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SPACE VEHICLE CONCEPTS

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ABSTRACT

This paper presents several concepts of chemical-propulsion Space Vehicles (SVs) for manned Mars landing missions. For vehicle sizing purposes, several specific missions were chosen from opportunities in the late 1990's and early 2000's, and a vehicle "system" concept is then described which is applicable to the full range of missions and opportunities available. In general, missions utilizing planetary opposition alignments can be done with smaller vehicles than those utilizing planetary opposition alignments (reference 1) . The conjuction missions have a total mission time of about 3 years, including a required stay-time of about 60 days. Both types of missions might be desirable during a Mars program, the opposition type for early low-risk missions and/or for later unmanned cargo missions, and the conjunction type for more extensive science/exploration missions and/or for Mars base activities. Since the opposition missions appeared to drive the SV size more severely, there were probably more cases examined for them.

Some of the concepts presented utilize all-propulsive braking, some utilize an all aerobraking approach, and some are hybrids. Weight statements are provided for various cases. The aerobraking cases have significant advantages in size and weight. Cryogenic propellants were used for the main propulsive elements in all cases, due to their significant weight advantage over storable propellants (reference 1). Extensive use is made of existing propulsive elements and other systems.

Most of the work was done on 0-g vehicle concepts, but partial-g and 1-g concepts are also provided and discussed. A recommendation is added that efforts be made to find ways to offset the long-term 0-g effects on the crew, other than providing a g-field for the total SV or spacecraft, since this causes significant design and operations impacts.

Several options for habitable elements are shown, such as large-diameter modules and Space Staion (SS) types of modules. The latter were used as a reference because of their cost advantage as existing elements.

Several options are shown for the Mars landing vehicle, and a landing "system" is recommended which makes use of a large aeroshell to allow landing of payloads of various sizes and shapes over the course of a multi-year program.

Because of the large size and weight of the SV it will be necessary to launch individual elements and assemble them in low Earth orbit (LEO). A configuration of one potential assembly concept is provided.

ALL-PROPULSIVE OPTION

Figure 1 illustrates an all-propulsive option which is sized for propulsive braking maneuvers (no aerobraking at Mars or Earth return) using LO_2/LH_2 propellants. This vehicle is sized for the 1999 mission opportunity, using an opposition-type trajectory. The concept utilizes 3 propulsion stages for the mission which accomplish LEO departure, Mars arrival and departure, and Earth braking, respectively. The stages are jettisoned after use, including jettison of external hydrogen tankage prior to departure from Mars. This figure also provides the terminology used for the configuration elements. Figure 2 illustrates the concept at different stages during the mission. In the Earth-Mars transit phases, the normal vehicle orientation is with its long axis towards the sun, to minimize propellant boiloff losses. Other orientations can be effected occasionally, as long as they are kept within reasonable limits.

The stage sizing and tank arrangements were influenced by the size and delivery capability of the launch vehicle used for delivery of elements to LEO, with a significant amount of on-orbit propellant transfer necessary to fill the propellant tanks.

The engines for the first stage are Shuttle-derived Space Transporation Main Engines (STME's), as defined in reference 2. The first stage tanks are derivable from the SDV-3R Earth-to-orbit (ETO) vehicle (see reference 3) or from the Shuttle External Tank (ET). The second and third stage engines are Orbital Transfer Vehicle (OTV) - derived RL-10 engines, as defined in reference 2. The second stage tankage should be derivable from the SDV-3R (or ET) and the OTV, and the third stage tanks should be derivable from the OTV. The tanks are insulated with 4 inches of multilayer insulation and are outfitted with vapor-cooled shields, to

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FIGURE 1
MANNED MARS MISSION
1999 OPPOSITION
ALL PROPULSIVE OPTION

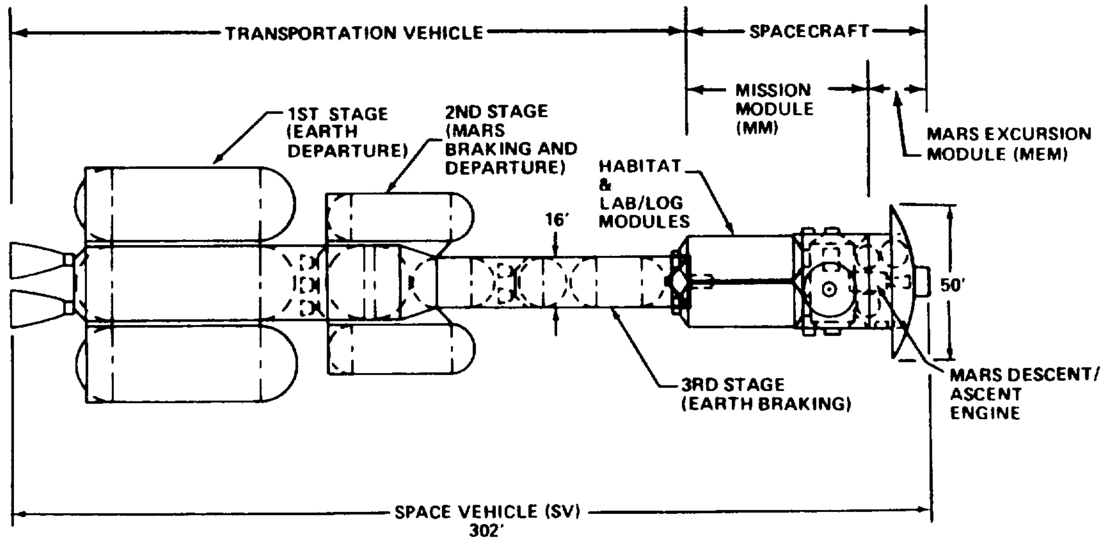

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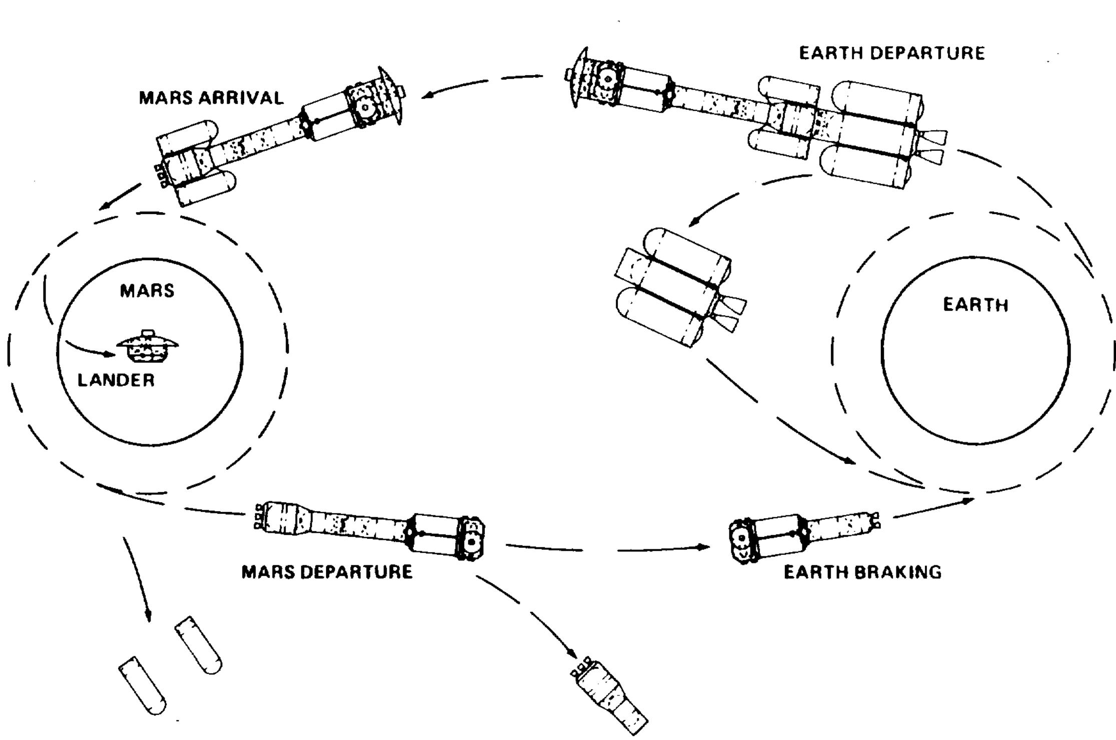
FIGURE 2.
1999 OPPOSITION
MANNED MARS MISSION
ALL PROPULSIVE STAGING



minimize cryogen boiloff for each stage. A discussion on insulation thickness trades is provided in reference 4.

The spacecraft portion of the vehicle consists of the Mission Module (MM), (which includes 3 Space-Station (SS)-type modules), the Mars Excursion Module (MEM), (which consists of a lander and ascent stage for the Mars surface), and experiments and experiment probes for deployment during the mission. Weight of these elements is important because of the effect it has on propulsive stage sizing (particularly the round trip portion). The SS-type modules shown in the MM include 2 Habitability Modules and a Laboratory/Logistics Module, as modified for the Mars mission. The MM remains in Mars orbit with a crew of 2 persons, while the MEM descends to the surface with a crew of 4, during opposition missions; all 6 crewmen would descend to the surface during a conjunction mission.

Most Spacecraft subsystems technology/designs were assumed to be the SS-type, for sizing and costing purposes. Although SS modules and subsystems are still in a very early stage of definition, it appears that a closed-loop (except for the food loop) ECLSS will be used there. The Spacecraft power source was asssumed to be a Radioisotope Thermoelectric Generator (RTG) - type (non-SS), operating at a power level of 25kw during the transit phases (MEM and MM systems active) and having 10kw for the surface phase (MEM).

The spacecraft concept shown is based on a "0-g" in-transit environment for the crew, which provides the simplest configuration approach. Several options considered for the MM are illustrated in Figures 3 and 4. Figure 3 is provided primarily to show the relative size comparison of a single module concept from reference 5 with a twin SS module concept having approximately equal volume. The volumes shown here are not adequate for the Mars mission currently being discussed. Also, the single module from reference 5 provides no safe haven volume in case of emergency. A large tunnel could be installed down the center of the single module to provide such a region. The larger-diameter module has advantages in volumetric and weight efficiency, and probably allows better utilization of the basic equipment weight for radiation shielding. However, it would be a new design, and would not allow as much cost-savings benefit as the concept which utilizes SS modules.

FIGURE 3. MISSION MODULE OPTIONS

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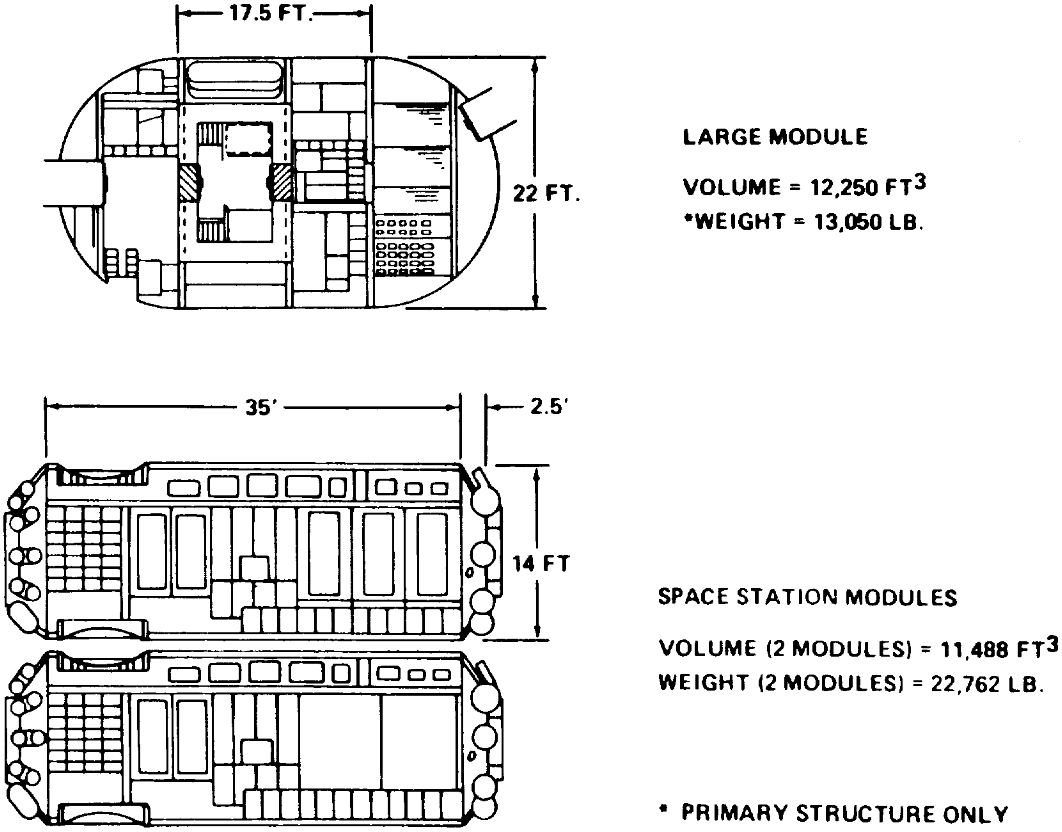

FIGURE 4. MANNED MARS HAB MODULE CONCEPT

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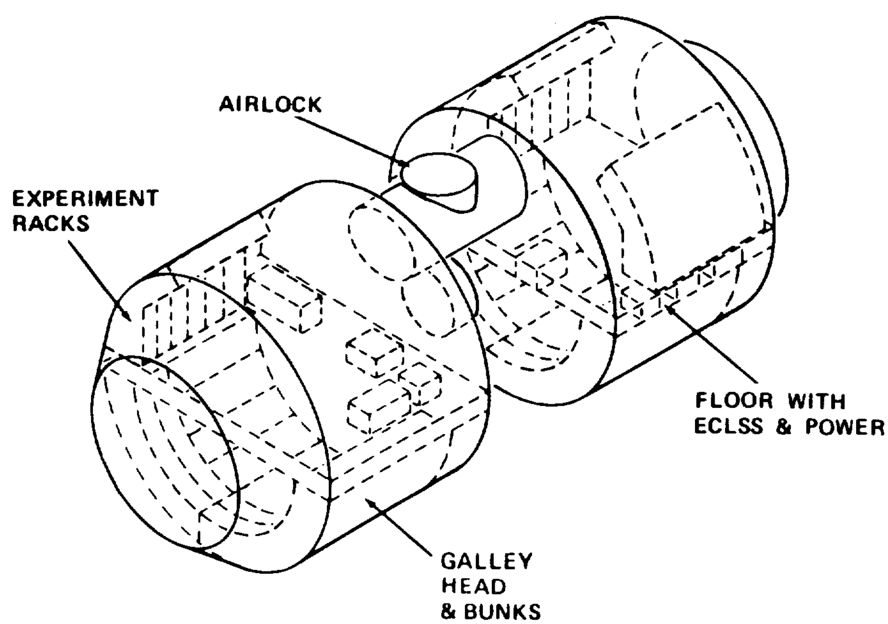

Figure 4 provides a concept which uses 2 end-to-end large-diameter modules. The modules shown here utilize a floor across their midsections which would house much of the ECLSS, power and other required equipment, leaving the cylindrical walls free for experiments, bunks, and other facilities. The EVA airlock rests between the two modules, granting access from both. For all options, it was assumed that a minimum of 2 separate pressurized compartments was necessary in case of an incident that required evacuation and isolation of an area.

As discussed later, preliminary calculations showed that the total spacecraft systems mass should be sufficient to provide adequate protection from background radiation and solar flares, if its distribution could be effected properly. Such detailed layout activity was beyond the scope of this study, but such an approach seems feasible. This consideration would necessitate packaging most of the spacecraft equipment around the walls of the pressure vessels, for maximum shielding effectiveness. Retention of expended propulsive stages during the long coast phases of the mission may also benefit the radiation protection for the crew. Packaging of fluids such as propellants (especially H_2) and water around the habitable modules would add significantly to the radiation protection, but no viable concept of this sort has been developed yet. Boil-off, tank weight, interfaces/integration, and module visibility are difficulties associated with such a concept. Figure 5 depicts the spacecraft used as a reference for this study. It provides more details on the MM concept utilizing SS modules. Three modules are required to provide the necessary volume for the Mars mission. Figure 5 also provides details of the MEM. The MEM consists of a descent stage which stays on the Mars surface and an ascent stage for return of the crew and samples to Mars orbit for rendezvous with the MM. Existing solid rocket de-orbit motors as defined in reference 2 are used for de-orbiting the MEM prior to Mars landing. An entry heat shield is provided for deceleration and protection during entry, and propulsive braking and attitude control are used for landing. The pressurized portion of the ascent stage is occupied by the 4-person crew during the Mars entry and landing. Descent engines are arranged such that one is subsequently reused for ascent. (Liquid oxygen and monomethylhydrazine (LO_2/MMH) propellant is used.) These engines are defined in reference 2, and would be a new

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FIGURE 5. MANNED MARS MISSION SPACECRAFT CONCEPT (SPACE STATION HABITAT MODULES)

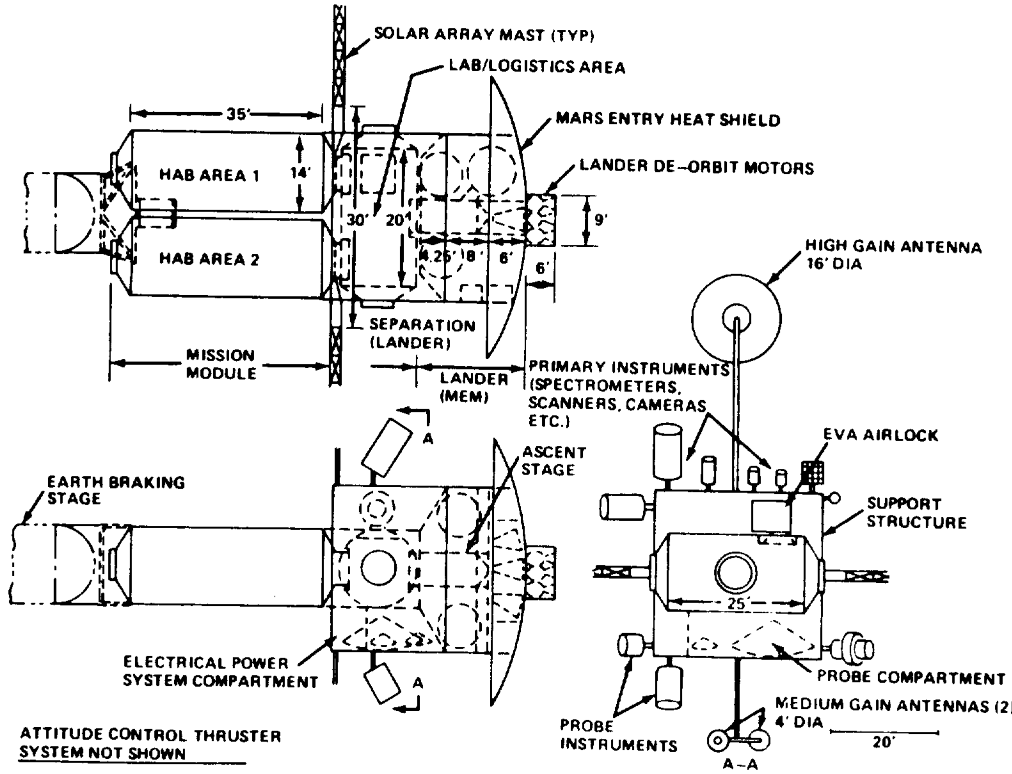

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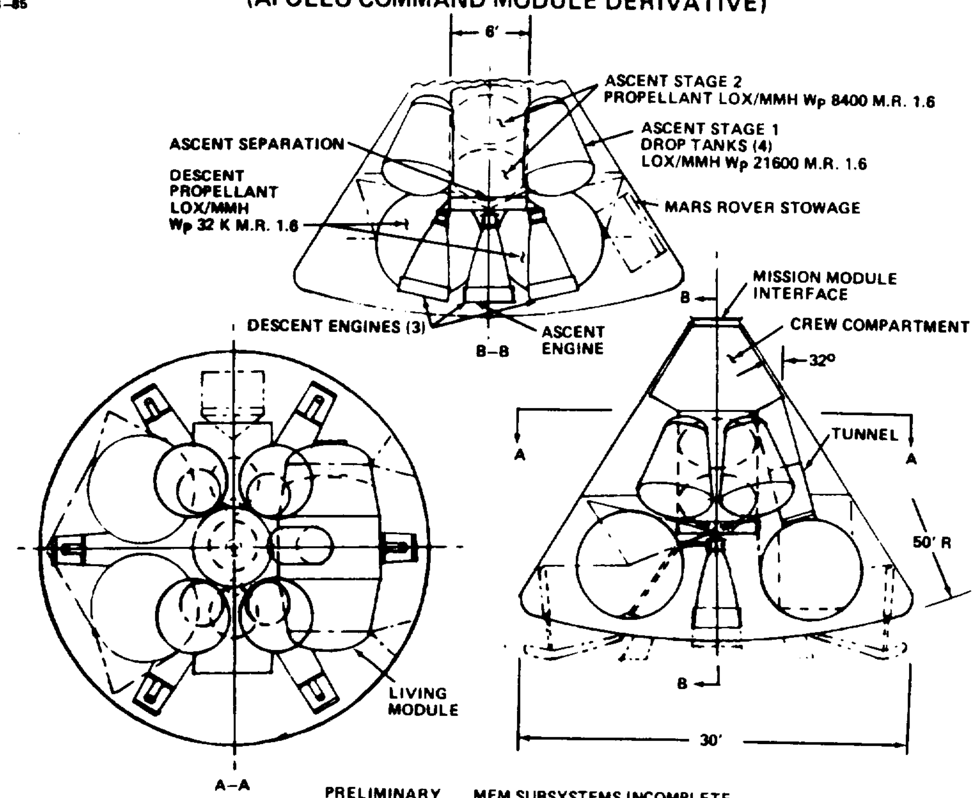
FIGURE 6 MARS EXCURSION MODULE CONCEPT (APOLLO COMMAND MODULE DERIVATIVE)



PRELIMINARY MEM SUBSYSTEMS INCOMPLETE

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design. The lander portion of the vehicle includes a pressurized crew module/laboratory, experiments, and exploration provisions (including surface mobility provisions such as a rover vehicle having power, communications, and thermal control capability). EVA capability is provided from the crew module. Upon completion of the surface mission, the crew and samples return to Mars orbit in the ascent stage, leaving most of the landed mass on the surface. After rendezvous with the orbiting vehicle, the crew and samples are transferred and the ascent stage is jettisoned prior to Mars orbit departure.

Figure 6 depicts a MEM option which is a derivation of the Apollo Command Module, and is a modified version of a concept from reference 6. This concept imposes severe packaging shape and size/weight constraints on the equipment and habitability volumes necessary to be transported to the surface, particularly that for longer-duration missions. Such a concept might suffice for very limited early missions, but would be dead-ended from a growth standpoint.

In contrast, the large aeroshell approach previously showm (Figure 5) allows implementation of a surface delivery "system" concept, wherein the aeroshell is used to accommodate small or large payloads, with minimum impact on their shape, size, or weight. A cylindrical shell is shown behind the aeroshell to serve as a heat shield, but this item may not be required.

ALL-AEROBRAKE OPTION

An all-aerobrake option of the Manned Mars Space Vehicle is shown in Figure 7 for the 2001 opportunity, using an opposition-type trajectory. This concept utilizes the same spacecraft as the all-propulsive versions, but uses aerobraking instead of propulsive braking for Earth and Mars capture. This design, therefore, uses much less propellant and has a much lower weight (discussed later) at Earth departure than the all-propulsive version. Aerobraking concepts were assumed to be derivatives of those utilized for the OTV and STS concepts. The OTV is expected to be operational in the mid-to-late 1990s.

The first stage is expended after departure from Earth and is returned to LEO (Figure 8). The propellant tanks of the first stage were sized to take advantage of current hardware; the diameter and bulkheads have commonality with the STS External Tank. The second stage can also

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FIGURE 7.
2001 OPPOSITION
MANNED MARS MISSION
AEROBRAKE OPTION

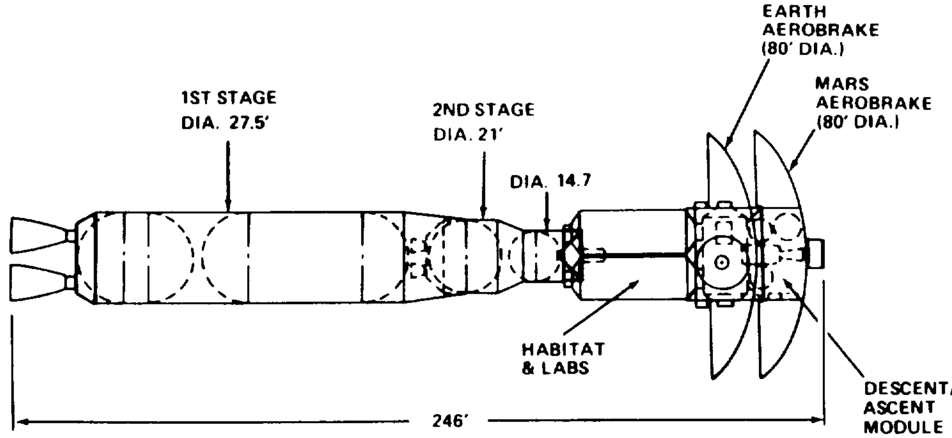

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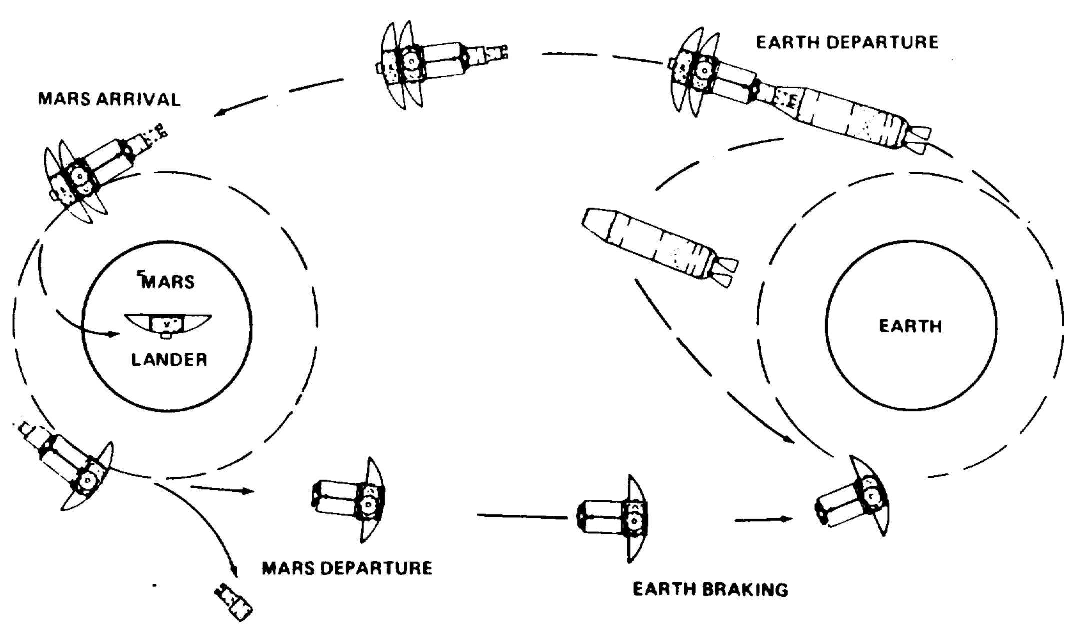
FIGURE 8.
2001 OPPOSITION
MANNED MARS MISSION
AEROBRAKE STAGING



make use of then-existing designs, specifically OTVs. Of course, both stages can grow by adjustments to their cylindrical lengths. As with the all-propulsive vehicle, the first and second stages utilize engines derived from existing (or then-existing) vehicles (Shuttle and/or SDV-3R, and OTV). An 80 ft. diameter aerobrake provides the braking for Mars arrival. This aerobrake can be jettisoned, revealing a separate 50 ft. heat shield for the MEM, or only part of the aerobrake may be jettisoned reducing it to a reusable 50 ft. diameter heat shield for the MEM. Another option is to reuse the entire 80 ft. diameter aeroshell for the MEM heat shield. A third option is to reuse the 80 ft. aeroshell for Earth braking, and provide a separate 50 ft. heat shield for the MEM.

As shown in Figure 8, once the MEM ascent stage returns to the MM in Mars orbit, the crew and cargo are transferred, and the ascent stage is jettisoned. The second propulsive stage provides Mars departure velocity and is discarded. The vehicle then attains Earth orbit with the use of the 80 ft. diameter Earth-braking shield.

HYBRID OPTION

Another option is a hybrid vehicle which uses aerobraking at Mars and then propulsive braking for Earth return (Figures 9 and 10). The same spacecraft as utilized in the other options was also used here, except as noted below. This vehicle is sized for the 1999 opportunity, using an opposition-type trajectory. Utilizing an opposition-type trajectory at this opportunity results in an energy level which will produce a high g-level if the total spacecraft is aerobraked into Earth orbit. The crew may be especially susceptible to g-level effects if they have been in a reduced-g or 0-g field for a long period of time. To keep the g-level within acceptable bounds (estimated to be about 3g to 5g) for the crew, it is necessary to do propulsive braking just prior to Earth orbit entry. However, if the entire spacecraft is propulsively braked, the addition of a fairly large 3rd stage and significant growth in the first and second stages would be required. An alternative approach, used for this concept, was to retain the MEM ascent stage, to jettison the MM near Earth, then propulsively brake only the MEM ascent stage using MEM engines or a small third stage. Once the energy level is reduced to this acceptable limit, very little additional propulsive braking would be required to brake into Earth orbit. This approach was selected rather

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FIGURE 9.
1999 OPPOSITION
MANNED MARS MISSION
AEROBRAKE OPTION

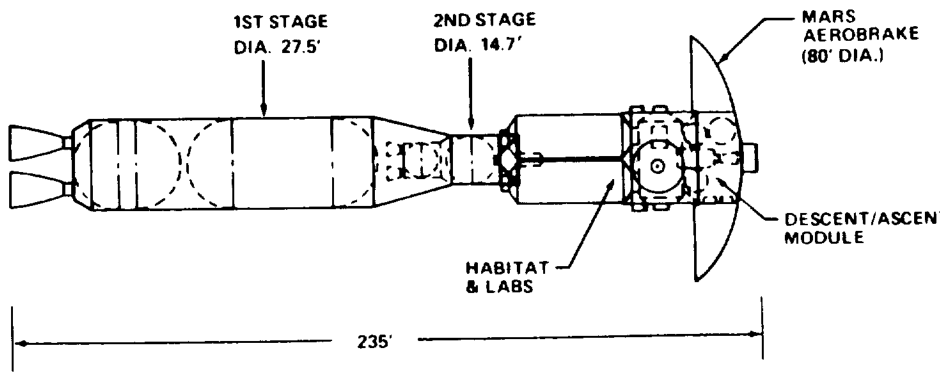

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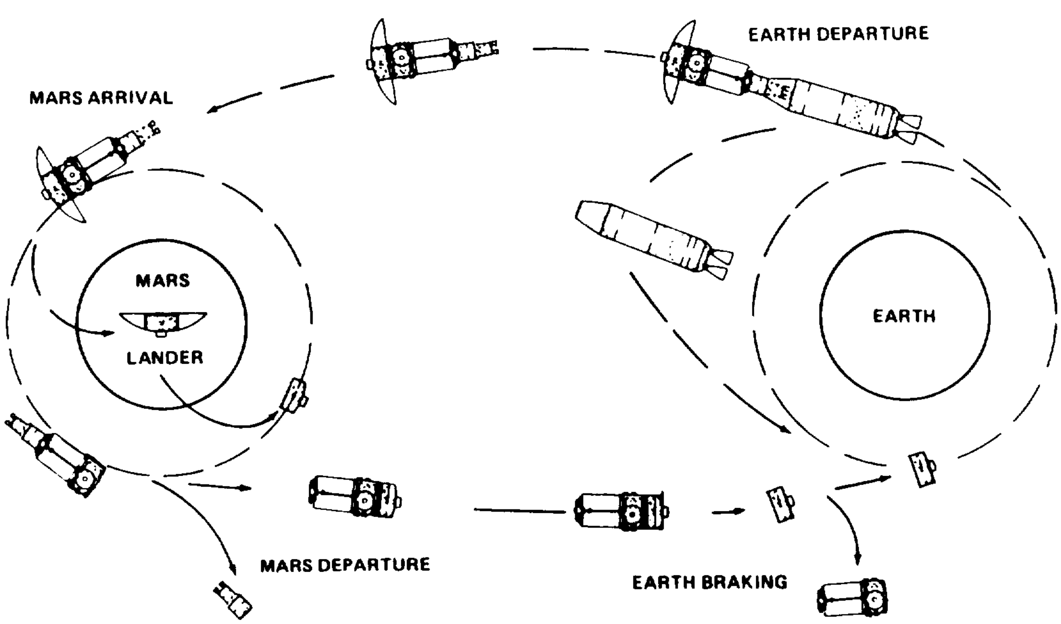
FIGURE 10.
1999 OPPOSITION
MANNED MARS MISSION
AEROBRAKE STAGING



than aerobraking for this configuration. Weights are considerably lower using this option than using the all-propulsive vehicle.

SV "SYSTEM"

The concepts described above for the 1999 and 2001 missions are summarized graphically in Figure 11. The conjunction-type missions are generally easier to accommodate configuration-wise than opposition missions. (See references 1, 7, 8 and 9). This is especially true for all-propulsive vehicles. However, the use of aerobraking concepts allows much easier accommodation of opposition missions, and allows development of a vehicle "system" which can perform either oppostion or conjunction missions at any opportunity and which can be used for manned or unmanned payloads (see references 8 and 9). About 65-70% of the opposition missions do produce acceptable g-levels when aerobraking is used at Earth.

The large aeroshells delivered to the Martian surface may provide useful structures for habitation or storage. Much of the aerobraking technology required should be developed as part of the OTV program, now in progress.

The 3-year (conjunction) missions allow a one year or so stay at Mars, which offers science benefits and may be more useful for more mature, Mars-base-era operations. However, the 2-year (opposition) missions, with their 60-day or so stay time at Mars, may be more attractive for earlier and/or simpler missions, or for unmanned cargo or other flights in the later timeframes. The "system" identified herein appears to offer a good bit of versatility to the user, for any of these applications.

The greatest contribution that the vehicle designer might make to the program is to provide a high degree of versatility to accomodate various mission and program options, at reasonable cost. Thus, an early flyby mission might be accomplished readily, and yet, the elements selected for such a mission would not be dead-ended, but would serve efficiently for follow-on exploration and utilization.

Some of the critical ingredients of such a vehicle system will be modularity and technology transparency. Vehicle designs must have multiple stages, add-on tanks, etc., to be able to accommodate greater payloads (or the same size payloads in years having less favorable opportunities), and must be able to incorporate newer technology systems as

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FIGURE 11.
AEROBRAKING OPTIONS
FOR 1999 AND 2001 OPPORTUNITIES

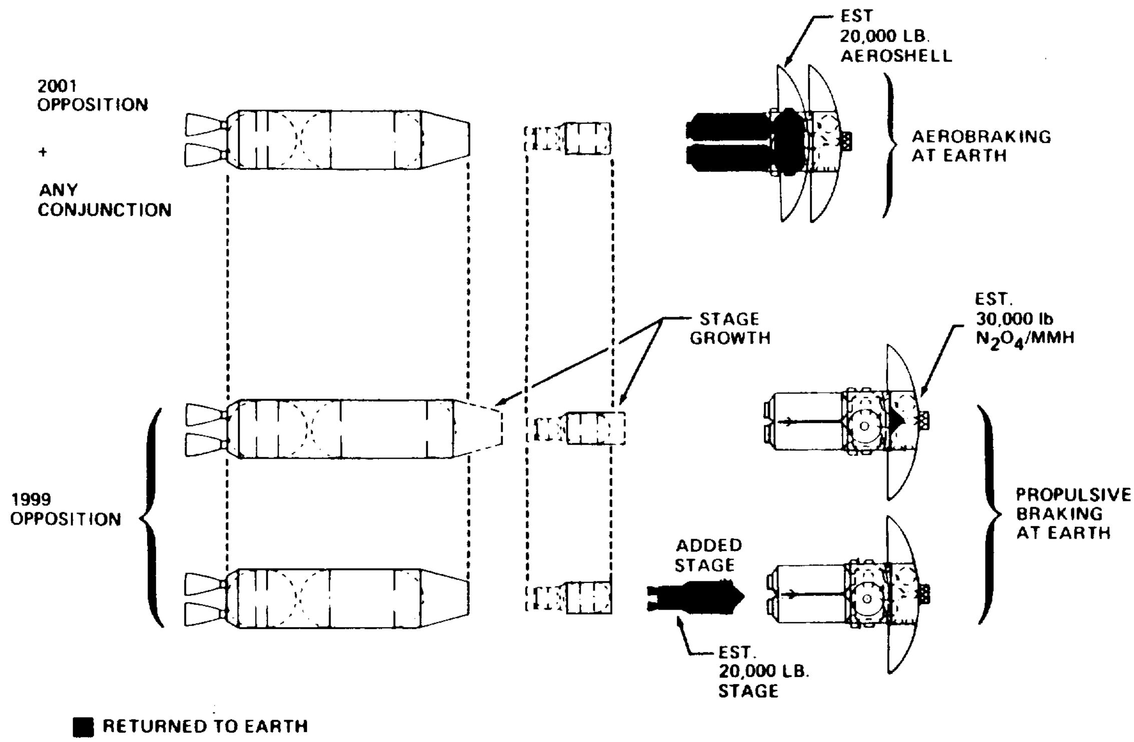

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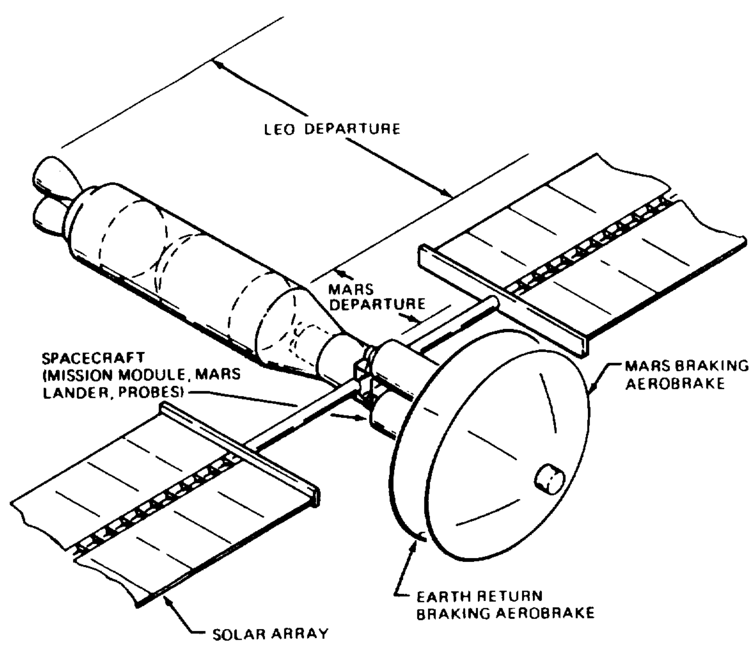
FIGURE 12.
ALL–AEROBRAKING MARS SPACE VEHICLE
UTILIZING SOLAR ARRAY



they become available with minimum impact on the rest of the vehicle. The vehicle should have adaptability to either manned or unmanned (cargo) missions, with minimum impact.

Figure 12 depicts an all-aerobraking concept which makes use of a solar array as part of the MM. The relative size of the solar array wings compared to the other elements can be seen here.

ON-ORBIT ASSEMBLY

Figure 13 dispicts one potential configuration of the SV undergoing on-orbit assembly in LEO. Here, a free-flying assembly "system" is being used, but other options range from using no assembly system to using the SS as the assembly system. References 12 and 14 provide futher discussion of assembly options. The assembly system shown here consists of a piece of the SS truss structure, including SS Attitude Control System elements and the Mobile RMS (MRMS).

GRAVITY-FIELD CONCEPTS

Some solution must be found to ameliorate the deleterious effects on the crew of long-term weightlessness. Hopefully, solutions to this problem will not require the total SV to provide a gravity field. While not impossible to do, this adds complexity to the SV which should be avoided unless absolutely necessary. If artificial-g is required, it might be acceptable to have less than 1g, but this is unknown. Configurations providing several different g-levels have been investigated, and some of these are discussed below.

Physiological constraints limit the rotation rate to a maximum of 4 RPM (reference 10). The spacecraft must thus have a radius of rotation of 200 ft. in order to obtain 1g acceleration (see Figure 14). This vehicle is based on the all-propulsive version, with the addition of two 200 ft. arms to support the MM and MEM. These arms would most likely be deployable beams such as those utilized as Space Station structure. Tunnels would probably be desired between modules, and would be a major difficulty due to their length. Environmental Control and Life Support System (ECLSS) control for the tunnels could be a significant problem. The 2 modules at the end of the 200 ft. arms must be fairly close to the same weight for good balance. The entire spacecraft or just the habitat section could be spun up, but if the entire vehicle is spun, the communication antennas, some science equipment, and possibly the solar arrays

FIGURE 13. ON-ORBIT ASSEMBLY OF MANNED MARS SPACE VEHICLE

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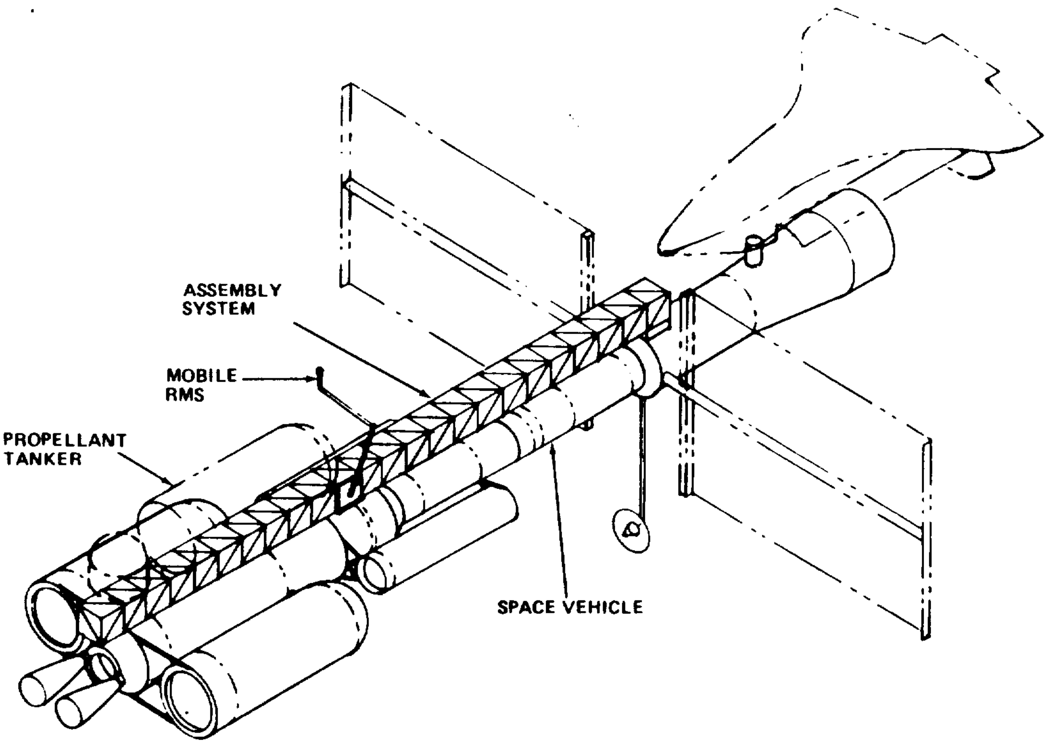

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FIGURE 14
MANNED MARS MISSION
1–G OPTION

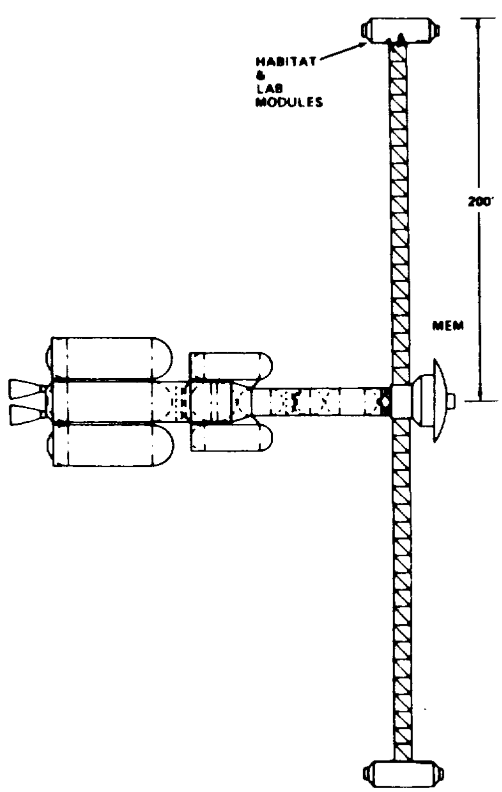
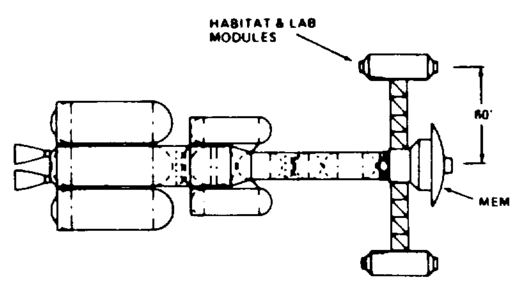


FIGURE 15
MANNED MARS MISSION
0.4–G OPTION



(if used) would have to be despun. Figure 15 depicts a vehicle option designed to generate .4 g radial acceleration. This vehicle is derived from the 1g design, the only change being the shorter 60 ft. radius of rotation.

Mass must be added to the SV for: (1) the RCS system required for spinup and maintenance of the spin rate; (2) the truss structure supporting the modules; and (3) the tunnels and their ECLSS equipment, additional shielding weight, etc.

Design and operational complexities are introduced since: (1) efficient utilization of the habitable environment is difficult due to the distances involved; (2) frequent traversing between modules would tend to produce sickness due to the varying g-levels experienced, (3) systems and living quarters would have to operate and be functional in 0g, partial g, and 1g environments, with the latter two involving two different g-force directions (ground and on-orbit); (4) some of the modules and other elements would have to be relocated to the region behind the aeroshell of an all-aerobraking concept for capture at Mars and Earth; and (5) EVA activities would necessitate stopping the rotation. The booms may have to be adjustable length-wise to balance the changing masses as the configuration changes over the two-or three-year length of the mission.

Some elements of the SV (astronomy instruments, guidance sensors, etc.) would have to be de-spun to allow their proper operation and others (appendages, etc.) would have to be stiffened to withstand the g-forces.

WEIGHTS

Weight summaries for four different manned Mars propulsion vehicles are shown in Tables 1 through 4. Propellant weights are from reference 1. Weights are included for interstages and payload adapters to connect stages together as well as for the spacecraft propulsive vehicle and crew. The number of engines in the propulsion system is shown in parentheses for each stage. The avionics weights for the propulsive stages are minimal, since the main avionics system would be in the spacecraft. A fifteen percent contingency is added to all the dry weights, since most of the hardware is new and considered to be current technology equipment. Boiloff propellants are included for the vehicle after Earth departure only, since it was assumed the propellants could be "topped off" just prior to Earth departure. The aerobrake/heat shield weight for the MEM

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TABLE 1
WEIGHT SUMMARY (POUNDS)
ALL PROPULSIVE CRYOGENIC VEHICLE FOR 2 YEAR 1999 OPPOSITION MISSION

	1ST STAGE EARTH DEPARTURE	2ND STAGE MARS ARRIVAL & DEPARTURE	3RD STAGE EARTH BRAKING
PROPELLANT TANKS	37529	13083	3650
STRUCTURES	11436	7524	2030
INSULATION & VAPOR COOLED SHIELDS	23996	12865	3520
ENGINES & PROPULSION SYSTEM	(2) 24903	(5) 6737	(2) 2293
AVIONICS (MINIMAL ONLY)	800	500	200
CONTINGENCY (15%)	14800	6106	1754
RESIDUALS	8948	4101	1278
SUBTOTAL BURNOUT WEIGHT	122412	50916	14725
BOILOFF PROPELLANTS	–	1680	704
USABLE PROPELLANTS	2265472	671420	160222
STAGE LAUNCH WEIGHT (LEO)	2387884	724016	175651
SPACECRAFT (LAUNCH)		291,203	
TOTAL SPACE VEHICLE AT LEO LAUNCH		3,578,754	

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TABLE 2
WEIGHT SUMMARY (POUNDS)
AEROBRAKING CRYOGENIC VEHICLE FOR 2 YEAR 1999 OPPOSITION MISSION

	1ST STAGE EARTH DEPARTURE	2ND STAGE MARS DEPARTURE	3RD STAGE EARTH BRAKING
PROPELLANT TANKS	21991	1404	267
STRUCTURES	14631	2150	925
INSULATION & VAPOR COOLED SHIELDS	10303	1521	773
ENGINES & PROPULSION SYSTEM	(2) 24213	(2) 1939	(2) 1773
AVIONICS (MINIMAL ONLY)	800	200	200
CONTINGENCY (15%)	10791	1082	590
RESIDUALS	4334	901	425
SUBTOTAL BURNOUT WEIGHT	87063	9197	4953
BOILOFF PROPELLANTS	–	335	105
USABLE PROPELLANTS	902938	50830	15000
STAGE LAUNCH WEIGHT (LEO)	990,001	60,362	20,058
AEROBRAKE FOR MARS ARRIVAL (80 FEET DIA)		38,893	
SPACECRAFT (LAUNCH)		291,203	
TOTAL SPACE VEHICLE AT LEO LAUNCH		1,400,517	

3302–85

TABLE 3
WEIGHT SUMMARY (POUNDS)
AEROBRAKING CRYOGENIC VEHICLE FOR 3 YEAR 1999 CONJUNCTION MISSION

	1ST STAGE EARTH DEPARTURE	2ND STAGE MARS DEPARTURE
PROPELLANT TANKS	18234	1334
STRUCTURES	14631	2150
INSULATION & VAPOR COOLED SHIELDS	8680	1470
ENGINES & PROPULSION SYSTEM	(2) 24115	(2) 1914
AVIONICS (MINIMAL ONLY)	800	200
CONTINGENCY (15%)	9969	1060
RESIDUALS	3730	876
SUBTOTAL BURNOUT WEIGHT	80159	9004
BOILOFF PROPELLANTS	–	1600
USABLE PROPELLANTS	724706	43528
STAGE LAUNCH WEIGHT (LEO)	804,865	54,132
AEROBRAKE FOR MARS & EARTH ARRIVAL (80 FEET DIA.)	38,893	
SPACECRAFT (LAUNCH)	383,510	
TOTAL SPACE VEHICLE AT LEO LAUNCH	1,281,400	

3303–85

TABLE 4
WEIGHT SUMMARY (POUNDS)
AEROBRAKING CRYOGENIC VEHICLE FOR 2 YEAR 2001 OPPOSITION MISSION

	1ST STAGE EARTH DEPARTURE	2ND STAGE MARS DEPARTURE
PROPELLANT TANKS	24381	3959
STRUCTURES	15222	2697
INSULATION & VAPOR COOLED SHIELDS	10734	3273
ENGINES & PROPULSION SYSTEM	(2) 24265	(2) 2287
AVIONICS (MINIMAL ONLY)	800	200
CONTINGENCY (15%)	11310	1862
RESIDUALS	4568	1269
SUBTOTAL BURNOUT WEIGHT	91280	15547
BOILOFF PROPELLANTS	–	705
USABLE PROPELLANTS	977280	157804
STAGE LAUNCH WEIGHT (LEO)	1,068,560	174,056
AEROBRAKE FOR MARS & EARTH ARRIVAL (80 FEET DIA.)	38,893	
SPACECRAFT (LAUNCH)	291,203	
TOTAL SPACE VEHICLE AT LEO LAUNCH	1,572,712	

is included in the MEM weights. The eighty foot reusable aerobrake weight shown for the aerobraking vehicles was estimated and includes heat tiles (Orbiter type). This eighty foot aerobrake could be constructed so that the outer section could be jettisoned and left at Mars, and the remaining part used for Earth aerobraking if a smaller aerobrake is desired.

The MEM propulsion systems are shown in Table 5 for two different concepts. The N_2O_4/MMH (storable) concept is shown as the reference and includes the descent and ascent stages. The number of engines which are included in each stage are shown in parenthesis. All three engines are used during descent to the Mars surface, but only one is used for the ascent phase of the mission. The LOX/MMH option shows a large boiloff of LOX during the 60-day stay on the Mars surface. This boiloff of LOX could possibly be used by the ECLSS or the power system if fuel cells were used, but mission time would be limited. The total MEM propulsion system weights and stage weights are shown at launch from LEO. The deorbit propulsion system (solids) are not included on this chart, but they are included with the spacecraft and payload weights in Table 7.

Preliminary weight estimates for crew consumables are provided in Table 6; totals are given for an opposition (approximately a 2-year mission). The weight summary for the spacecraft for two and three year missions are shown in Table 7; for the 3-year mission, all 6 men go to the surface. The weights are shown separately for the Habitability Module #1, Habitability Module #2, Laboratory/Logistics Module, the MEM, and the Science Probes. The micrometeoroid shield and outer insulation weights are included with the structures. An airlock weight is shown for the Lab/Log Module, and on the same line, an aerobrake/heat shield is shown for the MEM. The main avionics, power, and ECLSS are shown in the Habitability Modules and the MEM. The Lab/Log Module would be supplied power and ECLSS from the Habitability Modules. A fifteen percent contingency is included on all the dry weights, since most of the hardware is new and considered current technology equipment. Spares are included for non-structural weights at three percent per year. Further study and analysis should be done in estimating spares. Fluids, consumables, and propellants are shown separately for each module. The deorbit propulsion system includes extra propellants for limited plane changes and landing

TABLE 5

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MEM PROPULSION SYSTEM WEIGHT SUMMARY (POUNDS)

	REFERENCE N_2O_4/MMH SYSTEM		OPTIONAL LOX/MMH SYSTEM	
	DESCENT STAGE	ASCENT STAGE	DESCENT STAGE	ASCENT STAGE
PROPELLANT TANKS	287	346	305	420
STRUCTURES	700	350	700	350
INSULATION	173	187	181	222
ENGINES & PROPULSION SYS	(2) 2014	(1) 1115	(2) 1906	(1) 1055
AVIONICS (MINIMAL ONLY)	100	100	100	100
CONTINGENCY (15%)	491	315	479	322
RESIDUALS	434	294	426	306
SUBTOTAL BURNOUT WEIGHT	4199	2707	4097	2775
BOILOFF PROPELLANTS	–	–	–	7200
USABLE PROPELLANTS	34000	38400	31250	35250
STAGE LAUNCH WEIGHT (LEO)	38,199	41,107	36,347	45,225
PROPULSION SYSTEM WEIGHT (LAUNCH)	79,306		80,572	

1184–85

TABLE 6
PRELIMINARY WEIGHT ESTIMATE OF CREW CONSUMABLES (POUNDS)

1. BASIC WEIGHTS						
AMBIENT FOOD	2.6 #/MD	608D	6 MEN	9,500 #		
FROZEN FOOD	1.0 #/MD	608D	6 MEN	3,600		
CLOTHING	.2 #/MD	608D	6 MEN	700		
MISCELLANEOUS	1.6 #/MD	608D	6 MEN	5,800		
				19,600 #	19,600 #	
2. Δ FOR ORBITING CREW						
AMBIENT FOOD	2.6 #/MD	60D	2 MEN	310 #		
FROZEN FOOD	1.0 #/MD	60D	2 MEN	120		
CLOTHING (REUSABLE)	.2 #/MD	60D	2 MEN	20		
MISCELLANEOUS	1.6 #/MD	60D	2 MEN	190		
				640 #	640 #	
SUBTOTAL (STOWED IN MISSION MODULE)					20,240 #	20,240 #
3. Δ FOR SURFACE CREW						
AMBIENT FOOD	2.5 #/MD	60D	4 MEN	600 #		
CLOTHING (EXPENDABLE)	1.2 #/MD	60D	4 MEN	290		
MISCELLANEOUS	1.6 #/MD	60D	4 MEN	380		
SUBTOTAL (STOWED IN MEM)				1,270 #	1,270 #	1,270 #
TOTAL						21,510 #

MD= MAN–DAY

3303-85

TABLE 7
WEIGHT SUMMARY (POUNDS)
MANNED MARS SPACECRAFT FOR 2 & 3 YEAR MISSIONS

SUBSYSTEMS	HAB MOD # 1 (LBS)	HAB MOD # 2 (LBS)	LAB/LOG MOD (LBS)	MEM (LBS)
STR. MECHANISMS	1500	1500	1000	1500
PRESS. STRUC. (3)	5250	5250	4750	4125
SECONDARY STRUC.	1500	1500	1000	1500
MICR/INSULATION	900	900	700	470
INTERFACE STR/SHELLS	1200	1200	6800	4100
AIRLOCK/HEAT SHIELD	–	–	1500	4000
STRUCTURES SUBTOTAL	10350	10350	15750	15695
THERMAL CONTROL	1177	1177	50	1527
ELECTRICAL POWER	3000	3000	120	5475
COMM. & DATA	2027	2027	150	2220
GN&C	833	–	–	833
CREW SYSTEMS	5482	2937	4260	6545
ECLSS	7324	7324	233	2733
PROPULSION SYSTEM W/CONTIN.				6956
CONTINGENCY (15%)	4529	4022	3084	5254
SPARES (3%/YEAR) (NON–STRUCT.)	1369	1136	332	1334
SUBTOTAL (DRY)	36091	31,973	23,979	48572
FLUIDS, THERMAL	140	140	–	140
FLUIDS, ELECTRICAL	55	55	–	–
ECLSS CONSUM.	5394	5394	–	1920
CREW SYS. COMSUM.	4800	4800	9715	1140
PROPULSION DEORBIT & PLANE CHANGE CAPABILITY				7791
PROPELLANTS DESCENT & ASCENT				72004
MISSION/SCIENCE	4430	4430		1480
CREW (6)	2280			
TOTAL (LAUNCH)	53190	46792	33694	133047

SCIENCE PROBES	24480
TOTAL MISSION MODULE (LAUNCH)	133676
TOTAL MEM (LAUNCH)	133047
TOTAL SPACECRAFT (LAUNCH) 2 YEAR MISSION	291,203 LBS
ADDITIONAL MISSION/SCIENCE EQUIPMENT	10,920
ADDITIONAL CREW SYSTEMS, ECLSS, & CONSUMABLES	51,825
ADDITIONAL STRUCTURES AND SUBSYSTEMS	29,562
TOTAL SPACECRAFT (LAUNCH) 3 YEAR MISSION	383,510

site selection capability. The mission/science weights are only representative and would change as requirements are established. The crew weights include six men with flight suits. The total launch weights are for a two year mission at launch from LEO. Additional equipment, consumables, structures, and subsystems would need to be added (mostly to the MEM) for a three year mission, as shown. Shielding could be provided in the modules, mostly from the equipment and consumables shown on this chart, provided that the layout of each module is carefully done with shielding as the driving requirement. The effective thickness of aluminum for shielding of each module has been estimated to be approximately 1.5 inches for the Habitability Modules and 1.86 inches for the Lab/Log module, assuming even distribution of equipment throughout each module.

Reference 11 indicates that 1.75 inches is required. Hence, a primary challenge for spacecraft designers is to package equipment sufficiently densely, in at least a "storm shelter" region, so that no additional weight will have to be added for shielding. In addition to the SV elements, other items must be transported to LEO for the Missions to Mars. Some of these are listed in Table 8. If an assembly system is required in LEO, for the Missions to Mars, it must be transported there. Propellant which boils off during the assembly period must be placed. Assembly can last several months to a year or more, for some cases considered (see Reference 12), and boiloff can amount to half a million pounds or so, as shown in Table 8. Aerobraking vehicles, of course, would suffer much less boiloff of propellants than the all-propulsive case shown here. Ideally, the SV elements would be launched and assembled dry, then propellants would be added. This would minimize boiloff. However, to gain maximum efficiency from the ETO launch vehicles (see reference 3), the SV elements must be launched "wet", or at least partically wet.

The crew consumables used during on-orbit assembly must also be replenished, and the SV must be re-boosted occasionally in LEO to offset orbit decay and/or to maintain proper phasing with respect to the co-orbiting SS. Reference 13 discusses potential roles of the SS in more detail. If the assembly period lasts a long time, there will probably need to be a crew rotation every 3 months or so. Weights are not shown for this.

4514-85

TABLE 8
TOTAL WEIGHT * TO BE TRANSPORTED FROM EARTH TO EARTH ORBIT

● SPACE VEHICLE WEIGHT		2,701,810
– DRY	284,939	
– FLUIDS, CONSUMABLES, PROPELLANTS, ETC	2,416,871	
● ASSEMBLY SYSTEM (W/CMG'S AND MRMS)		16,000
● PROPELLANT BOILOFF REPLENISHMENT		528,258
– LEO ASSEMBLY	469,362	
– 30–DAY DEPARTURE WINDOW	58,896	
● CREW CONSUMABLES REPLENISHMENT (LEO ASSEMBLY)		32,165
– GN_2	3,752	
– FOOD, MEDICAL, PERSONAL, HOUSEKEEPING, ETC.	28,413	
● REBOOST PROPELLANT DURING ASSEMBLY		7,000
		3,285,233 LB.

* FOR 1 SPACE VEHICLE, 1999 OPPOSITION MISSION, ALL–PROPULSIVE CONCEPT

In some options, the SS may serve as the assembly system, and may also provide the crew, related resources, and possibility the reboost propellants during LEO assembly. If so, these would all be subtracted from the list of items (Table 8) that must be furnished by the Mars program separately.

REFERENCES

1. Paper in Section II of this report, entitled "Mission and Vehicle Sizing Sensitivites", by Archie Young of MSFC.
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3. Paper in Section III of this report, entitled "Earth to Orbit Launch Vehicles", by Milton Page of MSFC.
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5. Final Oral Report of the "Integrated Manned Interplanetary Spacecraft Concept Definition (IMISCD), D2-113543-3, January 1968, by the Boeing Co., NASA Contract NAS1-6774.
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7. Paper in Section II of this report, entitled, "Mars Mission Concepts and Opportunities," by Archie Young of MSFC.
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10. Space Systems Technology, edited by Regis D. Heitchue, Jr., published by Reinhold Book Corp., N.Y., Copyright 1968, pp. 148-150.
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12. Paper in Section III of this report, entitled "Manned Mars Mission ETO Delivery and On-Orbit Assembly", by B. Barisa and G. Solmon of MSFC.
13. Paper in Section IX of this report, entitled "Space Station Utilization and Commonality", by John Butler of MSFC.