The Space Congress® Proceedings

1991 (28th) Space Achievement: A Global Destiny

Apr 23rd, 2:00 PM - 5:00 PM

Paper Session I-B - Nuclear Thermal Rocket Propulsion Application to Mars Missions

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Nuclear Thermal Rocket Propulsion Application to Mars Missions

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Abstract

This paper discusses vehicle configuration options using nuclear thermal rocket (NTR) propulsion application to Mars missions. A reference mission in 2016 using an opposition-class Mars transfer trajectory is assumed. The total mission duration is 435 days. A single 75,000-lb-thrust nuclear engine is used for all major propulsive maneuvers. The studies indicate that three perigee "kick" burns upon leaving Earth result in the lowest stage weights required in low Earth orbit (LEO). The stay time on Mars is assumed to be 30 days. On the interplanetary return leg en route to Earth, a gravity assist by Venus is employed.

The reference mission assumes that the nuclear engine delivers a specific impulse of 925 s with an engine thrust-to-weight ratio of 4. The total stage thrust-to-weight ratio was 0.06. To determine which engine parameters were most critical to good mission performance, calculations were performed over a range of specific impulses and thrust-to-weight ratios. One of the major conclusions resulting from this study is that engine specific impulse is the single most important engine parameter in reducing overall stage weight, provided the engine thrust-to-weight ratio is above approximately 4. Lower engine thrust-to-weight ratios were found to incur severe performance penalties.

Introduction

The Space Exploration Initiative provides a framework within which various elements and approaches to human exploration of the Moon and Mars can be examined. In order to provide the data necessary to make assessments, several reference approaches were selected during the Lunar and Mars exploration 90-day study conducted during the fall of 1989.

The reference approaches were used to determine which parameters drive such things as cost, schedule, complexity, and program cost. The launch opportunity for the Mars Exploration Initiative considered for this study is February 2016, which is consistent with the 90-day study. The reference trajectory of the 2016 Mars mission is shown in figure 1. It is a 435-day opposition-class trajectory which employs an inbound

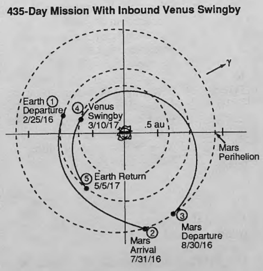
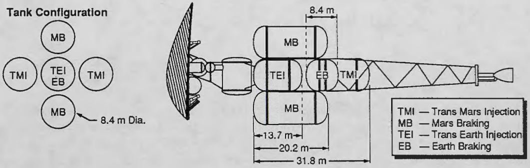


Figure 1. NTR Reference Trajectory

Venus swingby. The mission begins on February 25, 2016, with a three-impulse departure from the Space Station *Freedom* orbit. To enter the proper trajectory, a 7° plane change is needed. The use of a three-impulse departure minimizes the gravity losses, which results in a propellant savings. The outbound trip to Mars lasts 157 days. Upon arrival at Mars, the vehicle performs a propulsive maneuver to enter a 250×33,793-km-altitude parking orbit. This orbit has a period of one Mars day, or one sol. At the end of the 30-day Mars stay, the vehicle leaves Mars on a trajectory that brings it inside the orbit of Venus. At about 192 days after Mars departure, the vehicle uses a Venus swingby to change the trajectory. The Venus swingby gives a gravity assist which reduces the trajectory energy level at Mars departure and Earth arrival. Earth arrival would occur on May 5, 2017. The total return trip would last 248 days.

The Mars transfer vehicle (MTV), which is shown in figure 2, was designed to make use of elements already in use for the space shuttle external tanks (ET's). The hydrogen tank end domes and cylindrical sections could be used for the MTV propellant tanks if proper insulation is added. Hopefully, this would allow the use of ET manufacturing facilities. The tanks are arranged as two sets of drop tanks around a core tank set. The NTR engine is placed at the end of a truss structure to provide separation between the nuclear reactor and the crew module. This arrangement minimizes the reactor shield mass. The hydrogen in the propellant tanks also augments the shielding by having the Earth-return propellant in the central core tanks. The interplanetary payload which the NTR vehicle carries is a 34.9 t mission module and a Mars excursion vehicle which weighs 73.1 t. The 73.1 t includes a 25 t cargo payload to the surface of Mars.



- Allows use of existing ET manufacturing facilities.
- Larger diameter decreases the surface area-to-volume ratio and may decrease thermal insulation mass.
- Core tanks (TEI, EB) augment radiation shielding when full.
- Truss is used to keep propellant tanks within a 15° half-cone angle in front of engine.
- Tanks are dropped in pairs to maintain vehicle center of mass on the line of thrust.
- Initial Mass In Earth Orbit (IMIEO) = 638 t

Figure 2. MTV Configuration Using STS External Tank Elements

The mission profile of the MTV for the 2016 Mars mission is illustrated in figure 3. The MTV is assembled in LEO, possibly at the space station or some other LEO assembly node. The mission would depart from LEO using a three-impulse maneuver to minimize gravity losses. Shortly after the completion of this maneuver, two propellant tanks are jettisoned. The vehicle then coasts to Mars, making the necessary midcourse corrections along the way. When the vehicle reaches Mars, it performs a propulsive orbit insertion maneuver to enter a 250×33,793-km parking orbit. Two more propellant tanks are jettisoned in Mars orbit. At the end of the Mars stay, the MTV departs Mars and returns to Earth. The core of the MTV is reusable, so the vehicle performs a propulsive orbit insertion upon Earth return. The interplanetary mission module returns to an Earth elliptical orbit with a 24-hour period and a perigee of 427 km, 20 km above Space Station *Freedom's* orbit altitude. The interplanetary mission module with a crew of four is retrieved by a space transfer vehicle for return to S.S. *Freedom's* orbit.

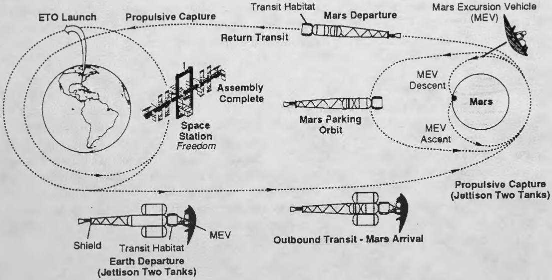


Figure 3. Mars Initiative NTR Vehicle Mission Profile

Trade Studies

In order to assess some of the mission and vehicle design issues that exist for nuclear-powered spacecraft, several trade studies were performed. These trade studies were primarily concerned with the effects of NTR engine parameters on the performance of the vehicle. In this study, the total vehicle mass at Earth departure was used as a measure of performance. The trades that were performed include the NTR engine thrust level, engine thrust-to-weight ratio, and engine specific impulse.

The initial weight in LEO is a function of the NTR total thrust level. This is due to the fact that there is a tradeoff between the gravity losses penalty experienced during the thrusting maneuvers and the weight of the NTR propulsion system. Parametric analysis was performed for the trans-Mars injection maneuver using a one-burn Earth departure and a three-burn Earth departure maneuver. An NTR total thrust-level range from 25,000 to 250,000 lb was used. The trans-Mars injection energy required was a hyperbolic energy (C_3) level of 10.3 km^2/s^2 with the hyperbolic excess velocity vector having a declination of –35.9°. Therefore, the NTR propulsion system had to perform a plane-change energy maneuver in addition to the ΔV energy maneuver(s). The plane-change maneuver was required because S.S. *Freedom* has an orbit inclination of 28.5°, thus the excess velocity vector of –35.9° is 7.4° below the space station orbit plane. A plane change of 7.4° must be performed in order to achieve the desired trans-Mars target conditions.

As shown in figure 4, the initial mass required in LEO is greater for a one-burn Earth departure than for a three-burn Earth departure. This increased mass in LEO associated with the one-burn departure is due to the increased gravity loss experienced compared to a three-burn departure. The optimum total thrust level for a one-burn departure maneuver is about 225,000 lb; a total thrust level of 150,000 to 250,000 lb would be an effective thrust-level range for an NTR engine. The optimum total thrust level for a three-burn maneuver is 107,000 lb as indicated in figure 4. An effective NTR thrust-level range for a three-burn maneuver would be between 75,000 and 150,000 lb.

Based on the NTR engine thrust-level trade results given in figure 4, a mass savings of 74 t in LEO can be realized by using a three-perigee-burn Earth departure compared to using a one-perigee-burn Earth departure. The 74 t represent a decrease of 10.5 percent in mass required in LEO. A three-burn Earth departure trajectory profile is presented in figure 5. The NTR engine characteristic for the three-burn profile has an engine Isp of 925 s and a thrust level of 75,000 lb. The initial thrust-to-weight (F/W_0) ratio is 0.06. The first NTR engine burn is 35.4 min and results in an elliptical orbit with a period of 3.11 h and an apogee of 8,504 km (1.33 Earth radii). The second burn is for 37.1 min and results in an elliptical orbit with a period of 37.6 h and an apogee of 99,922 km (15.67 Earth radii). A plane change of 7.4° is made at apogee in order to achieve the desired hyperbolic inclination after the third perigee burn is executed. The apogee plane-change burn requires a thrusting time of 3.2 min. The third perigee burn requires 16.7 min of NTR thrusting which places the NTR interplanetary vehicle on the desired hyperbolic energy (C_3) level of 10.3 km^2/s^2 with a declination of −35.9°.

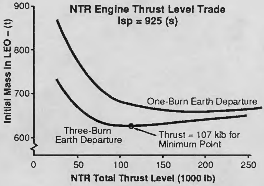


Figure 4. All Propulsive NTR Maneuvers for the Human Mars Initiative

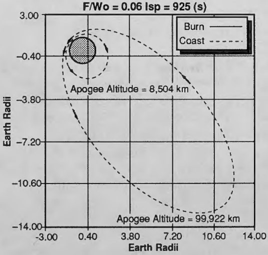


Figure 5. Trans-Mars Three-Burn NTR Maneuver

Another engine parameter which has a significant impact on vehicle performance is engine Isp. The most noticeable effect of Isp on vehicle performance is the impact on the propellant requirement for the mission. Since most of the vehicle mass is propellant, the effect of Isp can be shown by examining the sensitivity of vehicle initial mass in LEO to engine Isp.

An NTR engine Isp trade analysis was conducted for an engine-thrust level of 75,000 lb, and the summary results are presented in figure 6. Two different technology levels were considered which include: (1) nuclear engine for rocket vehicle application (NERVA) technology and (2) advanced NERVA technology. Results for Isp values of 850, 925, 1,050, and 1,200 s are given for comparison to the chemical-plus-aerobrake interplanetary vehicle. The initial mass in LEO required for the chemical-plus-aerobrake is 830 t, which will require seven heavy lift launch vehicle (HLLV) launches. The HLLV LEO payload capability is 140 t. The Mars interplanetary vehicle, which has an NTR engine with an Isp of 850 s, has an initial mass in LEO of 729 t using NERVA technology and 681 t using advanced NERVA technology. Advanced NERVA technology results in a decreased mass in LEO of 48 t or a decrease of 6.6 percent. The difference in initial mass in LEO using an Isp of 1,200 s is more pronounced between NERVA technology and advanced NERVA technology; the difference in mass in LEO is 110 t or a difference of 20.6 percent. To place the interplanetary vehicle into LEO, four HLLV launches are required with an NTR engine of 1,200 s Isp and advanced NERVA technology. The four HLLV launches have a payload margin of 91 t for this interplanetary vehicle.

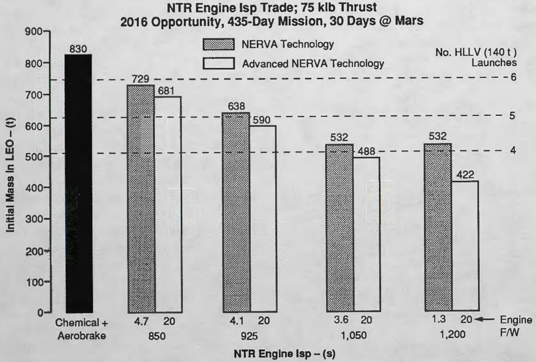


Figure 6. All Propulsive NTR Maneuvers for the Human Mars Initiative

A second important effect of engine Isp on the performance of a nuclear-powered spacecraft is the effect that Isp has on the engine mass. This effect can be seen in the sensitivity of vehicle initial mass to the engine thrust-to-weight ratio for several values of Isp. If the engine can be built to achieve the optimum thrust-to-weight ratio, considerable vehicle mass savings are possible. Figure 7 shows the results of an engine thrust-to-weight ratio trade. The trade was performed for four values of Isp. The reference NTR engine design point considered was the advanced NERVA engine characteristics with an Isp of 925 s and a thrust-to-weight ratio of 4.1. As shown in figure 7, when the thrust-to-weight ratio becomes less than 6, the impact on initial weight in LEO increases exponentially. When the NTR engine thrust-to-weight ratio can be designed and built with a ratio of 20 or greater, the initial weight in LEO sensitivity becomes small for a given engine Isp. NTR engine Isp level has a significant impact on the required initial weight in LEO. Operating with an engine Isp of 1,200 s with a thrust-to-weight ratio of 6 requires 440 t weight in LEO; operating with an engine Isp of 850 s requires 715 t weight in LEO or an increase in required weight of 63 percent above that for an Isp of 1,200 s. A change of 100 s in Isp will reflect a change in weight in LEO of 85 t.

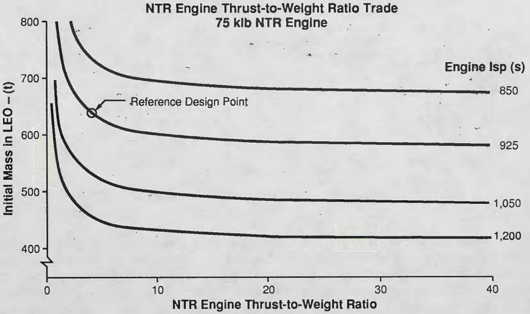


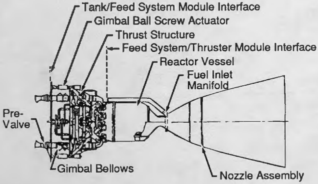
Figure 7. All Propulsion NTR Maneuvers for the Human Mars Initiative

Vehicle Description

The MTV is assembled in LEO and requires six heavy lift launches of 140 t to transport the various modules to the assembly point. Most of the initial mass for the vehicle is propellant. The vehicle is comprised of a crew compartment, a Mars lander with low-energy aerobrake, and a main propulsion system consisting of a single NERVA-class nuclear engine with hydrogen tank sets.

Nuclear Engine: The Mars transportation system (MTS) uses a single nuclear engine to provide thrust for all major propulsive maneuvers. The reference engine chosen for this study is based upon the NERVA design with certain modest extensions in technology to increase the Isp above that demonstrated in the original NERVA program. Figure 8 illustrates the design of this engine. An expander engine cycle was assumed for this study since it is more efficient than the original NERVA hot-bleed cycle. Composite fuel elements were also assumed for the present study since they provide an increase in engine performance over the coated particles in graphite matrix design, extensively tested during the NERVA program, and represent only a modest advance in technology. Reactor control is accomplished through the use of reflector control drums and actuators. The engine includes a small internal shield to protect critical engine components during operation. A larger external disk shield (not shown) is also required for propellant tank and core shielding.

Higher engine efficiencies could be attained in this design through the use of carbide fuel elements which could increase the nozzle chamber temperature to approximately 3,100 K, corresponding to an Isp of about 1,050 s. Other reactor designs could offer even higher performance engines corresponding to the other curves shown in figure 7. Engine designs based upon the particle bed concept could increase the engine thrust-to-weight ratio by factors of between 3 and 5. If engine operation at high chamber temperatures (>3,300k) and low chamber pressures (<1 atm) proves feasible, hydrogen dissociation/recombination in the thrust chamber and nozzle could yield Isp values in excess of 1,200 s.



Isp:	925 s
Thrust:	75,000 lb
Weight:	8,228 kg
Expansion Ratio:	500:1
Reactor Power:	1,645 MW
Chamber Temperature:	2,700 K
Chamber Pressure:	68 atm
Engine Width:	4.15 m
Engine Length:	7.9 m (stowed) 13.4 m (extended)

Figure 8. Reference Nuclear Engine, NERVA Type-Expander Cycle

Crew Compartment: The MTS crew compartment provides living and working space for a crew of four during the voyage between Earth and Mars. The crew compartment provides a habitable volume of 448 m^3 and weighs approximately 35,000 kg. It is constructed in the shape of a cylinder 9 m long (excluding the ellipsoidal end caps) and 7.6 m in diameter. Radiation protection is provided by including "storm shelters" within the compartment where the crew resides during propulsive maneuvers using the nuclear engine and during solar storms. The shielding materials used for the storm shelters consist of food and other consumables. The present design does not provide artificial gravity.

Propellant Tanks: The MTS tanks were designed based on a given propellant mass, internal pressure, tank diameter, and safety factor. The internal pressure of 241 kPa (35 psi) was determined from the operating pressure requirements of the NERVA engines. Tank diameter was based on the Space Transportation System (STS) ET diameter of 8.41 m (27.58 ft) in order to utilize existing tooling and facilities for manufacturing. A factor of safety of 1.5 and a minimum margin of safety of 0.15 was used for all calculations. Aluminum-lithium was selected for the tank material for its high strength, low weight, weldability, and current availability.

Wall thicknesses for each ellipsoidal dome and the cylindrical portion of the tanks were calculated based on internal pressure and hydrostatic loads. Launch loads were evaluated and dismissed as not being design drivers since internal pressure represents most of the design load. A check of the analysis method was performed using the propellant loads for the STS ET liquid hydrogen tank. The ratio of tank weight to propellant for the MTS tanks is comparable with the values for the STS ET liquid hydrogen tank.

In addition to the tank sizing, the trans-Earth injection and Earth braking tanks were examined for possible buckling during the Earth braking burn. Based on the tank diameter and the location of the thrust loads, it was determined that with an internal pressure of approximately 17.2 kPa (2.5 psi) no buckling would occur. The tank set and propellant weights are given in table 1.

Table 1. Propellant Tank Weights

Tank Set	Tank Weight (kg)	Propellant Weight (kg)
TMI-1	12,516	117,591
TMI-2	12,516	117,591
MB-1	7,018	71,932
MB-2	7,018	71,932
TEI	4,252	46,091
EB	2,216	25,364
Total	45,536	450,500

Figure 9 shows the absorbed heat rate (Q) on the NTR propellant tanks for the transit portions of the 435-day mission. The hot orientation is the case in which the vehicle is oriented such that the Sun vector is perpendicular to the sides of the tanks. This provides a larger heat absorbing area, thus a greater absorbed heat rate. For the cold case, the vehicle is oriented such that the Sun vector is directed perpendicular to the end caps of the tanks. This provides a smaller heat absorbing area, resulting in a lower absorbed heat rate.

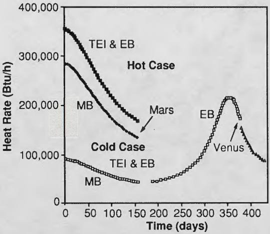


Figure 9. Heat Rate to NTR Propellant Tanks

The tanksets include the trans-Earth insertions (TEI) and Earth braking (EB) tanks and the two Mars burn (MB) tanks. For the Earth-to-Mars transit, all tanks were included in the analysis. The TEI and MB tanks are dropped before going into the Mars-to-Venus transit, leaving only the EB tank to complete the mission.

The top curve shows the amount of absorbed Q on the TEI and EB tankset in the Earth-to-Mars transit for the hot case. The curve just below it is the amount of absorbed Q on one of the MB tanks. The brake in the curve is for the 30-day orbit at Mars. The remainder of the curve shows the amount of absorbed Q for the EB tank. Because heat flux on a surface is proportional to the inverse of the square of the distance of an object from the Sun, the heat rate decreases in the Mars transit as the tanks get further from the Sun, as can be seen on the plot. As the EB tank goes toward Venus, it passes very close to the Sun, which causes the increase in heat rate shown by the peak in the curve. The heat rate drops as the tank goes on from Venus and back to Earth.

For the cold case, the TEI and EB tankset and the MB tanks have approximately the same heat rate because the tank end caps, which are directed toward the Sun, have the same surface areas. The absorbed heat rate on the EB tank for the cold case is approximately the same as it is for the hot case, because the heat absorbing area is approximately the same for either of the cases due to the near spherical shape of the EB tank.

Advanced Propulsion Options Comparison

Earth-to-orbit (ETO) mass required to be delivered to LEO by the Earth launch vehicle is given in figure 10 for four different advanced propulsion options used to accommodate the humans on a Mars mission. Two different launch opportunities to Mars are used to give the minimum and maximum requirement for a given Earth-to-Mars launch cycle. The 2018 launch opportunity is an easy year, and the 2024 is a hard year. Launch opportunities in any other years would require values somewhere between the 2018 and 2024 values. Total mission duration considered ranges from 300 to 600 days. This range of mission duration is associated with the opposition and Venus-swingby mission mode class. In order to accommodate mission time as low as 300 days in the 2018 easy year, the nuclear electric propulsion (NEP) system requires an electrical power level of 120 MW and a specific mass, $\propto$, of 3 kg/kW. Because of the very low mass fraction, the solar electric propulsion (SEP) system can only accommodate a mission duration as low as 500 days.

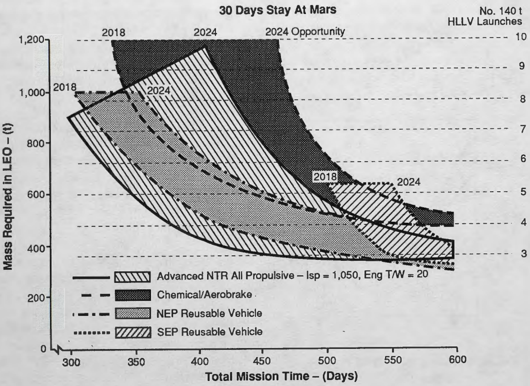


Figure 10. Advanced Propulsion Options Comparision

Acknowledgments

The authors would like to acknowledge the contributions made to this paper by the following individuals: Susan Spencer, Margaret Elrod, Greg Hajos, and Reggie Alexander.