Propulsive Options for a Manned Mars Transportation System

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In this investigation, a comparison of four potential manned Mars transportation systems is made. These options include 1) a single vehicle using chemical propulsion, 2) a single vehicle using nuclear thermal propulsion, 3) a single vehicle using nuclear electric and chemical propulsion, and 4) a dual vehicle (a nuclear electric cargo spacecraft and a chemical manned vehicle). In addition to utilizing the initial vehicle mass in low-Earth orbit (LEO) as a measure of mission feasibility, this study addresses the major technological barriers each propulsive scenario must surpass. It is shown that instead of a single clearly superior propulsion system, each of the four means of propulsion may be favored depending on the specified exploration policy, technology readiness level, and the acceptable manned flight time. The effect that aerobraking has on mission feasibility is also considered. Although the use of aerobraking at both Earth and Mars is shown to make chemical propulsion more competitive with the other means of transportation, this option is still not optimal from an initial LEO mass standpoint. This study also shows that the reduced initial LEO mass requirements associated with both aerobraking and nuclear thermal propulsion do not necessarily supplement each other.

Nomenclature

α	= powerplant specific mass, kg/kW
CHEM	= chemical propulsion
ERV	= Earth return vehicle
GEO	= geosynchronous orbit
HT	= high thrust
Isp	= specific impulse/s
LEO	= low-Earth orbit
LH_2	= liquid hydrogen
LOX	= liquid oxygen
LT	= low thrust
MEM	= Mars excursion module
M_{LEO}	= initial LEO mass
NEP	= nuclear electric propulsion
NERVA	= Nuclear Engine for Rocket Vehicle Applications
NTP	= nuclear thermal propulsion
SOI	= sphere of influence
STV	= space transfer vehicle

Introduction

RECENT studies have sparked a renewed interest in manned missions to Mars by showing that such a mission would be technically feasible in the early 21st century.[1,2] To accomplish the interplanetary transfer, various high- and low-thrust propulsive systems have been suggested. In this analysis, a number of transportation scenarios for a manned Mars mission have been studied. In particular, the use of chemical, nuclear thermal, and nuclear electric propulsion has been addressed.

Presented as Paper 89-2950 at the AIAA/ASME/SAE/ASEE 25th Joint Propulsion Conference, Monterey, CA, July 10-12, 1989; received July 11, 1989; revision received June 21, 1990; accepted for publication July 17, 1990.

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Because a specific manned Mars exploration strategy has not been selected, the concept of an optimal means of transportation is relatively vague. Current mission plans range from short opposition class transfers to longer, more energy-efficient conjunction class trajectories. The opposition class missions are characterized by total trip times as short as possible (on the order of 1.0 to 2.0 years), which generally includes a 30- to 90-day Mars stopover. In a conjunction class scenario, the emphasis is shifted from minimizing trip time to reducing the initial LEO mass. Typically, these missions are characterized by a longer Mars staytime (from 1 to 3 years) in which a larger crew lives in a base camp environment and is more independent of their transfer ship. Furthermore, many of these scenarios utilize advanced mission features such as the use of Martian propellant productions to decrease the required initial LEO mass.[3] A summary of the major interplanetary transfer options that correspond to various high-thrust exploration strategies is provided by Hoffman et al.[4]

The present investigation focuses on defining the advantages and disadvantages inherent to various propulsive options. This is accomplished by utilizing the initial vehicle mass in LEO as a figure of merit and identifying the key technological barriers each scenario must overcome. The propulsive comparison is based on the general guidelines of an initial manned exploration scenario that includes 1) a crew of 6, 2) a 1.0- to 2.5-year total trip time that includes a 60-day Mars stopover, and 3) launch from LEO in the time frame 2010 to 2025. Missions utilizing a single vehicle as well as a dual vehicle option (nuclear electric cargo vehicle and chemical manned vehicle) were studied. Additionally, the effect of aerobraking on each high-thrust propulsive option is analyzed. This assessment includes an identification of the reduced initial LEO mass requirements associated with various aerobraking mission profiles, as well as a preliminary characterization of the aerothermodynamic environment by using entry velocity as a figure of merit.

For consistency, similar payload systems (habitation modules, truss structure, and MEM) and mission profiles were utilized throughout the study. Transportation infrastructure assumptions include the use of an advanced launch vehicle to transfer the required mass into LEO and the use of an operational space station for on-orbit assembly of the interplanetary vehicle. Additionally, the low-thrust options require the use of

a STV vehicle to ferry supplies and personnel between LEO and GEO.

Computational Methods and Tools

To perform the required trajectory simulations, two preliminary-level trajectory optimization tools were utilized. For the high-thrust systems (CHEM and NTP), the SWISTO program was used.[5] This program is capable of simulating either round-trip or one-way missions that utilize as many as four propulsive maneuvers and a planetary swingby on either the inbound or outbound transfer leg. The trajectories are simulated using a three-dimensional patched conic approach and an impulsive velocity addition is assumed for each burn. For each LEO launch date, trajectories were simulated over a range of total trip time and the optimal transfer was identified based on minimizing the initial mass required in LEO.

The low-thrust trajectory simulations (NEP) were performed with QUICKTOP.[6] Actually, QUICKTOP is a driver program that accesses the CHEBYTOP III trajectory optimization program.[7] Rather than employing numerical integration or a calculus of variations approach, this algorithm is based on a polynomial representation of the key trajectory and propulsion parameters. By approximating the optimal trajectory solution with a series of Chebychev polynomials, the problem is reduced from its precise variational form to one of ordinary calculus. The long execution times and convergence problems associated with a more exact approach are avoided and an approximate solution ideal for a preliminary, systems-level study is obtained. As in the high-thrust simulations, for each LEO launch date, trajectories were simulated over a range of total trip time, and the optimal transfer was determined based on minimizing the required initial LEO mass.

Baseline Mission Profile and Vehicle Descriptions

In this section, a general description of each propulsive option and the mission profile is provided. Furthermore, significant interplanetary trajectory characteristics are identified and discussed. The following four vehicle options were simulated: 1) a single vehicle CHEM, 2) a single vehicle NTP, 3) a single vehicle NEP/CHEM hybrid, and 4) a dual vehicle (NEP cargo vehicle and CHEM manned spacecraft). For comparison, the dry mass breakdown associated with each option is listed in Table 1.

Baseline Mission Profile

To aid the comparison, a common set of Mars and Earth parking orbits was utilized in the simulations. In each option, the mission begins with the interplanetary vehicle departing the space station in LEO. Upon Mars arrival, the vehicle achieves an elliptical orbit characterized by a 24.6-h period (1 Martian day) and 500-km periapsis altitude. Although in an actual mission scenario parking orbit selection requires matching the orientation of the arrival and departure hyperbolic asymptotes, orbital precession and declination issues were not examined in this investigation. Hence, the propulsive requirements were based solely on energy considerations. After performing the excursion operations and discarding unnecessary components, the vehicle begins the inbound (Mars-Earth) transfer. The total time of Mars operations is 60 days. The mission is concluded with the vehicle's return to LEO.

In each of these vehicle options, the required velocity changes were performed propulsively; however, the reduced initial LEO mass requirements associated with aerobraking at Earth and Mars were identified for each of the high-thrust options.

Single Vehicle CHEM Option

The chemically propelled vehicle was originally developed by Tucker et al.[8] A LOX/LH_2 propulsion system characterized by an *Isp* of 480 s was assumed and the stage tankage is assumed to be 10% of the propellant mass required. The mission strategy requires an outbound (Earth-Mars) payload of 1.37×10^5 kg to be delivered into orbit about Mars. Of this outbound payload, 6.14×10^4 kg are returned to Earth upon completion of the mission. The interplanetary trajectory simulations showed that mission performance is affected by 1) the planetary geometry, 2) the total transfer time, and 3) the interplanetary transfer mode chosen (i.e., direct, inbound or outbound Venus swingby).

The feasibility of performing an energy-efficient chemically propelled transfer is greatly dependent on the alignment of Earth and Mars. The relative angular and absolute heliocentric position of the two planets vary cyclically with periods of 25.6 months (the Earth-relative synodic period of Mars) and 15 years, respectively. The effect of these two cycles is shown in Fig. 1, which presents the minimum required initial mass in

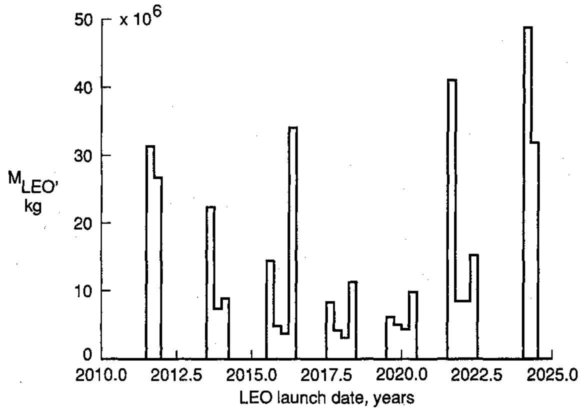

Fig. 1 Minimum initial LEO mass in each 3-month launch period vs launch date; single vehicle CHEM option (direct missions, total trip time: 1.0 to 1.5 years).

Table 1 Interplanetary transfer vehicle dry mass estimates

				Dual CHEM/NEP	
Vehicle component	CHEM, kg × 10^3	NTP, kg × 10^3	NEP/CHEM, kg × 10^3	CHEM, kg × 10^3	NEP, kg × 10^3
Two habitation modules[8]	50	50	50	50	——
Truss structure and support equipment,[9,14]	11	11	61	11	61
Propulsion system (returned)	——	12	25[a]	——	25[a]
Total inbound payload	61	73	136	61	86
Payload discarded at Mars (MEM and probes)[9,14]	76	76	75	——	75
Propulsion system (discarded)	——	24	——	——	——
Total material discarded at Mars	76	100	75	——	75
Total outbound payload	137	173	211	61	161

[a]For baseline mission, $\alpha = 5$ kg/kW; 5 MW.

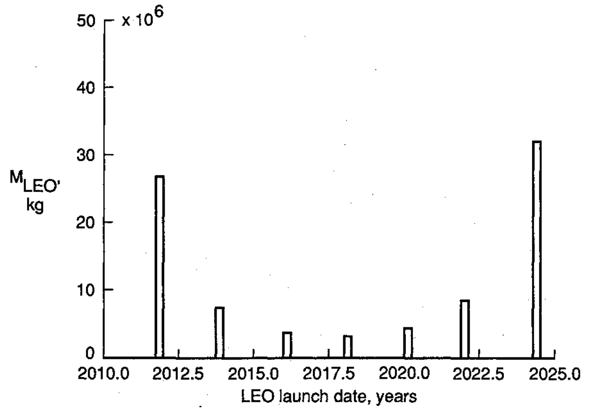

Fig. 2 Minimum initial LEO mass for each opportunity vs launch date; single vehicle CHEM option (direct missions, 10-day launch window, total trip time: 1.0 to 1.5 years).

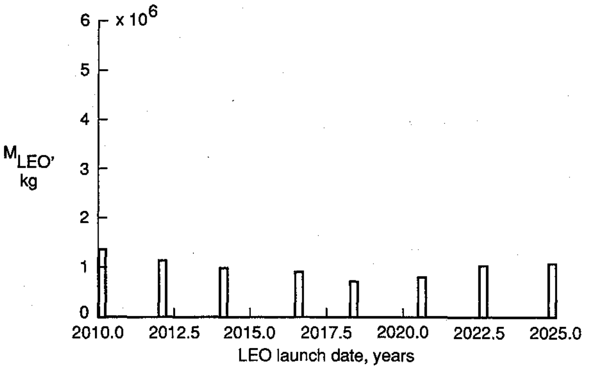

Fig. 3 Minimum initial LEO mass for each opportunity vs launch date; single vehicle CHEM option (direct missions, 10-day launch window, total trip time: 2.0 to 2.5 years).

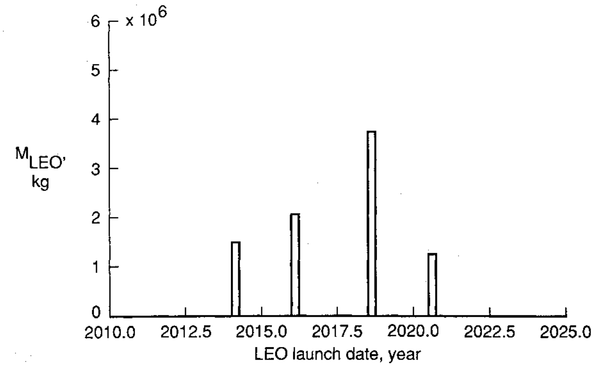

Fig. 4 Minimum initial LEO mass for each opportunity vs launch date; single vehicle CHEM option (inbound Venus swingby missions, 10-day launch window, total trip time: 1.0 to 1.5 years).

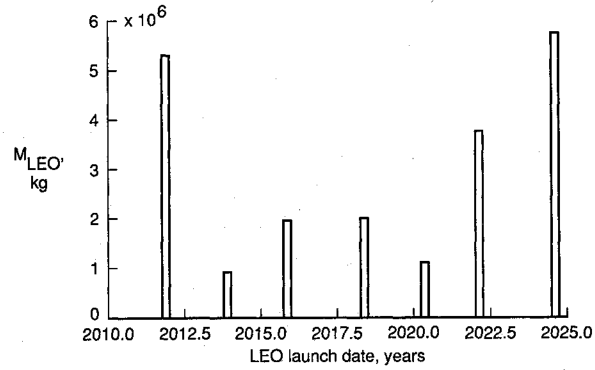

Fig. 5 Minimum initial LEO mass for each opportunity vs launch date; single vehicle CHEM option (inbound Venus swingby missions, 10-day launch window, total trip time: 1.5 to 2.0 years).

LEO for any direct (Earth-Mars-Earth) transfer of 1.0 to 1.5 years.

The results presented in Fig. 1 were determined in the following manner. First, the 2010 to 2025 Earth departure timeframe was divided into 3-month departure periods. Each departure period was also divided into 20 departure dates (a 5-day departure date increment). Then, all potential direct missions of 1.0 to 1.5 years for each departure date were simulated and the trajectory that resulted in the minimum initial LEO mass over the entire 3-month launch period was identified. Note that the data presented in Fig. 1 are not divided by the transfer type (i.e., Type I, Type II), but are presented as a function of the Earth departure date, interplanetary transfer mode (in this case, direct), and total trip time (in this case, 1.0 to 1.5 years).

Figure 1 shows that optimum mission opportunities exist approximately every 26 months, whereas the initial LEO mass requirements rise dramatically for departure dates in between. With a 10-day launch window assumed, the minimum initial LEO mass required within each departure opportunity is depicted in Fig. 2. (A 10-day launch window is utilized in all subsequent work.) Figure 2 illustrates that optimum performance repeats approximately every 7 opportunities or about every 15 years. Note that even the most mass-efficient opportunity requires a relatively large initial LEO mass of 3.07×10^6 kg, whereas other opportunities require an order-of-magnitude increase in the initial LEO mass. In all subsequent work, a maximum initial LEO mass of 6.0×10^6 kg is assumed.

By increasing the total trip time, the required initial LEO mass is reduced. Figure 3 shows the results of the same vehicle following a 2.0- to 2.5-year round-trip direct transfer. Once again, a mission opportunity exists approximately every 26 months. However, these trajectories require a much lower initial LEO mass (as low as 7.85×10^5 kg). Additionally, because of the reduced energy requirements inherent to missions with longer trip times, the effect of the 15-year ephemeris cycle is less noticeable.

Another means of reducing the required initial LEO mass is through the use of a Venus swingby. By either gaining or losing energy relative to the sun, this maneuver yields a reduction in the vehicle's propulsive requirements; therefore, a lower initial LEO mass results.[10] Minimum initial LEO mass requirements for the inbound Venus swingby mission opportunities with a round-trip time of 1.0 to 1.5 years are depicted in Fig. 4. In comparison to a direct mission of equivalent transfer time (Fig. 2), the significantly lower initial LEO mass requirements of a Venus swingby mission (as low as 1.35×10^6 kg for a 1.0- to 1.5-year mission) are evident. Furthermore, as shown in Fig. 5, the initial required mass is reduced further (as low as 9.86×10^5 kg) if the total transfer time is extended to 1.5 to 2.0 years. Although not shown, the same effect is noted if an outbound Venus swingby mission sequence is selected. These outbound swingby mission opportunities complement the set of inbound opportunities such that an even larger number of potential missions exist that span the 15-year Earth departure cycle.[9]

Additional types of interplanetary trajectories exist for a chemically propelled vehicle, including transfers that incorporate more than one planetary swingby, a more complex burn sequence, or cyclic behavior.[4] However, these scenarios were not examined in the present investigation.

In an aerobraking scenario, a Venus swingby may be utilized to lower the encountered atmospheric entry velocity (hence, reduce the severity of the aerothermodynamic environment) upon Earth return.[11] For example, the direct mission profiles of total trip time 1.0 to 1.5 years (Fig. 2) are characterized by Earth re-entry velocities in the range of 11.5 to 18.0 km/s. With the use of either an inbound (Fig. 4) or outbound Venus swingby, the Earth re-entry velocities range from 11.5 to 14.0 km/s. Furthermore, through a judicious selection of the Earth departure date and the use of a Venus

swingby, numerous mission opportunities may be identified in the 2010 to 2025 time frame in which the Earth re-entry velocities range from 11.5 to 12.5 km/s.[9]

Single Vehicle NTP Option

In this analysis, the nuclear thermal propulsion system was modeled on the NERVA experimental results. Through the NERVA program, both the NRX series (3.34×10^5 N of thrust) and Phoebus series (1.11×10^6 N of thrust) nuclear reactors were developed and tested.[12] Because the NRX class reactors were more thoroughly developed, the characteristics of this nuclear engine were simulated in the present analysis. This engine has undergone ground testing and development under simulated space conditions. However, an actual flight-test program had not been performed before the project's cancellation. Each reactor system weighs 1.18×10^4 kg, of which 5.0×10^3 kg is allocated for a tungsten/LiH radiation shield to protect the crew.[12] Although an *Isp* on the order of 900 s was estimated as being achievable by the NERVA program, the inclusion of postburn propellant losses, which are required to cool the reactor, and degradation of the fission elements within the nuclear core significantly reduce the effective *Isp*. Thus, in this study (as in other analyses[12,13]), an *Isp* of 825 s was chosen.

The NTP mission profile requires the same Mars arrival and Earth return payload as in the CHEM simulations (Table 1) and a 10% LH_2 stage mass was assumed. For the Earth departure burn, three NRX class engines are utilized to reduce gravity losses. Two of these engines are discarded at Mars and the third NRX system is used to provide both the Mars-departure and Earth-arrival propulsive maneuvers. Thus, a NRX restart capability is assumed.

The NTP interplanetary trajectory simulations showed that, as in the CHEM option, mission performance is affected by the 1) planetary geometry, 2) total transfer time, and 3) transfer mode selected. However, the constraints that each of these

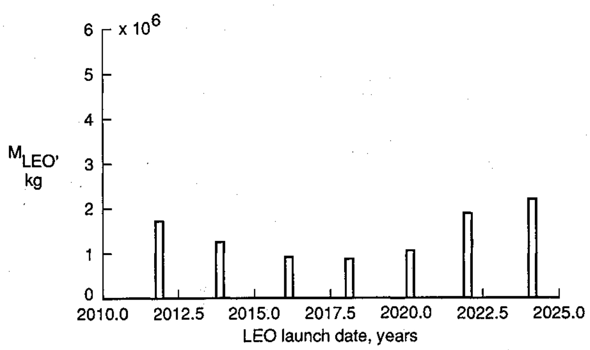

Fig. 6 Minimum initial LEO mass within each opportunity vs launch date; single vehicle NTP option (direct missions, 10-day launch window, total trip time: 1.0 to 1.5 years).

factors place on the mission profile are significantly reduced by the higher *Isp* of the NTP system. Figure 6 presents the results of the 1.0- to 1.5-year direct transfers, whereas the results of the 1.0- to 1.5-year inbound Venus swingby missions are shown in Fig. 7. By comparing these two figures with Figs. 2 and 4, it is evident that the NTP vehicle is capable of achieving shorter, higher energy transfers than an equivalent chemically powered spacecraft. These higher-energy transfers (which induced significant mass penalties on the CHEM option) yield a greater number of mass-efficient potential mission opportunities. This is evident by noting that because of initial LEO mass requirements, the 2012.5, 2022.5, and 2025.0 mission opportunities shown in Fig. 7 do not appear in Fig. 4. Additionally, longer departure windows result within each mission opportunity.

To accomplish a specific transfer, Figs. 6 and 7 show that a vehicle using nuclear thermal propulsion requires much less initial LEO mass than a similar CHEM spacecraft. For the direct 1.0- to 1.5-year transfer mode, as little as 9.00×10^5 kg is initially required in LEO (this is approximately the same mass that the most efficient CHEM option required for a 2.0- to 2.5-year transfer); furthermore, when a Venus swingby is included in the mission plan, as little as 5.28×10^5 kg is required. Thus, in comparison to the CHEM option, initial mass reductions on the order of 65% are achieved by the use of nuclear thermal propulsion for this class of transfer time (1.0 to 1.5 years). Figure 8 presents the results of the NTP simulations for direct transfer with a 2.0- to 2.5-year total trip time. Once again, in comparison to the CHEM option (Fig. 3), a reduced initial LEO mass requirement results if the mission is based on nuclear thermal propulsion (as low as 4.45×10^5 kg). This reduction (on the order of 43%) is not quite as substantial as in the shorter-duration transfers because of the CHEM option's increased performance with total trip time.

Although mission flexibility is enhanced with the NTP option (in comparison to the CHEM option), in an aerobraking scenario, the use of a higher-energy interplanetary transfer may result in a significantly increased entry velocity during Mars entry or Earth re-entry. For example, recall that in the 1.0- to 1.5-year inbound Venus swingby transfer mode, the trajectories shown in Fig. 4 (CHEM option) encountered Earth re-entry velocities in the range of 11.5 to 14.0 km/s, whereas the trajectories illustrated in Fig. 7 (which are powered by the NTP system on trips of 1.0 to 1.5 years and include mass-efficient, higher-energy transfers) are characterized by Earth re-entry velocities of 11.5 to 22.0 km/s. Earth aerocapture at the high end of this velocity range is certainly unfeasible given the severe aerothermodynamic flight environment that would result. In comparing the CHEM option and NTP option direct transfer mode, the variation in entry velocity is even more pronounced. Hence, the reduced initial LEO mass requirements associated with both aerobraking and nuclear thermal propulsion do not necessarily supplement each other.

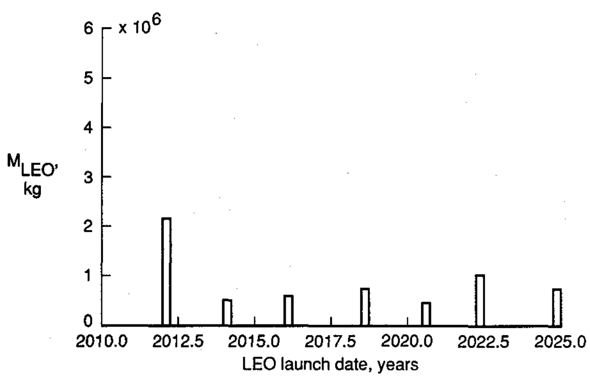

Fig. 7 Minimum initial LEO mass within each opportunity vs launch date; single vehicle NTP option (inbound Venus swingby missions, 10-day launch window, total trip time: 1.0 to 1.5 years).

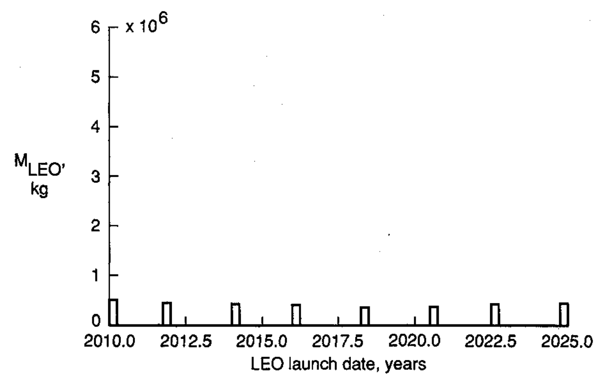

Fig. 8 Minimum initial LEO mass within each opportunity vs launch date; single vehicle NTP option (direct missions, 10-day launch window, total trip time: 2.0 to 2.5 years).

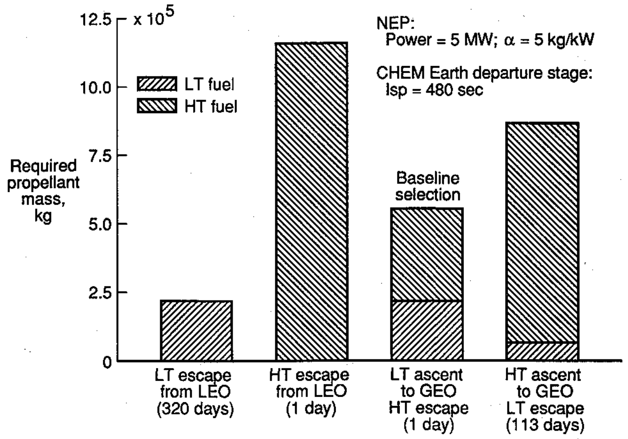

Fig. 9 Earth departure options; single vehicle hybrid NEP/CHEM option.

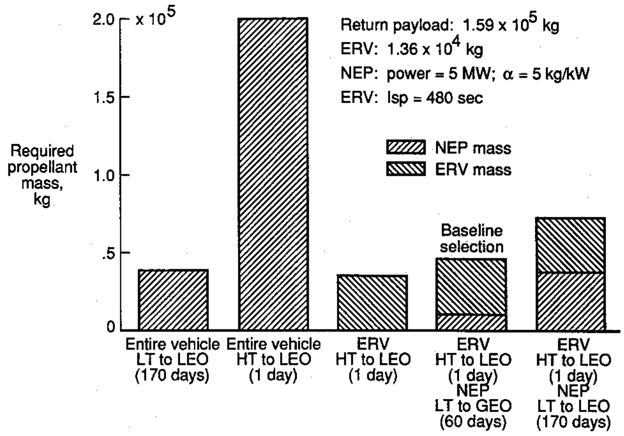

Fig. 10 Earth return options; single vehicle hybrid NEP/CHEM option.

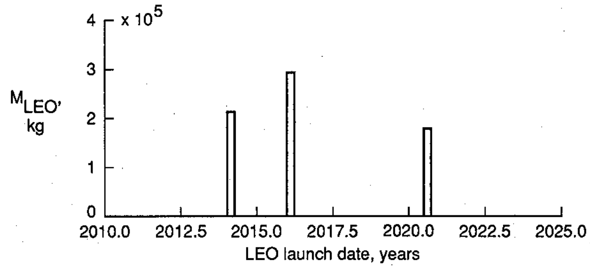

Fig. 11 Minimum initial LEO mass (manned vehicle) within each opportunity vs launch date; dual vehicle CHEM/NEP option (inbound Venus swingby missions, 10-day launch window, total trip time: 1.0 to 1.5 years).

Single Vehicle Hybrid NEP/CHEM Option

The hybrid NEP/CHEM vehicle simulated in this study is a low-thrust interplanetary transfer vehicle concept that has been augmented with a chemical Earth departure stage. A dry mass breakdown of the hybrid vehicle is presented in Table 1. Primary vehicle propulsion is characterized by a low-thrust propulsion system that incorporates an argon ion thruster subsystem with a vacuum-specific impulse rating of 3,000 s. The required electrical power is generated by a nuclear reactor capable of a total power output capacity of 5 MW. The power system specific mass α is specified as 5 kg/kW and the chemical Earth departure booster is a LOX/LH_2 propulsion stage characterized by an *Isp* of 480 s. In addition, the interplanetary vehicle carries two smaller spacecraft, a MEM and an ERV. Both craft utilize LOX/LH_2 propulsive systems characterized by an *Isp* of 480 s.

After assembly in LEO, the mission begins as the vehicle initiates an unmanned spiral maneuver to GEO utilizing the NEP system. Once in GEO, the crew rendezvous with the vehicle via an STV. At this point, the Earth departure maneuver is performed by the high-thrust chemical stage. As detailed in Fig. 9, this departure mode represents a compromise between utilizing a relatively slow but fuel-efficient low-thrust propulsion system and an inherently faster, but less mass-efficient high-thrust booster. In this manner, the manned Earth departure time is on the order of that provided by the CHEM and NTP options; however, the departure stage mass requirements are reduced.

By applying a variable thrust profile during the interplanetary transfer, the vehicle is able to control its trajectory such that the ephemeris requirements of a particular planetary alignment are met throughout the 26-month cycle. Hence, because a low-thrust system was utilized to perform the interplanetary transfer, mission flexibility is enhanced. After 60 days, a low-thrust escape and transfer back to Earth is performed. Once the Earth's SOI is reached (radial distance of 9.235×10^5 km), the crew departs the main ship and returns to LEO in a chemically propelled Earth return capsule while the interplanetary vehicle (using its low-thrust propulsion system) spirals to GEO. The interplanetary vehicle is left in GEO so that the initial LEO-GEO spiral would only need to be performed once. As shown in Fig. 10, this Earth return option was selected according to a rationale similar to that used in the Earth departure choice. Note that because this option relies on a low-thrust system as its primary source of propulsion, aerobraking was not considered.

This hybrid option represents a series of mission tradeoffs, with operational staging, high- and low-thrust propulsion, and vehicle system performance parameters chosen such that mission performance is enhanced. By combining favorable high- and low-thrust trajectory characteristics, the vehicle is capable of providing round-trip flights of approximately 1 year with an associated initial LEO mass of approximately 1.15×10^6 kg. A more detailed description of the interplanetary, operational, and logistical issues examined is provided by Blersch.[14]

Dual Vehicle CHEM/NEP Option

The dual vehicle CHEM/NEP option mission profile simulated in this investigation is comparable to that proposed by Niehoff.[15] A dry mass breakdown of both vehicles in the dual vehicle CHEM/NEP option is presented in Table 1. In this option, a NEP vehicle serves as an unmanned cargo spacecraft that carries as much of the outbound (Earth-Mars) payload as possible on a low-thrust, fuel-efficient trajectory. A second vehicle (utilizing a LOX/LH_2 chemical propulsion system with an *Isp* of 480 s) serves as a manned transport craft and is responsible for transferring the crew to and from Mars as quickly as possible. The mission is enhanced further by transporting the Mars departure and Earth return chemical stages aboard the NEP cargo spacecraft. In this manner, the outbound payload mass of the manned vehicle is decreased, thereby diminishing its propulsive requirements and allowing for a shorter manned transfer to be flown. The cargo vehicle utilizes a low-thrust propulsion system that incorporates an argon ion thruster subsystem with a specific impulse of 3000 s. As with the nominal hybrid NEP/CHEM mission scenario, primary power is provided by a nuclear reactor that generates 5 MW of usable energy and an α of 5 kg/kW was utilized.

The mission begins by launching the NEP cargo vehicle on a low-thrust trajectory from LEO toward Mars. The total transfer time is on the order of 2.5 years (this includes an approximate 1-year spiral time for Earth departure and a 3-week spiral into Mars orbit). The NEP cargo vehicle's total outbound payload mass is 4.95×10^5 kg (including the chemical Mars departure and Earth return stages for the manned vehicle). The NEP vehicle's initial LEO mass including the argon propellant and nuclear power system is 7.00×10^5 kg. After the cargo vehicle is inserted and autonomously certified in its orbit about Mars, the manned vehicle is launched from LEO

on the next opportunity. As alluded to previously, this vehicle contains only supplies necessary for the outbound trajectory leg (the Mars departure and Earth return stages are carried aboard the NEP vehicle). Figure 11 presents the three most efficient departure opportunities in terms of initial LEO mass for the chemically propelled manned vehicle. As shown in this figure, the initial LEO mass of this vehicle is affected by launch date (from 2.03×10^5 kg for LEO departure in 2020 to 3.00×10^5 kg for departure in 2016). Because one of the primary reasons for using a dual vehicle option is to lower the initial LEO mass of the manned spacecraft, only the three most efficient opportunities over the 15-year launch span were considered.

Upon Mars arrival, the two vehicles rendezvous and the manned vehicle's return (Mars-Earth) supplies and fuel are transferred onboard. At this point, approximately 4.0 years have elapsed since the NEP cargo vehicle departed LEO. After performing the required Mars exploration activities, the landing party returns to the manned vehicle, discarding the MEM; the total time of Mars operations is 60 days. The manned vehicle is then launched on a quick return trajectory. To increase mission efficiency from a mass standpoint, an inbound Venus swingby maneuver is performed. Additionally, the manned vehicle could aerobrake at both Earth and Mars (with entry velocities comparable to that of the CHEM option). The crew's mission is concluded with the manned vehicle's return to LEO; total manned flight time is approximately 1.0 to 1.5 years. If warranted by the exploration strategy, the NEP cargo vehicle may be returned to either LEO or GEO for refurbishment.

As mentioned previously, the mass of the low-thrust cargo vehicle is relatively insensitive to the launch date. However, through the inclusion of high-thrust propulsive components, the required initial LEO mass is dependent on the launch date. Overall, the vehicles initially require between 9.09×10^5 and 1.02×10^6 kg of total mass in LEO for a manned flight time in the range of 1.3 to 1.5 years. In addition to this option's sensitivity to the launch date, mission feasibility is influenced by the reliability of the tankage hardware, as well as the effects of a relatively large propellant storage time (approximately 3.5 to 4.5 years) and Mars rendezvous issues.

Comparative Results

The initial LEO mass required to perform each of the various propulsive options is shown in Fig. 12 over a range of total manned flight time. In this figure, each of the four options is represented; additionally, the effect of either an inbound or outbound Venus swingby is included for the high-thrust options. The masses shown in this figure are indicative of the minimum initial LEO mass for each particular mission

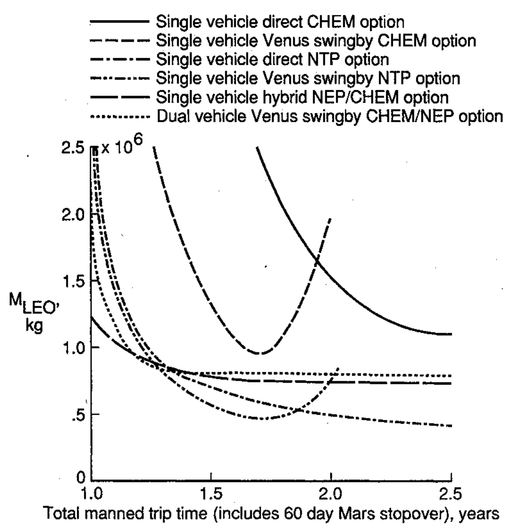

Fig. 12 Initial LEO mass vs manned flight time; 2010-2025 minimum mass opportunity, 10-day launch window.

option over the 15-year range of LEO departure dates, 2010 to 2025.

As shown in Fig. 12, because of the significant increase in initial LEO mass requirements as the trip time is shortened, the single chemically propelled vehicle does not appear competitive with the other options for missions of less than approximately 1.25 years. Furthermore, even with the associated benefits of a Venus swingby maneuver, the single vehicle CHEM option requires the most LEO mass, regardless of flight time. On the other hand, over a limited range of manned flight times (1.6 to 1.8 years for the Venus swingby mode and 2.2 to 2.5 years for the direct mode), this propulsive means becomes competitive with the hybrid NEP/CHEM option and dual vehicle CHEM/NEP option (initial LEO mass of approximately 1.0×10^6 kg).

As is evident from Fig. 12, the means of propulsion that requires the least LEO mass is significantly dependent on the acceptable range of trip time. For a total mission duration greater than 1.3 years, the most mass-efficient means of transportation is provided by the NTP option. Over this entire range of flight time, the NTP option can be performed for under 1.0×10^6 kg. Furthermore, as little as 5.0×10^5 kg may be required (for a manned flight time of 1.6 years). Thus, by nearly doubling the *Isp* (in comparison to the CHEM option), the required initial LEO mass is nearly halved. Note that a single vehicle NEP option is generally considered competitive with the NTP option in terms of initial LEO mass requirements.[16] However, a pure NEP option was not simulated in the present analysis.

For a round-trip transfer of 1.2 to 1.3 years, the dual vehicle CHEM/NEP scenario is the most mass-efficient option. By maximizing the low-thrust cargo vehicle's potential as an efficient transfer vehicle, the chemically propelled manned spacecraft can complete a quick transfer relatively efficiently. However, because trip times shorter than 1.2 years require a large increase in the size of the manned spacecraft's chemical propellant requirements, this option's efficiency is sacrificed. For trip times less than 1.2 years, the hybrid NEP/CHEM scenario was found to be the most efficient option from an initial LEO mass standpoint. Because this option relies less on the use of chemical propulsion, it is less affected by decreasing manned flight times than the dual vehicle CHEM/NEP option. Therefore, depending on the total trip time, one of three propulsive options provides the most mass-efficient means of transportation.

The degree to which aerobraking at both Mars arrival and Earth return enhances the mission depends largely on the type of propulsion system selected, as well as the acceptable range of manned flight time. Aerobraking will reduce the required initial LEO mass for the high-thrust options. However, because in this investigation the trajectory optimization process was based on minimum initial LEO mass rather than minimum flight time, the effect of aerobraking on a low-thrust mission concept is negated by the inclusion of planetocentric spiral maneuvers. In this analysis, the aerobrake masses were assumed to be 15% of the vehicle mass prior to atmospheric entry.

As shown in Fig. 13, aerobraking is of the most benefit to the short-duration, chemically propelled mission option (an approximate 50% reduction was achieved for the 1.25-year manned flight time). However, as the high-thrust system's *Isp* is increased (as in the NTP option), the relative mass reduction is not as significant (on the order of 20%). Furthermore, because the use of NTP allows higher-energy interplanetary transfers to be achieved, the aerobrake may have to withstand a more severe atmospheric flight environment than that encountered after following a lower-energy (as in the CHEM option) interplanetary flight path. Thus, not only does the initial LEO mass reduction become less significant, but the feasibility of a successful atmospheric passage diminishes as the interplanetary transfer's energy is increased. The effect of aerobraking on the manned craft of the dual vehicle CHEM/NEP option is relatively small (an approximate 10% initial mass re-

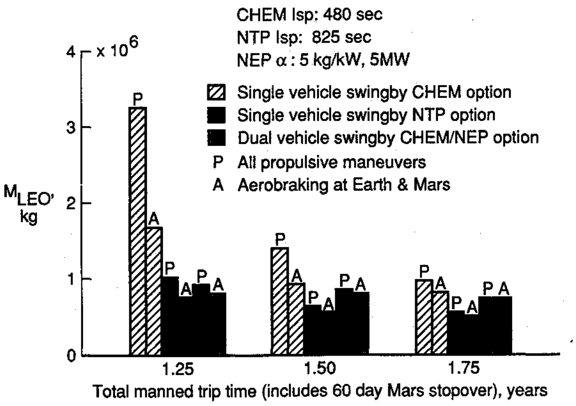

Fig. 13 Effect of aerobraking on high-thrust propulsive options.

duction) due to this vehicle's low dry mass. Because of this small reduction, the NTP aerobraking scenario becomes the most mass-efficient option for a total manned flight time of 1.25 years. Recall that for a purely propulsive option, the dual vehicle CHEM/NEP option was the most efficient for a flight time of 1.2 to 1.3 years.

As the mission's total flight time is increased, Fig. 13 shows the diminishing returns associated with including an aerobraking maneuver. As the trip time approaches 2.0 years, the initial LEO mass reduction of the CHEM option is decreased to approximately 15% (in comparison to the 50% savings noted for a mission of 1.25 years). Additionally, the savings in initial LEO mass associated with the NTP and dual vehicle CHEM/NEP options are diminished and become relatively insignificant as the manned flight time approaches 2.0 years. Finally, note that although aerobraking allows the CHEM option to be more competitive with other means of transportation, this option is still not the most efficient from an initial LEO mass standpoint.

Major Issues

As just discussed, the most efficient option from an initial LEO mass standpoint depends on the acceptable manned flight time. However, the required initial LEO mass is not the only significant factor that affects mission performance and feasibility. Of equal importance in selecting an optimal means of transportation are the technological advances required to successfully perform each propulsive scenario. These factors are summarized in Table 2. Although it is evident that the CHEM option is not the most efficient form of transportation from an initial LEO mass standpoint, it is the only option that relies on established technology. Because chemical rocket systems are space-proven, minimal advances in technology are required if this option is chosen; additionally, this system has been accepted by the general public for use in space transportation. If manned flight time is not an issue (manned flight time greater than 2.0 years), then the large LEO mass requirements associated with a short chemically propelled transfer diminish and the system begins to become competitive with the other options. However, the CHEM option's performance is very dependent on the Earth departure date; hence, mission flexibility is limited. If a trip time of less than 2.0 years is a mission constraint, performance-enhancing features (i.e., aerobraking or a planetary swingby) are required such that the CHEM option becomes more practical. These enhancing features require a testing and validation program before possible application to a manned program. Finally, it should be noted that if manned exploration beyond Mars is a long-term objective, then reliance on chemical propulsion could prove prohibitive in terms of manned flight time and initial LEO mass requirements.

A majority of the disadvantages associated with the CHEM option are significantly diminished by the use of NTP. In particular, an achievable *Isp* on the order of 825 s provides mission flexibility (less dependency on the Earth departure date), as well as low initial LEO mass requirements. Because of the low LEO mass requirements associated with an all-propulsive mission, the use of aerobraking does not substantially increase the flexibility of the NTP option; hence, this feature is not required. Additionally, the efficiency and flexibility advantages of the NTP option increase in magnitude over the CHEM option for manned exploration strategies that include multiple cycles or travel beyond Mars. Currently, several developmental concerns remain that must be solved before a manned mission featuring NTP is attempted. These include degradation of the nuclear reactor's core and propellant losses due to postburn cooling requirements. Unfortunately, major drawbacks to this option are the safety and environmental ramifications of not only the NTP mission, but the required development and testing program. If selected, technological development could begin with the NERVA program results[5]; hence, an NTP program would not have to begin anew.

The hybrid NEP/CHEM vehicle concept is nearly as reusable as a pure low-thrust concept and is characterized by a minor dependency on the departure date. Thus, it is a flexible option that is particularly useful in a short transfer (1.0 to 1.2 years) scenario. Furthermore, this concept may be readily applied to manned exploration beyond Mars. However, the de-

Table 2 Major issues affecting each propulsive option

Option	Pros	Cons
CHEM	Established technology Competitive long duration missions	Date-dependent Requires testing and validation of enhancing features Limited planetary use
NTP	Low initial LEO weight requirements Minor date dependence NERVA results Favorable in multiple cycle scenario and general planetary use	Public support and operational confidence
Hybrid NEP/CHEM	Applicable to short transfers High reusability Minor date dependence General interplanetary use	Development and testing of multimegawatt nuclear electric power source Public support and operational confidence
Dual CHEM/NEP	Applicable to short transfers Broadest advances in propulsion technology	Logistical problems at Mars Long-term propellant storage Complex abort strategy Most extensive development program

velopment and establishment of operational confidence in a space-proven nuclear reactor system capable of generating 5–10 MW of electrical power are required before this option may be considered a feasible candidate for a manned Mars transportation system.

The dual vehicle CHEM/NEP option requires a relatively low initial LEO mass and provides a short manned trip time. However, the required Mars rendezvous and long exposure of the Mars departure and Earth return propellant stages (on the order of four years) is risky from an operational viewpoint. The development and establishment of operational confidence in a multimegawatt nuclear power system and advances in long-term cryogenic fuel storage are this option's major technological issues. Furthermore, because both a high- and low-thrust vehicle are utilized, the development program associated with the dual vehicle CHEM/NEP option is viewed as being the most extensive.

Another issue that remains unresolved pertains to the time constraints under which the manned Mars mission is attempted. If an early exploration requirement is imposed on the analysis (e.g., launch by the year 2000), the advanced technology propulsive options (e.g., electric and nuclear thermal propulsion) would be discounted due to the extensive development programs required. Under the limitations imposed by this scenario, a reliable, chemically propelled vehicle is the only option. On the other hand, if a long-term exploration strategy is chosen (e.g., launch by the year 2020), many of the advanced technology features would be given the necessary developmental time; hence, the choice of an optimal transportation system may be significantly different.

Conclusions

This investigation has shown that, instead of a single clearly superior propulsive option, each of a number of various means of propulsion may be favored, depending on the specified exploration policy, technology readiness level, and acceptable manned flight time. The propulsive option that requires the lowest initial LEO mass is the hybrid NEP/CHEM option for flight times of 1.0 to 1.2 years, the dual vehicle CHEM/NEP option for flight times of 1.2 to 1.3 years, and the NTP option for flight times of 1.3 to 2.5 years. In addition to the initial LEO mass requirements, the major technological issues associated with each propulsive means were addressed. A single vehicle CHEM option was also simulated. Although this option is not optimal from an initial LEO mass standpoint (regardless of the acceptable range of trip time), chemical propulsion is the most proven means of space transportation and requires the least technological development.

The effect of aerobraking on the required initial LEO mass was also studied. It was shown that the inherently lower LEO mass requirements associated with an aerobraking mission diminish as either the total manned flight time increases or the *Isp* of the high-thrust system increases. Hence, the reduced initial LEO mass requirements associated with both aerobraking and nuclear thermal propulsion do not necessarily supplement each other. Additionally, this investigation has demonstrated that, although the use of aerobraking at both Earth and Mars allows the CHEM option to be more competitive with other means of transportation, this option is still not optimal from an initial LEO mass standpoint.

Acknowledgments

This work was performed at the NASA Langley Research Center; partial support was provided through Cooperative Research Agreement NCC1-104 between the Langley Research Center and the George Washington University. The authors would like to point out that a number of others have contributed to this investigation. In particular, the results of J. Chris Edelen's research on the dual vehicle CHEM/NEP option have been included in this comparison. Additionally, this work was guided by Jerry Walberg, Bob Tolson, Delma Freeman, Alan Wilhite, and Dick Powell of NASA Langley. Finally, thanks should be extended to Anne Costa for preparing the figures and final text form of this paper.

References

[1]National Commission on Space, "Pioneering the Space Frontier," Bantam Books, May 1986.

[2]Ride, S.K., "Leadership and America's Future in Space," NASA, August 1987.

[3]French, J. R., "The Impact of Martian Propellant Manufacturing on Early Manned Exploration," American Astronautical Society 84-178, 1984.

[4]Hoffman, S. J., McAdams, J. V., and Niehoff, J. C., "Round Trip Trajectory Options for Manned Exploration of Mars," American Astronautical Society 89-201; see also "Orbital Mechanics and Mission Design," *Advances in the Astronautical Sciences,* Univelt, Inc., Vol. 69, April 1989, pp. 663–679.

[5]Mead, C. W., and Jones, M. F., "Optimization of Ephemeridal Parameters for Minimum Propellant Requirements on Multiplanet Roundtrip Swingby-Stopover Missions," Lockheed Missiles & Space Company, Huntsville Research & Engineering Ctr., NAS8-20082, May 1968.

[6]Mascy, A. C., Schaupp, R. W., and Hayes, S. A., "QUICKTOP III—A Computer Program for Low-Thrust Trajectory and Mass Optimization," NASA TM X-62305, Sept. 1973.

[7]Johnson, F. T., "Improvement of the QUICKTOP Digital Computer Program (CHEBYTOP III)," NAS2-7049, Final Rept. D180-15371-1, NASA Ames Systems Study Division, April 1973.

[8]Tucker, M., Meredith, O., and Brothers, B., "Space Vehicle Concepts," NASA TM 89320 (or NASA M002), Manned Mars Mission Conference, June 1986.

[9]Braun, R. D., "The Influence of Interplanetary Trajectory Options for a Chemically Propelled Manned Mars Vehicle," *The Journal of the Astronautical Sciences,* Vol. 38, No. 3, July-Sept. 1990, pp. 289–310.

[10]Deerwester, J.M., "Initial Mass Savings Associated with the Venus Swingby Mode of Mars Round Trips," AIAA 65-89, Jan. 1965.

[11]Sohn, R.L., "Venus Swingby Mode for Manned Mars Missions," *Journal of Spacecraft and Rockets,* Vol. 1, No. 5, Sept.-Oct. 1964, pp. 565–567.

[12]Howe, S.E., "Assessment of the Advantages and Feasibility of a Nuclear Rocket for a Manned Mars Mission," NASA TM 89320 (or NASA M002), Manned Mars Mission Conference, June 1986.

[13]Babb, G.R., and Stump, W.R., "Comparison of Mission Design Options for Manned Mars Missions," NASA TM 89320 (or NASA M002), Manned Mars Mission Conference, June 1986.

[14]Blersch, D.J., "An Analysis of Trajectory Options for Manned Mars Missions Utilizing Electric Propulsion," MS Thesis, George Washington Univ./JIAFS program, Nov. 1988.

[15]Neihoff, J., "Piloted Sprint Missions to Mars," Science Applications International Corporation, SAIC-87/1908, Schaumburg, IL, Nov. 1987.

[16]Keaton, P.W., and Tubb, D.J., "Nuclear Electric Propulsion," NASA TM 89320 (or NASA M002), Manned Mars Mission Conference, June 1986.