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CONCEPT FOR A MANNED MARS FLYBY

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ABSTRACT

A concept is presented for a three man crew to fly by the planet Mars. The groundrule for the study is to execute the mission as quickly as possible which dictates using late 1990's technologies and space infrastructure. The proposed mission described herein uses a preliminary concept for the agency's Manned Orbit Transfer Vehicle (MOTV) and proposed Space Station elements. The space vehicle will depart from the LEO Space Station and is delivered to Low Earth Orbit (LEO) by a future launch vehicle of a Shuttle Derived Launch Vehicle (SDV) class. The trajectory parameters are chosen such that the mission duration is on the order of one year, with a two and one-half hour period within ten planetary radii of Mars. If the issues of acceptable crew "g" loads and entry vehicle heat load can be resolved, then the returning vehicle can aerobrake at Earth into a Space Station compatible orbit. Otherwise, a propulsive maneuver will be required to reduce vehicle velocity prior to Earth entry interface. It is possible to execute a mission of reasonable capability at an initial LEO departure weight of 716,208 pounds for the aerobraked case of 1,350,000 pounds for the propulsive case.

INTRODUCTION

The collection of rationales for a manned Mars mission divides into categories of: (1) science and exploration, (2) the manifest destiny of man in space, (3) the benefits of technology spin-offs, and (4) geopolitical issues such as national pride and prestige. A manned flyby mission is a mission that principally responds to the last category; probably such a mission would arise in an atmosphere of competition with the Soviets in a race in the geopolitical arena where the prize is an addition to the trophy case of national pride and prestige. Although the intangible benefits can be significant, a flyby mission should be carefully balanced between the perceived "value" of national pride and prestige, the value of the scientific return, mission costs, mission timeliness, and usability of the hardware for follow-on missions. Timeliness is addressed in reference 1 and indicates that the preponderence of the evidence, as based on the activities within the Soviet Union, point to a Soviet manned Mars flyby mission in the late 1990's. If the U. S. is to respond, existing or near term, vehicles and space infrastructure must be used in order to save, or at least share, development costs and assemble and execute a mission as quickly as possible.

ASSUMPTIONS

Transfer Vehicle

The civilian space agency is in the early phases of defining the next generation of vehicles for space transportation. In pursuit of this goal, the NASA Marshall Space Flight Center is managing the Phase A studies for the Orbit Transfer Vehicle (OTV). One of the competing vehicle configurations under study is a manned OTV of lunar and geosynchronous delivery capability. A description of this vehicle is given in NASA technical memorandum number (TM) 58264 (reference 2). This vehicle is adopted as the basic transportation unit for the Manned Mars Flyby Mission described herein. Figure 1 is an artist's concept of this vehicle at the Space Station with a Mars sample return mission payload being attached. Figure 2 is a sketch of the vehicle. Additional "drop-tanks" will be required in order to increase the propellant capacity of the vehicle. These tanks, and possibly some advanced power systems, are the only unique developments for this mission as outlined herein.

Mission Module

The concept for the Mission Module (MM) was taken from a Lunar Base Study performed by the Johnson Space Center. The basis for the data in that report came from NASA TND-6349. The MM is a Space Station derivative and is fully equipped with life support systems, health maintenance facilities, galley, and sleeping areas. It will contain, or have attached to it, a solar flare storm shelter scaled down from the design in reference 4 and contains an assumption that 1/2 the required shielding is contained within the vehicle mass. Figures 3 and 4 are from the Lunar Base Study and define the mass and geometry of the MM.

Command Module

The Command Module (CM) proposed for the flyby mission is based on a design for a manned geosynchronous sortie vehicle. The conceptual design for the geosynchronous CM was accomplished by the Johnson Space Center

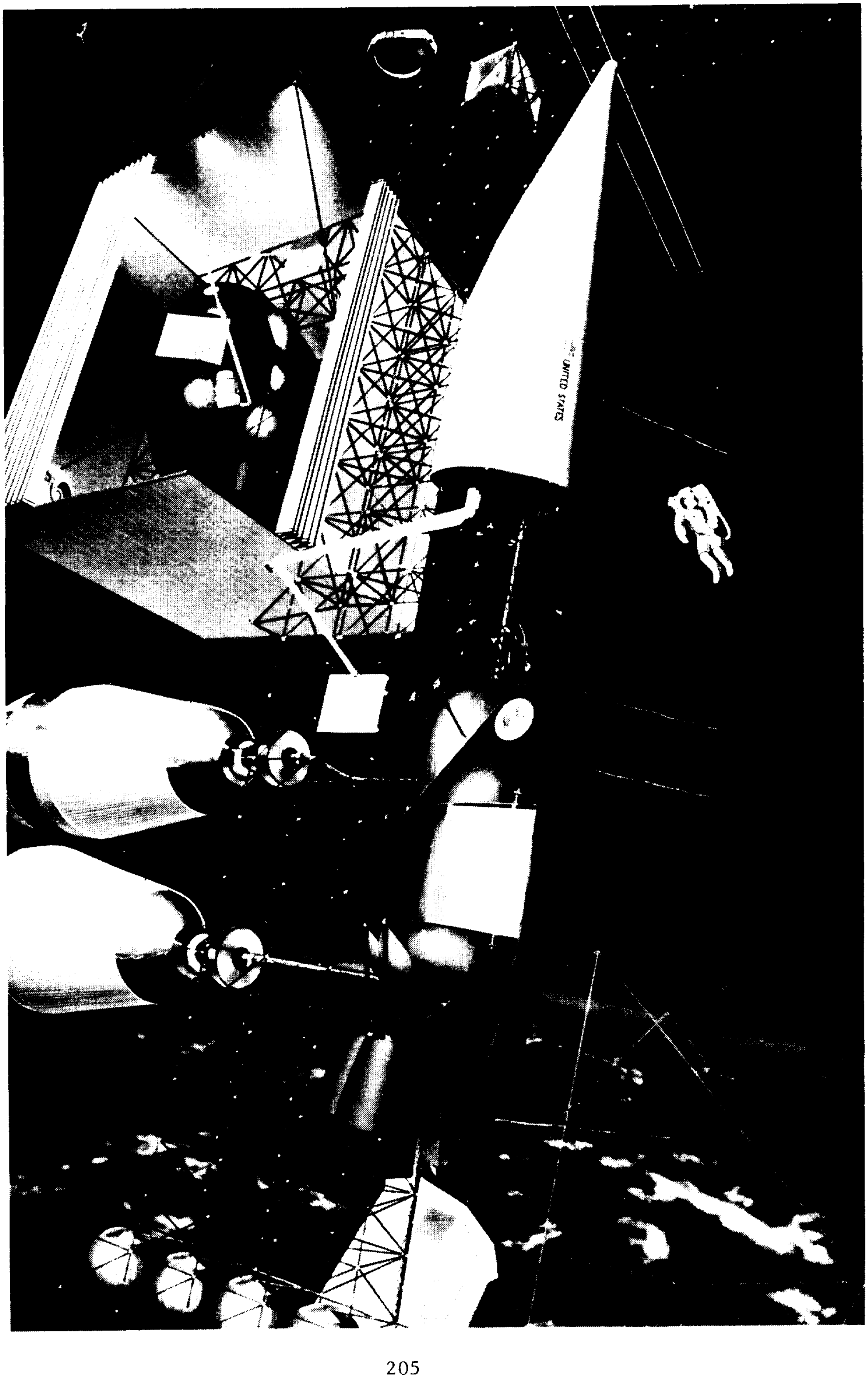

FIGURE 1 - ORBIT TRANSFER VEHICLE BEING MATED WITH A MARS SAMPLE RETURN MISSION

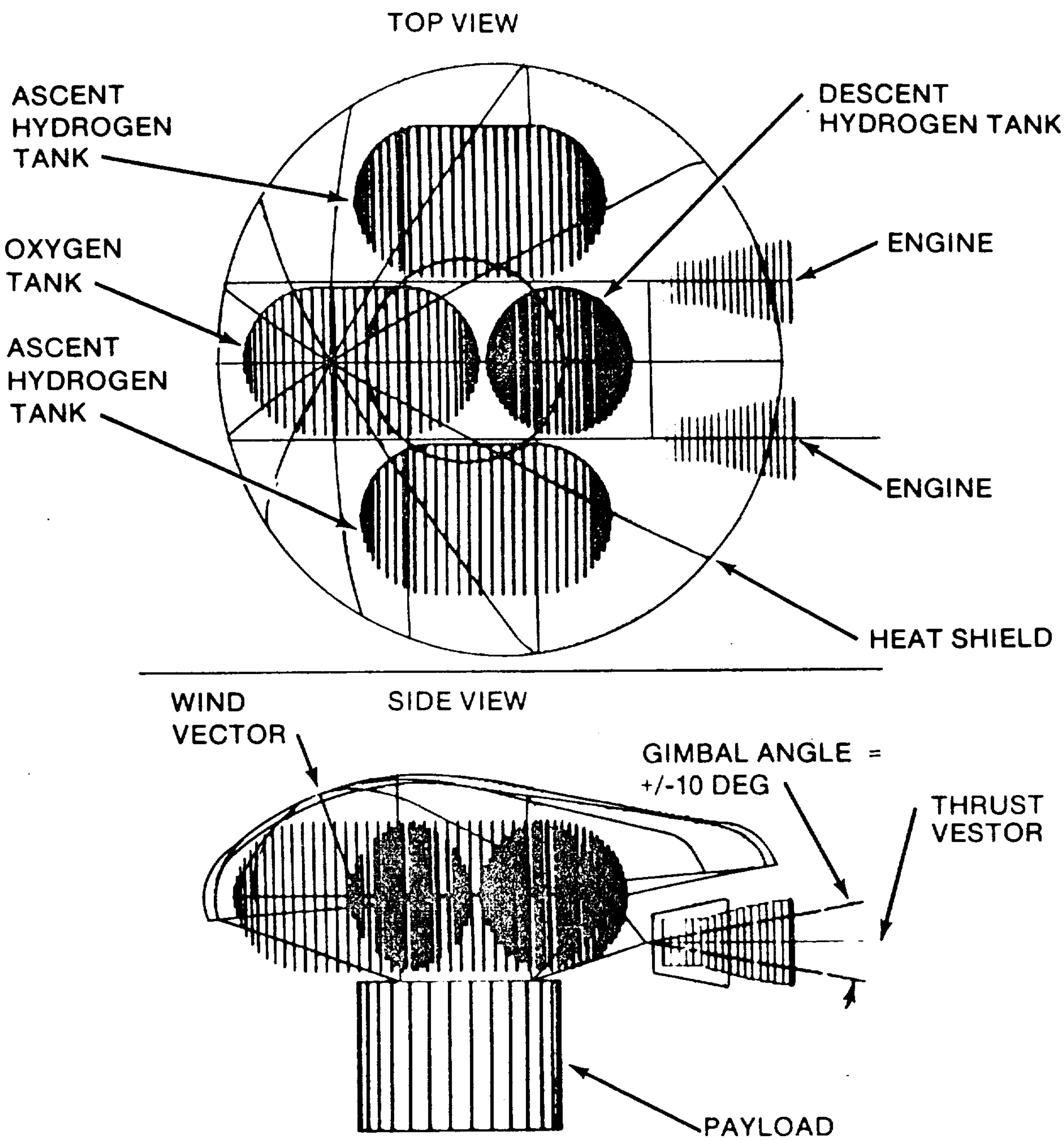

Figure 2. - Integrated AOTV concept with 12-m (40-ft) diameter heat shield.

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FIGURE 3 - MODULE SIZING	ADVANCED PROGRAMS OFFICE (ED13)	
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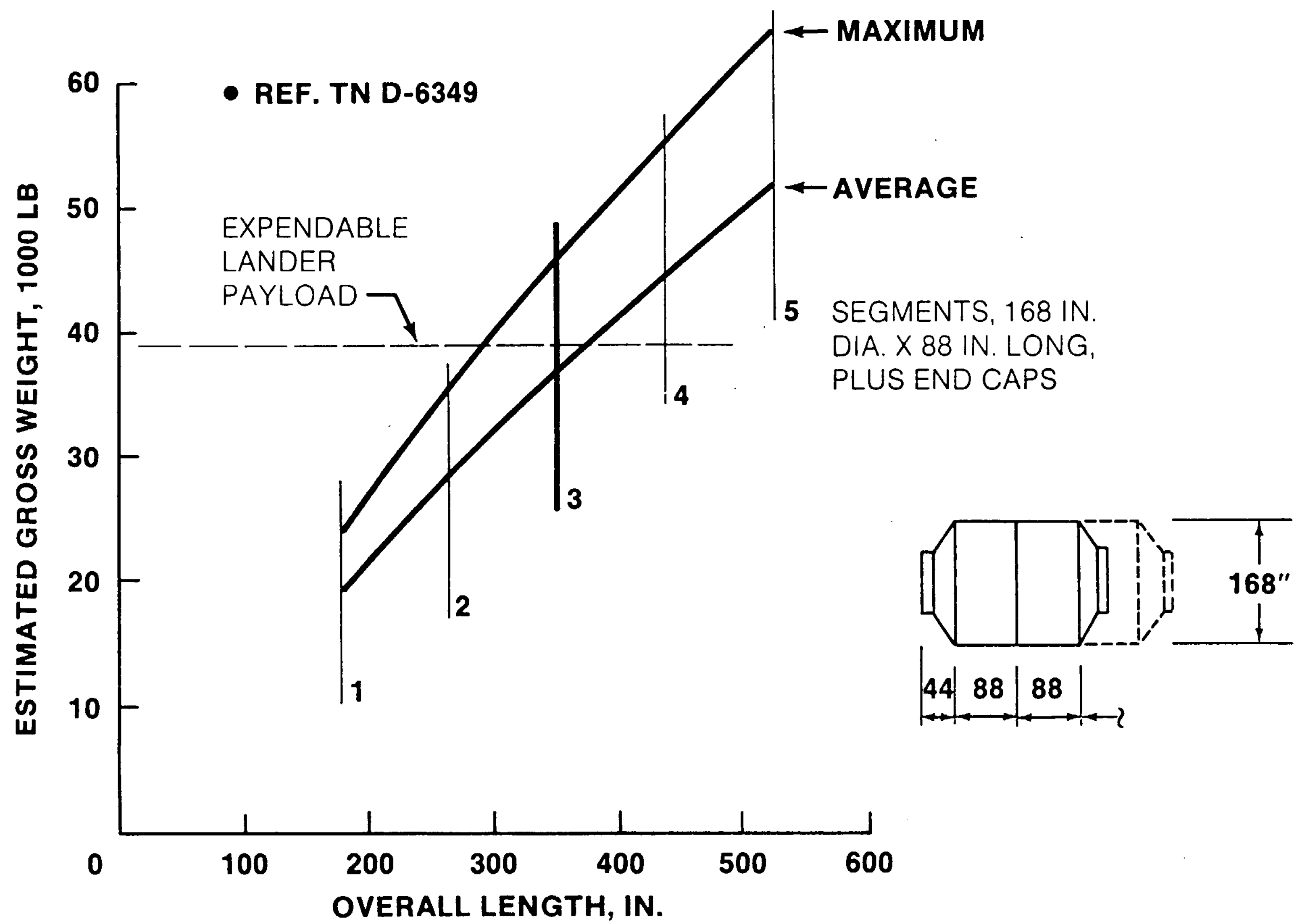

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FIGURE 4 - MODULE SIZING	ADVANCED PROGRAMS OFFICE (ED13)	
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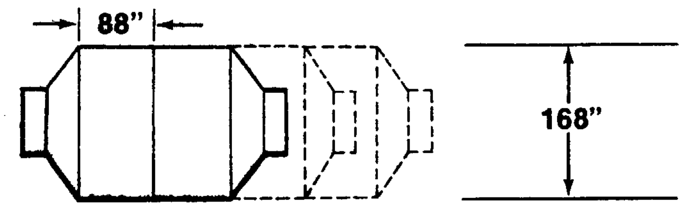
ASSUME 88" SEGMENTS PER TIGER TEAM STUDY:



NO. OF SEGMENTS	VOLUME, FT^3	GROSS WT., LB (AVG)	GROSS WT., LB (HIGH)
2	2718	27,900	34,900
3	3847	36,200	45,300
4	4976	43,900	54,900
5	6105	51,200	64,000

- **IF OTV PAYLOAD IS 38,600 LB, THREE-SEGMENT MODULES ARE MAXIMUM USABLE SIZE**
- **IF MODULES ARE APPRECIABLY HEAVIER THAN AVERAGE CURVE (E.G., FOR STRUCTURAL BEEF-UP TO RESIST REGOLITH - BURIAL LOADS), THREE SEGMENTS (264" CYLINDER) ARE TOO LARGE**
- **IF SPACE STATION STRUCTURAL ELEMENTS ARE NOT USED, DIAMETER IS STILL CONSTRAINED BY ORBITER PAYLOAD BAY. NEW STRUCTURAL DESIGN WILL NOT APPRECIABLY ALTER THE ABOVE RESULTS**

in 1983. It can carry three men and has all the necessary systems for command, communication, control, and life support.

Trajectory Data

Trajectory data were taken from reference 5. There are two key variables that determine the propulsive requirements for this mission. They are mission duration and mission date (planetary alignment). Reference 5 has a table of delta velocities as a function of the mission variables: duration and launch date. For the case discussed in this paper, a representative set of delta velocities was chosen for a one year mission. They are: (1) Earth depart - 28,200 ft/sec, and (2) Earth return - either zero for advanced Thermal Protection System (TPS) systems or non-reusable ablative systems or 20,000 ft/sec to reduce the vehicle's energy to parabolic with respect to the Earth.

Obviously, some additional comments are necessary to explain the choice of velocity change for Earth return. The velocity at perigee of the returning vehicle is approximately 55,000 ft/sec. At these velocities, the aeroheating to the returning vehicle will most likely exceed the limits of state-of-the-art reusable TPS (see reference 2) available for the entry heatshield. To aerocapture the returning vehicle at these velocities will require advanced TPS or ablative systems. Also, the g-levels experienced by the crew may be exhorbitant at the aerobraking reentry velocities shown. Reference 5 has incorporated, as an option, an impulsive rocket burn that will place the return vehicle in a parabolic orbit. This maneuver should reduce the aerothermal and g-level environments to a level that current state-of-the-art TPS and crew can withstand. Thus, the choice of technology for the heatshield and crew g-level considerations will affect the main rocket impulse requirements which in turn greatly impact the initial weight in LEO.

Configuration and Mission Scenario Configuration

Figure 5 shows the configuration in LEO at departure for a one year flyby mission with drop tanks sized for the case of no propulsive burn on Earth return. Two lunar OTVs, from reference 2, are mated at a docking ring on the MM. At the forward end of the MM, the CM is mated at a docking ring. Two propellant tanks are attached to the trunnion pins on the MM and are dropped prior to Earth entry.

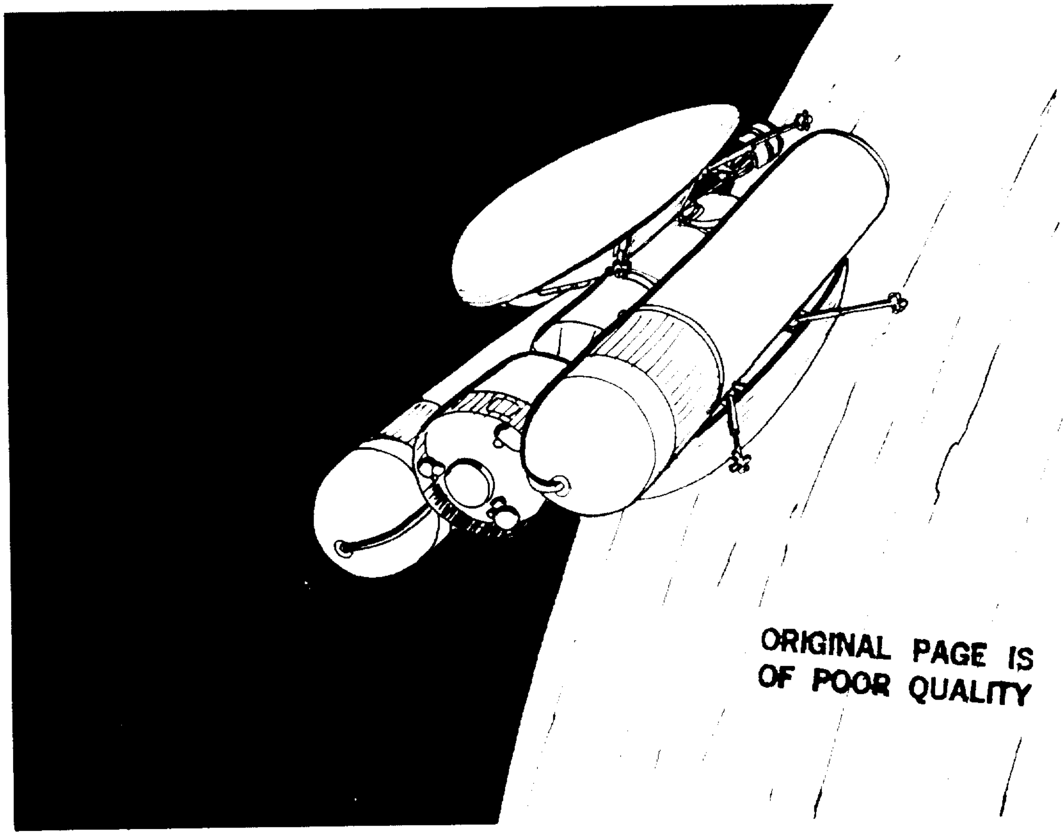


FIG. 5 - START BURN IN LEO

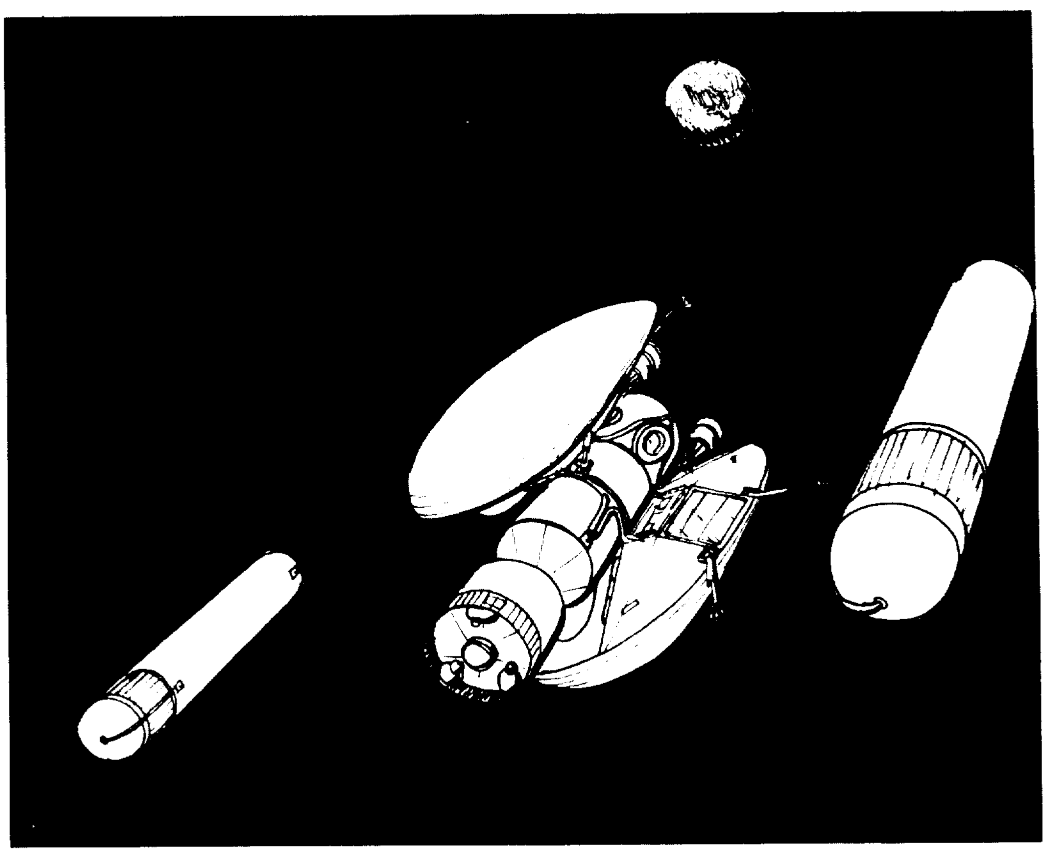


FIG. 6A - PREPARATION FOR ENTRY, DROP PROPELLANT TANKS

WEIGHT STATEMENT FOR ONE YEAR MISSION WITH NO EARTH RETURN BURN

1.	OTVs (2) @ 11,500 each		23,000
2.	Mission module		36,000
3.	Storm shelter		3,000
4.	Food & water (closed LSS)		2,300
5.	Scientific equipment		7,000
6.	Command module		12,000
7.	Drop tanks (mass fraction = .95)		23,245
8.	Main propellants	441,663	609,663
	- Drop tanks	273,663	
	- OTV tanks	168,000	
			716,208

Mission Scenario

The following brief notations describe the mission scenario:

Assembly at the Space Station

The MM and two drop tanks are delivered to LEO. The CM and manned OTVs are assumed to be operational space-based elements of the transportation system and available for this mission. All elements of the configuration mate at docking rings except the drop tanks, which will require mating to the trunnion pins on the MM and connection of umbilicals for propellants and electrical signals.

Trans-Mars Injection

Propellants are delivered by a SDV and transferred to the stacked configuration. At the start of the burn, the thrust-to-weight ratio is on the order of .1, and the total burn time is approximately one hour. To keep gravity losses to a minimum, the burn may be split into two burns if necessary. The start burn configuration is shown in Figure 5.

Trans-Mars Coast

The propellant tanks can be dropped at this time, however, since they can provide some additional shielding to the MM for meteoroids and solar storms, it might be advisable to keep them attached until just prior to Earth entry.

Mars Encounter

The encounter period (within 10 planetary radii) will be approximately 2 1/2 hours and the periapsis velocity at Mars will be approximately 26,000 ft/sec at an altitude of 160 n.m. (reference 5).

Return to Earth

As the vehicle returns to Earth, the OTVs are uncoupled from the MM, and the CM is docked with one OTV for aerocapture at the Earth (Figures 6A and 6B). The second OTV is jettisoned unless the heating problem is resolved, because no propellants have been saved for return of this vehicle. As an alternate, this vehicle does have Mars entry capability and could be used to place a payload on the Martian surface. The MM is jettisoned. If a burn is needed to reduce the velocity for TPS heating constraints or to meet g-level constraints, it will be done at this time. The OTV returns to the Space Station after passing through the atmosphere and performing some orbit adjustments. Figure 7 is an artist's concept of entry.

PERFORMANCE CONSIDERATIONS

The OTVs, storm shelter, and CM are fixed weights that cannot be manipulated; however, the MM (which would include choices on open or closed life support system), scientific equipment, and consequently, the drop tanks, are parameters that can be varied to perform some sensitivity studies. The payoff function for these sensitivity studies will be weight in LEO (W_{LEO}) at Earth departure, since this parameter has been generally accepted as an economic indicator of mission cost. Using the rocket equation, a relationship can be established for the weight in LEO for this mission, it is:

$$WLEO = \frac{\lambda_T}{T^{-1+e^{-a_1}}} \quad \frac{S \div CM}{e^{-a_2}} \div W_S \;\hat{-}\; \frac{2\ (1 - T)}{T} W_{PF} \div W_{MM} \div W_{BO} \quad (1)$$

where:

λ_T = mass fraction for the drop tanks; $.90 < \lambda_T < .96$

$a_{1,2} = \frac{\Delta v_{1,2'}}{Ig_o}$ where g = 32.174 ft2/sec2

Δv = velocity change, ft/sec

I_{sp} = specific impulse, sec.

W_{PF} = Full propellant load of the OTV described in reference 2 = 84,000 lbs.

W_{BO} = Propellant boil-off prior to the Earth entry burn, lbs.

W_S = Stage weight = 11,500 lbs.

W_{CM} = Command module weight = 12,000 lbs.

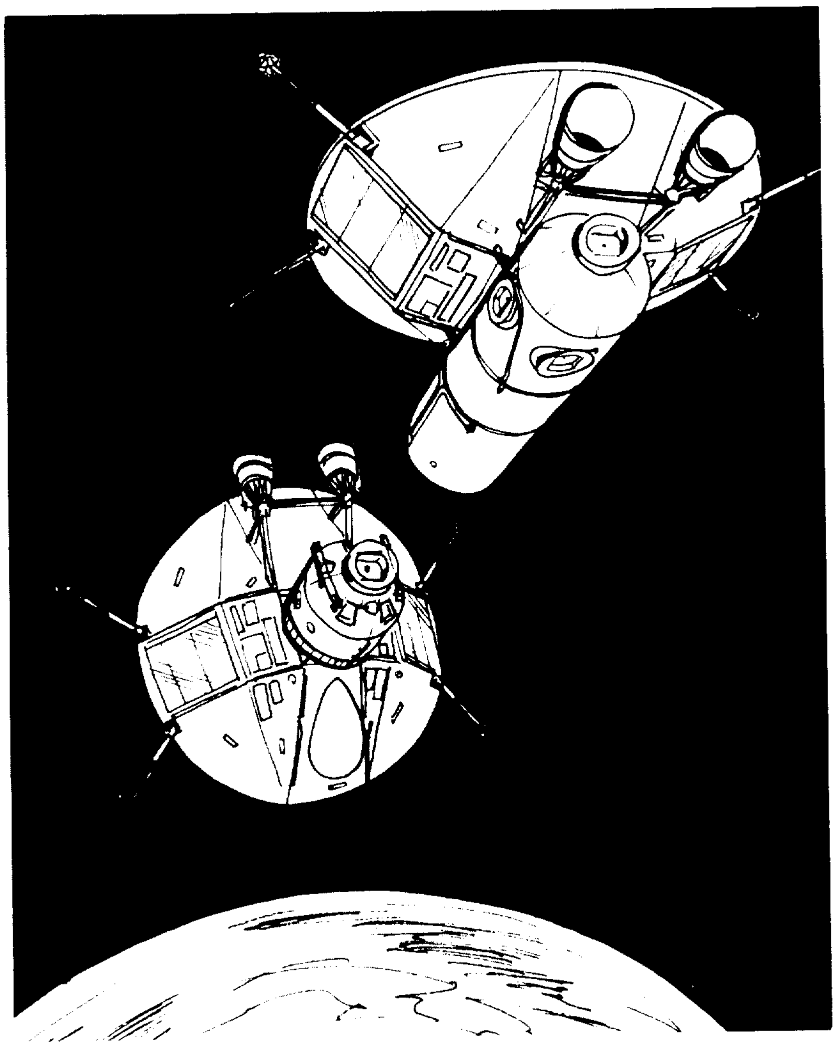


FIG. 6B - PREPARATION FOR ENTRY; ATTACH COMMAND MODULE AND DROP MISSION MODULE

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FIGURE 7

ARTIST'S CONCEPT OF ENTRY OF OTV

MM = Weight, in lbs., of MM including:

- o Solar storm shelter
- o Scientific equipment
- o Consumables
- o Life support systems

If it is assumed that:

1. Boil-off can be reduced to one pound per hour
2. I_{SP} = 460 sec. (RL-10 IIB)
3. The ΔVs are as stated earlier, and
4. λ_T = .95,

then the variation of weight in LEO as a function of W_{MM} is as shown in Figure 8. The design point for the weight statement given in the section "Configuration and Mission Scenario" is indicated on the plot. The impact of making the second burn to parabolize the Earth relative trajectory is also shown. Note that when all other considerations are equal, the decision to include this second burn impacts the LEO weight by approximately half a million pounds (all in propellant and larger drop tanks), which at forecast heavy lift vehicle delivery costs of $500 per pound, equates to an additional mission cost of $250 million (approximately $750 million for shuttle delivery). Three other points are indicated on the plot in addition to the previously discussed "design point"; one of these is an indicator of what might represent the absolute minimum mission. This point is for a mission in which the MM is replaced with a small (10,000 lbs) logistics module, principally designed for food and water storage but also providing some minimum increase in living space. Health maintenance equipment and science equipment are the most notable omissions. This minimum configuration will have a LEO depart weight of 465,000 lbs. It should be noted by the reader that although the physical relationship of W_{LEO} to W_{MM} is precise, there is not much rigor in weight estimates presented in this paper for the MM. The OTV stage weight, CM, and propellant tank weights are higher quality and would change little with detail design. However, the bottom line is probably valid and it is that a manned flyby of Mars of minimum capability can be executed with late 1990's technologies and potential space infrastructure for a LEO weight of 500-750 thousand pounds.

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FIGURE 8 W_{LEO} VS W_{MM}	ADVANCED PROGRAMS OFFICE (ED13)	
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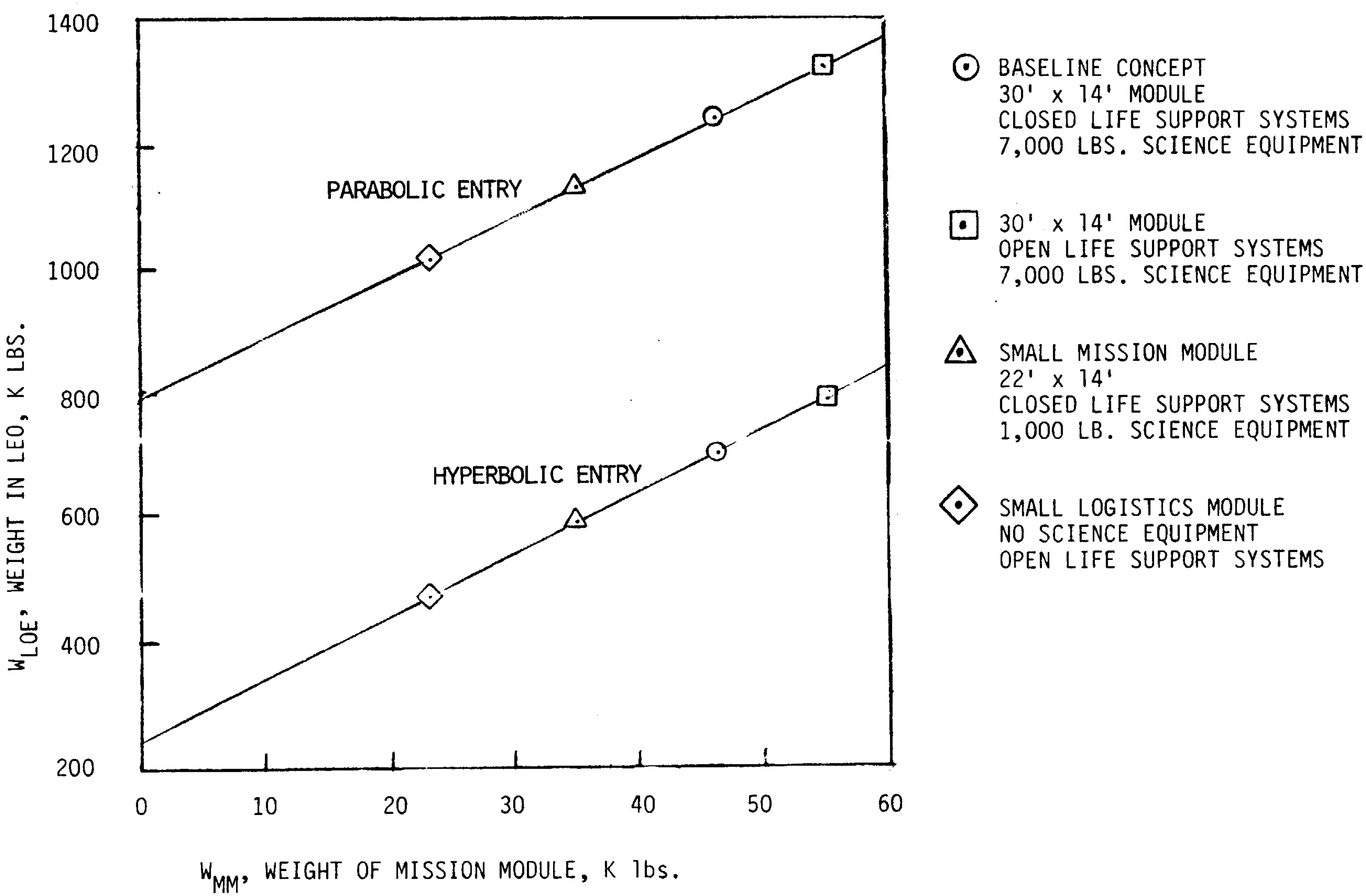

SUMMARY AND CONCLUSIONS

Summary

(1) A manned flyby of Mars would most likely be conceived in a competitive environment and mandate use of late 1990's technologies and space infrastructure.

(2) Proposed concepts for advanced space transportation system elements along with a Space Station derivative MM would satisfy the requirements for a vehicle for this mission.

(3) It is most likely impractical to utilize the reusable TPS planned for the proposed OTV for this mission due to the significant weight increase in LEO required for the Earth arrival burn. Ablative systems or advanced TPS concepts are required.

(4) A minimum mission can be performed for an initial weight in LEO of 500-750 thousand pounds.

Conclusions

It should be noted that there are no firm plans by the agency to put any of the elements discussed herein into development. The only element that is beginning to solidify is the Space Station module. The real value of this paper is to put in place the special requirements of a Manned Mars Flyby Mission using these elements such that if, and when, their development is approved, decisions will be made so as not to exclude the opportunity to use these elements to configure for this mission. On the other hand, should the decision for a Mars flyby preceed the development of the needed elements, the flyby mission components should be designed to support the Manned GEO and Lunar Base objectives.

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