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Propulsion Heat Analysis Nuclear Trajectory Optimization (PHANTOM)

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PROPULSION HEAT ANALYSIS FOR NUCLEAR TRAJECTORY OPTIMIZATION MODELING

By

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Final Report for 4600:461 Senior/Honor Design, Spring 2024

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Abstract

Interplanetary Travel requirements necessitate the use of increasingly complex propulsion systems. Nuclear Thermal Propulsion (NTP) is a cutting-edge concept that holds immense potential for the future of space exploration. By utilizing nuclear fission to heat a propellant and generate thrust, NTP offers the potential for unprecedented efficiency, enabling faster interplanetary travel and extended mission capabilities. A nuclear thermal rocket engine is a type of propulsion system that utilizes the heat generated by a nuclear reactor to heat and expel a propellant, typically hydrogen, at high velocities to produce thrust. This project will focus on the development of a thermal-hydraulic model created in MATLAB to analyze NTP system performance characteristics, the design of a Venusian Transfer Vehicle (VTV) and associated mission, and the use of orbital mechanics software for the evaluation of Venusian flight trajectories utilizing NTP. This endeavor demonstrates how nuclear-powered missions can simplify logistics for interplanetary missions, reducing time and costs, therefore enabling streamlined deep-space exploration.

1. Introduction

Interplanetary travel requirements necessitate the use of increasingly complex propulsion systems. Nuclear Thermal Propulsion (NTP) is one such propulsion system that has immense potential for space exploration. A Nuclear Thermal Rocket Engine (NTRE) is a type of propulsion system that utilizes the heat generated by a nuclear reactor to heat and expel a propellant, typically hydrogen, at high velocities to produce thrust. NTREs offer the potential for increased efficiency compared to chemical rocket engines, enabling faster interplanetary travel and extended mission capabilities.

Figure 1 - Overview of a solid nuclear thermal propulsion module by Borowski et al. (2009)
1.1 Nuclear Thermal Propulsion Advantages

The key advantage of a nuclear thermal rocket engine is its high specific impulse. Specific impulse refers to the efficiency of a propulsion system, measured by the amount of thrust generated per unit of propellant consumed. Specific Impulse can be expressed as follows:

\[ I_{sp} = \frac{F}{m} = AC_f \sqrt{\left( \frac{T_c}{M} \right)} \]  (1.1)

Typical chemical propulsion reactions produce combustion gases composed of carbon dioxide, water vapor, nitrogen, and trace quantities of other byproducts. These byproducts have molecular weights that contribute to the overall mass of the exhaust gases, reducing the specific impulse and limiting the efficiency of chemical propulsion systems. By comparison, hydrogen used in an NTP system has a molecular weight of two. The absence of combustion in a nuclear thermal rocket engine enables the optimization of the radical term in the above equation, resulting in a substantially higher specific impulse. NTP's superior specific impulse allows for more efficient use of the fuel and enables higher velocities compared to traditional chemical rocket engines.

NTP offers the potential to revolutionize space travel by significantly reducing mission durations. Conventional chemical propulsion systems often require long transit times due to their limited specific impulse. In contrast, NTP's high specific impulse allows for higher velocities, enabling spacecraft to reach destinations such as Mars in a fraction of the time required by chemical propulsion. By shortening travel durations, NTP reduces the exposure of astronauts to the hazards of prolonged spaceflight, including radiation and physiological challenges, ultimately enhancing the safety and feasibility of crewed missions.

In addition to reducing mission durations and expanding the frequency of space missions, nuclear thermal propulsion (NTP) brings forth the advantage of enhanced payload capabilities. The higher specific impulse offered by NTP allows spacecraft to carry larger payloads while still achieving the desired velocities for interplanetary travel. This increased payload capacity opens opportunities for more extensive scientific instruments, additional crew supplies, and even equipment for in-situ resource utilization (ISRU) on other celestial bodies. With greater payload capabilities, NTP enables more ambitious and comprehensive exploration missions, empowering scientists and engineers to gather more data, conduct more experiments, and pave the way for future exploration and colonization endeavors. By maximizing the efficiency of propellant usage and optimizing payload-carrying capabilities, NTP proves instrumental in propelling humanity further into the depths of space and unlocking new frontiers of knowledge.

Furthermore, the reduced mission durations enabled by NTP not only open extended mission capabilities but also present exciting possibilities for increased exploration and scientific discovery. The shorter travel times enable a higher frequency of missions within a given timeframe, allowing for more expeditions to various destinations in space that were once constrained by long transit durations. This increased cadence of missions not only accelerates our understanding of the solar system but also enhances the potential for scientific research, resource utilization, and the establishment of a sustained human presence beyond Earth.
1.2 Nuclear Rocket Engine Principles

In a nuclear rocket engine, the source of heat is a controlled fission chain reaction, as illustrated in Figure 2. The fuel in a nuclear core is typically Uranium-235, which is the same type broadly used in nuclear power plants, whose atoms can easily undergo fission\footnote{[1]}\textsuperscript{1}. In the present, all reactors have been designed to conform to the policies by the U.S. Nuclear Regulatory Commission, which are expected to operate with high-assay low enriched uranium (HALEU) containing 5-20\% of U-235 content\footnote{[2]}\textsuperscript{2}.

A nuclear rocket engine core is essentially a heat exchanger that is heated up by a controlled fission chain reactor and cooled down by the propellant, usually hydrogen (H\textsubscript{2}). The hydrogen is stored in a liquid state and heats up as it flows through the core propellant channels.

![Figure 2 - Nuclear fission chain reaction by NASA](image)

1.3 History of NTP Systems

Project Rover and the NERVA program, short for Nuclear Engine for Rocket Vehicle Application, epitomized the ambitious spirit driving space exploration in the mid-20th century. Conceived through collaboration between NASA and the Atomic Energy Commission from 1955 to 1973, Project Rover introduced a revolutionary nuclear thermal rocket technology. While NASA led the charge in the United States, concurrently, the Soviet Union pursued a parallel path in developing Nuclear Thermal Propulsion (NTP). This dual effort showcased a shared commitment to advancing propulsion capabilities for space travel, although neither NERVA nor its Russian counterpart saw active service in crewed missions. The intertwined history of NERVA and Soviet NTP illuminates an era marked by innovative strides and a collective determination to conquer the challenges of exploring the cosmos.
Rover/NERVA

Project Rover consisted of four major segments: KIWI, PHOEBUS, NERVA, and RIFT (Reactor in Flight Test). The KIWI series of reactors were a non-flyable configuration meant to test the most fundamental aspects of hot hydrogen reactor technology. This series of reactors were eight in total and were built/tested between 1965 and 1969. PHOEBUS was a series of high power and long duration nuclear test reactors. Three reactors were built and tested under this segment between 1965 and 1968. It was PHOEBUS 2A that achieved the highest thrust (200,000 lbf) and the highest power level (4,100 MW) ever achieved with a solid core NTRE. NERVA was an advanced phase of the Rover Program and is often perceived as the entirety of the American effort to develop NTREs in the 20th century. While not true, this perception is understandable as the NERVA segment of testing saw a fully successful ground test program utilizing a flight geometry NTRE (XE Prime). In total, six reactors were built and tested under NERVA from 1964 to 1969. The primary objective of RIFT was to design, develop, fabricate, and fly a NERVA powered upper stage for a Saturn Class vehicle. This was seen as an essential step towards furthering humanity’s place in the solar system. RIFT was cancelled due to difficulties experienced in early KIWI reactor testing.

Figure 3 - NERVA Program timeline
Soviet NTP

Soviet research into NTP began in the late 1950’s, and testing of developmental fuel for solid-core NTREs began in 1962\[^3\]. During this time, Soviet engineers viewed an expedition to Mars as the next step in human exploration after manned missions to the moon. When considering possible mission architectures for a human led mission to Mars, scientists and engineers gradually began looking to NTP as a viable alternative to chemical propulsion systems\[^4\]. While solid core and gas core reactors were considered, it was determined that solid core reactors were much closer to implementation.

The outcome of the Soviet effort to develop an NTRE was the manufacture and testing of the RD-0410. This was an NTRE capable of approximately 35 kN (7,868 lbf) of thrust and a specific impulse of 910 seconds\[^4\]. A major difference between the RD-0410 and the American Project Rover developed reactors was the form of the fuel elements used within the reactor. While American reactors adopted the hexagonal shape with fuel channels running through the element, the Soviet design resembled a twisted ribbon. This was intended to prevent the loss of nuclear material should cracking occur within the reactor elements.

![Twisted ribbon fuel elements](image)

Figure 4 - Twisted ribbon fuel elements

It was the intent of the soviet NTRE development program to continue development of the NTRE by designing and building the RD-0411. The engine was never realized but was designed to increase the thrust to approximately 392 kN (88,125 lbf). This iteration was intended to be used in lunar and martian missions.


1.4 Modern NTP Systems

The recent years have been marked by a joint effort to make nuclear propulsion a reality. Besides governmental organizations such as national laboratories and NASA, universities and commercial companies are heavily involved to push the boundaries of knowledge and feasibility of the technology, addressing the current challenges that surround nuclear propulsion reactors and rockets.

In the beginning of 2023, NASA administrator Bill Nelson officially announced the agency’s collaboration with the Defense Advanced Research Projects Agency (DARPA) to develop the Demonstration Rocket for Agile Cislunar Operations (DRACO)\(^5\). The project is intended as a proving ground to nuclear thermal engines that is expected to be launched as early as 2027. This endeavor includes the participation of Lockheed Martin and BWX Technologies (BWX-T\(^6\)).
2. Initial Design of Chamber and Nozzle

Using Solidworks modeling program, a Computer Aided Design of the combustion chamber and nozzle was created.

2.1 Computer Aided Design (CAD)

The design includes a 3-D Printed combustion chamber and nozzle, and a 3-D printed nozzle extension. The expansion ratio was selected to be $\frac{A_e}{A_t} = 100$ and the contraction ratio was selected to be $\frac{A_c}{A_t} = 17$. The two are fastened together using 18 x 1.5-6 countersunk bolts. At the top end of the chamber, there is a flange with 18 1.5-6 threaded holes for fastening the motor to a nuclear reactor. Additionally, regenerative cooling channels were added for physical representation, but their thermal and fluid characteristics were not analyzed. A rendering of the CAD model is found below in Figures 7 and 8.

![Figure 7 - Isometric view 1 of CAD design](image-url)
2.2 Finite Element Analysis

To further validate the model before using it for hydraulic analysis, a structural Finite Element Analysis (FEA) was created using ABAQUS CEA. The assumed material was Inconel 718\cite{13} where the Young’s Modulus and Poisson’s Ratio were input into the model. The back flange of the chamber was given a fixed boundary condition, and a load of 1,000psig was applied to all pressure bearing surfaces. The mapped von mises stress is shown below in Figure 9.
The max stress of the model was shown to be 98ksi, which corresponds to a margin of safety of 1.67 on the structure of the material. This was found acceptable to continue with hydraulic model analysis.

3. Thermal Hydraulic Model

Testing remains the ideal benchmark for system verification. However, the complexity, cost, and risk associated with NTP testing, both on ground and in space, creates the need for improved modeling of NTP systems. In the case of ground testing NTREs, even though they were not employed in NERVA testing, substantial effluent cleanup systems play a crucial role in future testing. These systems are designed to remove fission products from the reactor exhaust before release of the cleaned gas[8]. This helps to manage and mitigate the potential negative environmental impacts of the engine's exhaust. Meanwhile, in-space testing introduces a plethora of concerns encompassing regulations, logistics, costs, and public perception. Figure 10 shows a proposed effluent cleanup system for modern day NTP testing.
For these reasons, multiphysics modeling is employed to ascertain the performance of NTP systems in a theoretical scope. These forms of analyses serve to reduce boundaries to entry for those interested in contributing to the field. An NTRE consists of three main subsystems: the nuclear reactor, the propellant flow system, and the nozzle.

The nuclear reactor is the heart of the engine and generates heat through nuclear fission reactions. The heat produced by the nuclear reactions is transferred to the propellant via channels through the fissile material. It is in these channels where the majority of the heat is deposited into the propellant achieving extremely high temperatures before the propellant is expelled through the nozzle. The reactor is designed to operate at these extremely high temperatures to maximize the thermal energy available for propulsion.

The propellant flow system transports the propellant throughout the engine, ensuring a steady supply from onboard storage tanks or external sources. It incorporates turbomachinery to facilitate propellant circulation and maintain nominal propellant flow conditions. Additionally, the flow system incorporates regenerative cooling loops to manage the high temperatures generated by the engine. The cooling loops circulate the propellant around the engine to absorb and dissipate the excess heat, preventing damage to the engine components. This ensures the engine operates within safe temperature limits, optimizing its performance and reliability.

The nozzle is where the heated propellant is expelled at high velocities, creating thrust. The nozzle is designed to efficiently convert the high temperature and pressure of the propellant into directed exhaust flow, generating thrust according to Newton's third law of motion.
3.1 Model Setup

This analysis employed the use of MATLAB to develop a script capable of determining the propellant properties of hydrogen flowing through cooling channels in the reactor core and subsequent expulsion through the nozzle. This work does not attempt to model the propellant flow system upstream of the reactor inlet plenum. Figure 11 shows the elements of the system modeled for this work.

Component blocks are used to model the change between each state point. Each component block has a set of governing equations. Inputs are provided to the block and outputs are received from the block. The reactor outputs are used as inputs for the nozzle component block.

Cooling Channel Physics

The cooling channels were modelled in mass and an average power curve was used for the heat deposition equations. Heat deposition into the propellant was modelled using existing propellant property tables for hydrogen and the following governing equations:

**Heat Deposition:**

\[ Q = \dot{m}h \rightarrow h = \frac{\dot{Q}}{\dot{m}} \frac{dh}{dx} \rightarrow h_2 = h_1 + \frac{\dot{Q}}{\dot{m}}(x_2 - x_1) \]

**Pressure Drop:**

\[ \Delta P = f \frac{\rho \Delta x V_{\text{ave}}^2}{2D} + \frac{\rho}{2}(V_{i+1}^2 - V_i^2) \]

**Friction Factor:**

Laminar Flow - \( f = \frac{64}{Re} \) (Re < 2300)

Turbulent Flow

\[ f = 0.094 \left( \frac{D}{\epsilon} \right)^{0.325} + 0.53 \frac{D}{\epsilon} + 88 \left( \frac{D}{\epsilon} \right)^{0.44} \frac{1}{Re^{0.5}} \] (Re > 10,000)

Where: \( \varphi = 1.62 \left( \frac{D}{\epsilon} \right)^{0.434} \)

Nozzle Thermodynamics

To determine nozzle exit conditions, mach area ratios were used to relate local flow velocities to local nozzle areas. Mach area ratios play a crucial role in nozzle thermodynamics by determining the expansion of gases as they flow through a nozzle. These ratios are used to determine the exit velocity of the propellant from the nozzle and determine thrust. Propellant properties at the exit of the reactor were provided to the nozzle component block as propellant
inlet properties. The following governing equations were used to model the nozzle thermodynamics:

_Thrust/Specific Impulse:_

\[
F_{thrust} = \dot{m} V_e + (P_e - P_w) A_w
\]

\[
I_{sp} = \frac{1}{\dot{m}} \left\{ \frac{2 \gamma RT}{\gamma - 1} \left( 1 - \frac{P_2}{P_1} \right)^{\frac{\gamma - 1}{\gamma}} \right\}
\]

_Nozzle Thermodynamics:_

\[
R = C_p - C_v
\]

\[
\gamma = \frac{C_p}{C_v}
\]

\[
\frac{A}{A^*} = \frac{1}{M} \left[ \frac{2(1 + \frac{\gamma - 1}{2} M^2)}{\gamma} \right]^{\frac{\gamma + 1}{\gamma - 1}}
\]

\[
\frac{P}{P^*} = \left[ \frac{\gamma + 1}{2(1 + \frac{\gamma - 1}{2} M^2)} \right]^{\frac{\gamma - 1}{\gamma + 1}}
\]

\[
\frac{T}{T^*} = \left[ \frac{\gamma + 1}{2(1 + \frac{\gamma - 1}{2} M^2)} \right]^{\frac{\gamma - 1}{\gamma + 1}}
\]

\[
V = Mc = M\sqrt{\gamma RT}
\]

### 3.2 Analysis

The exit temperature of the hydrogen propellant from the nuclear reactor was found to be 2,712K. This was used in conjunction with the exit pressure to determine the enthalpy at this state point. Figure 12 shows the rise in temperature and enthalpy along the length of the reactor.

![Figure 12 - Temperature (left) and Enthalpy (right) rise through cooling channel](image-url)
The temperature and pressure at the reactor exit was then used in the nozzle equations to determine thrust and specific impulse. The thrust for this engine design was found to be 15,284 lbf and the specific impulse was found to be 849 seconds. These values clearly demonstrated the higher efficiency of Nuclear Thermal Propulsion in comparison to Traditional Chemical Propulsion technologies.

4 Interplanetary Transfer Analysis

The goal of this analysis was to scientifically examine the use of using nuclear thermal propulsion systems for transit between celestial bodies within our solar system. Nuclear thermal propulsion provides many benefits to interplanetary transit, and a key aim of this work was to demonstrate those benefits.

4.1 Concept of Operations

The scope of this paper was to evaluate the performance of an NTP engine to serve as the propulsion method for interplanetary transportation from Earth to Venus. The metric for this evaluation was one mission from Earth to Venus and back to Earth again. Figure 13 shows the overall mission design.

![Figure 13 - Orbit transfer between Earth and Venus](image)

4.2 Theory and Methods

For interplanetary transfer, an important parameter to consider is delta-V (delta velocity) of the vehicle. Two important values that are heavily dependent on delta-V is overall transit time and ability to leave Earth’s orbit and become captured in the orbit of another planetary body. Delta-V relates to rocket engine design through the Tsiolkovsky Rocket Equation shown below:
\[ \Delta V = I_{sp} g_0 \ln \left( \frac{m_0}{m_f} \right) \]

Where:

- \( I_{sp} = \text{Engine Specific Impulse} \)
- \( g_0 = \text{Standard Gravity} \)
- \( m_0 = \text{Wet Mass of Vehicle} = m_f + m_p \)
- \( m_p = \text{Mass of Propellant} \)
- \( m_f = \text{Dry Mass of Vehicle} \)

This equation was restructured to the following:

\[ m_p = m_f e^{I_{sp} g_0} - m_f \]

This equation was used to directly compare how much propellant is necessary to complete one burn to obtain a required delta-V. According to the concept of operations listed above, the burns necessary for one total mission are shown in the figure below:

![Diagram showing burns for one total mission](image)

Hence, the following equation can be used to determine the total propellant necessary for one mission:

\[ m_t = m_{exit} + m_{enter} + m_{exit} + m_{enter} \]

\( I_{sp} \) for the NTP engine was given by the hydraulic model shown earlier in the paper, while the \( I_{sp} \) for the chemical engine was assumed using statistics from the state-of-the-art SpaceX Raptor engine. Additionally, dry mass comes from spacecraft design, and standard gravity is constant. The delta-V requirement, however, varies depending on planetary body, and was found through conducting a Hohmann orbital transfer, which is currently one of the most efficient methods of moving from the orbit of one celestial body to another\[^{10}\]. A visual example of this is shown in Figure 14.
The Hohmann’s transfer requires four total burns to complete one mission:

1. Changing velocity from Earth’s orbit around the sun to enter the Hohmann’s transfer orbit.
2. Changing velocity from the Hohmann’s transfer orbit to enter Venus’ orbit around the sun.
3. Changing velocity from Venus’ orbit around the sun to enter Hohmann’s transfer orbit.
4. Changing velocity from Hohmann’s transfer orbit to enter Earth’s orbit around the sun.

The timing of such maneuvers was assumed to be optimal based on Earth and Venus’ position around the Sun, but the exact timing and date selection for this mission was not explored and is outside the scope of this paper. For this paper, the assumption was made that the Earth is at its closest point to the sun in its elliptical orbit. Venus’ orbit is mostly circular, so any deviations from concentricity was assumed negligible and its average radius from the sun was used. To calculate the velocities of these orbits, the vis-viva equation was used\(^8\):

\[
v_{\text{orbit}} = \sqrt{\frac{2}{r} - \frac{1}{a}}
\]

Where:
\[ \mu = \text{standard gravitational parameter of the central body} \]
\[ r = \text{radius between celestial bodies around the sun} \]
\[ a = \text{semi – major axis between the two celestial bodies} \]

4.3 Calculations

To calculate the mass of propellant necessary for the mission, calculating the delta-V necessary to complete the four necessary burns must be found. These are found through the four equations listed below:

\[
\begin{align*}
\Delta V_{\text{exit}} & = v_{\text{Transfer}} - v_{\text{Earth}} \\
\Delta V_{\text{enter}} & = v_{\text{Venus}} - v_{\text{Transfer}} \\
\Delta V_{\text{exit}} & = v_{\text{Transfer}} - v_{\text{Venus}} \\
\Delta V_{\text{enter}} & = v_{\text{Earth}} - v_{\text{Transfer}}
\end{align*}
\]

For the sake of simplifying this report, these calculations were consolidated into an excel spreadsheet. Rather than try to come up with a dry mass estimate for a theoretical space vehicle, the dry mass estimates were first assumed to be the same (85T), emulating the SLS and SpaceX Starship. However, Nuclear Thermal Propulsion engines, in their current technology readiness, are noticeably heavier than chemical ones; subsequently, a plot was created to show how much heavier a NTP spacecraft is allowed to be before the allowable propellant mass ends up being the same. The mass of the rest of the spacecraft was assumed to be the same since this is a comparison of propulsion methods. The results are shown below:

<table>
<thead>
<tr>
<th></th>
<th>Earth to Transfer</th>
<th>Transfer to Venus</th>
<th>Venus to Transfer</th>
<th>Transfer to Earth</th>
<th>Units</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \mu )</td>
<td>1.33E+20</td>
<td>1.33E+20</td>
<td>1.33E+20</td>
<td>1.33E+20</td>
<td>m^3/(s^2)</td>
</tr>
<tr>
<td>Radius of Earth</td>
<td>1.50E+11</td>
<td>1.50E+11</td>
<td>1.50E+11</td>
<td>1.50E+11</td>
<td>m</td>
</tr>
<tr>
<td>Radius of Venus</td>
<td>1.09E+08</td>
<td>1.09E+08</td>
<td>1.09E+08</td>
<td>1.09E+08</td>
<td>m</td>
</tr>
<tr>
<td>Semi-Major Axis of Earth/Venus</td>
<td>7.49E+10</td>
<td>7.49E+10</td>
<td>7.49E+10</td>
<td>7.49E+10</td>
<td>m</td>
</tr>
<tr>
<td>Delta-V</td>
<td>-2.87E+01</td>
<td>1103.775656</td>
<td>-1103.775656</td>
<td>28.67943354</td>
<td>km/s</td>
</tr>
</tbody>
</table>

Table 1 - Delta-V Calculations

<table>
<thead>
<tr>
<th></th>
<th>Chemical Propulsion Spacecraft</th>
<th>Nuclear Thermal Propulsion Spacecraft</th>
<th>Units</th>
</tr>
</thead>
<tbody>
<tr>
<td>Dry Mass</td>
<td>85000</td>
<td>85000</td>
<td>kg</td>
</tr>
<tr>
<td>Isp</td>
<td>363</td>
<td>950</td>
<td>Sec</td>
</tr>
<tr>
<td>Gravitational Constant</td>
<td>9.81</td>
<td>9.81</td>
<td>m/(s^2)</td>
</tr>
<tr>
<td>Burn 1 Propellant Needed</td>
<td>-681.8144472</td>
<td>-261.1733224</td>
<td>kg</td>
</tr>
<tr>
<td>Burn 2 Propellant Needed</td>
<td>30886.46002</td>
<td>10687.57669</td>
<td>kg</td>
</tr>
<tr>
<td>Burn 3 Propellant Needed</td>
<td>-22654.4939</td>
<td>-9493.855417</td>
<td>kg</td>
</tr>
<tr>
<td>Burn 4 Propellant Needed</td>
<td>687.3277411</td>
<td>261.978284</td>
<td>kg</td>
</tr>
<tr>
<td>Total Propellant Needed</td>
<td>8237.479412</td>
<td>1194.52623</td>
<td>kg</td>
</tr>
</tbody>
</table>

Table 2 - Calculations for propellant required
These results show that the requirement of propellant for a chemical propulsion spacecraft is approximately 6.9 times larger than that of a nuclear thermal propulsion spacecraft when assuming the same dry weight. This shows that the fuel tank requirements are much larger for a chemical propulsion spacecraft, which can increase the amount of allotted weight needed for fuel storage and decrease the amount of allotted weight in the spacecraft for payloads. Additionally, this demonstrates that for the same mass of allotted fuel and the same assumed dry mass, a nuclear thermal propulsion spacecraft can conduct 6 times as many missions as a chemical propulsion one. However, this efficiency goes down when NTP spacecraft become much heavier than chemical ones, so for NTP to become viable for a mission to Venus, its dry mass must not exceed the dry mass of a chemical engine by more than 6.88 times, which is shown in Figure 15.

5 Conclusion

In conclusion, when compared to current state-of-the-art chemical engines, the modeled engine used 6.88x less propellant for one mission from Earth to Venus and back, which means approximately 7 missions could be carried out consecutively with the same propellant mass using the modeled nuclear thermal propulsion system. Another way to interpret this is that the modeled engine could carry 6.88x more payload weight and use the same propellant as a current state-of-the-art chemical engine.


5.1 Accomplishments
This project won 1st Place / Best Overall Research Project at the University of Akron’s Senior Design Day. Additionally, the three minute video (link below) made for the project won Best Video for the Research Project section.

[https://youtu.be/nx1hf5Fwz8g](https://youtu.be/nx1hf5Fwz8g)

5.2 Future work
- Continued Development of the thermal hydraulic modeling tool to include transient analysis and upstream system contributions
- Further orbital trajectory analysis utilizing commercial simulation tools (ex. Orbital STK) to define the capabilities provided with the engine design.
- Transit time analysis using different orbital maneuvers to more completely grasp the impact NTP has on space exploration
- Higher fidelity engine design in CAD allowing for manufacturability and further analysis.
References


