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STUDY OF TECHNOLOGY REQUIREMENTS FOR ATMOSPHERE BRAKING TO ORBIT ABOUT MARS AND VENUS

Volume I - Summary

by E. M. Repic

Prepared by
NORTH AMERICAN ROCKWELL CORPORATION
Downey, Calif.
for Ames Research Center

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FOREWORD

This volume presents a condensed summary of the study. It is being submitted in accordance with Paragraph 2.6 of Specification A-12241 contained in Contract NAS2-4135, Study of Technology Requirements for Atmosphere Braking to Orbit About Mars and Venus, which was issued by the Mission Analysis Division, National Aeronautics and Space Administration. The data were generated between January and October 1967 and are presented in three volumes:

Volume I - Summary (SD67-994-1)

Volume II - Technical Analyses (SD 67-994-2)

Volume III - Appendices (SD 67-994-3)

Volume IV - Final Briefing (SD 67-994-4)

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1.0 INTRODUCTION

Manned missions to Mars and Venus, although not a stated national goal, are a logical extension of the current space program. Overall technical feasibility of such missions appears within the current and projected near-term state of the art. However, the eventual accomplishment is constrained by certain key issues which must be resolved in the early planning phases. One key decision which must be made early in the program involves the choice of the mode to effect planet capture (i.e., retrobraking or aerobraking). Each mode obviously must be studied intensively to develop the requisite background data for a rational selection at the earliest date.

Previous studies have shown that atmosphere braking to orbit about Mars and Venus (shown schematically in Figure 1) offers the potential of significantly lower gross system weights in Earth orbit as compared to retropropulsive capture. Weight comparisons for aerobrakers using space-storable propellants for planet-orbit departure and all nuclear retrobrakers are shown in Figures 2 and 3 for Mars and Venus mission opportunities from 1980 to 2000. Both orbiter and lander missions which are shown

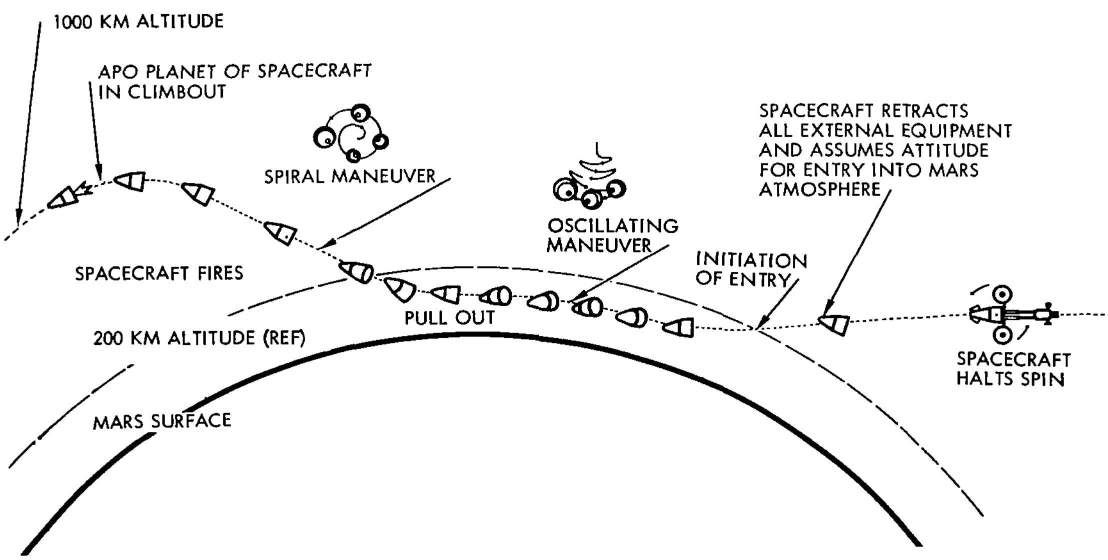


Figure 1. Mars Aerobraking Maneuver

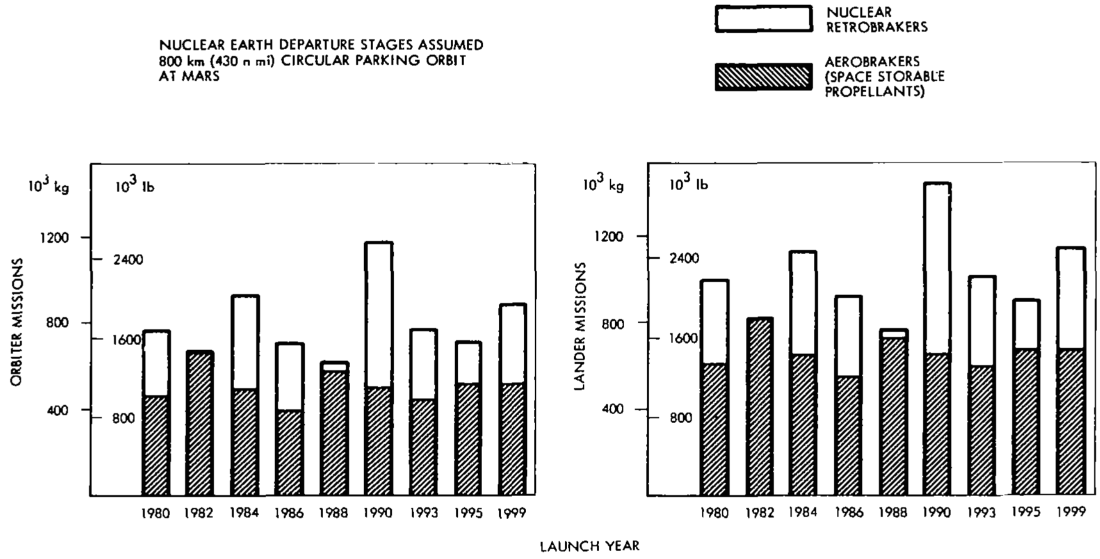


Figure 2. Initial Mass in Earth Orbit—Mars Missions

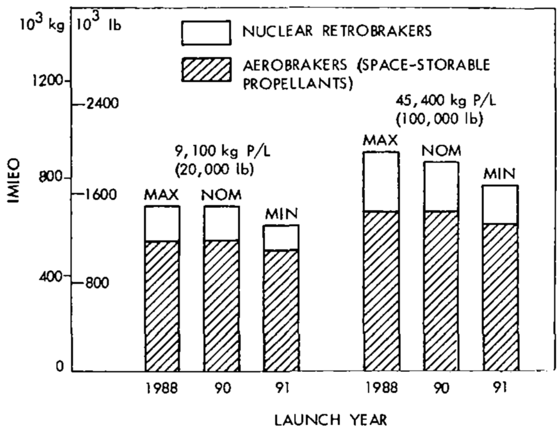


Figure 3. Initial Mass in Earth Orbit—Venus Missions

were synthesized using ΔV requirements determined by Deerwester(1) for Venus-swing-by missions for Mars, while the Venus data were obtained from the recently completed Commonality study(2). The three opportunities for Venus represent the minimum, nominal, and maximum energy requirements for Venus missions in the 1980 to 2000 period.

The use of elliptical parking orbits at Mars and Venus result in significant weight reductions for both aerobrakers and retrobrakers. For example, results obtained for Venus orbiter missions showed that the nuclear retrobraker was approximately 50 percent heavier for circular orbits and 20 percent heavier for elliptical orbits than the aerobraker utilizing space-storable propellants

(1) Deerwester, J.M. and S.M. D'Haem. "Systematic Comparison of Venus Swingby Mode With Standard Mode of Mars Round Trips." Journ. of Spacecraft and Rockets. Vol. 4 No. 7 (July 1967), pp. 904-912.

(2) Codik, A. and R.D. Meston, "Final Report - Technological Requirements Common to Manned Planetary Missions." North American Rockwell Corp., Space Division, SD 67-621 (Dec. 1967).

for all possible mission years. The aerobraking maneuver is potentially more complex than the retrobraking maneuver, however, and the spacecraft designs contemplated are more than an order of magnitude larger in mass and volume than the largest entry vehicles yet considered. Potential technology problems in the fields of gasdynamics, thermodynamics, structures and materials, and guidance and control could possibly negate some of the indicated weight savings; additionally, no serious consideration has previously been given to the problems of system test and qualification.

The primary objective of this study was to conduct a detailed investigation of integrated aerobraking spacecraft and the associated technology implications so that both the advantages and disadvantages of this mode could be evaluated. Included were parametric and conceptual design studies to define the spacecraft weights and their sensitivity to variations in the environmental models assumed, crew size, mission profile, propellants, etc., as well as internal packaging arrangements. A secondary objective was to analyze the requirements for system simulation, testing, and qualification.

The study was conducted in accord with the logic diagram shown in Figure 4. Initially, the aerobraking vehicle requirements, including entry corridors, heating, loads, and packaging, were determined parametrically for a wide range of mission and vehicle parameters. With these requirements established, a

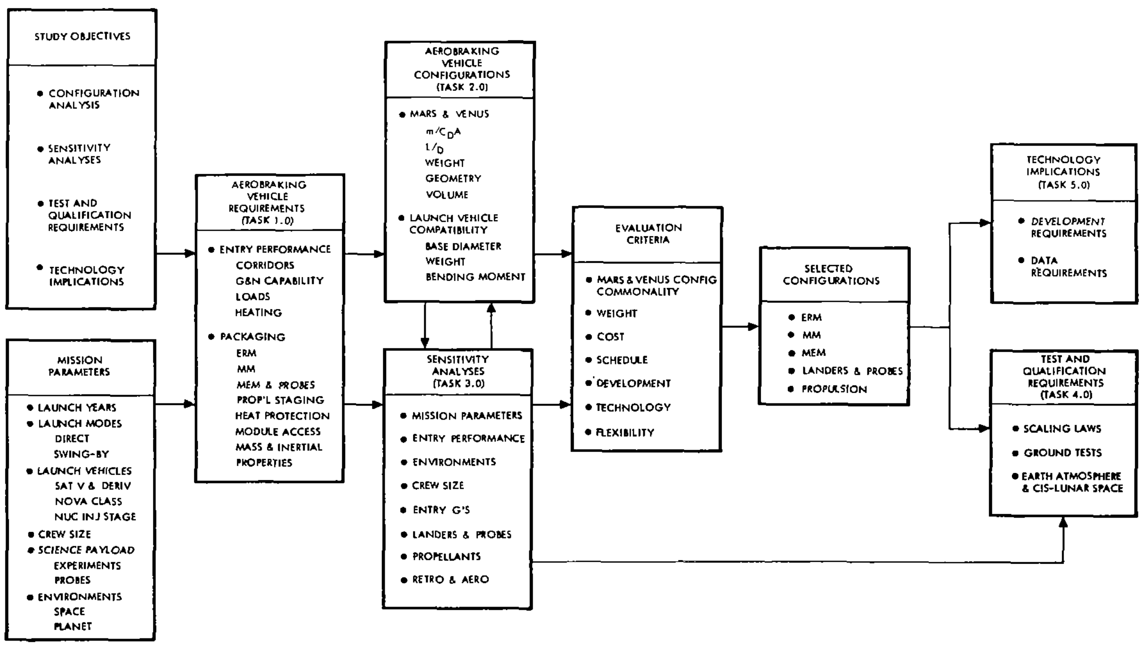


Figure 4. Study Logic Diagram

configuration analysis was initiated. Concurrently, sensitivity analyses were conducted to establish the effects of variations in mission parameters, propellant selection, and module selection on the vehicle designs. Over 30 detailed designs were generated and evaluated on comparative performance, size, weight, and technology requirements. In the final phase, the four most attractive configurations were selected, and the technology implications associated with each were delineated. In addition, preliminary consideration was given to test and qualification requirements. The pertinent results are summarized in this volume; more detailed analyses are presented in the volumes entitled, Technical Analyses and Appendices, SD 67-994-2 and 67-994-3.

2.0 STUDY GUIDELINES AND SCOPE

The candidate configurations were derived from a family of symmetrical, biconic shapes trimmed to operate at lift-to-drag ratios of from 0.5 to 1.0. In addition, a blunt, Apollo-type configuration was analyzed to determine its applicability to the aerobraking mission. The basic biconic configuration is shown in Figure 5.

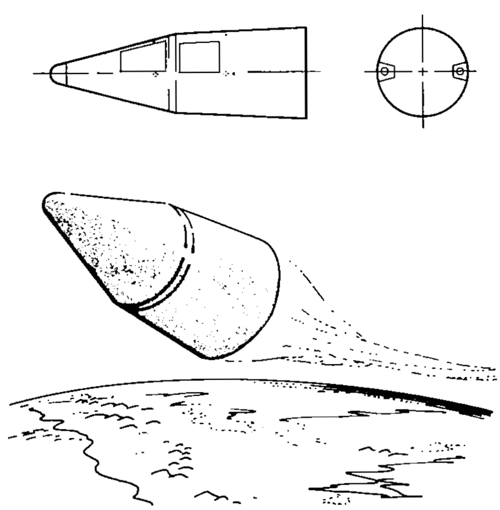


Figure 5. Flared-Cone Concept

Entry trajectories were determined for the entry velocities and atmosphere models specified in Table 1. Allowable entry corridors were determined assuming a 10-km (32,810-ft) minimum-altitude constraint at Mars and 5- and 10-g undershoot trajectories at Venus. An approach corridor guidance-accuracy capability of 20 km (10 n. mi.)

Table 1. Entry Parameters

Planet	Entry Velocity	Atmosphere Models
Mars	10 km/sec (32, 810 ft/sec)	JPL VM-7[3], VM-8, MSC-3[4]
Venus	12 km/sec (39, 370 ft/sec)	MSFC LDM, MDM, UDM[4]

[3] Martin, C. D., Physical Characteristics and Atmospheric Data for Mars, NAA/SID 65-1684 (Nov. 1965).

[4] Venus and Mars Nominal Natural Environment for Advanced Manned Planetary Mission Studies. NASA SP-3016 (1967).

was assumed. Flow fields and convective and radiative heat-transfer rates were determined; nonadiabatic and self-absorption effects were considered. Thermal protection requirements were established using Avcoat 5026 (Apollo material) as the reference ablator.

The entry corridor and heating data were then employed to identify the allowable range of values for the ballistic parameter (m/C_DA) and L/D for each atmosphere. Packaging studies were conducted concurrently using these m/C_DA and L/D requirements to establish initial geometric relationships and weight distribution requirements for a 10 m (33 ft) base diameter configuration (to be compatible with the current Saturn V). A multiloop iteration then was performed to establish configurations which were acceptable from both aerodynamic and packaging considerations. Structural analyses, including provisions for radiation and meteoroid protection, were conducted in support of the packaging studies.

Mission ground rules specified the use of an eight-man crew for both Mars and Venus missions, having four men available at Mars to land an excursion module. The mission module considered for the interplanetary journey was based on the designs developed for the manned Mars fly-by mission(5). The Mars excursion module was obtained from a concurrent study conducted for NASA/MSC(6); the Earth-return module, and other major subsystems were obtained from previously completed NASA studies(7), (8). In addition to the basic modules and subsystems required for the mission, a 9100 kg (20,000 lb) probe complement was included in each design. The trip times, which size the consumables and meteoroid protection and establish subsystem requirements, are shown in Table 2. Internal propulsion stages employed for planet departure were sized for the orbits and hyperbolic excess velocities shown in Figure 6 and were equal to or exceeded the requirements for 80 percent of the swing-by mission opportunities in the metonic cycle.

Table 2. Mission Trip Times (Days)

Planet	Outbound	Stay	Homebound
Mars	160	30	240
Venus	80	30	270

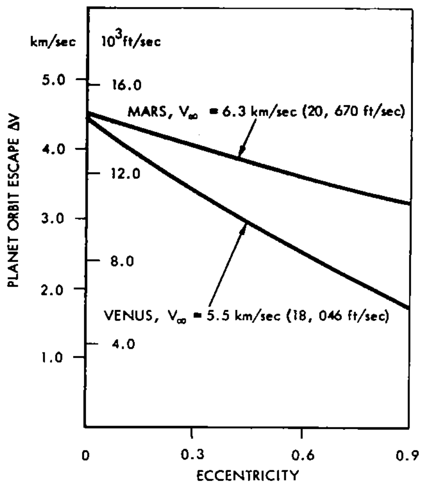
Propellant combinations considered include space storables (OF_2/MMH and FLOX/CH_4), cryogenics (LO_2/LH_2 and LF_2/LH_2), and a nuclear (H_2) system. The respective

(5) Manned Planetary Flyby Missions Based on Saturn/Apollo Systems. (Contract NAS8-18025) NAA S&ID, SID 67-549 (1 Aug. 1967).

(6) Definition of Experimental Tests for a Manned Mars Excursion Module. Contract NAS9-6464 (Nov. 1966).

(7) Technological Requirements Common to Manned Planetary Missions, Second Interim Report (Contract NAS2-3918) NAA S&ID, SID 67-294-1 (10 Mar. 1967).

(8) Study of Manned Vehicles for Entering the Earth's Atmosphere at Hyperbolic Speeds, Final Report NAS2-2526, Lockheed, LMSC 4-05-65-12 (Nov. 1965).



ORBIT PARAMETER*	MARS			VENUS		
	e=0	0.6	0.9	0	0.6	0.9
APOAPSIS ALTITUDE 10^3 km	0.3	11.3	66.5	0.3	19.8	117.3
ORBITAL PERIOD hours	1.9	7.4	59.5	1.6	6.4	50.8

* PERIAPSIS ALTITUDE, 300 km

Figure 6. Mars and Venus Parking-Orbit Characteristics

performance capabilities and packaging problems associated with both bell- and plug-nozzle engines were considered. Inasmuch as the tankage requirements for hydrogen-propelled nuclear system configurations may become excessive, other propellants than hydrogen (e.g., LiH, NH_3, etc.) were explored to determine if any advantage accrues from using higher-density propellants to reduce tankage requirements (and sizes).

3.0 AEROBRAKING MISSION-SYSTEM DESCRIPTION

A typical mission profile was developed so that the various operations and systems involved in a Mars landing mission could be viewed in perspective. The mission is shown schematically in Figure 7, and an inboard profile of the aerobraker vehicle is shown in Figure 8. An outbound Venus swing-by mode was chosen, but the profile developed could apply (with minor changes) to either Mars or Venus orbiting or landing missions.

The fully assembled aerobraking vehicle is launched unmanned to an assembly orbit. The crew is delivered to the assembly orbit in a logistics spacecraft which, presumably, has been developed in support of other Earth orbital programs (e.g., space stations). After the crew is transferred to the aerobraker, injection stages are mated to the vehicle, and rendezvous with the tankers is effected.

The fully fueled and checked-out aerobraker is shown to be injected into a transfer orbit to Mars by nuclear modules. (Two modules may be used for the first-stage injection, a third module being used after the first-stage modules are jettisoned.) If an abort situation should occur, the Mars-orbit-escape propulsion

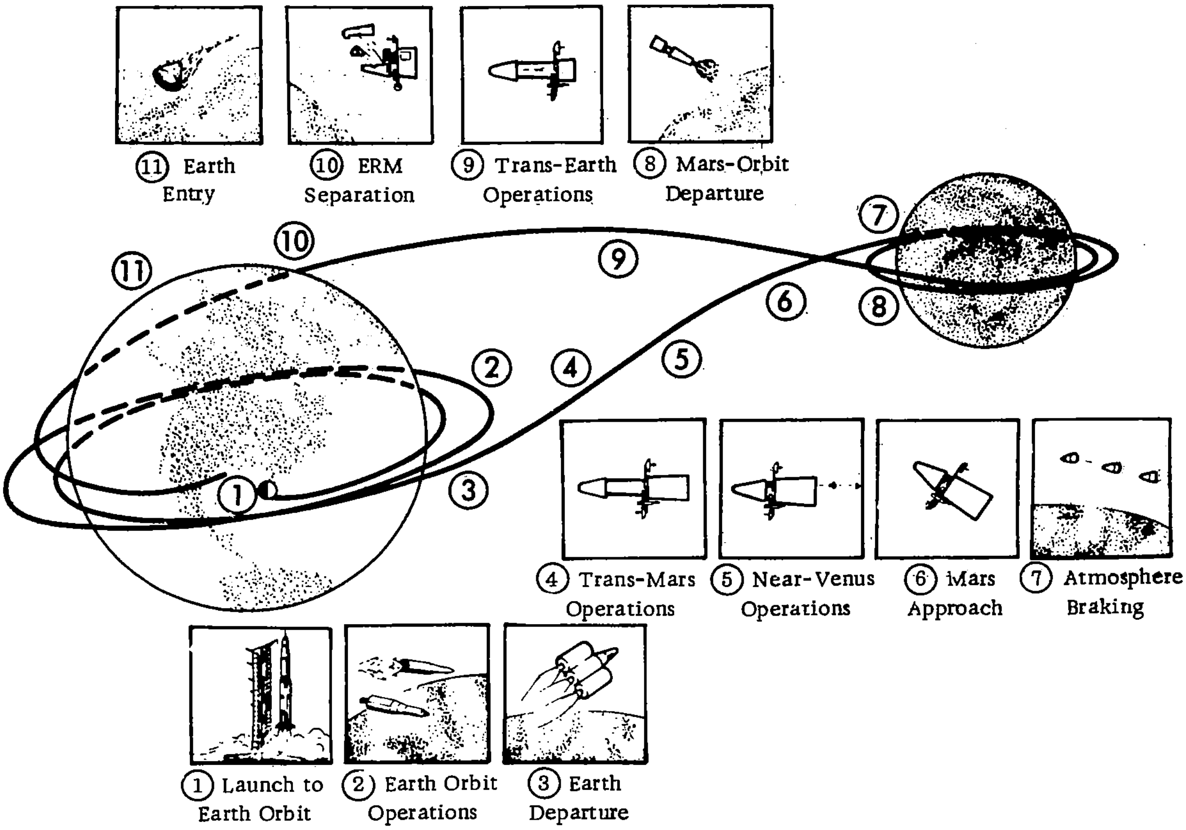


Figure 7. Mars Aerobraker Mission Profile

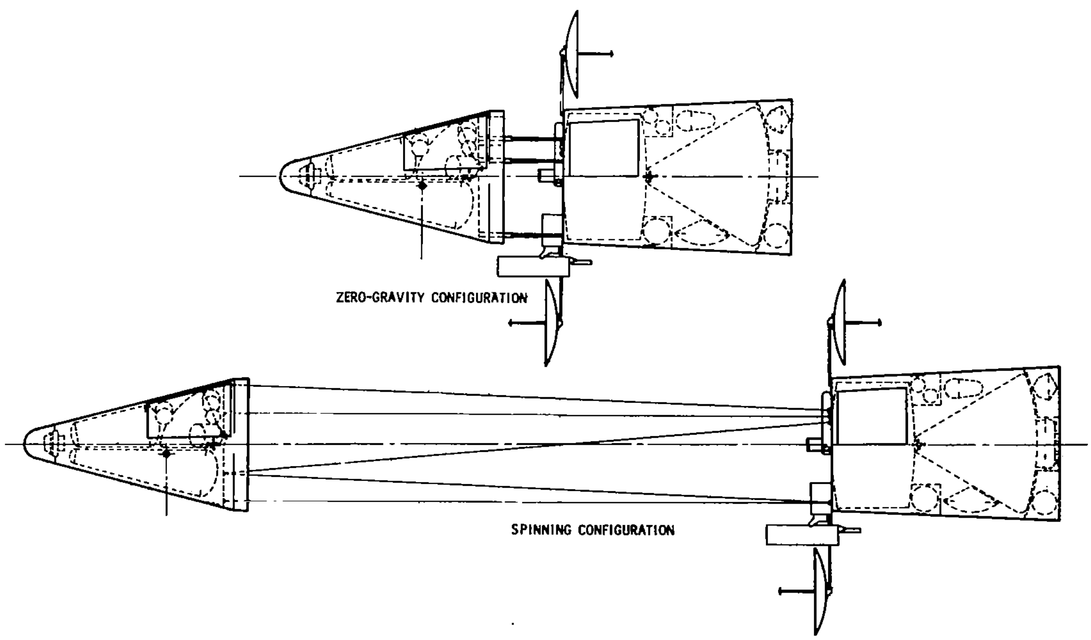


Figure 8. Spacecraft Configurations

system is used to place the spacecraft on an intercept trajectory with Earth. When Earth is approached, the crew enters the Earth reentry module (ERM) and effects a normal entry and recovery.

After injection, if a requirement for artificial gravity is established, the mission module (MM) and MEM may be extended on cables after the course correction and the resulting configuration spun up to achieve about 1/3 g. The electrical power system radiators, communication antenna, and instruments (e.g., telescope) then are extended and adjusted.

As the spacecraft approaches Venus along the outbound swingby trajectory, the Venus probes are checked out and injected into their proper transfer trajectories. Upon approaching Mars, the final course correction is made, the spacecraft's attitude is adjusted for entry, and all external appendages are retracted and stowed or jettisoned. The spacecraft encounters the atmosphere at an altitude of about 200 km (656×10^3 feet), with a fixed attitude, and at an entry angle (with respect to the local horizontal) near the middle of the allowable corridor. As the aerodynamic forces increase, deceleration rates are measured and time integrated to sense and generate real-time data on the entry flight path. The trajectory is matched to a preprogrammed reference trajectory, and deviations are corrected by modulating roll.

After pullup, the roll angle is adjusted to maintain constant-altitude flight until the proper velocity-density condition (which is a function of the chosen parking orbit characteristics) is reached for initiation of the exit maneuver. The maneuver is initiated by reducing the roll angle so that the vehicle ascends through the atmosphere. The atmospheric exit trajectory is guided and controlled in the same manner as entry to achieve the desired exit velocity and path angle.

After the spacecraft leaves the atmosphere, it coasts up towards the apoapsis altitude previously determined. The heat shield panels are jettisoned during coast. A powered maneuver at apoapsis delivers the spacecraft to its orbit, and the orbit parameters are then determined. Observations of the surface can be made to determine potential MEM landing sites. The spacecraft crew performs scientific observations in orbit, including the injection of a probe complement, after the MEM and its crew descends to the surface. When the planned surface staytime has been completed, the MEM ascends, rendezvouses with the spacecraft, and the crew transfers to the MM. Final checkout is initiated, and the procedure culminates with ignition of the planet-orbit-departure stage. After burnout, the trajectory parameters are determined in conjunction with the DSIF, and the time and attitude for the first midcourse correction are established. Upon approaching the Earth, the ERM is separated from the MM and propulsion units and is properly oriented for entry into the Earth's atmosphere. Entry and recovery are similar to current Apollo lunar mission procedures.

4.0 SIGNIFICANT RESULTS

The study results have shown that the selected aerobraker configurations exhibit satisfactory performance and packaging characteristics. There is considerable commonality between the spacecraft systems required to accomplish the Mars and Venus missions if the parking orbit eccentricity and probe payload parameters are adjusted appropriately. This finding resulted in the development of a modular approach to spacecraft design for Mars and/or Venus missions. These areas and other pertinent results are discussed in the paragraphs which follow.

4.1 AEROBRAKING VEHICLE REQUIREMENTS

Mission Modes—As can be seen in Figure 6, departure ΔV requirements for orbits with an eccentricity of 0.6 are approximately 25 and 50 percent lower than those required for departure from Mars and Venus circular orbits with corresponding "periapsis altitudes." The lower energy requirements can be translated to reductions in aerobraker entry weights of 40 percent at Mars and 55 percent at Venus for orbital missions. Introduction of a manned excursion module for a landing at Mars reduces the effect of the elliptical orbit inasmuch as the excursion module ascent propellant requirements increase with orbital altitude and/or eccentricity.

The MEM rendezvous propulsion requirements prompted an examination of an additional mission mode, i.e., a staged-aerobraker lander which employs an aerobraking excursion module to establish an intermediate circular orbit (at lower altitudes) prior to deorbiting the lander. The propulsion stage in the intermediate orbit later effects rendezvous with the parent spacecraft in the elliptical parking orbit. A laboratory attached to the propulsion stage could gather low altitude data and photographs. Use of this mode, along with highly elliptical parking orbits (i.e., $e > 0.6$), results in lower total system weights. For eccentricities lower than about 0.6, however, a weight penalty is incurred. Detailed comparisons of possibly enhanced operational flexibility versus the increase in system complexity as it effects crew safety and mission success were beyond the scope of this study.

Entry Performance—Performance studies defined the aerobraking entry corridors for Mars and Venus as a function of L/D, m/C_DA, atmosphere model, and undershoot criteria (i.e., minimum altitude or maximum deceleration). Assumptions included an approach navigation corridor capability of 20-km (~10 n. mi.), a 10-km (33,000-ft) minimum pullup altitude at Mars, a 10-g undershoot limit at Venus, and a 10-m (33-ft) spacecraft base diameter. The L/D requirement at Mars increases monotonically with the ballistic parameter (m/C_DA) for the several atmospheres investigated. Figure 9 shows this variation for the "worst case" atmosphere (VM-8) for a range of velocity while Figure 10 shows typical values of L/D as a function of C_DA and angle-of-attack for a given set of vehicle geometric parameters. It should be noted a given L/D ratio can be achieved by flying at both low and high angles of attack. These data essentially define the biconic configuration requirements, i.e., packing density and geometry. For example, inasmuch as preliminary estimates of Mars vehicle weights on the order of 150,000 to 200,000 kg (330,000 to 440,000 lb) were indicated, the need for a high angle of attack is immediately apparent to satisfy the m/C_DA - L/D relationship indicated by Figure 9 for the low-density VM-8 atmosphere.

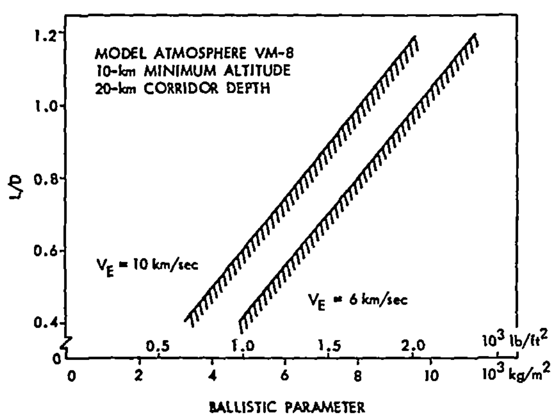


Figure 9. Effect of Orbit Eccentricity on Vehicle Weight

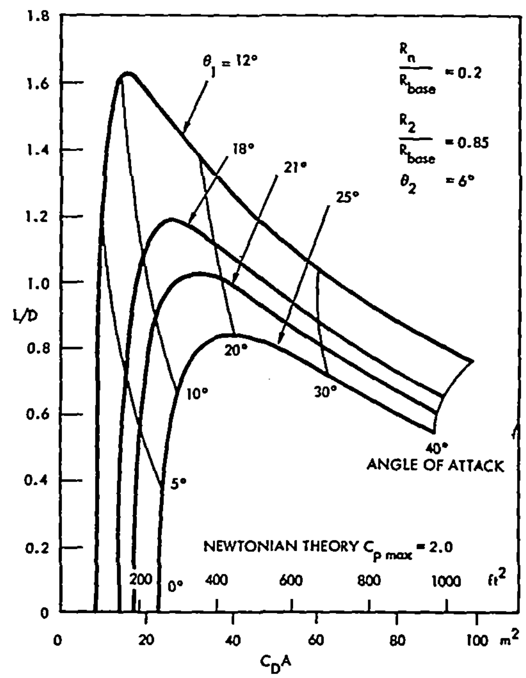


Figure 10. Lift-to-Drag Ratio Requirements

In the dense Venus atmospheres, the effect of m/C_DA on required L/D is eliminated inasmuch as m/C_DA only affects the position of the corridor (i.e., height) in the atmosphere. In the future, an altitude limit may be placed on Venus entries to assure pullup prior to cloud penetration. Such a limit was not considered at this time because of the lack of suitable estimates of cloud-layer heights. Inasmuch as m/C_DA is not a problem, the Venus-entry vehicles can be operated at low angles of attack.

Gasdynamic Heating — The atmospheric density profile and composition, entry conditions and corridor boundary, and vehicle characteristics (e.g., L/D, m/C_DA, geometry) all significantly affect aerobraker gasdynamic heating rates and loads.

Heating-rate differences induced by the density variation across the assumed 20-km entry corridor were greater than those due to uncertainties in the atmospheric density profile or composition. Consideration of nonadiabatic and self-absorption phenomena in determining radiation heating further reduces the effect of variations in atmosphere composition. Figure 11 shows the effect of m/C_DA and analytical model on radiative heating which predominates at the stagnation point. Nose bluntness and forebody cone-angle are the dominant geometric parameters with regards to heating at other locations; the afterbody half-cone angle has a negligible effect.

The heat loads developed were used to determine heatshield requirements for the many possible variations in entry conditions and geometry. Ablator thicknesses were obtained at different points on the vehicle, and thickness distributions were developed. The ablator weight contributes about 2/3 of the total heatshield weight; the other 1/3 is support structure and insulation. The total heatshield weight fraction as a function of entry velocity is shown in Figure 12 for both Mars and Venus vehicles. The bands shown represent a range of ballistic parameters and show that the heatshield weight fractions range from 5 to 20 percent of the aerobraker entry weight. If these ratios are interpreted as mass fractions, an equivalent heatshield I_{sp} for Mars is approximately 7000 seconds ($\Delta V \sim 4$ kilometers per second), while for Venus ($\Delta V \sim 3$ kilometers per second) it approaches ~2000 seconds.

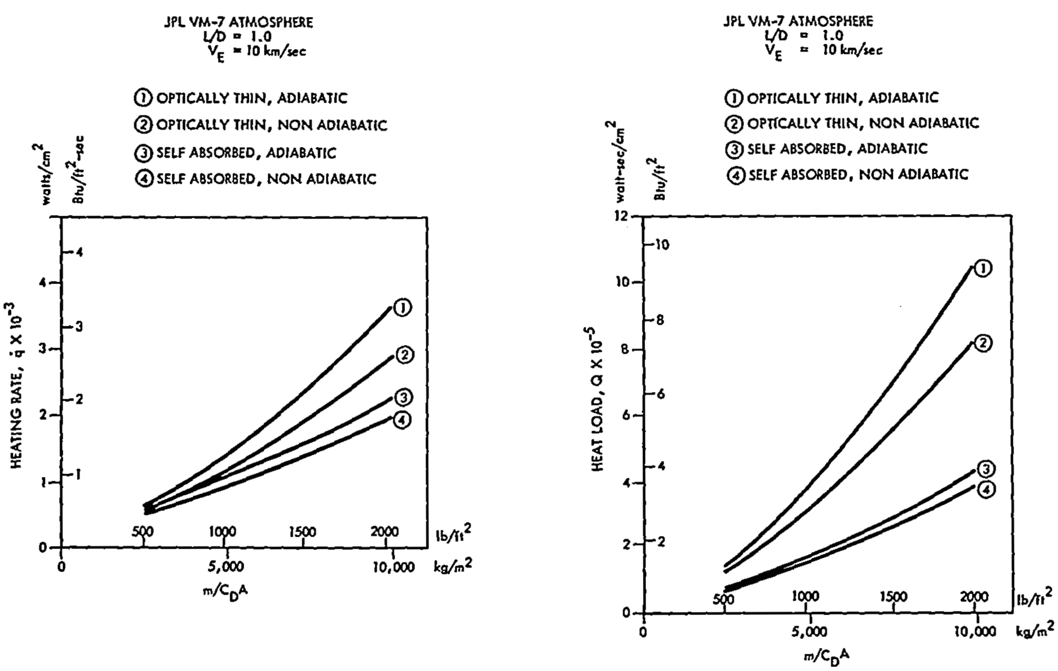


Figure 11. Effect of m/C_DA and Analytical Model on Mars Aerobraker Stagnation-Point Heating

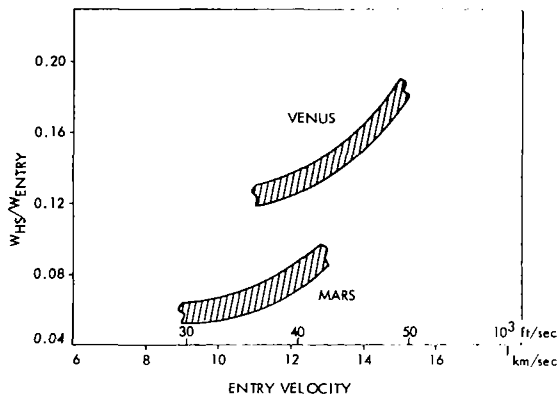


Figure 12. Effect of Entry Velocity on Heatshield Weight

4.2 CONFIGURATION ANALYSIS

Mars aerobrakers for orbiter and lander missions and Venus designs for orbiter missions (i.e., no manned excursion module) were developed for space-storable, cryogenic, and nuclear propulsion planet-orbit-departure systems for the several parking orbits of interest. Indicated lengths for the nuclear vehicles were in excess of 40 m (132 ft) if the base diameter is constrained to 10 m (33 ft); furthermore, the resulting configurations did not provide satisfactory aerodynamic characteristics. Difficulty was encountered in achieving reasonable stability margins with cryogenic propellants (i.e., LO_2/LH_2).

Of the propellant combinations considered, the space-storables (e.g., OF_2/MMH or FLOX/CH_4) show the most promise for aerobrakers.

Further analyses indicated that FLOX/CH_4 is the most desirable based on performance, storability, and engine cooling characteristics; this combination was selected for the final designs.

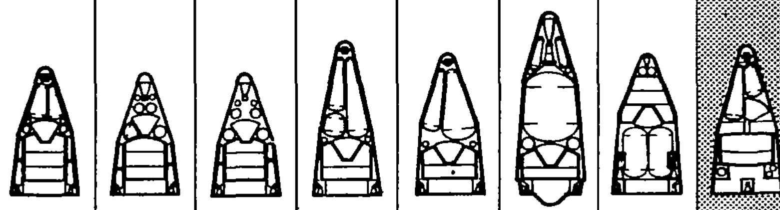
In addition to the propulsion analyses, structural concepts for the thermal-structural system, load-carrying structure, and tank and module supports also were investigated. A skin-stringer concept was selected to accommodate the high anticipated in-plane loads; the skin-stringer unit weights of 15 to 20kg/m^2 (3 to 4 lb/ft^2) were approximately 10 percent lighter than equivalent honeycomb or truss-core concepts.

A study of spinning-configuration dynamics was conducted for typical aerobraker designs, in the event that artificial gravity is required. The results indicated no significant problems with this mode and that the propellant requirements for spin and despin are on the order of 3 to 5 percent of the vehicle weight. (All of the designs considered are applicable to either the spinning or nonspinning mode.)

Matrices of the designs considered are presented in Tables 3 through 6. The configurations shown were chosen to compare the effects of circular and eccentric parking orbits; storable (OF_2/MMH or FLOX/CH_4), cryogenic (LO_2/LH_2), and nuclear-LH_2 propulsion systems; and module selection on the aerobraker packaging arrangement.

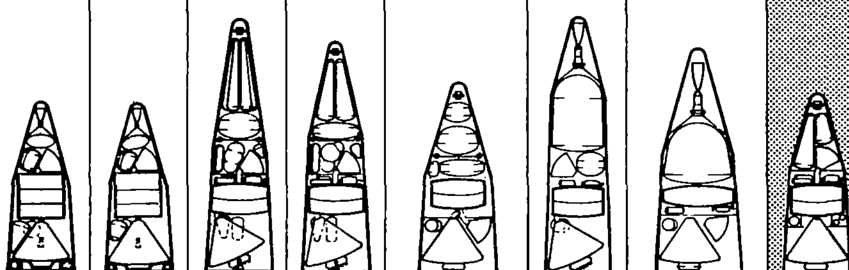
The Mars orbiters shown in Table 3 are grouped by propellant type, the heaviest vehicles being

Table 3. Mars Orbiter Mission Vehicles



ERM: Apollo MM: 8-man, 430-day Periapsis altitude 300 km Base diameter 10 m (33 ft)								
Eccentricity	0	0.6	0.9	0.6	0.9	0.6	0.9	0.6
Propellants	S	S	S	C	C	N	N	S
Length								
m	19.7	18.8	18.8	23.4	21.5	29.6	21.5	22.8
ft	65	61	61	77	71	97	71	75
Entry weight								
10^3 kg	265	186	165	188	161	164	147	177
10^3 lb	584	410	364	413	353	362	324	390
Injected weight								
10^3 kg	279	195	173	197	168	173	155	189
10^3 lb	614	430	382	434	370	380	341	417

Table 4. Mars Lander Mission Vehicles



ERM: Apollo MEM: direct, Apollo MM: 8-man, 430-day Periapsis altitude 300 km Base diameter 10 m (33 ft), except as noted					$D_B = 11.7$ m (39 ft)		$D_B = 13.3$ m (44 ft)	
Eccentricity	0	0.6	0.6	0.9	0.6	0.6	0.6	0.6
Propellants	S	S	C	C	C	N	N	S
Length								
m	26.6	26.6	40.0	34.6	30.0	40.6	36	28.4
ft	87	87	127	113	98.5	138	118	93
Entry weight								
10^3 kg	296	247	244	227	242	220	218	224
10^3 lb	652	543	536	500	533	484	480	495
Injected weight								
10^3 kg	310	260	256	239	254	231	230	244
10^3 lb	685	571	563	526	560	509	505	539

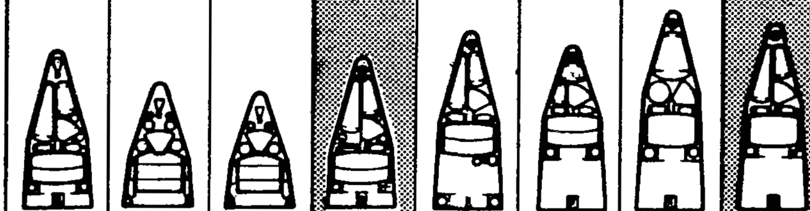
Table 5. Venus Orbiter Mission Vehicles

ERM: Apollo

MM: 8-man, 380-day

Periapsis altitude 300 km

Base diameter 10 m (33 ft)



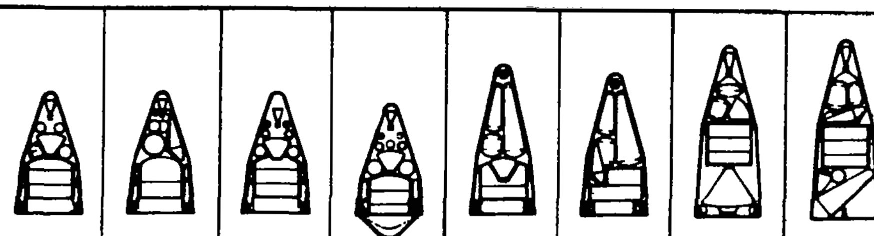
Payload weight								
10^3 kg	9.1	9.1	9.1	9.1	45.5	45.5	45.5	45.5
10^3 lb	20	20	20	20	100	100	100	100
Eccentricity	0	0.6	0.9	0.6	0.6	0.9	0.6	0.6
Propellants	S	S	S	S	S	S	C	S
Length								
m	23.8	18.8	18.0	22.8	27.0	25.0	30.4	28.4
ft	78	61	59	75	89	82	100	93
Entry weight								
10^3 kg	276	135	111	146	234	174	230	194
10^3 lb	607	297	243	321	516	382	507	427
Injected weight								
10^3 kg	290	142	116	152	246	185	242	202
10^3 lb	638	312	256	335	542	408	533	446

Table 6. Module Shape Variations

MM: 8-man, 430-day

Periapsis altitude 300 km

Base diameter 10 m (33 ft)



Eccentricity	0.6		0.9		0.6		0	
Propellants	S		S		C		S	
Mission	Mars orbiter		Mars orbiter		Mars orbiter		Mars lander	
ERM	Apollo	Conic seg	Apollo	Apollo	Apollo	Conic seg	Apollo	Conic seg
MEM	——	——	——	——	——	——	Apollo	Lifting body
Length								
m	18.8	18.8	18.8	21.0	23.4	21.9	26.6	28.7
ft	61	61	61	69	77	72	87	94
Entry weight								
10^3 kg	186	184	165	165	187	186	297	298
10^3 lb	410	407	364	364	413	410	652	652
Injected weight								
10^3 kg	195	194	174	174	197	196	310	310
10^3 lb	430	427	382	382	434	431	685	685

located to the left of the chart. It is readily apparent that, while the cryogenic and nuclear propellant aerobrakers are lighter for a given mission, they are also much longer. The three smallest orbiters use storable propellants for planet departure; all are approximately the same size, with length-to-diameter ratios of about two. This allows a fairly large aft-cone angle, resulting in a center of pressure located aft of the cone intersection. In order to achieve an acceptable center of gravity, the POE propellant is stored in the annulus around the three-floor mission module.

Based upon the investigations conducted during the study, a recommended Mars aerobraker orbiter, indicated by the shading in Table 3, was developed. This configuration is powered by a single stage for planet departure, using storable $FLOX/CH_4$ propellants and an aerospike engine. The mission module chosen has two floors, curved bulkheads, and a diameter of 8.2 m (27 ft). It incorporates an equipment bay on the forward end in which antennas, telescopes, solar panels, and scientific instruments are stored. The aft end of the MM provides storage for life-support gases, outbound-course-correction propellant tanks, and a scientific laboratory area. The probe compartment is located at the aft end of the vehicle.

Aerobraker vehicles for the Mars landing missions are shown in Table 4. The configurations afford a comparison of the effects of parking orbit on vehicle design for each of the propulsion systems considered; i.e., storable (OF_2/MMH, $FLOX/CH_4$), cryogenic (LO_2/LH_2), and nuclear (LH_2). The comparison also is made between the several propulsion systems for a given orbit (e.g., e = 0.6). This group of landers is arranged similarly to the orbiter family, the heaviest vehicle shown to the left.

Mars vehicles using nuclear (LH_2) propulsion are the lightest of the total Mars lander spectrum; their weights run about 10 percent less than the cryogenic landers and 11 percent less than the storable landers for a given orbit. The 10-m (33-ft) base diameter configuration sized for the 0.6 e orbit approaches a length-to-diameter ratio of about five, because the LH_2 located forward of the MM requires such a large volume that only a small aft-cone angle is allowed. The MEM and probes preclude the propellant from being in the aft end, with its larger volume, because of their own size and position requirements. These lengths and packaging requirements create difficult stability problems and preclude the use of nuclear (LH_2) propulsion for aerobraker vehicles.

An integrated lander design, indicated by the shading in Table 4, was developed. This configuration achieves a proper balance between cg and cp and makes use of $FLOX/CH_4$ propellants in an aerospike engine. The $FLOX/CH_4$ combination appears to present less long-term storage problems than the other storables considered (OF_2/MMH).

For the Venus orbiter family shown in Table 5, aerobrakers carrying an Apollo-shaped ERM and either a light or a heavy probe complement were developed to compare the effects of parking orbits and propulsion systems. All the spacecraft shown are similar in arrangement and carry the single planet-departure propulsion stage in the nose, followed by the ERM, MM, and probes. A recommended design, indicated by the shading in Table 5 was generated. For purposes of commonality, the exterior shape and basic arrangement are identical to the equivalent Mars aerobraker, even though the Venus mission requires smaller POE tanks. If the POE tanks were filled, an orbit with an eccentricity as low as 0.2 could be achieved at Venus.

Table 6 illustrates a number of Mars aerobrakers (both orbiters and landers) configured to compare the effect of the conic segment ERM and the lifting body MEM. Also, two Mars orbiter designs, sized for highly eccentric orbit (i.e., e = 0.9) and using storable propellants, were developed to compare the effect of an external probe compartment on the aerobraker length from base plane to nose.

Little or no difference was found in the length or packaging arrangements between the Apollo-shaped and conic segment ERM. Storing all the probes externally aft of the bulkhead resulted in a 5-percent shorter Mars orbiter. The lifting body MEM requires about the same length compartment as the Apollo-shaped MEM. No appreciable change in spacecraft gross weight is noted for these shape variations.

Figure 12 presents a summary of the vehicle injected weight variation with parking orbit eccentricity for both Mars and Venus missions. Also shown are the weights for direct and staged Mars excursion modules.

4.3 RECOMMENDED CONFIGURATIONS

Recommended aerobraker designs include Mars and Venus orbiters and Mars lander, Venus orbiter vehicles sized for a planet departure ΔV of 3.6 km/sec (11,800 ft/sec), equivalent to an eccentricity of 0.6 at Mars and 0.2 at Venus. These designs used $FLOX/CH_4$ for the departure propulsion and carry Apollo-shaped Earth-reentry modules. The Mars lander also carries an Apollo-shaped excursion module. Schematic profiles of the four recommended designs and pertinent weight data are shown shaded in Tables 3 through 6.

Heatshield weight breakdowns for the four selected configurations are shown in Figure 13. The weights ranged from 7 to 15 percent of the vehicle entry weight; the Venus orbiter required the highest heatshield design was considered for both Mars and Venus entry (assuming the spacecraft could be trimmed at the proper angle of attack). The weight penalties imposed on the recommended designs for a common heatshield can approach 50 percent of the heatshield weight. This penalty can be reduced

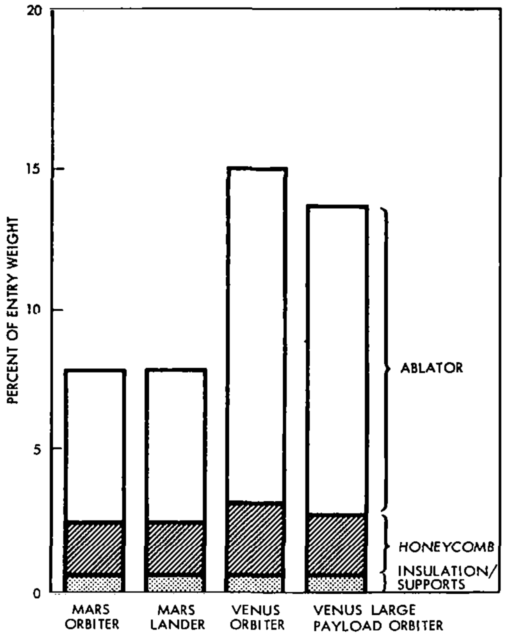


Figure 13. Heatshield Weight Breakdowns

by judicious selection of the mission and vehicle parameters (e.g., V_e, m/C_DA, α, corridor depth).

4.4 SENSITIVITY ANALYSES

Sensitivity analyses were conducted to determine the effects of changes in assumptions, mission parameters, crew size, propulsion requirements, and packaging arrangements on the selected configurations, and the data are summarized in Table 7. The Earth-departure weights are most sensitive to changes in the planet-orbit escape (POE) ΔV requirement, and in the mission (MM) weight. The MM weight is a function of crew size, mission duration, and type of life support system. For the subsystems considered, the MM weight increases by 10 percent for a 25 percent increase in the number of man-days.

The POE ΔV requirement is a function of the parking orbit selected and the departure V_∞. Changing the parking orbit from 0.6 eccentricity to a 300-km circular orbit increases the POE ΔV 20 percent, resulting in a 20- to 40-percent increase in Earth departure weight.

Performance - Entry corridor sensitivities to the environments, flight modes, mission parameters, and guidance concepts were established to define the approach navigation and maneuvering requirements. A vehicle with an L/D of 1.0 and an $m/C_DA \leq 8500$ kg/m^2 (1750 lb/ft^2) could accomplish the aerobraking orbital capture maneuver at Mars in VM-8, the most tenuous atmosphere postulated. At Venus, the maneuver can be accomplished in the atmosphere models considered with an L/D of 0.5 to 1.0 and $m/C_DA \leq 20{,}000$ kg/m^2 (4000 lb/ft^2).

Flight-mode performance was evaluated in terms of exit velocity and flight-path-angle accuracies required to achieve a prescribed parking orbit versus the applied ΔV correction. An almost one-to-one relationship was found between the deviation in exit velocities from the prescribed value and the applied ΔV required. Flight modes considered included constant flight path angle and constant altitude trajectories; the choice of mode produced a negligible effect on exit velocity accuracy. Required roll rates for both modes were estimated to be on the order of 3 degrees/second.

Table 7. Aerobraker Spacecraft Weight Sensitivities

Module			Change	Percent Change in Injected Weight			
				Mars		Venus Orbiter	
				Lander	Orbiter	Large Payload	Small Payload
Mission module			±10% in module weight				
8 men - 430 days				±4.1	±5.1		
8 men - 380 days						±3.3	±4.1
Earth reentry module							
8 men - Apollo shape				±1.1	±1.4	±1.0	±1.3
Probes							
9,080 Rg (20,000 lb)				±0.5	±0.7	±0.7	±0.9
36,320 Rg (80,000 lb)						±2.4	
Mars excursion module							
4 men - 30 days Apollo shape				±1.9			
Propulsion System	ΔV m/sec	ΔV ft/sec					
Outbound course correction	76	250	±20% ΔV			±0.4	±0.4 ±0.4
	152	500	±20% ΔV	±0.8	±0.8		
Outbound spin and despin	132	440	±2 cycles*	±1.0	±1.0		
	66	220	±1 cycle			±0.5	±0.5
Planet orbit attainment and maintenance	152	500	±20% ΔV	±0.8	±0.8		
	76	250	±20% ΔV			±0.4	±0.4
Planet orbit spin and despin	66	220	±1 cycle	±0.2	±0.2	±0.3	±0.4
Planet orbit escape	3,600	11,800	±10% ΔV	+10.7 / - 7.6	+13.8 / - 9.8		
	2,700	8,200	±10% ΔV			+4.3 / -3.4	±5.8 / -4.7
Return spin and despin	66	220	±1 cycle	±0.3	±0.4	±0.4	±0.5
Return course correction	76	250	±20% ΔV	±0.3	±0.4	±0.3	±0.3

*1 cycle = 22 m/sec (67 ft/sec)

Heating and Heat Protection - Convective and radiative heating rates for the selected vehicles were computed using methods representative of the current state of the art. It was found that the choice of analytical model used to obtain the radiative heating (i.e., adiabatic, nonadiabatic with self-absorption, etc.) was more significant than the variations in the atmospheric composition. At Mars, there was a decrease in maximum heating rate of 15 percent in going from VM-7 to the MSC-3 atmosphere. Changing the analytical model from the simple adiabatic case to the sophisticated nonadiabatic, self-absorption approach results in a stagnation heating rate almost 50 percent lower, while peak heating at the other locations is reduced by as much as a factor of five. A 20-percent increase in entry velocity increased the peak stagnation heating rate by a factor greater than two; increasing the angle of attack from 8 degrees to 28 degrees increased the radiative heating rate on the conical surface by more than an order of magnitude.

The effects of these uncertainties on the heat protection system weights were reflected in heatshield weight variations of almost 100 percent, or vehicle entry weight variations of from 7 to 15 percent. The heatshield design for the Mars mission was 35 percent lighter than the Venus heatshield. If a common heatshield design were used for both missions, the weight-in-Earth-orbit penalty for the Mars mission would be 3 to 8 percent. This penalty can effectively be eliminated by suitable compromises in the mission and vehicle parameters.

Structure and Shielding - Integrated structure unit weights of 15 to 20 kg/m^2 (3 to 4 lb/ft^2) were found for the range of configurations studied. These weights include the load-carrying structure, as well as meteoroid protection requirements to assure a 99-percent probability of no penetration in 430 days, given the nominal meteoroid flux model.(9) The unit weights derived were more sensitive to the analytical method applied than to the mission assumptions.

Design Integration - Sensitivities of the configuration packaging arrangements to changes in the orbital parameters and candidate propulsion system for Mars lander vehicles were illustrated earlier in Table 3 through 6. The Earth reentry module (ERM) shape was found to have no effect on the vehicle length or arrangement. An Apollo-shaped Mars excursion module (MEM) was selected; use of the 16-m (35-ft) long lifting body MEM does not severely penalize the packaging arrangement. The effect of 4- to 12-man crews on the packaging arrangements is not clear because crew free-volume requirements are poorly defined. For example, a mission module designed for 8 men with a free volume per man of 20 m^3 (700 ft^3) could be used for 12 men at 13 m^3 (470 ft^3) free volume per man; recommended

(9) Manned Planetary Flyby Missions Based on Saturn/Apollo Systems, op. cit.

free volumes range from 10 to 20 m^3 per man. [10]

4.5 MODULAR APPROACH TO VEHICLE SYNTHESIS

In the latter phases of the configuration analysis, it became apparent that a series of common modules could be used to synthesize the aerobraker. The approach, illustrated in Figure 14, is attractive because only four basic segments are needed for all the orbiting and landing missions of interest.

(10) Davenport, E.W., Congdon, S.P., and Pierce, B.F. "The Minimum Volumetric Requirements of Man in Space," AIAA Paper No. 63-250, presented at AIAA Summer Meeting in Los Angeles, California (17-20 June 1963).

The vehicle is comprised of three major modules; i.e., a propulsion (or nose module), a mission module, and a probe or probe-and-lander module. Initially, the mission and probe modules for each configuration were developed as a unit to satisfy the cp-cg relationships, volumetric requirements, and the base-diameter constraints. After the configurations were analyzed, it appeared possible to treat these modules as discrete components in order to establish a mission module shape common to both the orbiter and lander configurations.

Each spacecraft heatshield initially was treated as a unique design; if common elements were to be specified for the propulsion, mission, and probe modules, a common heatshield capable of satisfying multimission

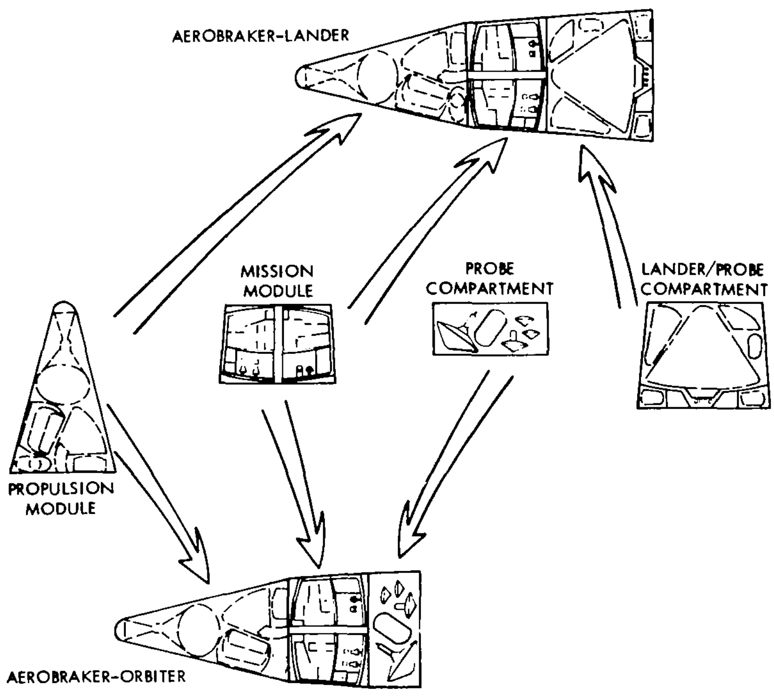


Figure 14. Modular Approach to Aerobraker Synthesis

and spacecraft requirements might be expected to suffer some weight disadvantage. It was found that although the Mars heatshield weights increased 33 to 36 percent to achieve commonality, the gross spacecraft weight increase would be less than 3 percent; the Venus heatshield weight would increase 2 to 6 percent and the spacecraft gross weight would increase less than 1 percent. These increases could be reduced to negligible values by altering the mission-system parameters slightly (i.e., velocity and L/D). Although some penalty would be incurred in the injection stage, the obvious advantage of a common heatshield appears to make this a worthwhile trade-off.

4.6 TEST AND QUALIFICATION CONSIDERATIONS

Acceptance of the aerobraker mode will require sufficient testing to establish a high level of confidence in the aerobraking mission. The requirements for these tests were considered briefly. It was found that the aerobraking maneuver is of the same order of complexity as return and entry along a maximum range trajectory on current Apollo lunar missions where the Apollo exits the atmosphere after being slowed to near orbital velocity and eventually enters again to achieve the desired range. This entry profile is, in almost all respects, a valid simulation of the aerobraking concept.

Additional aspects of the vehicle's design and performance could be tested in a combination of ground, earth-atmosphere, and cislunar space environments. No planetary-based tests would be required beyond the currently planned unmanned data-gathering probes. If the modular approach to the aerobraker design is adopted, parts of the vehicle could be used as Earth-orbital space stations while being tested for the more demanding planetary missions. Although this approach is most promising, it has not been studied in detail.

5.0 RELATIONSHIP TO OTHER NASA PROGRAMS

The current study was performed for the Mission Analysis Division of NASA's Office of Advanced Research and Technology and is the fifth in a related series of manned interplanetary mission requirements studies conducted by North American Rockwell. The series started in 1964 with Contract NAS2-1408, "Manned Mars Landing and Return Mission Study." The baseline aerobraker configuration used in the present investigation was developed in 1964 under Contract NAS9-1748, "Study of Subsystems Required for a Manned Mars Mission Module"; a sensitivity analysis of the baseline aerobraker vehicle was conducted in 1965 under Contract NAS2-2477, "Study of Unmanned Systems to Evalute the Martian

Environment." The results of a concurrent study of the Mars excursion module, conducted under Contract NAS9-6464. entitled "Definition of Experimental Tests for a Manned Mars Excursion Module," were used directly in this study. Two other parallel studies which furnished valuable data in some areas were Contract NAS8-18025, "Study of Manned Planetary Flyby Missions Based on Saturn/Apollo Systems," and Contract NAS2-3918, "Study of Technological Requirements Common to Manned Planetary Missions."

More comprehensive data have been generated in this investigation of the aerobraking mode to effect planet-orbital capture. The size- and weight-scaling relationships developed can readily be extrapolated to assess effects of mission date, duration, and flight mode. The designs developed illustrate possible packaging arrangements and volume utilization and will serve as the baselines for the next series of manned planetary capture and landing studies.

6.0 SUGGESTIONS FOR FUTURE WORK

Two broad areas of future studies are recommended as a result of the work accomplished. The first is described as "Planetary Capture Mission-System Analyses" and the second as "Atmosphere Braking Vehicle Technology Studies."

6.1 PLANETARY-CAPTURE MISSION-SYSTEM ANALYSES

The current study provides data which indicates that the aerobraking mode has a potentially significant advantage over the retrobraking mode when the two are compared on the basis of weight in Earth orbit. Furthermore, the technology development requirements in propulsion (i.e., nuclear systems) are more demanding for retrobraking than for aerobraking, inasmuch as the latter can be accomplished with a chemical planet-orbit-departure stage. Aerobraking appears feasible in the currently postulated family of Mars and Venus atmospheric models; a segmented modular vehicle concept which affords multiplanet, multimission capability and has obvious economic attractions has been defined. Consequently, detailed comparative analyses of aerobraking and retrobraking should now be performed for a wide range of mission opportunities. The analyses should consider developmental problems, cost, reliability, and schedule risk, as well as the system requirements.

The availability of an acceptable test and qualification plan for an aerobraker spacecraft system, preferably one which could be conducted in the near-Earth environment, would go far in promoting acceptance of this mode. This program should encompass both scaled models and the modularized segments to man rate the total system; detailed sizing, scaling, and simulation would be a prerequisite analysis. The modular approach to vehicle synthesis suggested by this investigation should be extended to both Earth-orbital and planetary-flyby missions, and the associated compromises (and penalties) in the mission and spacecraft requirements and capabilities should be identified.

6.2 ATMOSPHERE-BRAKING-VEHICLE TECHNOLOGY STUDIES

Suggested technology studies of the atmosphere-braking vehicle are presented briefly in these concluding paragraphs.

Crew Systems and Functions - Establish crew timelines, functional operations, and associated crew system requirements (e.g., free volume per man, displays and controls, living quarter arrangements, etc.). These data are needed for the design and planning of ground and Earth-orbital tests.

Vehicle Design - Determine the design requirements for the modular approach to spacecraft synthesis, including the arrangements of the major modules (i.e., ERM, MEM, probes) within the spacecraft, heatshield joints, separation planes, external appendages (such as radiators and antennas), and manufacturing considerations.

Aerodynamics and Entry Performance - Establish the aerodynamic coefficients and stability characteristics of the modular spacecraft. Evaluate candidate triconic configurations to assure adequate stability and lift capability. Study entry dynamics in sufficient detail to derive the guidance and control system requirements, including the reaction control system. Parking orbit characteristics for selected mission opportunities must be examined in more detail so that orbit precession is taken into account as well as the proper orientation of the approach and departure V_∞ vectors.

Heating and Heat Protection - Refine the computation of the radiative heating environment to include the contribution of atomic lines considering a self-absorbed, nonadiabatic flow field. The effects of ablation-product radiation in the flow field and wake should be examined in detail. Investigate the properties of high-density carbon-based ablative materials in the regions of high heat flux. Experimental verification of ablation rates and material reactions in chemical models of the candidate atmospheres must be obtained. The heatshield design currently is envisioned to be composed of a large number of panels which must cover a surface area of approximately 750 m^2 (8000 ft^2). These panels are jettisoned after exit from the atmosphere. Determine the performance of such a segmented heatshield design and investigate other candidate heatshield concepts. Determine entry-velocity and angle-of-attack requirements for a common heatshield for both planets.