Original Research Article

Feasibility and Parametric Analysis of Nuclear Thermal Propulsion for Crewed Earth-Mars Transfers

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ABSTRACT

Colonizing Mars is considered an important objective for long-term human exploration, scientific discovery, and technological advancement. However, crewed missions to Mars using chemical propulsion take around 7-10 months at a minimum, posing significant challenges for crew health and mission reliability. One promising solution to reduce this travel time and still maintain feasibility is by employing Nuclear Thermal Propulsion, or NTP. This study simulates an Earth-Mars transfer, using circular and coplanar orbital mechanics as well as patched-conic and two-body approximations, to evaluate the effectiveness of NTP in reducing travel times to Mars. These results show that NTP can feasibly reduce travel times to 70-90 days during the optimal launch window while achieving a hyperbolic excess velocity, or V-infinity of 10 km/s. This result considers the trajectory and launch window of such a mission, the propellant mass fractions of the rocket, travel time, and reserve propellant. These findings indicate that NTP represents a technically realistic and advantageous option for future crewed missions to Mars, and eventually, the rest of the Solar System. NTP offers a significant reduction in transit time when compared to chemical propulsion but remains consistent with current reactor technology and demonstrated technological advancements.

Keywords: Nuclear Thermal Propulsion (NTP); Transit Time Reduction; Hyperbolic Excess Velocity (V_∞); Delta-V (ΔV) Optimization; Propellant Mass fractions

INTRODUCTION

NTP is important because even though the concepts and ideas of having such a rocket have not been developed in the current era of space exploration and colonization, these types of rockets are virtually imperative for sustaining a large colony on Mars. Thus, a mission profile will be presented and evaluated purely on its ability to reach Mars in a timely and feasible manner. Understanding how NTP works – and why it is much more efficient than chemical propulsion is essential in understanding its potential. In short, this propulsion works by having a nuclear reactor in the back of the rocket, and a fuel tank with liquid hydrogen, cooled to around 20k or -253°C. When the reactor is turned on, the liquid hydrogen runs through and is superheated to 2500k or around 2226C (1-2). This superheated gas is ejected through the rocket nozzle, which generates thrust for the rocket. As a result, the specific impulse for NTP is 900s, compared to 450s for Chemical Rockets. For reference, specific impulse is a measure of fuel efficiency in rocket engines.

Currently, missions to Mars take 7-10 months (3). This is because chemical rockets follow a Hohmann transfer, a transfer orbit where Earth and Mars are in an

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Accepted March 11 2026
https://doi.org/10.70251/HYJR2348.4290101

orbital position, such that orbital energy and fuel usage are kept at a minimum. Furthermore, the Hohmann transfer only occurs once every 26 months (4). This means that if there is some type of issue on the colony, an emergency mission would be very inefficient in terms of resources used, and travel time. Even though NTP does have limitations on its transfer windows to Mars, there is a greater range of time in which a Mars mission is feasible, which means that the colony will not have to wait as long for help to arrive. This study uses two-dimensional orbital mechanics, astronautics, as well as a mission profile to analyze if an NTP transfer vehicle between Earth and Mars is feasible and realistic. Also, propellant mass percentage that is needed to reach the Delta-V requirements are evaluated to determine if NTP is a practical upgrade.

The objective of this study is to evaluate the feasibility and performance of Nuclear Thermal Propulsion for crewed Earth-Mars missions through parametric trajectory analysis. This study examines how variations in key parameters such as specific impulse, departure velocity and propellant mass fractions affect the feasibility of rapid transfers between Earth and Mars. The results aim to assess the potential advantages of NTP compared to chemical propulsion while identifying constraints that will be pertinent to future mission designs.

METHODS AND MATERIALS

Propulsion and *ΔV* Modeling

It is important to understand the mass limitations and the propellant mass percentage of the rocket, to reach the desired Delta-V. For every spacecraft the uses propellant, which includes Nuclear Thermal Propulsion, the spacecraft performance and ability to achieve a wanted ΔV is governed by the Tsiolkovsky Rocket Equation:

$$\Delta V = V_{exhaust} \cdot ln\left(\frac{M_{before\ burn}}{M_{after\ burn}}\right)$$

Where ΔV is the velocity change (m/s), $V_{exhaust}$ or exhaust velocity (m/s) is taken by multiplying specific impulse (s) to g_0, which is 9.81 m/s². In addition, the mass before and after the burn are both in kg.

This equation takes into account the specific impulse, also known as the Isp, of the rocket, which when multiplied by g_0, which is 9.8, gives $V_{exhaust}$ or exhaust velocity, as well as the increasing acceleration as mass is expelled from the rocket. Before this equation is used, the masses of the crew module and the nuclear reactor need to be found. For this study, the mass of the crew capsule is 200,000 kg (including outer covering and radiation shield). When compared to the SpaceX Starship, which has a dry mass of 85,000 kg (5), the conceptual crew module is more than twice as heavy. Even though the crew module has more mass, it also allows more crew comfort and space for cargo. Also, since NTP has at least twice the Isp, and more thrust, when compared to Chemical Rockets, larger payloads can be launched. A truss separates the crew module from the reactor, which further protects the crew from radiation coming out of it. The truss is expected to weigh 3,000 kg. Finally, the rocket consists of 5 parallel nuclear reactors, each with a rocket nozzle, and a shared fuel tank filled with liquid hydrogen. Each reactor is about 18,000 kg (6), so the reactors will have a combined mass of 90,000 kg. However, when considering the mass of the structure of the propellant tank, the total mass of the propellant tank and reactors could be 103,000 kg. Thus, after combining the mass of the reactor and propellant tank to the crew module, the dry mass of the rocket is 303,000 kg. In terms of calculating the propellant, it is important to consider not only the departure burns, but also the arrival burns, reserves, and enough fuel to return to Earth with the help of possible limited refueling. A propellant mass of 1,600,000 kg takes all these factors into account and brings the total mass to 1,903,000 kg. The rocket equation is then solved for propellant mass fraction $(\frac{M_1}{M_0})$

$$\frac{M_1}{M_0} = e^{\frac{-\Delta v}{V_{exhaust}}}$$

To find the amount of propellant remaining after a burn, you can further rearrange the equation to solve for the mass after burn (M_1)

$$M_1 = M_0\, e^{\frac{-\Delta v}{V_{exhaust}}}$$

Mission Architecture

It may seem obvious that this rocket would be too heavy to launch from Earth, and this does not consider the public opinion or environmental impacts of the effects if something goes wrong during the launch. Thus, this rocket should be assembled in space. Firstly, the crew module, which includes the crew living area and life support, as well as the cargo holds for payload, in addition to the truss separating the module from the reactor are 200,000 kg combined. This high of a mass makes it extremely heavy and difficult to launch into orbit, meaning that each part should be launched separately and docked together in orbit.

Based on the total vehicle mass and current heavy-launch capabilities in the range of 70-100 metric tons (7-8), or around 70,000 to 100,000 kg, the complete vehicle assembly would involve launching 4-5 rockets. Furthermore, when considering the in-space refueling before departure and that the maximum amount of fuel was loaded, this would add another 8-14 launches. Thus, the total amount of the launches that would be required for both refueling and assembly would be around 12-19 total. Even though this may seem incredibly difficult when considering the numerous launch procedures, orbital rendezvous and docking procedures, assembling such a rocket is not an unreasonable task. In fact, such operations are consistent with the assembly and docking capabilities of the International Space Station, which itself involved about 42 launches to build (9). The spacecraft is composed of several major sections, including the propulsion system, propellant tanks, habitat module, and payload components. The assumed mass values and sources for each subsystem used in the mission design are summarized in Table 1. The overall configuration of the spacecraft is illustrated in Figure 1.

Additionally, since landing a nuclear reactor with this amount of mass on Mars or Earth would be incredibly difficult with current technology, the rocket should start and end its mission in orbit. As a result, the mission architecture allows for partial reuse of major propulsion and life support components but are subject to structural lifetime constraints and performance.

Table 1. *Value, unit and source/basis for each part of the concept rocket. This includes the specific impulse of the spacecraft, and masses for the crew module, reactor, structure and propellant. Also, the initial spacecraft mass presents the mass of the spacecraft at departure.*

Parameter	Value	Units	Source/Basis
Specific Impulse	900	s	NERVA-class estimates
Crew Module Mass	200,000	kg	Architecture estimate
Reactor Mass	100,000	kg	NERVA-class estimates
Structural Mass	13,000	kg	Architecture estimate
Propellant Mass (fully loaded)	1,600,000	kg	Calculation estimate
Initial Spacecraft Mass	1,903,000	kg	Mission model

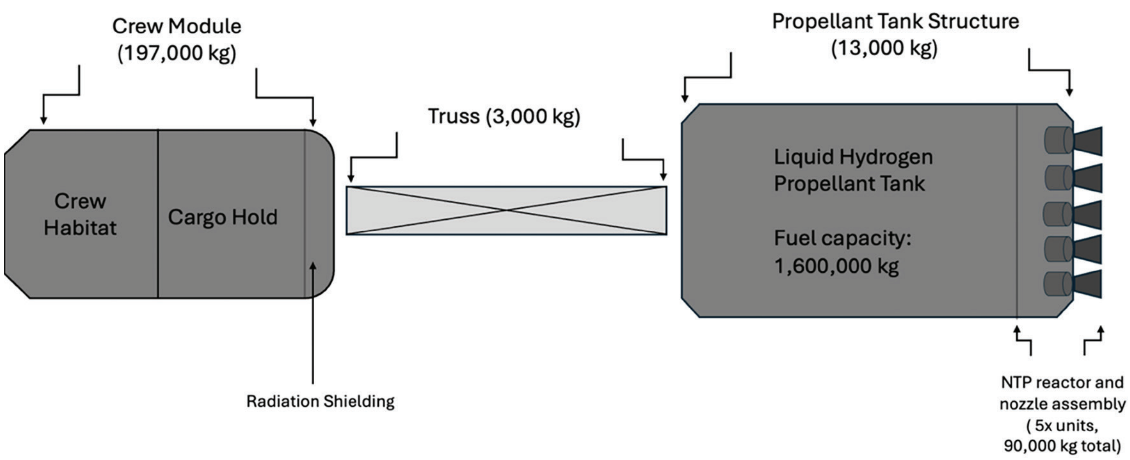

Figure 1. *Schematic diagram of the spacecraft showing the major sections and their relative positions. The mass of each section is indicated. The gaps between sections represent locations where in-space docking would occur to assemble the spacecraft components.*

The mission starts in Low Earth Orbit or LEO, at an altitude of 400 km. This altitude is like where the International Space Station orbits, and the time in LEO is low enough to prevent significant orbital effects from atmospheric drag. Furthermore, the combined reactor power for all 5 reactors is 6.5 GW. This number was considering the reactor powers of the NERVA program, during the 1960s. The reactor power for one engine is 1.1 GW reactor, so the combed power will be around 5.5 GW (10). When accounting for technological advancements in nuclear reactors in the 60 years since, 6.5 GW would be an accurate and realistic number.

To calculate thrust, the equation $T = \frac{2P}{V_e}$, where P is the power of the reactor in watts, and after it is multiplied by 2, the resulting number is divided by exhaust velocity. Since Isp is 900 and G_0 is 9.8, the exhaust velocity is always 8829, while the combined reactor power is 6.5 GW, the amount of thrust generated is 1.47 MN.

After, to calculate mass ejection rate, which will then be used to calculate burn time, the equation $M_{eject} = \frac{T}{V_e}$ was used. In this equation, T is the thrust in Newtons (N), which is again divided by exhaust velocity. In the end, the mass flow rate is 166.7 kg/s, meaning that every second, 166.7 kg of propellant is released during the burns. Then, to calculate burn time, the equation $T_B = \frac{M_{prop\ loss}}{M_{eject}}$ was used. Since there are multiple burns that are associated with this mission, the $M_{prop\ loss}$ changes, but thrust, power, exhaust velocity and mass flow rate are the same.

To summarize all the information mentioned and evaluate whether the amount of propellant is enough to deliver its crew and payload to Mars at a hyperbolic excess velocity of 10 km/s leaving Earth's Sphere of influence, will be important to calculate the transit time. Table 2 shows the burn time, Delta-V changes and mass changes, as the rocket leaves Earth and approaches Mars, with the goal of reaching a stable orbit of 350 km above the Martian surface. For note, the spacecraft will start the mission at 400 km above Earth's orbit.

Table 2. *A mission profile from 400 km low Earth orbit (LEO) to a stable 350km low Mars orbit. The initial velocity, final velocity, Delta-V for burns, initial and final masses after each burn and the burn time are shown. Section A gives the checkpoints from 400km LEO to Earths sphere of influence (SOI), with a hyperbolic excess velocity of 10 km/s. Section B presents the Mars arrival, aerobraking and orbital insertions, leading to a 350km circular Mars orbit.*

Section A: Earth Departure Checkpoint #	**#1: 400 km orbit above Earth surface.**	**#2: Prograde burn to Mars**	**#3: Spacecraft leaves Earth sphere of influence**
$V_{initial}$ (km/s)	7.67 km/s	7.67 km/s	14.75 km/s
V_{final} (km/s)	7.67 km/s	14.75 km/s	10.00 km/s
ΔV (used in rocket Equation)	N/A	7.07 km/s	N/A
$Mass_{initial}$ (kg)	1,903,000 kg	1,903,000 kg	854,414 kg
$Mass_{final}$ (kg)	1,903,000 kg	854,414 kg	854,414 kg
Burn Time (s)	N/A	6290s	N/A

Section B: Mars Arrival Checkpoint #	**#4: Interplanetary space between Earth and Mars Sphere of Influence**	**#5: Retrograde burn to decrease velocity before aerobrake**	**#6: Aerobrake in Mars atmosphere**	**#7: Retrograde burn after leaving Mars atmosphere**	**#8: Trajectory correction to achieve stable 350 km orbit**
$V_{initial}$ (km/s)	10.00 km/s	8.55 km/s	6.39 km/s	5.50 km/s	3.298 km/s
V_{final} (km/s)	7.07 km/s	6.30 km/s	5.50 km/s	3.556 km/s	3.306 km/s
ΔV (rocket equation)	N/A	2.55 km/s	N/A	1.944 km/s	0.008 km/s
$Mass_{initial}$ (kg)	854,414 kg	854,414 kg	662,204 kg	662,204 kg	531,333 kg
$Mass_{final}$ (kg)	854,414 kg	662,204 kg	662,204 kg	531,333 kg	530,851 kg
Burn Time (s)	N/A	1153 sec	N/A	785 sec	3 sec

Trajectory Methods

Determining the transfer trajectory and the optimal angular positions of Earth and Mars, is important for knowing travel time. For modeling purposes, I assumed that both planets orbit in perfect circles and are coplanar. This should not seriously affect the result because according to Nasa's J2000 mean orbital elements, the orbital eccentricity of Earth is 0.0167 and Mars is 0.0934. In addition, the inclination of Mars is 1.85 degrees above the solar plane, while Earth is 0 degrees above (11). These small changes will not significantly alter the transfer orbit, or the travel time goal.

For the trajectory, gravitational parameters, orbital velocity, orbital distance, travel time, and hyperbolic excess velocities of Earth and Mars were all considered. These numbers were found by solving for the semi major axis (a) in the vis-viva equation. The semi-major axis is used to find eccentricity, orbital energy, perihelion and aphelion. For context, the vis-viva equation is shown below.

$$V = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}$$

Additionally, the vis-viva relationship was used to determine the V-infinity of Mars at varying V-infinity Earth values to ensure greater accuracy with the trajectory and travel time. This equation also relates the orbital speed to the position of the spacecraft along its trajectory if the gravitational parameter of $\mu_{sun} = GM_{sun}$.

While the heliocentric motion does govern the spacecraft's trajectory, the spacecraft's trajectory is also affected by the gravitation fields of Earth and Mars. To model these positions, patched conic approximations are employed. Under this framework, the spacecraft's trajectory is separated into two body segments. The gravitational influence of the sun and each planet's gravitational influence switching at their respectable sphere of influence are used.

$$\frac{d^2r}{dt^2} = -\frac{\mu_{object}}{r^3_{object}} r_{object}$$

Where r_{object} is the distance between the body's center of mass and the spacecraft. In this case, the bodies are Earth, Mars and the Sun. Table #3 clarifies the "object" that is being referenced in this equation, at different points of the trajectory (Table 3).

Trajectory Modeling Results

With these methods in mind, the vis-viva equation and patched conics were coded using python. The inputs are V-infinity values, both for leaving Earth and encountering Mars. To further simplify the code and calculations, the launch window was verified calculated for a given travel time, by considering the hyperbolic excess velocities of Earth and Mars. Figure 2 shows the optimal transfer window, trajectory and transit time of the rocket at the launch window. Furthermore, Figure 3 shows the full heliocentric orbit of the spacecraft, while Table 4 describes the parameters of this orbit. The parameters listed in Table 4 correspond to the full heliocentric orbit around the sun, which is determined by the departure velocity. Even though the aphelion does exceed Mars orbital radius from the sun, the spacecraft reaches Mars arrival position prior to reaching the aphelion. The launch window calculations in Figure 2 takes this, and the transit time calculations into account.

Table 3. *The solution to what the "object" is being referred to in the patched conic equation (shown above) is presented at positions along the spacecraft's trajectory.*

Position	μ_{object}	r_{object}
Near Earth Departure	Object = Earth	Object = Earth
Near Mars Arrival	Object = Mars	Object = Mars
Interplanetary Space	Object = Sun	Object = Sun

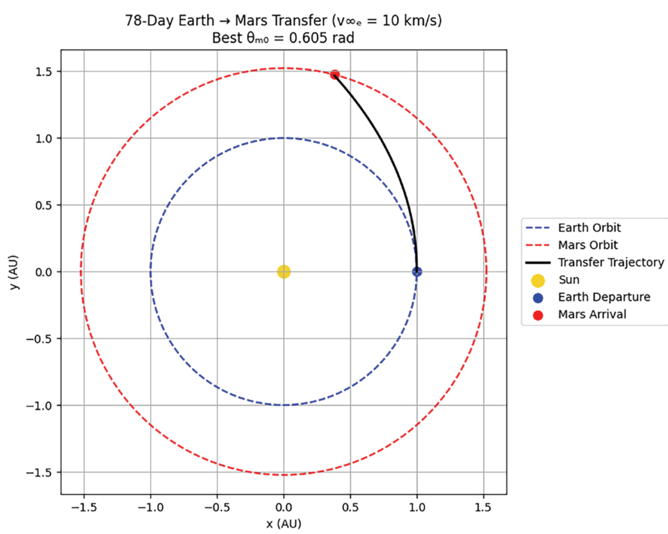

Figure 2. *Transfer window for the conceptual NTP mission. The optimal launch window () is presented as 0.605 radians, which is equivalent to around 34.6 degrees. This means that the optimal launch period is when Mars is 34.6 degrees ahead of Earth in its orbit. Also, the full transfer trajectory, and positions of Earth departure and Mars arrival are included.*

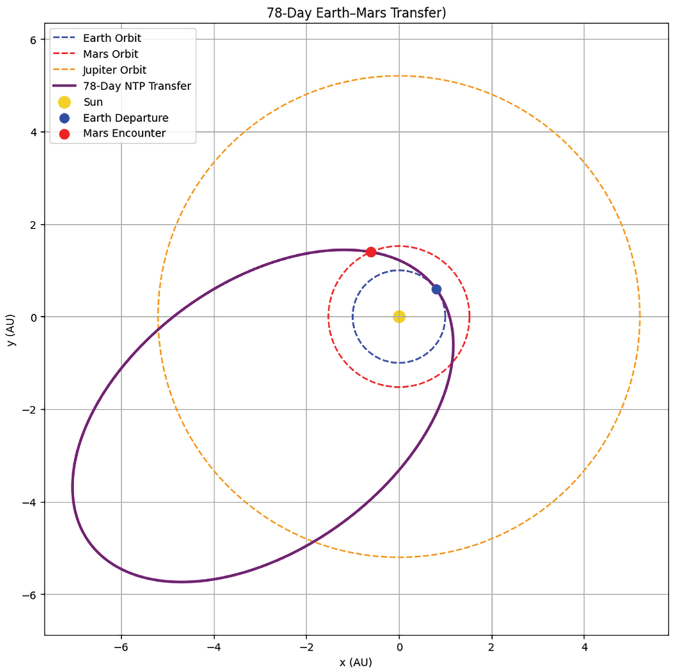

Figure 3. *The full trajectory of the conceptual spacecraft around the sun. Note that Earth departure and Mars arrival positions are plotted on the figure, as shown in the legend. Since there is greater orbital energy, as the hyperbolic excess velocity leaving Earth is 10 km/s, the aphelion swings out beyond Jupiter orbit, and Mars encounter occurs before the spacecraft reaches its farthest point from the sun.*

Table 4. *The semi major and minor axis, perihelion and aphelion, eccentricity and orbital period of the orbit shown in figure 3 are presented. Note that the aphelion distance exceeds Mars' radius because this transfer corresponds to a high energy transfer, which causes the aphelion distance to increase. Finally, as shown in the figure, Mars encounter happens before the spacecraft reaches aphelion.*

Semi Major Axis (AU)	4.64 AU
Semi Minor Axis (AU)	2.88 AU
Perihelion (closest point to Sun, AU)	0.9976 AU
Aphelion (farthest point to Sun, AU)	8.28 AU
Eccentricity	0.784
Orbital Period (years)	9.98 years

Scenarios Evaluated

To evaluate the feasibility of a crewed Earth-Mars Nuclear Thermal Propelled Rocket, multiple mission scenarios were evaluated, to find the most effective and realistic scenario. These different scenarios represented the trade-offs between travel time, propellant mass fractions and reactor power requirements.

In the scenario involving a hyperbolic excess velocity leaving Earth of less than 8 km/s, the transfer time to Mars is greater than 150 days. In this case, the potential of using NTP is not effectively utilized, and although the mass ratios are more favorable, the added complexity, costs and resources involved would not be worth the decreased travel time.

In the moderate scenarios involving a hyperbolic excess velocity leaving Earth of 10-12 km/s, an optimal balance is reached in terms of performance and feasibility, while utilizing the potential of NTP. Even though the mass ratios would be less favorable than the first case, the decrease in travel time would be more pronounced, with a travel time of around 70-90 days. Additionally, reactor power limits and the significant decrease in travel time allow the cost of using NTP to be worth the benefit. This scenario represents the most realistic mission profile and architecture for a manned Earth-Mars mission that utilizes NTP.

The high delta-V scenarios involving a hyperbolic excess velocity leaving Earth of greater than 12 km/s allow for the fastest transfers between Earth and Mars ($\leq$ 70 *days*). While achieving these Delta-V values are achievable with light, robotic probes, it would be impractical for a manned mission. This is because the required propellant mass fractions and reactor power increase dramatically, which leads to unfeasible spacecraft masses and reactor technology that don't currently exist. Therefore, when considering a manned Earth-Mars mission, this scenario is currently unrealistic with current technology.

Validation

It is important to compare the results of the Earth-Mars NTP to the current Hohmann Transfer to understand the value of using NTP. Firstly, the launch window for an NTP spacecraft, with an 80 day time of flight, is when Mars is 36.15 degrees ahead of Earth, according to figure 4. In comparison, the angular position for the Hohmann transfer to Mars is around 44 degrees ahead of Earth at launch (12). Furthermore, after launching the rocket from Earth orbit, the hyperbolic excess velocity is 10 km/s. As the spacecraft follows its heliocentric transfer trajectory to Mars, its approach velocity decreases to around 7.07 km/s, when entering the Martian sphere of influence. In comparison, for the average Hohmann transfer, the hyperbolic excess velocity is about 2.9 km/s, and the approach velocity of Mars sphere of Influence is 2.65 km/s, which were derived from the NASA Planetary Fact Sheet. This results in the NTP mission to Mars being around 80 days, while the Hohmann transfer has a travel time of about 259 days according to a graph of a Hohmann transfer to Mars, published by NASA (12).

RESULTS AND DISCUSSION

Δv capability

High propellant mass fractions are key to achieving high velocity, even with the extra Isp that's offered with using Nuclear Thermal Propulsion. For instance, with most chemical rockets, Delta-V values above approximately 4 km/s are impractical as propellant takes up almost 61% of all the mass of the rocket. On the other hand, Nuclear Thermal Propelled rockets can realistically achieve a delta V of more than 9 km/s due to the extra Isp, as shown in Figure 4. Furthermore, mass ratios need to take into account the change in velocity while burning to leave Earth and the burns needed for deceleration when near Mars. For this reason, the mission profile involves a Delta-V of 7.07 km/s and not 9 km/s in the burn leaving Earth's orbit. Even though that delta-V involves a 45% propellant mass burn, the rocket will still need enough propellant to do burns to decelerate when approaching Mars, as well as reserves for a possible return to Earth (since in-space refueling may not be feasible until much further in the future).

To find Propellant Mass percentage, the rocket equation was rearranged for ($\frac{M_1}{M_0}$), which yields the equation:

$$\text{Propellant Mass } \% = 100\left(1 - e^{\frac{-\Delta V}{V_e}}\right)$$

Where $V_e = Isp \times g_0$

Transit-Time Analysis

The reason why NTP is considered, is because of its ability to drastically reduce travel time to Mars. Interplanetary space is not a safe place for humans to be in for extended time, due to weightlessness, radiation exposure, the need of resources such as food and water, and the limited emergency response capability. Due to this, the travel time to Mars should be shortened by as much as possible, to ensure the greatest chance of success. For greater detail, Figure 5 compares the transfer orbits of the Hohmann transfer, the NTP case transfer and the higher-energy NTP transfer. Furthermore, Table 5 uses a table to compare important mission parameters for the Hohmann Transfer and NTP transfer.

Lastly, it is important to consider the V-infinity effects on the transfer window, which drastically affects the travel time to Mars. As the V-infinity and Delta-V increase, the transfer window involves Mars being closer ahead to Earth, while low velocity transfers, like the Hohmann, involve Mars being farther ahead in its orbit, relative to Earth. In this case, the transfer window for the NTP scenarios involves Mars being about 36 degrees ahead. To contrast, the Hohmann transfer involves Mars being 44 degrees ahead.

In addition, it is important to find the relationship between increasing excess hyperbolic velocity values when leaving Earth to transit time. To construct Figure

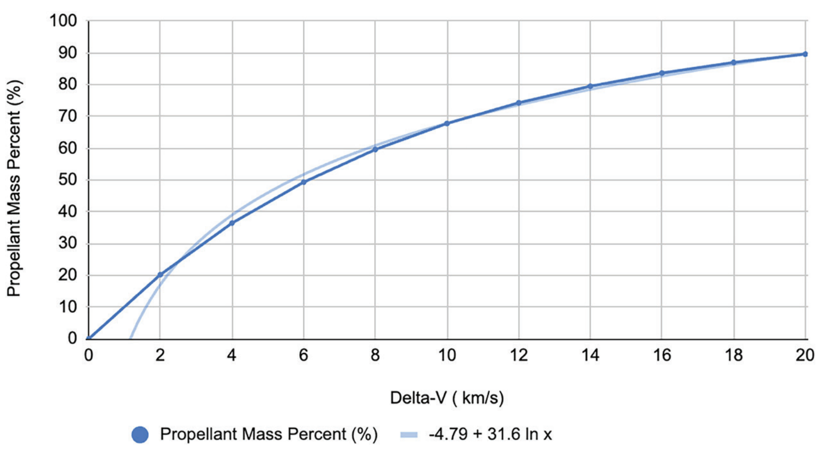

Figure 4. The relationship between Propellant Mass percent and Delta-V is concave down. This means that every successive increase in Propellant Mass Percentage results in a greater marginal Delta-V increase. The line of best fit, which has an equation of , and shown in light blue, has a correlation coefficient of 0.995. This line is used to input Delta-V values and output the corresponding propellant mass fractions, assuming that Isp is 900s.

6A and 6B, varying V-infinity values from 10-12 km/s were added with each value increasing by 0.25 km/s. Figure 6A shows only the Earth-Mars trajectory for each scenario. Meanwhile, Figure 6B zooms out and shows the full orbit of each of the 9 scenarios. Table 6 shows the hyperbolic excess velocity values of Mars from each corresponding V-infinity of Earth and shows the transit time for each scenario.

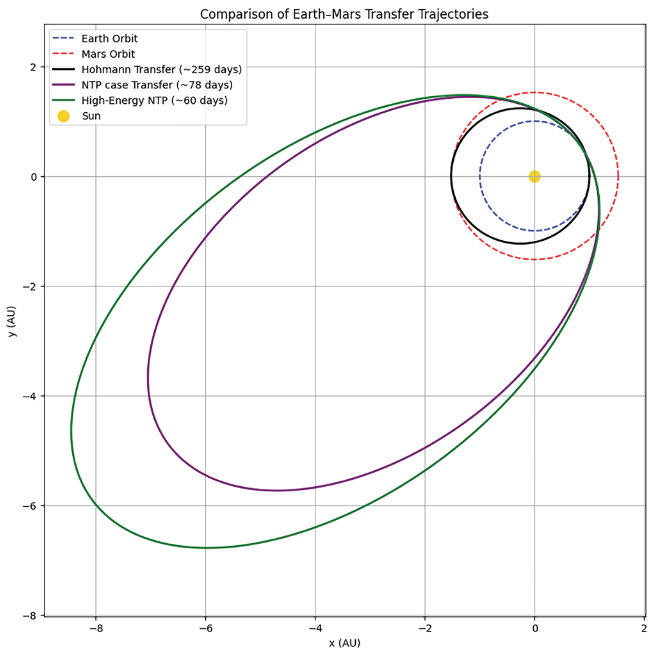

Figure 5. *Comparison between the Hohmann Transfer, high energy NTP transfer, and the conceptual NTP transfer from Earth to Mars. The difference in orbital energy and transfer time is clearly noticed in the orbits. The Hohmann transfer barely makes it to Mars orbit and takes longer to intersect with its orbit. On the other hand, the NTP transfers intersect Mars orbit earlier and their apoapsis extends far beyond Mars orbit.*

Table 5. *Comparison of the transfer window, specific impulse, propellant mass fraction, delta-v, v-infinity and transit time for the conceptual NTP transfer and the Hohmann transfer that is commonly used by chemical propulsion.*

Parameter	Nuclear Thermal Propulsion	Chemical propulsion
Transfer Window	36 degrees	44 degrees
Isp	900s	450s
Propellant Mass Fraction	0.55	0.80
Delta-V	7.08 km/s	3.6 km/s
V-infinity	10 km/s	2.9 km/s
Travel Time	~78 days	~259 days

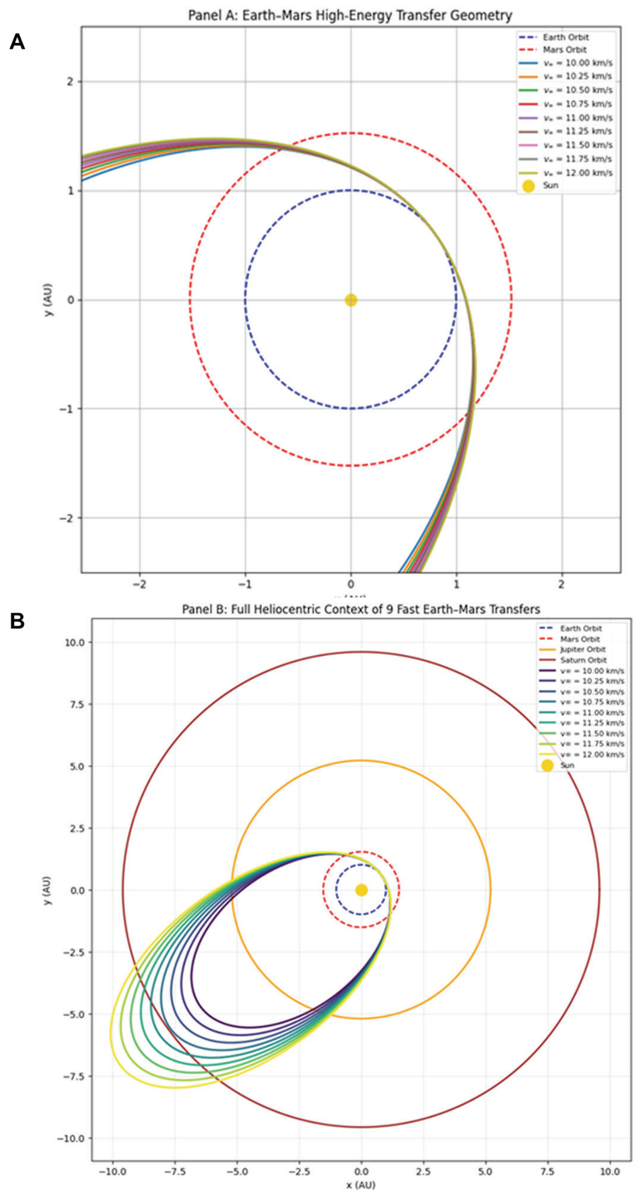

Figure 6. *Earth–Mars transfer trajectories for nine mission scenarios with increasing orbital energy. (A) Transfer paths from Earth to Mars illustrating decreasing transit time as orbital energy increases; differences in travel time become progressively less pronounced at higher energies. (B) Full heliocentric orbits for the nine transfer scenarios. Increasing orbital energy results in a near-linear increase in apoapsis distance. All scenarios originate from the same launch position, while launch windows and arrival positions are adjusted to maintain realistic Earth–Mars alignment.*

Table 6. *For each of the 9 scenarios, the corresponding hyperbolic excess velocities of Earth and Mars, in addition to its transit time are presented. The scenario that each row corresponds to is shown in the legend of figure 6 A and B.*

Earth V-Infinity (km/s)	Mars V-Infinity (km/s)	Travel Time (days)
10	7.07	78.7
10.25	7.38	77.4
10.5	7.70	76.2
10.75	8.01	75.1
11	8.33	74.0
11.25	8.64	72.9
11.5	8.96	71.9
11.75	9.27	70.9
12	9.58	70.0

Sensitivity Analyses

While these optimized transfer cases demonstrate travel time benefits of NTP, it is necessary to also evaluate the sensitivity these results are, compared to changes in mission parameters and the implications they have for the mission.

One parameter that can be changed that could affect the mission is differences in delta-v. This change in Delta-V when doing the low-Earth-orbit burn to Mars has a positive relationship with v-infinity. This is because all scenarios start at 400 km above Earth's surface and have a constant orbital velocity there. Thus, as Delta-V is increased, a higher velocity is reached, allowing the rocket to leave Earth and arrive at Mars faster. Figure 7 shows that the curve between travel time and departure delta-v is concave up but decreasing, meaning that there are diminishing returns for travel time in every successive increase in delta-v.

The specific impulse (Isp) is another parameter that could affect the mission. Isp is directly related to exhaust velocity, which is used for computing propellant mass ratios, as illustrated in the rocket equation. Since Isp and exhaust velocity have a positive linear relationship, an increase in Isp results in an increase in exhaust velocity. For this paper, the NTP rocket discussed has an Isp of 900s, which is a conservative estimate as this number was taken from the NERVA program, during the 1960s and 1970s. It is important to consider that anything above 1000s is unfeasible with current technology. Figure 8 shows the relationship between Isp and propellant mass ratio, for a delta-V of 7 km/s. This specific delta-v was used as this value is consistent with the delta-v for the conceptual mission profile in Table 2. This table shows that with an Isp of 450s, around 80% of all the wet mass of the rocket would need to be propellant to achieve this delta-v. On the other hand, with Isp of 900s, around 55% of all the wet mass would need to be propellant.

Lastly, reactor power is another parameter that could affect the mission as well. As shown in section 2.2,

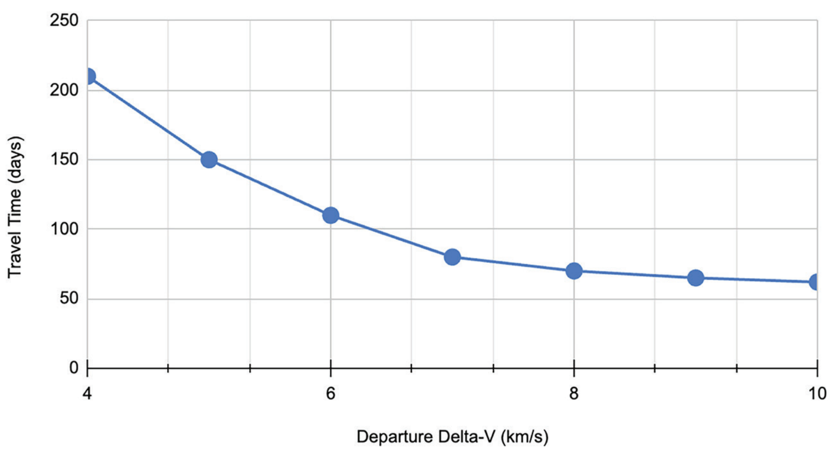

Figure 7. The relationship between departure delta-v when in a 400 km orbit (7.67 km/s orbital velocity) above Earth and the corresponding transit time to Mars introduces diminishing returns for each increase in delta-V. Since the table is made by making intervals of 1 km/s, a delta-V of 3 km/s was not included as the rocket would fail to leave Earth's sphere of influence. The transit time was calculated with the help of two body and patched conic approximations during the optional launch window.

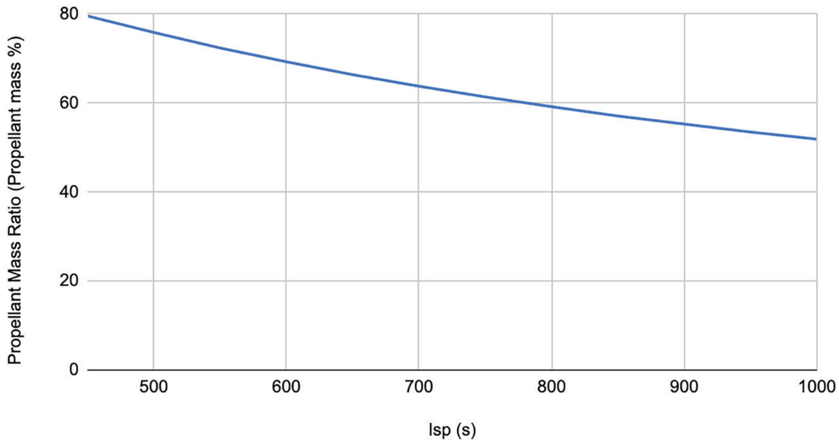

Figure 8. *For a burn involving a delta-v of 7km/s, an increase in isp leads to a decrease in the mass percentage of propellant that is needed to reach this velocity. Also, this curve is concave up but decreasing, which represents diminishing returns, meaning that each increase in Isp contributes slightly less to the decrease in propellant mass ratio.*

reactor power has a positive relationship to thrust, which has a positive relationship to propellant mass flow rate and burn time. For this paper, after considering technical advances since NERVA and the usage of 5 reactor-thruster components, the reactor power was assumed to be 6.5 GW. As a result, the burn time when leaving Earth orbit is 6290 seconds, with a propellant mass flow rate of 166.7 kg/s. On the other hand, if the reactor power was 7 GW for instance, the mass flow rate would go to 180 kg/s, which would shorten the Earth orbit burn to 5825 seconds.

Feasibility of Round Trips

For any crewed mission to Mars, feasibility must extend beyond the one-way transit and consider the capability to return to Earth. Even though in-space refueling for this mission is purely speculative and conceptual, it is imperative to evaluate whether such a rocket could potentially support a round trip.

As illustrated in 2.1 and 2.2, the NTP spacecraft maintains a favorable propellant mass fraction at high Delta-V and V-infinity values. After reaching Mars orbit of 350 km, around 227,000 kg of propellant remain. Using the rocket equation, in the case the rocket uses all remaining propellant, the spacecraft has an available departure Delta-V of 5 km/s from Mars orbit. This leads to an excess hyperbolic velocity of 7 km/s. This is enabled by Mars' lower escape velocity compared to Earth, meaning that the hyperbolic trajectory can be achieved at lower velocities. However, since this case uses all the remaining fuel, it represents a limiting scenario and is not intended to be used, due to the difficulty of deceleration when arriving at Earth.

Instead, the return could be possible if the rocket is partially refueled (150,000 – 200,000 kg refilled) while in orbit on Mars, which could be done with the help of a Mars base and taking advantage of the several water-ice reservoirs on the planet. Still, taking advantage of these reserves are futuristic and should not be used in mission profiles for near-future missions. Instead, the rocket could achieve a lower Delta-V and V-infinity when leaving Mars, which can save on fuel and make it possible to return to Earth with the help of burns and aerobraking, and still possibly reduce return time to Earth as well.

Limitations and Future Work

This study is subject to several modeling and methodological limitations that could affect the applicability of the results. While these simplifications may slightly influence numerical values, they do not alter the trends or conclusions that were produced.

First, orbital mechanics were modeled using heliocentric and patched conic approximations. As discussed in 2.3, coplanar and circular orbits were used to simplify the model. Although these assumptions introduce only minor deviations, real interplanetary missions must account for eccentricities, inclinations and the gravitational interaction between the planets. Additionally, mid-course and trajectory corrections were not explicitly modeled, but were considered when calculating the amount of propellant. Such scenarios in which corrections are needed regularly occur in spaceflight.

Furthermore, the nuclear reactor power was based on scientific estimates when adding NERVA class results to likely technological advancements made sense. Actual flight-ready missions would need to accurately determine reactor power, when considering subtle extra masses and power due to thermal losses, and structural mass growth. Finally, the spacecraft mass models were not explicitly modeled during the study. Even though the truss mass is completely conceptual, the actual mass of the truss would be small relative to the much larger masses of the crew module and reactor. For the heavier parts, such as the crew module and reactor, scientific estimates were used to consider the masses, factoring in all aspects that would need to be included. For example, the crew module estimate was 200,000kg after considering the possible masses of radiation shielding, living areas, life support and cargo holds.

Future work should specialize on a detailed engineering analysis of the spacecraft architecture. This includes analyzing structural design, thermal power management, radiation shielding, costs, and engineering feasibility.

CONCLUSION

The usage of NTP can offer benefits to crew safety, payload capacity and reusable architecture. NTP's shortened transit time improves crew safety by reducing exposure to microgravity, space radiation and psychological challenges that could affect astronauts. NTP can also enable larger payloads of cargo to be launched while still achieving rapid transfers. This capability is key for supporting a Mars presence as large payloads are required for maintaining habitats and life-support, while also bringing supplies and scientific equipment. Finally, the NTP architecture supports the partial reusability of such a spacecraft which lowers mission costs and enabling repeated, rapid Earth-Mars transfers.

Overall, this study evaluates the feasibility of Nuclear Thermal Propulsion for a crewed Earth-Mars mission with the objective of reducing travel time while maintaining feasibility. The results show that with the increased Isp, NTP can achieve higher departure velocities, enabling transfer times of 70-90 days. Meanwhile, the Isp levels and reactor power are consistent with nuclear reactor advancements from NERVA-class technology. Sensitivity analysis further confirms that increases in delta-V continue to reduce travel time, while ensuring that Isp and reactor power are consistent in the results. These results indicate that reduced travel times can be achieved without requiring speculative technology. Additionally, the remaining propellant margins following Mars orbit, suggests that with the help of lower energy trajectories or possible partial refueling in future missions, round trip mission architectures are achievable. Overall, Nuclear Thermal Propulsion represents a technically realistic and advantageous option that should be considered for future study and implementation in the future.

ACKNOWLEDGEMENT

The author would like to thank Kyle Kennedy from Polygence for his mentorship and his valuable contributions to this project.

FUNDING

The author(s) received no specific funding for this work.

CONFLICT OF INTEREST

The author declares that there are no conflicts of interest related to this work.

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