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MISSION AND VEHICLE SIZING SENSITIVITIES

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ABSTRACT

Representative interplanetary space vehicle systems are sized to compare and show sensitivity of the initial mass required in low Earth orbit to one mission mode and mission opportunity. Data are presented to show the requirements for Earth-Mars opposition and conjunction class roundtrip flyby and stopover mission opportunities available during the time period from year 1997 to year 2045. The interplanetary space vehicle consists of a spacecraft and a space vehicle acceleration system. Propellant boil-off for the various mission phases is given for the Lox/LH (Liquid Oxygen / Liquid Hydrogen) propulsion systems. Mission abort information is presented for the 1999 Venus outbound swingby trajectory. transfer profile.

INTRODUCTION

This paper presents information on performance and operational requirements and their sensitivity to flyby, Venus swingby with stopover, and conjuction class missions to Mars with stopover missions. The time period considered in developing this information is 1996 to 2045. The initial mass required in low Earth orbit was determined for each launch opportunity associated with the three classes of missions. The Mars flyby is a nonstop encounter with Mars; the Venus swingby mode opposition-class mission is a mission of less duration than the conjunction class mission but only allows a short stopover time of 60 days at Mars. Conjunction class missions require longer stopover times, up to 550 days, at Mars, but require less propellant.

Information developed in this paper is not final, as configurations of the transportation vehicles are not firm. Different values of the Mission Module (MM), Mars Excursion Module (MEM), and Mars probes may appear. The important thing to note is the relative comparative values presented for the different mission modes.

ASSUMPTIONS

Pertinent assumptions used in this study are given for the departure and capture orbit parameters, propulsion stages and planetary spacecraft elements (Figure 1). The interplanetary space vehicle was assumed to be assembled in, and depart from the 270 nm altitude, 28.5 degrees inclination, Space Station circular orbit. For the all propulsive case, required interplanetary velocity increments are achieved by three propulsive stages. The first propulsion stage effects the Earth escape maneuver, the second stage brakes the spacecraft and Earth braking stage into the Mars elliptical capture orbit and effects the escape maneuver from the Mars elliptical orbit. The third propulsion stage brakes the MM into a 24-hr elliptical orbit at Earth return. Each of the three propulsion stages' mass fractions were developed using scaling equations. For the Mars aerocapture and Earth return aerobraked case, the interplanetary velocity increments are achieved by two propulsive stages. The first and second stages were used to effect the Earth and Mars escape maneuvers, respectively.

Venus swingby, outbound, inbound, or double swingby, was used to lower the energy required for the Mars opposition class missions. The Venus closest approach distance was constrained to be equal to or greater than 0.1 planet radii (330 nm).

For the conjunction class missions, type I (<180 deg) or type II (>180 deg) Hohmann transfer trajectories were used. The Mars stopover time was optimized to achieve minimum initial weight in Earth orbit.

Interplanetary trajectory parameters (launch dates, trip times, heliocentric transfer angles, etc.) have been determined which result in a minimum total initial weight to be assembled in the Space Station's orbit. The variable propulsion stages were sized using general scaling weight laws which are dependent upon propellant loading. These coefficients are input to the interplanetary trajectory shaping program. Up to six major interplanetary maneuvers can be optimized.

<u>INTERPLANETARY SPACE VEHICLE</u>

The spacecraft is made up of a MM (the living and work area for the crew), a MEM and experimenter accommodations. A number of unmanned probes and orbiters are included to complement the manned activity. Major elements of the spacecraft are interconnected by pressurized tunnels allowing shirt sleeve passage between them. A minimum crew of 6 is necessary to operate the space systems and perform a reasonable scientific exploration program.

MARS EXPLORATIONS
POST SPACE STATION MISSIONS

STUDY ASSUMPTIONS

FOR VENUS SWINGBY MISSION MODE

TIME PERIOD OF CONSIDERATION: YEAR 1997 TO 2045

PLANET DEPARTURE AND CAPTURE ORBIT PARAMETERS

EARTH DEPARTURE	CIRCULAR ORBIT ALTITUDE = 270 N. MI
MARS CAPTURE	24 HR ELLIPTIC ORBIT PERIAPSIS ALTITUDE = 270 N. MI
MARS ESCAPE	24 HR ELLIPTIC ORBIT PERIAPSIS ALTITUDE = 270 N. MI
EARTH CAPTURE	24 HR ELLIPTIC ORBIT PERIAPSIS ALTITUDE = 270 N. MI

HELIOCENTRIC PROFILE

VENUS SWINGBY MODE (OUTBOUND, INBOUND OR DOUBLE SWINGBY)

VENUS MINIMUM CLOSET APPROACH EQUAL 0.1 PLANET RADII (330 N. MI)

CONJUNCTION CLASS MISSION USES TYPE I OR TYPE II TRAJECTORIES

INTERPLANETARY SPACE VEHICLE (1)

			MARS FLYBY	OPPOSITION MISSION	CONJUNCTION MISSION
SPACECRAFT:	MISSION MODULE WEIGHT	=	88,500 (2)	113,633	160,000
	MARS EXECURSION MODULE WEIGHT	=	N/A	133,047	133,047
	PROBES WEIGHT	=	20,000	44,523	60,000

PROPULSION STAGES	FIRST STAGE	SECOND STAGE	THIRD STAGE
MASS FRACTION (λ)	S. EQ.	S. EQ.	S. EQ.
I_{SP} (SEC)	461	482	482
PROPELLANT	LOX/LH_2	LOX/LH_2	LOX/LH_2
I_{SP} (SEC)	331.6	345.4	345.4
PROPELLANT	N_2O_4/MMH	N_2O_4/MMH	N_2O_4/MMH

(1) POST YEAR 2000 USES DIFFERENT WEIGHTS THAN THESE

(2) INCLUDES A 7,500 LB EARTH RETURN MODULE

FIGURE 1

Two interplanetary space vehicle configurations for the opposition class mission via an outbound Venus swingby for the year 1999 opposition opportunity are given in Figure 2. Information for each of the propulsion stages and the total interplanetary vehicle weight is given. The total initial mass required in the Space Station orbit for the all propulsive configuration is 3,575,321 lb; for a configuration that utilized aerobraking at Mars capture and Earth return, the total initial mass required in the Space Station orbit is 1,433,294 lb.

Earth return with aerobrake entry has been analyzed and results show that with an Earth return C_3 greater than 25 km^2/sec the g-load will be in excess of 5 g's. This high g-load probably cannot be tolerated by the crew. Earth return with C_3 greater than 25 km^2/sec^2 will require propulsive braking in order to stay within the g-load constraint.

MISSION AND VEHICLE SIZING SENSITIVITY

In mission profile design and vehicle sizing there are many variables which influence the resultant mission profile and space vehicle configuration. Some of the more significant variables include: (1) Earth launch window duration, (2) Stay time at Mars, (3) MM weight, and (4) MEM weight, including Mars lander capsule weight and Mars ascent capsule weight.

Sensitivities to the Earth launch window duration and Mars stay time for the 1997 and 1999 opportunity Venus swingby mission profiles is given in Table 1. For the 1997 opportunity, a 40 day stay time at Mars and an Earth launch window of 10 days requires 1,591,700 pounds initial weight in low Earth orbit to perform the mission. A 60 day stay time and an Earth launch window of 30 days will require 1,949,700 pounds of initial weight in low Earth orbit to perform from the 1997 launch opportunity; this weight is an increase of 22 1/2 percent over a 40 day stay time and 10 day launch window case. The 1999 launch opportunity is not as sensitive as the 1997 opportunity. A 60 day stay time and a launch of 30 days requires an initial weight in low Earth orbit of 1,434,200 pounds; this weight is an increase of 6 3/4 percent over a 40 day stay time at Mars and a 10 day Earth launch window.

The interplanetary space vehicle sensitivity to changes in MM and MEM weight is shown in Table 2 for an aerobrake (at Mars capture and Earth return) space vehicle configuration. An initial weight of

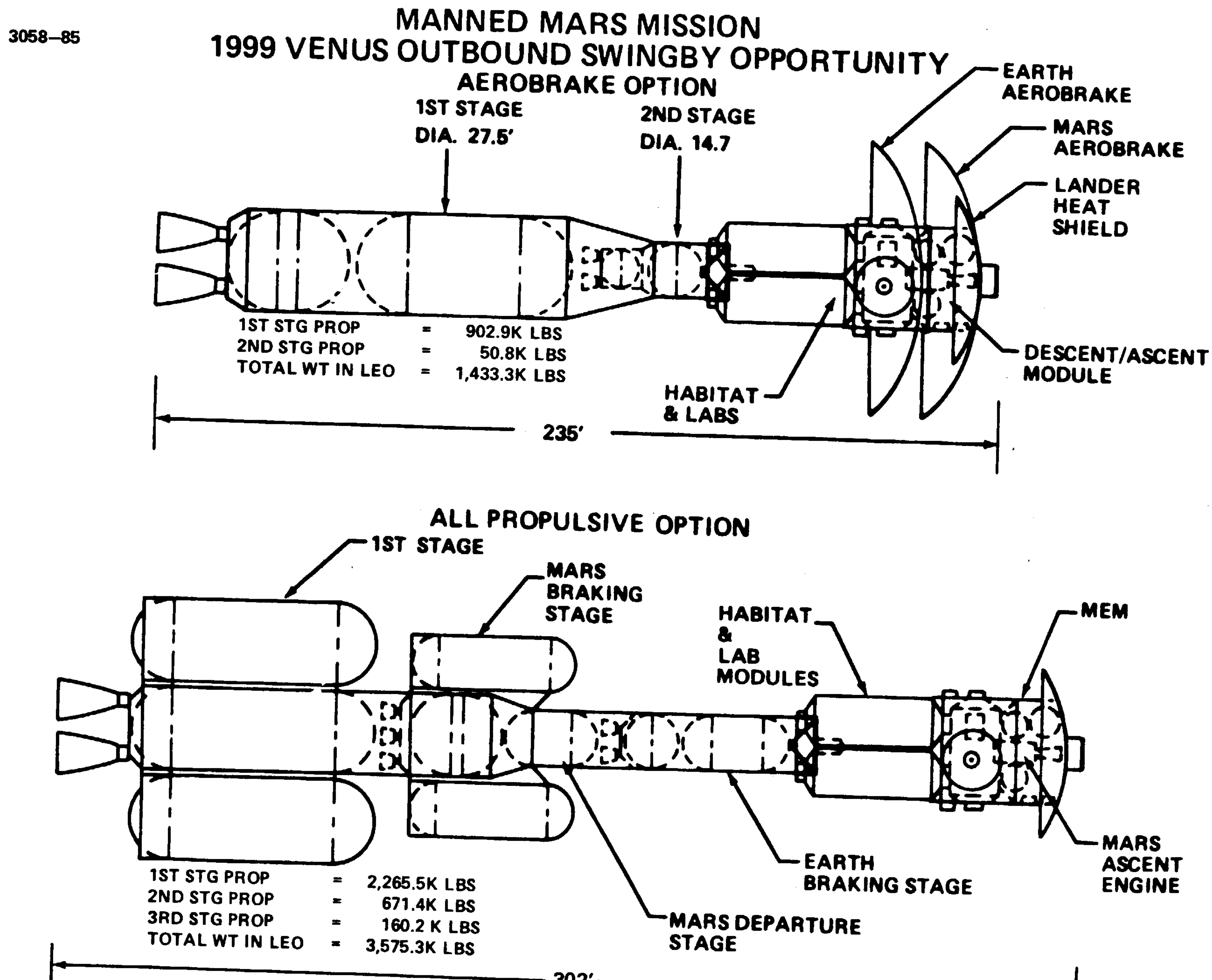

FIGURE 2 INTERPLANETARY VEHICLE CONFIGURATION

MANNED MARS EXPLORATION
ALL CHEMICAL PROPULSION MANEUVERS
AEROBRAKE AT MARS AND EARTH RETURN
WEIGHT REQUIRED IN 1000'S OF LBS

DOUBLE SWINGBY FOR 1997 OPPOSITION OPPORTUNITY

EARTH LAUNCH WINDOW → / MARS STAY TIME ↓	10	30
40	116.6 (1)	125.8
	1008.5 (2)	1115.5
	1591.7 (3)	1712.0
60	179.5	200.0
	1129.4	1270.4
	1782.6	1949.7

OUTBOUND SWINGBY FOR 1999 OPPOSITION OPPORTUNITY

EARTH LAUNCH WINDOW → / MARS STAY TIME ↓	10	30
40	60.0	60.0
	825.3	901.1
	1343.2	1421.5
60	64.6	64.6
	833.3	908.8
	1356.3	1434.2

(1) 2ND STAGE PROPELLANT
(2) 1ST STAGE PROPELLANT
(3) TOTAL WEIGHT IN EARTH ORBIT

TABLE 1

3062–85

MANNED MARS EXPLORATION
1999 VENUS OUTBOUND SWINGBY OPPORTUNITY
WEIGHT SUMMARY (IN LBS)
AEROBRAKE AT MARS AND EARTH RETURN

	NOMINAL CASE	Δ MM BY 15%	Δ MEM BY 15%
2 STAGES DRY WEIGHT	89,529	91,739	90,779
2 STAGES PROP WEIGHT	974,064	1,037,407	1,012,306
MISSION MODULE WEIGHT	134,585	154,773	134,585
MARS EXCURSION MODULE WEIGHT	137,881	137,881	158,563
PROBE & MARS AEROBRAKE WEIGHT	98,156	98,156	98,156
TOTAL WEIGHT IN LEO	1,434,215	1,519,956	1,494,389

TABLE 2

3061–85

MANNED MARS MISSION

EXCHANGE FACTOR

TYPE PROPELLANT	$\frac{\partial \text{ MEM WT}}{\partial \text{ LANDER WT}} \left(\frac{\text{LB}}{\text{LB}}\right)$	$\frac{\partial \text{ MEM WT (LB)}}{\partial \text{ ASCENT WT (LB)}}$
N_2O_4/MMH	1.37	8.30
LOX/MMH	1.33	7.08

TABLE 3

1,434,215 pounds is required in low Earth orbit for the nominal case. If the MM weight is increased by 15 percent, the initial weight in low Earth orbit is increased by 6 percent over the nominal case. If the MEM weight is increased by 15 percent, the initial weight is an increased by 4.2 percent over the nominal case.

The MEM initial weight sensitivity to variation in Mars lander capsule and Mars ascent capsule weights is given in Figure 3. The exchange factors are given in Table 3 for two different types of propellants, N_2O_4/MMH and LOX/MMH.

The initial mass required in low Earth orbit for each mission opportunity is given in Table 4 and Figures 4 and 5. The initial mass required ranges from 1,280,001 to 3,575,321 lb for LOX/LH propellant. These values do not include propellant boiloff in low Earth orbit during orbital assembly time. The initial mass required in low Earth orbit for the 1999 opposition outbound Venus swingby using N_2O_4 /MMH propellant is 8,869,090 lb. The initial mass in Earth orbit can be equated to cost and used to determine the most favorable mission opportunities and the most effective type of propellant for the propulsive stages.

PROPELLANT BOILOFF

The Mars mission is characterized by different mission environments including LEO buildup, interplanetary transit, and Mars orbit. The passive thermal protection on the cryogenic propellant tanks consists of 1 to 4 inches of MLI on the first stage and 4 inches MLI on the second and third stage tanks. Vapor cooled shields are utilized on all tanks. Table 5 relates cryogenic boiloff rate ranges for the different mission environments using the all cryogenic vehicle configuration.

The boiloff rates for LEO were calculated with 1 and 4 inches of MLI on the first stage. The boiloff rates in LEO are relatively high due to large tank areas and albedo (reflected thermal energy from the Earth) heating effects. The interplanetary transit mission phase is characterized by relatively low boiloff because of reduced vehicle tank area (stages two and three) and lower environmental heating. The lower heating is contributable to transit vehicle orientation during flight to minimize solar flux on tank wall areas and greater distance from the Earth. Any deviation from the preferred orientation will result in increased boiloff. The Mars orbit stages experience medium boiloff rates

1567—85

MARS EXPLORATION MEM DESCENT AND ASCENT FOR 165° INCLINATION SENSITIVITY DATA

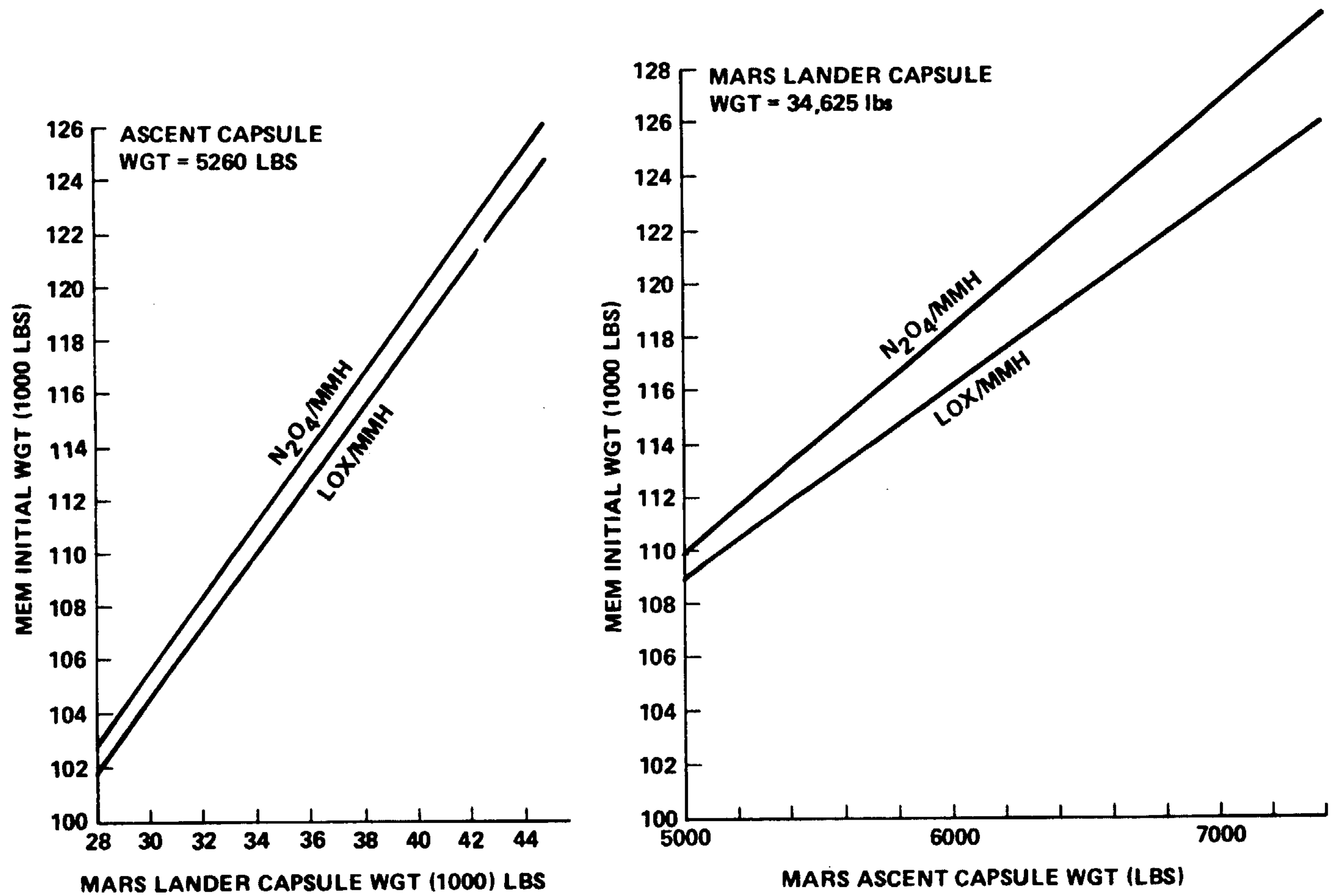

FIGURE 3

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MARS EXPLORATION
MASS IN EARTH ORBIT REQUIREMENTS

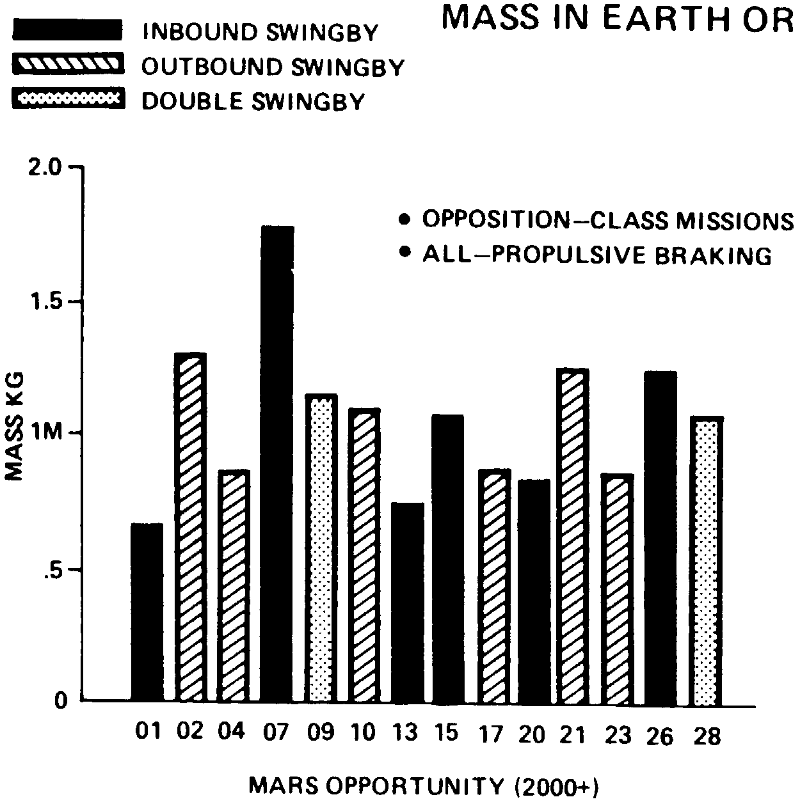

FIGURE 4

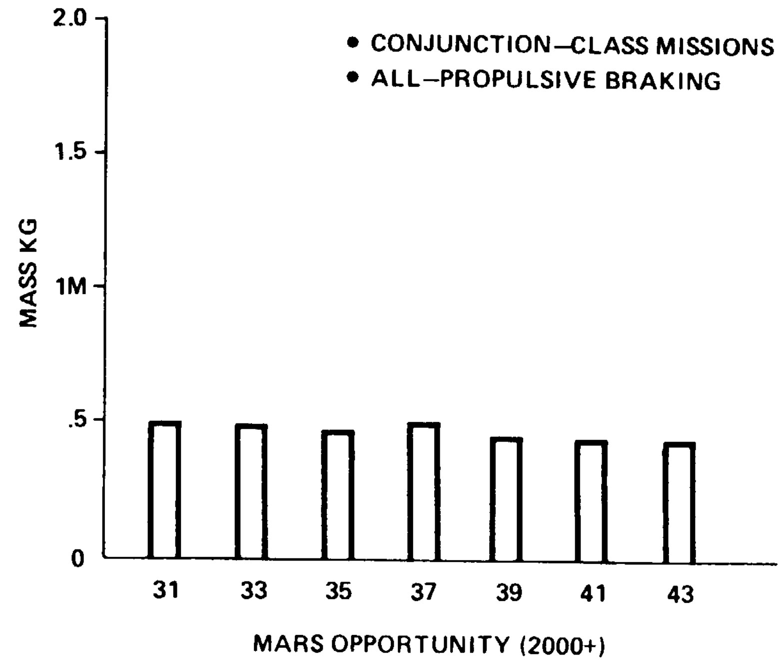

FIGURE 5

MANNED MARS EXPLORATION
PROPULSION STAGE SIZE AND LEO WEIGHT REQUIRED

MISSION/ VEHICLE	ΔV EARTH ESCAPE (KM/SEC)	ΔV MARS CAPTURE (KM/SEC)	ΔV MARS ESCAPE (KM/SEC)	ΔV EARTH RETURN (KM/SEC)	PROPELLANT LOADING (1000 LBS) 1ST STAGE	2ND STAGE CORE	2ND STAGE TANKS	3RD STAGE	TOTAL WEIGHT IN LEO (1000 LBS)
1999 ONE YR FLYBY ALL PROPULSION	7.0690	0	0.406	6.1636	1,075.2 (1)	15.9	N/A	45.8 (2)	1,346.4
1999 VENUS SWINGBY ALL PROPULSION	4.4289	2.7569	1.6238	3.7246	2,265.5	594.7	76.8	160.2	3,575.3 (3)
1999 VENUS SWINGBY AERO CAPTURE @ MARS AND EARTH	4.4289	0	1.6238	0	902.9	50.8	N/A	N/A	1,433.3
1999 CONJUNCTION ALL PROPULSION	3.7169	1.4365	0.9294	1.1135	1,004.9	211.2	26.4	47.6	1,764.7
1999 CONJUNCTION AEROCAPTURE @ MARS AND EARTH	3.7169	0	0.9294	0	724.7	43.7	N/A	N/A	1,280.0
1999 VENUS SWINGBY ALL PROPULSION (4)	4.4289	2.7569	1.6238	3.7246	6,673.2	1,436.0	N/A	262.6	8,869.1
2001 VENUS SWINGBY ALL PROPULSION	4.1535	1.6744	3.2567	1.4963	1,483.3	426.6	37.5	48.1	2,431.6
2001 VENUS SWING BY AERO CAPTURE @ MARS AND EARTH	4.1535	0	3.2567	0	977.3	157.8	N/A	N/A	1,602.0

(1) THIS VALUE COULD BE REDUCED CONSIDERABLE BY OPTIMIZING MISSION TIME

(2) EARTH RETURN WITH RETURN CAPSULE ONLY

(3) EQUALS 3,772.1 FOR REUSABLE FIRST STAGE

(4) N_2O_4/MMH PROPELLANT ALL OTHER CONFIGURATIONS USES LOX/LH_2 PROPELLANT

TABLE 4

MANNED MARS MISSION
1999 VENUS OUTBOUND SWINGBY
TOTAL VEHICLE BOILOFF RANGES ~ LB/HR

ENVIRONMENT	BOILOFF RANGE	
	LH_2	LO_2
LOWER EARTH ORBIT – 1 IN MLI ON FIRST STAGE	16.2 ~ 18.3	10.4 ~ 12.6
LOWER EARTH ORBIT – 4 IN MLI ON FIRST STAGE	6.8 ~ 9.0	3.8 ~ 5.0
*VEHICLE TRANSIT	.06 – .10	.04 – .08
MARS ORBIT	.7 ~ 1.0	.5 ~ .8

ASSUMPTIONS

- ALL CRYO CONFIG.
- STAGE 1 W_p = 2266K
- STAGE 2 W_p = 671.4K
- STAGE 3 W_p = 160.2K
- NO ACTIVE TPS
- VCS UTILIZED
- 4 IN. MLI ON 2ND & 3RD STAGES

* BASED ON PREFERRED VEHICLE ORIENTATION. DEVIATION FROM ORIENTATION WILL RESULT IN INCREASED BOILOFF RATES.

TABLE 5

through potential environmental heating due to vehicle orientations driven by mission requirements. Preferred orientation in Mars orbit to reduce environmental heating would lower the boiloff rate.

1999 VENUS OUTBOUND SWINGBY MISSION ABORT

In the final selection of trajectories for the manned Mars stopover missions, many factors other than vehicle weight must be considered. Abort capability of the vehicle is one of these factors. It is, therefore, necessary to plan and prepare for the possible irreparable failures at some point during the mission.

Abort situations can be characterized as occuring in two different phases of the mission which can be defined as (1) Earth departure phase and (2) Heliocentric orbit phase. If abort maneuvers are executed within 30 minutes after trans-Venus injection, return to low Earth orbit can be achieved within two days. The interplanetary vehicle is within Earth's gravity sphere of activity up to 1 3/4 days after trans-Venus injection; if abort maneuvers are undertaken within this time span, an elliptical orbit return to low Earth orbit can be achieved within 18 days.

Heliocentric orbit phase is reached 1 3/4 days after trans-Venus injection. The interplanetary vehicle (aero capture at Mars and aero brake at Earth return) delta V capability after trans-Venus injection is in excess of 9 km/sec for a small Earth return capsule; the Mars excursion module has a 7.2 km/sec delta V capability and the second stage main propulsion system has a 1.6 km/sec delta V capability with the total mission module weight of 113,633 lb. If mission abort is executed sometime less than 40 days into the mission, an Earth return rendezvous trajectory can be achieved which returns back to low Earth orbit within 80 to 250 days. After 180 days into the mission, the interplanetary vehicle is committed to a Mars flyby which would return to Earth in 560 days.

The above description of recovery from orbit conditions emphasized minimum delta V requirement for the return to Earth trajectory. Other abort situations (i.e., abort after 40 days, braking into orbit at Mars, Mars landing, Mars escape, etc.) need to be studied in more detail.

CONCLUSION

Comparative and sensitivity data have been developed for an opposition class Mars flyby and 60 day stopover missions to Mars. Also, data were developed for conjunction class stopover missions. The 60 day stopover mission utilized the Venus swingby mode in order to reduce the propulsive energy required.

There is a great variation in initial mass required in low Earth orbit for the all propulsive interplanetary space vehicles over a number of mission opportunities. This variation is due to the eccentricity of Mars orbit which has a perihelion distance of 1.38 A.U. and an apahelion distance of 1.66 A.U. The wide variation in initial mass may be reduced by aerocapture at Mars and Earth return or by only returning to Earth capture orbit with a small Earth return module and leaving the heavier Mission Module in an Earth-Mars periodic orbit. The variation in initial mass for the conjunction class mission over a number of mission opportunities is relatively small because there is more freedom to optimize the outbound transfer to Mars and the return transfer to Earth.

Mission abort capability, for the 1999 Venus outbound opportunity, can extend out to 40 days after trans-Venus injection. In order to minimize required weight in low Earth orbit, 4 inches of MLI on all stages seems to be the most effective.

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