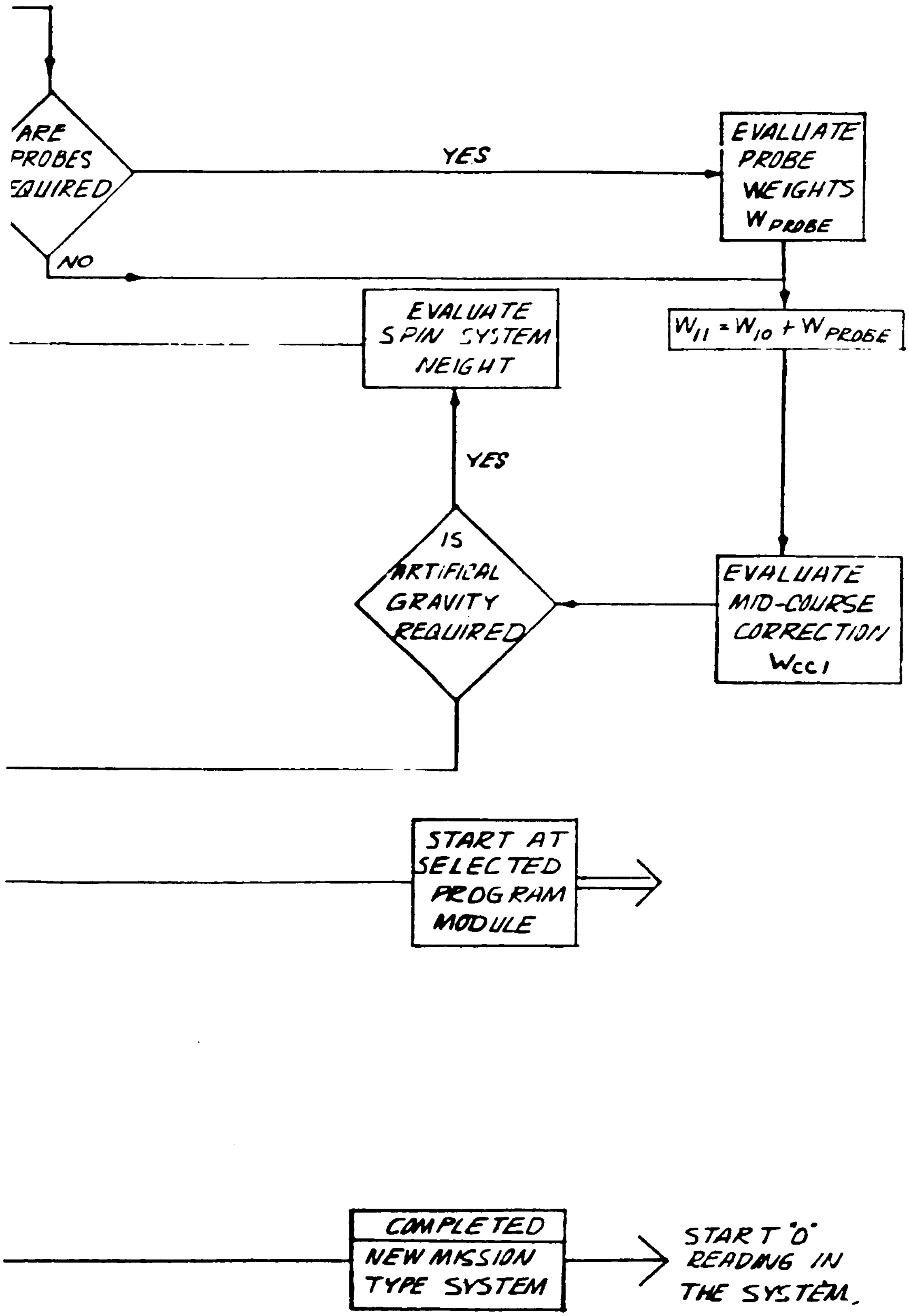

gure 15. Planetary Mission Weight Synthesis Program Logic

WEIGHT SYNTHESIS PARAMETRIC ANALYSIS

Generalized weight synthesis data were generated for the Earth Reentry Module (ERM), Mission Module (MM), Planetary Excursion Modules (PEM), Propulsion Modules (PM), and the aerobraker spacecraft. The data are presented in parametric form, which permits the rapid determination of the approximate mass requirements of the individual modules and the mass on earth orbit. The mass requirements for the individual modules, which are presented in this section, do not include the effects of environmental considerations (thermal insulation, meteoroid protection, and radiation protection) since the environmental effects depend upon the mission objective and, to a lesser extent, the mission opportunity. The effects of environmental considerations are included in a subsequent section (Mission/System Design) where the total system mass requirements are determined for specific missions. The aerobraker mass data, however, include nominal environments.

EARTH REENTRY MODULE

The ERM mass depends upon the module shape, crew size, and earth reentry speed. The mass requirements for the Apollo, biconic, and conic ERM's are presented in Figures 16 through 18 for crew sizes from 4 to 20 men and for the range of reentry speeds applicable to the study. A comparison of the three ERM's is shown in Figure 19 for a crew size of eight men. Below a reentry speed of approximately 14.2 km/s, the Apollo shape is the most advantageous on the basis of mass alone. Between 14.2 and 17.5 km/s the biconic shape is the lightest; above 17.5 km/s the conic shape is the lightest. By comparing Figure 16 through 18, it can be seen that the reentry speeds at which the mass requirements intersect are approximately the same for all crew sizes.

The reentry speeds are less than 14.7 km/s for all mission objectives except Mercury and Ceres provided the Venus swingby mode is used for the Mars missions. For the majority of the missions being considered, the Apollo shape imposes the lowest ERM mass requirements. The reentry speeds for Mercury missions are between 14.5 and 17.5 km/s and, in general, the biconic configuration imposes the lowest mass requirements. The reentry speeds can be reduced to less than 12.4 km/s for the Mercury missions if the Venus swingby mode is used during earth return. These missions, however, impose higher total incremental velocity requirements than the minimum-energy direct missions during the same year. The biconic shape results in a mass savings of approximately 11.5 percent (900 kilograms) when compared with the Apollo shape for the highest Mercury reentry speed considered for the direct missions.

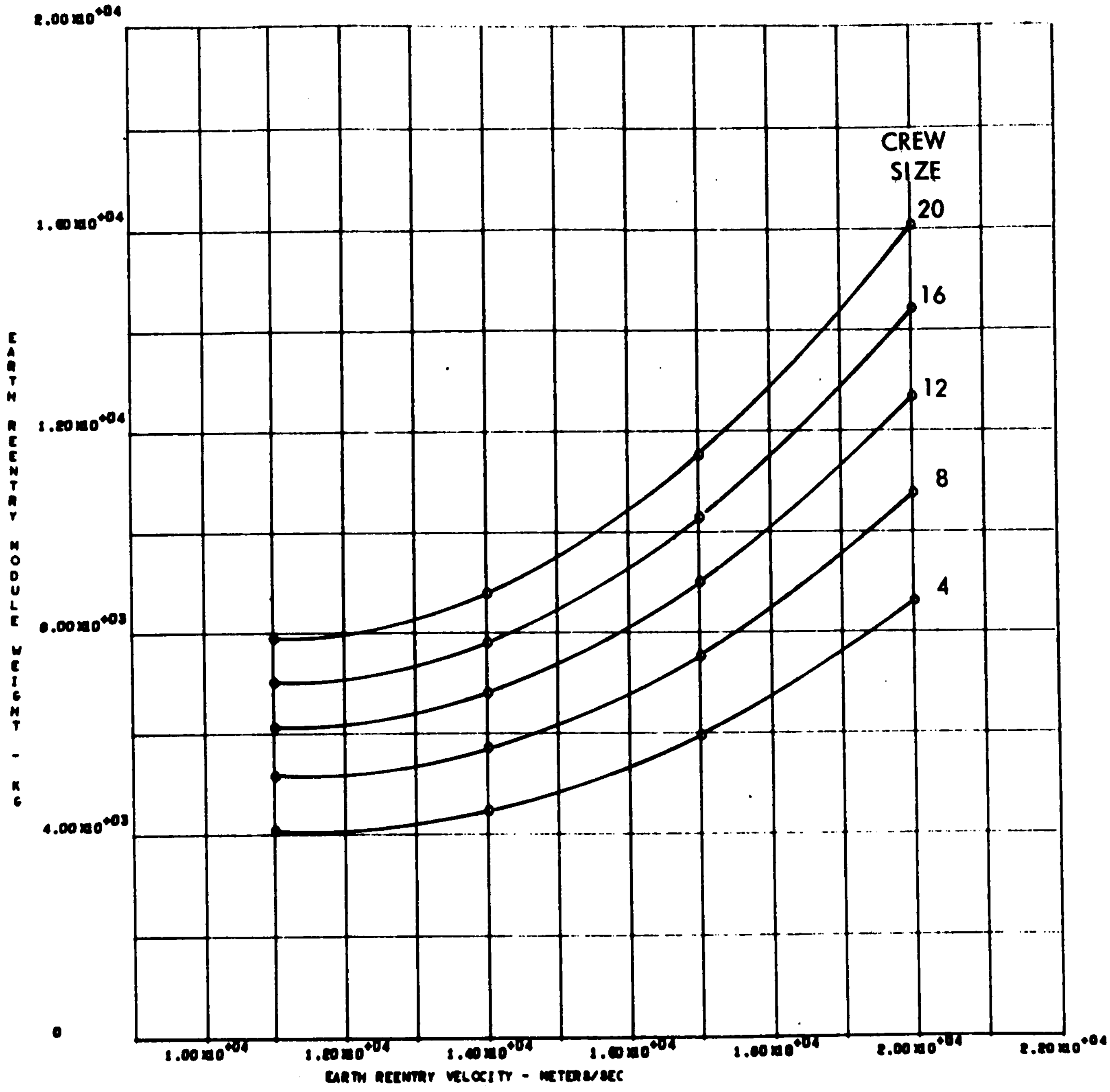

Figure 16. Apollo Earth Reentry Module Mass

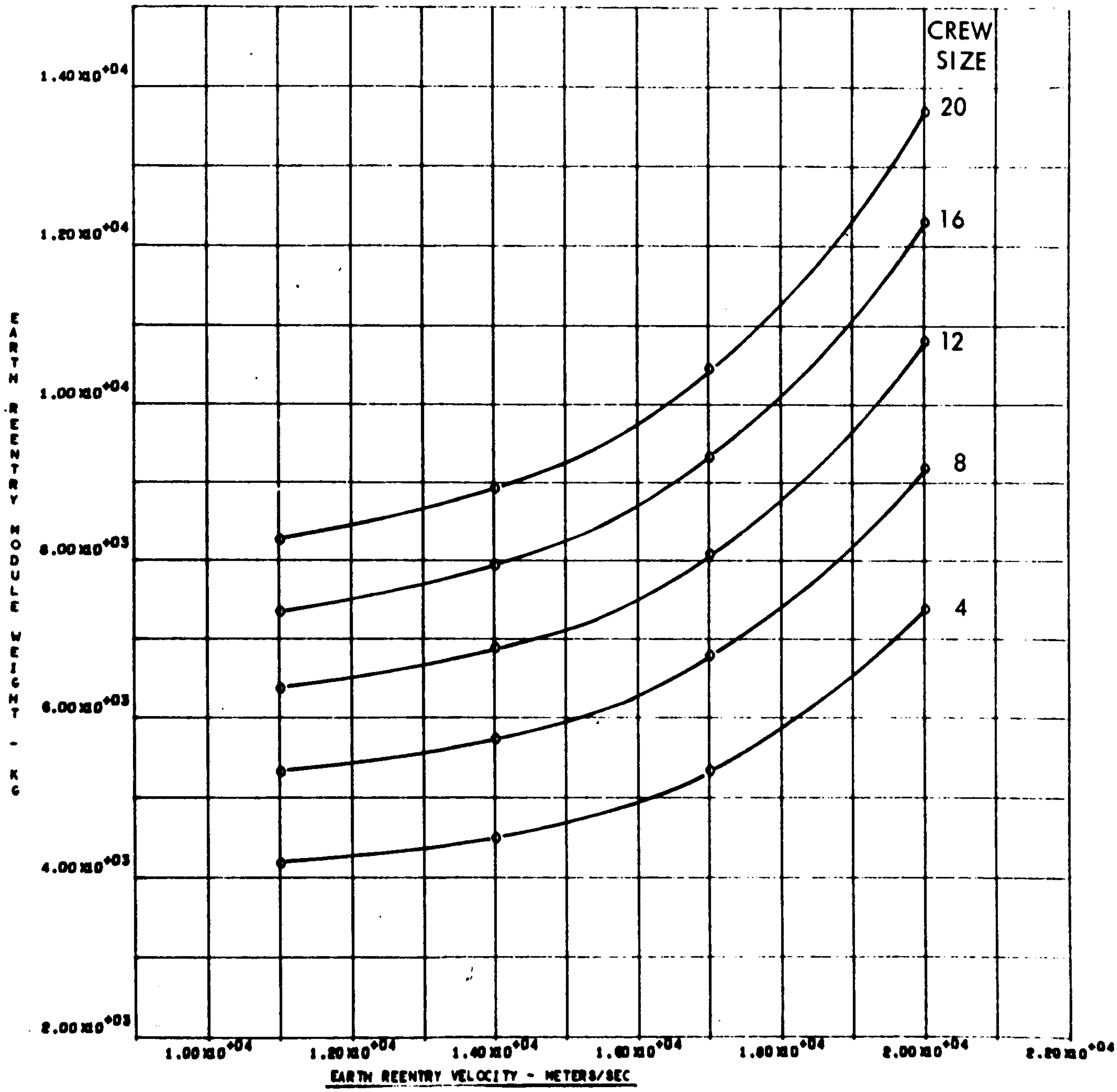

Figure 17. Biconic Earth Reentry Module Mass

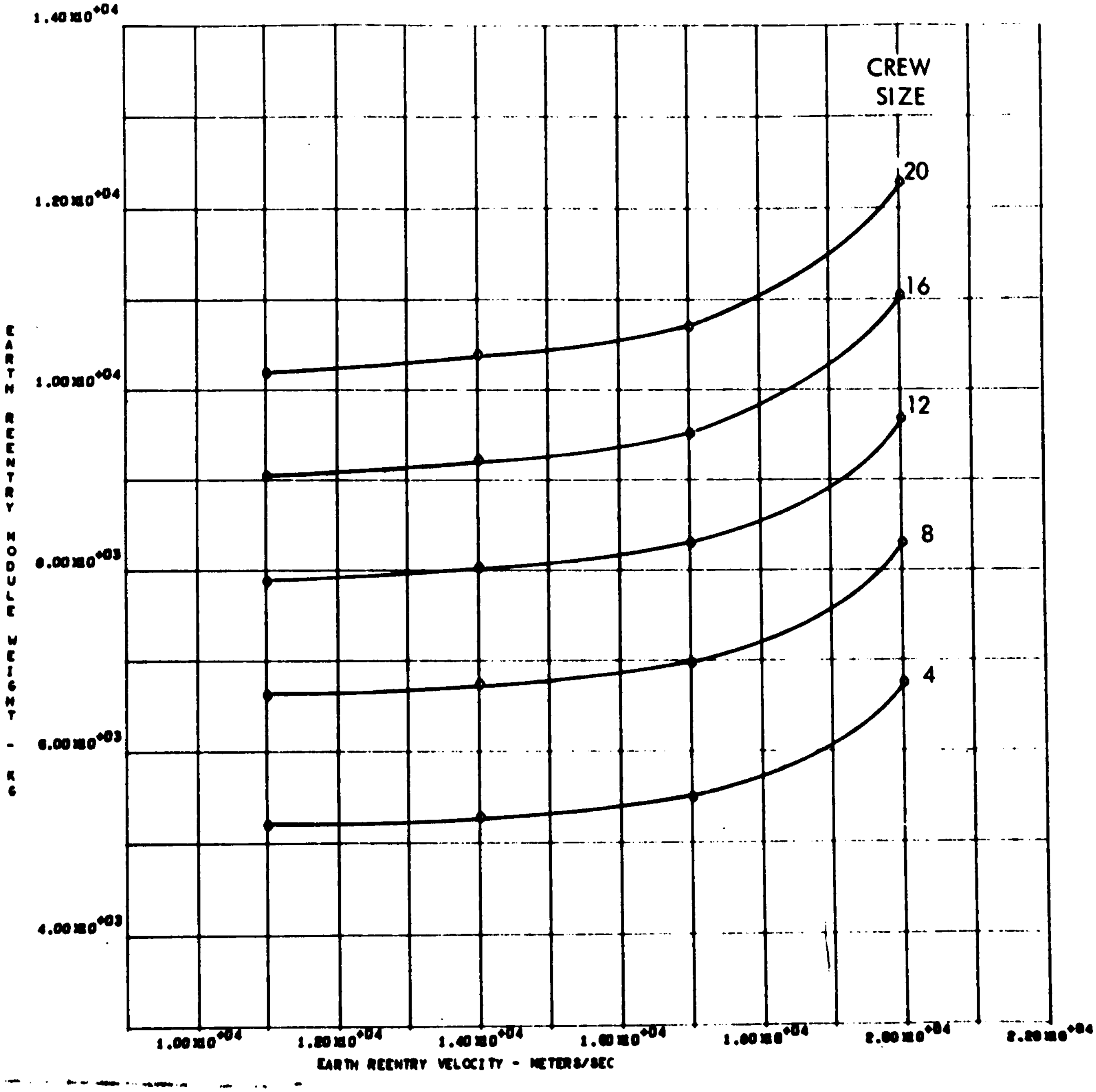

Figure 18. Conic Earth Reentry Module Mass

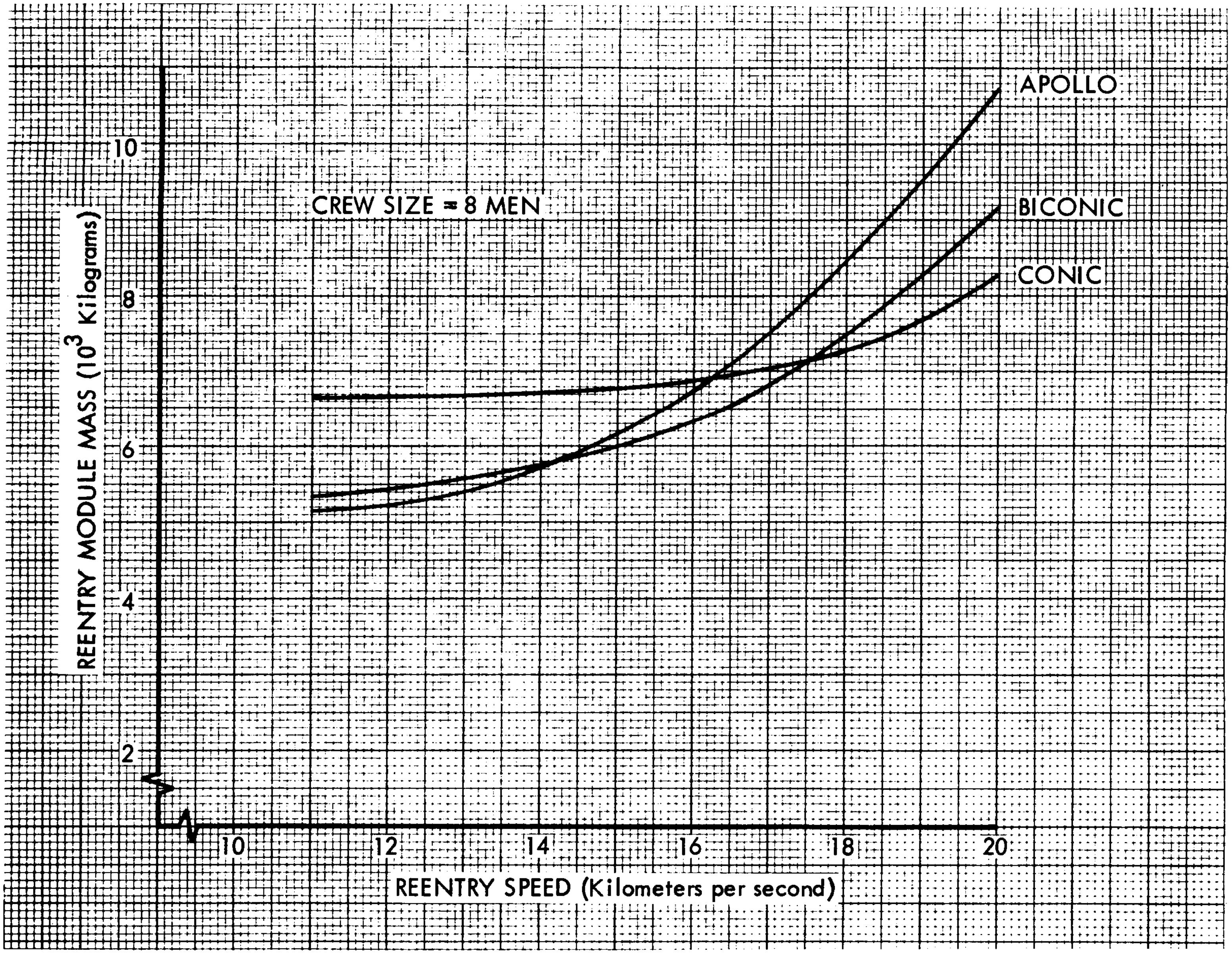

Figure 19. Reentry Module Mass Comparison

MISSION MODULE

The MM mass depends upon the spacecraft crew size, mission duration, free volume per man, number of floors, and subsystem types. The effects of the first four parameters on the mass requirements are shown in Figures 20 through 24, assuming an environmental control and life support subsystem with water and oxygen recovery and an isotope mercury Rankine electrical power subsystem. Of the parameters shown, the crew size and mission duration have the most significant effect on the module mass, and the number of floors the least effect. For a free volume of 750 cubic feet per man, a 20-man crew, and a mission duration of 1500 days, the mission module mass is increased by approximately one percent (800 kilograms) if the number of floors is decreased from four to three.

The effects of free volume per man (from 400 to 1200 cubic feet per man) varies from seven percent to seventeen 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/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.

Of the mission module subsystems considered in detail, the type of environmental control and life support subsystem has the predominant effect on the module. The scaling equations defining the open, water recovery, and water and oxygen recovery systems are

$$W_{OPEN} = 408 + 330\,N_C + 0.09\,t + 11.317\,N_C t + W_{LCS};\ \text{kg}$$

$$W_{\left(\substack{H_2O\\ REC}\right)} = 468 + 367\,N_C + 0.09\,t + 1.981\,N_C t + W_{LCS};\ \text{kg}$$

$$W_{\left(\substack{H_2O + O_2\\ REC}\right)} = 471 + 323\,N_C + 0.09\,t + 0.997\,N_C t + W_{LCS};\ \text{kg}$$

where

N_C = crew size

t = mission duration; days

W_{LCS} = cabin repressurization mass, kg.

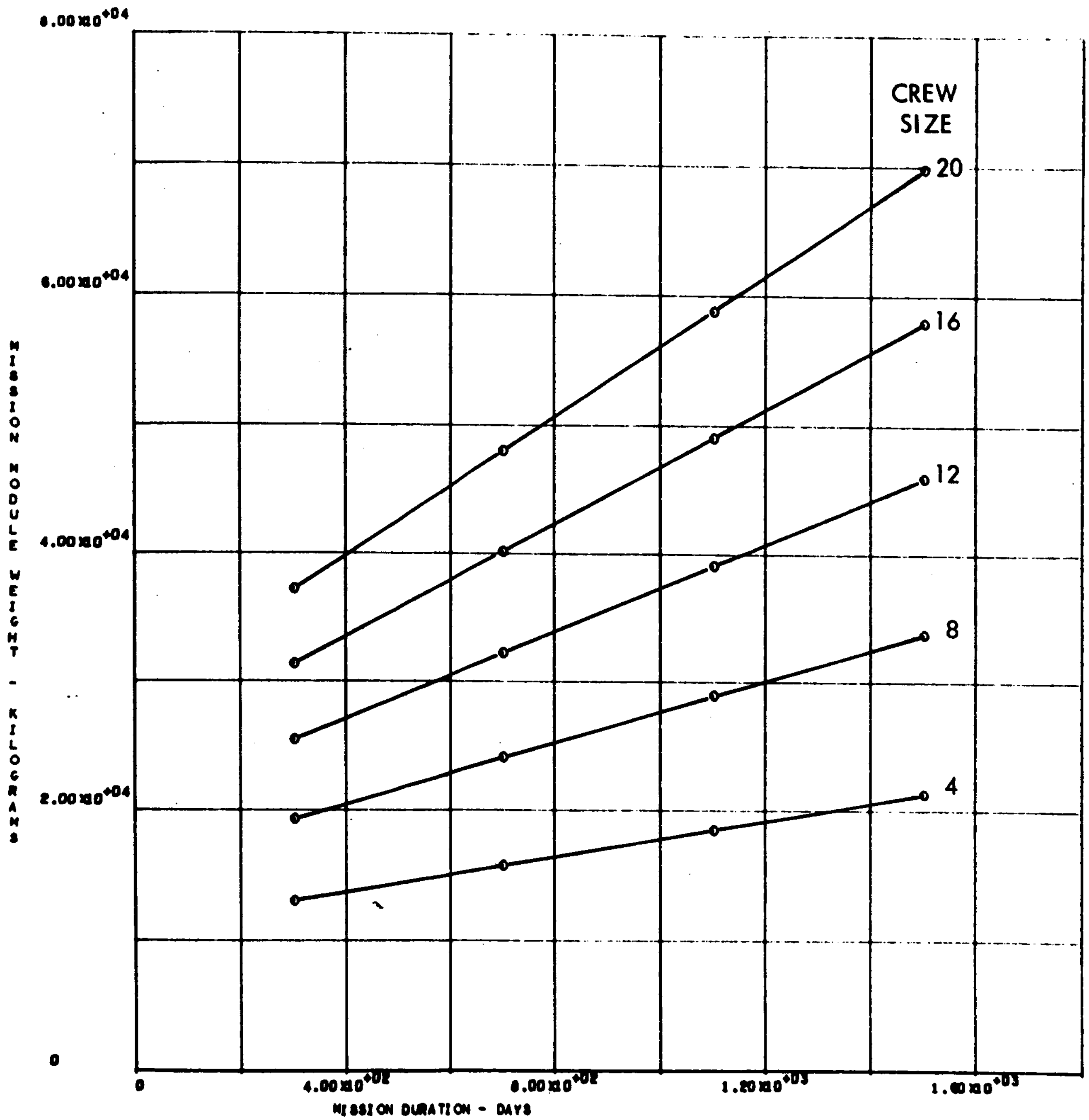

Figure 20. Mission Module Mass (750 Cubic Feet Per Man, 3 Floors)

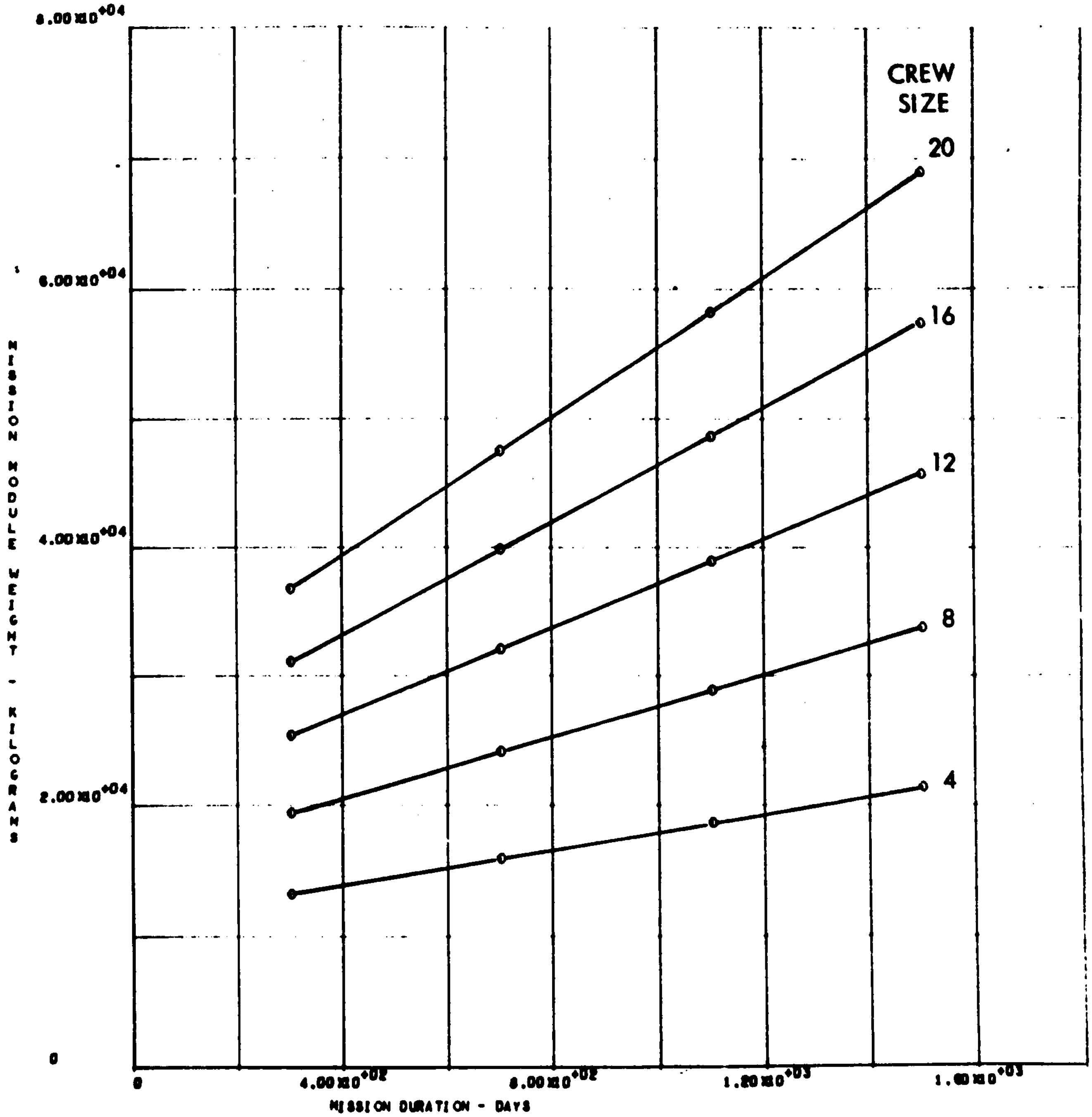

Figure 21. Mission Module Mass (750 Cubic Feet Per Man, 4 Floors)

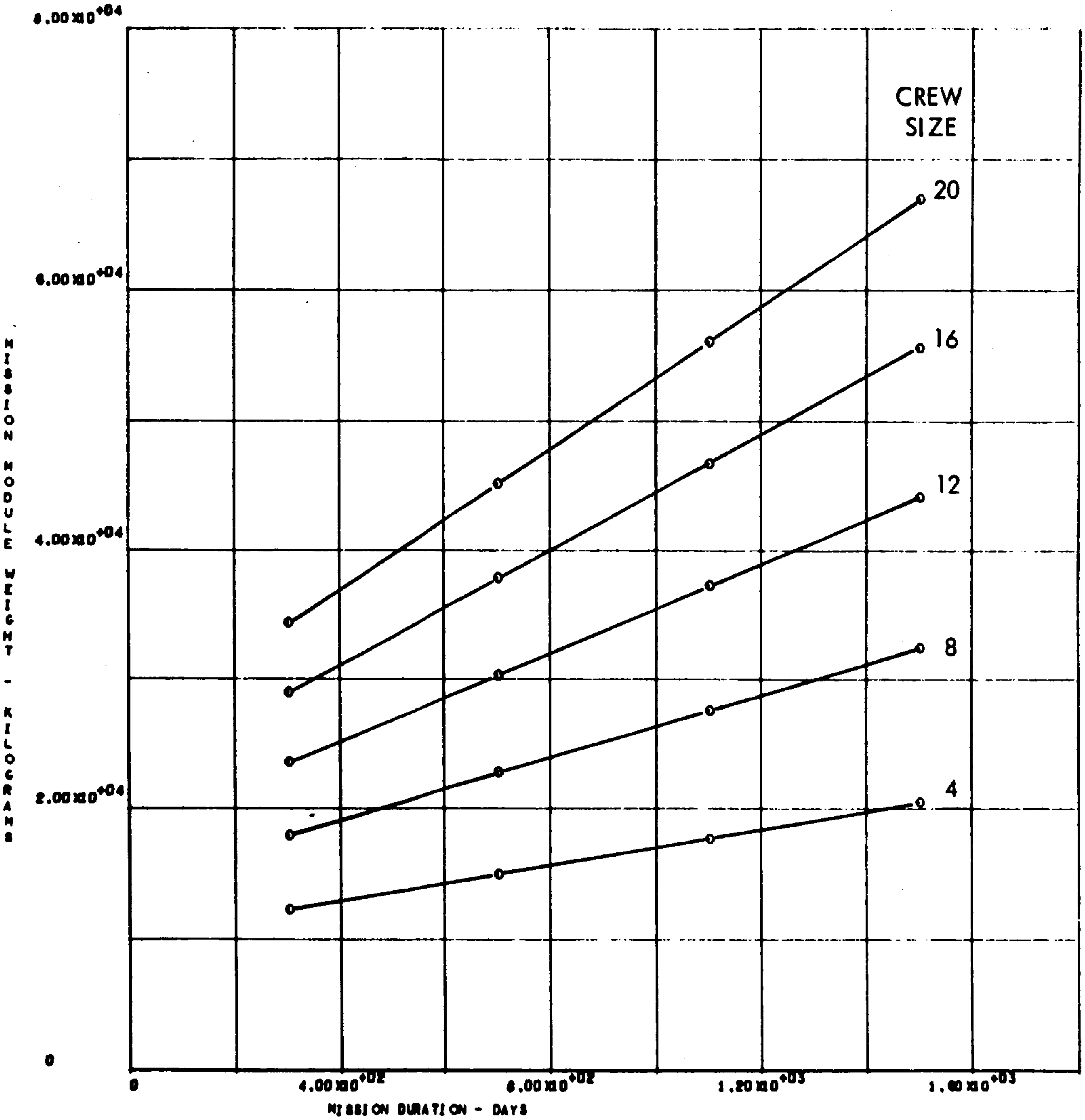

Figure 22. Mission Module Mass (400 Cubic Feet per Man, 3 Floors)

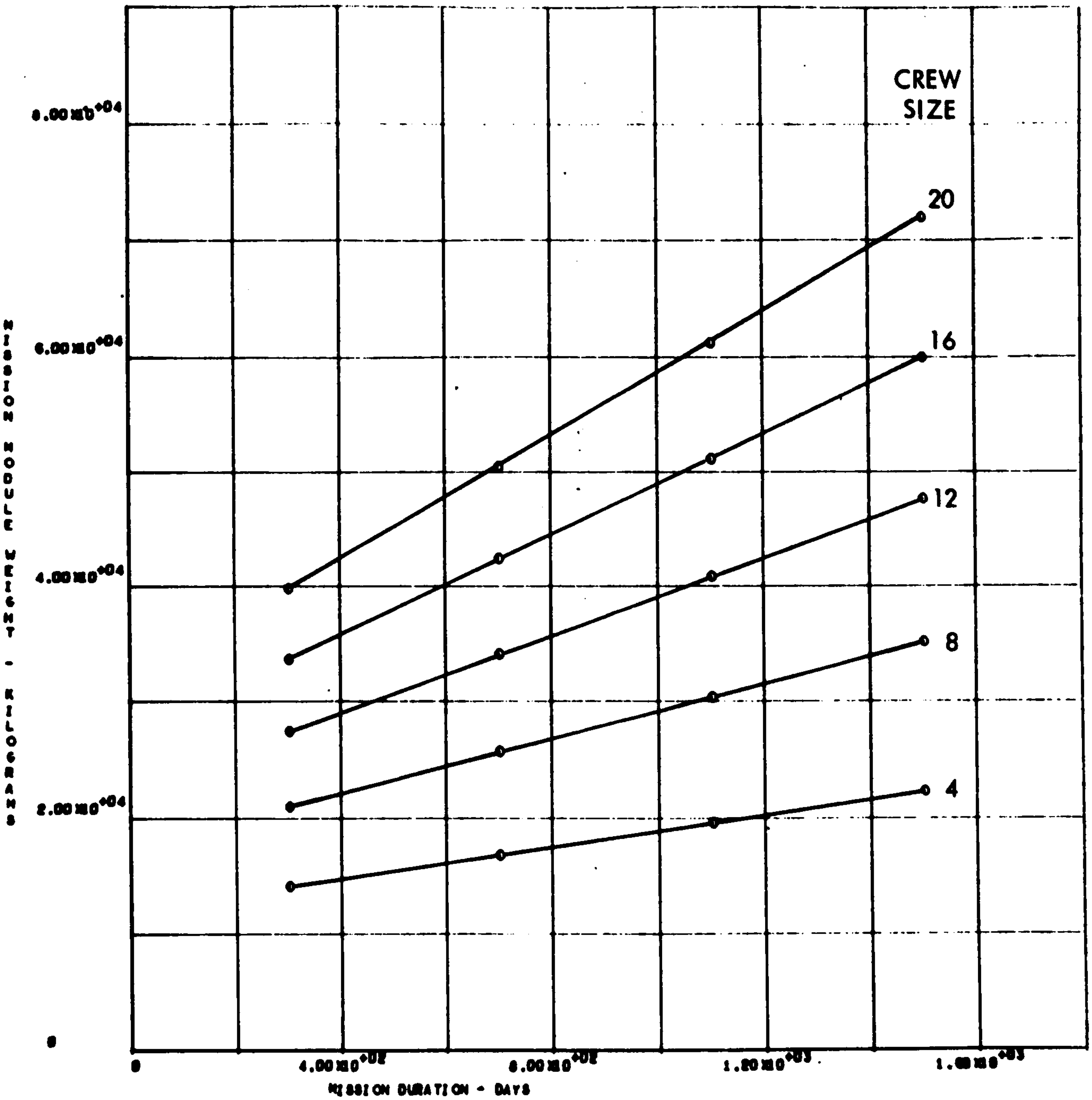

Figure 23. Mission Module Mass (1200 Cubic Feet per Man, 4 Floors)

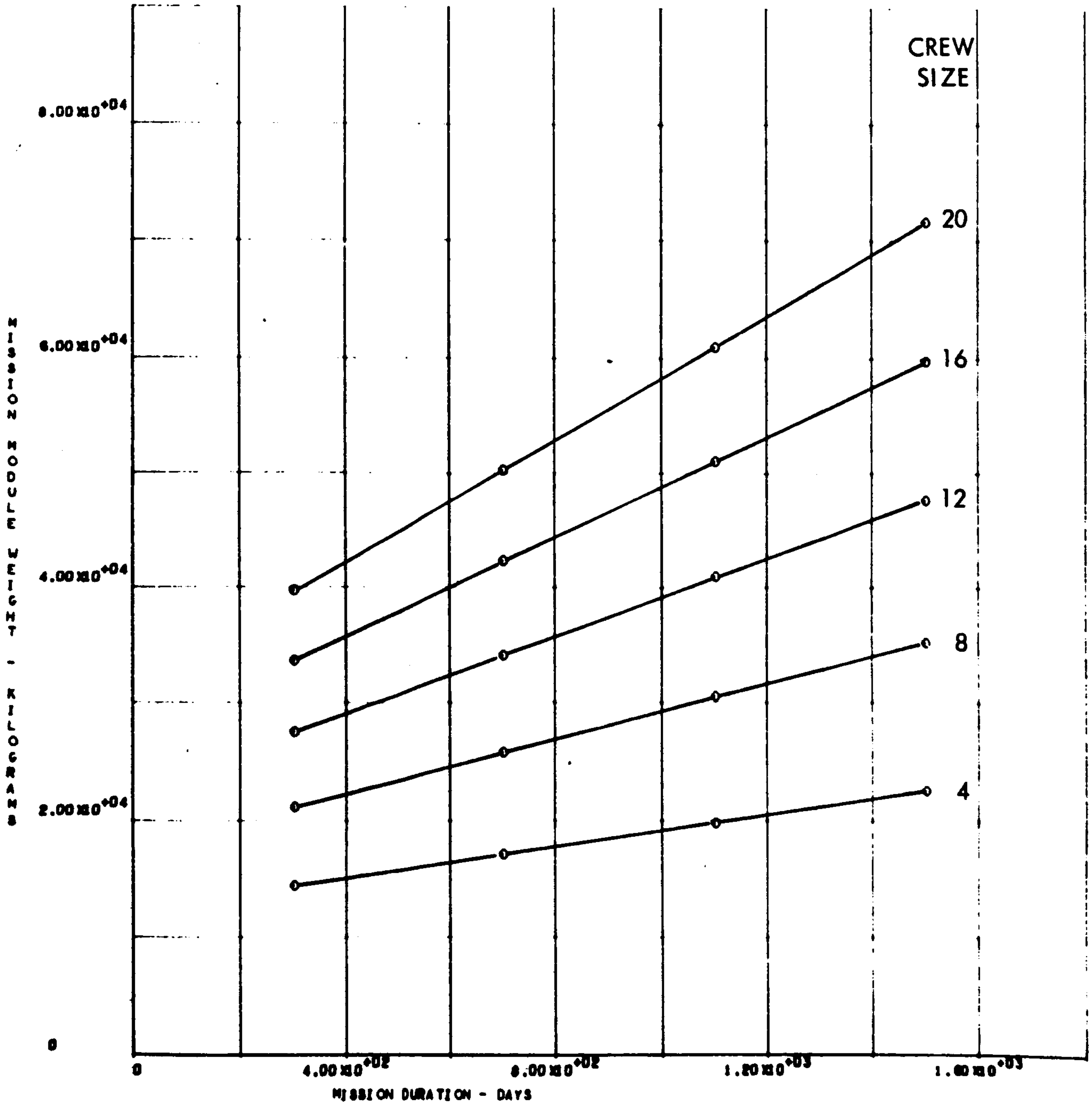

Figure 24. Mission Module Mass (1200 Cubic Feet per Man, 5 Floors)

The resultant effects of the life support system mass on the module mass on Earth orbit are

$$W_{OPEN} - W_{\begin{pmatrix}H_2O\\REC\end{pmatrix}} = -60 - 37\,N_C + 9.336\,N_Ct;\ kg$$

$$W_{OPEN} - W_{\begin{pmatrix}H_2O\\REC\end{pmatrix}+O_2} = -63 + 7\,N_C + 10.320\,N_Ct;\ kg$$

$$W_{\begin{pmatrix}H_2O\\REC\end{pmatrix}} - W_{\begin{pmatrix}H_2O\\REC\end{pmatrix}+O_2} = -3 + 44\,N_C + 0.984\,N_Ct;\ kg$$

The above mass differences do not include the effects of the change in structural mass due to the change in volume or the electrical power mass changes. These effects are small, however, when compared with the differences due to the basic systems. The severe penalty imposed by the open system can be seen by considering the Mercury missions, which have the shortest mission durations (300-400 days). 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 (2,690 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 mass requirements for manned PEM's were investigated parametrically for all mission objectives except Venus. Results are presented in Figures 25 through 30 as a function of crew size and occupancy time. In all cases, including Mars, a propulsive landing mode is assumed. The mass requirements of both the ascent and descent stages are based on a specific impulse of 387 seconds and a propellant bulk density of 1233 kg/m^3 (77.0 lb/ft^3). Two configurations, the Apollo and the Aeronutronic shapes, were considered for the Mars excursion modules. The Apollo shape (Figure 26) is from 21 to 27 percent lighter than the Aeronutronic shape.

The data presented in Figures 25 through 30 are for circular planetary parking orbits at the altitudes indicated on the figures and do not include the environmental protection requirements. The effects of planetary parking orbit eccentricity and environmental considerations are included in the data presented in the Mission/System Design section.

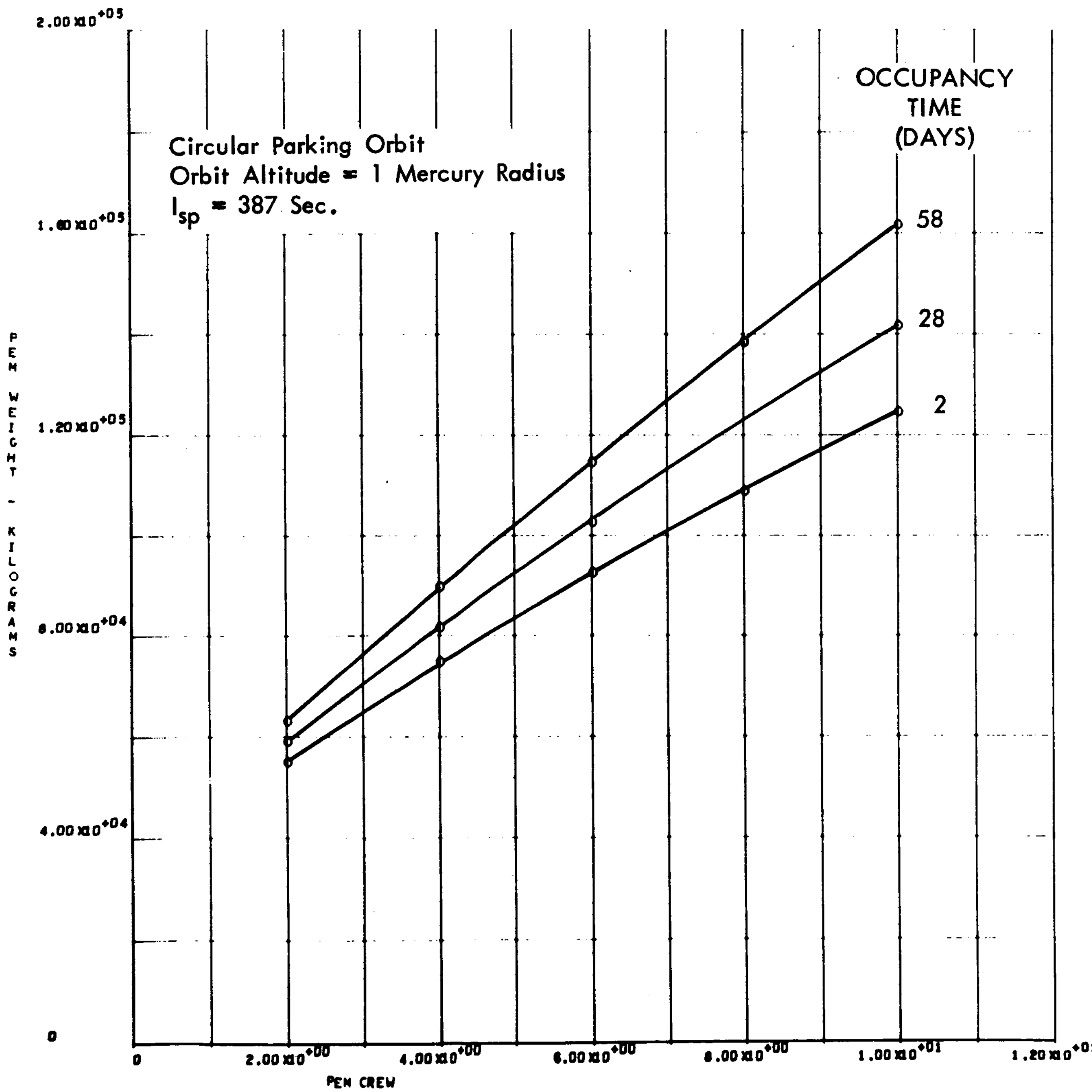

Figure 25. Mercury Planetary Excursion Module Mass.

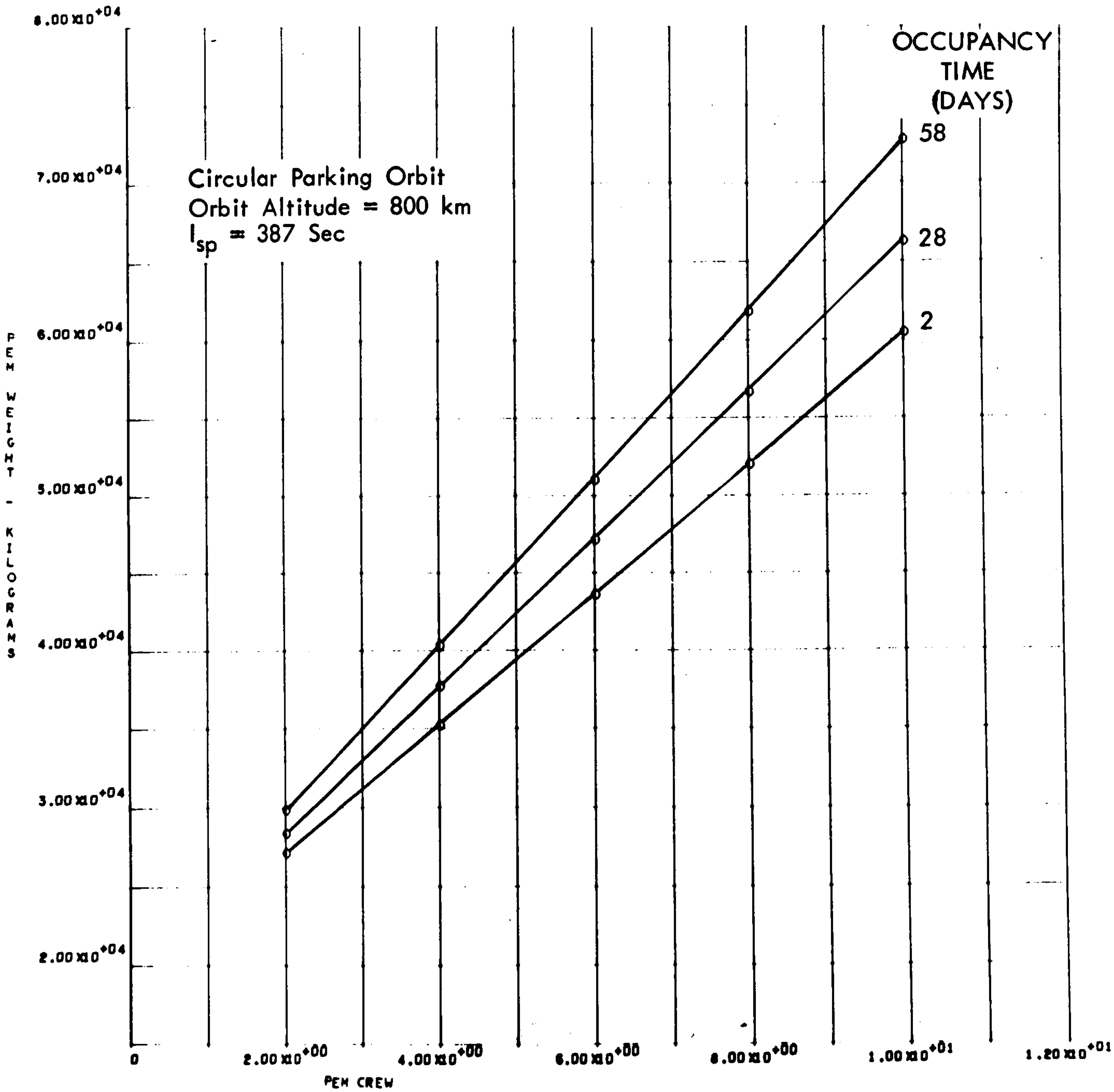

Figure 26. Mars Planetary Excursion Module Mass (Apollo Shape)

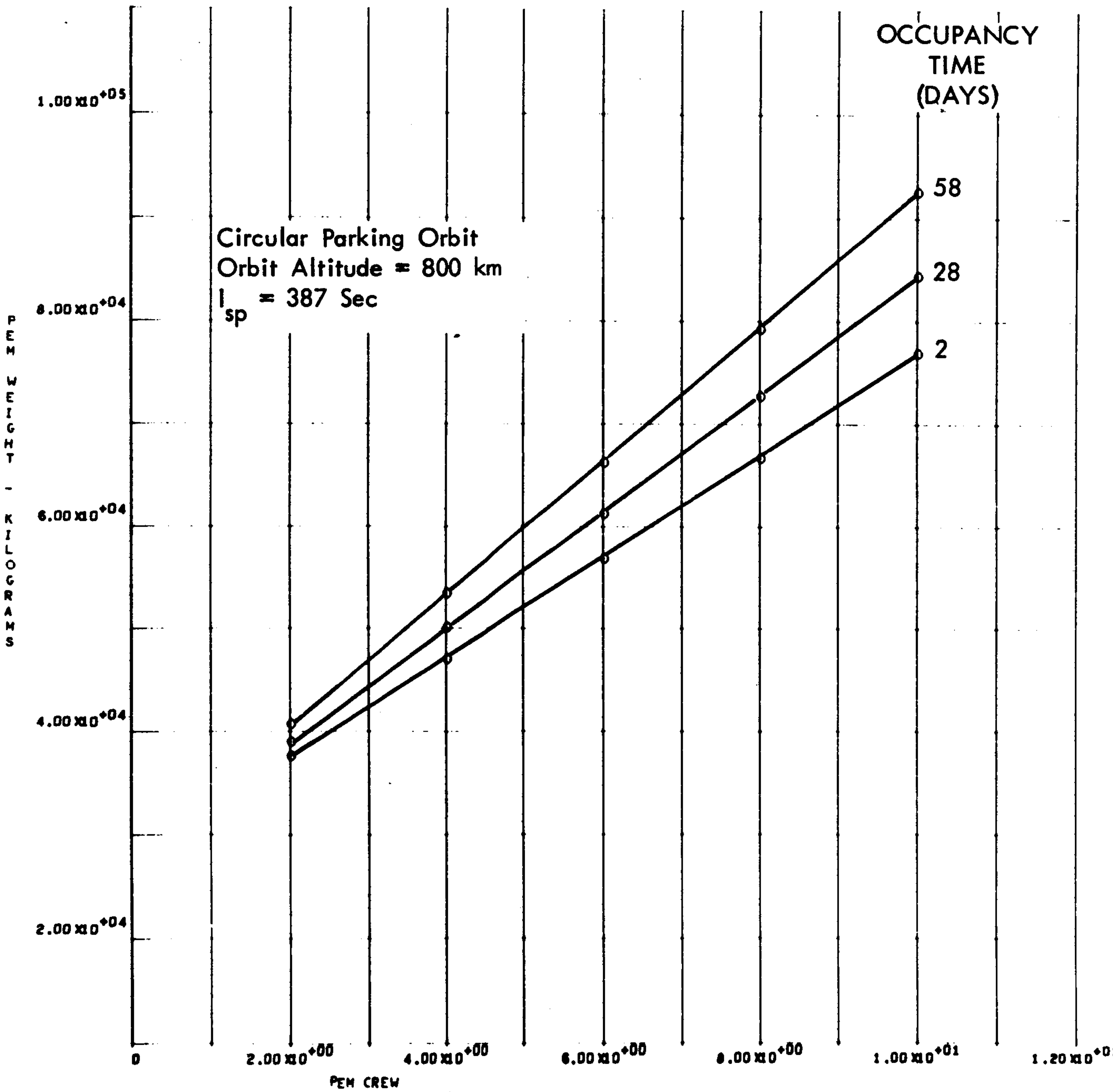

Figure 27. Mars Planetary Excursion Module Mass (Aeronutronic Shape)

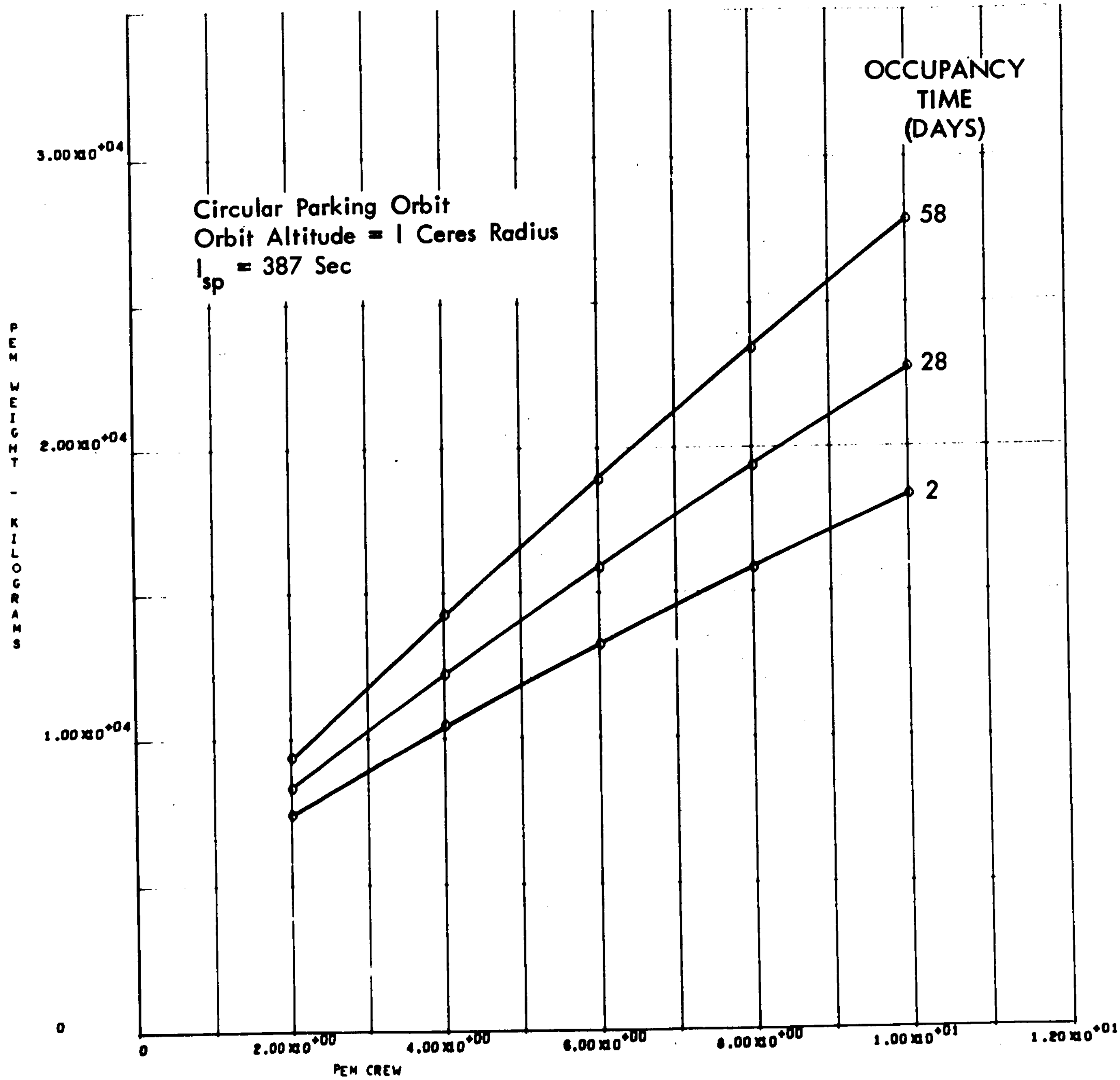

Figure 28. Ceres Planetary Excursion Module Mass

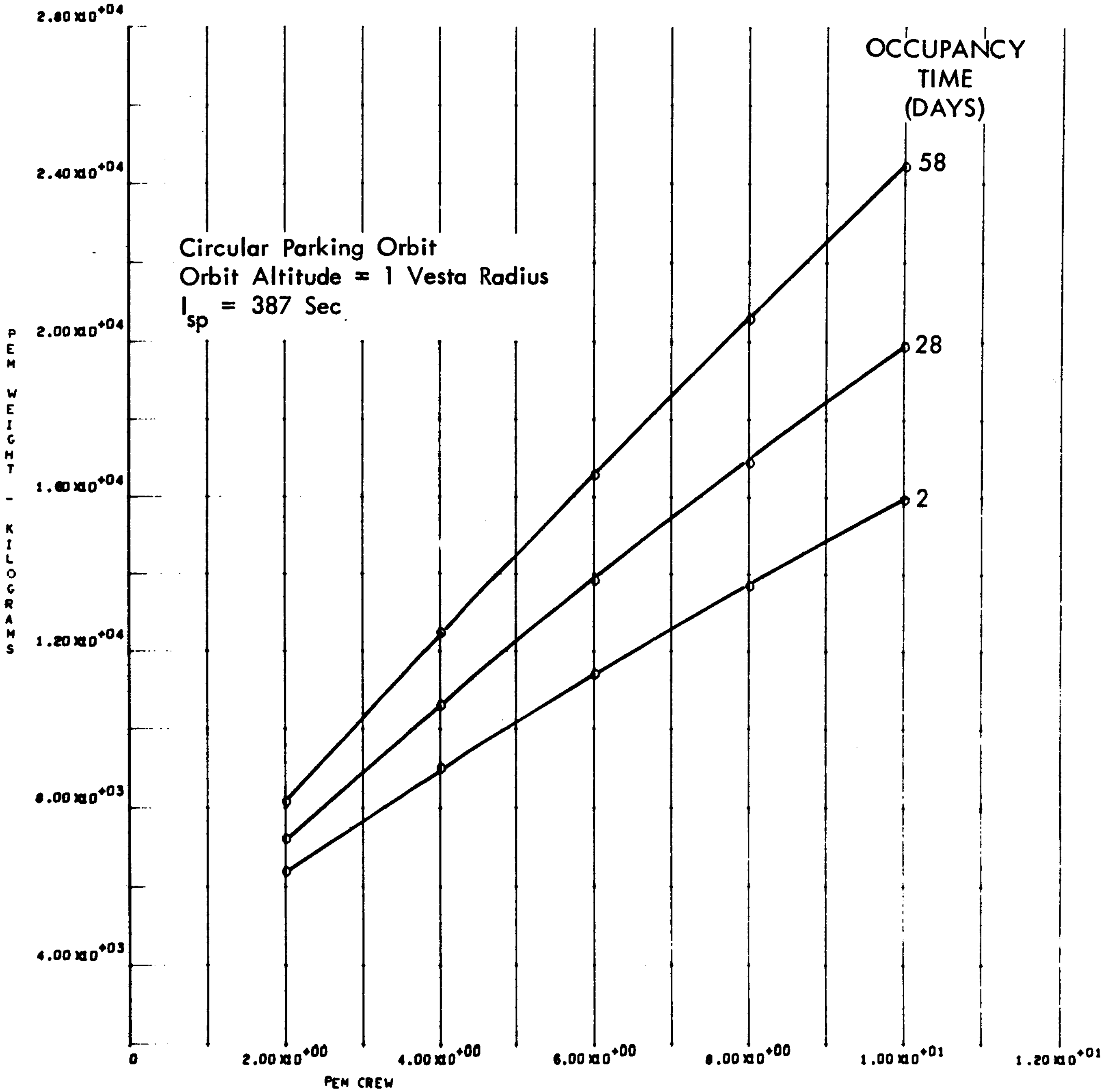

Figure 29. Vesta Planetary Excursion Module Mass

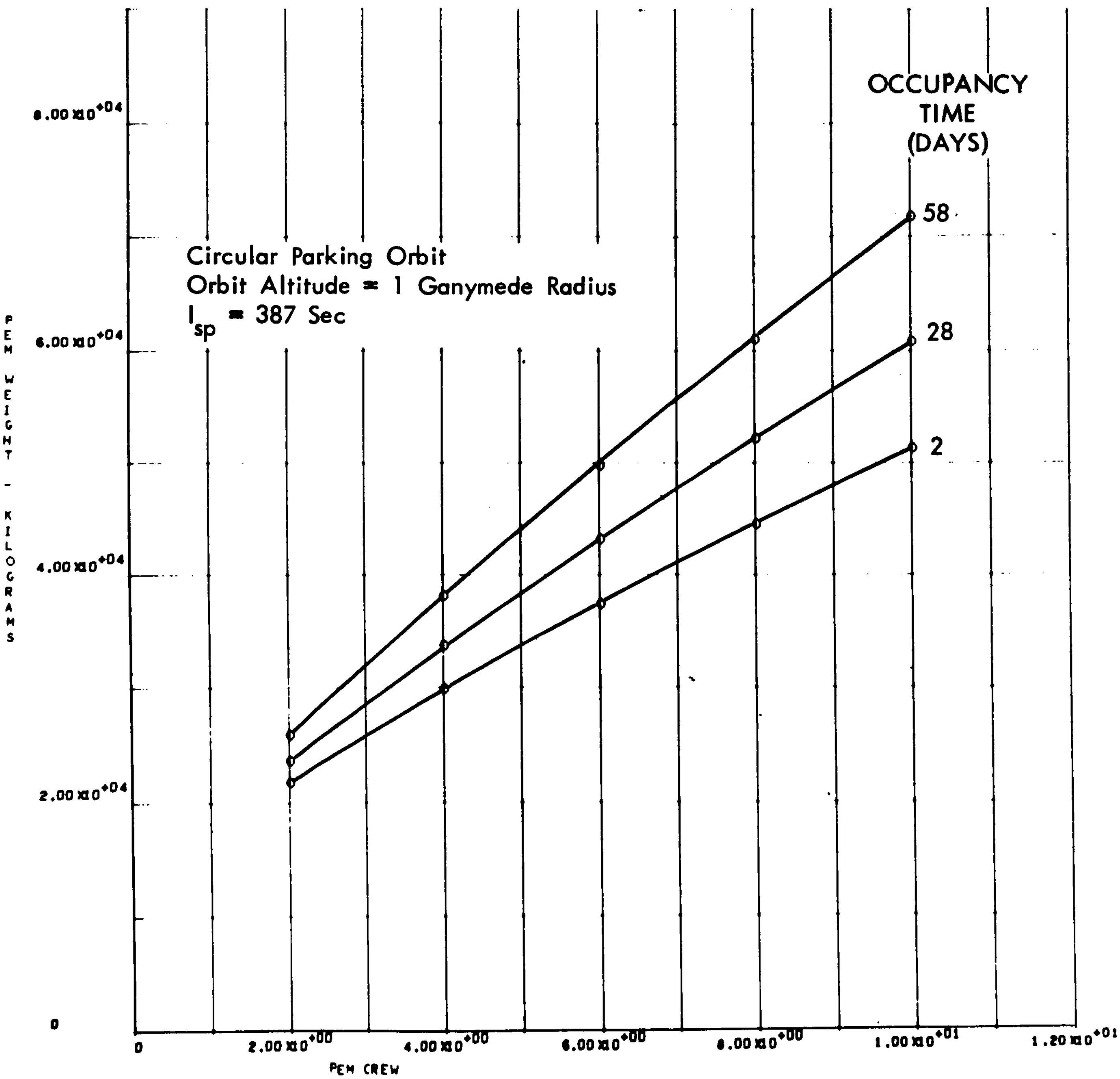

Figure 30. Ganymede Planetary Excursion Module Mass

PROPULSION MODULE

The propulsion module mass requirements were determined as a function of the characteristic velocity and payload mass for chemical (earth storable and cryogenic), solid-core nuclear, and gaseous-core nuclear stages. The mass data presented include the basic structure, engine, propellant, and interstage; the incremental mass requirements for thermal and meteoroid protection are excluded. The effects of thermal and meteoroid protection requirements are included in the data presented in the Mission/System Design section.

The effects of variations in the basic structural mass, the specific impulse, and the gaseous-core nuclear engine thrust-to-weight ratio were evaluated. The nominal structure is based on the assumed post-1980 technology represented by a 50-percent reduction in the required structural mass relative to currently developed stages. The effects of utilizing the current technology and an even more advanced structure 75-percent reduction relative to current technology are shown.

The mass of the chemical modules is presented in Figures 31 through 41 for specific impulse values of 387 and 450 seconds. The lower value is representative of storable propellants, while the upper value represents the expected specific impulse for cryogenic systems during the post-1980 era. The nuclear solid-core data are presented in Figures 42 through 49 for specific impulse values of 800, 820, and 900 seconds. A value of 820 seconds was assumed as the nominal value in all systems synthesis analyses. Specific impulse values of 2,000, 2,500, and 3,000 seconds were considered for the gaseous-core nuclear engines. Results are presented in Figures 50 through 57. A specific impulse of 2,500 seconds and an engine thrust-to-weight ratio of 8 were assumed as the nominal values.

The data presented in Figures 31 through 57 include the gravity losses associated with escape from Earth orbit at an initial parking orbit altitude of 300 kilometers. Although the gravity losses will vary with central body, the effects are not critical and the data are applicable with minor errors for all target bodies considered in the study.

AEROBRAKER

Results of the aerobraker syntheses (Figures 58 through 63) define the mass requirements as a function of the crew size and planetary orbit escape incremental velocity for orbital and landing missions. The Apollo shape was assumed for the earth reentry module in all cases but both the Apollo and Aeronutronic shapes were considered for the Mars excursion module (MEM). Figures 58, 59, and 60 are based on a planetary orbit escape propulsion module specific impulse of 387 seconds. Figures 61 through 63 are based on 450 seconds.

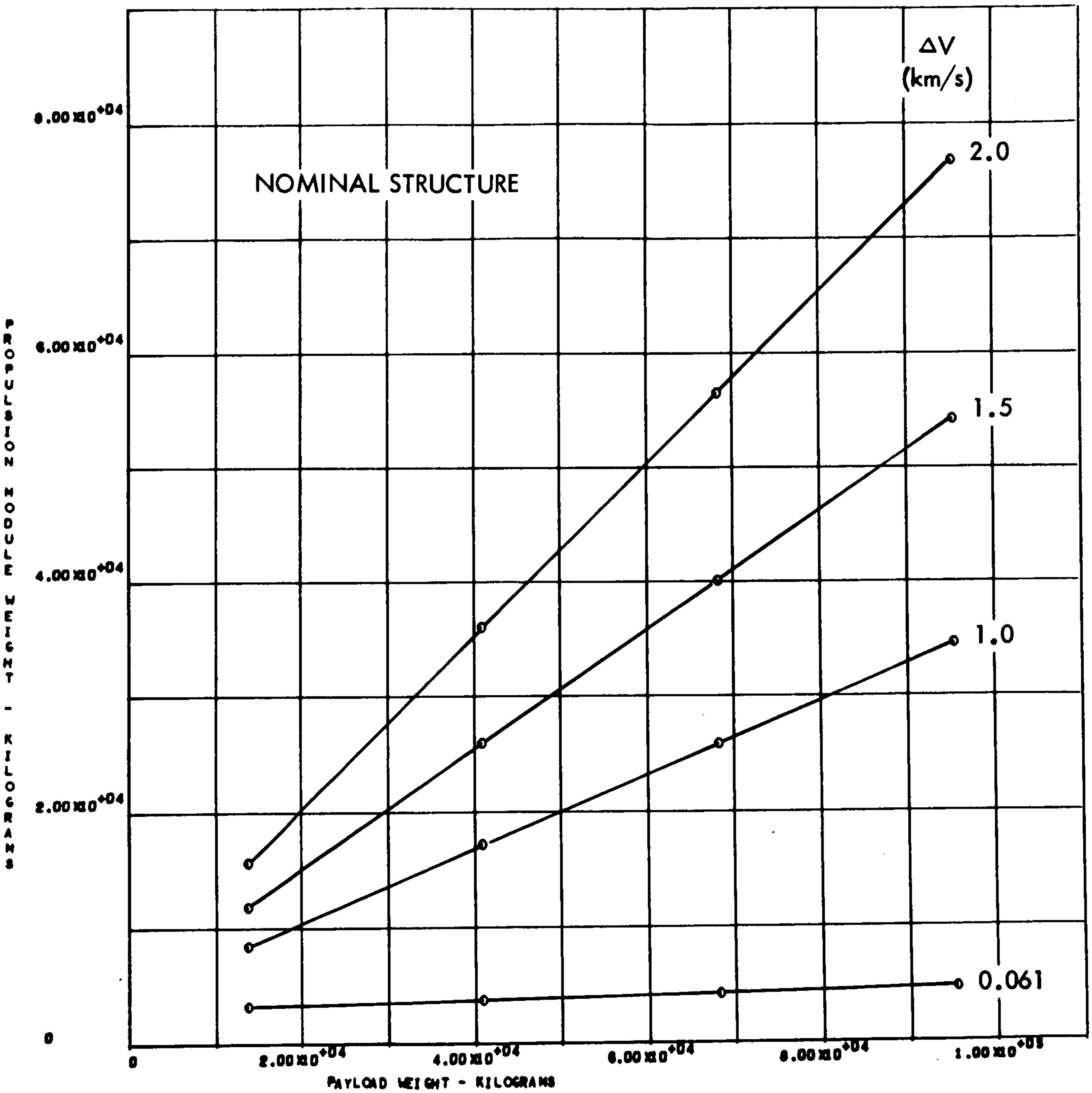

Figure 31. Propulsion Module Mass (I_{sp} = 387 Seconds, ΔV = 0.061 to 2.0 km/s, Maximum Payload = 100,000 kg)

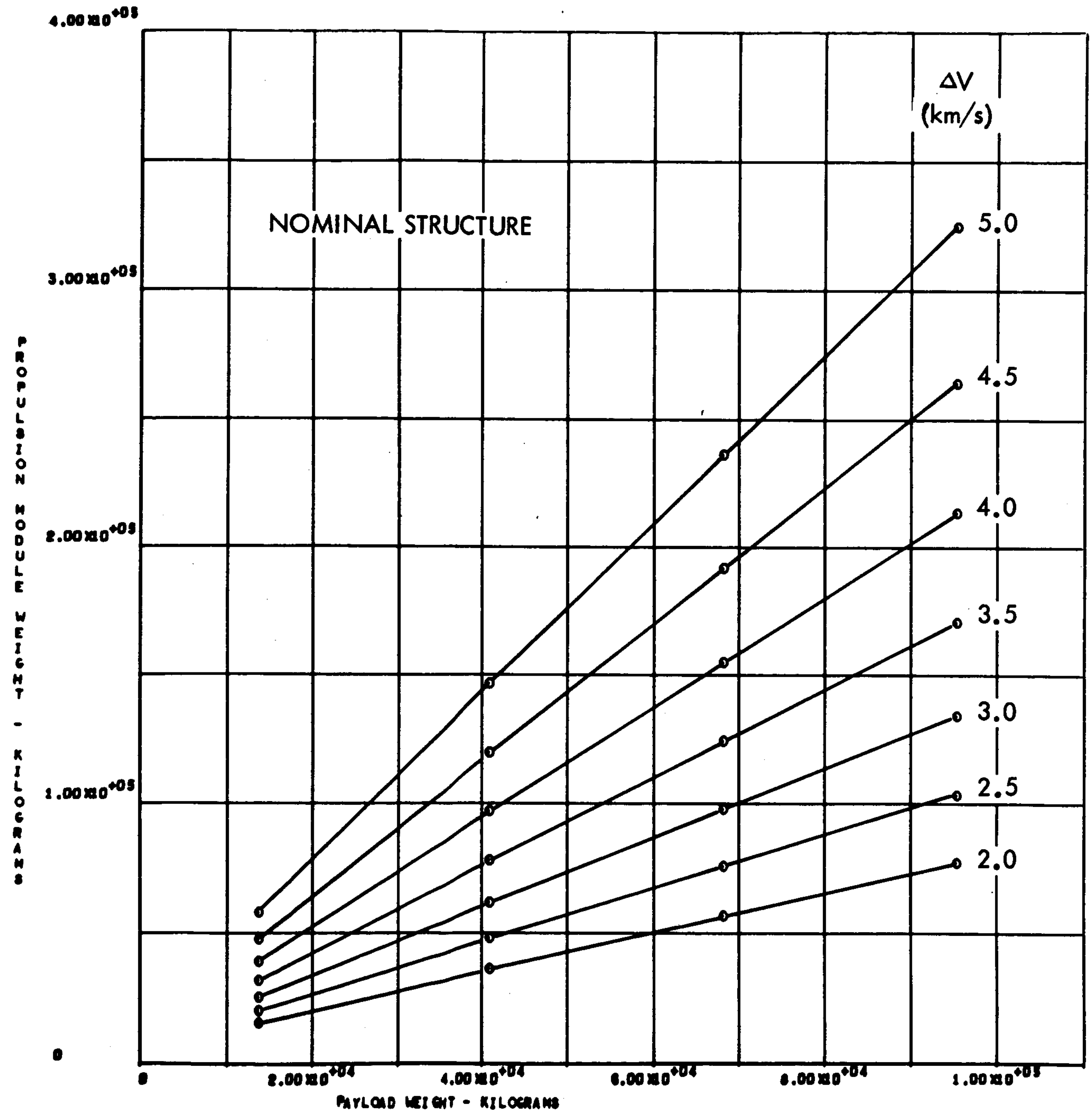

Figure 32. Propulsion Module Mass (I_{sp} = 387 Seconds, ΔV = 2 to 5 km/s, Maximum Payload = 100,000 kg)

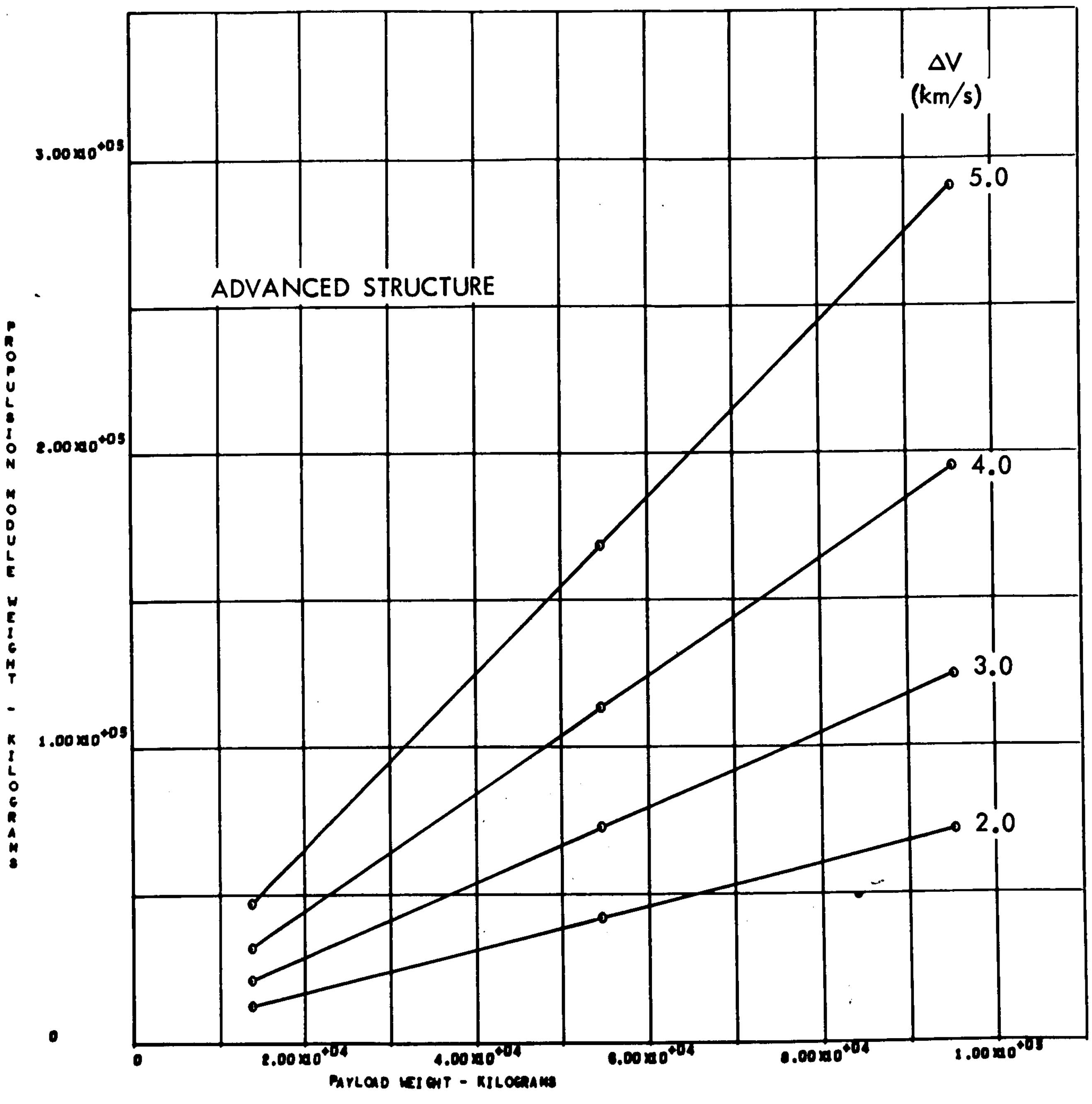

Figure 33. Propulsion Module Mass (I_{sp} = 387 Seconds, ΔV = 2.0 to 5.0 km/s, Maximum Payload = 100,000 kg)

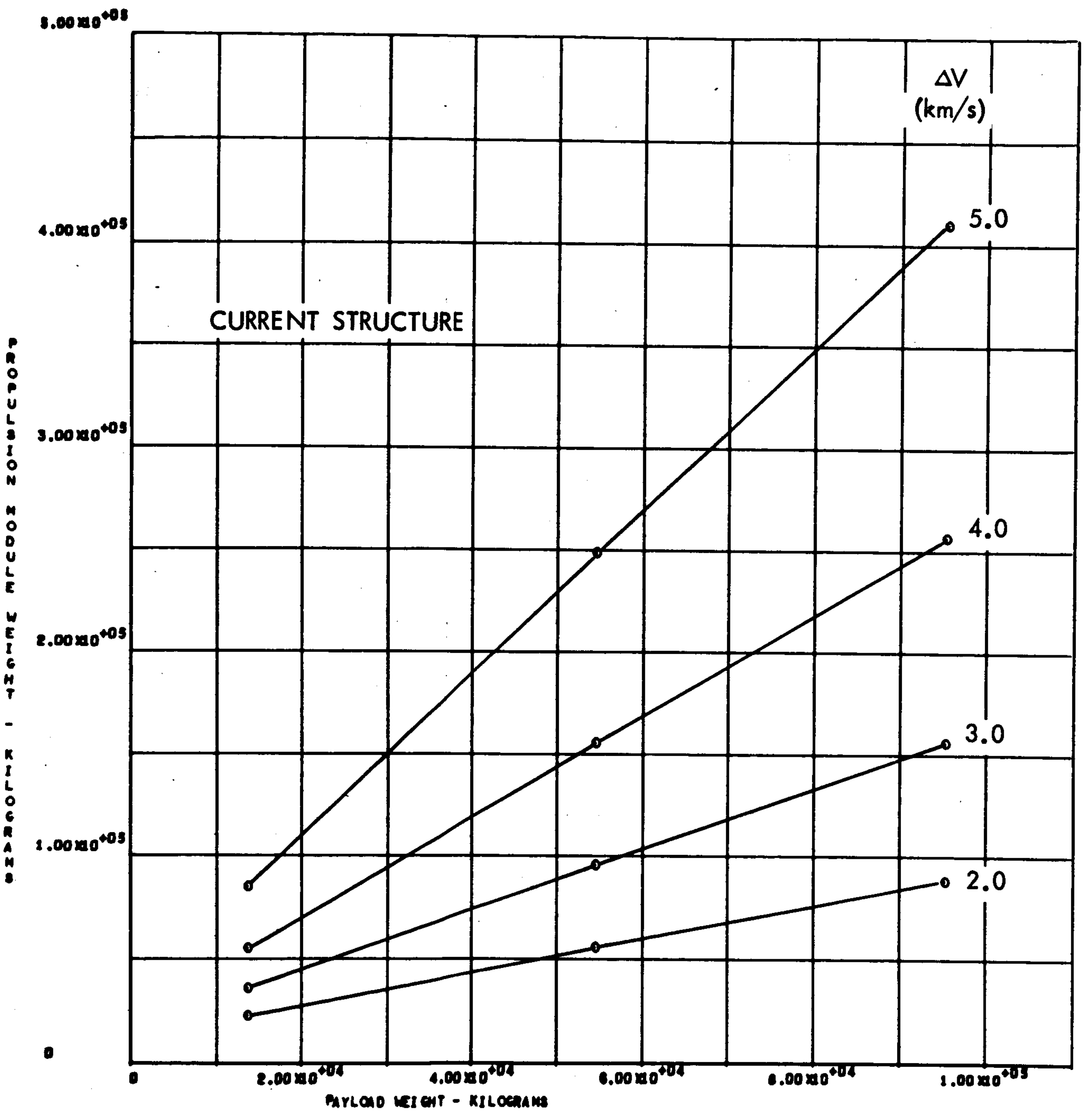

Figure 34. Propulsion Module Mass (I_{sp} = 387 Seconds, ΔV = 2.0 to 5.0 km/s, Maximum Payload = 100,000 kg)

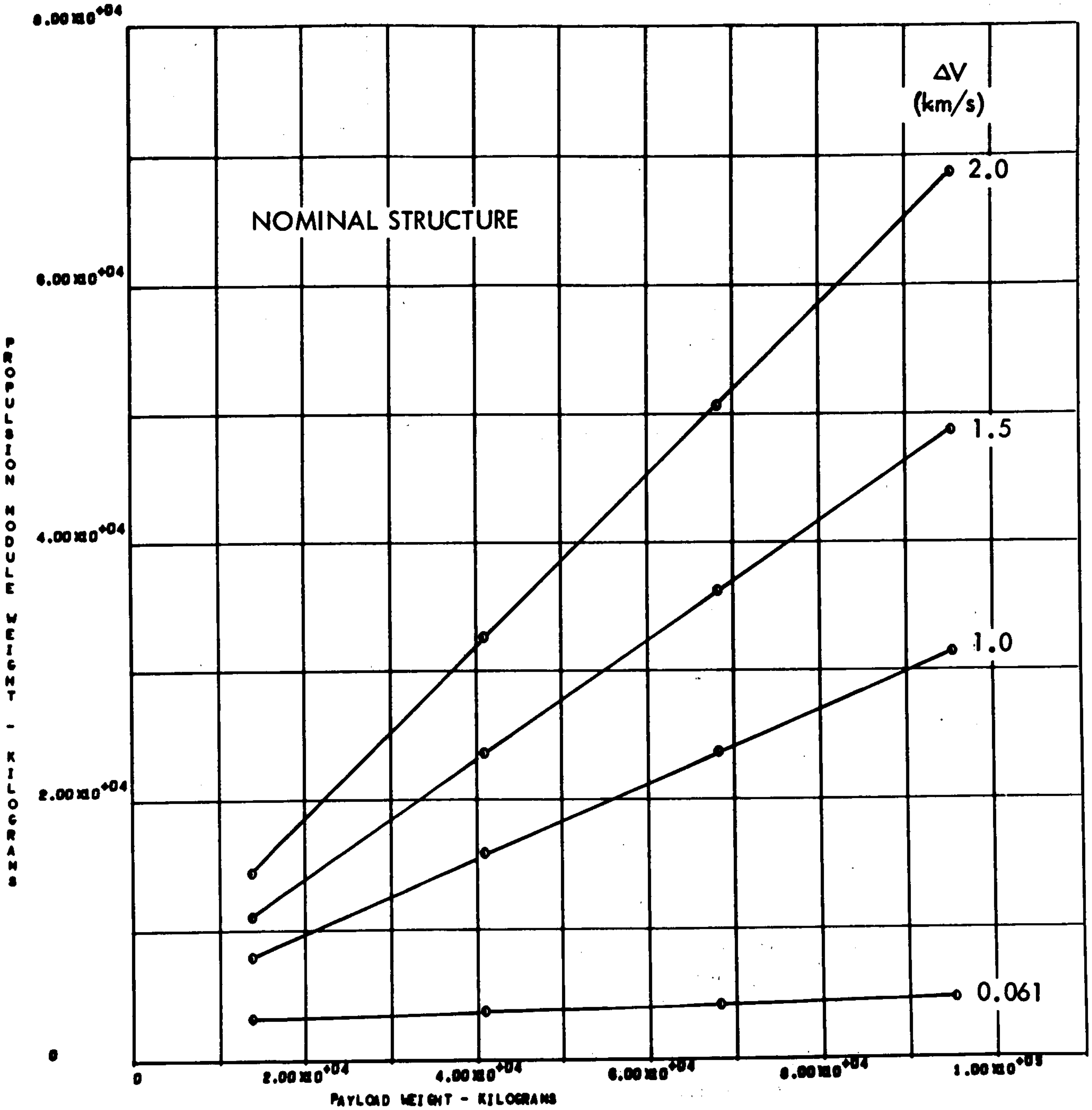

Figure 35. Propulsion Module Mass (I_{sp} = 450 Seconds, ΔV = 0.061 to 2.0 km/s, Maximum Payload = 100,000 kg)

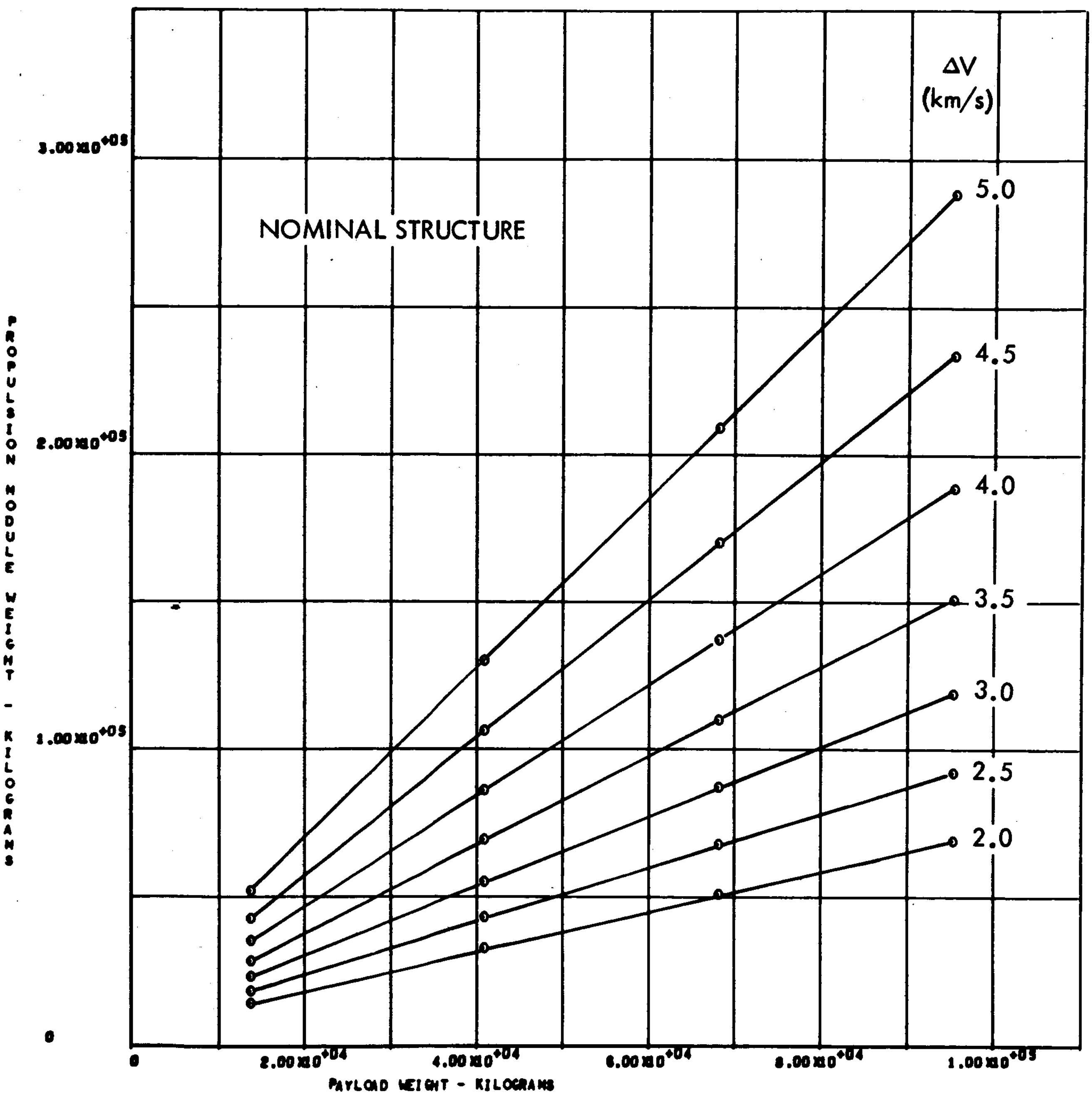

Figure 36. Propulsion Module Mass (I_{sp} = 450 Seconds, ΔV = 2.0 to 5.0 km/s, Maximum Payload = 100,000 kg)

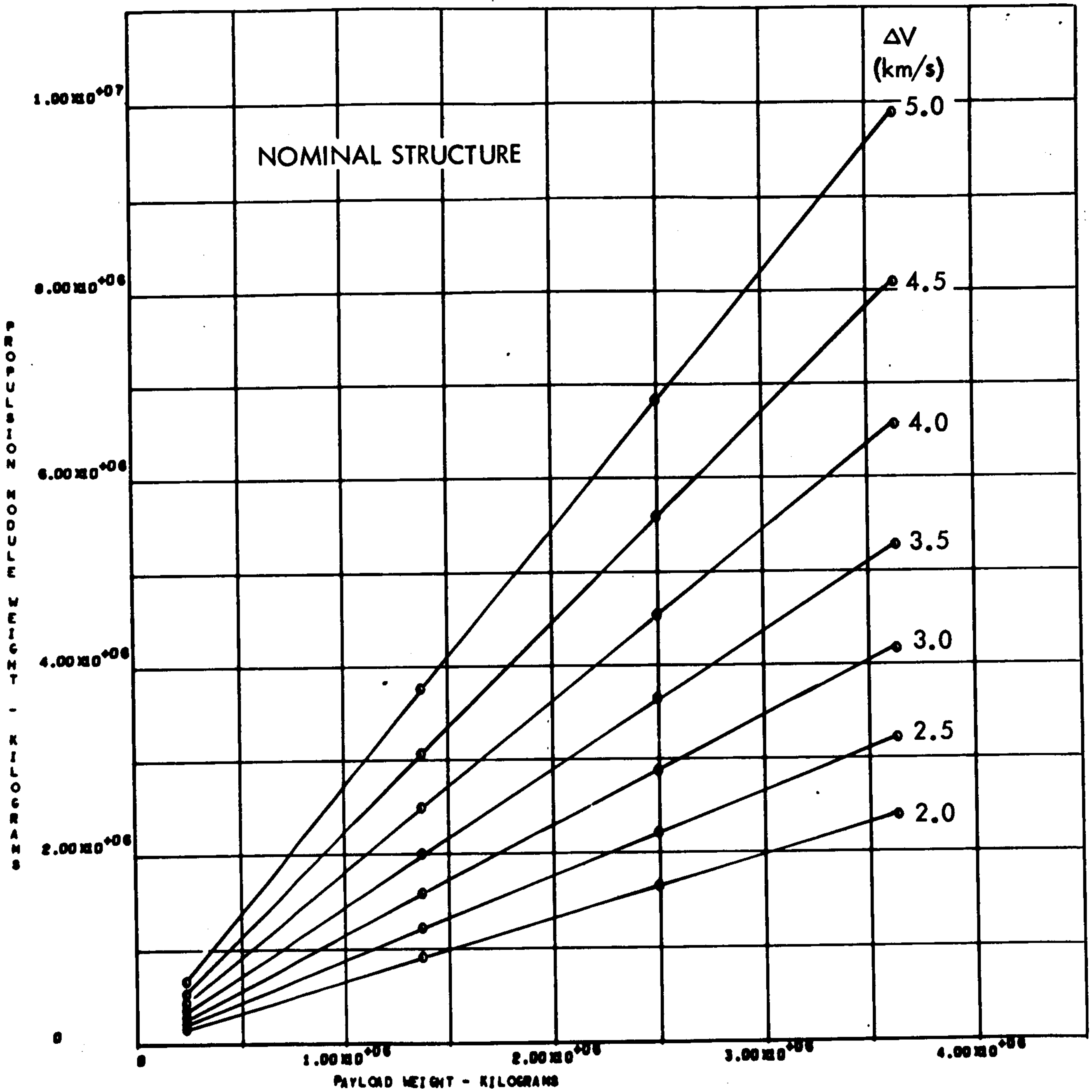

Figure 37. Propulsion Module Mass (I_{sp} = 450 Seconds, ΔV = 2.0 to 5.0 km/s, Maximum Payload = 4,000,000 kg)

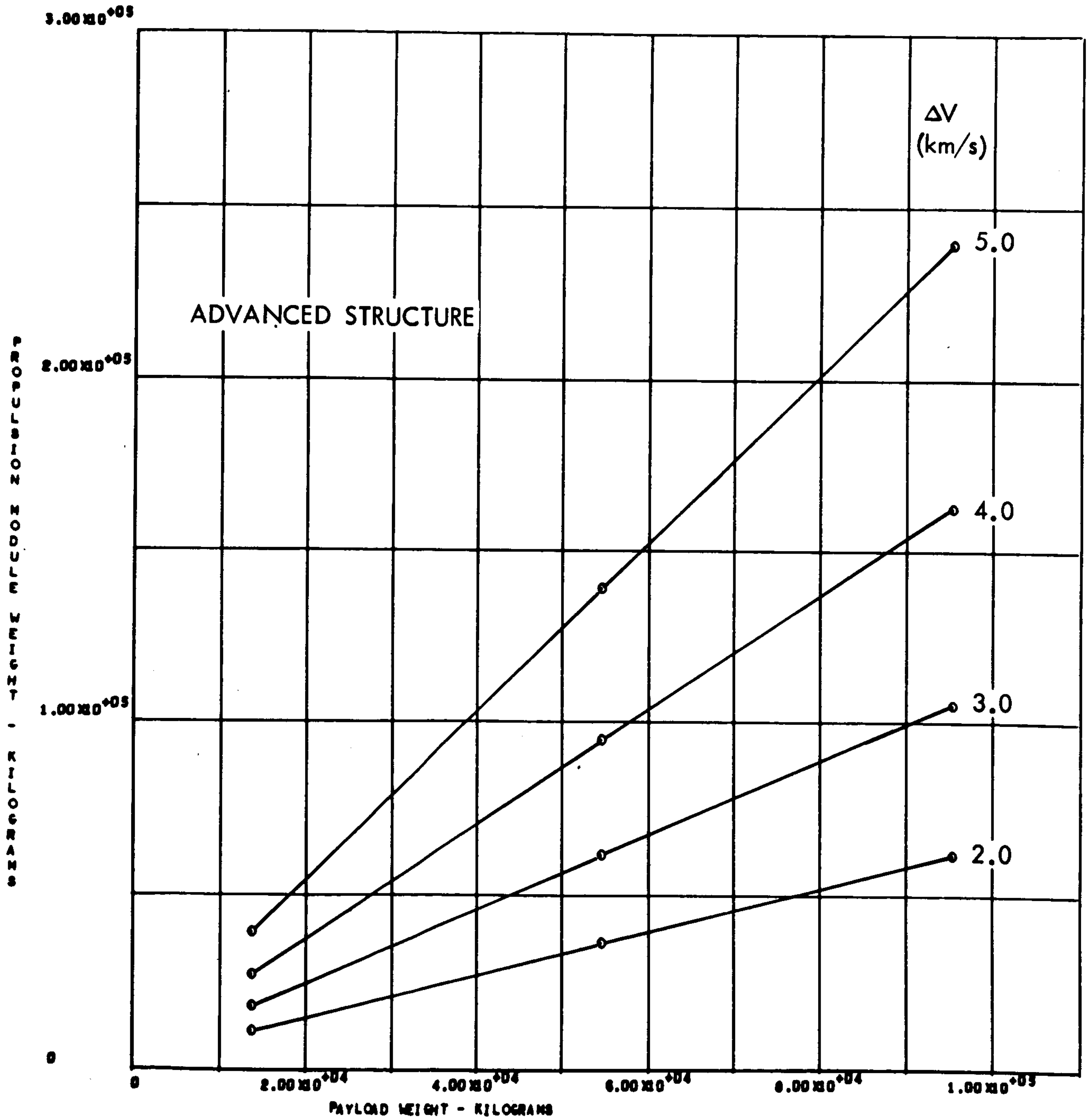

Figure 38. Propulsion Module Mass (I_{sp} = 450 Seconds, ΔV = 2.0 to 5.0 km/s, Maximum Payload = 100,000 kg)

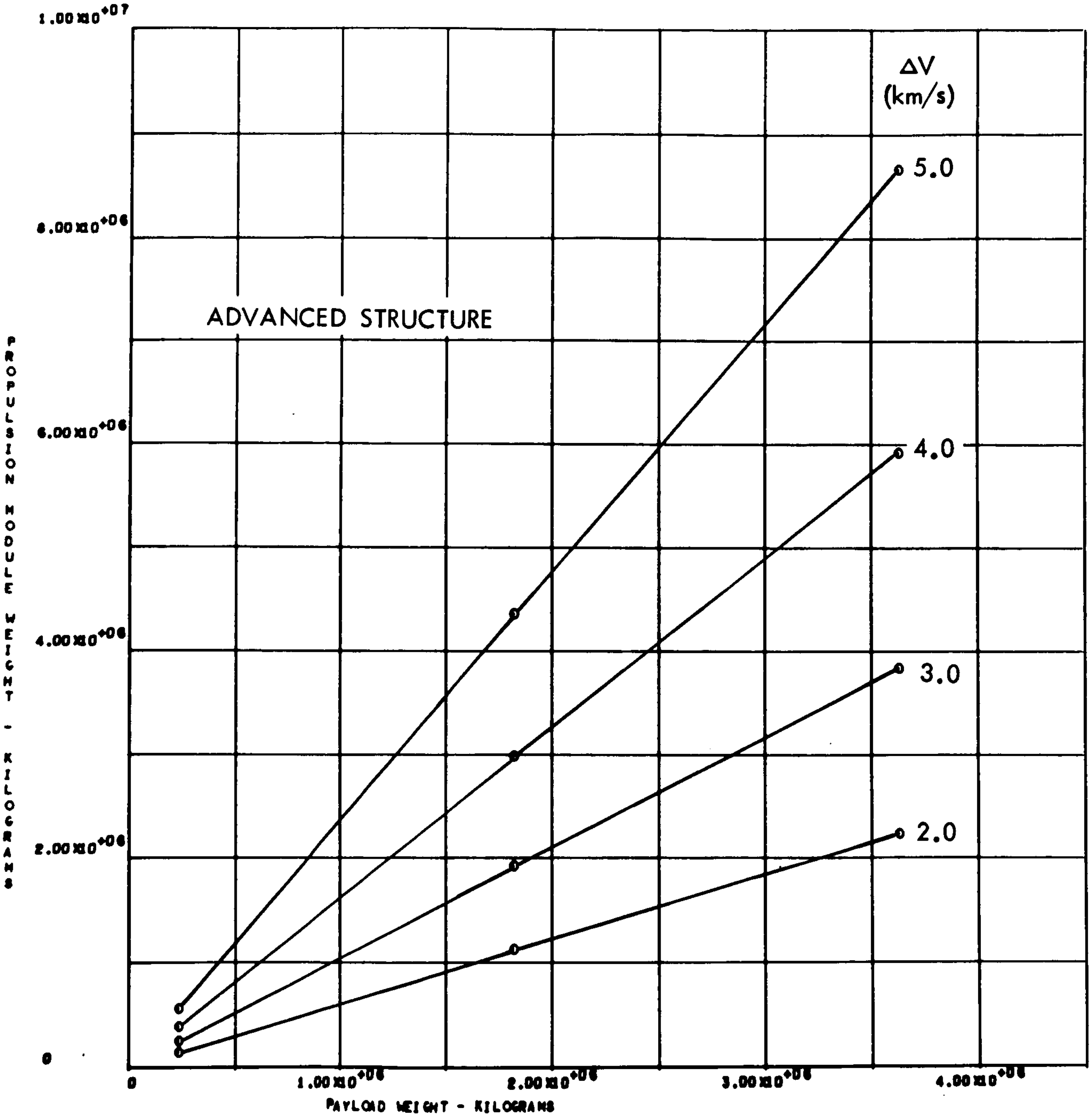

Figure 39. Propulsion Module Mass (I_{sp} = 450 Seconds, ΔV = 2.0 to 5.0 km/s, Maximum Payload = 4,000,000 kg)

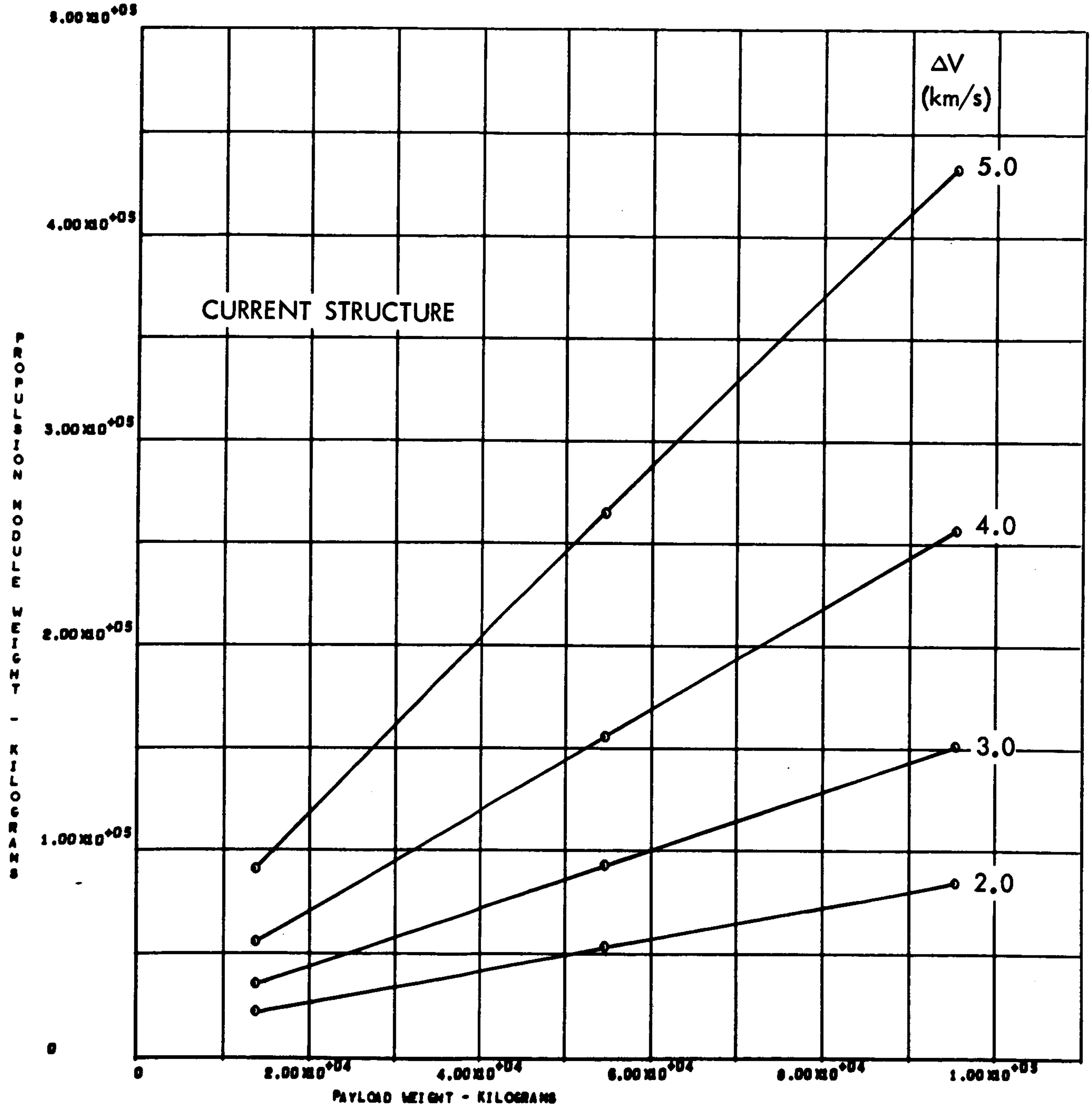

Figure 40. Propulsion Module Mass (I_{sp} = 450 Seconds, ΔV = 2.0 to 5.0 km/s, Maximum Payload = 100,000 kg)

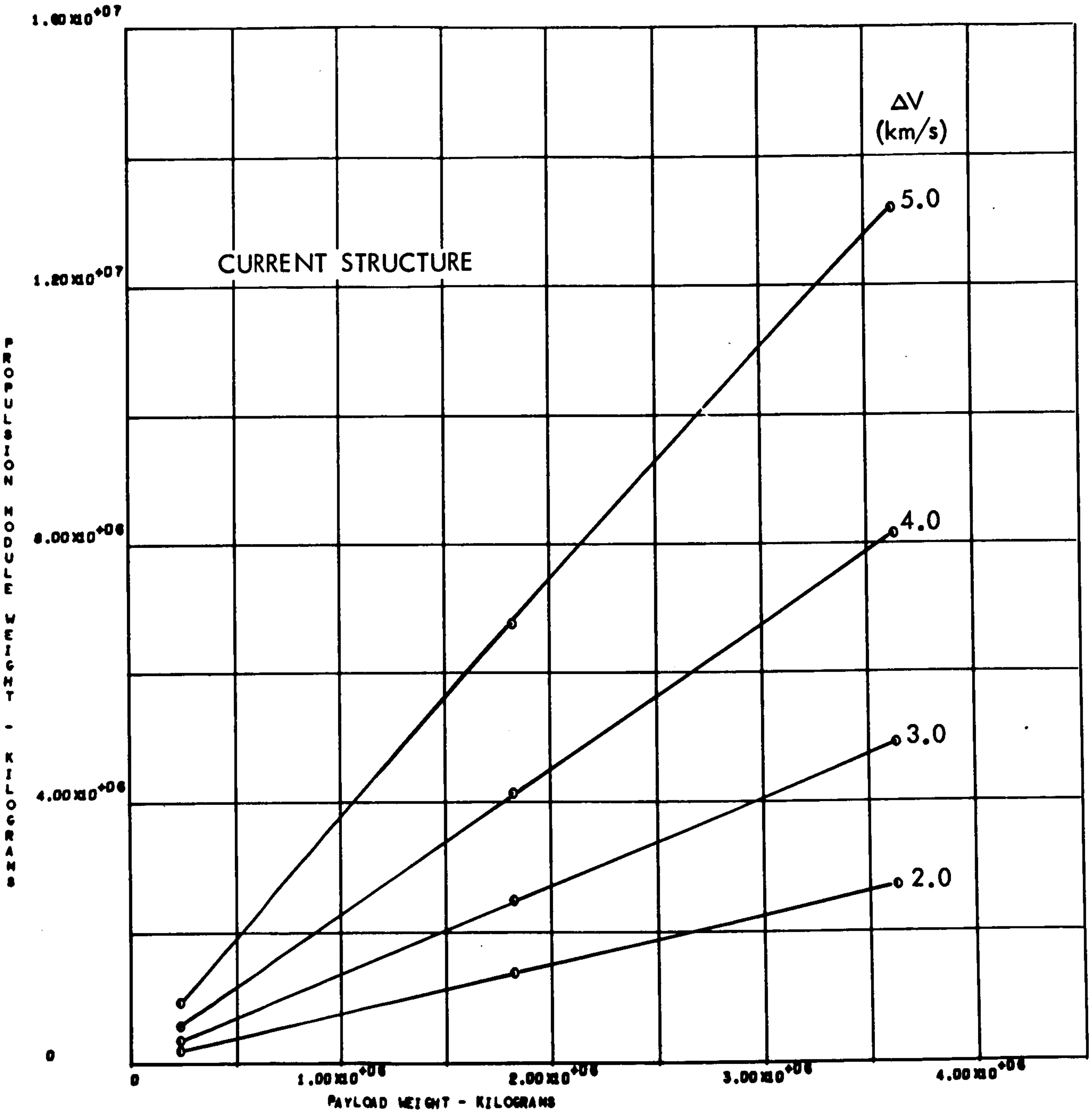

Figure 41. Propulsion Module Mass (I_{sp} = 450 Seconds, ΔV = 2.0 to 5.0 km/s, Maximum Payload = 4,000,000 kg)

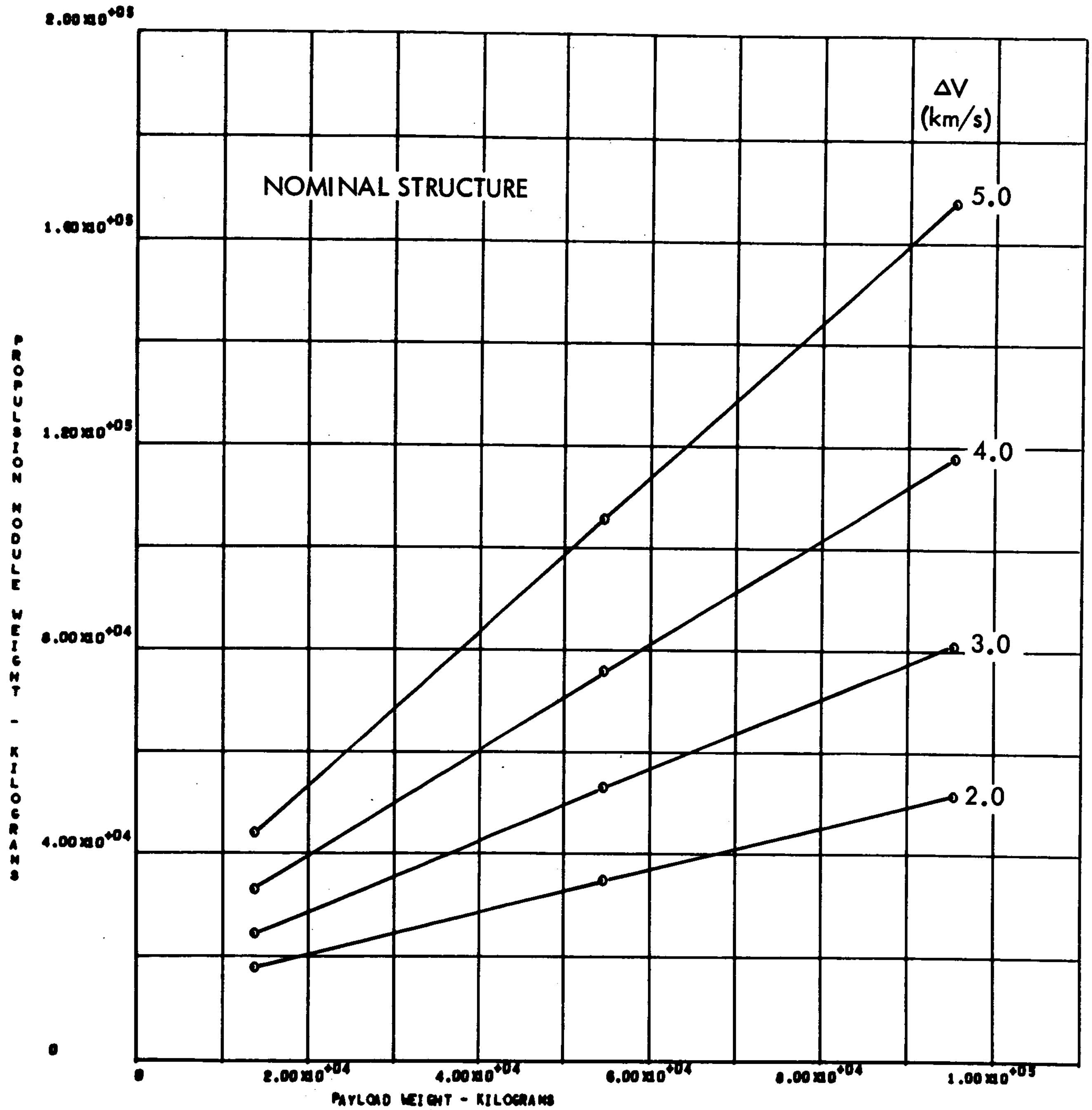

Figure 42. Propulsion Module Mass (I_{sp} = 800 Seconds, ΔV = 2.0 to 5.0 km/s, Maximum Payload = 100,000 kg)

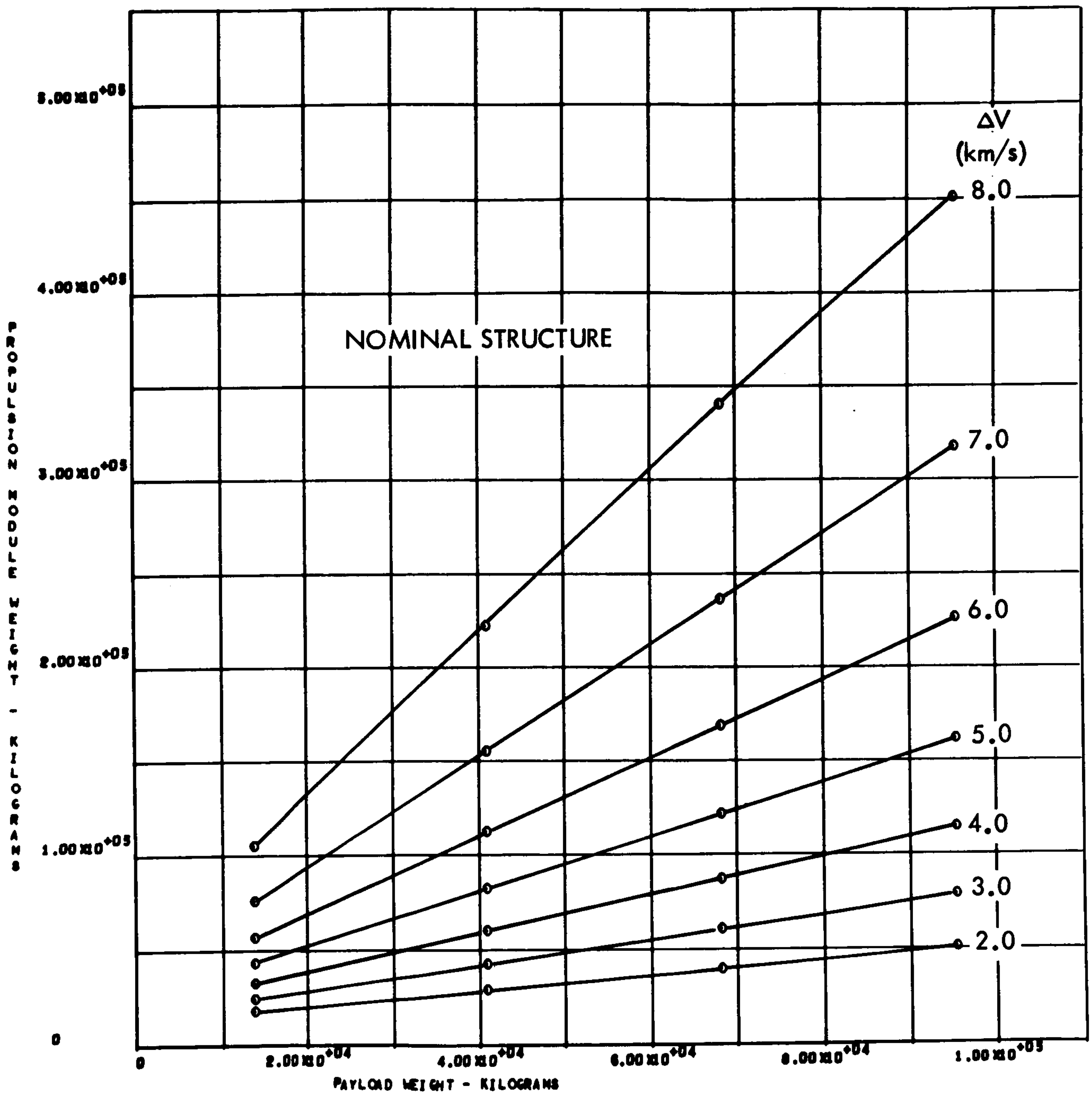

Figure 43. Propulsion Module Mass (I_{sp} = 820 Seconds, ΔV = 2.0 to 8.0 km/s, Maximum Payload = 100,000 kg)

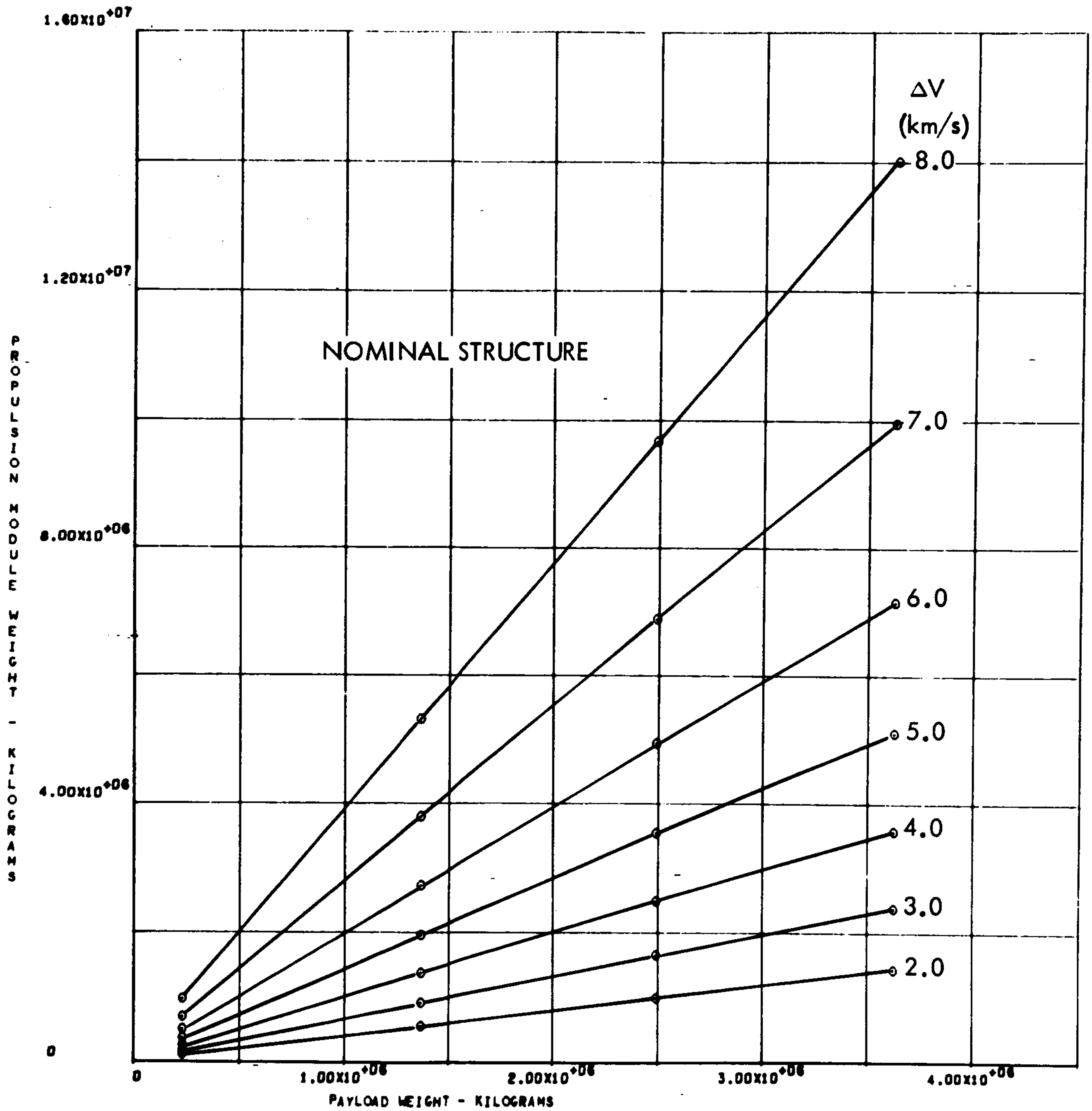

Figure 44. Propulsion Module Mass (I_{sp} = 820 Seconds, ΔV = 2.0 to 8.0 km/s, Maximum Payload = 4,000,000 kg)

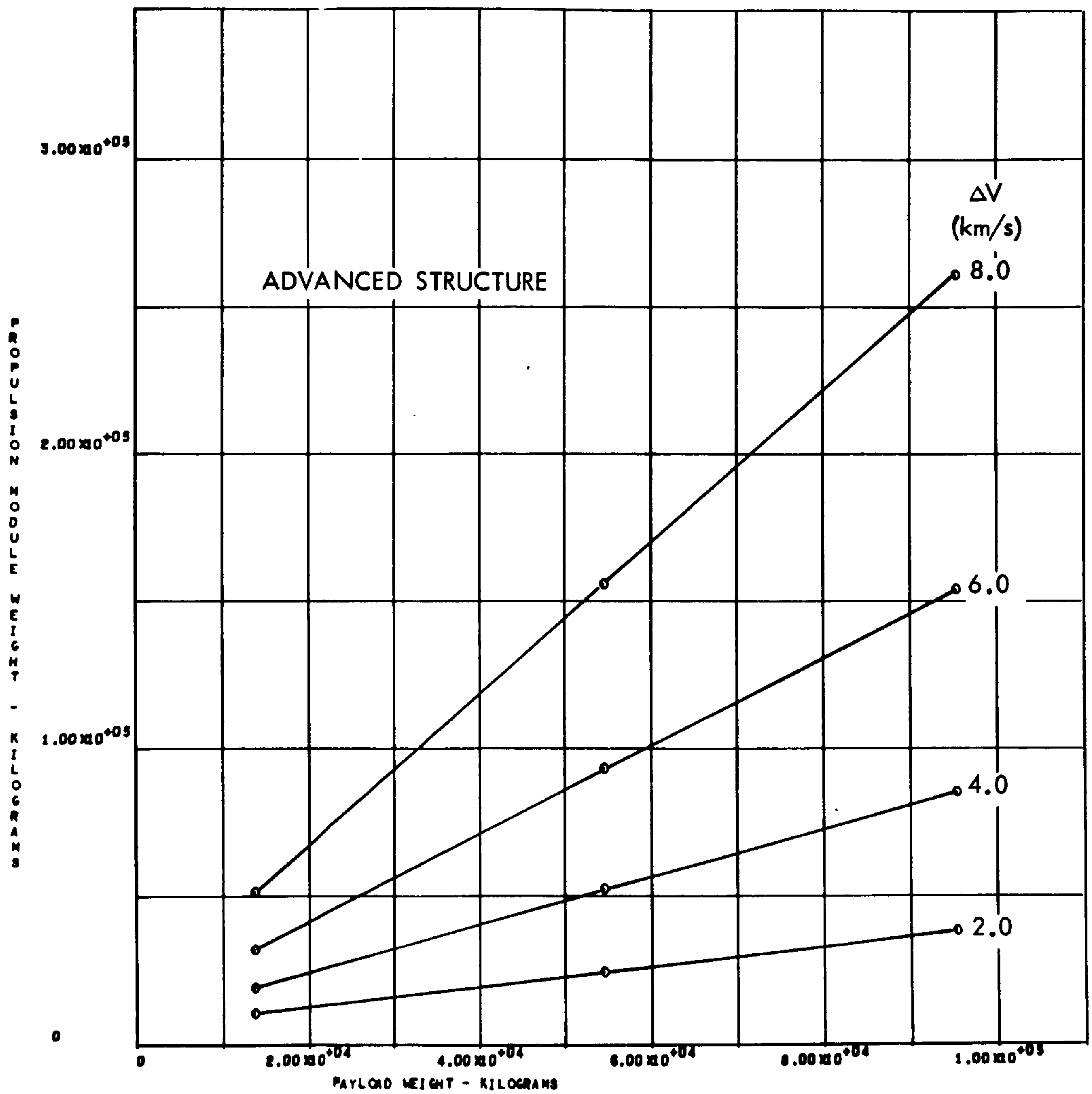

Figure 45. Propulsion Module Mass (I_{sp} = 820 Seconds, ΔV = 2.0 to 8.0 km/s, Maximum Payload = 100,000 kg)

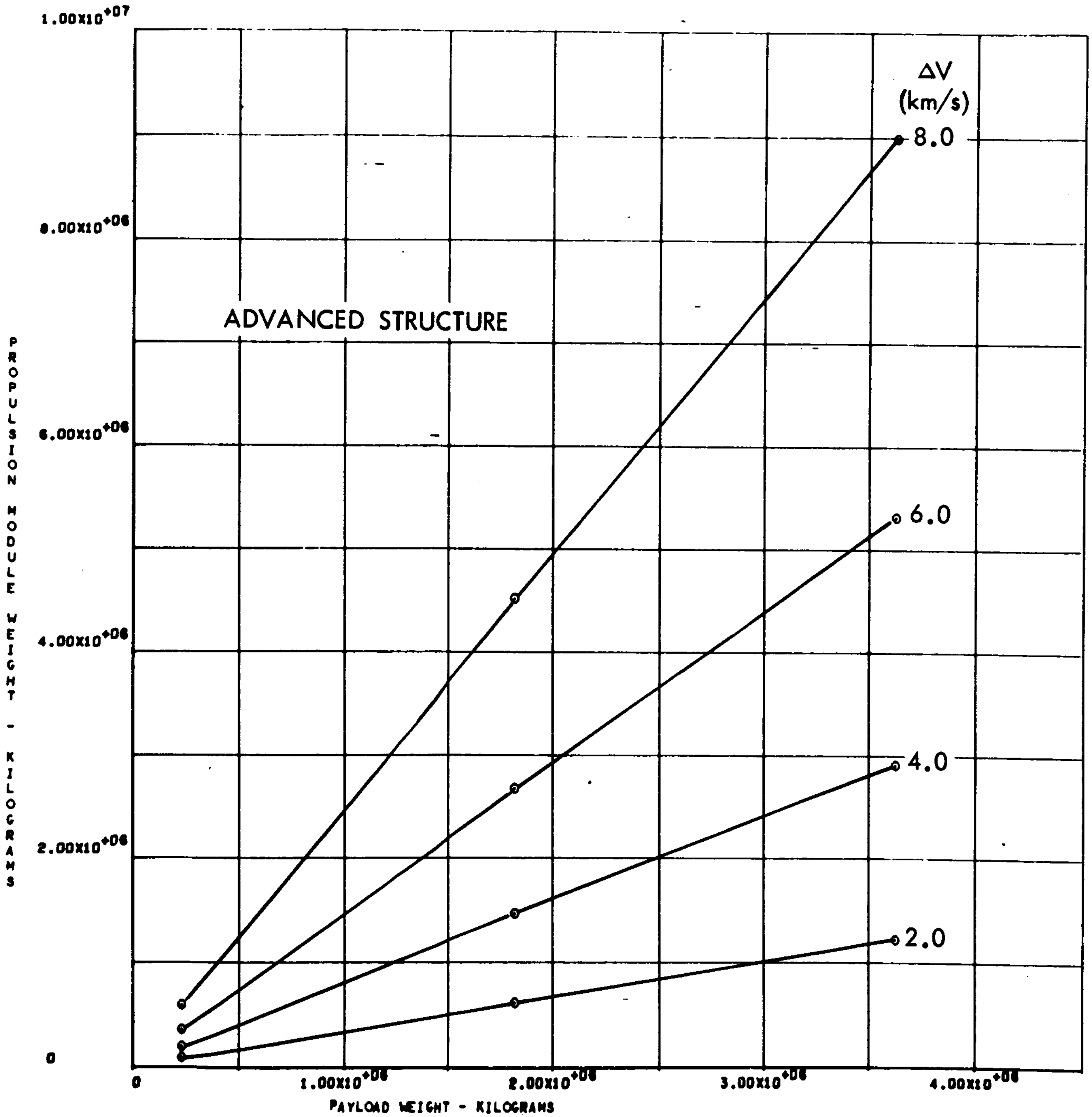

Figure 46. Propulsion Module Mass (I_{sp} = 820 Seconds, ΔV = 2.0 to 8.0 km/s, Maximum Payload = 4,000,000 kg)

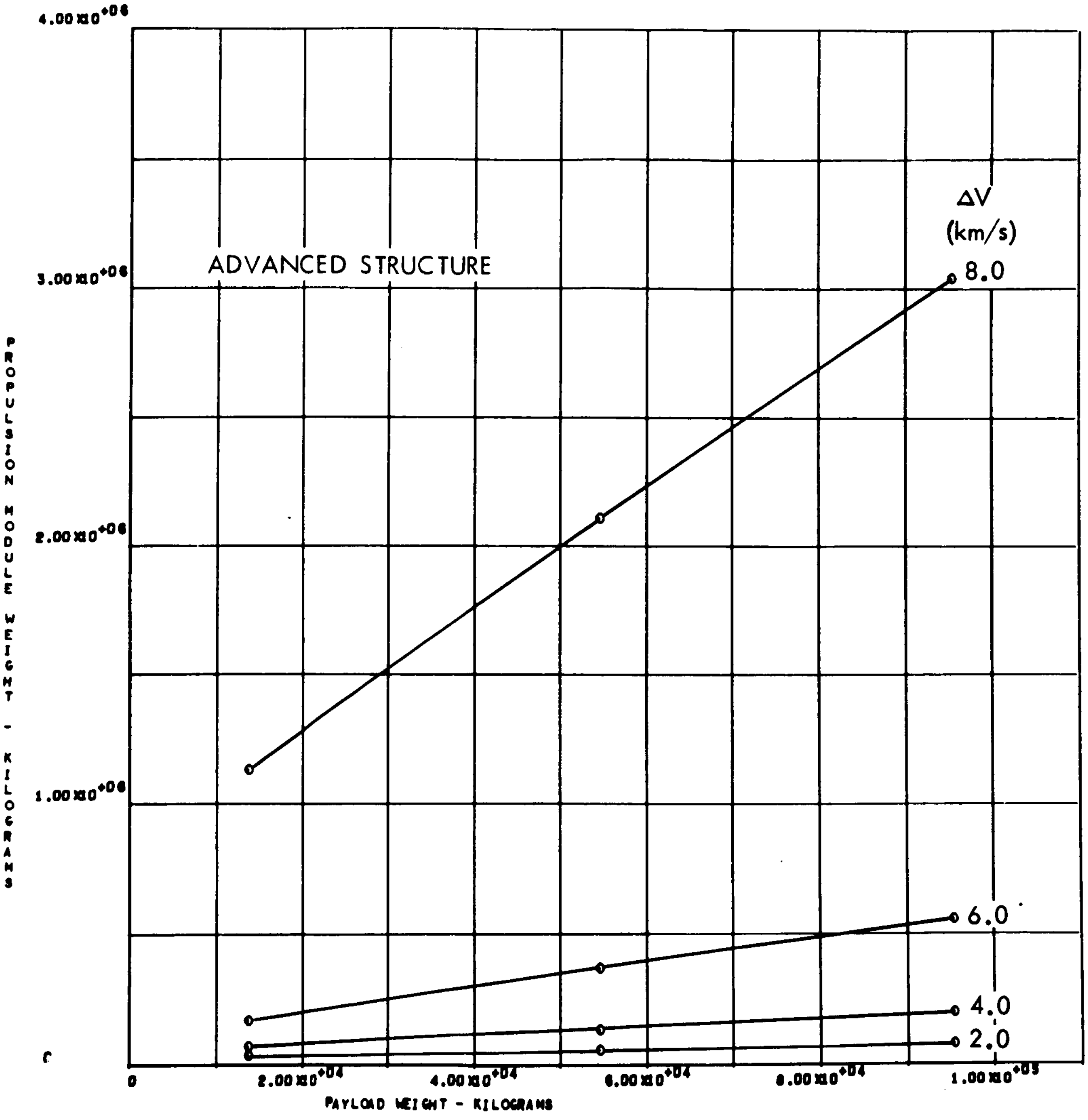

Figure 47. Propulsion Module Mass (I_{sp} = 820 Seconds, ΔV = 2.0 to 8.0 km/s, Maximum Payload = 100,000 kg)

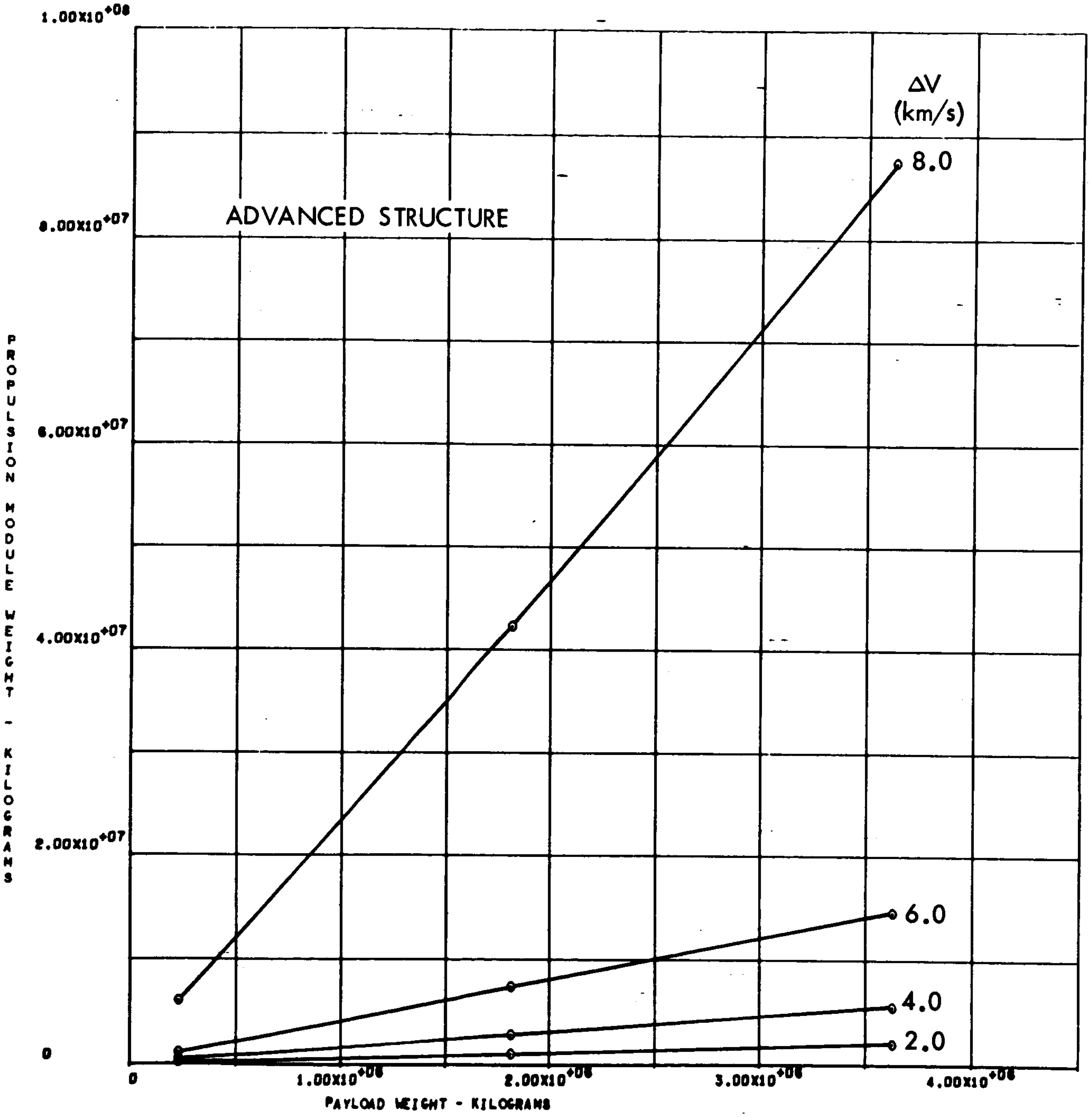

Figure 48. Propulsion Module Mass (I_{sp} = 820 Seconds, ΔV = 2.0 to 8.0 km/s, Maximum Payload = 4,000,000 kg)

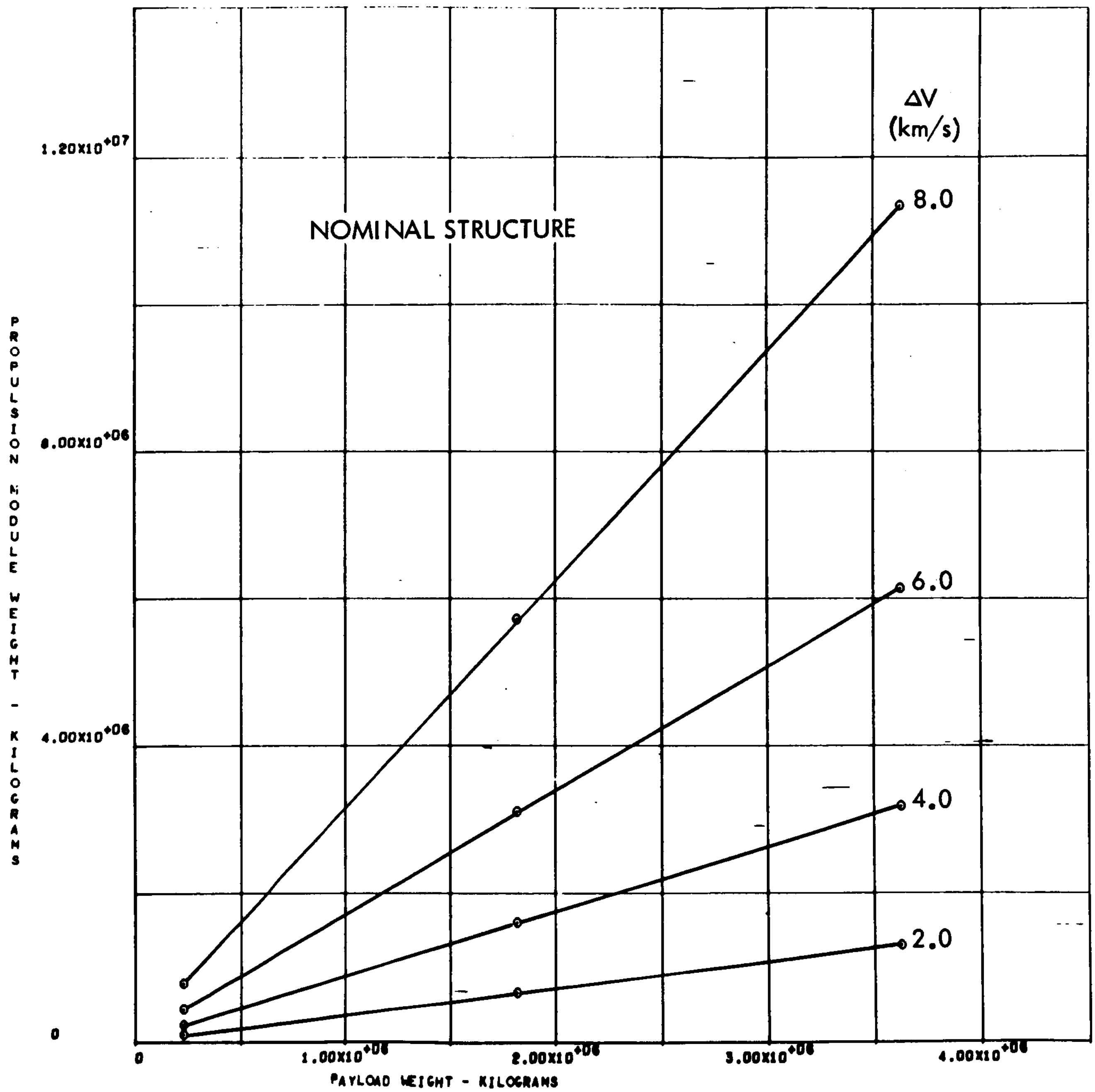

Figure 49. Propulsion Module Mass (I_{sp} = 900 Seconds, ΔV = 2.0 to 8.0 km/s, Maximum Payload = 4,000,000 kg)

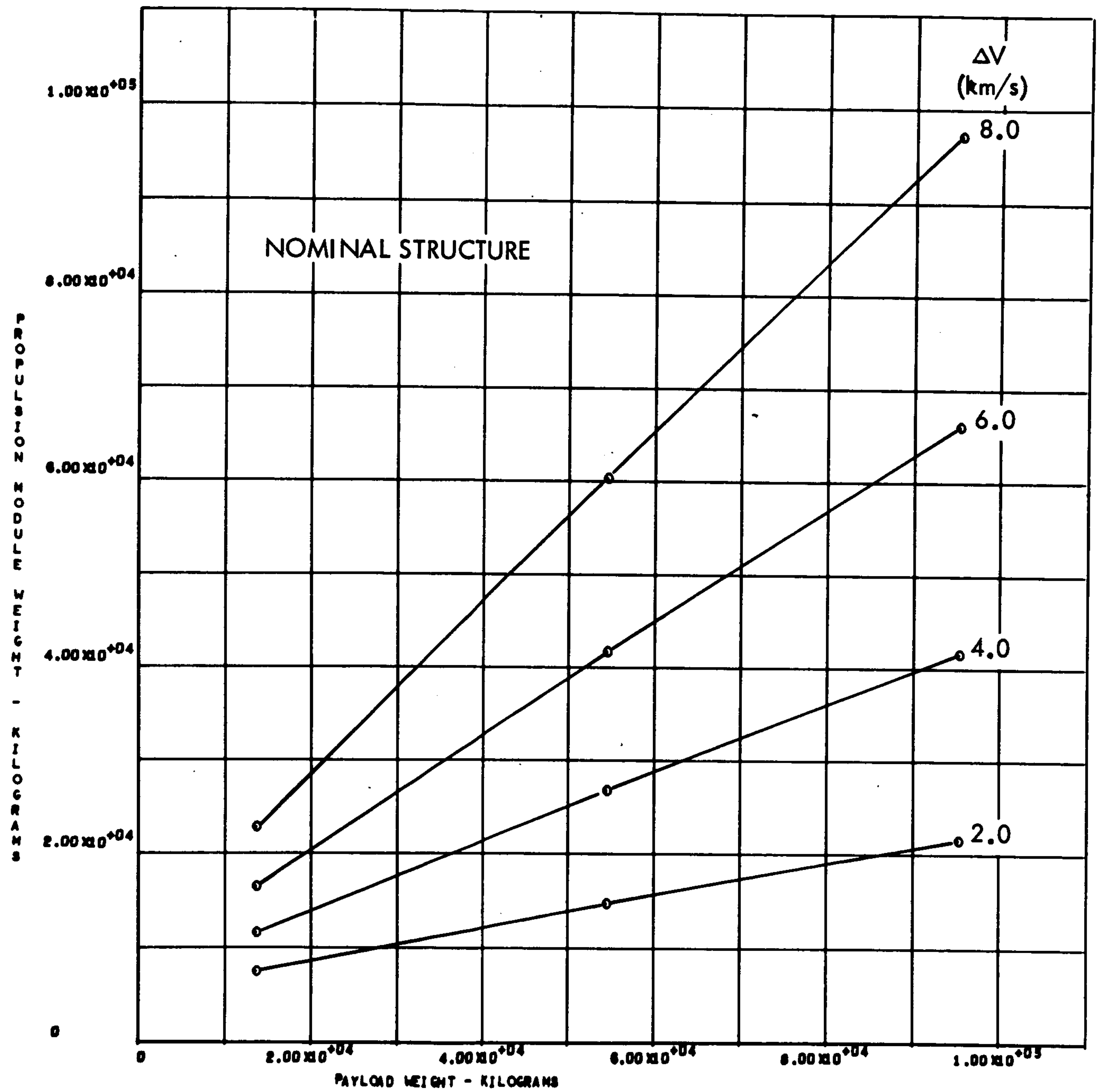

Figure 50. Propulsion Module Mass (I_{sp} = 2000 Seconds, T/W_e = 8.0, Maximum Payload = 100, 000 kg)

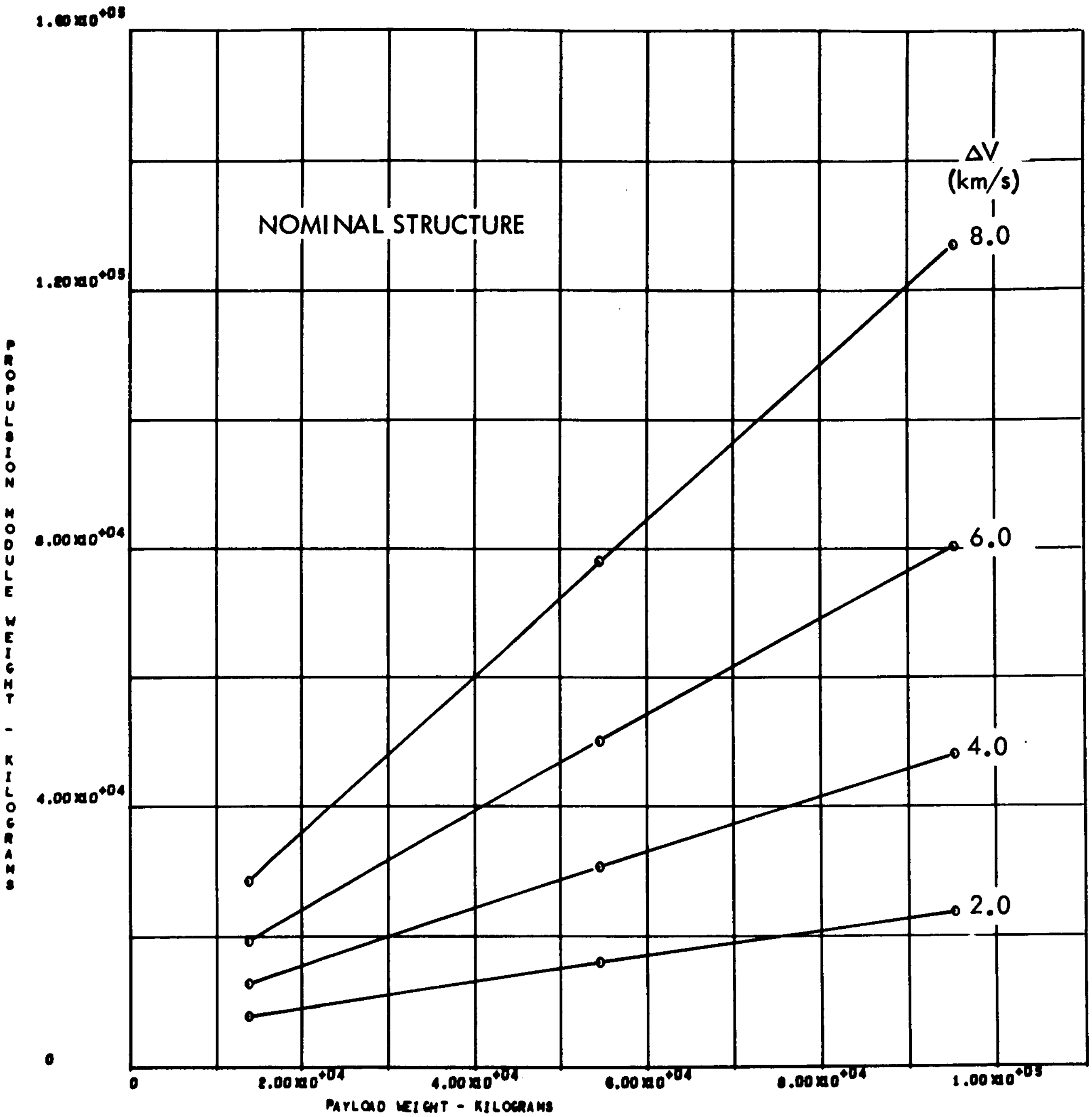

Figure 51. Propulsion Module Mass (I_{sp} = 2500 Seconds, T/W_e = 3.0, Maximum Payload = 100,000 kg)

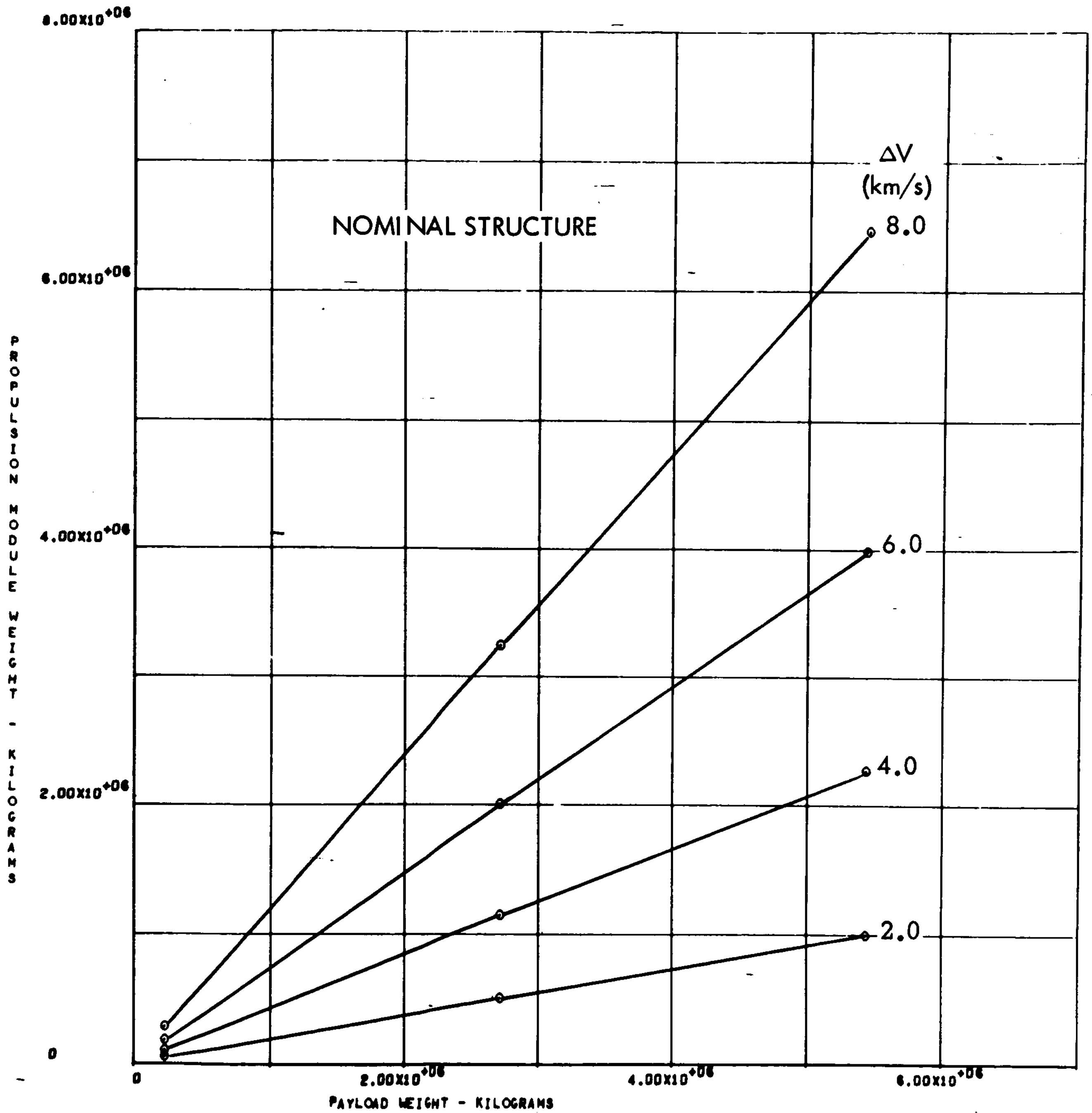

Figure 52. Propulsion Module Mass (I_{sp} = 2500 Seconds, T/W_e = 3.0, Maximum Payload = 6, 000, 000 kg)

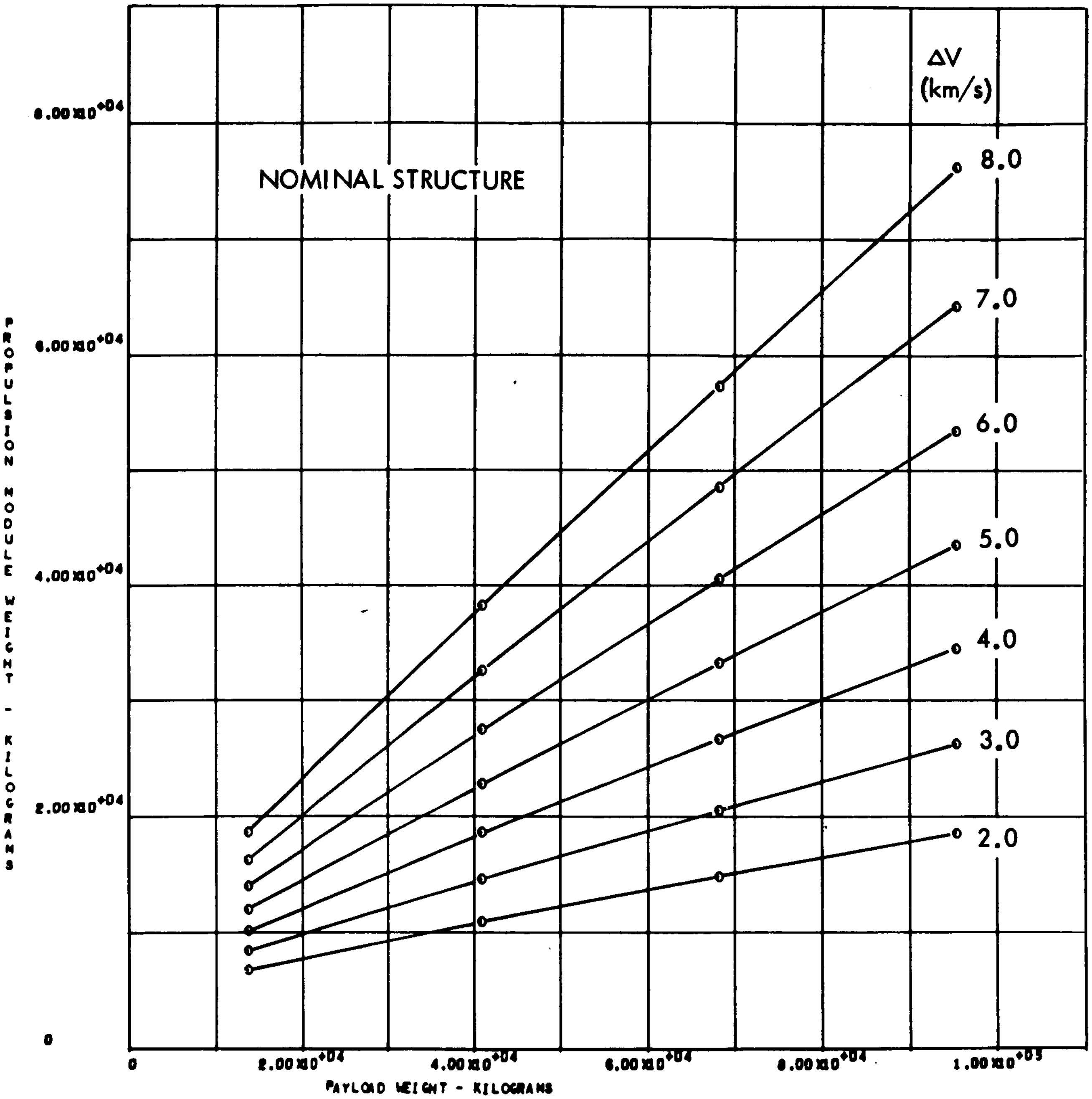

Figure 53. Propulsion Module Mass (I_{sp} = 2500 Seconds, T/W_e = 8.0, Maximum Payload = 100,000 kg)

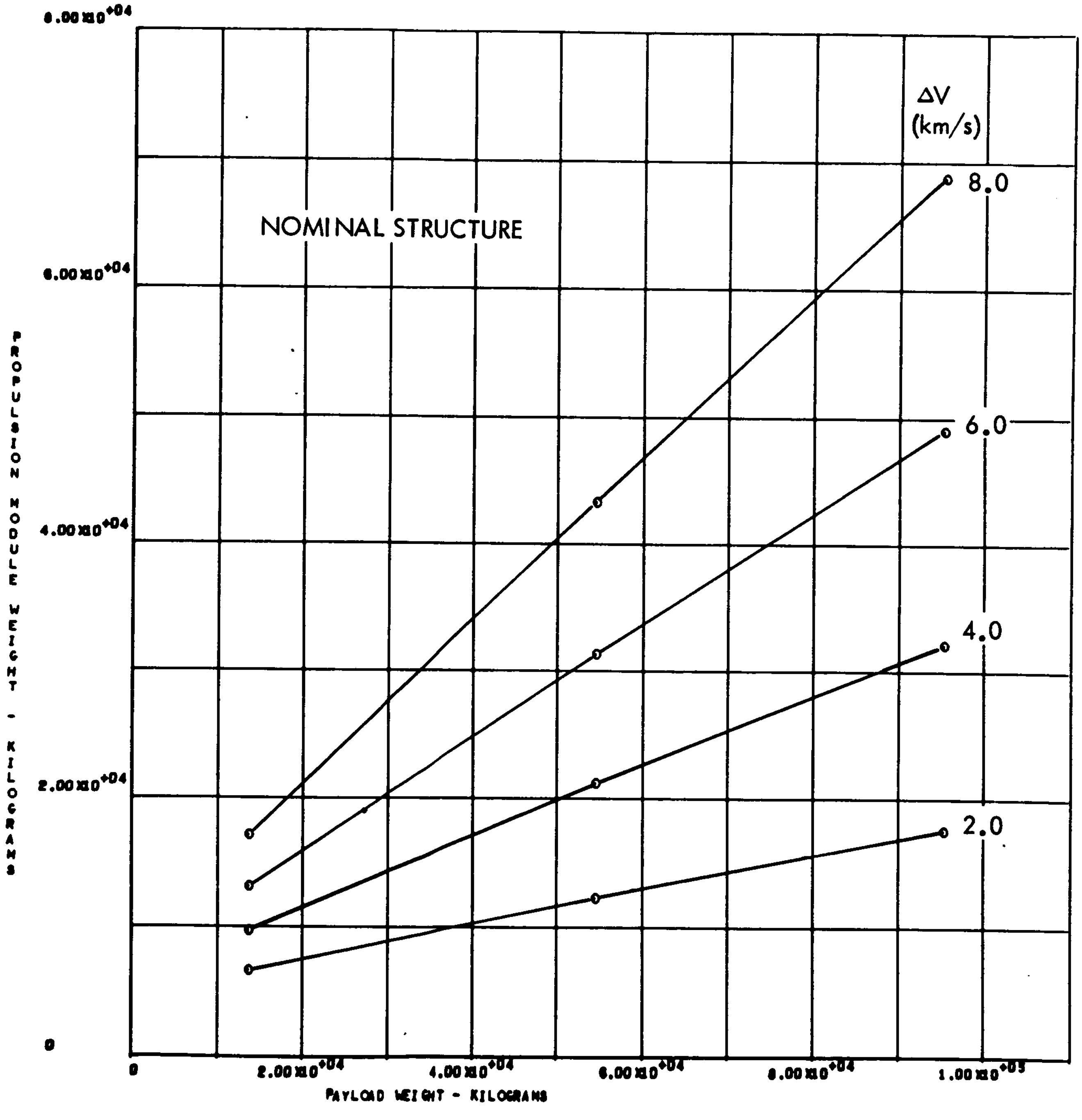

Figure 54. Propulsion Module Mass (I_{sp} = 2500 Seconds, T/W_e = 12.0 Maximum Payload = 100, 000 kg)

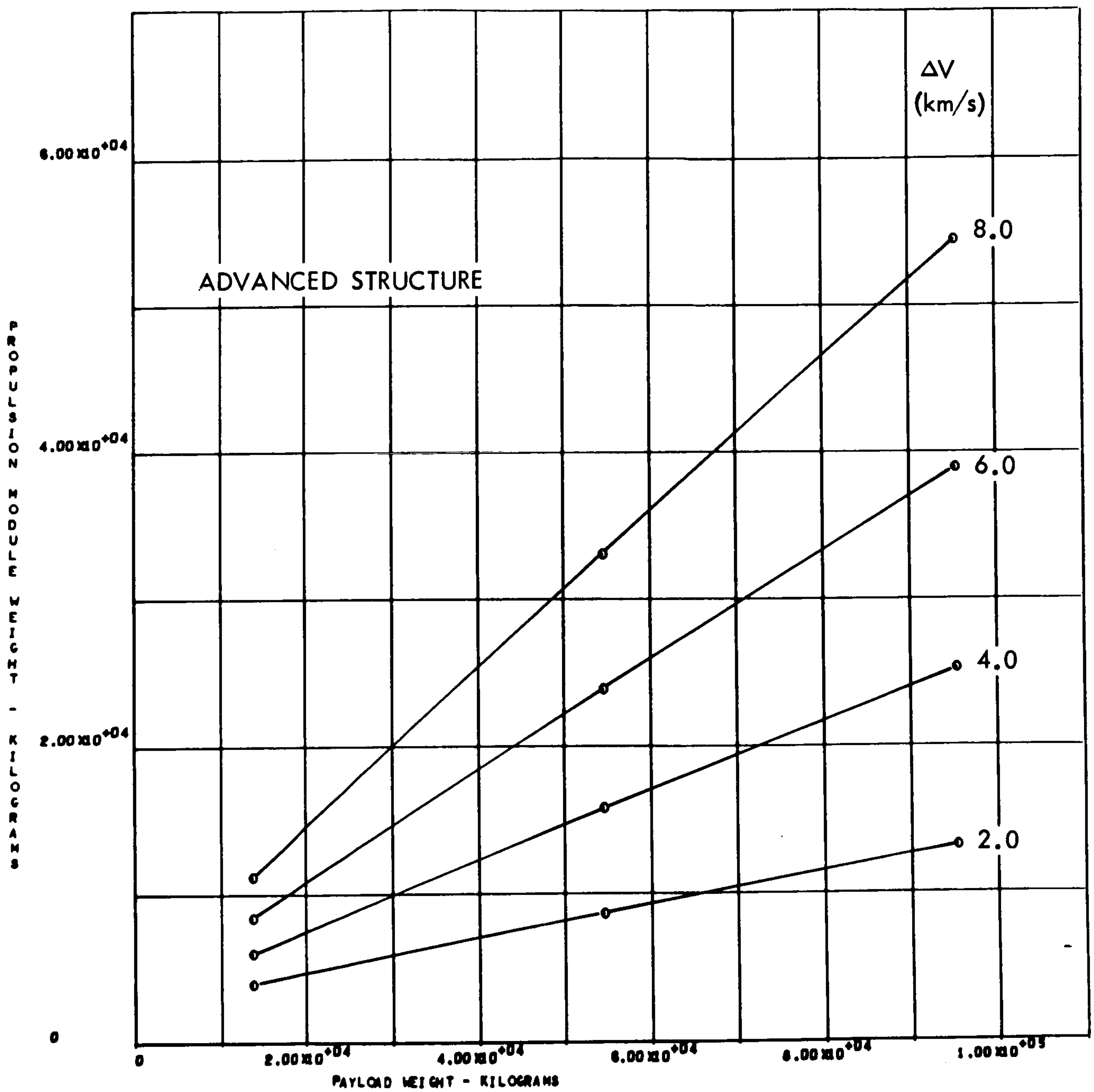

Figure 55. Propulsion Module Mass (I_{sp} = 2500 Seconds, T/W_e = 8.0, Maximum Payload = 100,000 kg)

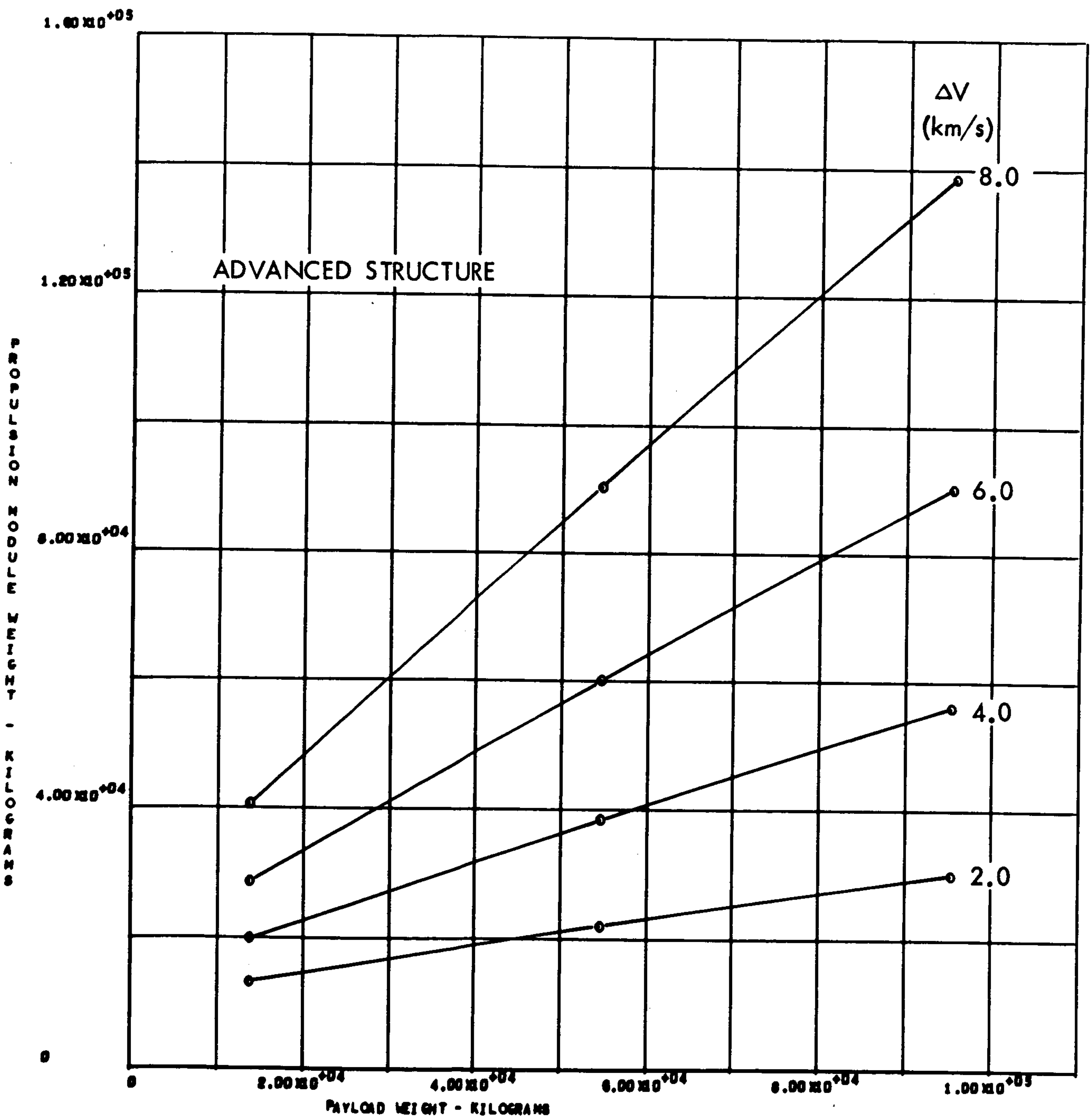

Figure 56. Propulsion Module Mass (I_{sp} = 2500 Seconds, T/W_e = 8.0, Maximum Payload = 100,000 kg)

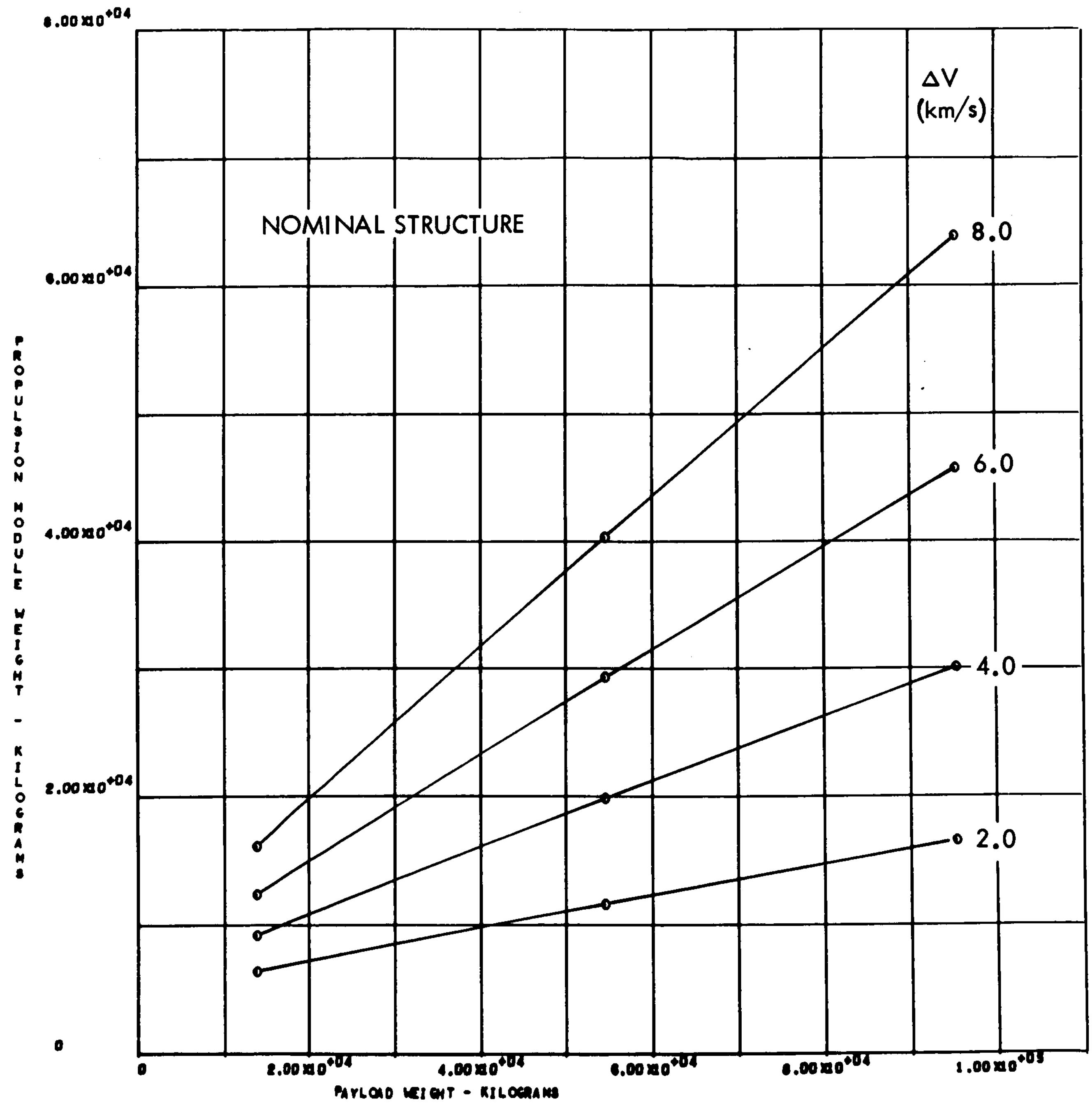

Figure 57. Propulsion Module Mass (I_{sp} = 3000 Seconds, T/W_e = 8.0, Maximum Payload = 100,000 kg)

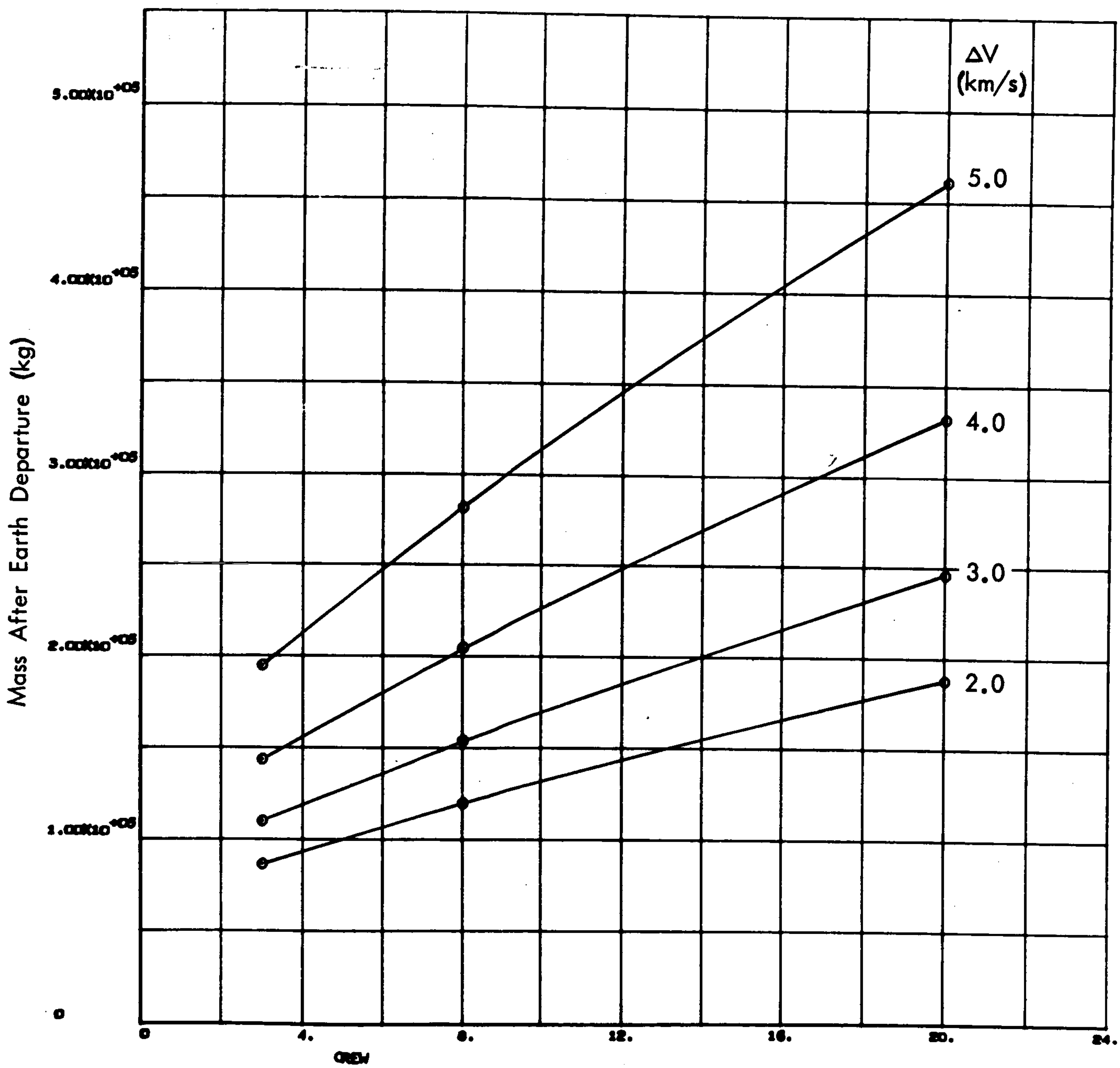

Figure 58. Mars Aerobraker (Apollo ERM, No MEM, I_{sp} = 387 Seconds)

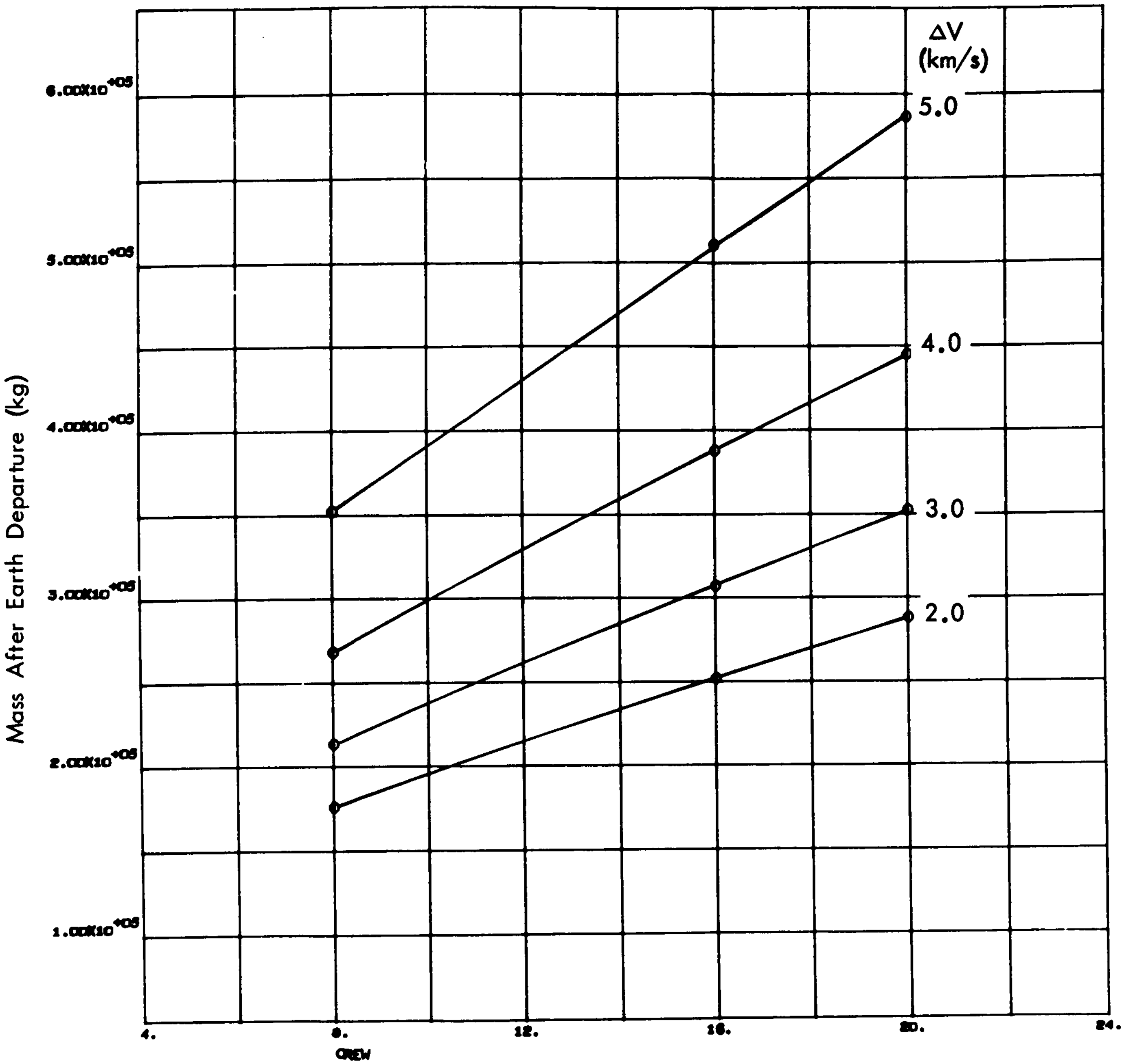

Figure 59. Mars Aerobraker (Apollo ERM, Apollo MEM, I_{sp} = 387 Seconds)

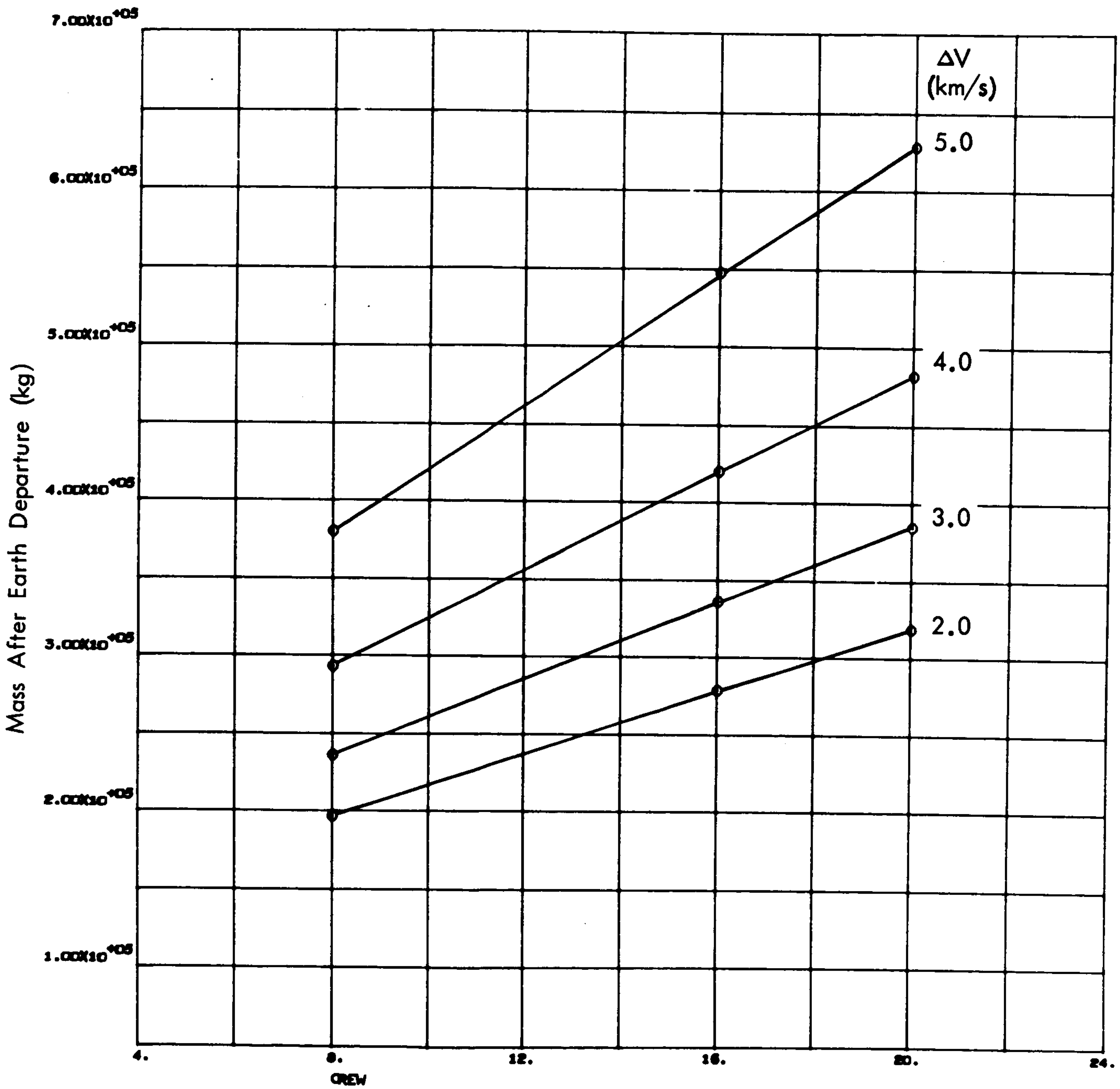

Figure 60. Mars Aerobraker (Apollo ERM, Aeronutronic MEM, I_{sp} = 387 Seconds)

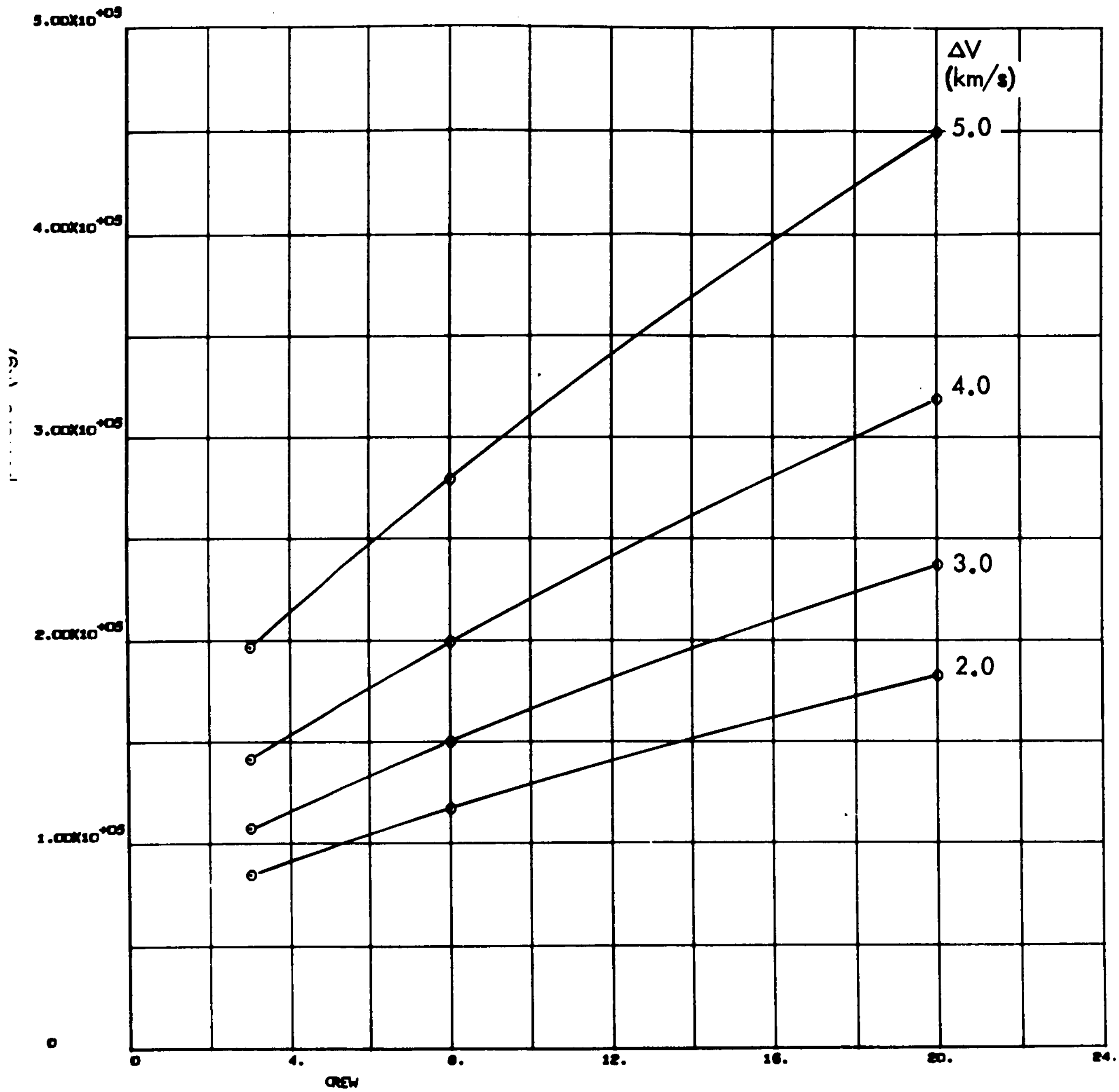

Figure 61. Mars Aerobraker (Apollo ERM, No MEM, I_{sp} = 450 Seconds)

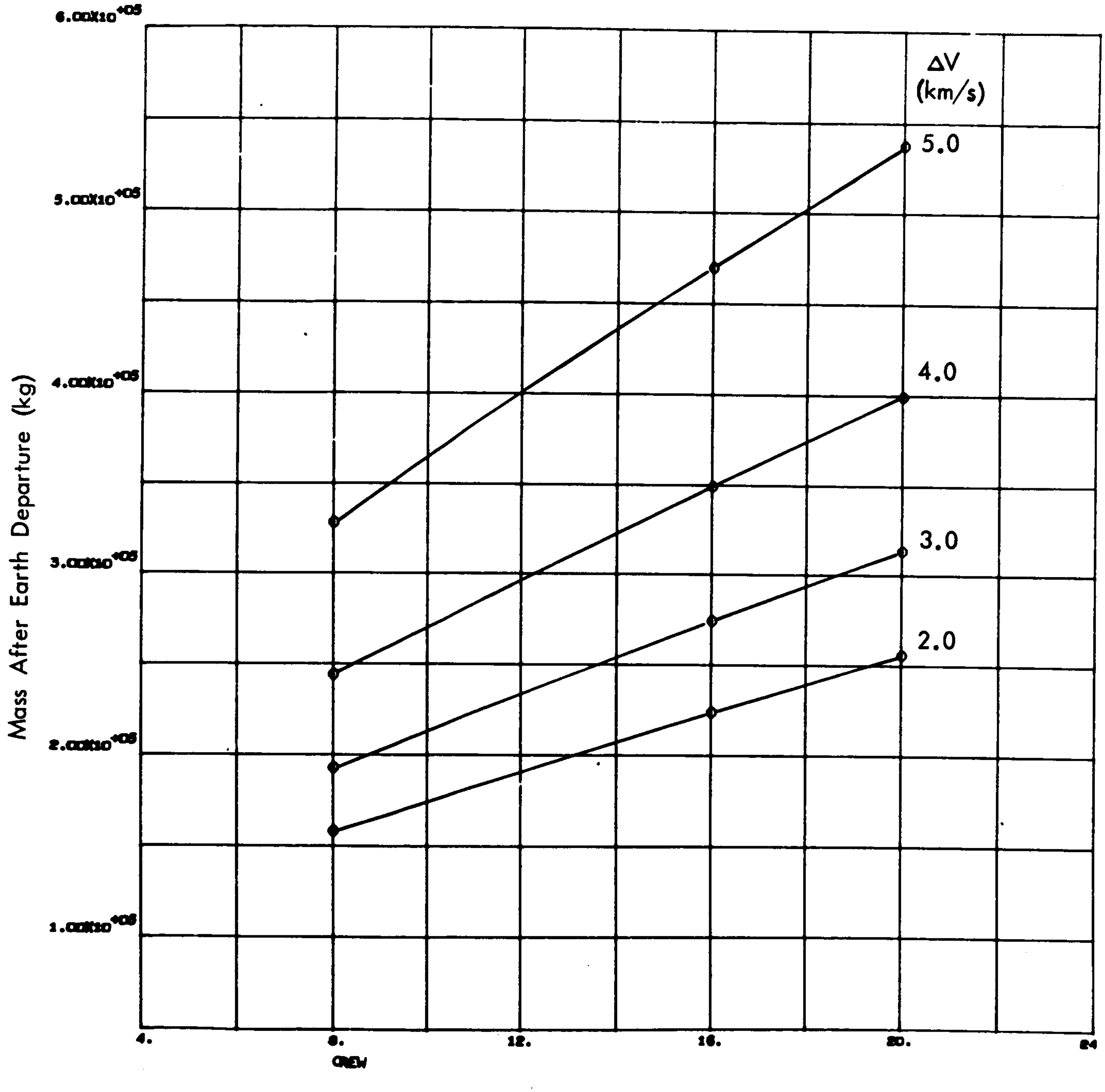

Figure 62. Aerobraker Mass (Apollo ERM, Apollo MEM, I_{sp} = 450 Seconds)

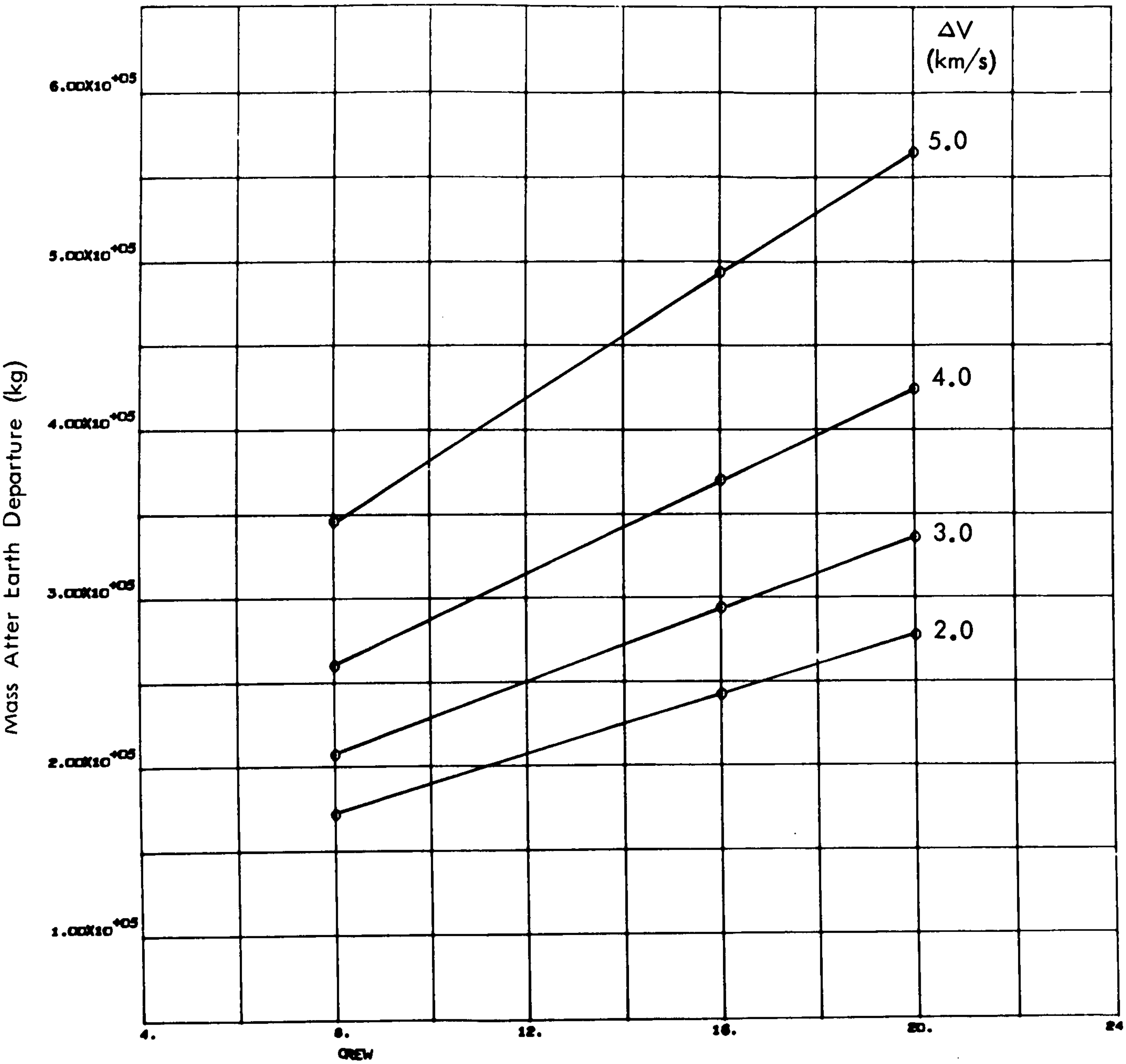

Figure 63. Mars Aerobraker (Apollo ERM, Aeronutronic MEM, I_{sp} = 450 Seconds)

The aerobraker mass includes the earth reentry module, mission module, trans-Earth midcourse correction stage, planetary orbit escape propulsion module, Mars excursion module, orbit circularization propulsion module, trans-Mars midcourse correction stage, and the aerobraker shroud. The trans-Mars and transearth midcourse correction incremental velocities were assumed to be 60 m/s. An additional 305 m/s was provided to circularize the elliptical parking orbit which results from the aerobraking maneuver. For the landing missions, the planetary stay time is 28 days, and the MEM crew size is half to total crew. All data are based on an earth reentry speed of 15.0 km/s and a Mars entry speed of 12.0 km/s.

The aerobraker data presented are based on the weight scaling equations presented in the Weight Scaling Equations section.

MISSION/SYSTEM DESIGN

The requirements of all of the missions which might be considered in any future manned planetary exploration program must be evaluated simultaneously to ensure an efficient over-all program. The establishment of the design requirements for modules for the nearer-term missions must include an evaluation of the requirements of the more advanced missions. Such an approach will ensure the maximum utilization of all modules developed, and an efficient expenditure of national resources for manned planetary exploration.

The characteristics of the individual modules and subsystems have been presented in parametric form in the previous sections of this Appendix and in Appendix C. These data, although useful in defining the subsystem and module characteristics as a function of the design parameters, do not conveniently define the total system mass requirements for the diverse mission objectives and mission opportunities which have been considered. The total system mass requirements are established in the following sections for representative mission opportunities for each of the mission objectives. The mass requirements of the manned modules and propulsion modules are then examined and potential common modules established. Finally, some of the penalties and advantages associated with the use of the common modules are evaluated.

It has been determined that it is feasible and, in some cases, advantageous to use common modules for a select family of missions. The penalties in mass in Earth orbit are dependent upon the mission objective, mission opportunity, and the criteria used to select common modules. Advantages include the use of a propulsion module which is designed by a mission which has high incremental velocity requirements to decrease the mission duration, increase the planetary stay time, or increase the payload for a mission which has lesser nominal requirements. However, a mass penalty would still exist since the distribution of the incremental velocity requirements would, in general, be such that one or more of the propulsion modules in the total system would be off-loaded in propellant.

STUDY APPROACH

The initial analyses of the mass requirements were based on the performance requirements for circular planetary parking orbits. The circular orbit restriction was imposed at the onset of this 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 this study, however, have shown that only modest performance penalties are incurred for performing off-pericenter planetary orbit insertion and escape maneuvers. Maneuvers carried out as much as 60 degrees in true anomaly from pericenter can result in increases in the incremental velocity requirements of only about 7 percent. These penalties are much less than the velocity reductions inherent in the use of elliptical orbits. [1]

The effects of using elliptical planetary parking orbits on the propulsion module mass requirements were considered for 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 such orbits would not result in significant performance advantageous due to the small mass of the asteroids.

The total system requirements were first established assuming the individual modules, and thus the total system, were designed by the particular mission requirements, e. g., incremental velocity, payload, mission duration, Earth reentry speed, crew size, environment, etc. These data provided the basic information required to select common modules and to evaluate the penalties and/or advantages associated with the use of common modules.

The initial examinations of common modules were based on the utilization of a common Earth reentry module (ERM) and a common mission module (MM). The modules which were selected satisfied the requirements of the majority of the missions. The selection of a common ERM was based on the Earth reentry speed for the majority of the missions considered in the study. The selection included the elimination of some mission objectives, mission opportunities, and mission modes due to excessive requirements which would unduly penalize the majority of the missions. The selection of a common mission module was based on the longest mission duration and consumables were off-loaded as required for missions of shorter duration. The investigations of the effects of using a common ERM and MM were performed by determining the total system mass assuming the propulsion modules and the environmental protection requirements of all modules were sized by the particular mission. In this manner it was possible to determine the effects of the common manned modules on the mass requirements of the propulsion modules and the mass in Earth orbit.

[1] The velocity reductions which can be achieved using elliptical planetary parking orbits are presented in Appendix A.

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.

The investigations of common propulsion modules were based on mass requirements only. Many other factors will also effect the selection of future propulsion modules — for example, the development cost and development time. Other factors which must be included in the ultimate selection of future propulsion modules are the operational considerations. These include the compatibility of the propulsion 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 manned module weights and scientific mission objectives insofar as they influence spacecraft weight.

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

Only the Venus swingby mission mode was considered for Mars missions during the mission/system design analyses. The Venus swingby missions have, in general, lower total incremental velocity requirements, lower Earth reentry speeds and reduced velocity sensitivity to launch delays compared to the direct mission.

CIRCULAR PLANETARY PARKING ORBITS

Operational considerations could be imposed which would limit the planetary parking orbits to circular orbits. Since circular orbits could conceivably be required, the analyses of such orbits are presented separately. During the subsequent discussions of elliptical planetary parking orbits, circular orbits are again considered but only in the context of elliptical orbits of zero eccentricity.

Optimized Manned Modules

The basic mass requirements of the manned modules (Earth reentry module, mission module, and planetary excursion module) were presented in parametric form in the Weight Synthesis Parametric Analysis section of this Appendix. The additional mass requirements for meteoroid protection, thermal insulation and radiation protection must be added to the basic module mass in order to define the total mass for a given mission. The resultant mass (measured at the beginning of the mission) of the Earth reentry module, mission module, and trans-Earth midcourse correction propulsion module are shown in Figure 64 for representative mission opportunities for crew sizes of eight and twenty men. The data are based on a low L/D (Apollo) Earth reentry module and a mission module with a partially regenerative environmental control and life support subsystem (water and oxygen recovery) and an isotope/mercury Rankine electrical power subsystem. A nominal mission module free volume per man of 750 ft^3/man was used and the number of floors were varied with crew size in order to maintain a module diameter of 10 meters or less. (It has been shown in the previous section - Weight Synthesis Parametric Analysis - that the number of floors has a negligible effect on the module mass.) The trans-Earth midcourse correction propulsion module mass requirements are based on an incremental velocity of 60 meters per second per mission leg, a specific impulse of 387 seconds, and a stage mass fraction of 0.85. A constant stage mass fraction was assumed for the trans-Earth midcourse propulsion module since the module sizes were outside the range of applicability of the stage weight scaling equation defined in the Weight Scaling Equations section of this Appendix. Inordinately high values would have been obtained had the scaling equations been employed.

The effects of the mission opportunity on the mass requirements for radiation protection can be seen by comparing the requirements for the 1988 and 1992 Mercury missions. The 1988 mission occurs when the projected solar activity will be a minimum, and the 1992 mission occurs when the activity is expected to be a maximum. The mass differences due to radiation shielding requirements alone are approximately 7,000 kg for an eight-man crew, and 11,000 kg for a twenty-man crew. The remaining differences are due to the slightly higher Earth reentry speed (15.59 km/sec versus 15.02 km/sec) and longer mission duration (364 days versus 311 days) for the 1992 mission.

The mass requirements of the planetary excursion modules are presented in Figures 65 through 67. The data include the mass of the interstage which houses and provides meteoroid protection of the PEM during the trans-planet mission phase. The data are based on the use of storable propellants with a specific impulse of 387 seconds and a bulk density of 1233 kg/m^3 (77 lb/ft^3). The palnetary parking orbit altitude at which the descent maneuver is initiated and the ascent maneuver is terminated is one planetary radius in all cases except Mars. The Mars parking orbit altitude was assumed to be 800 km.

Environmental and performance considerations precluded extensive analyses of manned landings on Venus and Jupiter. A brief investigation of the mass requirements of a Venus excursion module (Configuration Design section) resulted in an estimated minimum module mass in excess of

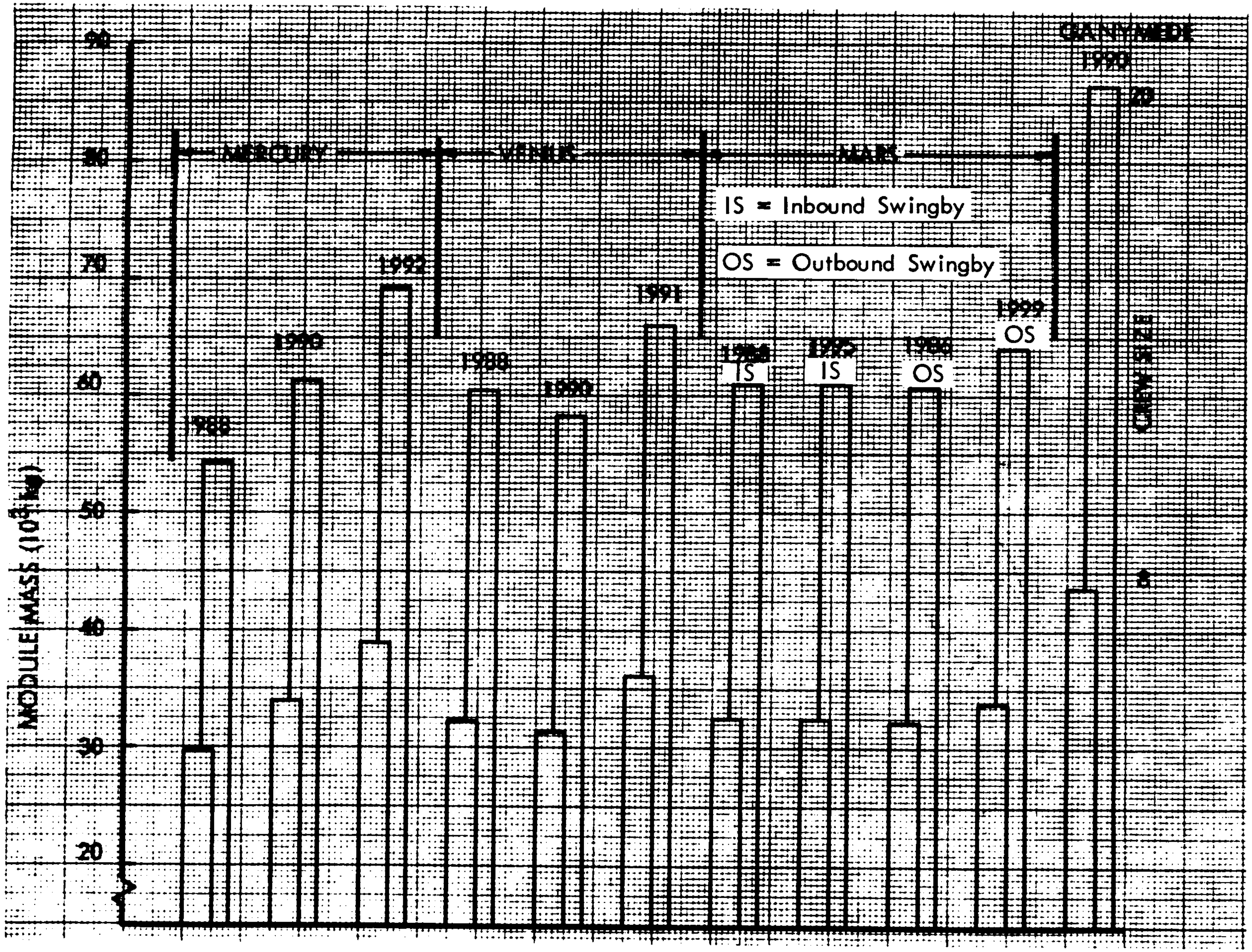

Figure 64. Transearth Mass Requirements

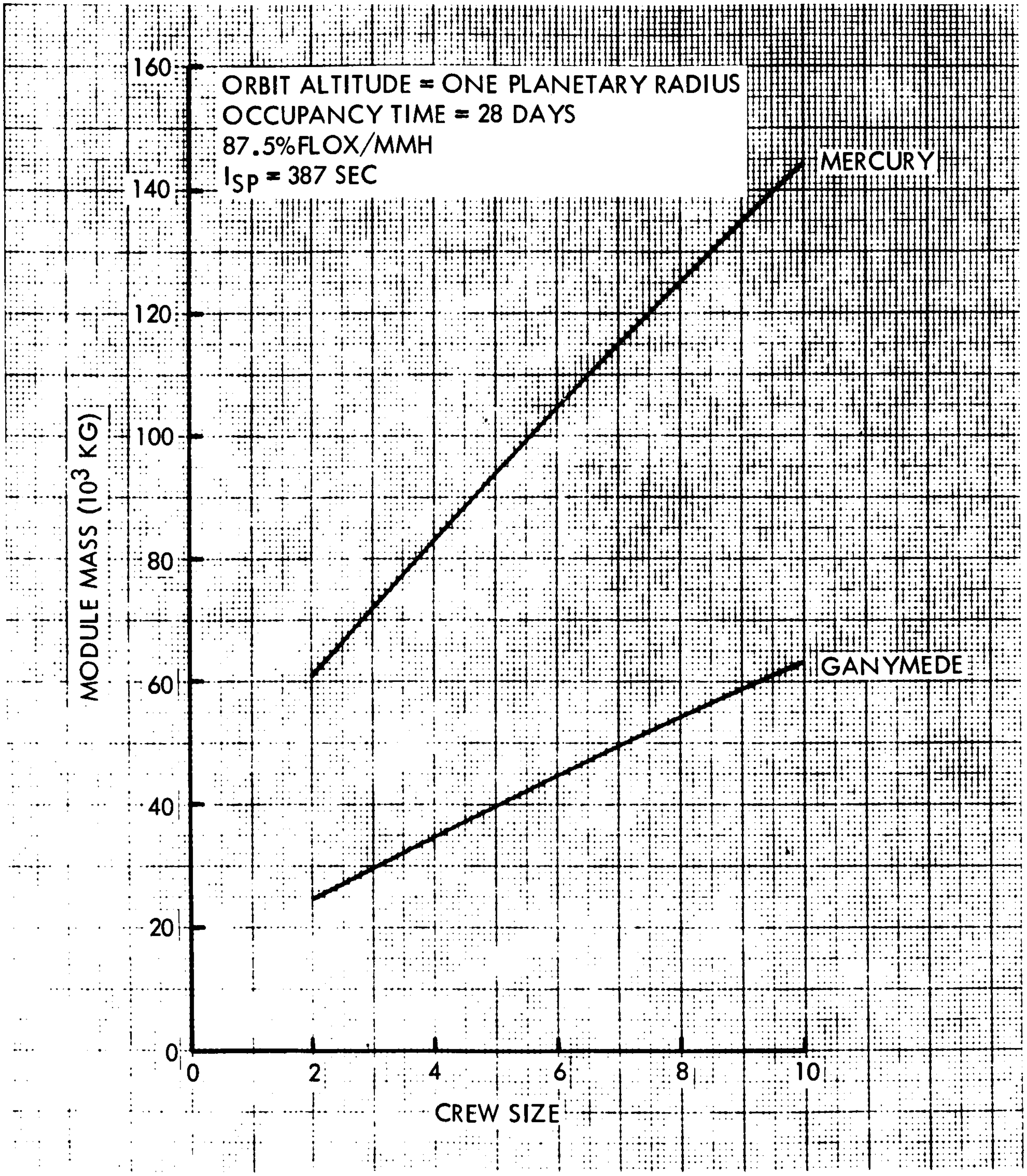

Figure 65. Planetary Excursion Module Mass (Mercury and Ganymede)

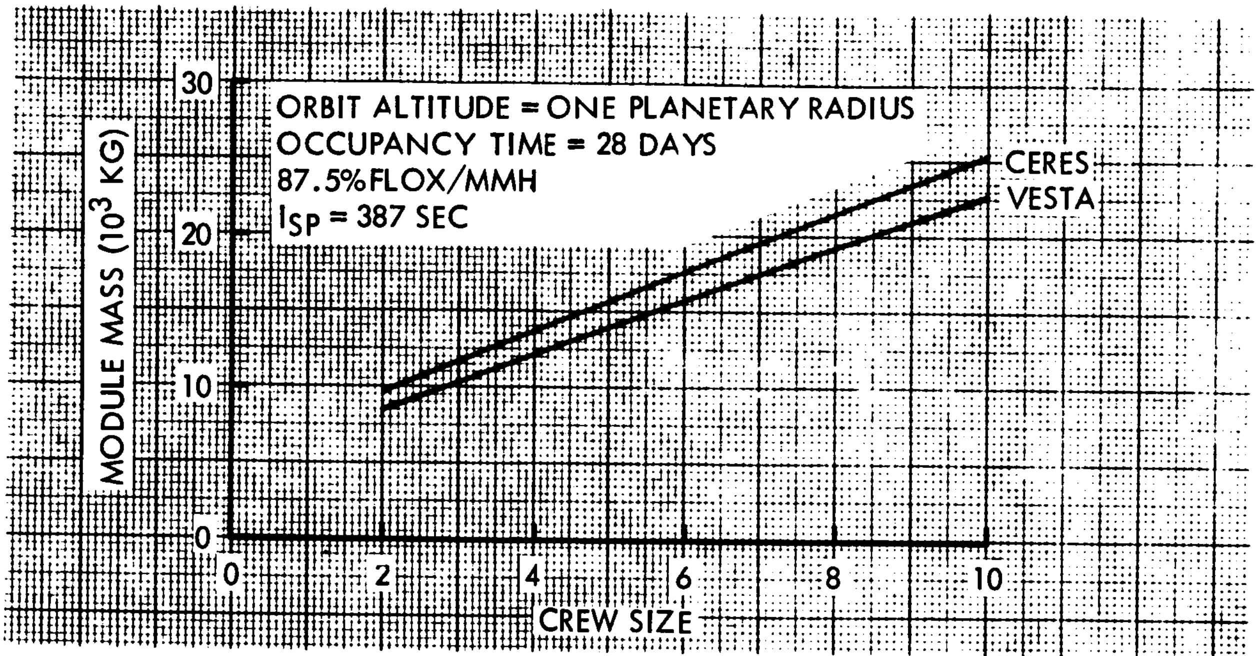

Figure 66. Planetary Excursion Module Mass (Ceres and Vesta)

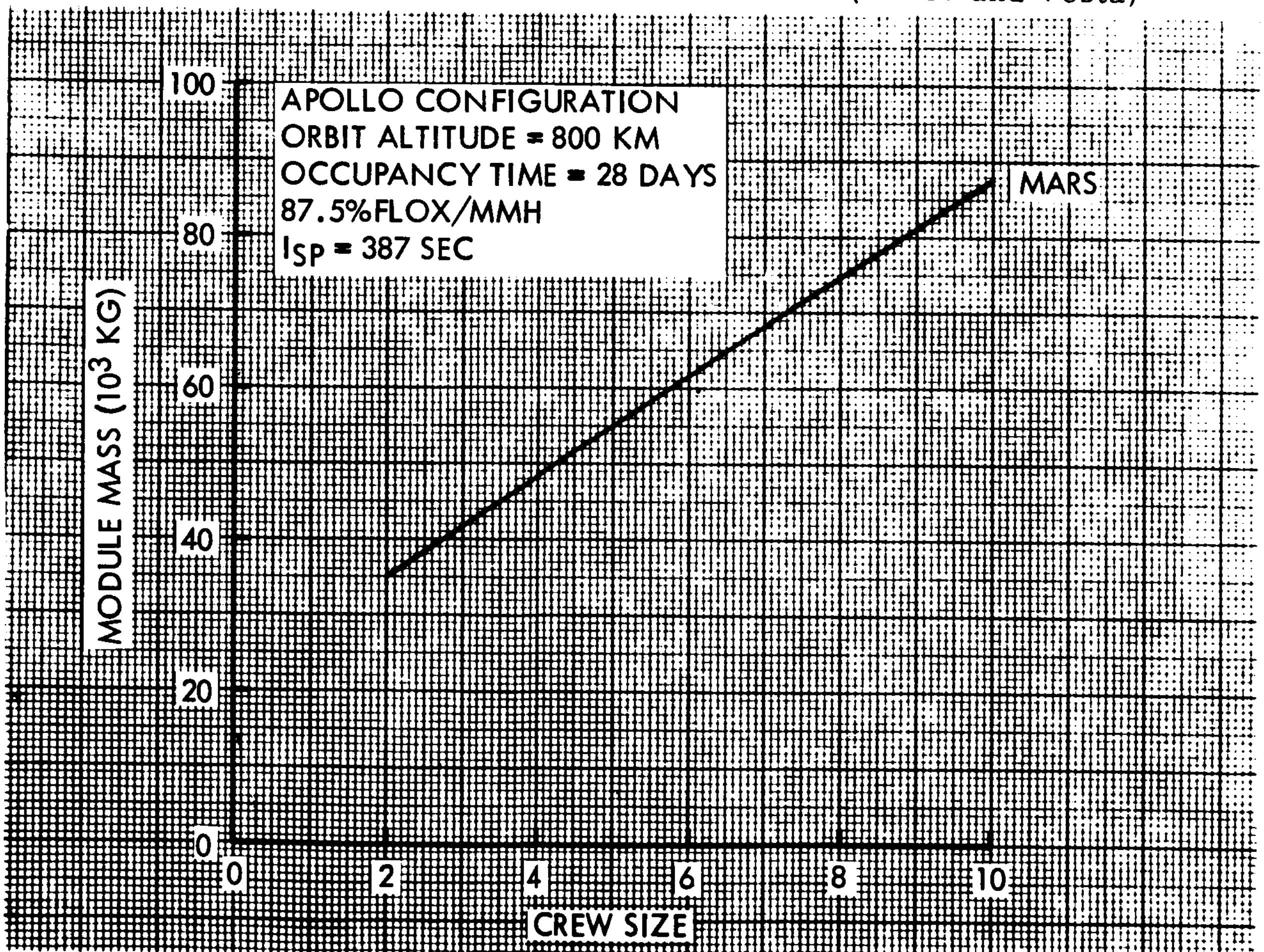

Figure 67. Planetary Excursion Module Mass (Mars)

363,000 kilograms. This was considered to be excessive and was not investigated during the mission/system design analyses. The effects of the payload mass which is landed (or left in orbit about the planet) were investigated for both Venus and Jupiter by using arbitrary probe masses of 10,000 and 50,000 kg during the system weight synthesis analyses.

Optimized Propulsion Modules

The major propulsion modules for retrobraker missions are the Earth orbit escape module (EOE), planetary orbit insertion module (POI), and planetary orbit escape module (POE). The mass requirements of the individual modules which are sized by the payload, incremental velocity requirement, and central body are discussed in this section.

The results which are presented in this section are based on the use of an Apollo type ERM and a MM with a partially closed (water and oxygen recovery) EC/LSS and an isotope/mercury Rankine EPS. The PEM mass requirements are based on the use of storable propellants with a specific impulse of 387 seconds and a bulk density of 1233 kg/m^3. In all cases, the PEM crew size is assumed to be half the total crew size, i.e., half the size of the crew size used in the determination of the ERM and MM mass.

The total propulsion module mass consists of the basic shell (tankage, accessories, etc.), engine, propellant (including boil-off propellant), meteoroid protection, insulation, and interstage structure. The engine thrust (and thus mass) was determined by optimizing the initial thrust-to-weight ratio. Engine burn time limits of 600 and 1200 seconds were assumed for chemical and nuclear engines, respectively. If the thrust-to-weight ratio would nominally be optimum with a burn time in excess of the above limits, the thrust corresponding to the maximum burn time was used to determine the engine mass.[1] The insulation and boil-off propellant requirements were optimized for each module by minimizing the total system mass in Earth orbit (see Appendix B). 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 ignition.

Chemical Propulsion Modules

The examinations of the chemical propulsion module mass requirements were limited, in general, to Mars and Venus missions. The mass requirements for representative Mars and Venus mission opportunities are shown in

[1] A brief investigation of the effects of the burn time constraints showed that the assumed limits did not unduly penalize the total system mass. The thrust-to-weight ratio had either been minimized prior to reaching the burn time constraint or was near the minimum at the limiting values.

Figure 68 for crew sizes of eight and twenty men. A limited investigation has shown that the mass requirements for intermediate crew sizes can be estimated quite accurately by linear interpolation. The data are based on a specific impulse of 450 seconds for all propulsion modules. As can be seen from the figure, the propulsion module mass requirements are continuous when variations in crew size are considered.

Solid Core Nuclear Propulsion Modules

The mass requirements of solid core nuclear propulsion modules are shown in Figure 69 for representative mission opportunities for all mission objectives. The mission opportunities which were considered during this analysis are summarized in Table 14. The detailed characteristics of the missions can be obtained from Tables through of Appendix A. All of the results presented in Figure 69 are based on a specific impulse of 820 seconds. Included in the data, however, are the nuclear Earth orbit escape propulsion module mass requirements for Mars and Venus retrobraker missions and Mars aerobraker missions using cryogenic upper stages.

Certain similarities in the propulsion module mass requirements can be observed from Figure 69. The planetary orbit insertion requirements for Vesta and Ganymede missions are comparable to the nuclear propulsion modules required for planetary orbit escape 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 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.

The Earth orbit escape propulsion module mass requirements for the baseline Vesta, Ceres, and Jupiter flyby missions are shown in Table 15. Also shown in the table are the mass requirements of the manned modules (Earth reentry module and mission module at the time of Earth orbit escape) and the mass in Earth orbit requirements. The effects of mission opportunity on the mass requirements are most significant for the Ceres missions and the least significant for the Jupiter missions. The large variations in the requirements for the Ceres missions are due to the high inclination and high eccentricity of the orbit of Ceres. The Earth orbit escape propulsion module mass requirements are comparable to the requirements for circular planetary orbit insertion and escape for the Mars and Venus missions even if the worst Ceres missions are considered.

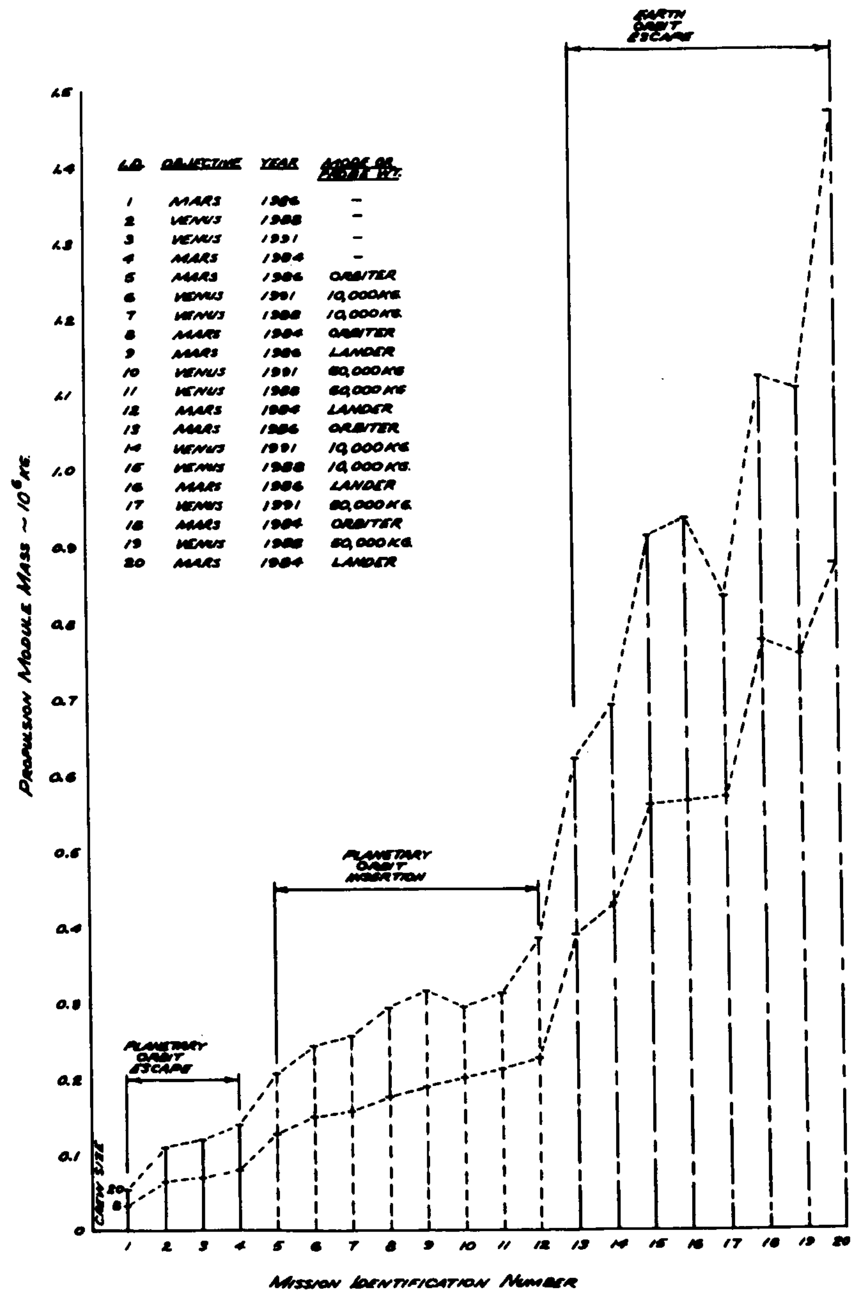

Figure 68. Propulsion Module Mass Requirements
(Specific Impulse = 450 Seconds)

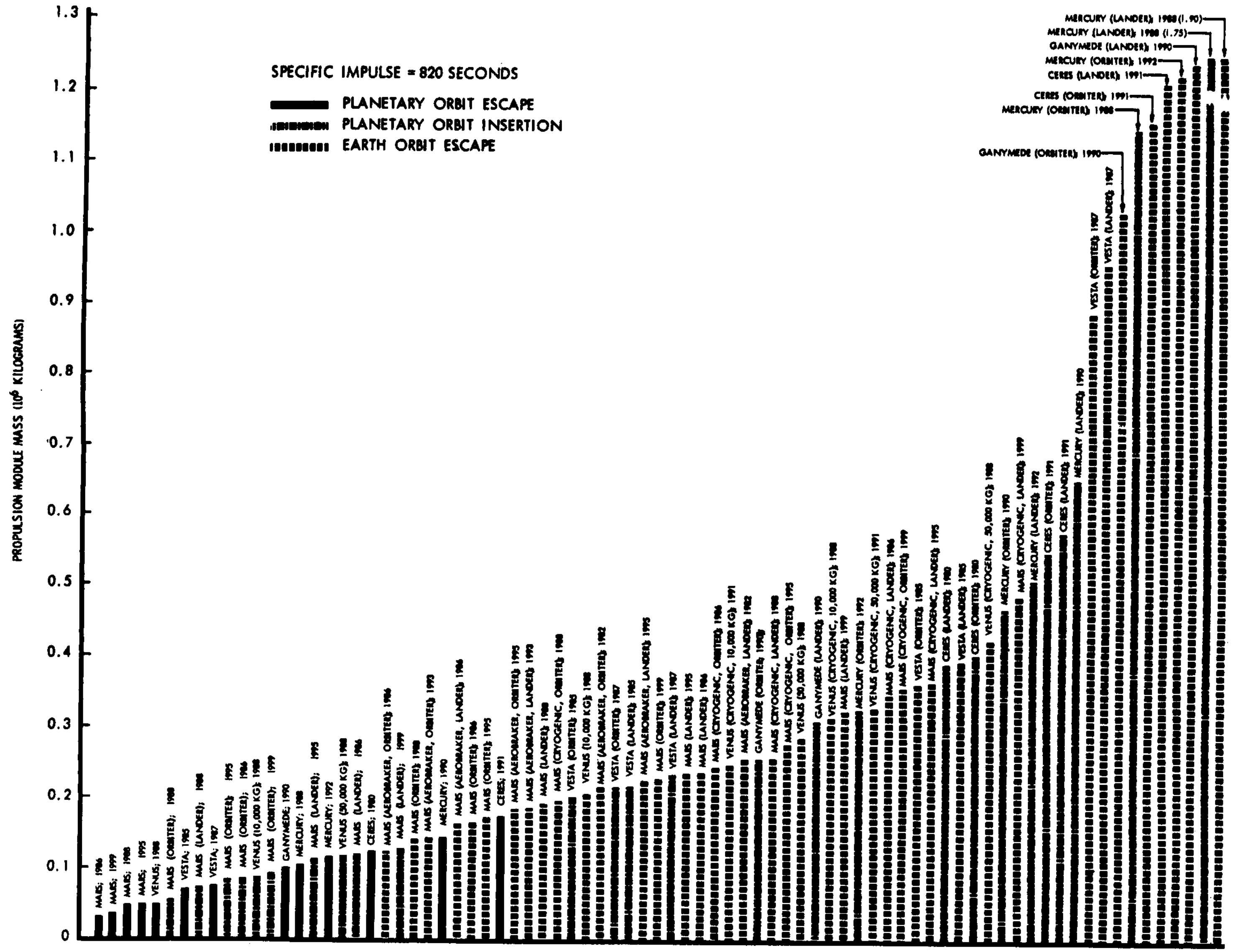

Figure 69. Nuclear Propulsion Mass Module Requirements (8-Man Crew)

Table 14. Mission Opportunities for Solid Core Nuclear Propulsion Module Mass Requirements Analyses

Mission Objective	Mission Mode	Mission Years Considered
Mercury	Direct	1988, 1990, 1992
Venus	Direct	1988
Mars	Venus Swingby	1986, 1988, 1995, 1999
Vesta	Direct	1985, 1987
Ceres	Direct	1980, 1991
Ganymede	Direct	1990

Table 15. Flyby Mass Requirements (Eight-Man Crew)

Objective	Year	Manned Modules (kg)	Earth-Orbit Escape Stage (kg)	Mass on Earth Orbit (kg)
Vesta	1991	30,700	57,200	94,300
Vesta	1993	35,700	105,500	147,800
Ceres	1993	35,500	79,600	121,600
Ceres	1992	36,700	198,100	241,400
Jupiter	1991	42,200	137,600	186,600
Jupiter	1985	36,200	134,100	176,900

A brief investigation was conducted to determine the effects of the mission profile and the meteoroid environment on the incremental velocity requirements and the mass requirements for missions to Ganymede. The nominal mission profile consists of a single plane transfer from Earth to Jupiter/Ganymede and from Jupiter/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 AU out of the plane of the ecliptic at the radius of the center of the asteroid belt (2.8 AU). The two plane transfer consists of an initial heliocentric conic which is in or near the plane of the orbit of the departure planet about the Sun. Since this trajectory will not, in general, result in a rendezvous with the target planet, a second maneuver is required. The second maneuver consists of a pure plane change maneuver which results in a second heliocentric conic which is designed to avoid passage through the asteroid belt. The resultant incremental velocity requirements and the mass requirements are shown in Table 16. It can be seen from the table that 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 meteoroid environment. Although the incremental velocity requirements are higher, meteoroid protection is required only for the cometary flux resulting in lower shielding requirements for all modules (manned and propulsive). If the maximum environment is considered with the nominal (single plane transfer) mission profile, additional shielding is required for all modules which increases the mass in Earth orbit by a factor of 3. 14.

The relatively small increase in the mass in Earth orbit requirements associated with the out-of-the ecliptic profile and the uncertainty in the asteroidal environment make the out-of-the ecliptic mission mode particularly attractive. On the basis of the limited analyses conducted during this study, it appears that this mission mode should be given serious consideration during the definition of the mission/system requirements for all (manned and unmanned) missions to Jupiter.

Gaseous Core Nuclear Propulsion Modules

The propulsion module mass requirements for Mercury, Vesta, Ceres and Ganymede missions using gaseous core nuclear propulsion modules are shown in Figure 70. The data are based on a specific impulse fo 2500 seconds and an engine thrust-to-weight ratio of eight. The upper bar represents the effects of varying the crew size from eight to twenty men. By comparing Figures 69 and 70, it can be seen that the gaseous core nuclear propulsion module mass requirements are approximately an order of magnitude less than the requirements for solid core modules.

Although Mars and Venus missions were not considered during the analyses of the gaseous core propulsion module mass requirements, it is appropriate to determine the effects of using modules designed by the advanced missions for the nearer-term missions. Modules designed by the requirements of the advanced missions could be used to either reduce the mission duration or increase the payload for Mars and Venus missions. The effects of using the propulsion modules designed by the 1990 Jupiter mission to perform the 1995 Mars mission are shown in Table 17. The table shows the reduction in the trip time and mass in Earth orbit which can be achieved. The results are based on minimizing the total trip time by maximizing the utilization of the fixed propulsion modules. Since the modules are not designed by the requirements of the Mars mission, the total capability of all propulsion modules can not be used. For the mission opportunity examined, only the planetary orbit insertion stage is fully loaded. Both the planetary orbit escape and the Earth orbit escape modules were off-loaded in propellant.

Common Manned Modules

Earth Reentry Module

An examination of the Earth reentry speeds presented in the Mission Requirements Section of Appendix A shows that the reentry speeds are generally less than 15 km/sec. The major exceptions are the Ceres and

Table 16. Jupiter Out-Of-The Ecliptic Mission (Ganymede 1990 Mission)

Missile Profile	Meteoroid Environment	Manned Modules (kg)	Total ΔV (km/s)	Mass on Earth Oribt (kg)
Nominal	Nominal	34,600	17.54	1,951,806
Out-of-the-ecliptic	Nominal	32,910	18.75	2,120,258
Nominal	Maximum	63,370	17.54	6,119,961

Table 17. Mars Mission Using Gaseous Core Nuclear Propulsion Modules

Propulsion System	Trip Time		Mission Duration	Mass in Earth Orbit (kg)
	Trans-Mars	Trans-Earth		
Cryogenic	180	260	470	1,215,000
Nuclear Gaseous Core	130	195	355	213,000

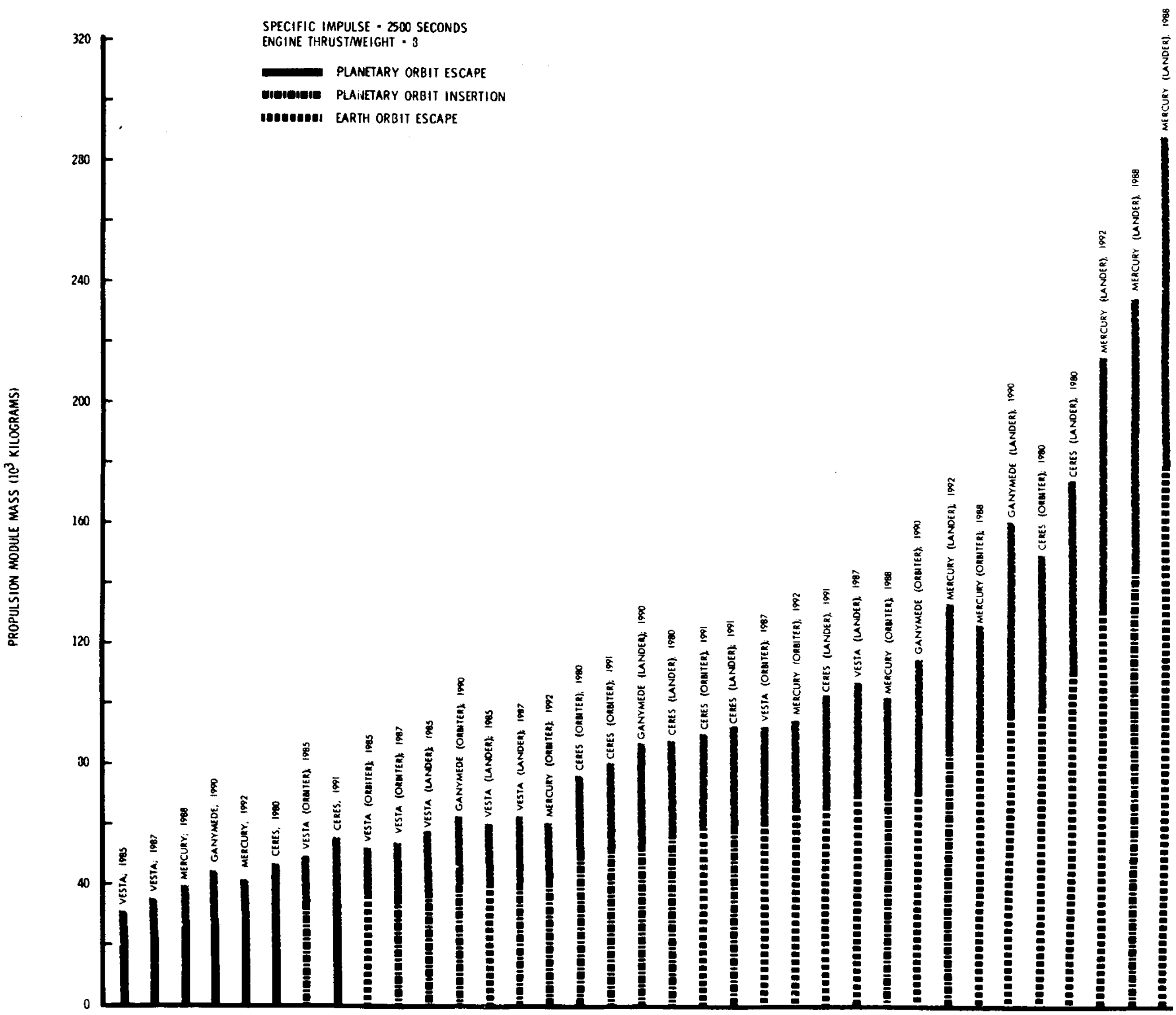

Figure 70. Gaseous-Core Nuclear Propulsion Module Mass Requirements (8-Man Crew)

Mercury missions and the direct Mars missions. The Earth reentry speeds for the Ceres missions can be reduced by increasing the incremental velocity requirements. The high reentry speeds associated with the direct Mars missions can be avoided by considering only the Venus swingby mission mode which is also the likely mission mode when propulsive requirements are considered.

The Earth reentry module mass requirements were shown in Figure 19 of the Weight Synthesis Parametric Analysis section of this Appendix for a crew size of eight men. As can be seen from the figure, the Apollo configuration is the lightest for reentry speeds below 14.2 kilometers per second. In the area between 14.2 and 17.5 kilometers per second the biconic is the lightest; and above 17.5 kilometers per second, the conic is the lightest. From the standpoint of mass considerations, the development of a conic configuration would not be required if the alternate class Ceres missions and only the Venus swingby Mars missions are considered.

The reentry speeds for Mercury missions can be limited to less than 16.0 km/s by limiting the mission opportunities which are considered. Limiting the missions to those opportunities which have the lower reentry speeds is also compatible with the elimination of mission opportunities on the basis of excessive performance requirements. For the remaining mission objectives (Venus, Mars missions with Venus swingby, Vesta, and Jupiter/Ganymede) the reentry speeds are less than 15 km/s. Between 14.2 km/s and 15.0 km/s, the biconic configuration has a slight mass advantage. In terms of the total mass of the manned vehicles, the mass advantage of approximately 200 kg represents less than one percent of the total mass of the transearth spacecraft. It is therefore concluded that, on the basis of the parameters which have been considered in the present study, the low L/D (Apollo) configuration will be adequate for future manned planetary missions. Other considerations which may make the development of a second configuration desirable, e.g. abort, have not been considered.

Mission Module

The mission module mass is primarily dependent upon the crew size, mission duration, and the types of subsystems assumed. 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 in the module 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, for the mission duration. As the crew size increases, another floor could be added and the additional consumables provided.

An alternate approach would be to develop a single module which is designed for a given mission duration and crew size and to then off-load crew and consumable as required for missions which impose lesser requirements. The missions which have the longest duration and those which impose the maximum requirements in terms of consumables are the Jupiter/Ganymede missions. The mission duration for these missions are slightly over 1400 days.

Regardless of which approach is used, it is assumed that the meteroid and radiation protection would be sized for the particular mission. This assumption is 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.

Planetary Excursion Modules

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 planetary excursion modules would be in the amount of propellant provided for ascent and in the ascent stage engine thrust. The environmental protection requirements (thermal) would be designed for the given mission objective. 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 relative small module for landings on Ceres and Vesta. As can be seen from Figure 65, the mass of the Mercury module is approximately twice the mass of the Ganymede module. The mass differences in the ascent stages are 4000 kilograms; the differences in the descent stage requirements are 12000 kilograms. The differences in the mass requirements are due to the differences in characteristic velocity requirements and thus propellant requirements. Thus, two common descent stages could be developed which are sized on the basis of the requirements for the Mercury and Ceres missions.

Effects of Common Manned Modules

The effects of using manned modules which were sized by the requirements of the limiting missions were evaluated assuming a crew size of eight men. The earth reentry module requirements were dictated by the 1992 Mercury mission which has a reentry speed of 15.59 km/s. The Mission Module was sized by the Jupiter/Ganymede missions based on a mission duration of 1416 days.

The effects on propulsion module mass requirements of using manned modules which were designed for the eight-man crew but off-loaded in consumables as dictated by the mission requirements are shown in Figure 71. The lower bars represent the basic requirements, while the upper bars represent the penalties for using fixed manned modules. The effect of oversizing the earth reentry module can be seen by the relatively small increase in the propulsion module mass requirements for the Ganymede missions. The maximum effects of an oversized mission module can be seen by referring to the penalties for the 1992 Mercury mission.

Common Propulsion Modules

The determination of the future propulsion module requirements must be evaluated in the same manner as the manned modules requirements were evaluated in the previous section. Some of the basic questions which must be answered are: (1) can propulsion modules be developed for the nearer-term missions which are compatible with the requirements of the advanced missions; (2) can modules be developed which permit flexibility in the mission/system selection; and (3) what are the penalties and advantages which may result from the development of common modules?

The results which are presented in this section are based on the use of manned modules which are sized by the particular mission requirements. The characteristics of the manned modules are the same as those used during the investigations of the optimized propulsion modules.

Chemical Propulsion Modules

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 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.

By referring to Figure 68, it can be seen that the planetary orbit escape propulsion module mass requirements are less than 100,000 kg for the limiting

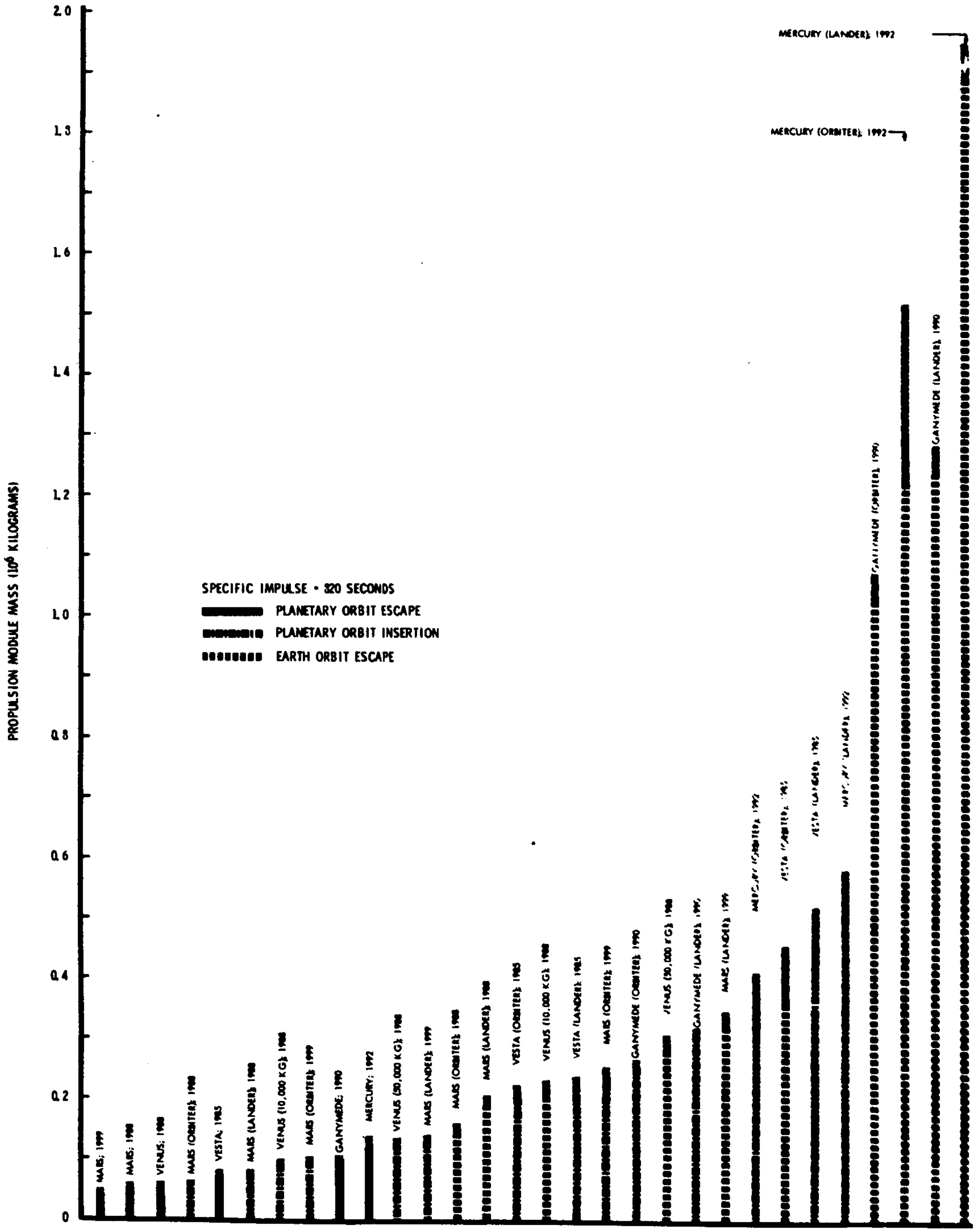

Figure 71. Nuclear Propulsion Module Mass Requirements (Common Manned Modules, 8-Man Crew)

mission opportunities considered. If such a module were developed, the same module could be used in pairs to accomplish the planetary orbit insertion maneuver. This module could not be used for Earth orbit escape, however, without excessive clustering.

A second module would have to be developed to accomplish the Earth orbit escape maneuver. This module could be on the order of 500, 000 kg and would be used either singularly, in pairs, or in combination with the 100, 000 kg module to accomplish the Earth orbit escape maneuver for all missions considered. An alternative would be to develop either a 300, 000 kg module or a 600, 00 kg module. The 600, 000 kg module could be used singularly to accomplish the Earth orbit escape maneuver for all missions except the more difficult Mars missions and the Venus missions with the larger probe mass (50, 000 kg). The 300, 000 kg module could never be used singularly but it would provide more flexilibility in crew size and/or probe mass.

Of the modules considered, the 100, 000 kg and 300, 000 kg modules appear to be the most attractive combination. If such an approach were adopted, there would be no commonality between the Earth orbit escape module and the modules required for the remaining mission maneuvers. Therefore, a storable module could be developed for the planetary orbit insertion and escape maneuvers without impacting the module commonality.

Nuclear Propulsion Modules

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 more appropriate for all missions.

It has been shown in Figure 69 that the propulsion module mass requirements are essentially continuous if all mission opportunities to all mission objectives are considered. In an attempt to produce discrete bands of propulsion module requirements, the number of mission opportunities which were considered were limited. The effects of limiting the mission opportunities for the more difficult missions (i. e., Mercury, Vesta, and Ceres) are shown in Figure 72. The lower bar corresponds to a crew size of 8 men; the upper bar corresponds to a crew size of 20 men. All mission opportunities for these mission objectives have been eliminated from consideration except those for which arrival at the target body occurs near the line of nodes. This restriction yields missions of lowest energy requirements. Also, the Mars and Venus missions with chemical upper stages are not shown in this figure. It can be seen that the propulsion module mass requirements are still continuous if crew sizes up to twenty men are considered. By limiting the crew size to eight men, a limited number of discrete bands can be obtained. As

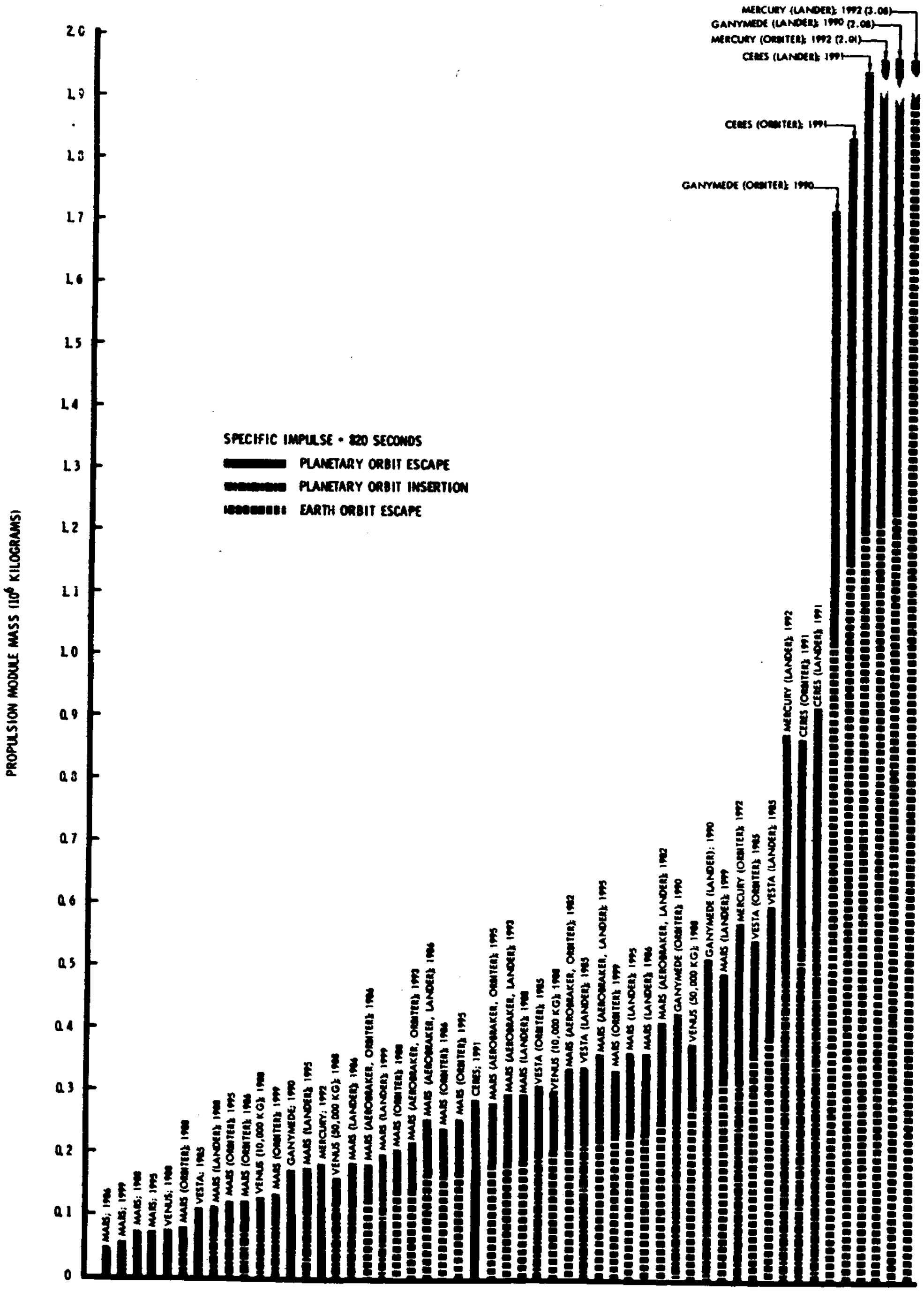

Figure 72. Nuclear Propulsion Module Mass Requirements (8- to 20-Man Crew)

can be seen from the figure, the mass requirements remain continuous up to approximately 600,000 kg even with the smaller crew size. This continuum includes all propulsion modules except the Earth orbit escape modules for the Mercury, Ceres and Ganymede missions. A lower continuum (mass $\leq$ 350,000 kg) exists which includes all propulsion modules up through the Earth orbit escape modules for the Mars, Venus, and Vesta missions. Included in the lower continuum are the planetary orbit insertion modules for the remaining Mercury lander missions and the Ganymede orbiter and lander missions. A second band of requirements exists between 1.0 and 1.2 million kilograms. Modules within this band would be required for Earth orbit escape for Ceres and Ganymede orbiter and lander missions and Mercury orbiter missions with an eight man crew.

The feasibility of selecting discrete propulsion modules within the lower band of requirements was investigated in detail assuming only two propulsion module sizes were to be developed. During the initial investigations a 100,000 kg module was assumed which could be used for planetary orbit escape for all mission objectives except Mercury and Ceres. The same module could be used either singularly or in multiples for planetary orbit insertion for Mars and Venus missions. However, it was found that an excessive penalty resulted since the module was extremely over-sized for the planetary orbit escape maneuvers for Mars and Venus missions. To reduce the penalty the module size was decreased to 75,000 kg. The 75,000 kg module could be used singularly for planetary orbit escape, and either one or two of the modules were required for planetary orbit insertion for all Mars and Venus missions. Two modules were required for planetary orbit escape for Mercury and Ganymede missions.

Prior to selecting a second propulsion module, the effects of using the smaller propulsion module on the mass requirements of the remaining modules were evaluated. After examining the propagation of the mass penalty for using the smaller module, a second module was selected which had a mass of 300,000 kg. The module could be used either singularly, in pairs, or in combination with the 75,000 kg module to satisfy the propulsion module requirements for all remaining maneuvers except the Earth orbit escape requirements for the Mercury, Ceres and Ganymede missions.

The effects of using the above common propulsion modules are shown in Figure 73 for an eight man crew. The lower bars represent the requirements if the modules are sized by the particular mission requirements while the upper bar represents the requirements which result from the use of the common modules. The data are based on the use of manned modules which are sized for the particular mission. The mass requirements which are shown include the additional requirements for meteoroid and thermal protection which were sized by the requirements of each mission. Also included is the mass of the interstage. The discontinuities in the requirements are due

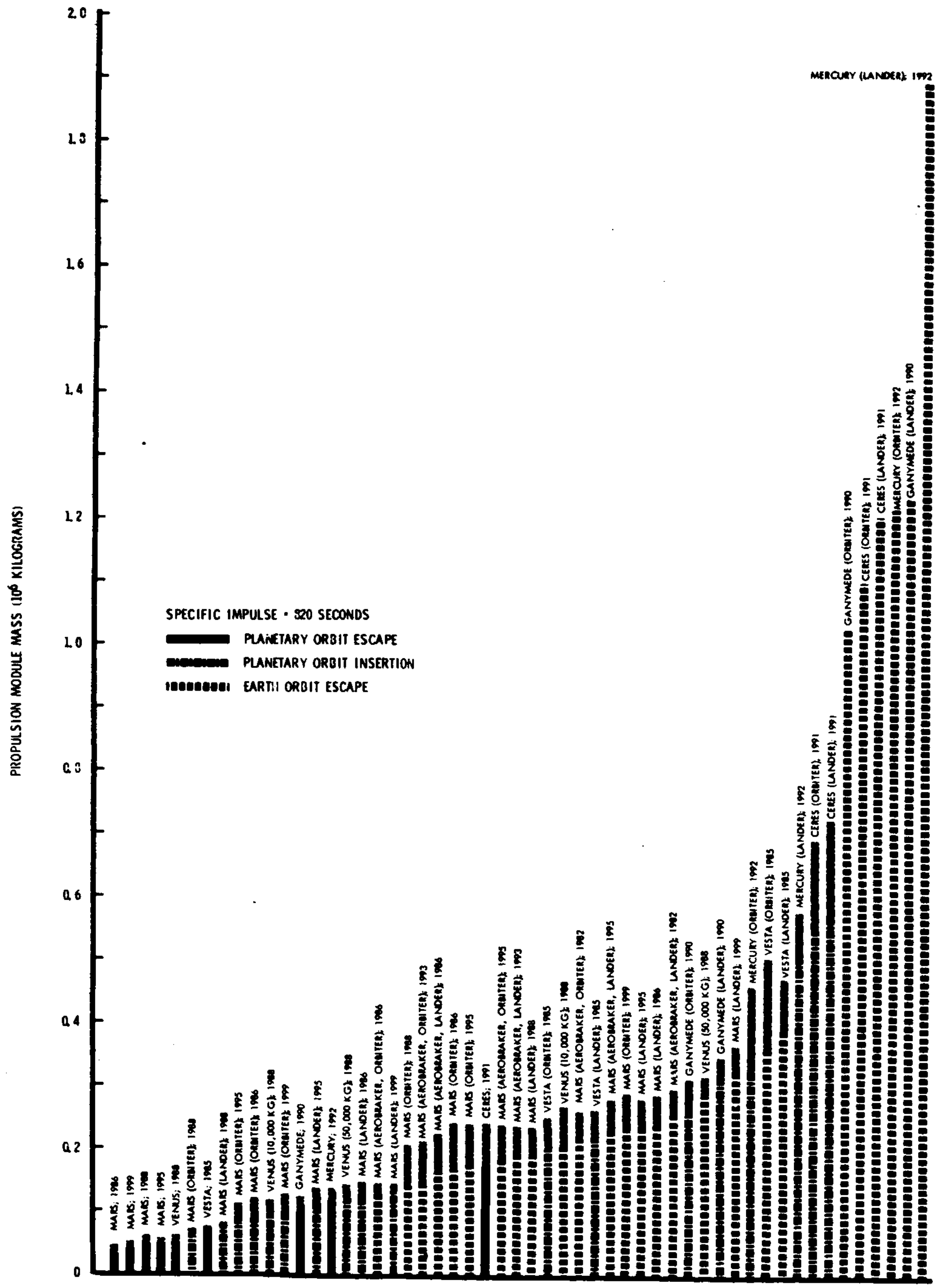

Figure 73. Propulsion Module Mass Requirements (Common Propulsion Modules, 8-Man Crew)

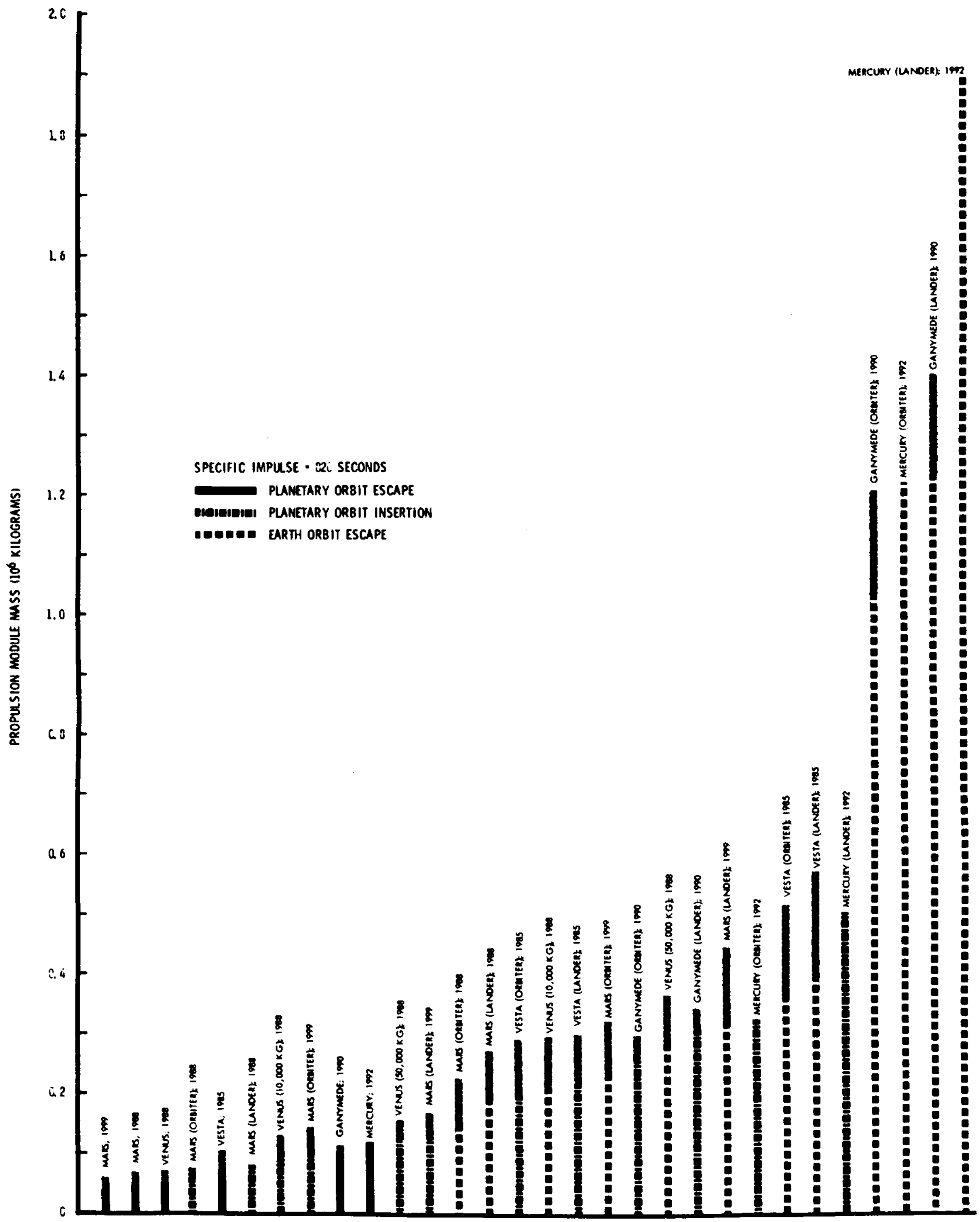

Figure 74. Nuclear Propulsion Module Mass Requirements (Common Manned and Propulsion Modules)

either to the addition of another propulsion module or due to changing from the 75,000 kg module to the 300,000 kg module.

Common Systems

The effects of using both the common manned modules and the common propulsion modules discussed in the previous sections were investigated for a limited number of mission opportunities. The resultant propulsion module mass requirements are shown in Figure 74. The lower bars represent the requirements if all modules are sized by the mission requirements while the upper bars represent the requirements resulting from the use of the common modules.

ELLIPTICAL PLANETARY PARKING ORBITS

Significant incremental velocity savings can be achieved if elliptical planetary parking orbits are considered. The magnitude of the savings, which are shown in the Performance Requirements section of Appendix A, are dependent upon the central body considered. The savings are most significant for Jupiter missions with low pericenter altitudes and the least significant for the asteroids. The effects of using elliptical planetary parking orbits were considered for Mercury, Venus, Mars, Jupiter and Ganymede missions and the results of the analyses are presented in this section.

The use of elliptical parking orbits will have no effect on the mass requirements of either the Earth reentry module or the mission module since they are independent of parking orbit eccentricity. Therefore, the data presented during the discussions of circular parking orbits are applicable to the present discussion. The use of such orbits will, however, effect the mass requirements of both the planetary excursion modules and the propulsion modules. The planetary excursion module mass requirements will increase with increasing eccentricity due to increased characteristic velocity requirements. The propulsion module mass requirements will decrease due to the decreased incremental velocity requirements.

Optimized Manned Modules

The planetary excursion module mass requirements are dependent upon the eccentricity of the planetary parking orbit since the characteristic velocity requirements vary with eccentricity. The effects of parking orbit eccentricity on the mass requirements of the Mercury, Mars, and Ganymede PEM's are shown in Figures 75 through 77 for crew sizes of four and ten men. The data include the mass of the interstage and meteoroid protection required during the transplanet mission phase. The requirements were determined using storable propellants with a specific impulse of 387 seconds and a bulk density of 1233 kg/m^3 (77 lb/ft^3). The mass requirements which are presented are

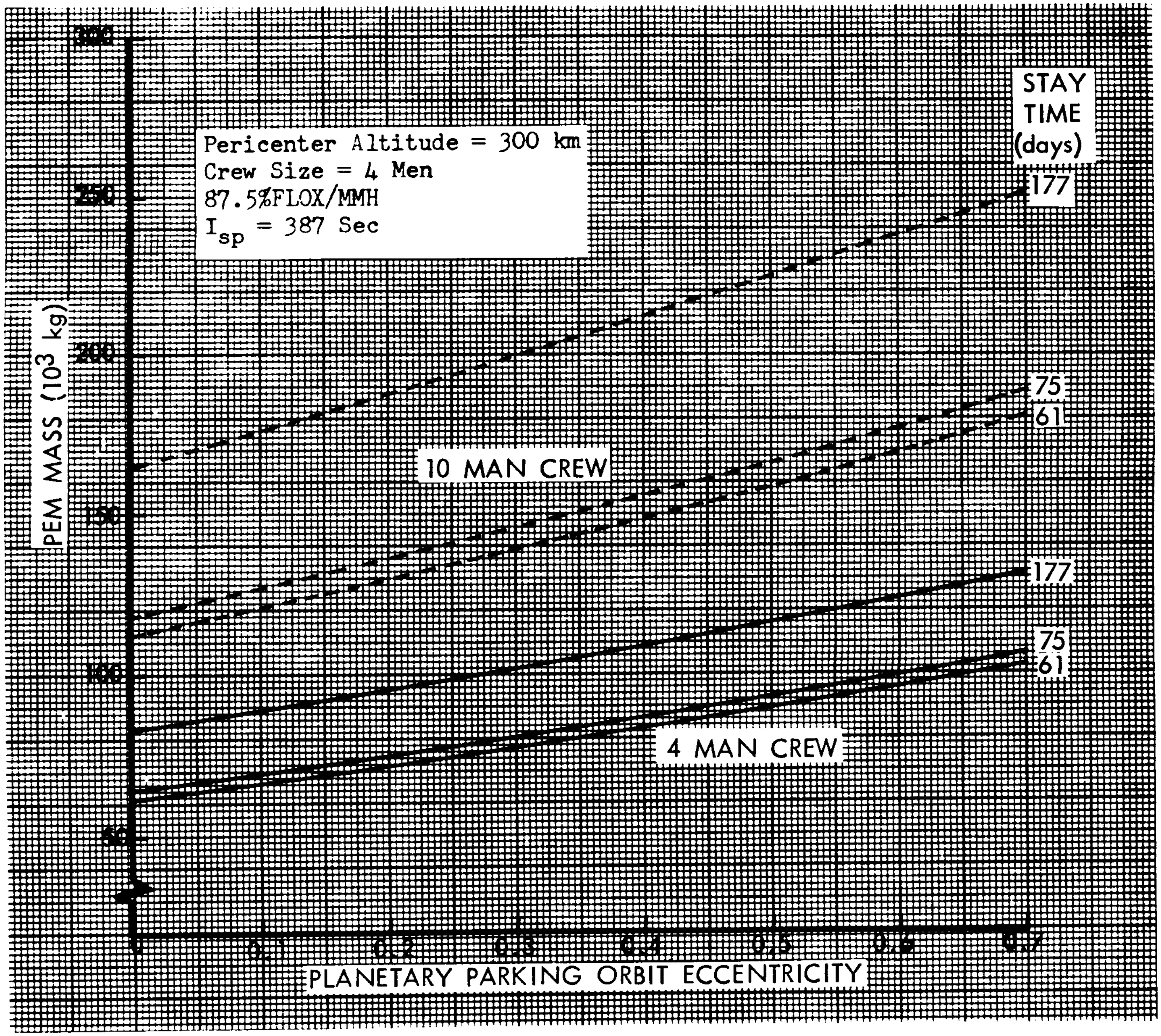

Figure 75. Mercury Planetary Excursion Module Mass

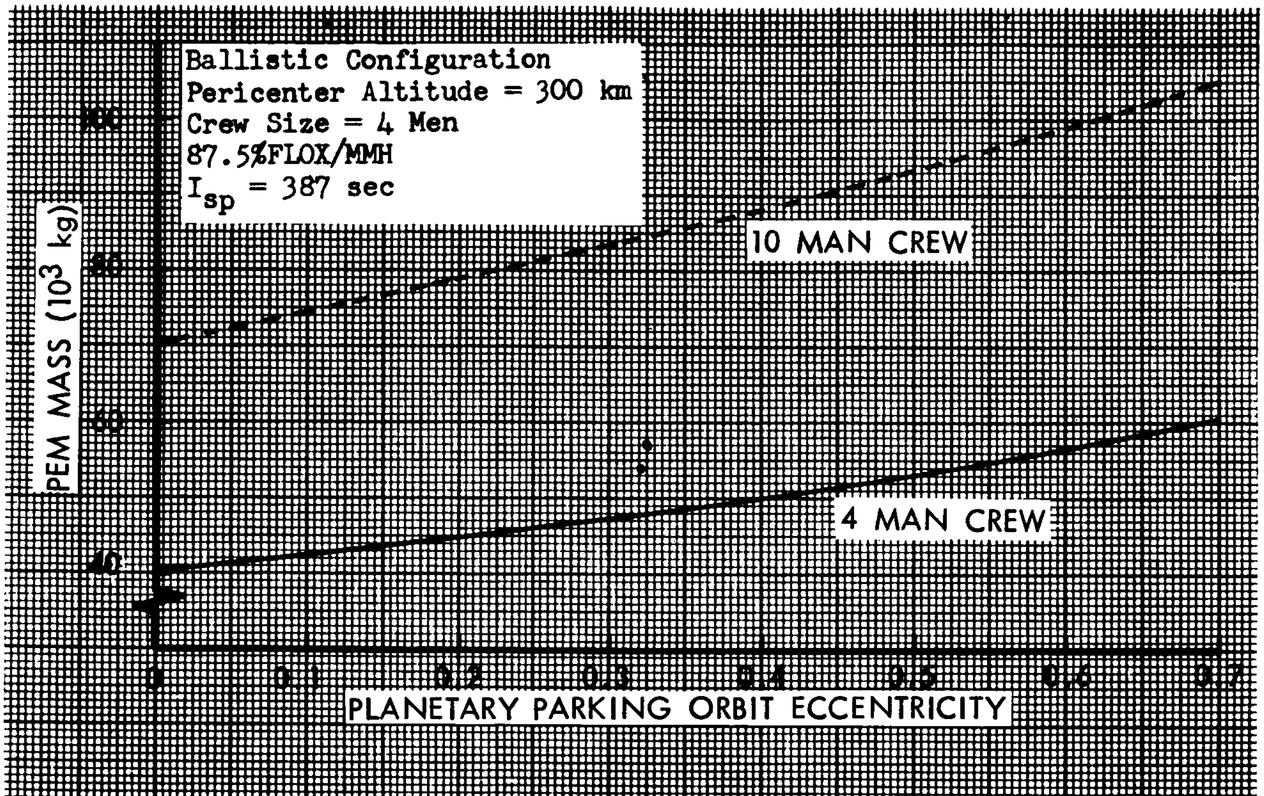

Figure 76. Mars Planetary Excursion Module Mass

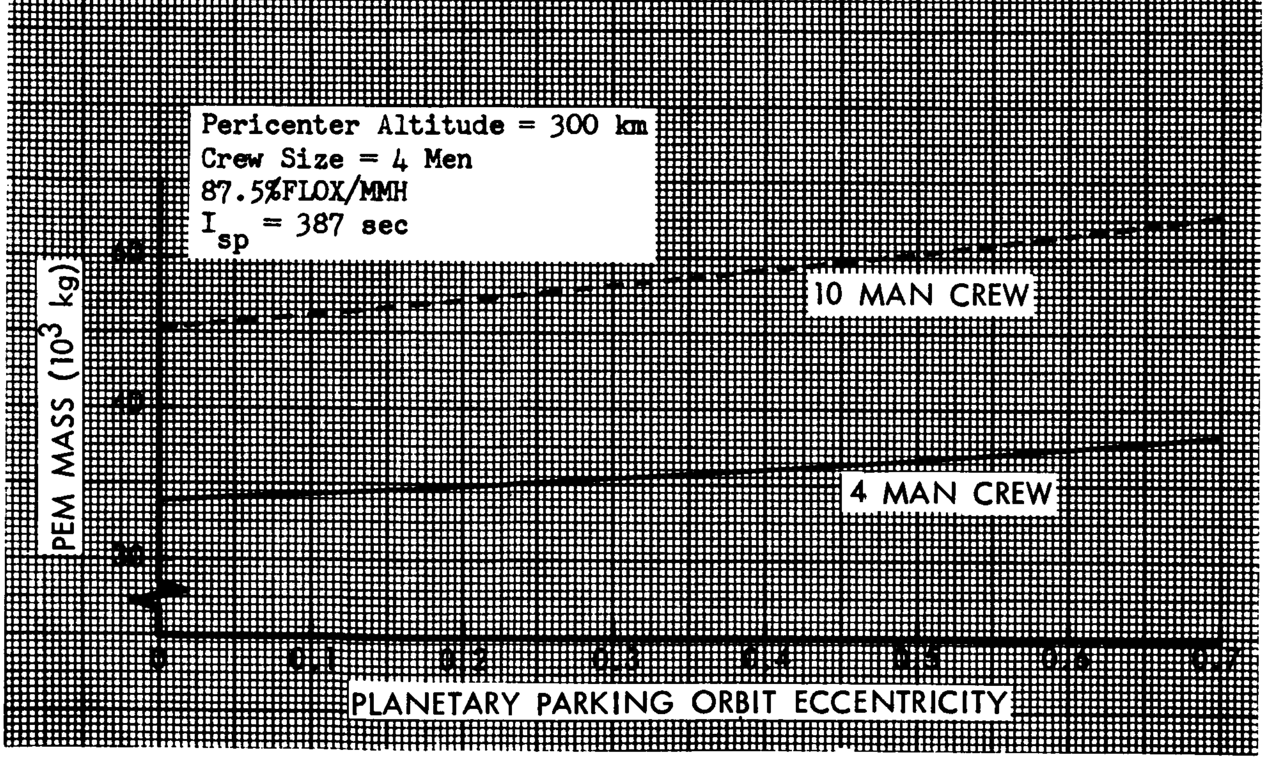

Figure 77. Ganymede Planetary Excursion Module Mass

based on an occupancy time of twenty-eight days for all mission objectives except Mercury. For Mercury missions, the mass requirements are shown for occupancy time of 61, 75, and 177 days. The occupancy times are two days less than the parking orbit stay times for the 1988, 1990 and 1992 mission opportunities, respectively.

The data presented in Figures 75 through 77 differ from the data presented in Figures 65 through 67 due to differences in the planetary parking orbit altitudes. During the investigations of the effects of elliptical parking orbits, new orbit altitudes were considered resulting in a change in the ascent and descent stage propellant requirements. The parking orbit altitudes used in the analyses are shown on the figures. The altitude differences can have sizeable effects on the planetary excursion module mass requirements, however, the effects on the total mass in Earth orbit are relatively small.

Optimized Propulsion Modules

During the investigations of the effects of elliptical parking orbits, only chemical and solid core nuclear propulsion modules were considered. The mass requirements of propulsion modules which are sized by the particular mission requirements are presented in the following paragraphs. The data are based on the use of the same manned module characteristics which were used during the investigations to the optimized propulsion modules for circular planetary parking orbits.

Chemical Propulsion Modules

The mass requirements of chemical propulsion modules for representative Mars and Venus mission opportunities are shown in Figures 78 through 80 as a function of the planetary parking orbit eccentricity for crew sizes of eight and twenty men. As noted previously, the mass requirements for intermediate crew sizes can be estimated quite accurately by linear interpolation. The planetary orbit insertion and escape module mass requirements are based on a specific impulse of 387 seconds while the Earth orbit escape requirements are based on a specific impulse of 450 seconds. Also included in the data are the Earth orbit escape module mass requirements for representative Mars aerobraker missions.

The significant effect of the Venus parking orbit eccentricity is quite apparent. It can be seen from Figure 78 that the planetary orbit escape propulsion module mass requirements can be decreased by over fifty percent by increasing the eccentricity from zero (circular orbit) to 0.7. The planetary orbit insertion requirements (Figures 78 and 79) can be decreased by over a factor of four while the Earth orbit escape requirements can be decreased by

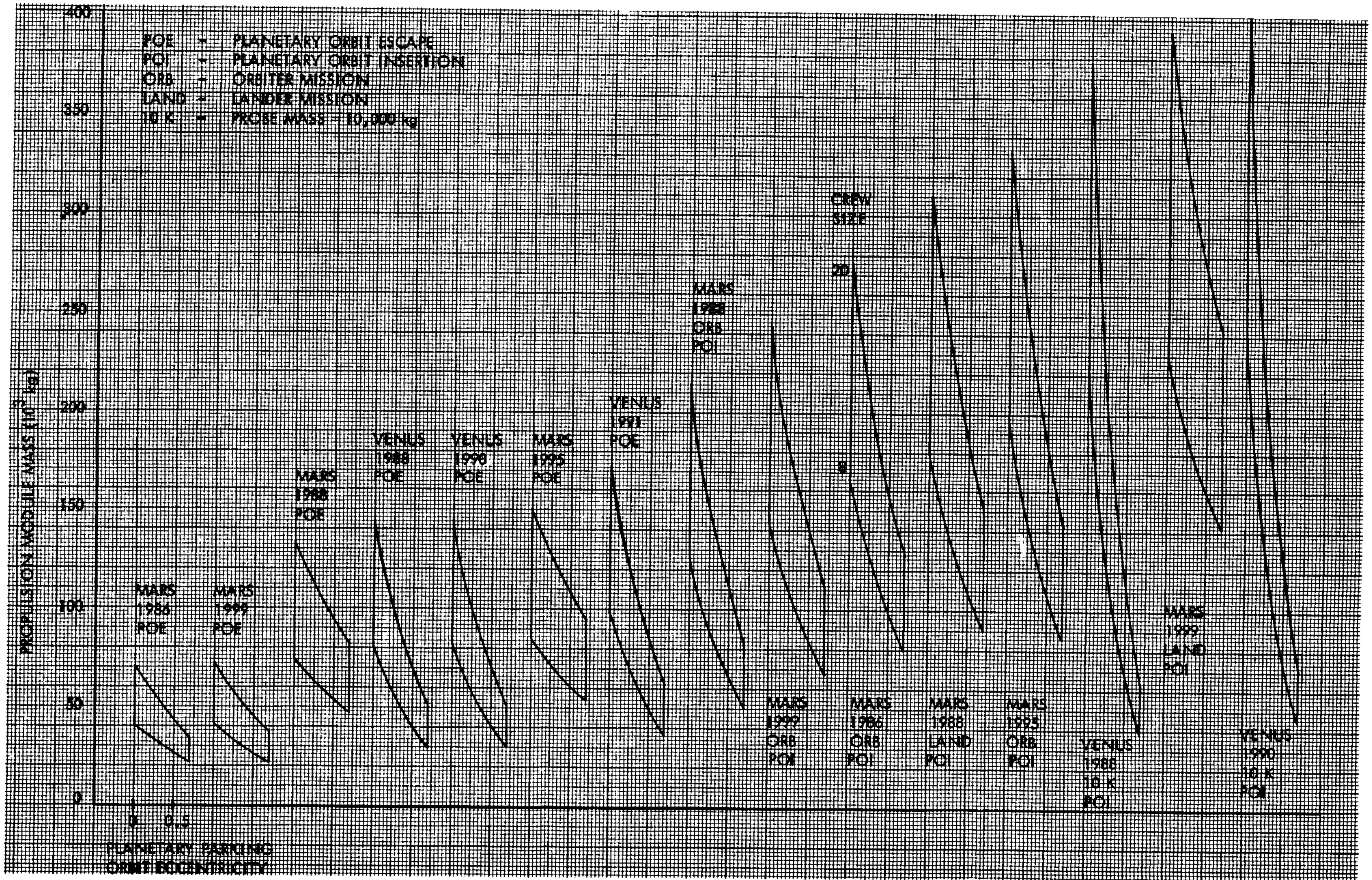

Figure 78. Chemical Propulsion Module Mass Requirements (0 - 400,000-kg)

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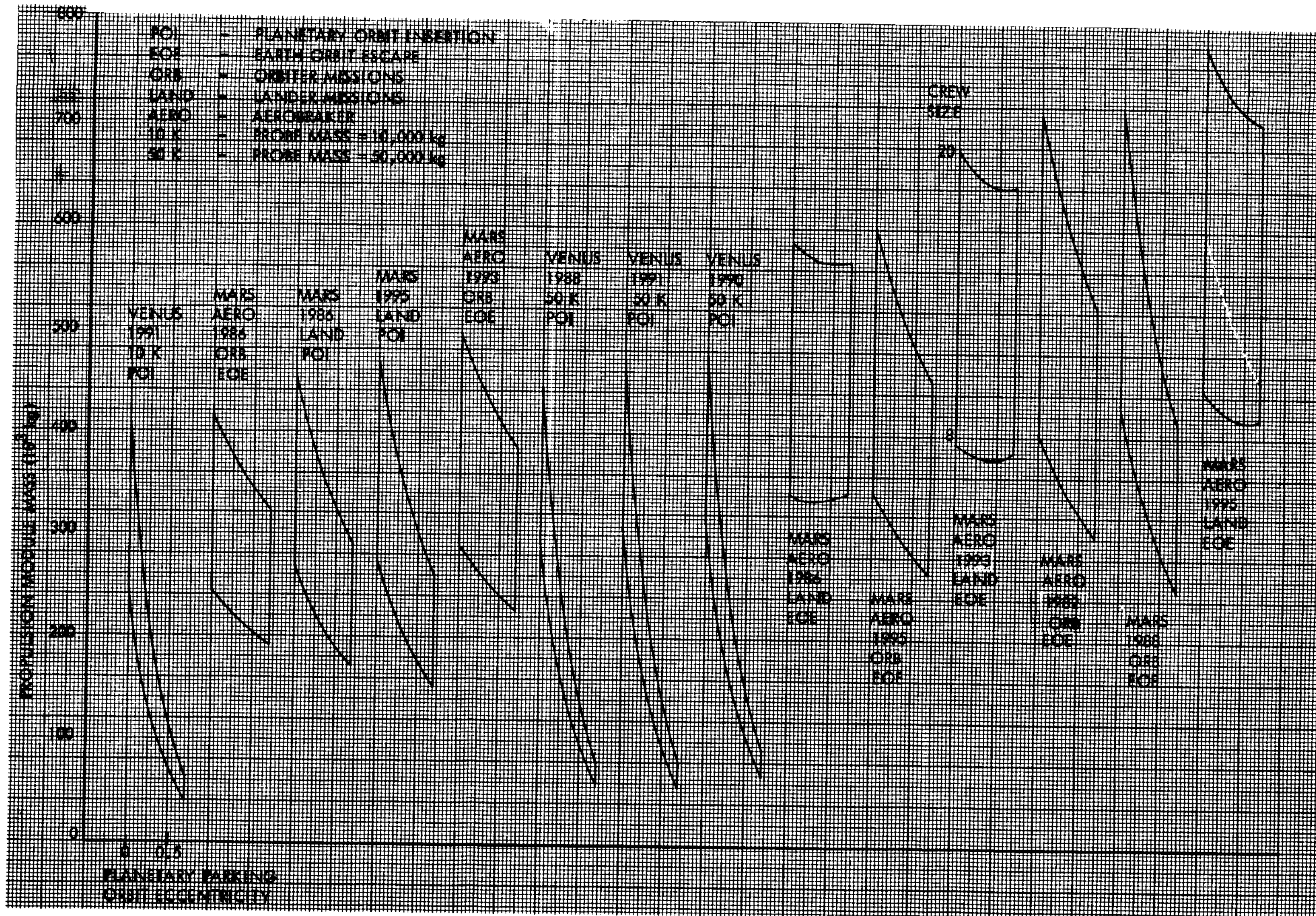

Figure 79. Chemical Propulsion Module Mass Requirements (0-800,000-kg)

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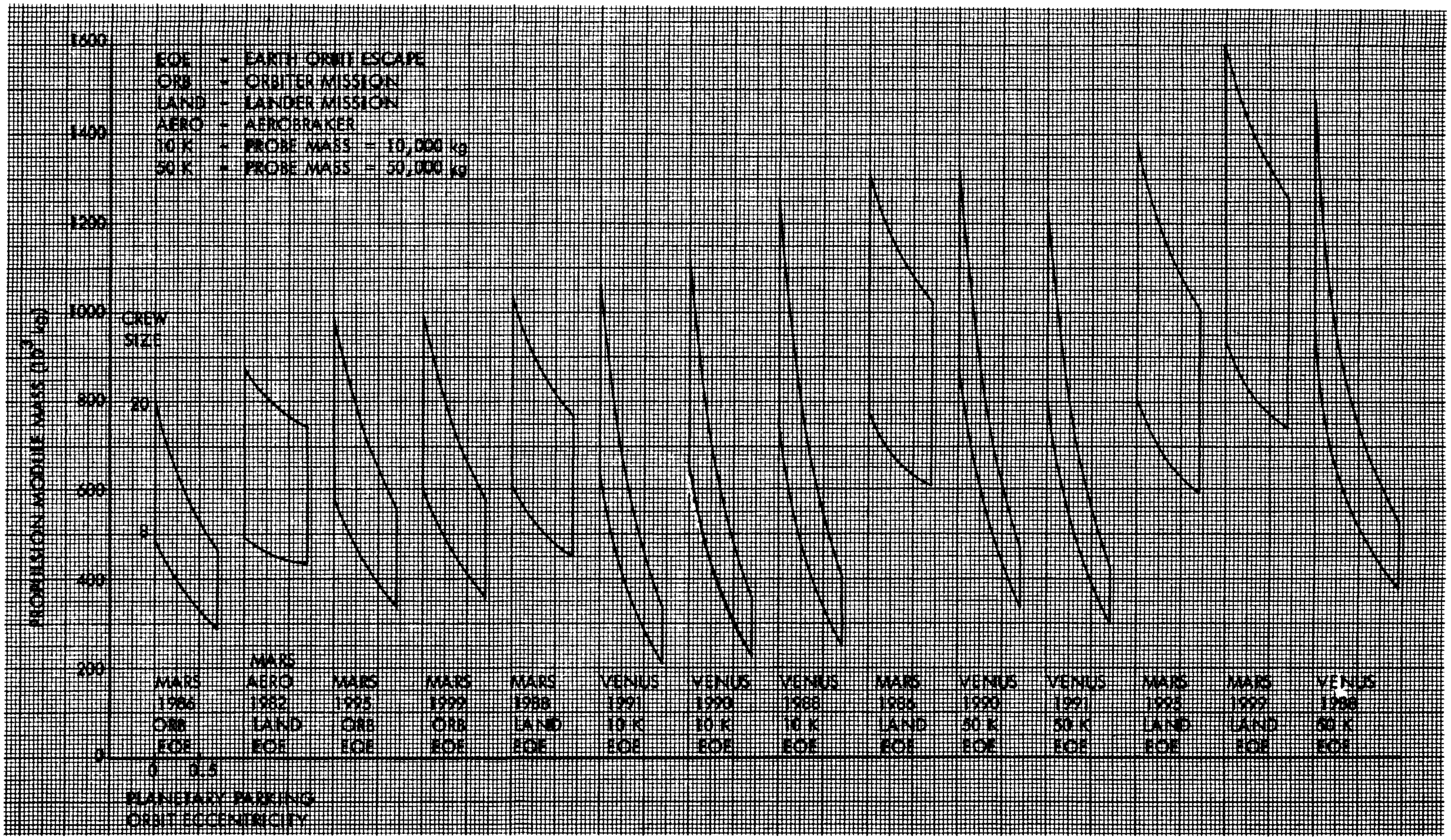

Figure 80. Chemical Propulsion Module Mass Requirements (0 - 1,600,000-kg)

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a factor of approximately three (Figure 80). 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 with the lower (10,000-kg) probe mass. Although the planetary orbit insertion module payload is greater, the insertion incremental velocity requirements are between 55 percent and 75 percent of the planetary orbit escape requirements resulting in nearly identical propulsion module mass requirements.

The effects of parking orbit eccentricity on the propulsion module mass requirements for Mars retrobraker missions are less significant due to the lower mass of Mars. By increasing the eccentricity from zero to 0.7, the mass requirements decrease by approximately thirty percent for planetary orbit escape. For orbiter missions, the decreases in the mass requirement are forty percent for planetary orbit insertion and twenty-five percent for Earth orbit escape. The corresponding decreases in the mass requirements for lander missions are 0 percent for planetary orbit insertion and 0 percent for planetary orbit escape.

Elliptic planetary parking orbits for Mars aerobraker lander missions do not produce very significant variations in the Earth orbit escape propulsion module mass requirements. For those missions which have the lower planetary orbit escape incremental velocity requirements, it is possible to minimize the mass requirements of the Earth orbit escape propulsion module by varying the parking orbit eccentricity (Figure 79). The optimization is a result of reduced planetary orbit escape requirements and increased planetary excursion module characteristic velocity requirements as eccentricity is increased. As the planetary orbit escape incremental velocity requirements increase, the eccentricity at which the mass requirements are minimized increases until the minimum mass is obtained at the maximum eccentricity considered (Figure 80).

The planetary orbit insertion and escape incremental velocity requirements for Jupiter orbiter missions with highly eccentric orbits are of the same magnitude as the requirements for Mars and Venus missions. Therefore, an investigation was conducted to determine the propulsion module mass requirements if chemical stages were used for these maneuvers. The resultant module mass requirements are shown in Table 18. The data are based on an eight-man crew, a probe mass of 10,000 kg, a Jupiter pericenter altitude of ten Jupiter radii, a specific impulse of 387 seconds for the planetary orbit insertion and planetary orbit escape propulsion modules, and a specific impulse of 820 seconds for the earth orbit escape propulsion module. Even with an eccentricity of only 0.3, the propulsion module mass requirements are not excessive when compared with the requirements for planetary orbit insertion and earth orbit escape for the Mars and Venus missions. For purposes of comparison, the corresponding mass requirements for an all nuclear system are also presented in Table 17. By comparing the all nuclear

Table 18. Chemical Propulsion Mass Requirements (Jupiter Mission)

Planetary Orbit Eccentricity	Planetary Orbit Escape I_{sp} (sec)	Planetary Orbit Escape Mass (kg)	Planetary Orbit Insertion I_{sp} (sec)	Planetary Orbit Insertion Mass (kg)	Earth Orbit Escape I_{sp} (sec)	Earth Orbit Escape Mass (kg)	Mass in Earth Orbit (kg)
0	387	363,000	387	2,820,000	820	8,848,000	12,142,000
0.3	387	179,000	387	798,000	820	2,866,000	3,917,000
0.7	387	76,000	387	183,000	820	900,000	1,219,000
0	820	169,000	820	642,000	820	2,386,000	3,267,000
0.3	820	98,000	820	256,000	820	1,146,000	1,562,000
0.7	820	51,000	820	89,000	820	564,000	762,000

systems with the systems that employ chemical modules at Jupiter, it can be seen that similar Earth orbit escape propulsion module mass requirements can be achieved if a higher eccentricity is employed when chemical modules are used. It will be shown in the next section, however, that the Earth orbit escape propulsion module mass need not exceed about 1,100,000 kg with nuclear upper stages if both Jupiter and Ganymede orbiter missions are considered. If the same Earth orbit escape module were used with chemical upper stages, an orbit eccentricity of approximately 0.66 would be required. This orbit has a period of 23 days for the assumed pericenter altitude. Based on the results of the limited analyses which were performed, it appears that the use of chemical propulsion modules for Jupiter planetary orbit insertion and escape will be limited by operational considerations and the mass of the earth orbit escape module rather than the mass of the chemical modules.

Solid Core Nuclear Propulsion Modules

The mass requirements of solid core nuclear propulsion modules are shown in Figures 81 through 87 for Mercury, Venus, Mars, Jupiter, and Ganymede missions with eccentric planetary parking orbits. The data are based on a specific impulse of 820 seconds. Included in the data are the nuclear Earth orbit escape propulsion module mass requirements for Mars and Venus missions using storable upper stages. The Earth orbit escape module mass requirements for Mars aerobraker missions using storable upper stages are also shown.

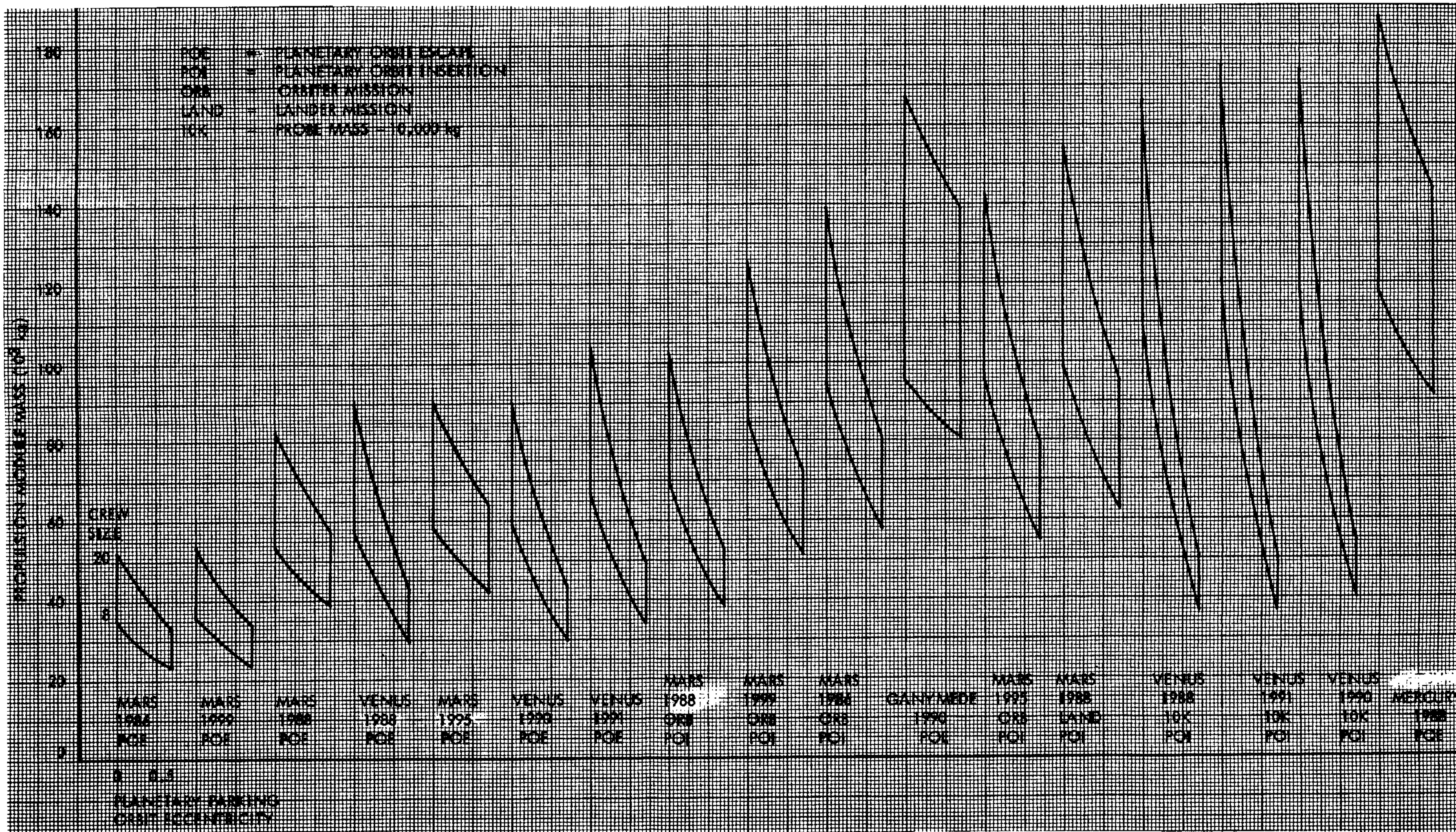

Figure 81. Nuclear Propulsion Module Mass Requirements (0 - 200,000 kg)

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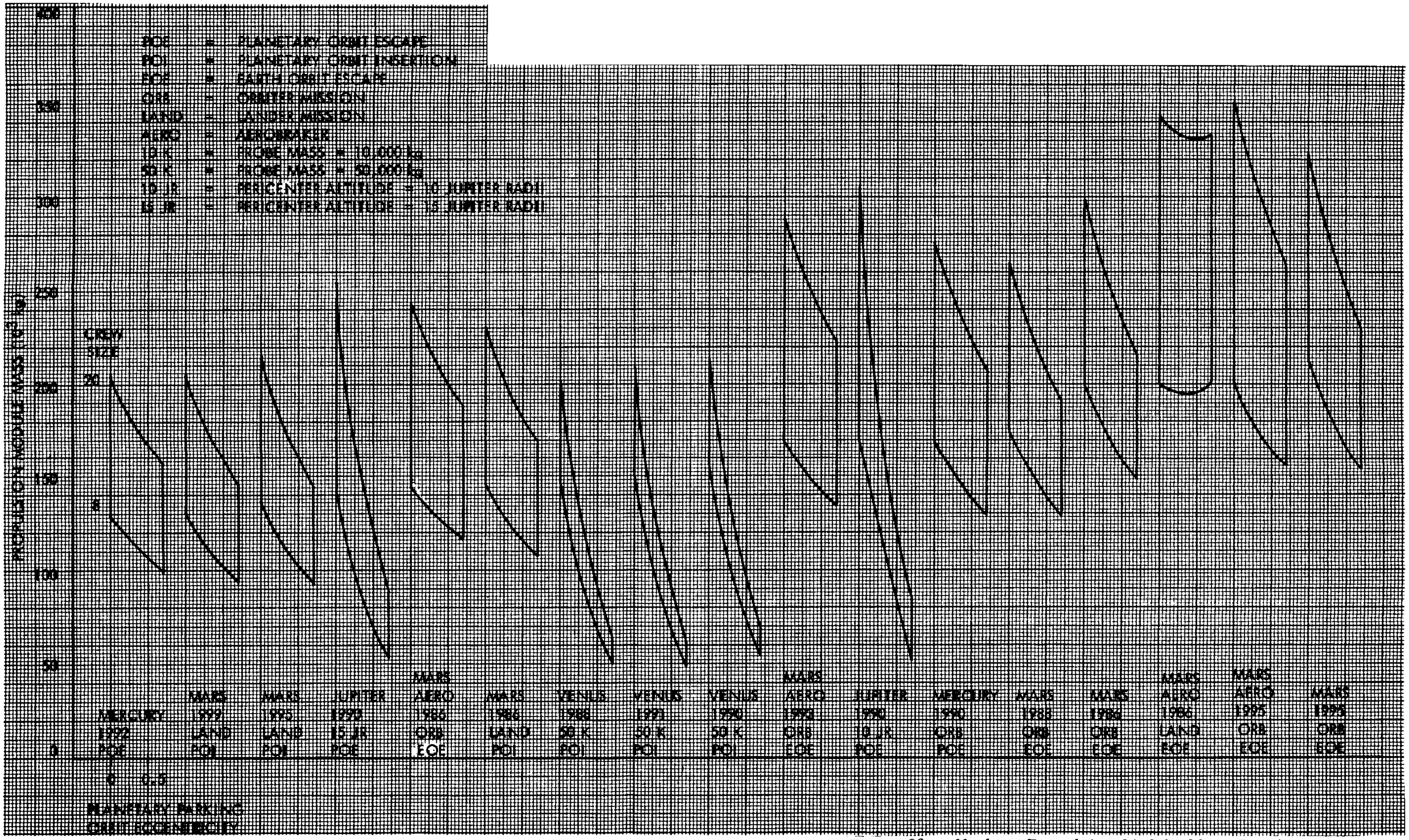

Figure 82. Nuclear Propulsion Module Mass Requirements (0 - 400,000-kg)

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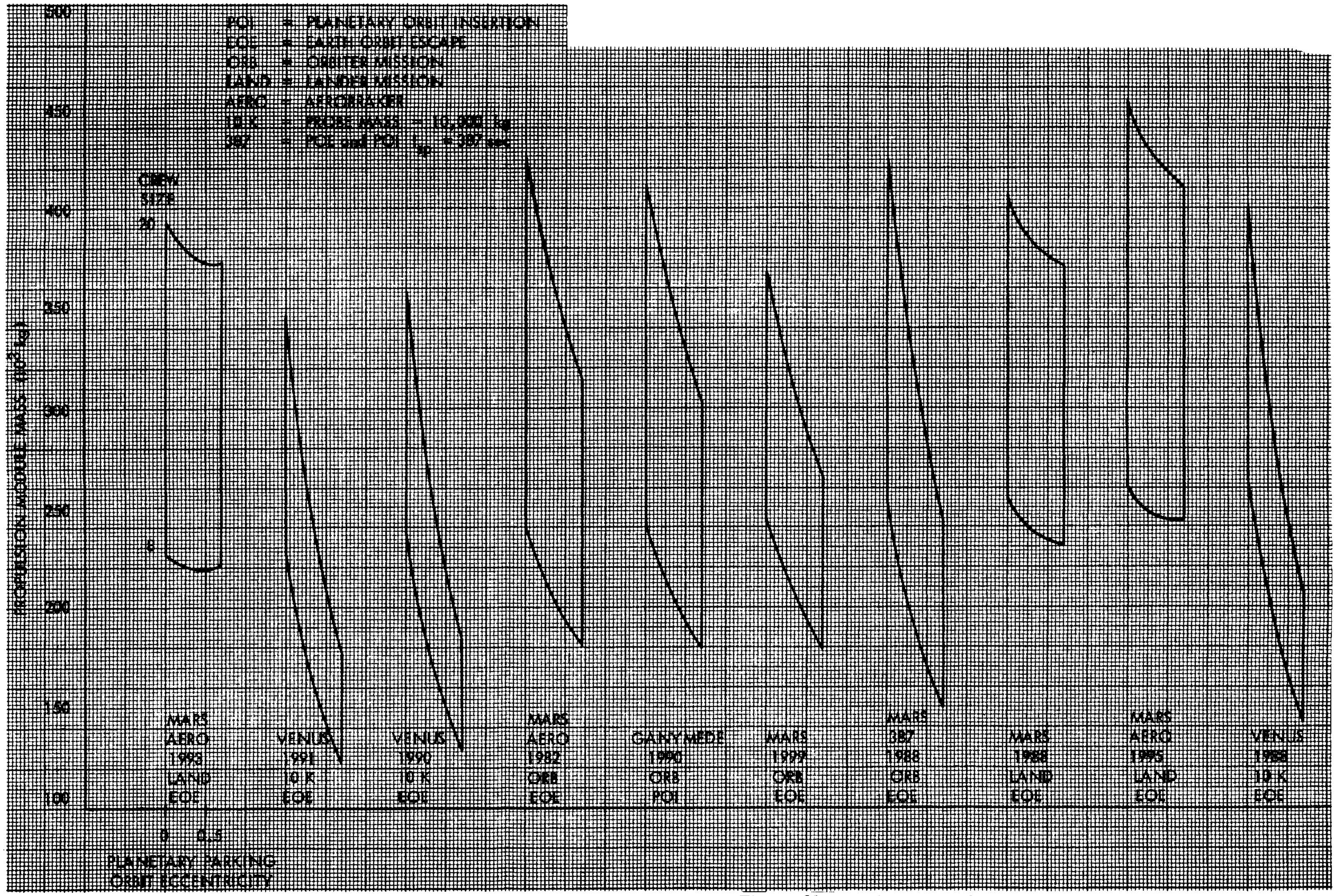

Figure 83. Nuclear Propulsion Module Mass Requirements (100,000 - 500,000 kg)

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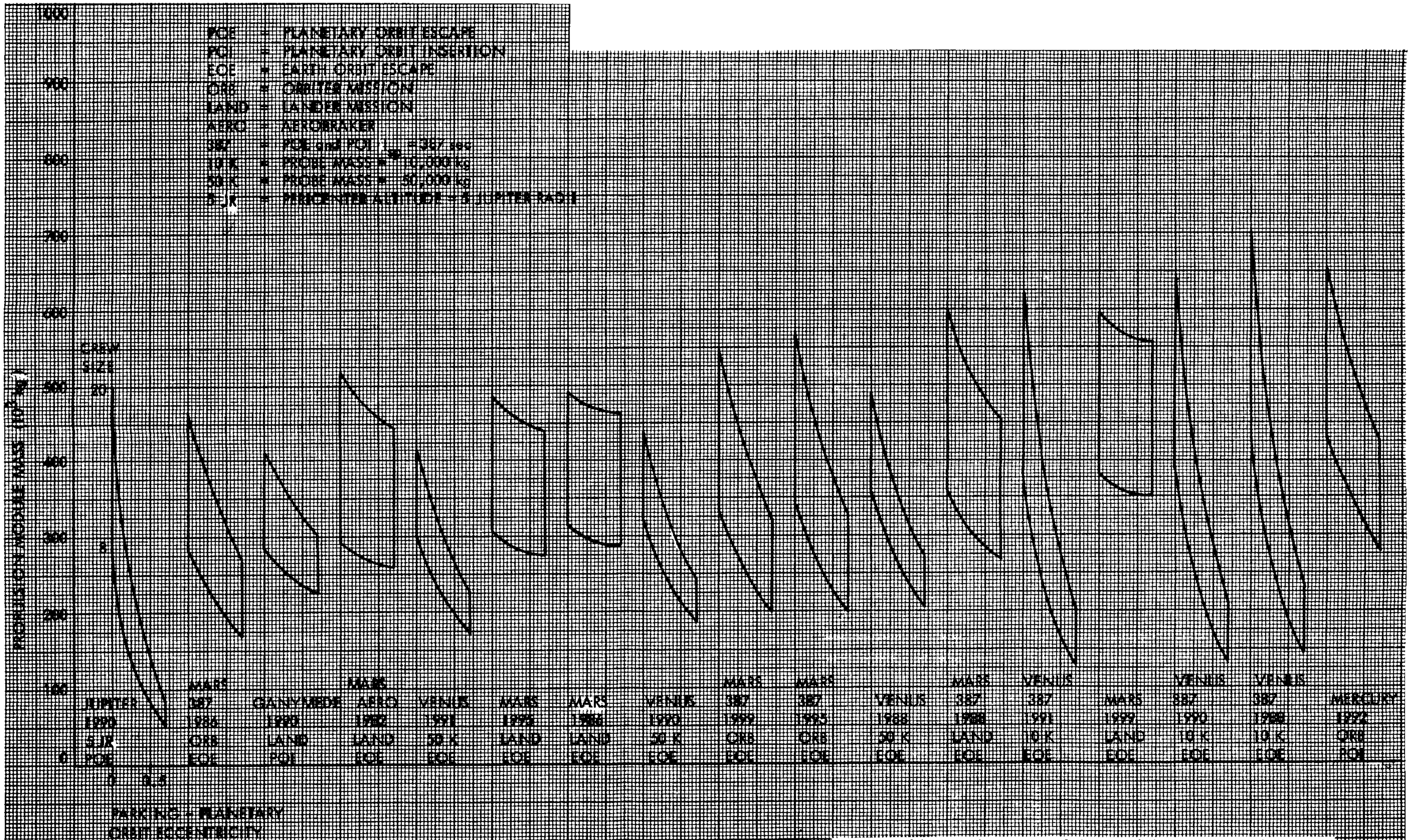

Figure 84. Nuclear Propulsion Module Mass Requirements (0 - 1,000,000 kg)

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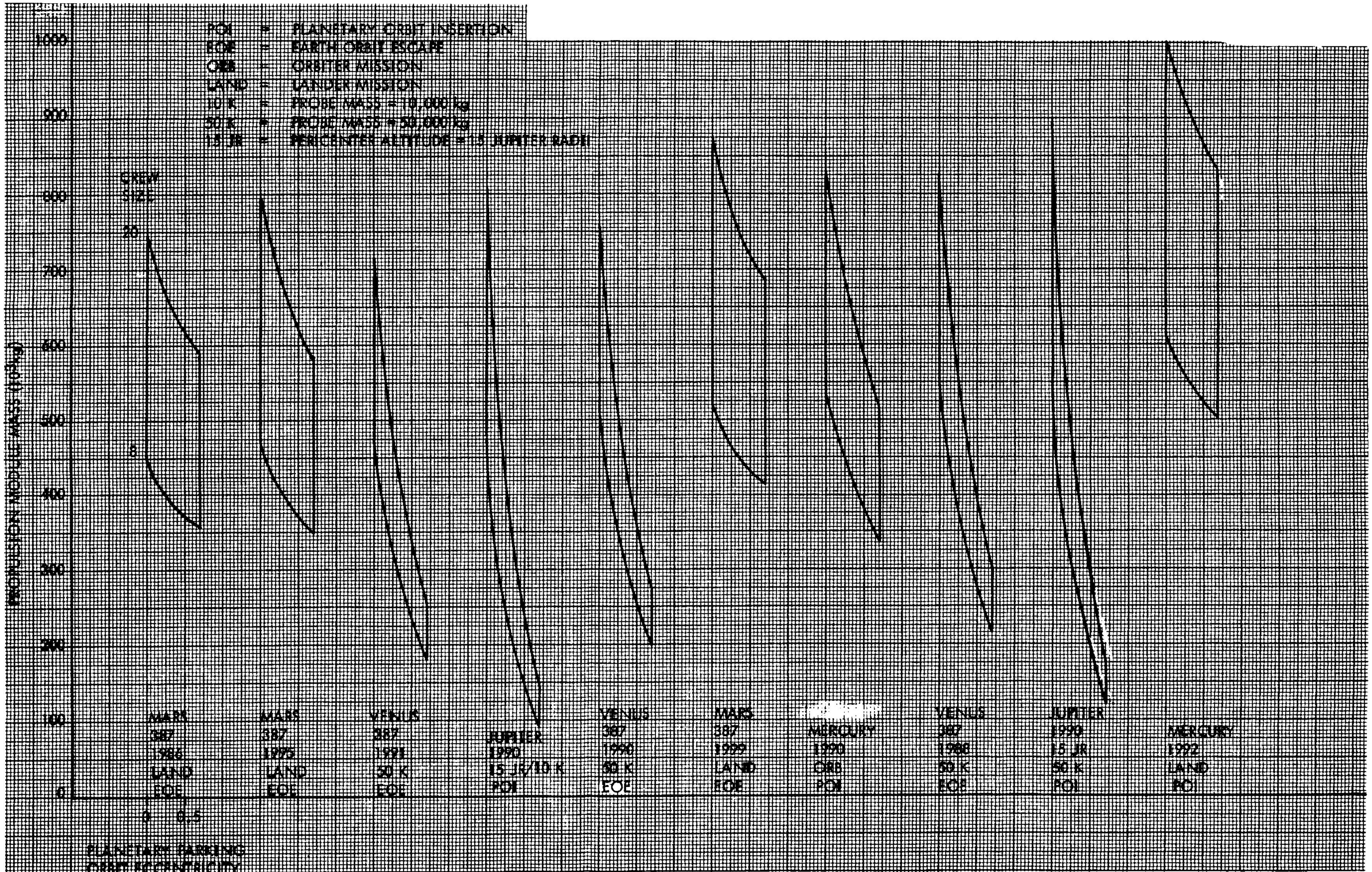

Figure 85. Nuclear Propulsion Module Mass Requirements (0 - 1,000,000-kg)

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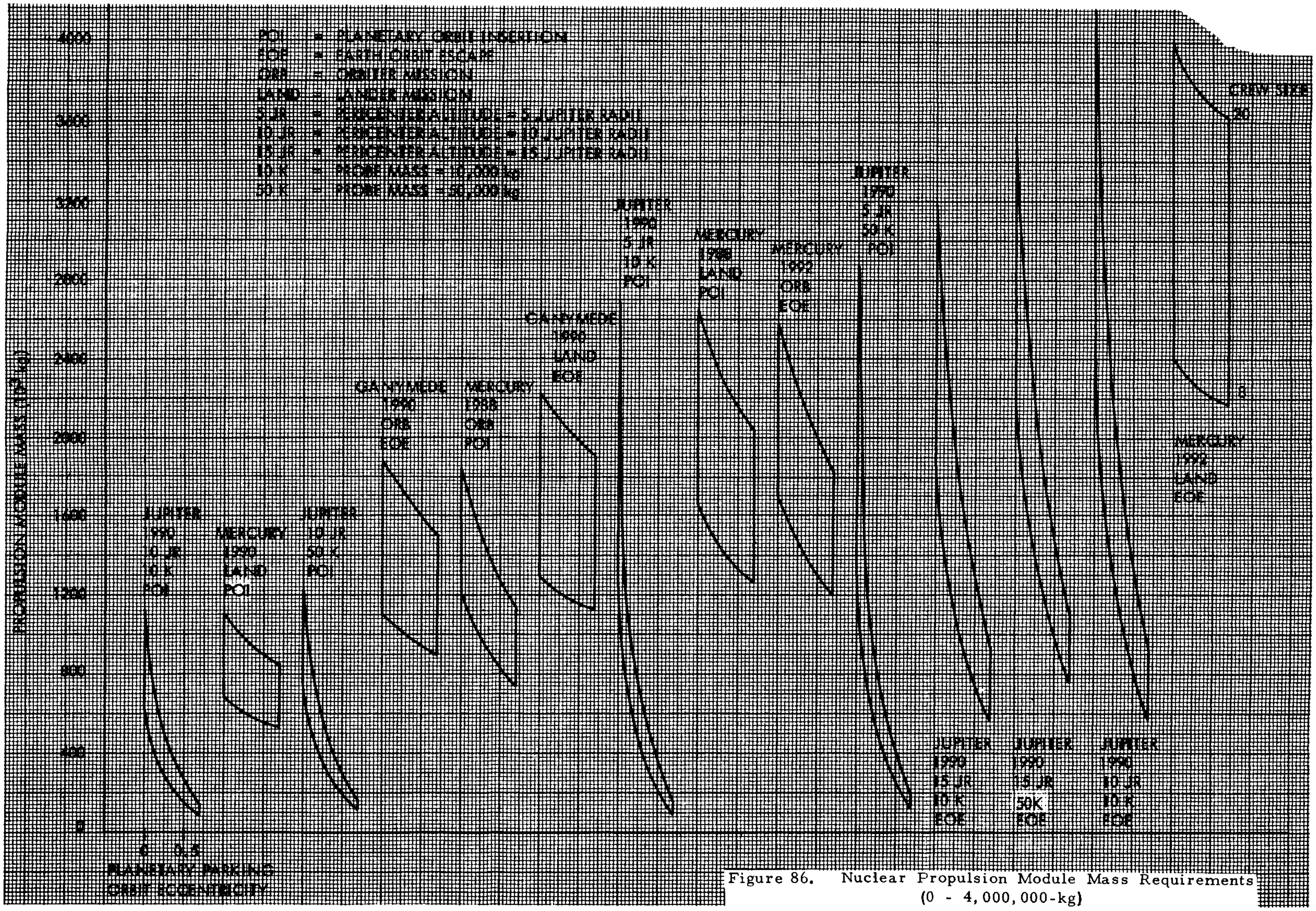

Figure 86. Nuclear Propulsion Module Mass Requirements (0 - 4,000,000-kg)

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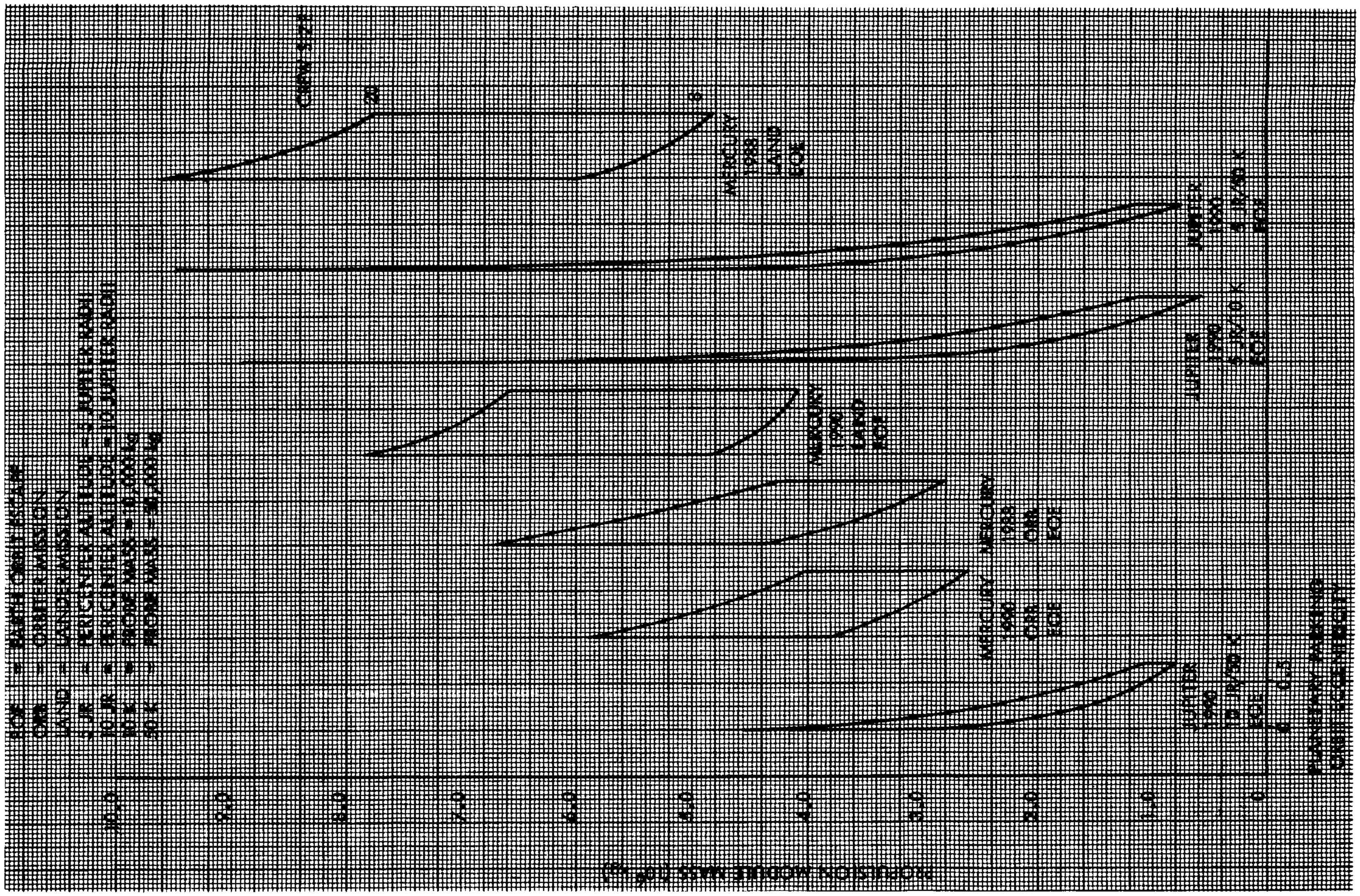

Figure 87. Nuclear Propulsion Module Mass Requirements (0 - 10,000,000-kg)

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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 from Figures 81 and 85 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.

Optimized Systems

The mass in Earth orbit requirements for the missions which have been considered in the previous sections are presented in Figure 88 for chemical propulsion modules, and Figures 89 through 92 for nuclear propulsion modules. The significance of elliptical orbits can be seen by comparing the mass in Earth orbit requirements for the Mars orbiter missions with chemical upper stages and the same missions with nuclear upper stages. As the eccentricity increases, the differences in the mass requirements decrease.

Common Manned Modules

During the investigations of the effects of elliptical planetary parking orbits, additional analyses were performed to determine the effects of using common manned modules. The Earth reentry module and mission module design parameters were the same as those used during the circular orbit analyses with the exception of crew size. For this investigation, it was assumed that the modules were designed for a crew size of twenty men but used by eight men. The consumables were provided only as required by the crew size and mission.

The resultant nuclear propulsion module mass and mass in Earth orbit requirements are shown in Figures 93 through 97. The mass requirements are represented by the solid lines while the broken lines show, for purposes of comparison, the mass requirements using modules which are designed by the crew size and mission. By referring to Figures 96 and 97, it can be seen that the mass in Earth orbit penalty for using the over-designed manned modules is between 20 and 30 percent. It can also be seen that for some cases, the same propulsion modules which are required for circular orbits with eight man modules could also be used with the off-loaded 20 man modules by using elliptical orbits.

Common Propulsion Modules

Within the constraint of employing circular capture orbits the establishment of common propulsion modules is relatively straightforward. Regions of common propulsion module requirements can be defined by

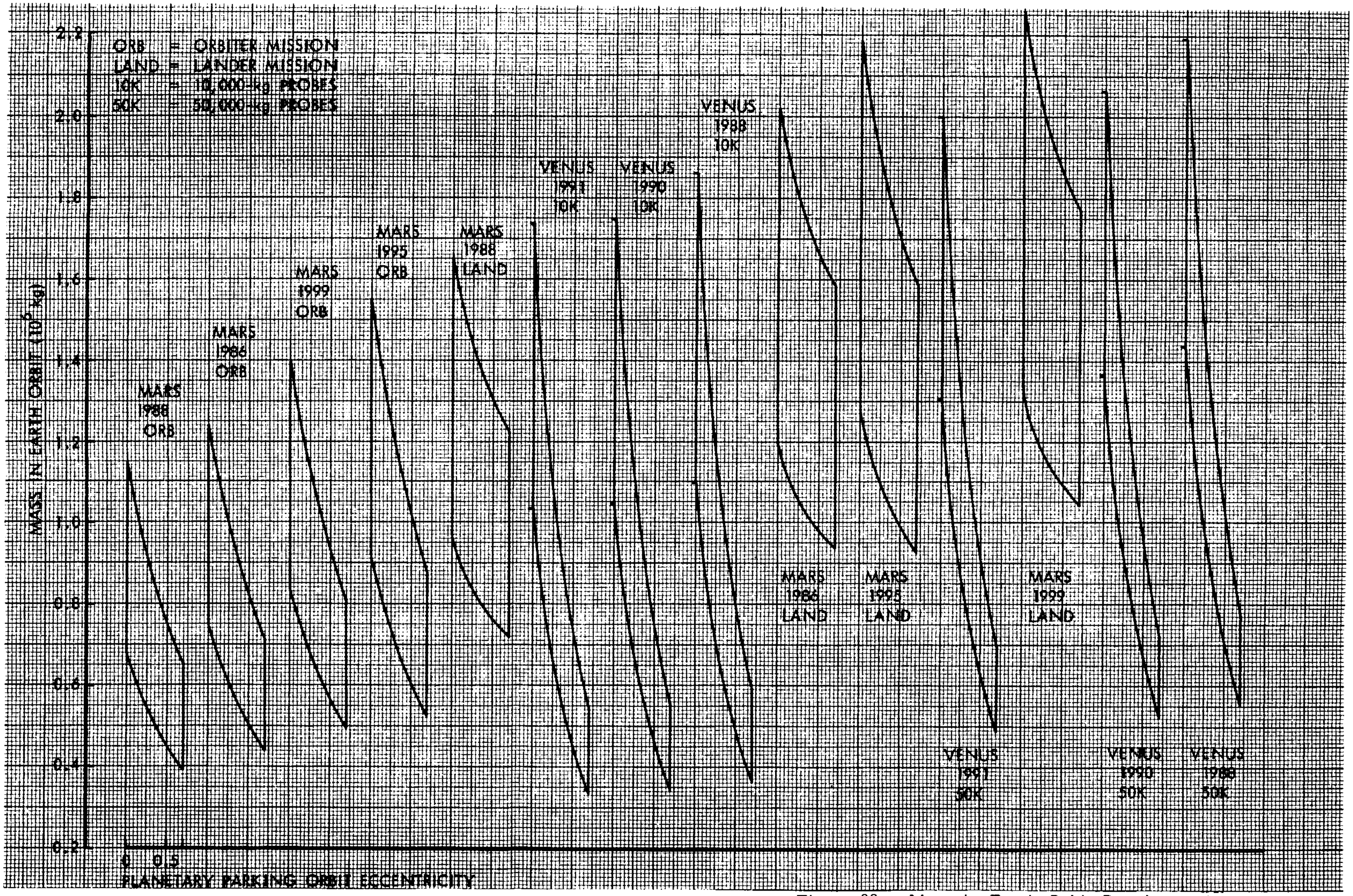

Figure 88. Mass in Earth Orbit Requirements (Chemical Propulsion Modules)

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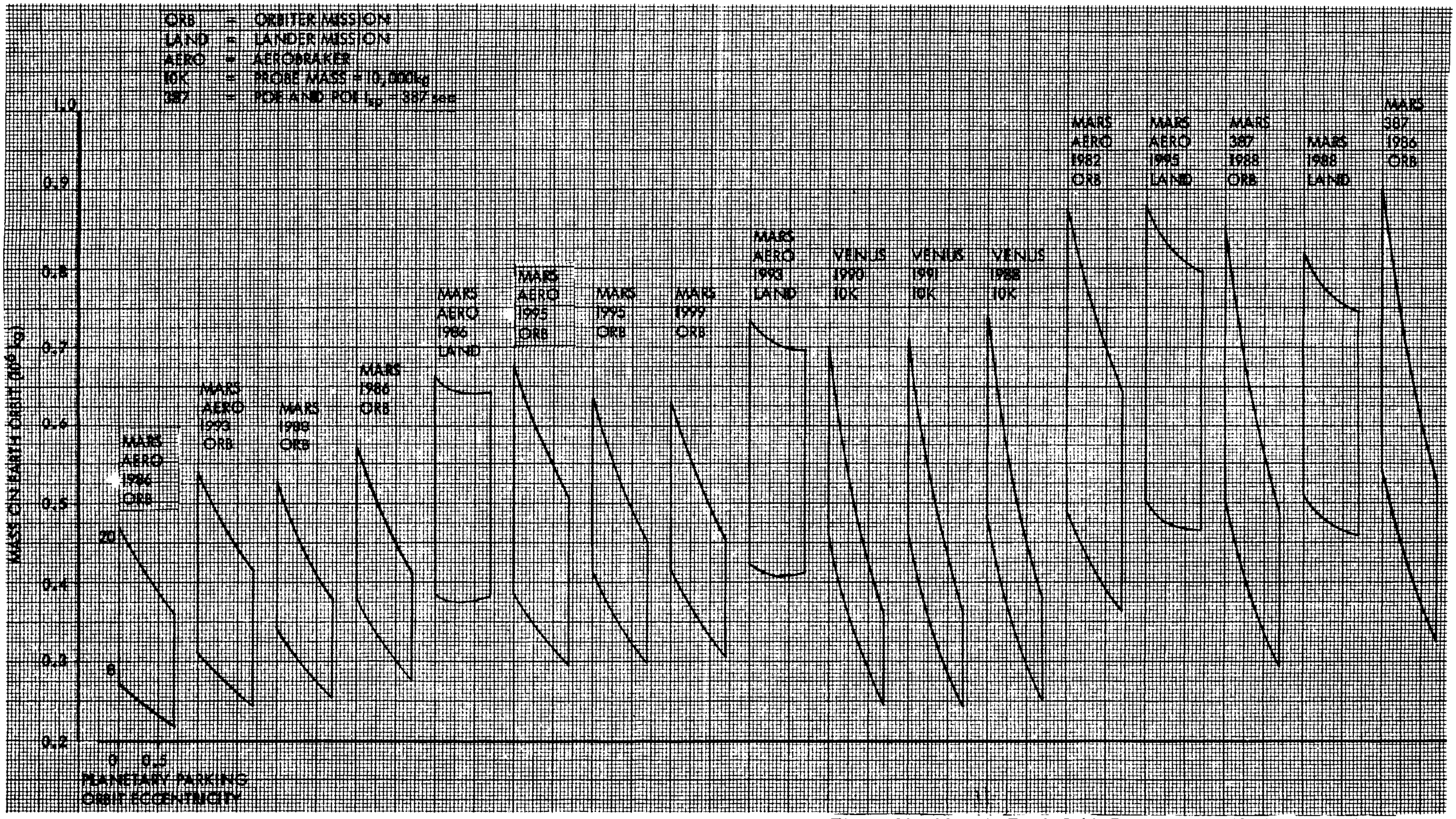

Figure 89. Mass in Earth Orbit Requirements Nuclear Propulsion Modules (200,000 - 1,000,000 kg)

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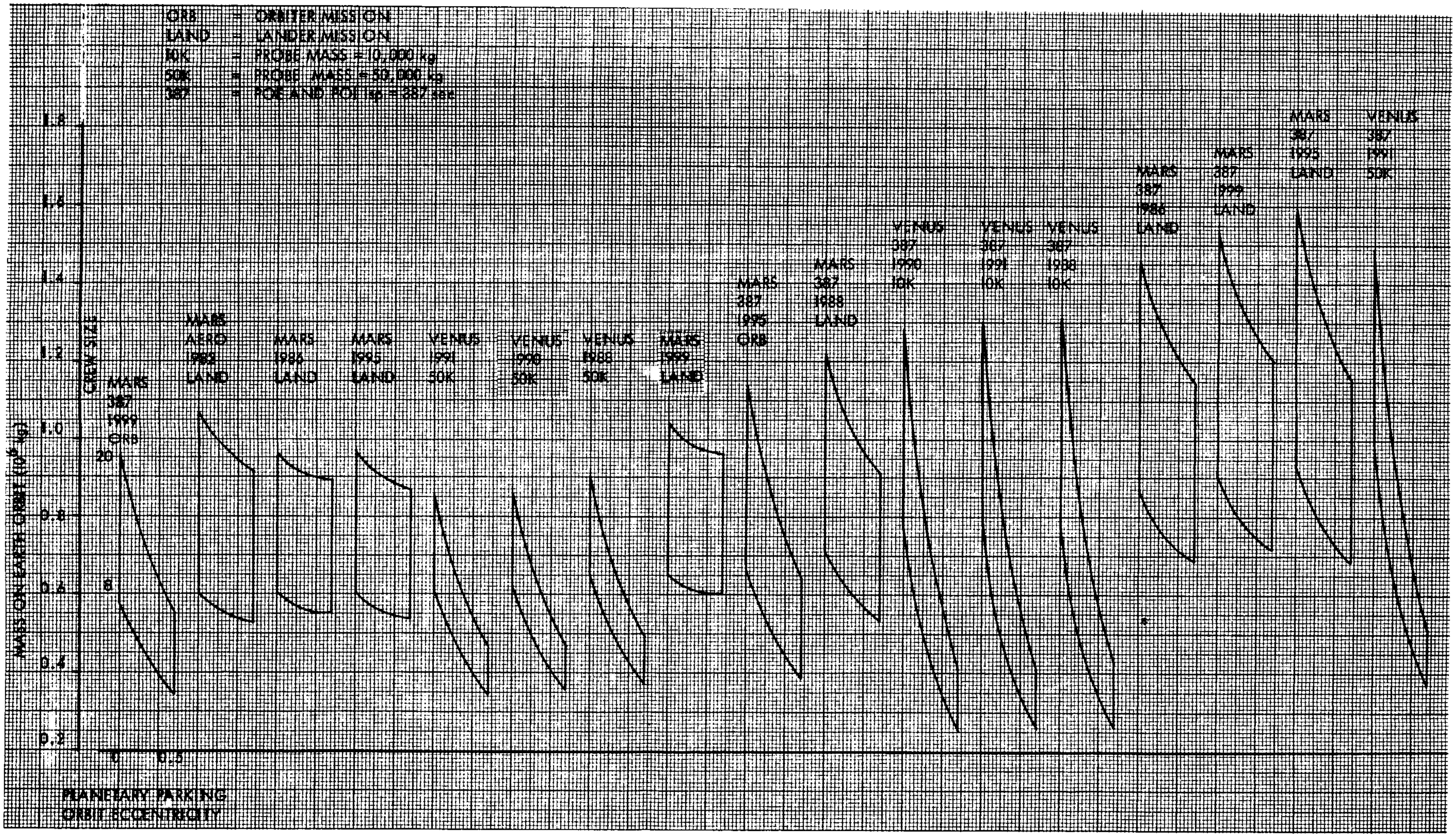

Figure 90. Mass in Earth Orbit Requirements Nuclear Propulsion Modules (200, 000 - 1, 800, 000 kg)

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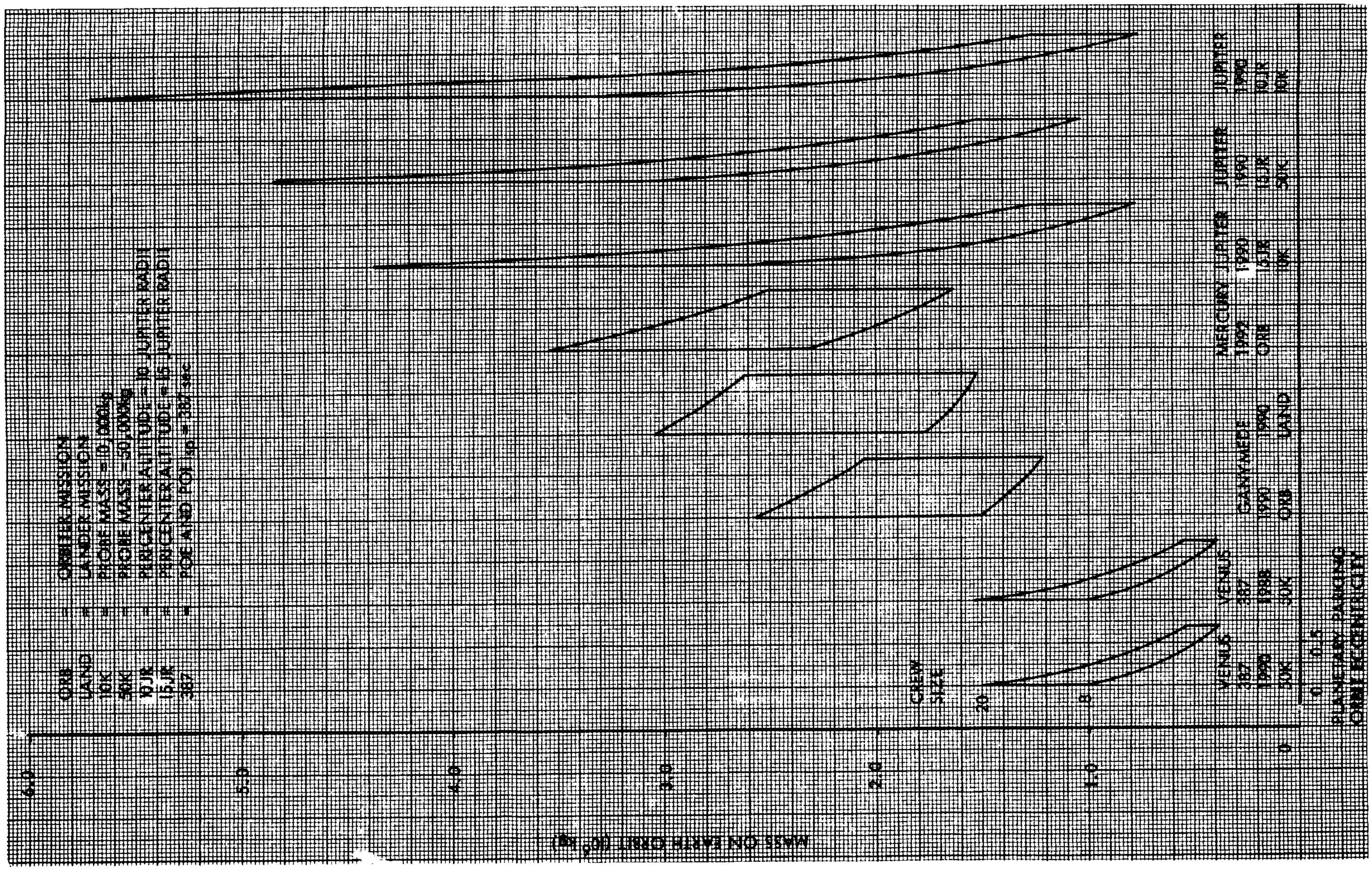

Figure 91. Mass in Earth Orbit Requirements for Nuclear Propulsion Modules (0 - 6,000,000-kg)

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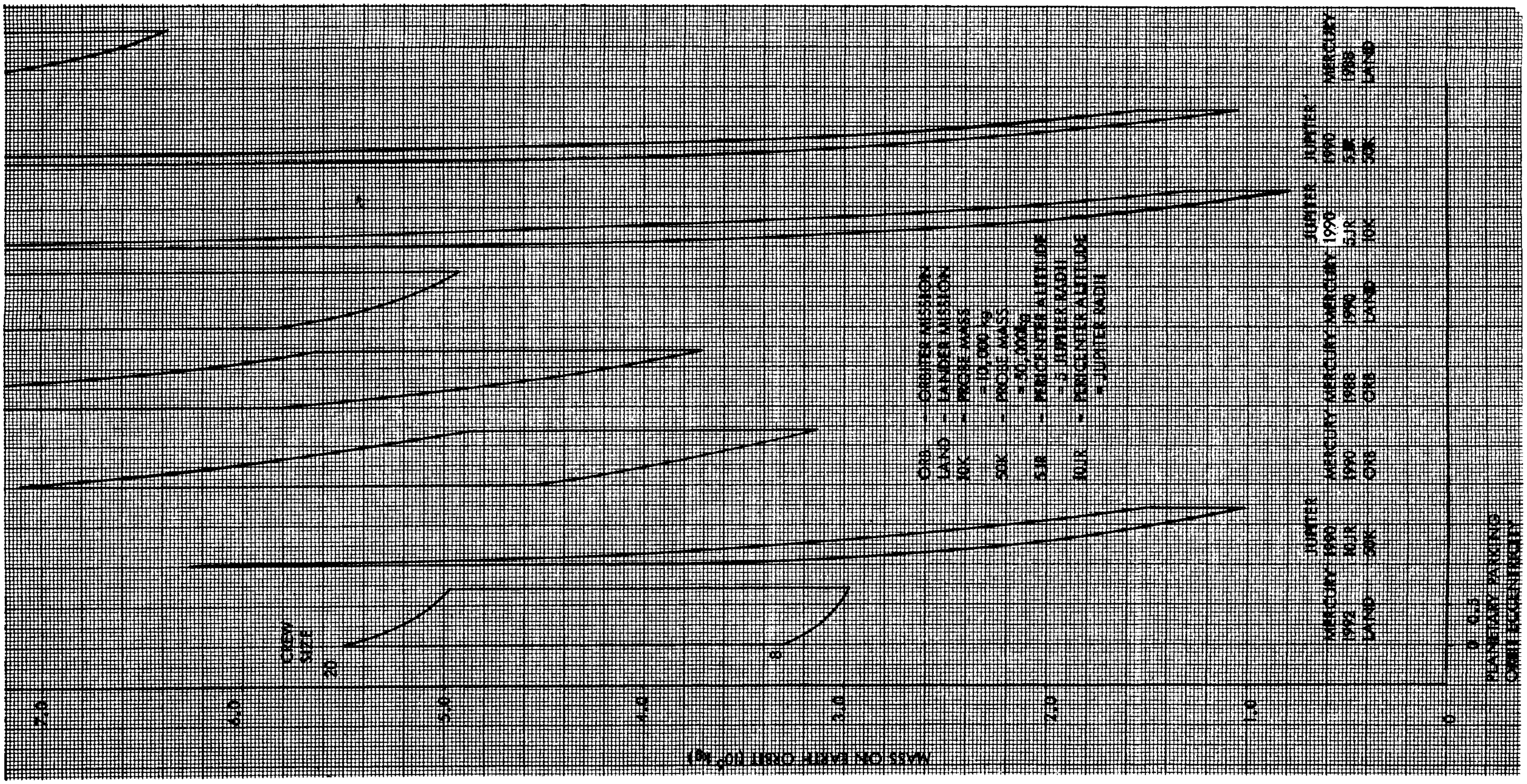

Figure 92. Mass in Earth Orbit Requirements for Nuclear Propulsion Modules (0 - 7,000,000-kg)

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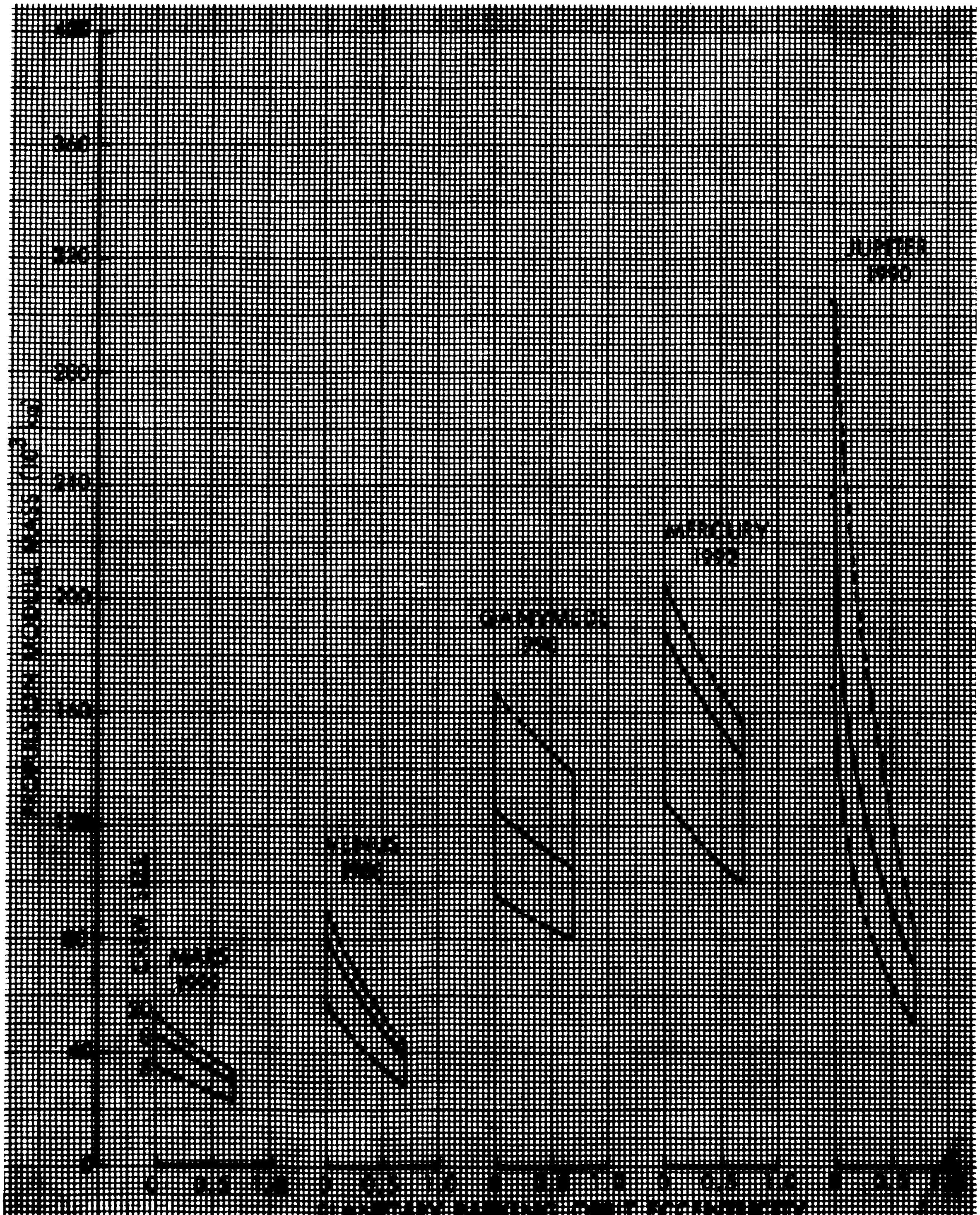

Figure 93. POE Propulsion Module Mass Requirements (20-Man Module with 8-Man Crew)

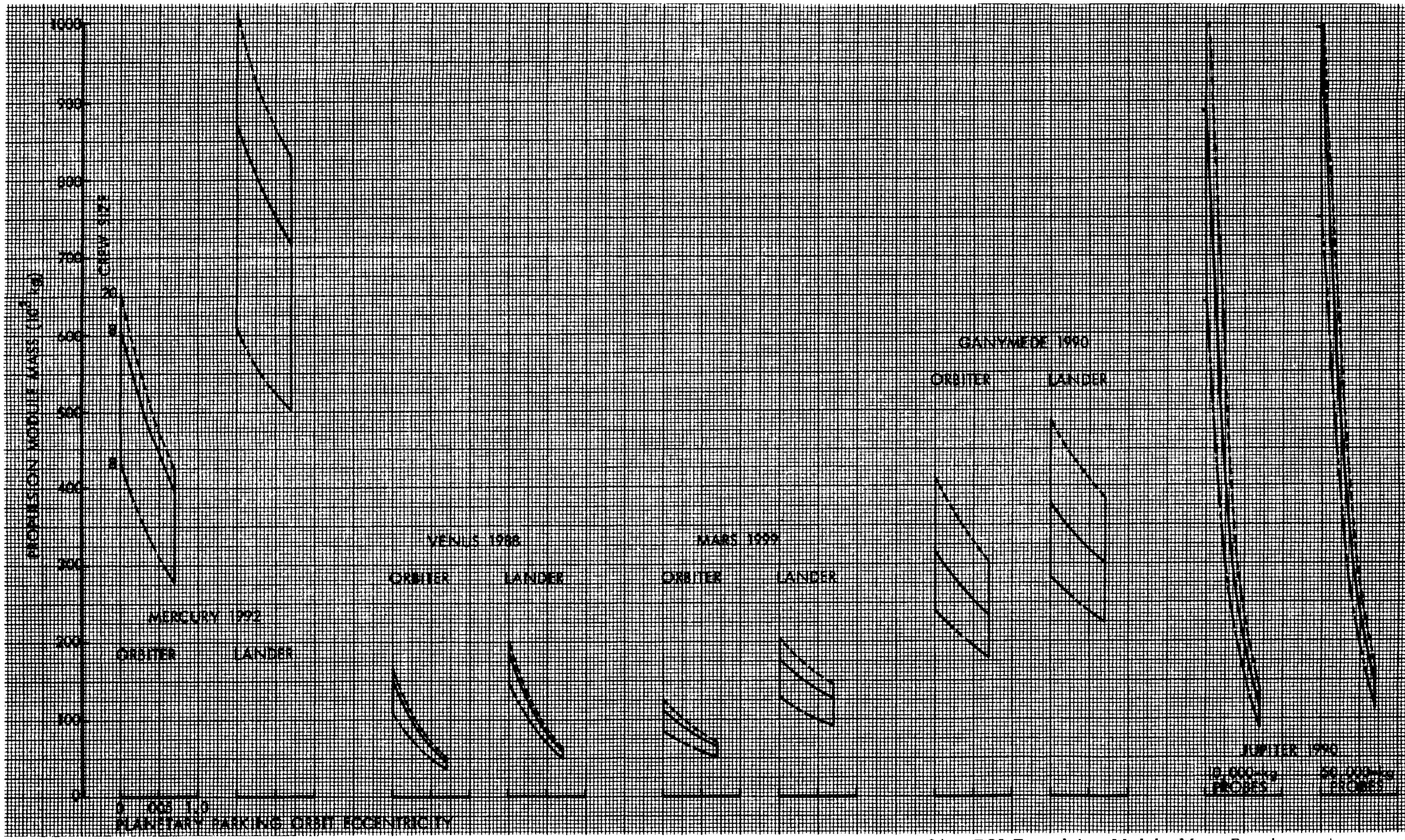

Figure 94. POI Propulsion Module Mass Requirements (20-Man Module with 8-Man Crew)

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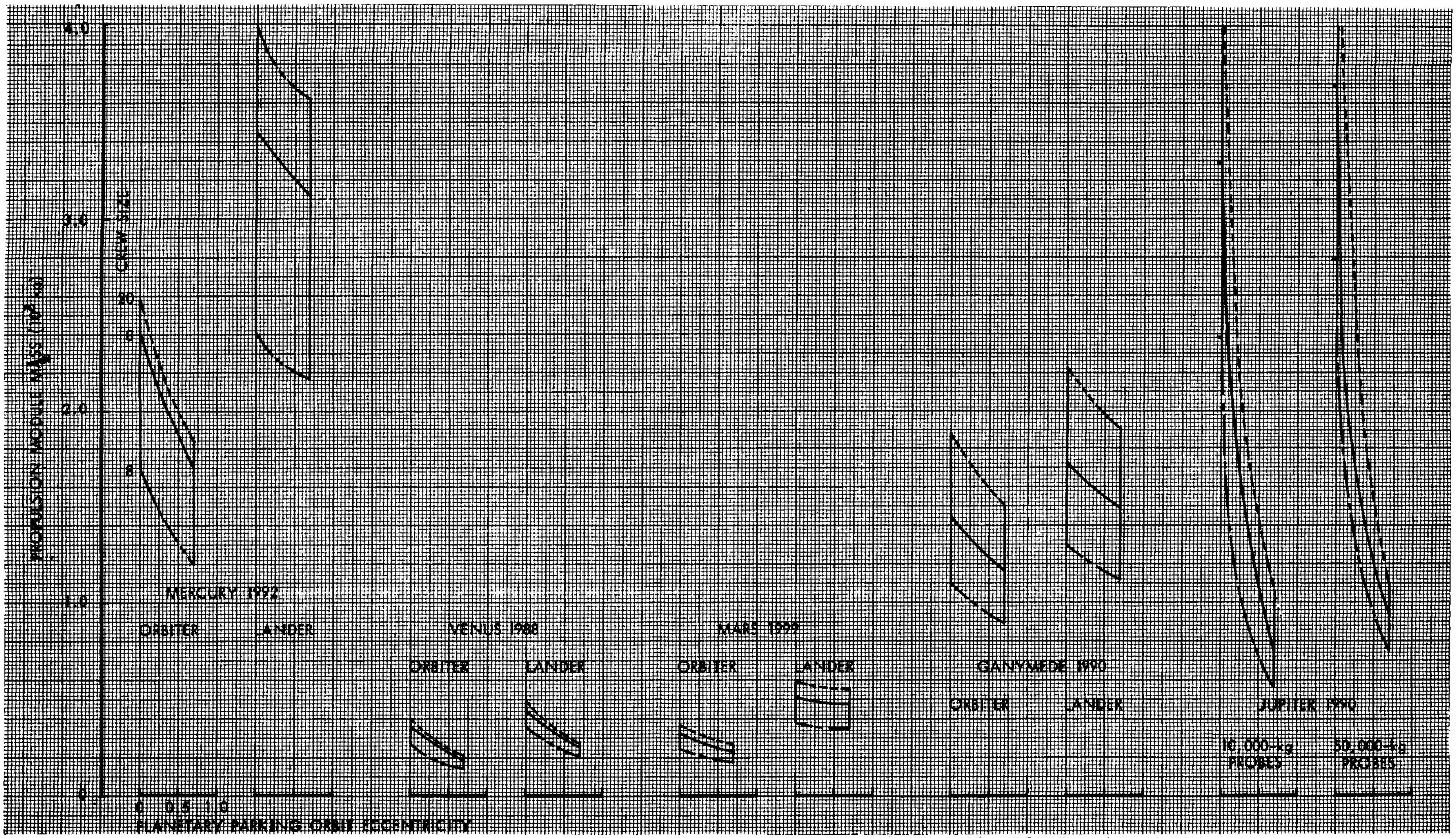

Figure 95. EOE Propulsion Module Mass Requirements (20-Man Module with 8-Man Crew)

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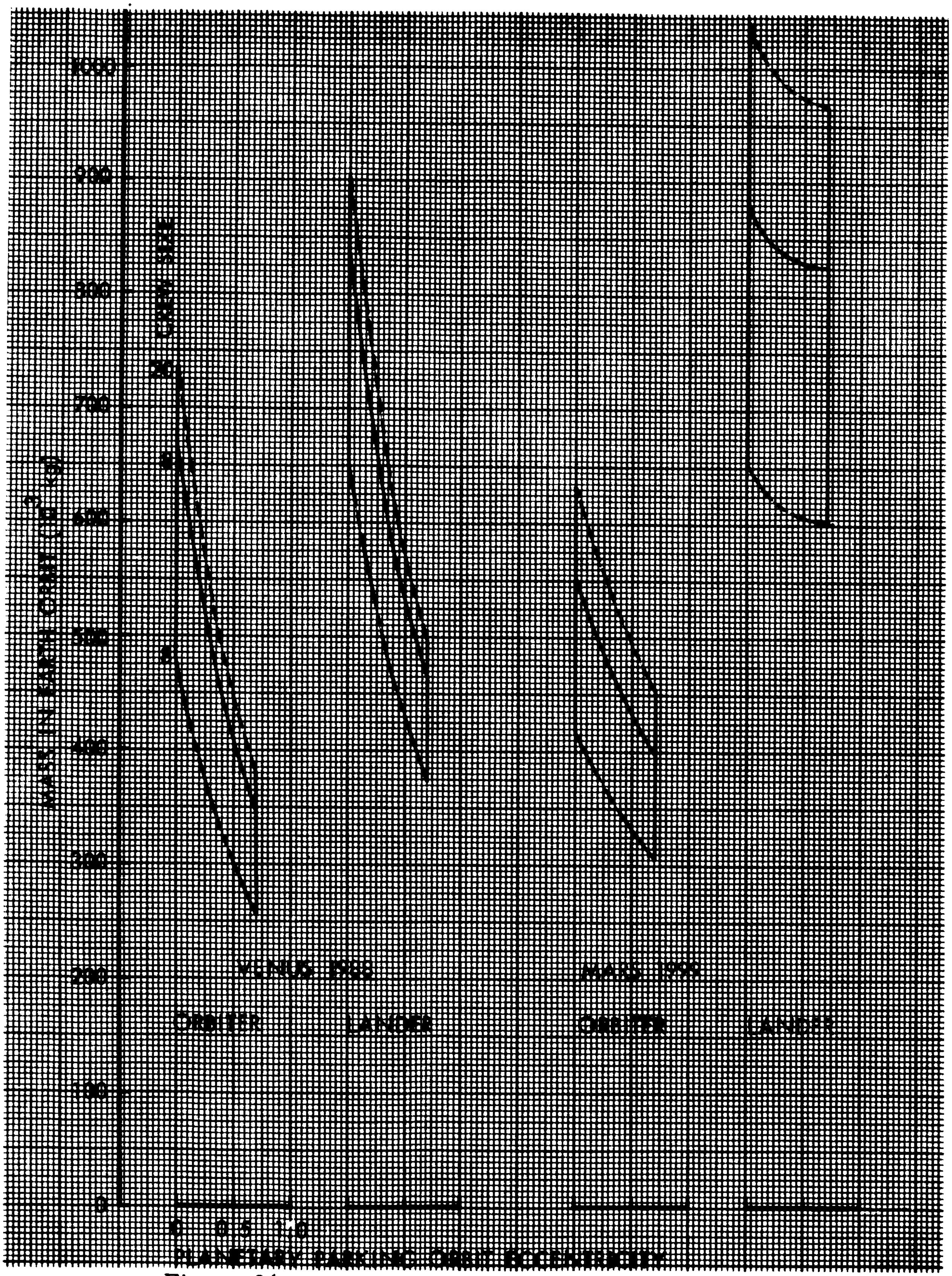

Figure 96. Venus and Mars Mass in Earth Orbit (20-Man Module with 8-Man Crew)

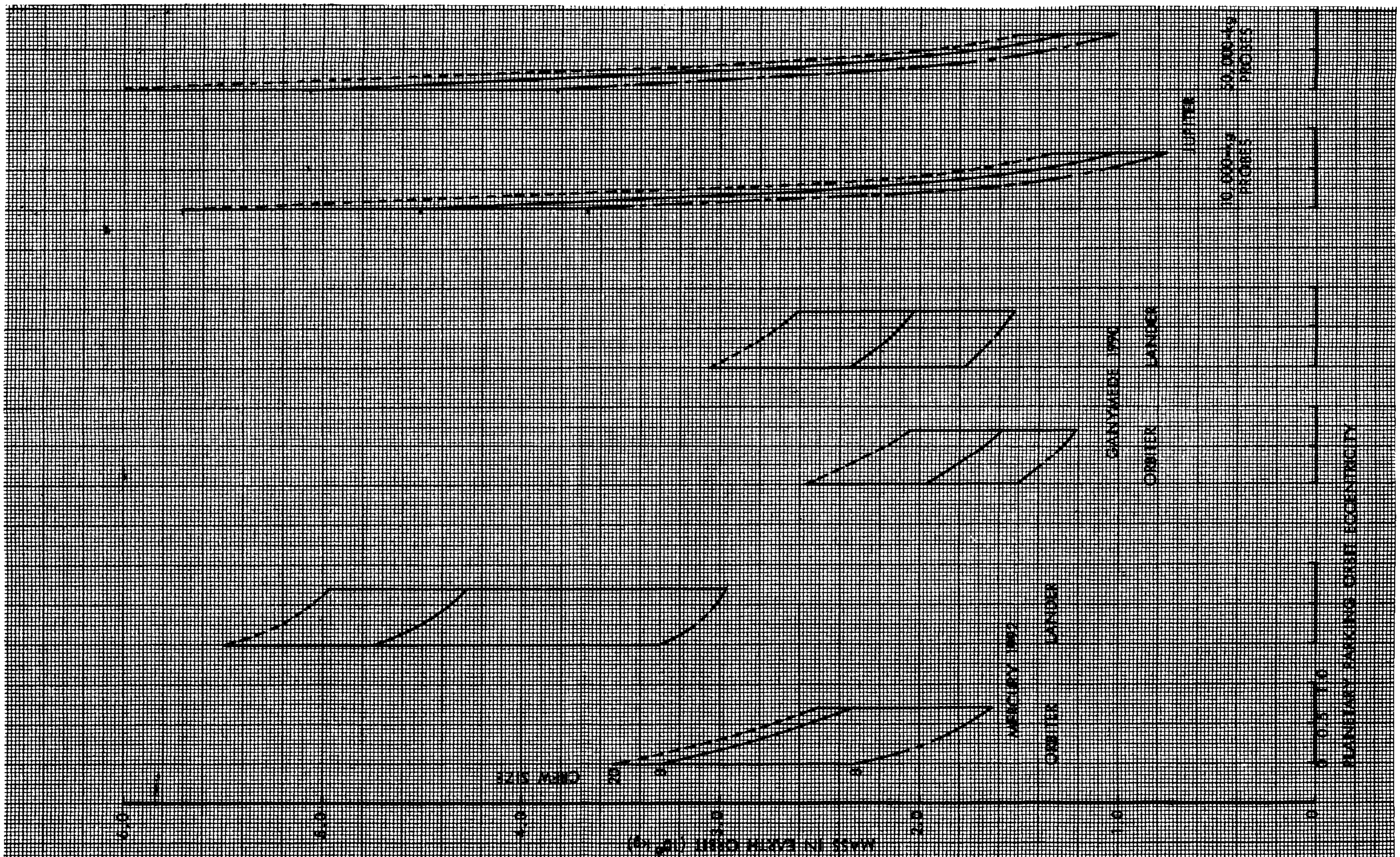

Figure 97. Mercury, Ganymede and Jupiter Mass in Earth Orbit (20-Man Module with 8-Man Crew)

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limiting the mission opportunities and the crew sizes. Regions of common requirements are not as apparent, however, when elliptical planetary orbits are considered because of the extreme variations in the propulsion module mass requirements. By examination of the various propulsion modules mass requirements shown previously in this section it is possible, however, to identify several propulsion module sizes that seem appropriate. These sizes are summarized in Table 19 for each of the propulsion system combinations considered. To interpret the format of the table consider the NNN system. The first option is to develop two modules of 75,000 kg and 300,000 kg; the second option is to develop three modules of 75,000 kg, 300,000 kg, 1,200,000 kg; and so forth.

Though the discussion of common propulsion module sizes could terminate with the matrix of Table 19, it seems desirable to place these results in some prospective. One way of accomplishing this is to contemplate various criteria which might, at some time in the future, be applied to the module selection process. At this particular time, of course, the fact that a specific module size is identified, as in Table 19, is of less importance than the conclusion that a module of about that size seems appropriate even if apparently contradictory criteria were to be imposed. The criteria which were considered are shown in Table 20. These criteria were not employed explicitly in the selection of the module sizes although such factors were considered implicitly simply to reduce the number of module sizes to a reasonable value.

Table 19. Candidate Common Propulsion Modules

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

Table 19. Candidate Common Propulsion Modules (Continued)

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

N = nuclear propulsion
C = chemical propulsion (cryogenic or space storable)
F = flyby mission

Table 20. Propulsion Module Selection Criteria

1. Compatible with Saturn V Launch Vehicle
2. No Impact by an Aerobraker versus Retrobraker Decision (Mars and Venus only)
3. Either Chemical or Nuclear Upper (i.e., POE and/or POI) Stages (Mars and Venus only)
4. Minimize the Number of Propulsion Modules Developed
5. Minimize the Number of Propulsion Modules in the Total System

Criterion 1.

The maximum payload in Earth orbit of uprated Saturn V launch vehicles which are currently being considered is approximately 220,000 kg (480,000 lbs). This limit was not imposed as a basic study constraint in order to permit complete freedom in the selection of common modules and because the mass in Earth orbit is generating so large that an excessive number of rendezvous would be required. This criterion, by itself, did not define common propulsion modules, but it did assist in the selection of common modules when alternatives were available.

Criterion 2

The aerobraker versus retrobraker criterion applies only to the Earth orbit escape propulsion module for Mars and Venus missions. The objective of this approach to establishing common module requirements would be to develop one or more propulsion modules which are compatible with the requirements of both mission modes. Such an approach would not require a decision of which mission mode would ultimately be employed prior to the development of the Earth orbit escape propulsion module.

The Earth orbit escape propulsion module mass requirements for the Mars and Venus aerobraker and retrobraker missions which were investigated varied from 120,000 kg to 380,000 kg for a crew size of eight men and to 600,000 kg for a crew size of twenty men. The above variations in the mass requirements are based on the use of nuclear upper stages for the retrobraker missions. The upper bounds are based on the requirements for circular orbits. Various alternatives exist for selecting a common propulsion module which will satisfy the earth orbit escape requirements of both the aerobraker and the retrobraker missions. A 600,000-kg module could be developed which would satisfy the requirements of all Mars and Venus missions for all crew sizes. This same module would satisfy the planetary orbit insertion requirements for Mercury orbiter and lander missions and the planetary orbit insertion requirements for Jupiter orbiter missions. A 600,000-kg module could also be used in multiples for earth orbit escape for Ganymede and Mercury orbiter and lander missions.

An alternate approach would be to develop a propulsion module with a mass of approximately 300,000 kg. The module could be used singularly for earth orbit escape for all aerobraker missions (orbiter and lander), all Mars orbiter missions, some Mars landing missions, and all Venus missions (orbiter and lander). A single 300,000-kg module could also be used for planetary orbit insertion for Ganymede orbiter and landing missions, Jupiter orbiter missions, and Mercury orbiter missions provided high eccentricity parking orbits are used. Two of the modules would satisfy the planetary orbit insertion requirements for Mercury lander missions and Jupiter orbiter missions. The Jupiter orbiter missions which fall within this range of capability need not be considered, however, since Ganymede missions can be performed with lesser total requirements.

A third alternative would be the development of a module with a mass of 150,000 to 200,000 kg which could be used either singularly, or in multiples, to satisfy the earth orbit escape requirements of all Mars and Venus missions. It would not be necessary to cluster more than four of these modules. A lesser number would suffice depending on the opportunity, crew size, and orbit eccentricity. A 150,000-kg module would also be compatible with the planetary orbit escape requirements of Mercury, Jupiter, and Ganymede missions. Two of the modules could be used for planetary orbit insertion for Ganymede and Jupiter missions. A 150,000-kg module would also satisfy the earth orbit escape requirements of all flyby missions considered (Vesta, Ceres, and Jupiter) except the Ceres missions in the more unfavorable launch opportunities. Of the alternatives considered, only the last one is compatible with the capabilities of the uprated Saturn V launch vehicles.

Criterion 3

The chemical versus nuclear upper stage evaluation criterion applies to the Earth orbit escape module for Mars and Venus retrobraker missions. The nuclear Earth orbit escape propulsion module requirements for the Mars missions investigated vary from 130,000 kg to 520,000 kg where the upper bound is defined by the 1999 Mars lander mission with chemical upper stages, a crew of eight men, and a circular parking orbit. The upper bound can be reduced to 420,000 kg when elliptical parking orbits are considered. The upper bound would increase to 700,000 and 900,000 kg for elliptical and circular orbits respectively if crew sizes of twenty men were considered. The propulsion module mass requirements will be comparable to the range of requirements which satisfy the aerobraker versus retrobraker criterion if the lander missions which have the higher requirements are restricted to crew sizes of less than twenty men and/or elliptical planetary parking orbits. Therefore, the modules which satisfy the aerobraker versus retrobraker criterion (i.e., the 150,000 kg, 300,000 kg and 300,000 kg modules) can also satisfy the chemical versus nuclear upper stage criterion.

Criterion 4

Minimizing the number of propulsion modules developed would be desirable from the standpoint of minimizing the number of development programs which must be undertaken. The minimization of the number of modules developed can best be accomplished by developing a limited number of modules which could be used either singularly or in multiples to accomplish all maneuvers for the majority of the mission objectives and mission opportunities. This is basically the criterion which was utilized in the investigation of the common module requirements for missions which employ circular planetary parking orbits. Applying the same criterion to the elliptical orbit missions could result in the same common module requirements but with increased mission capability. The increased capability could be an increase in crew size, probe mass, stay time, or combinations thereof. The increased capability would be achieved by utilizing eccentric planetary parking orbits during those missions which have the greater requirements. An alternate approach would be to develop smaller common propulsion modules (i.e., less than the 75,000 and 300,000 kg modules considered during the investigations of circular orbits) on the assumption that only eccentric planetary parking orbits would be used during those missions which impose the maximum requirements.

Criterion 5

Minimizing the number of propulsion modules in the total system could be desirable on the basis of decreased system complexity and increased system reliability. Minimizing the number of propulsion modules in the total system does not necessarily imply minimizing the total number of modules developed. For example, one module could be developed for performing each of the major propulsive maneuvers. This would result in only three propulsion modules in the total system for any mission but it would also require the development of at least three modules.

As an alternative, one module could be developed which could be used singularly for planetary orbit escape and in pairs for planetary orbit insertion. A second module could then be developed for Earth orbit escape. This would require the development of only two modules but would require four modules in the total system.

One other possibility which exists is the utilization of a single restartable module which could be used for multiple maneuvers. In the limit, this would result in only one module in the total system and one module developed. Such an approach, i.e., only one module, would not be feasible for the entire family of missions considered. The utilization of a restartable module does appear attractive, however, for performing both the planetary orbit escape and planetary orbit insertion maneuvers. One of the basic problems encountered when selecting common propulsion modules for the circular orbit missions is the selection of the small planetary orbit escape module. This module must be grossly overdesigned for the planetary orbit escape maneuvers in order to be capable of performing the planetary orbit insertion maneuver. This difficulty can be avoided by utilizing a single restartable propulsion module.

Some insight regarding the desirable mass of a restartable propulsion module can be obtained by considering the sum of the optimized planetary orbit insertion and escape modules. (The sum of these modules is not, of course, the module mass which would be achieved by optimizing the mass of a restartable module.) For crew sizes of eight men, the sum of the optimized planetary orbit insertion and planetary orbit secape propulsion modules varies from 75,000 kg for the Mars orbiter missions to approximately 200,000 kg for lander missions with crew sizes of eight men. It was shown previously that a module with a mass of 150,000 to 200,000 kg could satisfy the requirements of some mission phase for all mission objectives for the opportunities being considered. If the module had a restart capability, a single module could be used for both planetary orbit insertion and escape for Mars orbiter and lander missions with a crew size of eight men and for Venus missions with either larger crew sizes or larger probe complements. Multiples of the module could be used to perform both the planetary orbit insertion and escape maneuvers for all other mission objectives with the exception, perhaps, of the Mercury lander missions. As discussed previously, the same modules could be used without a restart capability for earth orbit escape for Mars and Venus missions.

A second propulsion module would be required for planetary orbit insertion for Mercury missions and for earth orbit escape for Jupiter or Ganymede missions if a single intermediate size restartable module were developed. The 600,000 kg propulsion module discussed previously appears to be a desirable module for performing the above maneuvers. The module could be used singularly for the Mercury planet orbit escape maneuver and multiple modules could be used for earth orbit escape for Ganymede and Mercury orbiter and lander missions and for Jupiter orbiter missions.

The effects of using single stage with a restart capability for performing both the planetary orbit insertion and the escape maneuvers were investigated for two Mars opportunities (1988 and 1999 oppositions) and one Venus opportunity (1988 inferior conjunction). The use of a restartable module has operational advantages since flexibility in the distribution of the incremental velocity requirements could be achieved. If single stages were used for each maneuver, even more propellant would be required for launch window purposes. By using restartable stages, any propellant which is not used due to a near nominal transplanet mission profile could be used to provide additional mission flexibility, i.e., longer stay time, shortened trans-Earth trip time, or more launch window capability at the planet. The use of a restartable stage also has disadvantages, e.g., restart of a nuclear engine, effective use of cool-down propellant and rendezvous near a hot reactor. These are probably not insurmountable if justification is sufficient. The above problems are avoided if chemical propulsion modules are employed.

The effects of using a restartable stage for performing the planetary orbit insertion and escape maneuvers for the three missions considered are shown in Figures 98, 99, and 100, respectively. The assumed propulsion module had a stage mass of 16,200 kg and a total propellant capacity of 100,000 kg. The engine was assumed to have a thrust of 889,640 newtons (200,000 lbs) and a mass of 15,000 kg. The stage mass and engine mass are based on the scaling equations defined in the Weight Scaling Equations section of this Appendix. The resultant total module mass (including insulation, meteoroid protection, and the interstage) was near the maximum capability of the Saturn V with uprated engines and a lengthened first stage (153,000 kg).

The resultant mass requirements are shown in Figure 98 for the 1988 Mars opportunity. The lower solid lines represent the sum of the optimally sized planetary orbit insertion and escape propulsion modules. The corresponding optimally sized Earth orbit escape propulsion module mass is shown by the upper solid curves. The effects of using the restartable module are shown by the dashed curves. The lower curves are for the planetary orbit insertion and escape maneuvers while the upper curves are for the Earth orbit escape maneuver. The numbers to the right of the curves indicate the number of propulsion modules required to accomplish the maneuvers. In the case of the Earth orbit escape curves, the first numeral defines the number of upper stages and the second the number of Earth orbit escape stages.

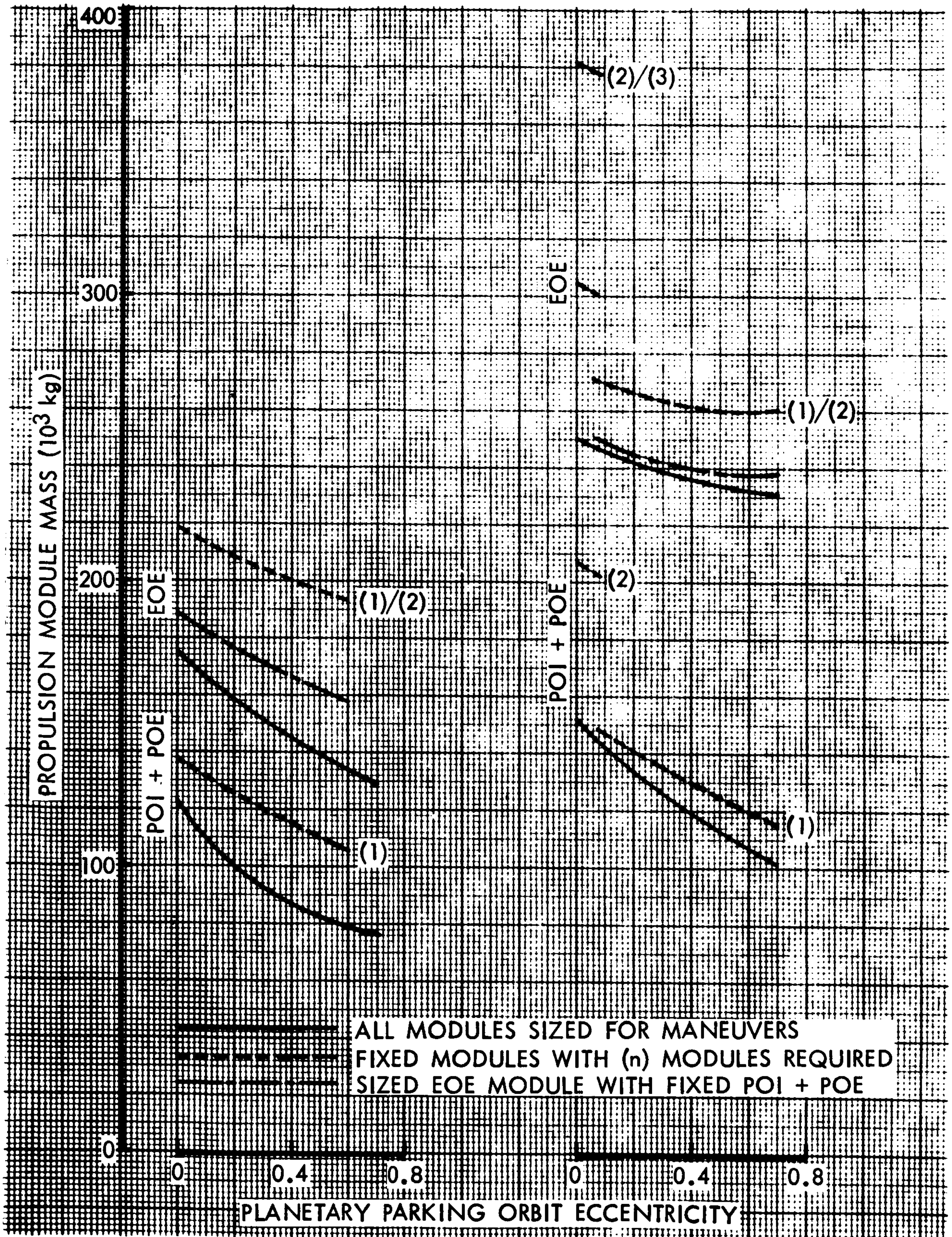

Figure 98. Fixed Propulsion Module Mass Requirements (1988 Mars Opposition)

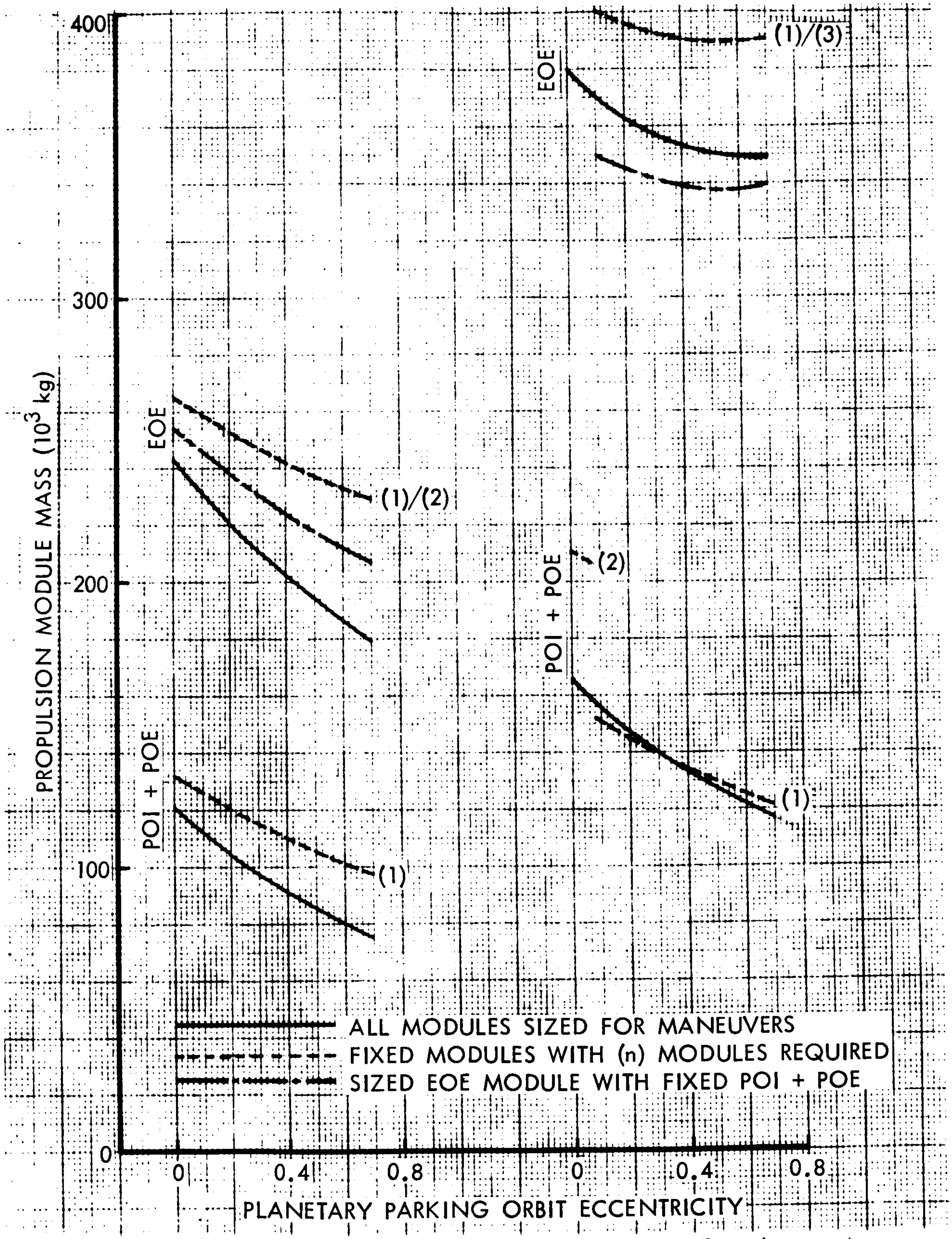

Figure 99. Fixed Propulsion Module Mass Requirements (1999 Mars Opposition)

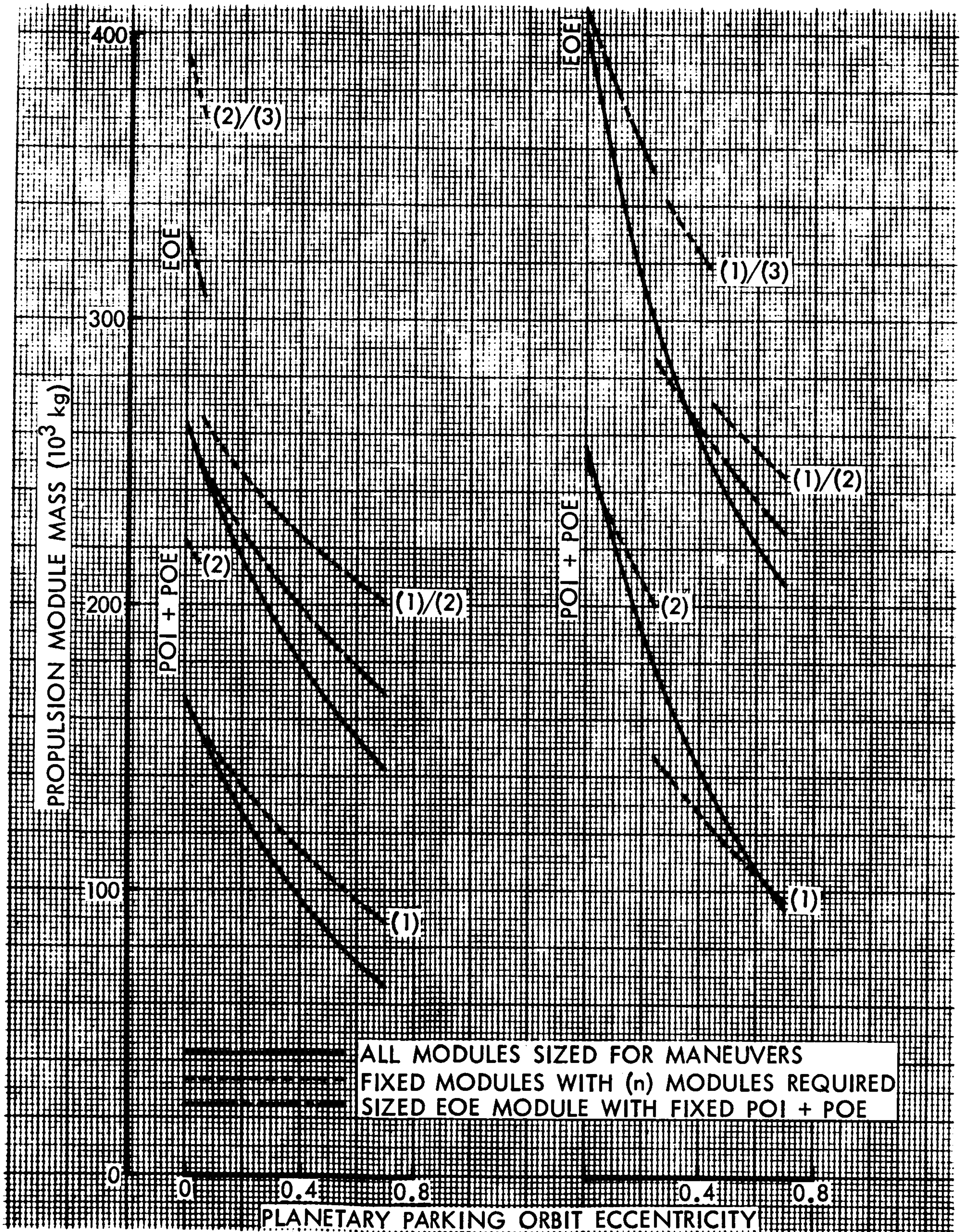

Figure 100. Fixed Propulsion Module Mass Requirements (1988 Venus Inferior Conjunction)

In the case of the 1988 Mars orbiter missions, a single restartable module can be used in the planetary vicinity and two modules for Earth orbit escape for all accentricities. For the lander missions, two modules are required for the planetary maneuvers at the lower eccentricities. Earth orbit escape can be performed using either two or, at the lower eccentricities, three modules.

Similar results were obtained (Figure 99) when the 1999 Mars opportunity was considered. However, three propulsion modules were required for Earth orbit escape for the lander missions even at the highest ecentricity considered.

The requirements for the Venus mission considered are shown in Figure 100. Again, no more than two modules were required in the planetary vicinity. As many as four modules were required for Earth orbit escape when the larger (50,000 kg) probe mass was considered.

CONCLUSIONS

The feasibility of developing common modules which are compatible with the requirements of both the nearer-term and the advanced manned planetary missions has been demonstrated. The specific modules which are ultimately developed will be dependent upon the constraints imposed during the selection of the missions and the modules.

It has been shown that it would be possible to develop a single Earth reentry module which could be used for the majority of the mission objectives and mission modes considered during this study. The use of the Apollo configuration would necessitate the elimination of the direct Mars missions and the Ceres missions unless a pre-entry retrobraking maneuver is utilized or the mission incremental velocity requirements are increased.

A single mission module could be utilized which is off-loaded in crew and consumables to satisfy particular mission requirements. An alternate, and perhaps more attractive approach, would be to utilize a modular concept wherein floors are added as required by the missions.

Finally, the feasibility of developing common propulsion modules has been demonstrated. Of particular significance is the conclusion that, if properly selected, the propulsion modules which will be required for the nearer-term missions can be utilized for the advanced missions. Therefore, prior to the definition of the design requirements of the next family of propulsion modules, the entire family of potential missions should be considered. In this manner, the modules which are developed for the nearer-term missions will also be capable of satisfying the requirements of more advanced missions.

The various propulsion module combinations which are considered to be particularly attractive have been shown earlier (Table 19). This section can be most appropriately concluded by summarizing the applicability of the various module sizes to the family of missions considered in this study. This summary is contained in Table 21 for both nuclear and chemical propulsion systems.

Several interesting conclusions are apparent from the table, e.g., (1) A 75,000 kg nuclear module is appropriate for all missions except Ganymede; (2) A 150,000 kg 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 used; (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 kg will be necessary; and (5) Missions to Mars and Venus can be carried out with chemical propulsion modules which do not exceed 300,000 kg in size.

Any conclusions concerning propulsion module mass requirements 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 time period. Unquestionably the values quoted herein are subject to further refinement. Nevertheless a fundamental conclusion has been reached, namely, that feasibility of the concept of commonality has been demonstrated.

Table 21. 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

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8. A Study of Technological Requirements Common to a Family of Manned Planetary Missions, First Interim Report, SID 66-1883 (16 December 1966).

ERRATA

Technological Requirements Common to Manned Planetary Missions

Final Report - Appendix D, SD67-621-5

1. Page 57 - last term inside brackets of equation which defines A_{mm} for elliptical bulkhead geometry should read as follows:

$$2\ AR\ \sqrt{AR^2-1} \quad -1$$

2. Page 88 - terms of left side of last two equations should read as follows:

$$W_{OPEN} - \left[W_{(H_2O)_{REC}} + O_2 \right]$$

$$W_{(H_2O)_{REC}} - \left[W_{(H_2O)_{REC}} + O_2 \right]$$

3. Page 139 - third sentence of second paragraph refers to Tables 4 through 19 of Appendix A.

4. Page 142 - Table 15: The mass in Earth orbit should equal the sum of the two preceeding weights.

5. Page 165 - last two sentences of second paragraph should read as follows:

   "For orbiter missions, the decreases in the mass requirements are fifty percent for planetary orbit insertion and forty percent for Earth orbit escape. The corresponding decreases in the mass requirements for lander missions are forty percent for planetary orbit insertion and twenty percent for Earth orbit escape."

6. Page 203 - Table 19: Replace column heading "N/C" by "600".

7. Page 204 - Table 19 (continued): The entries in the table should be on three separate lines.

8. Page 206 - last sentence of first paragraph: The value quoted for the third module weight should be 600,000 kg.

9. Page 209-211 (Figures 98-101): The left portion of each figure applies to orbiter missions; the right portion applies to lander missions.

10. Page 213 - first sentence of second paragraph: Delete the exclusion of Ganymede.

11. Page 214 - Table 21: Enter "F" under 100,000 kg chemical propulsion module for Vesta, Ceres, Jupiter flybys; for nuclear system modules mass for Vesta flyby is 75,000 kg-not 1,200,000 kg.