A Simulation and Comparison of Propulsion Systems for a Manned Mission to Mars

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Abstract: Recent developments suggest a manned mission to Mars as a realistic near future possibility, therefore requiring additional consideration into which propulsion system to use. Examples include the traditional chemical rocket; the nuclear pulse engine, which propels a spacecraft using nuclear explosions; the ion thruster, which accelerates propellant using electrically charged grids; and others. This paper evaluates six such propulsion systems using Python computer simulations. Each simulation models a SpaceX Starship spacecraft modified to use one of the six propulsion systems, calculating and returning values for transfer time and propellant mass consumption. These results are analyzed and ranked to determine how the different systems would compare in performance in an actual mission, specifically in the aforementioned criteria of travel time and propellant expenditure.

Introduction

Mars is the fourth planet from the Sun, orbiting at an average distance of about 228 million kilometers with a period of 687 days. It is a terrestrial planet, with a crust and mantle of silicates and a core of iron and sulfur. Possessing Earth-like geological features such as volcanoes, canyons, and water-rich ice caps, as well as seasons and weather patterns, Mars is considered the planet most similar to Earth in the Solar System [1].

Unmanned exploration of Mars began in 1964, with the United States' Mariner 4 space probe being the first successful flyby of the planet. This probe was followed by the first landings on Mars in 1975 by the Viking 1 and 2 spacecraft. Then, the Mars rovers: *Sojourner* in 1997, *Spirit* and *Opportunity* in 2004, *Curiosity* in 2012, and *Perseverance* in 2021 [1]. Today, NASA's Artemis program, whose goal is to resume manned exploration of the Moon, has the express long-term purpose of preparing for manned exploration of Mars [2].

Given that Mars is the most similar planet to Earth, it is the ideal planet for human beings to colonize. Large-scale colonization of Mars is a worthy pursuit since it would safeguard human survival in case of a global disaster that would otherwise render mankind extinct. To establish such a settlement on Mars, a space agency or corporation would need to construct hundreds of heavy-lift spacecraft to carry thousands of volunteers, launch them into Earth orbit, use one of multiple proposed propulsion systems to propel the spacecraft from Earth to Mars, and land them on the Martian surface to begin colonization. However, before long-term Mars habitation is

[1] Basic orbital structure of Python simulations written by project mentor Cody Waldecker. All planet- and propulsion-specific edits made by the author.

possible, it is first necessary to determine which propulsion system to use on the spacecraft to take the colonists there. To explore this question, this paper simulates a hypothetical manned mission to Mars.

One common method of planning a Mars mission is using a porkchop plot, a chart with departing dates on the x-axis, arriving dates on the y-axis, and each coordinate point representing a mission with the corresponding departure and arrival dates. Concentric loops represent groups of missions with certain energy requirements, with the innermost loop representing those missions requiring the smallest energy expenditures. Instead of this method, this paper models the transfer orbit of the spacecraft directly, using departure and arrival coordinates and radius from the Sun in a series of computer programs.

This simulation focuses on a mission to Mars using a Starship as the example spacecraft. Starship is a heavy-lift craft with a dry mass of 130 tonnes, a payload capacity of at least 90 t, and a propellant capacity of 1,200 t. After being launched into orbit by its own engines and the Super Heavy booster, it is meant to be refueled in orbit for a long-term trip [3]. In orbit, the spacecraft might also be refitted with different engines than those with which it is normally equipped, using different propulsion methods for a faster trip or a smaller propellant expenditure. This mission simulation assumes that the payload bay is filled to capacity. The type of mission simulated is the same one-way colonization trip described earlier; therefore, no propellant is carried for a return trip to Earth.

This paper analyzes the various propulsion systems that might take the spacecraft and its passengers on the hypothetical mission to Mars. The choice of propulsion is limited to those systems that have already been designed by engineers, have been tested in experimental settings, and are possible with already-existing sources of energy. This precludes any nuclear fusion engines, photon rockets, or other engines that exist solely in the realm of science fiction. This analysis focuses on six tested propulsion engines: the chemical rocket, the nuclear thermal rocket, the Project Orion nuclear pulse drive, the solar sail, the electrostatic ion thruster, and the electromagnetic plasma thruster. By modeling these six propulsion systems using Python simulations, this paper determines the relative performance of each in a hypothetical Mars mission in terms of reducing transfer time and propellant consumption mass.

Overview of Propulsion Systems

Rocket Equation

All rocket engines operate on the rocket equation, which can calculate the delta-V imparted to a rocket given the starting and ending masses, or vice-versa. This equation isolated for delta-V is

$$\Delta v = I_{sp} g_0 ln(\frac{m_0}{m_f})$$

(1)

Where I_{sp} is specific impulse, g_0 is Earth's gravitational acceleration in $\frac{m}{s^2}$), and m_0 and m_f are the masses of the spacecraft before and after the burn, respectively. Specific impulse is the measure of the efficiency of a rocket, defined as $I_{sp} = \frac{v_e}{g}$, where v_e is the exhaust velocity in $\frac{m}{s}$ and g is Earth's gravitational acceleration in $\frac{m}{s^2}$.

If m_f is equal to wet mass m_0 minus propellant expenditure m_p, the equation isolated for m_p is

$$m_p = m_0(1 - e^{-\frac{\Delta v}{I_{sp}g_0}}).$$

(2)

Mission Trajectories

There are multiple ways to travel between two planets with roughly circular orbits. The Hohmann transfer is the most energy-efficient method, producing the smallest delta-V and consuming the least possible propellant. In a Hohmann transfer, a spacecraft using a high-thrust propulsion system fires its engines tangent to Earth's orbit around the Sun, entering an elliptical orbit around the Sun with its periapsis tangent to Earth's orbit and its apoapsis tangent to Mars'. The formula for the delta-V of the spacecraft departing is

$$\Delta v_D = \sqrt{\frac{GM}{R_1}}(\sqrt{\frac{2R_2}{R_1+R_2}} - 1)$$

(3)

Where G is the gravitational constant, M is the mass of the Sun in kg, and R_1 and R_2 are the radii of Earth and Mars, respectively [4].

Once arriving at Mars 180 degrees around the Sun from its starting point, the spacecraft fires its engines tangent to Mars' orbit in the opposite direction from the previous burn, entering orbit around Mars. The formula for the delta-V of the spacecraft arriving is

$$\Delta v_D = \sqrt{\frac{GM}{R_1}}(\sqrt{\frac{2R_2}{R_1+R_2}} - 1).$$

(4)

The formula for the spacecraft's transfer time from Earth to Mars is

$$t = \frac{\pi}{\sqrt{GM}}(\frac{R_1+R_2}{2})^{\frac{3}{2}}.$$

(5)

This transfer takes a little over 259 days [4]. It is possible to reach Mars in shorter times by expending more propellant and obtaining a higher velocity. For example, a spacecraft could expend as much propellant as possible in the departing burn to produce the highest departing velocity possible while still leaving enough propellant to insert itself into orbit. Alternatively, the spacecraft could use the same amount of propellant to escape orbit as in the Hohmann transfer, then use a second low-thrust engine to accelerate slowly but continuously in the direction of the spacecraft's velocity.

Naturally, a lower travel time implies a higher propellant expenditure, and vice-versa. Which type of trajectory a Mars mission spacecraft should take depends on which of these two criteria the mission planners prioritize most.

Chemical Rocket

The chemical rocket is the default engine for the Starship spacecraft, as well as for most spacecraft currently in existence. It involves a fuel (not containing oxygen) and an oxidizer (containing oxygen) reacting chemically to produce a hot gas that expands out the nozzle to produce thrust. There are two types of chemical rockets: solid-propellant rockets, in which the fuel and oxidizer are suspended in a solid mixture; and liquid propellant-rockets, in which the fuel and oxidizer are stored in liquid form in separate tanks and pumped into a combustion chamber. Chemical rockets have low exhaust velocities and specific impulses (200–470 s) but high thrusts (up to several meganewtons). As a result, they are capable of escaping Earth gravity and making rapid maneuvers, but consume large amounts of propellant quickly [5].

The Starship spacecraft carries six of the Raptor engine, a liquid-propellant rocket that uses liquid methane as fuel and liquid oxygen as oxidizer. The Raptor 2 engine, the version currently in common use, produces 2198 kN of thrust in Earth's atmosphere and 2350 kN in the vacuum of space. It has a measured vacuum specific impulse of 351.5 s.

Nuclear Thermal Rocket

The nuclear thermal rocket (NTR) is a more efficient alternative to the chemical rocket, using a nuclear reactor core to heat a liquid propellant and expel it out of a nozzle to produce thrust. The less massive the propellant, the faster the exhaust velocity; therefore, liquid hydrogen is the preferred propellant for a NTR. The NTR achieves a faster exhaust velocity and a higher specific impulse (around 900 s for liquid hydrogen), but a lower thrust (kilonewtons as opposed to meganewtons) compared to a chemical rocket. Due to the danger of radiation, the NTR is not suitable for lifting a spacecraft off of Earth's surface. Rather, it is better suited for orbital transfer in space. Although prototypes have been built and tested extensively on the ground, the NTR has never been used in space [5].

NASA studied NTRs in 1955 under the Rover program. Over eighteen years, they produced and tested several nuclear reactors and rocket engines, their final product being the NERVA XE-Prime engine in 1969. This engine weighs 18,144 kilograms and possesses a specific impulse of about 841 seconds [6]. A single engine of this model propels the nuclear-propelled variation of the Mars mission propulsion, in place of the usual Raptor engines.

Nuclear Pulse Propulsion

Nuclear pulse propulsion differs from other reaction-based propulsion systems in that instead of expelling gas backward to produce thrust via Newton's third law, a spacecraft using this system expels a nuclear bomb. The bomb explodes and causes the debris to collide with the craft, imparting force and acceleration directly. Such a spacecraft has no rocket nozzles at the tail, but a circular "pusher plate" of steel or aluminum, with a hole through the center for the bomb or "pulse-unit" to pass through. The plate is supported by a two-staged shock-absorbing apparatus, the first stage consisting of multiple layers of toroidal, concentric gas bags, and the second stage consisting of long telescoping piston-legs. The pulse-unit, a fission bomb of one to five kilotons in yield, is shaped such that the energy of the nuclear explosion is directed toward a slab of tungsten propellant facing the spacecraft. The effect is that the tungsten is vaporized into a plasma that collides with the pusher plate, accelerating it by thousands of g's. The gas bags absorb this force and spread it out over time, reducing the acceleration to a few hundred g's, and the pistons further reduce it to one to two g's, making the trip survivable for human passengers. To prevent ablation of the pusher plate, a layer of oil is sprayed over the plate between explosions. To protect the crew from the radiation of the nuclear explosions, either the pulse-unit storage bays are placed between the engine and the crew habitat, or a shielded radiation shelter is built into the spacecraft structure [7].

This mode of transport was the focus of Project Orion, a U.S. government project from 1958 to 1965. This project developed the nuclear propulsion concept and ran experiments with miniature models propelled by non-nuclear bombs, but never produced a working prototype of a nuclear bomb-propelled engine. It was cancelled due to the Partial Test Ban Treaty of 1963, which prohibited nuclear tests in space over concerns of nuclear fallout and Cold War escalation [7].

Although Project Orion-type nuclear pulse propulsion is illegal and cannot be used in any space program, it is worth studying since it is capable of producing both high thrust and high specific impulse (high estimates range from 4,000 to 6,000 s), combining the best features of chemical rockets and electric propulsion, respectively. As a result, it could propel spacecraft as large as 4,000, 40,000, or 8 million tonnes anywhere in the Solar system in relatively short trip times. It is also the only propulsion system within the limits of modern technology capable of sending human beings to other star systems in less than 150 years [7].

The Project Orion engineers developed several different concepts of the nuclear pulse vehicle, the latest being the small-scale version developed for NASA's Mars mission plans. This design was to have a pusher plate 10 meters in diameter and an engine weight of 100 tons (90718.5 kg), use one-kiloton, 311-pound (141.067 kg) pulse-units, and be launched into space using the Saturn V rocket. This version is simulated in the hypothetical Mars mission, since its diameter best matches the width of the actual Starship spacecraft [7].

Solar Sail

The solar sail is arguably the simplest propulsion system that provides low continuous thrust to a spacecraft. It is a large sheet of thin, reflective material that receives solar radiation. As the photons collide with the sail, they exert a small pressure on it. Although this pressure is negligible over the short term, it increases the spacecraft's velocity over many months. The greatest advantage of the solar sail is that it requires no propellant. However, since the acceleration from sunlight is so low, other forms of propulsion are necessary to propel the spacecraft out of Earth's orbit and into Mars' within reasonable time frames [8].

Farres, Webster, and Folta give the formula for solar radiation pressure as

$$P_{srp} = P_0(\frac{R_0}{R_{sun}})^2 \tag{6}$$

In pascals, where $P_0 = 4.57 \times 10^{-6} N$ is the radiation pressure at 1 AU from the Sun, $R_0 = 150 \times 10^9 m$ is the distance from the Earth to the Sun, and R_{sun} is the distance from the spacecraft to the Sun [9]. Given that force is pressure times area, the acceleration caused by radiation on the sail is pressure times area over mass.

This propulsion system was first tested in 2010, on the Japanese IKAROS space probe [10].

Ion Thruster

The ion thruster is an electric propulsion engine, using electricity to accelerate propellant and produce thrust. Specifically, it is an *electrostatic* engine, which accelerates ions created by removing electrons from atoms. An ion thruster feeds neutral atoms into a chamber where a cathode bombards them with electrons, knocking out an electron from each atom's orbitals. The positively charged ions are directed toward negatively charged grids, which accelerate them to high speeds and expel them to produce thrust. Ion thrusters are low-thrust, producing millinewtons of thrust. As a result, they cannot escape Earth's atmosphere and take days to break out of Earth's orbit; they must be launched off the surface and out of orbit using more powerful engines. However, once out of orbit, they can accelerate continuously for months without having to refuel, shortening trip times by days. This is due to their high exhaust velocities and specific impulses, ranging from 2,500 to 10,000 s. Among propellants, xenon is preferred for an ion thruster because it is inert and has high atomic mass, producing more thrust per unit of power expended [11].

Goebel and Katz give the thrust of a xenon ion thruster as

$$T = 1.65 I_b \sqrt{V_b} \tag{7}$$

In millinewtons, here I_b is the beam current in amperes and V_b is the voltage. A thruster with a beam current of 2 A and a voltage of 1500 V produces a thrust of 122.4 mN [11].

They give the specific impulse of a xenon ion thruster as

$$I_{sp} = 123.6\gamma\eta_m\sqrt{V_b} \tag{8}$$

In seconds, where γ is the thrust correction coefficient and η_m is the mass utilization efficiency. If $\gamma = 0.958$ and $\eta_m = 0.9$, specific impulse is 4127 s [11].

Forty of these hypothetical thrusters are simulated in the ion-propelled modification of the Mars mission simulation. For escaping and entering orbit as well as powering the thrusters, a NERVA nuclear engine is included in the spacecraft mass.

Plasma Thruster

The magnetoplasmadynamic (MPD) thruster is another form of electric propulsion, an *electromagnetic* engine that uses magnetic fields to accelerate a plasma. Consisting of a hollow cylindrical anode around a central cathode, it relies on the Lorentz force between these two electrodes to accelerate the plasma and produce thrust. Due to the high power requirements, MPD thrusters tend to have the highest thrust of any electric propulsion engine [11]. Propellants used for MPD thrusters include hydrogen, lithium, and argon. Two main types of MPD thrusters exist: self-field and applied field [12], the latter type being simulated in this hypothetical mission. MPD thrusters have been tested in space as early as 1995 [13].

Myers gives the formula for the thrust of an applied-field MPD thruster as

$$T = \frac{1}{\sqrt{2}}BJr_a[1 - \frac{3}{2}(\frac{r_c}{r_a})^2] \tag{9}$$

In newtons, where B is the magnetic field strength in teslas, J is the discharge current in amperes, and r_c and r_a are the diameters of the cathode and anode respectively [12].

The applied-field thruster simulated in this hypothetical mission is one of a set of small prototypes developed by NASA in 1989. It uses argon propellant and has a cathode diameter of 12.7 mm, an anode diameter of 26 mm, a magnetic field strength of 0.3 T, a discharge current of 1500 A, and a measured specific impulse of about 1100 s [14]. Forty of these plasma thrusters, as well as the same NERVA engine as in the ion simulation, are simulated and included in the spacecraft mass.

Simulation of Propulsion Systems

As mentioned, each of the six propulsion systems is modeled in a Python simulation, representing a transfer of a spacecraft from Earth to Mars on a two-dimensional representation of the Solar System. (Although Mars has a 1.85 degree orbital inclination in respect to Earth's orbit, this difference is not modeled here for this initial investigation). At the start of each simulation, Earth is placed on the positive X-axis at the distance of its semimajor axis: 150 million km (Mars' orbit is not simulated until after Earth and the spacecraft have finished being simulated). Using the known values of $M = 1.989 \cdot 10^{30} kg$ for the mass of the Sun and $G = 6.67 \cdot 10^{-20} \frac{m^3}{kgs^2}$ for the gravitational constant, the velocity of Earth in the positive y-direction is calculated as

$$v = \sqrt{\frac{GM}{R}} \tag{10}$$

And imparts on all simulated objects a continuous gravitational acceleration toward the Sun expressed according to Newton's Law of Universal Gravitation as

$$a_x = -\frac{GM}{R^3}x \quad a_y = -\frac{GM}{R^3}y \tag{11}$$

Where R is the radius of the planet from the Sun in km, and x and y are the x- and y-coordinates of the object in km, respectively.

The simulated spacecraft starts at Earth's same position on the Cartesian plane, begins moving with positive y-velocity, is constantly under the gravitational influence of the Sun, and is considered to have arrived at Mars when its radius from the Sun has reached 228 million km, Mars' semimajor axis. After the spacecraft completes its trip, Mars' orbit is drawn in reverse starting from the spacecraft's position to finish at Mars' start point.

This simulation assumes circular orbits for both Earth and Mars, unlike reality in which Earth and Mars both have elliptical orbits with non-zero eccentricities. However, simulating circular orbits simplifies many design variables required to calculate travel time and propellant consumption, so circular orbits are simulated here.

Chemical Rocket

There are two simulations for this propulsion system. The first uses the Hohmann transfer formulas to simulate a trajectory from Earth to Mars using the Raptor chemical engines. It implements Equation 3 to calculate the mass of propellant consumed. This is achieved by changing m_f to $m_0 - m_{propellant}$, then isolating the equation for $m_{propellant}$. The specific impulse substituted into this equation is 351.5, the value measured for the Raptor engine [3].

The trajectory graph and data returned are as follows:

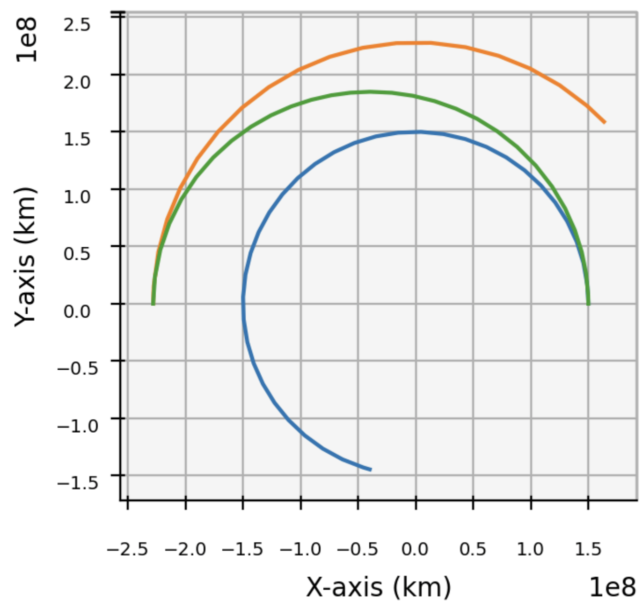

Fig. 1: Trajectories of Earth (blue), Mars (orange), and the spacecraft (green) under Hohmann transfer

Table 1: Chemical Rocket (Hohmann)

Transfer Time (days)	259.387
Delta V at Departure (km/s)	2.925
Delta V at Arrival (km/s)	2.632
Departing Propellant Expenditure (t)	640.543
Arriving Propellant Expenditure (t)	256.053
Total Propellant Expenditure (t):	896.596

By iteratively reducing the starting propellant mass and rounding up to a multiple of 50 t, the minimum possible starting propellant mass is 900 t.

The second simulation models a spacecraft that consumes the most propellant possible to produce the greatest departing delta-V while keeping total propellant consumption within the spacecraft's capacity of 1200 t.

The graph and data for this simulation are as follows:

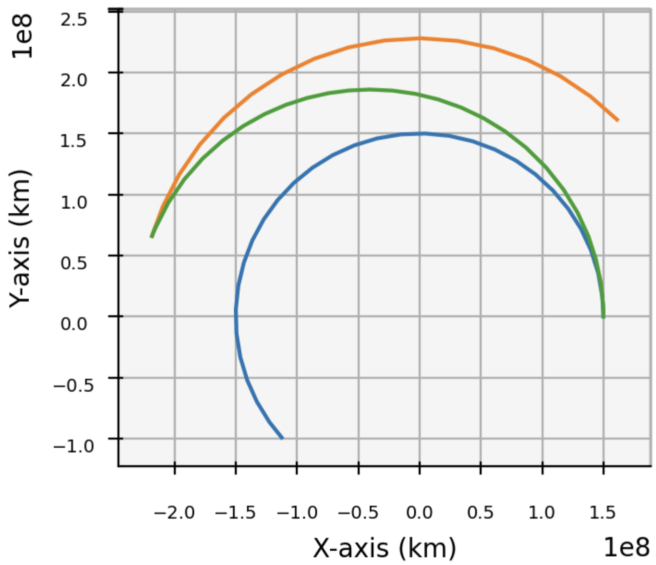

Fig. 2: Trajectories of Earth (blue), Mars (orange), and the spacecraft (green) under fast chemical propulsion

Table 2: Chemical Rocket (Fast)

Transfer Time (days)	225.586
Delta V at Departure (km/s)	2.998
Delta V at Arrival (km/s)	3.068
Departing Propellant Expenditure (t)	825.000
Arriving Propellant Expenditure (t)	350.700
Total Propellant Expenditure (t):	1175.700

By iteratively increasing the departing propellant expenditures and rounding down to a multiple of 25 t, the maximum possible departing propellant expenditure is 800 t. Although this approach uses more propellant, it reduces travel time by over a month.

Nuclear Thermal Rocket

Similarly to the chemical rocket, two simulations model this system: one where the spacecraft leverages a Hohmann transfer using the NTR, and another where it consumes the most propellant possible to produce the greatest departing delta-V. The specific impulse for this engine is 841 s, as recorded from the Rover project report, and the engine mass of 18,144 t is added to the vehicle mass [6].

The data for the nuclear thermal Hohmann transfer is as follows (the graph is the same as for the chemical Hohmann transfer):

Table 3: Nuclear Thermal Rocket (Hohmann)

Transfer Time (days)	259.387
Delta V at Departure (km/s)	2.925
Delta V at Arrival (km/s)	2.632
Departing Propellant Expenditure (t)	145.735
Arriving Propellant Expenditure (t)	93.565
Total Propellant Expenditure (t):	239.301

The amount of propellant necessary for the Hohmann transfer using the NTR engine is lower than for the chemical engine, due to the former's higher specific impulse.

By comparison, the graph and data for the fast NTR transfer are as follows:

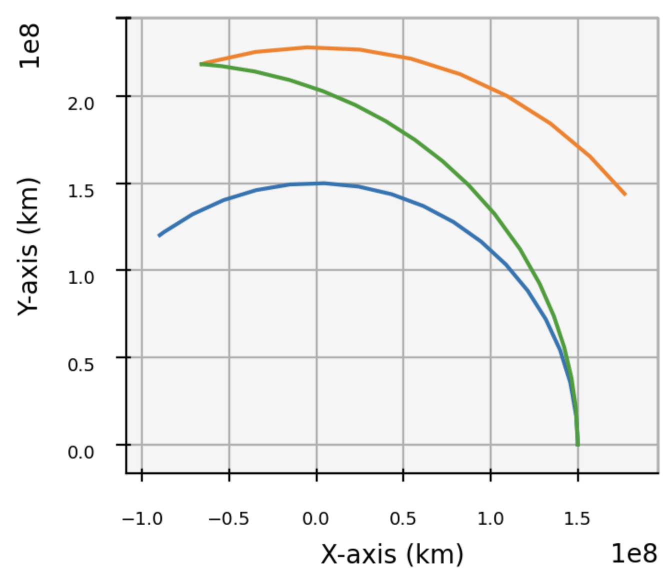

Fig. 3: Trajectories of Earth (blue), Mars (orange), and the spacecraft (green) under nuclear thermal propulsion

Table 4: Nuclear Thermal Rocket (Fast)

Transfer Time (days)	129.191
Delta V at Departure (km/s)	4.960

Delta V at Arrival (km/s)	8.913
Departing Propellant Expenditure (t)	650.000
Arriving Propellant Expenditure (t)	520.695
Total Propellant Expenditure (t):	1170.695

Using the same process as the fast chemical transfer, the resulting maximum departing propellant expenditure is 650 t. This amounts to a minimum transfer time of about 129 days, more than halving the duration of the Hohmann transfer at the cost of more than quadrupling the propellant expenditure.

Nuclear Pulse Propulsion

As with the NTR, this mode of propulsion has two simulations modeling two scenarios: a Hohmann transfer and a fast transfer. The specific impulse is estimated from Dyson at 3000 s, and the engine mass of 90,718.5 t is added to the vehicle mass. Propellant expenditure is reported in pulse-units rather than tonnes, with each pulse-unit having a mass of 141.067 kg. With the spacecraft's total propellant capacity being 1,200 t, 8,506 pulse-units can be carried aboard at maximum [7].

The data for the Hohmann transfer is as follows:

Table 5: Nuclear Pulse Propulsion (Hohmann)

Transfer Time (days)	259.387
Delta V at Departure (km/s)	2.925
Delta V at Arrival (km/s)	2.632
Departing Pulse Unit Expenditure (t)	36.113
Arriving Pulse Unit Expenditure (t)	29.624
Total Pulse Unit Expenditure (t)	65.737

The minimum number of pulse-units carried at the start, as calculated using the same process as for other Hohmann transfers, is 500. Of these 466 are expended, with a combined mass of 65,737 t.

By comparison, the graph and data for the fast transfer are as follows:

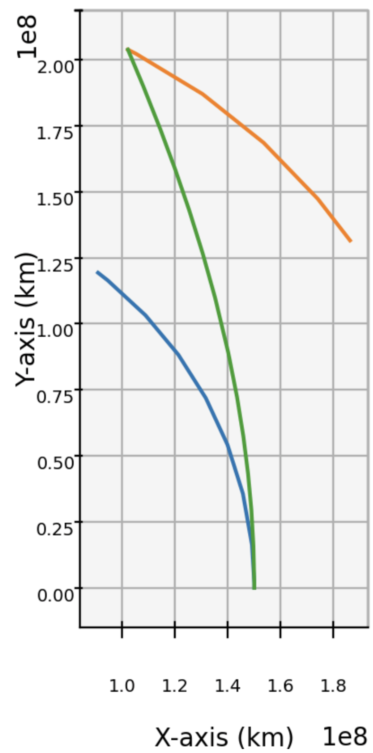

Fig. 4: Trajectories of Earth (blue), Mars (orange), and the spacecraft (green) under nuclear pulse propulsion

Table 6: Nuclear Pulse Propulsion (fast)

Transfer Time (days)	53.703
Delta V at Departure (km/s)	18.506
Delta V at Arrival (km/s)	27.804
Departing Pulse Unit Expenditure (t)	705.335
Arriving Pulse Unit Expenditure (t)	492.465
Total Pulse Unit Expenditure (t)	1197.800

Using the same process as previous similar simulations but rounding down to multiples of 500 pulse-units, the maximum departing pulse-unit expenditure is 5,000, with a mass of 705.335 t. Using this method, the transfer time is halved again to a few days under two months.

Solar Sail

The simulation modeling the solar sail uses the equation given by Farres, Webster, and Folta for solar radiation pressure. A square sail 1 km wide is represented, with mylar's density of 1400 $\frac{kg}{m^3}$ and an arbitrarily assigned thickness of 2.5 micrometers. These values are used to calculate the mass of the sail, which is added to the total vehicle mass. The simulated Starship is equipped with Raptor engines to escape and enter orbit, with propellant mass added to the vehicle mass for this purpose. The delta-V produced by the chemical engines to leave Earth orbit is the same as that produced for the Hohmann transfer. While the spacecraft is in transit, the sail, angled in the direction of solar radiation, accelerates the spacecraft out away from the Sun.

The graph and data are as follows:

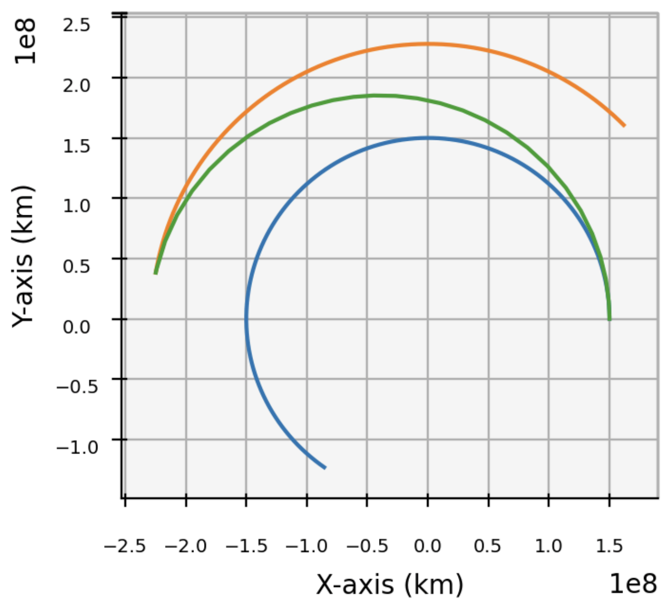

Fig. 5: Trajectories of Earth (blue), Mars (orange), and the spacecraft (green) under solar sail propulsion

Table 7: Solar Sail with Chemical Propulsion

Transfer Time (days)	239.607
Chemical Delta V at Departure (km/s)	2.925
Chemical Delta V at Arrival (km/s)	2.797
Departing Propellant Expenditure (t)	699.736
Arriving Propellant Expenditure (kg)	291.117
Total Propellant Expenditure	990.853

Due to the greater final velocity due to the acceleration produced by the solar sail, more chemical propellant (1,000 t) is required to insert the spacecraft into Mars orbit. In return, the acceleration from solar radiation shaves 20 days off the travel time.

Ion Thruster

The simulation modeling the ion thruster uses the equations for thrust and specific impulse given by Goebel and Katz. It also uses their hypothetical values for variables, that is, a beam current of 2 A, a voltage of 1500 V, $\gamma = 0.958$, and $\eta_m = 0.9$ [11]. After firing its NTR engine to escape Earth orbit with the same delta-V as in the Hohmann transfer, the simulated spacecraft accelerates continuously in the direction of its velocity using its 40 ion thrusters, then decelerates using the NTR to insert itself into Mars orbit. The delta-V produced by the NTR engines to depart is the same as for the Hohmann transfer.

The graph and data produced by this simulation are as follows:

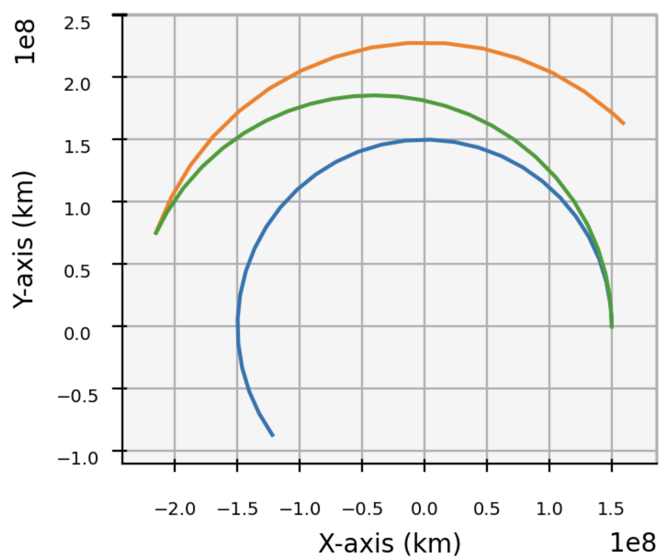

Fig. 6: Trajectories of Earth (blue), Mars (orange), and the spacecraft (green) under ion propulsion

Table 8: Ion Thruster with Nuclear Thermal Propulsion

Transfer Time (days)	219.621
NTR Delta V at Departure (km/s)	2.925
Ion Delta V (km/s)	0.247
NTR Delta V at Arrival (km/s)	3.259

NTR Propellant Expenditure at Departure (t)	160.663
Ion Propellant Expenditure (t)	2.296
NTR Propellant Expenditure at Arrival (t)	122.458
Total Propellant Expenditure (t)	285.417

A minimum starting propellant mass of 300 t can be estimated for this transfer. Although the delta-V produced by the ion thruster is small compared to those produced by the NTR, this small increment in velocity translates to nearly 40 days shaved off the transfer time compared to the Hohmann trajectory. It is also worth noting that the ion thruster consumed less than three tonnes of propellant, the least out of all propulsion systems other than the solar sail.

Plasma Thruster

The simulation modeling the MPD thruster uses the variable values from NASA's 1989 applied-field prototype: cathode diameter of 12.7 mm, anode diameter of 26 mm, magnetic field strength of 0.3 T, current of 1500 A, and specific impulse of about 1100 s [14]. The equation used to calculate thrust is the aforementioned Myers formula [12]. As with the ion thrusters, the NTR is used to escape and enter orbit using the same departing delta-V as for the Hohmann transfer, and 40 MPD thrusters provide continuous acceleration in the direction of its velocity.

The graph and data produced by this simulation are as follows:

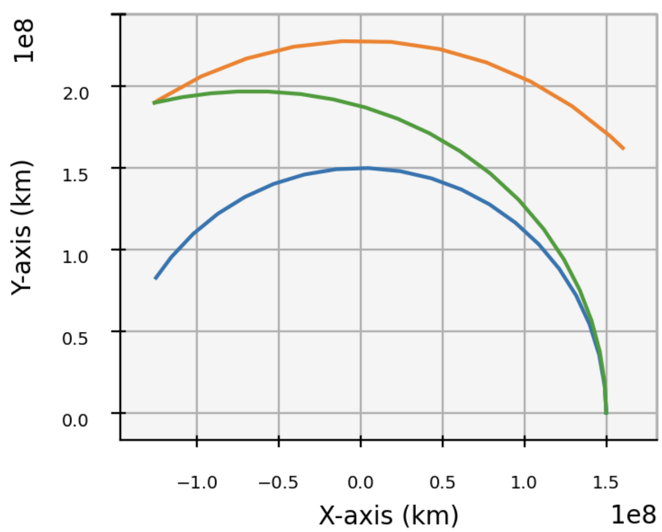

Fig. 7: Trajectories of Earth (blue), Mars (orange), and the spacecraft (green) under plasma propulsion

Table 9: Plasma Thruster with Nuclear Thermal Propulsion

Transfer Time (days)	149.300
NTR Delta V at Departure (km/s)	2.925
Plasma Delta V (km/s)	3.262
NTR Delta V at Arrival (km/s)	8.909
NTR Propellant Expenditure at Departure (t)	414.430
Plasma Propellant Expenditure (t)	254.098
NTR Propellant Expenditure at Arrival (t)	475.301
Total Propellant Expenditure (t)	1143.828

In this case, the minimum starting propellant mass is estimated to be 1150 t. The MPD thruster clearly consumes more propellant than the ion thruster due to its lower specific impulse, but this is somewhat compensated for by the greater thrust and shorter trip time, about 70 days shorter than the ion thrusters produce.

Table of Simulation Results

In the results below, the best value (depending on the criterion) is color-coded green, and the worst is color-coded red. Important values, namely the two performance criteria and the time-to-mass ratio, are in bright colors. For departing and low-thrust delta-V, the highest is the best because the transfer velocity becomes faster; for arriving delta-V, the lowest is the best because less propellant is consumed. If multiple propulsion systems have the same extreme value for a criterion, only one is color-coded.

Table 10: Simulation Results

	Chemical (Hohmann)	Chemical (fast)	Nuclear Thermal (Hohmann)	Nuclear Thermal (fast)	Nuclear Pulse (Hohmann)	Nuclear Pulse (fast)	Solar Sail	Ion Thruster	Magneto-plasma-dynamic
Transfer Time (days)	259.387	225.586	259.387	129.191	259.387	53.703	239.607	219.621	149.300
High Thrust Delta V at Departure (km/s)	2.925	2.998	2.925	4.960	2.925	18.506	2.925	2.925	2.925
Low Thrust Delta V (km/s)								0.247	3.262
High Thrust Delta V at Arrival (km/s)	2.632	3.068	2.632	8.913	2.632	27.804	2.797	3.259	8.909
Departing High Thrust Propellant Expenditure (t)	640.543	825.000	145.735	650.000	36.113	705.335	699.736	160.663	414.430
Low Thrust Propellant Expenditure (t)							0	2.296	254.098
Arriving High Thrust Propellant Expenditure (t)	256.053	350.700	93.565	520.695	29.624	492.465	291.117	122.458	475.301
Total Propellant Expenditure (t):	896.596	1175.700	239.301	1170.695	65.737	1197.800	990.853	285.417	1143.828
Decrease in Transfer Time from High Thrust Hohmann (days)							19.780	39.766	110.087
Increase in Propellant Expenditure from High Thrust Hohmann (t)							94.257	46.116	904.527
Ratio of Transfer Time Decrease to Propellant Expenditure Increase (days/t)							0.210	0.862	0.122

Conclusion

Overall, three ways were simulated to get from Earth to Mars: (1) using the Hohmann transfer to minimize propellant expenditure, (2) expending a maximum mass of propellant to minimize trip time, or (3) using a low-thrust propulsion system to accelerate constantly until arrival. Of the three propulsion systems capable of the Hohmann transfer—chemical, nuclear thermal, and nuclear pulse—the chemical rocket consumes the largest amount of propellant at about 897 t, followed by the NTR at about 239 t, and the nuclear pulse vehicle consumes the smallest amount at about 65.7 t. Using the same three systems but taking the second approach to minimize trip time, the chemical rocket takes the longest at 226 days, followed by the NTR at 129 days, and nuclear pulse takes the shortest time at 54 days.

In minimizing both propellant consumption and travel time, the nuclear pulse engine outperforms the nuclear thermal engine, which in turn outperforms the chemical engine. A lower propellant mass expenditure is advantageous since it reduces the cost of the mission; a lower trip time is advantageous as it minimizes the amount of deadly solar and cosmic radiation to which the crew are exposed. Since the ranking of the systems is the same for both criteria, it can be said that the nuclear pulse engine possesses higher overall performance than the nuclear thermal engine, which has higher overall performance than the chemical engine.

Given this ranking, it may seem that the nuclear pulse engine is the preferable choice for a high-thrust Mars mission. However, it has already been mentioned that the nuclear pulse engine is illegal under international law [7]. Barring the unlikely event that the Partial Test Ban Treaty is terminated, a Mars mission using a nuclear pulse engine is not politically feasible for the foreseeable future. The nuclear thermal engine is a more realistic alternative if lower trip time or propellant expenditure than traditional chemical propulsion is a major priority of the mission.

Of the low-thrust systems—the solar sail, the ion thruster, and the MPD thruster—the system that consumes the least propellant by itself is the solar sail at 0 t (all acceleration is from solar radiation pressure), followed by the ion thruster at 2.3 t, and the MPD thruster consumes the most at 254 t. (This ranking concerns the propellant consumed by the low-thrust engine itself, not including that consumed by the high-thrust engines used to enter or escape orbit). The system that produces the shortest travel time is the MPD thruster at 149 days, followed by the ion thruster at 220 days, and the solar sail has the longest travel time at 240 days.

Although the low-thrust engines themselves have low propellant consumption, the velocity that they add to the spacecraft results in a greater delta-V required to slow down at the end of the transfer. This increases the propellant mass expended by the high-thrust engine and the overall propellant expenditure compared to a Hohmann transfer using the same high-thrust engine. Taking into account the expenditures of the high-thrust engines, the ion/NTR system has the lowest propellant expenditure of 285 t, followed by the solar sail/chemical system at 991 t, and the MPD/NTR system has the highest at 1144 t. Unlike with the high-thrust system, there is no clear correlation between transfer time and propellant expenditure in the low-thrust systems, so it is not easy to rank their overall performance using these criteria alone.

One way to quantify the overall performance of the low-thrust systems is to calculate the ratio of the decrease in trip time to the increase in propellant consumption compared to a Hohmann transfer using the corresponding high-thrust engine. In equation form, this time-to-mass difference ratio is

$$R = \frac{t_h - t_l}{m_l - m_h} \tag{12}$$

Where t_h and t_l are the transfer times in days of the Hohmann and low-thrust transfers respectively, and m_h and m_l are the total propellant expenditures in tonnes. A higher ratio means more days subtracted from the transfer time for each tonne of propellant added using the low-thrust engine, therefore a greater improvement in performance over the corresponding high-thrust engine alone and a higher performance among the low-thrust systems.

The solar sail/chemical system decreases transfer time by 19.780 days while adding 94.257 t to the chemical Hohmann propellant expenditure; the ion/NTR system decreases transfer time by 39.766 days while adding 46.116 t to the NTR Hohmann propellant expenditure; the MPD/NTR system decreases transfer time by 110.087 days while adding 904.527 t to the NTR Hohmann propellant expenditure. The solar sail/chemical system produces a time-to-mass difference ratio of 0.210 days/t, the ion/NTR system produces 0.862 days/t, and the MPD/NTR system produces 0.122 days/t. Comparing the ratios, it appears that the ion/NTR system has the highest performance among the low-thrust systems, followed by the solar sail/chemical, and the MPD/NTR has the lowest performance.

While the low-thrust systems produce lower transfer times than Hohmann transfers using their corresponding high-thrust systems, the low-thrust systems have higher transfer times than the minimum times possible with high-thrust engines using maximum propellant expenditure. For example, the NTR alone can produce a minimum trip time 20 days shorter than the MPD/NTR's time with a propellant expenditure 27 t greater. Additionally, since high-thrust engines are needed to launch spacecraft using low-thrust engines out of and into orbit, it may be preferable to use the high-thrust engines alone rather than introduce additional variables into the mission by adding a second propulsion system.

Overall, the main conclusions are (1) that the nuclear pulse engine is superior in both transfer time and propellant consumption to the nuclear thermal engine, which in turn superior to the chemical engine; (2) that the ion/NTR system has a higher time-to-mass difference ratio, and therefore superior performance over the solar sail/chemical system, which in turn is superior over the MPD/NTR system; and (3) that while the low-thrust engines produce shorter maximum trip times than are possible with the Hohmann transfer, they produce longer trip times than the minimum possible with high-thrust engines. These conclusions are obtained by creating, running, and analyzing the results of the nine variations of the Mars mission simulations: by obtaining the necessary equations for thrust and specific impulse of each propulsion system, by using these equations and variable values obtained from the sources to simulate the acceleration and orbital trajectory of the spacecraft in relation to the Sun, Earth, and Mars, and finally by obtaining and comparing the results in the manner previously described in this section.

All simulations can be found at this link: https://github.com/thx1138bby/Mission-to-Mars.

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