Comparative Analysis of Nuclear Thermal Propulsion, Chemical Propulsion, and Electric Propulsion for a Mars Mission

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Contents

1 Introduction

The purpose of this project is to conduct a trade study to assess the feasibility of nuclear thermal propulsion. To achieve this, we will compare the requirements of the propulsion system driven by mission parameters for three propulsion systems: chemical propulsion, electric propulsion, and nuclear thermal propulsion.

Nuclear thermal propulsion is based on thermal reactions in which a nuclear core initiates a fission reaction to create heat. The propellant, chosen as hydrogen, is cryogenically stored and fed through the reactor core. The nuclear reaction heats the hydrogen to extremely high temperatures, and the gas is expanded through a nozzle. We hypothesize that the extreme temperature change in nuclear thermal propulsion will yield favorable trade-offs, provided that the materials can withstand the associated thermal stresses. It is important to note that there is no crew for the sample mission and the payload consists solely of instruments.

Current advancements in nuclear thermal propulsion include the Demonstration Rocket for Agile Cislunar Operations (DRACO) program, currently being developed by Lockheed Martin as part of the DARPA program with the ultimate goal of developing a spacecraft to carry crewed missions to Mars. This vehicle is to be launched by 2027. This nuclear thermal propulsion (NTP) engine operates using a fission reactor to transfer heat to a liquid propellant, specifically liquid hydrogen. This heat converts the hydrogen into a high-temperature gas, which expands through a nozzle to generate thrust. The reactor fuel consists of enriched uranium, primarily Uranium-238 combined with 20% Uranium-235 (the fissile isotope). This enrichment level, higher than the 3–5% typical of terrestrial light-water reactors but significantly lower than the 90% characteristic of weapons-grade material, was chosen to reduce regulatory and programmatic burdens while maintaining reactor performance. Particle bed reactor (PBR) designs, offering rapid heat transfer and compact configurations, are also under development. However, challenges remain in creating materials that can endure extreme temperatures and radiation, along with addressing regulatory, safety, and cost concerns. International interest, ongoing ground-based testing, and refined simulation tools are accelerating progress, positioning NTP as a key enabler for human exploration of Mars and beyond, provided these technological and logistical hurdles can be overcome.

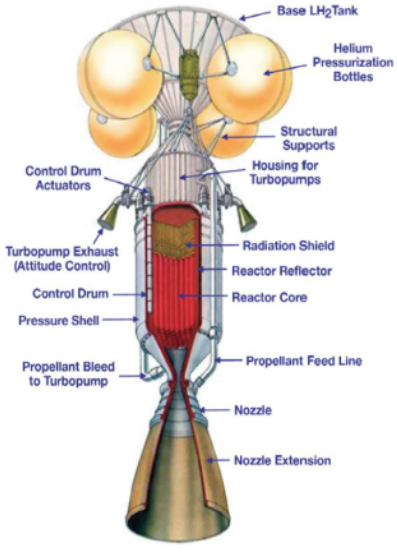


Figure 1: Detailed Schematic of Nuclear Thermal Propulsion System [1]

2 Mission Architecture

The mission under consideration involves the transfer from a low Earth orbit (LEO) to a low Mars orbit (LMO). Both impulsive and low-thrust propulsion systems were evaluated using Hohmann transfer and low-thrust maneuvers respectively, each consisting of three distinct phases. Although real-world orbital mechanics is more complex, requiring multibody simulations and careful consideration of launch timing, these factors were excluded from this analysis for simplicity.

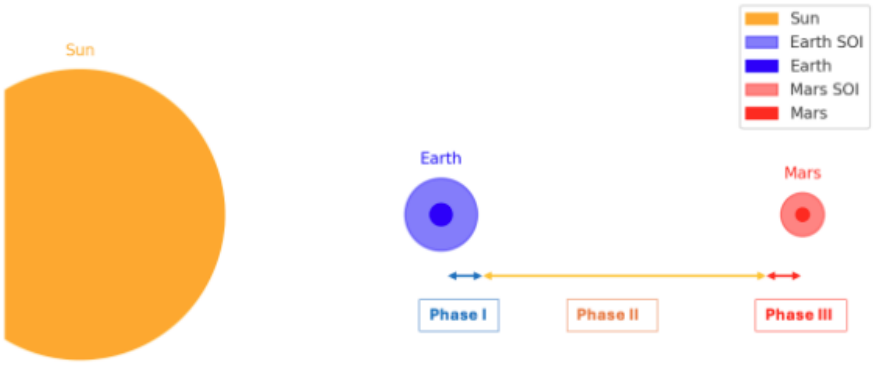


Figure 2: Schematic of the Three Phases of our Mission Based on Spheres of Influence

2.1 Hohmann Transfer

A Hohmann transfer is a two-impulse orbital maneuver designed to transfer a spacecraft between two circular orbits of different radii using an elliptical transfer orbit. This analysis assumes tangential impulsive maneuvers, where velocity changes are applied instantaneously. This assumption simplifies the calculations and is particularly suitable for determining the required Δv for chemical propulsion systems, which provide high thrust (on the order of 10^3 N) but relatively low specific impulse (250 - 450 s).

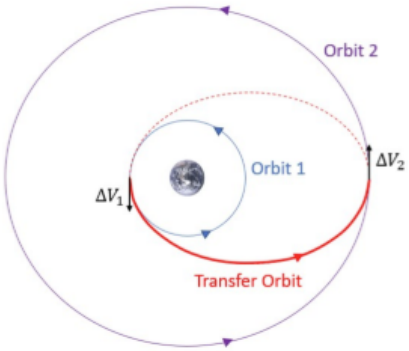


Figure 3: Hohmann Transfer Schematic [2]

The required formulas are as follows:

- **Semi-Major Axis**
  The semi-major axis (a_{transfer}) is the average of the radii of the initial (r_p) and final (r_a) orbits.

$$a_{\text{transfer}} = \frac{r_p + r_a}{2}$$

- **Orbital Velocity at a Given Radius**
  The orbital velocity (v) at any point in the orbit is calculated using the vis-viva equation:

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

  where μ is the gravitational parameter (GM), where G is the gravitational constant

$$G = 6.67 \times 10^{-11}\,\text{Nm}^2/\text{kg}$$

  and M is the mass of the central body.

- **Initial and Transfer Velocity**

$$v_{\text{initial}} = \sqrt{\frac{\mu}{r_p}}$$

$$v_{\text{transfer, periapsis}} = \sqrt{\mu \left(\frac{2}{r_p} - \frac{1}{a} \right)}$$

  The first impulsive burn is performed at periapsis, where the change in velocity is defined as:

$$\Delta v_1 = v_{\text{transfer, periapsis}} - v_{\text{initial}}$$

- **Final and Transfer Velocity**

$$v_{\text{final}} = \sqrt{\frac{\mu}{r_a}}$$

$$v_{\text{transfer, apoapsis}} = \sqrt{\mu \left(\frac{2}{r_a} - \frac{1}{a} \right)}$$

  The second impulsive burn is performed at apoapsis, where the change in velocity is defined as:

$$\Delta v_2 = v_{\text{final}} - v_{\text{transfer, apoapsis}}$$

  The total Δv is the sum of Δv_1 and Δv_2.

- **Orbital Period of the Elliptical Transfer Orbit** To determine the transfer time for a Hohmann transfer, the orbital period of the elliptical transfer orbit is calculated using the following formula:

$$T = 2\pi \sqrt{\frac{a_{\text{transfer}}^3}{\mu}}$$

  Since a Hohmann transfer involves traversing only half of the elliptical orbit, the transfer time is

calculated as:

$$t_{\text{transfer}} = \frac{T}{2} = \pi\sqrt{\frac{a_{\text{transfer}}^3}{\mu}}$$

For our mission, the required Δv to transfer a spacecraft from Low Earth Orbit (LEO) to Low Mars Orbit (LMO) was calculated using a series of Hohmann transfer trajectories. The analysis is divided into three distinct phases, each addressing a specific segment of the journey: escaping Earth's orbit, interplanetary transfer, and capturing into Mars orbit. The corresponding code is provided in Appendix B.

2.1.1 Phase I: Low Earth Orbit (LEO) to Escaping Earth's Sphere of Influence (SOI)

For this Hohmann transfer, it was assumed that the spacecraft's dynamics were solely influenced by Earth's gravity, and thus Earth's gravitational parameter was used. The radius marking the boundary of Earth's Sphere of Influence (SOI) was calculated using the following formula (from Dr. Peck's MAE 5065):

$$r_{\text{SOI}} = r_{\text{Earth-Sun}} \left(\frac{M_{\text{Earth}}}{M_{\text{Sun}}}\right)^{\frac{2}{5}}$$

Beyond this radius, Phase II began, where the spacecraft's dynamics became governed by the Sun.

2.1.2 Phase II: Interplanetary Transfer (Earth to Mars)

The second phase calculates the transition from Earth's orbit around the Sun to Mars' orbit using a Hohmann transfer in a heliocentric (Sun-centered) frame of reference.

2.1.3 Phase III: Capture into Low Mars Orbit (LMO)

The final phase determines the necessary burns to transition the spacecraft from Mars' sphere of influence (SOI) into low Mars orbit (LMO), utilizing Mars' gravitational parameters.

Figure 4: Summary of Orbit Radii Used in Transfers

	μ	Initial Orbit Radius	Initial radius (m)	Final Orbit Radius	Final Radius (m)
Phase I (LEO to Earth's SOI)	μ_E	$r_E + r_{LEO}$	6.976×10^6	$r_{Earth-SOI}$	9.201×10^8
Phase II (Earth to Sun)	μ_S	$r_{E-S} + r_{E-SOI}$	1.496×10^{11}	$r_{M-S} - r_{M-SOI}$	2.297×10^{11}
Phase III (Mars' SOI to LMO)	μ_M	$r_{Mars-SOI}$	5.778×10^8	$r_{LMO} + r_M$	3.590×10^6

2.2 Low-Thrust Maneuver

Electric propulsion systems with their low thrust output are not suitable for traditional Hohmann transfers because the assumption of impulsive burns—where velocity changes occur instantaneously—is no longer valid. Instead, spacecraft using electric propulsion follow a spiral-like trajectory as they gradually accelerate over extended periods. Despite this difference, the transfer can still be approximated using three phases

similar to those of a Hohmann transfer, as summarized in Table 1. These phases involve transitioning between orbits through continuous low-thrust propulsion.

In order to simulate these trajectories, ode45 in MATLAB computed the change in radius as the mass of the spacecraft changed during the burn. As the spacecraft traveled farther away from the Earth, the force of gravity became diminished and with reduced mass from expelling propellant, the spacecraft had significantly increased acceleration as the maneuver continued. This explains how the orbit snakes into less and less of a spiral, as evident in figures 4 and 5.

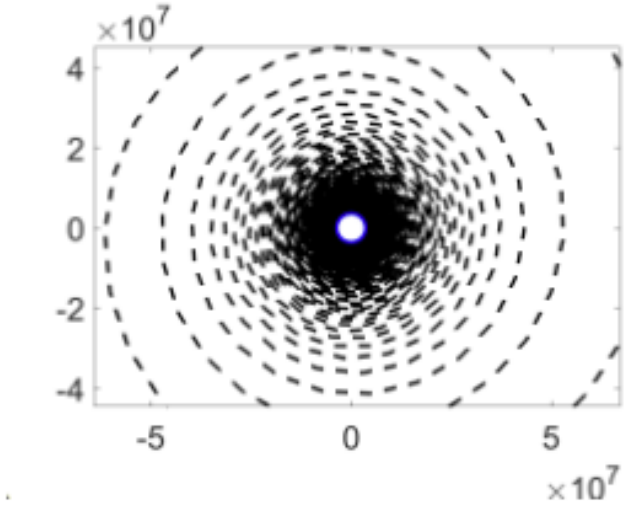


Figure 5: Zoomed In Earth Low Thrust Trajectory

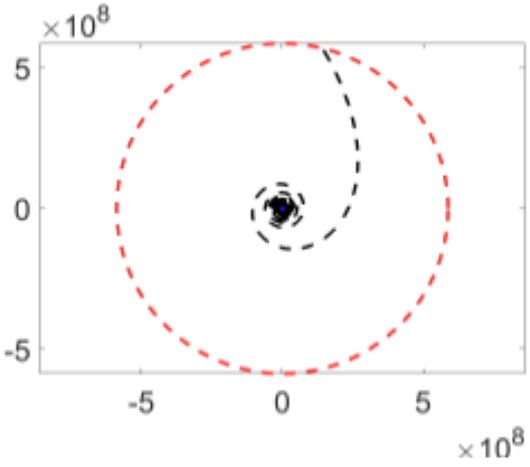


Figure 6: Earth Low Thrust Trajectory

The dynamics governing these trajectories were modeled in MATLAB using Newton's second law in a 2D polar coordinate system and `ode45`, a built-in numerical differential equation solver. The spacecraft's motion

is influenced by a balance of forces: gravitational attraction (F_g), thrust (F_T), and inertial forces. The radial and angular accelerations are derived as follows:

$$\sum \mathbf{F} = m\mathbf{a}$$

$$F_g\hat{e}_r + F_T\hat{e}_r = m\left((\ddot{r} - r\dot{\theta}^2)\hat{e}_r + (r\ddot{\theta} + 2\dot{r}\dot{\theta})\hat{e}_\theta\right)$$

where $F_g = \frac{\mu m}{r^2}$ and F_T is assumed to be constant and thus defined as $F_T = ma_r = m\ddot{r}$. By separating the r and θ components, the following equations can be obtained:

- **Radial acceleration (r)**
$$\ddot{r} = -\frac{\mu}{r^2} + r\dot{\theta}^2$$
- **Angular acceleration (θ)**
$$\ddot{\theta} = \frac{a_T - 2\dot{r}\dot{\theta}}{r} = \frac{F_\mathrm{T}/m - 2\dot{r}\dot{\theta}}{r}$$

Radial acceleration is influenced by gravitational attraction and centrifugal forces, while angular acceleration is governed by the spacecraft's thrust and the Coriolis force.

The equations are implemented in the function `odefun_skeleton`. This function takes the current state vector $(r, \dot{r}, \theta, \dot{\theta}, m)$ and computes the time derivatives $(\dot{r}, \ddot{r}, \dot{\theta}, \ddot{\theta}, \dot{m})$ to advance the simulation.

The initial state vector was:

$$[r_0, \dot{r}_0, \theta_0, \dot{\theta}_0, m_0] = [r_\mathrm{LEO}, 0, 0, \sqrt{\frac{\mu}{r^3}}, m_0]$$

A stopping condition, defined by the `opt1` event, ensured that the integration halts when the spacecraft reaches a target radius (r_p) for each phase of the transfer. The target radii correspond to the three key phases of the trajectory, enabling the spacecraft to achieve its desired orbital path. These simulations highlight the gradual nature of electric propulsion transfers and the efficiency of this technology for missions requiring high Δv over extended periods. The elapsed time for each burn in the provided MATLAB code is calculated directly from the output of the `ode45` solver.

3 Design

3.1 Chemical Propulsion

Upon reviewing literature for chemical rocket propulsion, we find a typical specific impulse value of approximately 300 s [3]. We consider hydrazine (N_2H_4) as our propellant due to its relatively higher specific impulse and proven track record in the industry. The accompanying oxidizer is commonly nitrogen tetroxide (N_2O_4) [4].

From our mission architecture, we know our Δv for the combined Hohmann transfer maneuver is approximately 10.1 km/s. Using this value along with our specific impulse, we can find the following relationship using the rocket equation:

$$m_0 = m_f e^{\Delta v / u_e}$$

Using this equation, where $u_e = I_{\text{sp}} g_0$, we find our initial mass as 7910 kg. Finally, we assume that the mass of the propellant is equivalent to the difference between the initial mass and the final mass, and deduce the mass of propellant for the chemical propulsion system to be 7660 kg.

3.2 Electric Propulsion (HET)

The input parameters of specific impulse and thrust were assumed based on typical values. The specific impulse is typically around 1500 s [5], and a typical value of thrust is 80 mN [6]. The propellant considered was Xenon due to its ease of storage, well-developed flight history, and widespread use in the industry. Its relatively high performance makes it suitable as a propellant for Hall Effect thrusters.

Using an assumed constant thrust force and exhaust velocity, the mass flow rate was determined:

$$\dot{m} = \frac{T}{u_e}$$

To calculate the initial mass, the MATLAB simulation was run varying the initial mass such that the final mass settled at 250 kg. The initial mass of the HET system would need to be 760 kg to reach Mars, using 509 kg of propellant. The cost would reach approximately half a million dollars [7].

The power required was calculated using the jet power equation:

$$P_{\text{jet}} = \frac{\dot{m} u_e^2}{2}$$

Given the assumed thrust and other performance parameters, the transfer time and Δv required were calculated from the MATLAB simulation, attached in the Appendices.

3.3 Nuclear Thermal Propulsion

Starting from typical design values for specific impulse and thrust (900 s for *Specific Impulse of NTP* [8] and 100 kN for *NTP Thrust* [9]), the mass flow rate can be found:

$$\dot{m} = \frac{T}{u_e} = 11.3\,\text{kg/s}$$

Using the equation for the specific impulse of the nuclear system, the change in temperature can be found:

$$u_e = \sqrt{2 c_p \Delta T}, \quad \Delta T = 2550\,\text{K}$$

Liquid hydrogen is usually stored at pressures of approximately 500 psi at temperatures above the boiling temperature of 25 K [10]. Since the incoming liquid hydrogen will be kept at cryogenic temperatures, T_0 is known, and T_e must be 2575 K. Therefore, the required heat input from the nuclear fission reactor must be enough to vaporize and heat the flow rate of LH2 [11]:

$$\dot{Q} = \dot{m} c_p \Delta T + \dot{m} \Delta H_{\text{vap}} = 446\,\text{MW}$$

To achieve this amount of heat transfer, the heat exchanger design is essential. Sizing the reactor is a function of the heat rate but is limited based on how much heat transfer can occur. Modeling the heat exchanger as

a set of pipes between the nuclear rods, the turbulent, fully-developed Nusselt number correlation is [12]:

$$\frac{hd}{k} = \text{Nu} = 0.023\text{Re}^{4/5}\text{Pr}^{0.4}$$

where A_s is also a function of diameter and the number of pipes. The heat flux is given by:

$$\dot{Q} = hA_s\Delta T$$

Considering a reactor length of about 3 m and setting the number of pipes as a solved parameter, the resulting solution minimizes the flow velocity to subsonic levels, minimizes the pipe diameter, and ensures the heat exchanger operates efficiently. The heat exchanger design is summarized in the following table:

Table 1: Summary of Heat Exchanger Design

Parameter	**Value**
Number of Pipes	16
Pipe Diameter	0.30 m
Reynold's Number	1.9×10^6
Convective Heat Transfer Coefficient	3800 W/m^2-K
Flow Velocity	250 m/s

The design is limited by the surface area provided by the pipe design, causing slightly unrealistic diameter sizes to compensate. The assumption made was that the flow traveled through the pipe at a constant speed; however, as the flow travels along the reactor, it accelerated as the density suddenly decreased with expansion and the temperature increased. This means the flow does not have a constant velocity through the pipe and therefore does not have a constant Reynold's number. Internal shock waves due to supersonic flows are also a concern in this design and should be considered in future development.

The propellant mass was calculated using the rocket equation:

$$m_0 = m_f e^{\Delta v/u_e}$$

which yields a propellant mass of approximately 535 kg.

4 Trade-off Analysis

Table 2: Summary of Design Parameters

Parameter	**Nuclear Thermal**	**Hall Effect Thruster**	**Chemical Propulsion**
Δv (km/s)	10.2	15.7	10.2
Specific Impulse (s)	900 [8]	1500 [5]	300 [3]
Thrust (N)	100,000 [9]	0.083 [6]	750 [13]
Transfer Time (yrs)	0.8	2.86	0.8
Burn Time (s)	47.2	Continuous	1210
Propellant	Hydrogen [14]	Xenon [7]	Hydrazine + N_2O_4 [15]
Propellant Mass (kg)	535	510	7660
Propellant Cost ($/kg)	30 [14]	1140 [7]	75.6 [15]
Total Cost ($)	16,050	581,000	580,000
Power Required (kW)	446,000 [16]	0.61 [6]	0

The total cost of the NTP system does not include the expensive nuclear fuel and extremely massive reactor apparatus. The NTP system is by far the heaviest in terms of propulsion system mass, which under the assumption of 250 kg final mass, retrospectively proves unrealistic for the nuclear thermal system, likely weighing a few tons to produce the thrust and heat loads studied. The nuclear thermal system, with its minimal propellant required and burn time, could take an alternate trajectory requiring more Δv but arriving much faster than 0.8 years.

5 Challenges

5.1 Chemical Propulsion

Chemical propulsion is the most established technology, providing high thrust (and low specific impulse) but at the cost of efficiency and practicality for interplanetary missions:

- **Propellant Mass:** The significant Δv requirements for interplanetary transfers result in large amounts of propellant ($\sim 7660\,\text{kg}$ for this mission). This limits payload capacity and increases launch costs.
- **Operational Complexity:** The use of hypergolic propellants (e.g., hydrazine and nitrogen tetroxide) simplifies ignition but requires extensive handling precautions due to their toxicity and reactivity.
- **Performance Limitations:** Specific impulse values ($\sim 300\,\text{s}$) are much lower compared to NTP and HET systems, reducing mission efficiency and increasing overall mission mass.

Despite these challenges, chemical propulsion remains reliable for rapid maneuvers and is well-suited for shorter-duration missions or applications requiring high thrust over short durations.

5.2 Electric Propulsion (HET)

Hall Effect Thrusters (HETs) are lightweight and highly efficient, but their reliance on continuous low-thrust operations introduces unique challenges:

- **Power Requirements:** While the power demand for Hall Effect Thrusters is relatively modest ($\sim 0.61\,\text{kW}$), it still requires a reliable power generation and storage system, such as solar arrays or nuclear power sources. For other systems, the power requirements may be significantly higher, potentially creating additional challenges. These systems increase the overall complexity and cost of the mission infrastructure, particularly as they often involve deployable structures. Furthermore, solar panels become ineffective for deep-space missions far from the Sun, necessitating alternative power solutions like radioisotope thermoelectric generators (RTGs) or nuclear reactors.
- **Propellant Cost:** Xenon, the preferred propellant, is extremely expensive ($\sim$ \$1140/kg). This increases mission costs significantly, particularly for high-Δv requirements.
- **Long Transfer Times:** The low thrust output of electric propulsion leads to extended mission durations, such as the 2.86 years required for the current mission. These prolonged timelines raise concerns about spacecraft reliability, system durability, and potential delays in meeting mission objectives. This challenge is particularly pronounced for smaller, cost-effective spacecraft like CubeSats, which often depend on off-the-shelf components with typical operational lifespans of only 3–5 years, making them less suited for long-duration missions.
- **Technological Integration:** While HETs are well-developed for near-Earth missions, scaling them

for interplanetary travel requires improvements in thruster lifespan, thermal management, and system redundancy.

5.3 Nuclear Thermal Propulsion

Nuclear thermal propulsion holds immense promise as a transformative technology for space exploration. However, significant design challenges, including propellant storage, efficient heat transfer, control of supersonic flows, and reactor stabilization, currently hinder its commercial viability. Despite these obstacles, it has the potential to revolutionize interplanetary travel and even enable interstellar missions in the future.

- **Propellant Storage:** Liquid hydrogen must be stored cryogenically at temperatures below its boiling point (25 K). Maintaining these temperatures over extended missions without excessive boil-off is a significant challenge and is nearly impossible with current technology.
- **Heat Transfer:** The nuclear fission reactor must generate and transfer heat efficiently to the propellant without causing structural failures. The current design, which uses pipes embedded in the reactor core, is constrained by surface area, leading to large pipe diameters that may be impractical. Improvements would consider alternate geometries to increase the effectiveness of the heat exchanger and integrate the reactor with it.
- **Reactor Stabilization:** The nuclear reactor must be controllable to prevent overheating and to adjust power output dynamically during the mission. This requires robust thermal management systems and safeguards to protect structural components. The system must be turned on and off suddenly to achieve the impulsive burns, which creates challenges to regulate reactor temperature. Future reactor cores of liquid or even gaseous products introduce temperatures far beyond current technology's material limits but promise specific impulse that is orders of magnitude greater.

The weight and complexity of the NTP system, including the reactor and associated shielding, also make it the heaviest propulsion option, posing significant integration and launch challenges.

6 Propulsion System Improvement

Assuming issues related to liquid hydrogen storage can be addressed, the performance of NTP can be improved by increasing heat transfer efficiency through the reactor. This elevates exhaust temperatures and increases specific impulse. Instead of the considered pipe layout, porous media can be used as the reactor, which allows increased heat transfer for reduced pressure drop. This technology is currently being explored.

To achieve the same thrust (100 kN) with an increased specific impulse of 1000 s, the required power would be 491 MW, which would have the reactor reach a maximum temperature of about 3000 K, where material limits are nearly reached. With the same thrust, it would have roughly the same burn time (with the same mission delta v required), but would weigh less, allowing more payload mass. The reduced propellant mass required would further reduce cost. The tradeoff lies in the increased power required and complexity of developing a porous media reactor.

Table 3: Comparison of Design Improvement

Parameter	**Nuclear Thermal**	**Nuclear Thermal Improvement**
Δv (km/s)	10.2	10.2
Specific Impulse (s)	**900**	**1000**
Thrust (kN)	100	100
Transfer Time (yrs)	0.8	0.8
Burn Time (s)	47.2	44.1
Propellant	Hydrogen	Hydrogen
Propellant Mass (kg)	535	450
Propellant Cost ($/kg)	30*	30*
Total Cost ($)	**16,050**	**13,500**
Power Required (MW)	446	491

Additional performance increases can be found in liquid or gaseous reactors. These media allow much higher temperatures since they have already melted or vaporized, allowing order of magnitude increases in exhaust temperature, velocity, and specific impulse.

7 Geometry Analysis

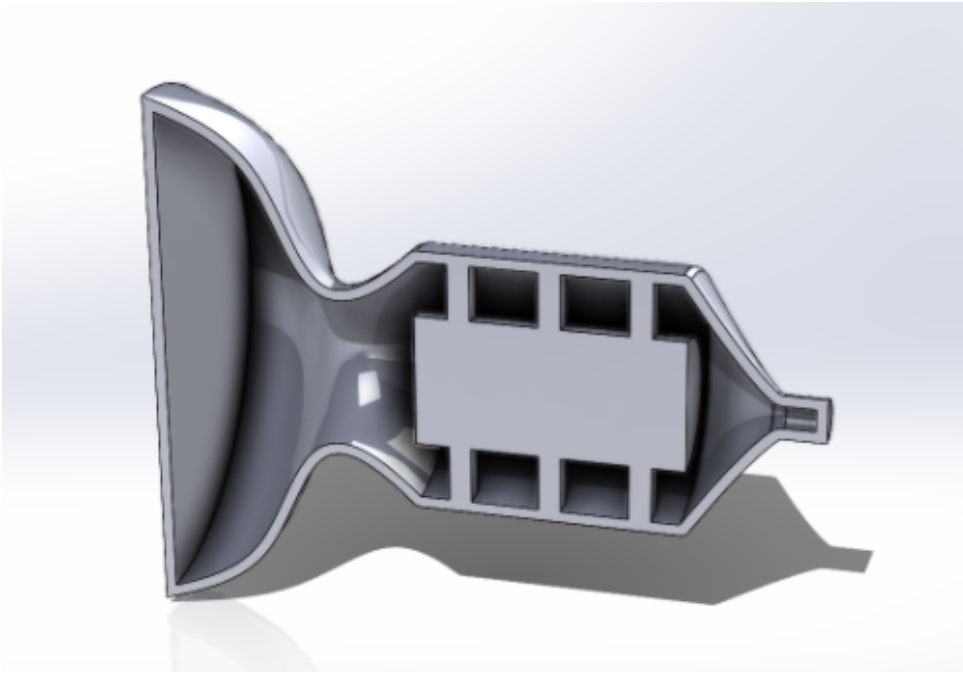


Figure 7: Nozzle Rendering

The geometry of the propulsion system can be described as an injector stage followed by a reaction chamber, ending with a convergent-divergent nozzle. The cylinder embedded within the reaction chamber represents the reactor core, in which the hydrogen is rapidly heated (the reaction is a fission reaction in the core). This cylinder is supported by struts to ensure the geometry prints properly.

In practice, hydrogen propellant is passed through pipes around the nozzle to regeneratively cool the structure

and vaporize the liquid propellant into a gas. We assume that this process occurs outside of our pictured geometry. The negative space in between the reactor core and chamber walls is where control channels would be placed, which ensure that the reactor core temperature does not jeopardize the structural integrity of the chamber walls by accelerating and decelerating the fission reaction.

When the system is not running, the reactor must be decelerated to near zero levels of heat generation where a complicated external heat exchanger (not pictured) rejects heat to an external radiator assembly. The dimensions match the analysis conducted in Section 3.1.

For a space application, where ambient pressure is 0, the optimal nozzle has an infinite expansion ratio to avoid plume divergence due to flow pressure gradients. The nozzle expansion ratio should be about 300:1 to ensure high efficiency of thrust delivery while not being too massive and costing extra mass, as studied in homeworks—which can be seen commonly in industry applications [17].

8 Recommendation

Throughout this report, the great potential of nuclear thermal propulsion has been proven. Its extremely high thrust and relatively high specific impulse surpasses the performance of both chemical and electric propulsion systems. However, the challenges of storing liquid hydrogen at cryogenic temperatures still hinder the use of NTP for a mission to Mars. LH2 is only used on launch vehicles due to the accessibility of extra LH2 as boil off happens on the launch pad right before launch.

Despite its immense potential, NTP is not yet feasible with current technology, leading us to recommend chemical propulsion for now. Comparing Hall Effect Thrusters (HET) with chemical propulsion reveals trade-offs: HET significantly reduces propellant mass, but the high cost of xenon makes it similarly priced while offering much slower transfer times. However, our analysis excludes launch costs, which would be considerably higher for chemical systems due to their greater weight and propellant demands. The key advantages of chemical propulsion over HET are its lower operational power requirements and significantly faster transfer times.

In the future, if cryogenic propellant storage in space becomes feasible, NTP could emerge as the leading propulsion system. It offers exceptional thrust efficiency while enabling the reactor to serve as a primary power source, greatly enhancing spacecraft capabilities. This concept is not impossible, as demonstrated by the James Webb Space Telescope, which maintains temperatures below 50 K for its instruments. Adapting similar hyper-insulative technologies could make efficient cryogenic storage a reality.

9 Conclusion

Nuclear thermal propulsion represents a transformative technology with the potential to significantly advance space exploration, particularly for interplanetary and potentially interstellar missions. Its high specific impulse and thrust make it a promising candidate for missions requiring efficient, long-duration propulsion systems. However, its commercial viability remains hindered by several critical challenges, including the cryogenic storage of liquid hydrogen, efficient heat transfer from the reactor to the propellant, management of supersonic flow dynamics, and stabilization of the nuclear reactor during operation.

Our trade study highlights the strengths and limitations of nuclear thermal propulsion compared to other systems such as chemical propulsion and Hall Effect thrusters. While chemical propulsion provides the high thrust required for quick orbital maneuvers, its low specific impulse necessitates substantial propellant mass,

which significantly limits payload capacity. On the other hand, Hall Effect thrusters, with their increased mass efficiency and low propellant mass requirements, are ideal for long-duration missions but suffer from extended transfer times and high propellant costs due to the reliance on xenon.

Nuclear thermal propulsion bridges the gap between these systems, offering a compelling balance of high thrust and efficiency. Our analysis indicates that, if the associated technological and logistical challenges can be addressed, nuclear thermal propulsion could drastically reduce mission durations while enabling more efficient payload delivery. The use of advanced materials for reactor components, innovations in heat exchanger design, and solutions to mitigate boil-off in cryogenic propellant storage are essential to overcome these barriers.

Furthermore, our exploration into design improvements, such as the use of porous media reactors to enhance heat transfer and increase specific impulse, underscores the potential for further optimization of nuclear thermal propulsion systems. By addressing these challenges, the technology could support ambitious missions, such as crewed exploration of Mars, and pave the way for future interstellar travel.

In conclusion, while nuclear thermal propulsion is not yet commercially viable, its development is critical for the advancement of space exploration. Continued research, investment in material science, and collaboration across international space agencies will be key to unlocking its potential. This technology not only holds the promise of revolutionizing space travel but also inspires humanity to push the boundaries of what is possible in our pursuit of the stars.

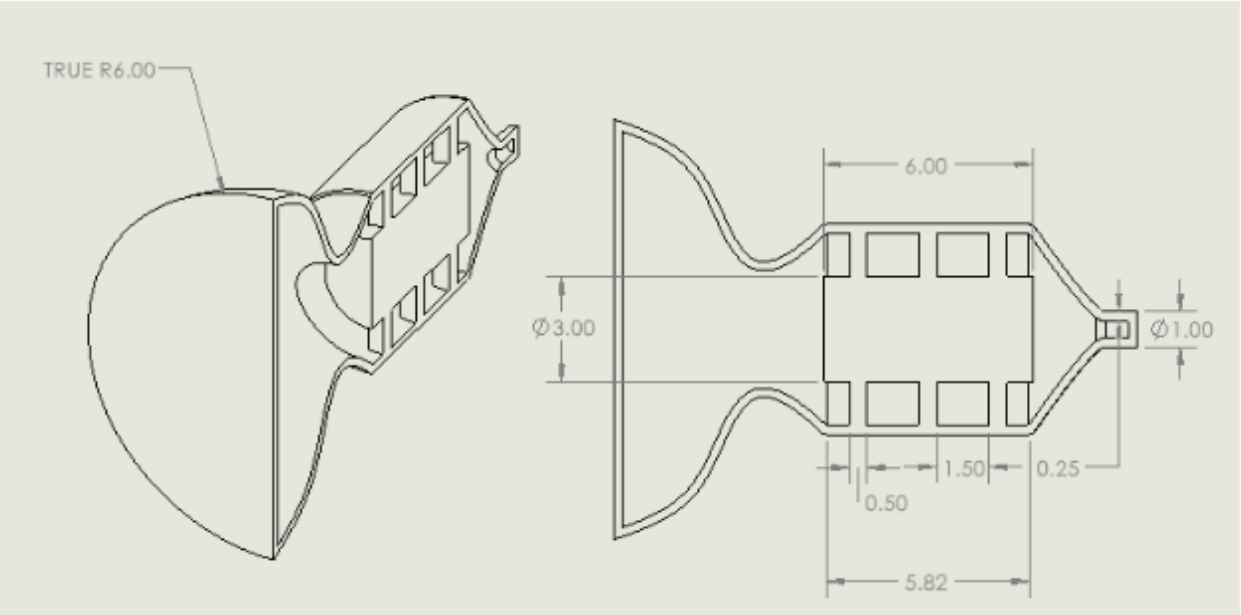
10 Division of Work

Justin: Introduction, Geometry Analysis, CAD design, Chemical Propulsion design
Eric: Nuclear Thermal Propulsion, Propulsion System Improvement, Electric Propulsion Code
Ana Sofia: Trajectory design, Chemical and Electric Propulsion Mission Architecture and code, Challenges
All: Trade-off analysis, report writing and presentation

11 Appendices

- A: Nozzle Geometry
- B: MATLAB Hohmann Transfer Code
- C: MATLAB Low Thrust Transfer Code

Appendix A



Appendix B

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2  Clear variables, command window, and close all figures
3 clear all;
4 clc;
5 close all;
6 %created by Ana Sofia Alonso (aa983)
7
8 % Gravitational Constant
9 G = 6.67430e-11;
10
11 % Masses of celestial bodies
12 m_E = 5.972e24; % Mass of Earth (kg)
13 m_M = 6.417e23; % Mass of Mars (kg)
14 m_S = 1.988e30; % Mass of the Sun (kg)
15
16 % Gravitational parameters
17 mu_E = G * m_E; % Earth (m^3/s^2)
18 mu_S = G * m_S; % Sun (m^3/s^2)
19 mu_M = G * m_M; % Mars (m^3/s^2)
20
21 %% Part 1: LEO to Escape (Earth-Centered Hohmann Transfer)
22 % Constants for Earth-centered orbit
23 R_E = 6378 * 10^3;      % Radius of Earth (m)
24 r_LEO = 6700 * 10^3;    % Radius of LEO (m)
25
26 % Escape radius of Earth's sphere of influence
27 r_ES = 149e9;           % Semi-major axis of Earth's orbit around the Sun (m)
28 r_esc = r_ES * (m_E / m_S)^(2/5); % Sphere of Influence (m)
29
30 % Hohmann transfer from LEO to escape
31 r_p = r_LEO;            % Periapsis (LEO)
32 r_a = r_esc;            % Apoapsis (escape)
33 a_transfer = (r_p + r_a) / 2; % Semi-major axis of transfer orbit
34
35 % Orbital velocities for Earth-centered transfer
36 v_LEO = sqrt(mu_E / r_LEO);                        % Initial velocity in LEO
37 v_transfer_p = sqrt(mu_E * (2 / r_p - 1 / a_transfer)); % Velocity at periapsis
38 delta_v1_LEO_to_escape = v_transfer_p - v_LEO;     % Delta-v for LEO to escape
39
40 % Escape velocity at Earth's sphere of influence
41 v_esc_SOI = sqrt(mu_E / r_esc);
42
43 T_transfer_1 = pi * sqrt(a_transfer^3 / mu_E);
44
45 %% Part 2: Interplanetary Hohmann Transfer (Earth to Mars)
46 % Constants for Sun-centered orbit
47 AU = 1.496e11;           % Astronomical Unit (m)
48 a_E = 1 * AU;            % Semi-major axis of Earth's orbit (m)
49 a_M = 1.524 * AU;        % Semi-major axis of Mars' orbit (m)
```

```
50
51 % Hohmann Transfer from Earth to Mars
52 r_p = a_E;               % Periapsis (Earth's orbit)
53 r_a = a_M;               % Apoapsis (Mars' orbit)
54 a_transfer = (r_p + r_a) / 2; % Semi-major axis of transfer orbit
55
56 % Orbital velocities in heliocentric orbits
57 v_E = sqrt(mu_S / r_p);                  % Earth's orbital velocity
58 v_M = sqrt(mu_S / r_a);                  % Mars' orbital velocity
59 v_transfer_p = sqrt(mu_S * (2 / r_p - 1 / a_transfer));  % Velocity at periapsis
    of transfer orbit
60 v_transfer_a = sqrt(mu_S * (2 / r_a - 1 / a_transfer));  % Velocity at apoapsis of
     transfer orbit
61
62 % Delta-v calculations for interplanetary transfer
63 delta_v2_departure = v_transfer_p - v_E; % Boost from Earth's orbit to transfer
    orbit
64 delta_v3_arrival = v_M - v_transfer_a;   % Mars insertion delta-v
65
66 % Total interplanetary delta-v
67 delta_v_interplanetary = delta_v2_departure + delta_v3_arrival;
68
69 T_transfer_2 = pi * sqrt(a_transfer^3 / mu_S);
70 %% Part 3: Mars to enter LMO
71 % Constants for Mars-centered orbit
72 R_M = 3390 * 10^3;       % Radius of Mars (m)
73 r_LMO = R_M + 200 * 10^3; % Radius of Low Mars Orbit (LMO) (m)
74 r_MS = 228e9;  %semimajor axis around mars sun (assuming circular orbits)
75 r_esc_M = r_MS*(m_M/m_S)^(2/5)
76 % Velocity at Mars SOI
77 v_M_SOI = sqrt(mu_M / r_esc);            % Velocity at Mars SOI (m/s)
78
79 % Hohmann transfer from Mars SOI to LMO
80 r_p = r_LMO;            % Periapsis (LMO)
81 r_a = r_esc_M;            % Apoapsis (Mars SOI)
82 a_transfer_Mars = (r_p + r_a) / 2;       % Semi-major axis of transfer orbit
83 % Orbital velocities for Mars-centered transfer
84 v_SOI_capture = sqrt(mu_M * (2 / r_a - 1 / a_transfer_Mars)); % Velocity at Mars
    SOI in transfer orbit
85 v_LMO = sqrt(mu_M / r_LMO);              % Velocity in LMO
86 v_transfer_p_Mars = sqrt(mu_M * (2 / r_p - 1 / a_transfer_Mars)); % Velocity at
    periapsis
87
88 delta_v4_SOI_to_LMO = v_transfer_p_Mars-v_LMO; % Delta-v for SOI to LMO
89
90 T_transfer_3 = pi * sqrt(a_transfer_Mars^3 / mu_M);
91 %% Total Delta-v
92 % Total delta-v including LEO escape, interplanetary transfer, and Mars capture
93 delta_v_tot = delta_v1_LEO_to_escape + delta_v_interplanetary +
    delta_v4_SOI_to_LMO;
94 T_transfer_total = T_transfer_1+T_transfer_2+T_transfer_3
```

```
% Display results
fprintf('Delta-v for LEO to escape: %.2f m/s\n', delta_v1_LEO_to_escape);
fprintf('Delta-v for interplanetary departure: %.2f m/s\n', delta_v2_departure);
fprintf('Delta-v for Mars arrival: %.2f m/s\n', delta_v3_arrival);
fprintf('Delta-v for Mars SOI to LMO: %.2f m/s\n', delta_v4_SOI_to_LMO);
fprintf('Total Delta-v: %.2f m/s\n', delta_v_tot);
fprintf('Transfer time for Mars SOI to LMO: %.2f seconds (%.2f days)\n',
    T_transfer_total, T_transfer_total/ (24 * 3600));
```

Appendix C

```
1
2  % Script for 3 LT maneuvers (LEO->MEO)
3  %created by Eric Yablonski (ery5@cornell.edu)
4  clear; close all; clc;
5  %% Parameters
6  %Hall Effect Thruster
7  T = 0.083;%N, assume constant thrust
8  Isp = 1500;%s, 1000-8000 for HET
9  %final mass
10 mo = 760;%kg, mass of structure, payload, etc. (nor propellant)
11 %% Constants
12 %gravitational constant
13 G = 6.6743e-11;
14 g0 = 9.81;
15 %masses
16 m_E = 5.972e24;%kg
17 m_M = 6.717e23;%kg
18 m_S = 1.988e30;%kg
19 %mu of each body
20 mu_E = G*m_E;
21 mu_S = G*m_S;
22 mu_M = G*m_M;
23 %useful variables
24 c.ue = Isp*g0;
25 ue = c.ue;
26 c.mdot = T/ue;
27 %Radii
28 %Earth
29 r_E = 6376*1000;
30 r_LEO = r_E + 600*1000;
31 r_ES = 149e9;%semimajor axis around sun
32 r_EE = r_ES*(m_E/m_S)^(2/5);
33 %Mars
34 r_M = 3390*1000;
35 r_LMO = r_M + 200*1000;
36 r_MS = 228e9;%semimajor axis around sun (assuming circular orbits)
37 r_ME = r_MS*(m_M/m_S)^(2/5);
38 %final time
39 tf = 1e11;%big enough
40 %% BURN 1: r_LEO -> r_EE
41 disp("Initial Mass: " + mo + "kg")
42 % [t, vec] = odeCall(r0,rf,mu,tf,mo,c)
43   [t, vec] = odeCall(r_LEO,r_EE,mu_E,tf,mo,c);
44 dv1 = sqrt(mu_E/r_LEO)-sqrt(mu_E/r_EE);
45 ti = t(end);
46 mf = vec(end,5);
47 disp("Burn 1 Time: " + t(end)/3600/24 + "days")
48 disp("Burn 1 Final Mass: " + mf + "kg")
49 %% BURN 2: r_ES + rEE -> r_MS - r_ME
```

```
% [t, vec] = odeCall(r0,rf,mu,tf,mo,c)
  [t, vec] = odeCall(r_ES+r_EE,r_MS-r_ME,mu_S,tf,mf,c);
dv2 = sqrt(mu_S/(r_ES+r_EE))-sqrt(mu_S/(r_MS-r_ME));
ti = ti + t(end);
mf = vec(end,5);
disp("Burn 2 Time: " + t(end)/3600/24 + "days")
disp("Burn 2 Final Mass: " + mf + "kg")
%% BURN 3: r_ME -> r_LMO
r0 = r_LMO;
% [t, vec] = odeCall(r0,rf,mu,tf,mo,c)
  [t, vec] = odeCall(r_LMO,r_ME,mu_M,tf,mf,c);
dv3 = sqrt(mu_M/r_LMO)-sqrt(mu_M/r_ME);
mf = vec(end,5);
ti = ti + t(end);
disp("Burn 3 Time: " + t(end)/3600/24 + "days")
disp("Burn 3 Final Mass: " + mf + "kg")
dv_total = dv1 + dv2 + dv3;
t_total = ti;
mp = mo-mf;
P = c.mdot*c.ue^2/2;
disp("Total Time: " + t_total/3600/24/365 + "years")
disp("Mp: " + mp + "kg")
disp("DeltaV total: " + dv_total/1000 + "km/s")
disp("Power Required: " + P/1000 + "kW")
% Retrieve data from the ODE45 Solution
r = vec(:,1);     %Hint: rmember the order you put your initial conditions in
rdot = vec(:,2);
th = vec(:,3);
thdot = vec(:,4);
m = vec(:,5);
%Convert to Cartesian coordinates for plotting
x = r.*cos(th);
y = r.*sin(th);
%% Plot the Spiral Trajectory
figure
plot(x,y,'k--');
hold on
th = linspace(0,2*pi,1000);
r2 = r_ME;
xLEO = r0.*cos(th);
yLEO = r0.*sin(th);
xGEO = r2.*cos(th);
yGEO = r2.*sin(th);
%
% % h = r(end)*thdot(end)*r(end);%angular momentum
% % p = h^2/mu;%constant
% % v_end = sqrt((r(end)*thdot(end))^2 + rdot(end)^2);
% % a = (2/r(end) - (v_end^2)/mu)^-1;
% % e = sqrt(1-p/a);
% % disp(e)
% % rNEW = p./(1 + e*cos(th));
```

```
% % xNEW = rNEW.*cos(th);
% % yNEW = rNEW.*sin(th);
%graph LEO
plot(xLEO,yLEO,'b','LineWidth',2)
%graph GEO
plot(xGEO,yGEO,'r--','LineWidth',2)
% %graph new orbit
% % plot(xNEW,yNEW,'y','LineWidth',2)
% %These are just plotting commands to make your graphs look nicer
% title('Spiral Trajectory','fontweight','bold')
% rmax = max(r);
% axis(1.1*[-rmax rmax -rmax rmax])
% axis square
% xlabel('x (m)', 'fontweight', 'bold')
% ylabel('y (m)', 'fontweight', 'bold')
% legend("Trajectory","LEO","GEO","New Orbit")
cost = mp*1140;
function [t,vec] = odeCall(r0,rf,mu,tf,mo,c)
    p.mu = mu;
    tspan = [0,tf];
    thetadot0 = sqrt(mu/r0^3);
    S0 = [r0, 0, 0, thetadot0, mo]';

    optS = @(t,S) opt1(t,S,rf);
    options = odeset('Events', optS);

    LTfun = @(t,S) odefun_skeleton(t,S,p,c);

    [t,vec] = ode45(LTfun,tspan,S0,options);
end
function [value, isterminal, direction] = opt1(~,X,rf)
value = X(1) - rf;
isterminal = 1;
direction = 0;
end
```

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