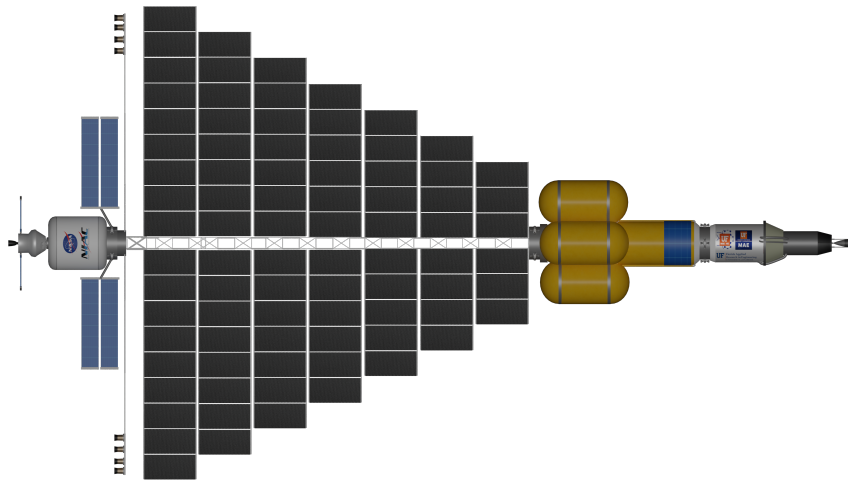


Wave Rotor Enhanced Nuclear (WREN) Propulsion

New Class of Bimodal NTP/NEP with a Wave Rotor Topping Cycle
Enabling Fast Transit to Mars

NASA Innovative Advanced Concepts (NIAC) - Phase I Final Report

Grant: 80NSSC23K0589



PI: Ryan Gosse

Co-PIs: Subrata Roy, Micheal Tonks, Justin Watson

Students: Garrison Osborne, Haris Mahmood, Sarah K. Burrows, Vishal Yadav,
Jeremy Smith, Tyler Mark, Jack Curley

University of Florida, Gainesville, Florida, 32611, USA

Co-PI: Anthony J. Colozza

HX5, LLC., Cleveland, OH 44135, USA

1 Nomenclature

A	Area (m^2)	PW	Power (kW)
a	Speed of Sound (m/s)	p	Pressure (kPa)
c_P, c_v	Specific Heat Capacity ($\frac{kJ}{kgK}$)	R	Universal Gas Constant
e	energy	SP	Specific Power ($\frac{kW}{kg}$)
F	Flux	T	Temperature, K
g	Earth Acceleration ($\frac{m}{s^2}$)	t	Time (s)
h	enthalpy	U	Conserved Variables
I_{SP}	Specific Impulse (s)	u	velocity
k	Structural Coefficient	V	Voltage (V)
M	Mach number	v	Velocity ($\frac{m}{s}$)
M_W	Molecular Weight ($\frac{g}{mol}$)	W	Work (kW)
m	Mass (kg)	w	shock speed (m/s)
$\dot{m}$	Mass Flow Rate ($\frac{kg}{s}$)	x	Distance (m)
PR	Pressure Ratio		

1.1 Greek

α	Specific Mass ($\frac{kg}{kW}$)	η	Efficiency
γ	Specific Heat Ratio	λ	Trajectory Characteristic
γ_{acc}	Thrust Acceleration ($\frac{m}{s^2}$)	Π	Pressure ratio across shock
ΔV	Impulse per unit Vehicle Mass ($\frac{m}{s}$)	σ	Stefan-Boltzmann Constant
ϵ	Emissivity	ϕ	Molar Fraction

1.2 Subscripts

avg	Average	Ma	Mars Arrival
Be	NEP Brayton Mode	o	Stagnation
Bt	NTP Brayton Mode	PL	Payload
$cable$	Power Distribution Cable	PP	Power and Propulsion
$comp$	Compressor	p	Propellant
Ed	Earth Departure	R	Right Side
e	Electric	rad	Radiator
et	Electric Thruster	$reject$	Heat Rejection
$exit$	Exit State	S	Structure
f	Final	s	Shock
Ht	Heliocentric Transfer	th	Thermal
i	Initial	tot	Total
ie	Induced by Expansion Fan	$trans$	Transit
is	Induced by the Shock	$turb$	Turbine
L	Left Side		

2 Executive Summary

Nuclear Thermal Propulsion (NTP) is identified as one of the preferred propulsion technologies for manned missions throughout the solar system (NASA MSFC).[1, 2] The state-of-the-art NTP cycle is based on a solid core Nuclear Engine for Rocket Vehicle Application (NERVA)[3] class technology (Fig. 1) that is envisioned to provide a specific impulse of 900 seconds doubling chemical rocket performance (450 seconds). Even with this impressive increase, the NTP NERVA designs still have issues providing adequate initial to final mass fractions for high ΔV missions.[4] Nuclear Electric Propulsion (NEP) can provide extremely high I_{sp} (2,000 to over 10,000 seconds) but with only low thrust and limits on mass to power ratios. The need for an electric power source also adds the issue of heat rejection in space where thermal energy conversion is at best 30-40% under ideal conditions. NASA Space Technology Mission Directorate (STMD) has recently expressed interest in finding advanced nuclear propulsion technology through the NASA Go:Thrust RFI.[5, 6] A novel Wave Rotor (WR) topping cycle has been proposed for our NASA NIAC concept. It promises to deliver similar thrust as NERVA class NTP propulsion, but with I_{sp} in the 1,200-2,000 second range. Coupled with an NEP cycle, the duty cycle I_{sp} can further be increased (1,800-4,000 seconds) with minimal addition of dry mass. This bimodal design enables fast transit trajectories for manned missions to Mars and revolutionizes the deep space exploration of our solar system.

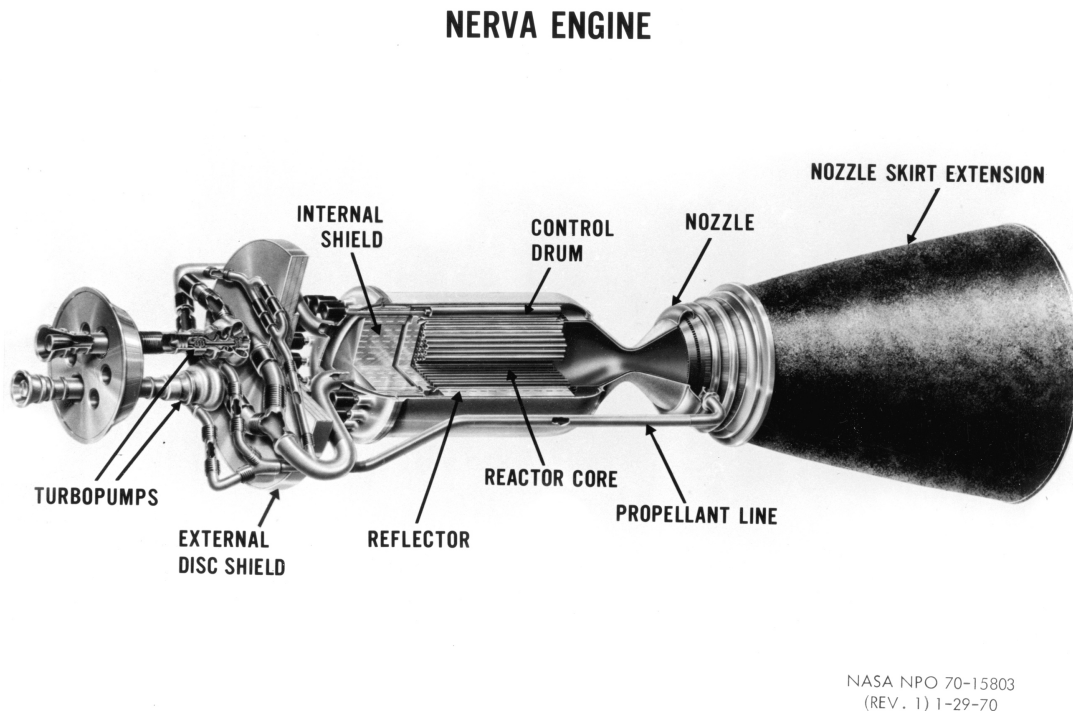


Figure 1: Drawing of the NERVA nuclear rocket engine.

3 Introduction

At a fundamental level, in-space Electric Propulsion (EP) is limited by the power cycle's ability to reject waste heat generated by its energy conversion process. This in turn produces a large heat load that requires an immensely heavy radiator, even at moderate thrust levels. Fig.2 shows the NASA Nuclear Electric Propulsion (NEP) - Chemical Propulsion (CP) Vehicle 1.2.[7] Note the large radiator area required for heat rejection of the 1.9 MW EP power system. Chemical and Nuclear Thermal Propulsion (NTP) designs, on the other hand, heat a working fluid and expel the hot gas with minimal heat transfer, reducing the cooling requirements to manageable levels at much higher levels of thrust.

The only way to reject heat in space is through radiation. If we assume the radiator is a black body, for discussion purposes, the power emitted is defined by,

$$PW_{\text{reject}} = A\sigma\epsilon(T_{\text{reject}}^4 - T_{\text{background}}^4) \quad (1)$$

In interplanetary space, the background temperature is typically 3-200K. Brayton cycles are typically considered for the electric generation for high power systems using a solid nuclear core as used in the NASA NEP-CP vehicle 1.2 reference vehicle. These systems typically provide a poor temperature to radiate waste heat. Focusing on the Brayton cycle turbine system to create electricity, higher rejection temperatures could be used but a fixed inlet temperature of the turbine means a lower output of power. To overcome this limitation, energy transfer processes with higher temperatures are required. In our work we explore the addition of wave rotor technology to overcome this limitation.



Figure 2: NASA NEP-CP hybrid concept.

Another important aspect for nuclear based propulsion is the selection of the motive gas for the rocket. The selection is directly impacted by the desired specific impulse (I_{sp}) and governed by the rocket equation as,

$$I_{sp} = \frac{1}{g} \sqrt{\frac{2\gamma}{(\gamma - 1)} \frac{RT}{M_W} \left(1 - \left(\frac{p_{exit}}{p_o} \right)^{\left(\frac{\gamma-1}{\gamma} \right)} \right)} \quad (2)$$

For a traditional chemical rocket utilizing hydrogen and oxygen, the combustion temperature is approximately 3420 K, the average molecular weight of the combusted gas is 13.8 g/mol, giving a maximum theoretical I_{sp} of 450 seconds with an optimized fuel rich mixing

ratio. For an NTP solid core based on the Nuclear Engine for Rocket Vehicle Application (NERVA)[3] design, reactor material limits the gas heating to a 2,700-3,000 K reactor exit temperature, but the rocket uses pure hydrogen that has a molecular weight of 2.016 g/mol. This gives the NTP a maximum I_{sp} of about twice that of the chemical rocket (900 sec). Reducing the molecular weight brings a significant performance increase. Fundamentally it would be ideal to reduce the molecular weight of the motive gas further. Fig.3 shows the phase change diagram for a pure hydrogen system. Over the pressure ranges relevant to rocket propulsion, complete atomization of hydrogen occurs in the range of 3150-5200 K. This is well beyond the service temperatures of solid core nuclear reactors. To reach these temperatures a liquid [8–12] or gas phase [13] reactor core would be required. Using an unsteady shock heating process, a wave rotor can theoretically reach liquid core performance while still using a nuclear reactor with a solid core. We have developed a unique cycle that leverages this capability.

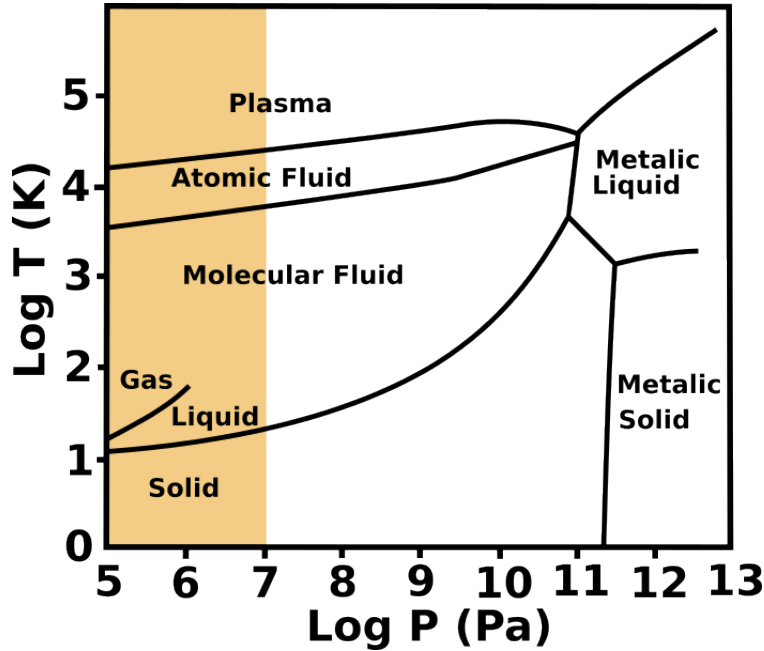


Figure 3: Phase change diagram for hydrogen system. Yellow box represents practical pressure ranges of in-space rockets.

Fig.4 lists nuclear core exit temperatures for the different classes (solid, molten, and gaseous). Liquid core designs are required to reach dissociation temperature. Current concepts involving molten core reactors rely on a centrifugal concept of rotating fuel elements [14]. The concept envisions heating hydrogen up to 5,000 K in the wave rotor. This leads to an estimated I_{sp} of 1,500 sec. Using unsteady shock heating processes, a wave rotor can reach liquid core performance with moderate temperature solid core heated gasses.

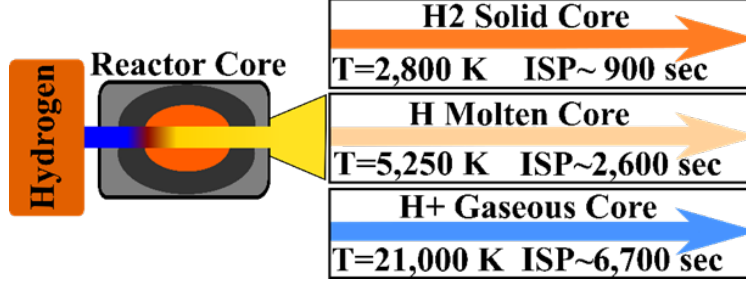


Figure 4: Solid, molten, and gaseous core nuclear reactor classes. Reactor core exit temperature, motive gas phase, and specific impulse are listed for each class.

4 Potential Impact

The proposed concept has the potential to revolutionize the design of NTPs based on solid cores. Materials available to build solid reactors for a traditional NERVA cycle place a hard limit on exhaust temperatures around 2850K. Advanced materials can push this further as discussed below but don't result in a revolutionary increase in I_{sp} compared to baseline NERVA technology. The addition of a wave rotor allows for additional heating with a driver gas that is heated to the 2850 K design limit. The unsteady nature of the wave rotor will allow for high temperature gasses to be processed using currently available high temperature materials. As laid out in the NASA STMD GO: Advanced Propulsion Capability Outcomes, hybrid NEP cycles are expected. This concept allow for the leapfrog of the "E3-A: NEP/CP Hybrid" outcome with a reduced nuclear core temperature (2600 K) "NTP-WR/NEP Hybrid" outcome greatly alleviating nuclear core material requirements. Then outcome "E3-B: NTP" can be replaced with an advanced optimized "NTP-WR/NEP Hybrid" outcome. Most significantly, both outcomes result in the ability to use fast-transit mission trajectories. Additionally, the technology would have the potential to transform the mission capabilities for all human and robotic deep-space missions.

As explained in the next section, the wave rotor operates using a driver gas that compresses the motive gas to be expelled out the spacecraft rocket nozzle. The two gasses do not have to be the same. The motive gas could be changed depending on the type of mission. For instance, the same reactor core could be used with LH₂, LOX, H₂O, or NH₃ motive gas if in-situ fuel processing plants are built on the Moon or Mars. The wave rotor may need to be changed out, but it is a simple small component. Different wave rotor driver gasses could also be used such as N₂, Ar, H₂O, and NH₃ compared to traditional nuclear power Brayton cycle gases such as He and Xe. Finally, the wave rotor could be equipped with variable timing ports on the end plates. Driver and motive gases would mix decreasing the gas temperature and increasing mass flow rate through the rocket nozzle. This would temporarily increase thrust (but reduce I_{sp}) for missions that require leaving a deep gravity well. The wave rotor also has gaps between the rotor and end plates, additional gas could be fed through gap leakage to trade thrust vs. Isp on a given mission. These types of analysis were beyond the motivation for the phase I study, but give motivation for the impact of the technology to other missions.

5 Technology Maturation

5.1 Wave Rotor Technology

A wave rotor, wave-in-rotor, or dynamic-pressure exchanger (primarily in Europe) is a type of turbomachine in which energy is exchanged using shock waves onboard the rotor. A wave rotor can be configured with a variety of ports controlling the processing of the gas in the rotor channels. We chose to investigate a four-port wave rotor as shown in Fig.5. This configuration consists of a rotor, two end plates and a set of transition ducts. On the left side of the wave rotor, a High Pressure Driver Gas [HPDG] and Low Pressure Compressed Gas [LPCG] are directed through the ducts. Inflow ports are sized on the end plate to induce a harmonic injection of the two gasses. The rotor is made of a series of channels that operate as individual rotating shock tubes, compressing the LPCG gas with the HPDG gas. The technology has been used in combustion applications, refrigeration cycles and test gas compression/heating. To our knowledge, this system has never been explored for application as a topping cycle for an NTP rocket.[15–22]

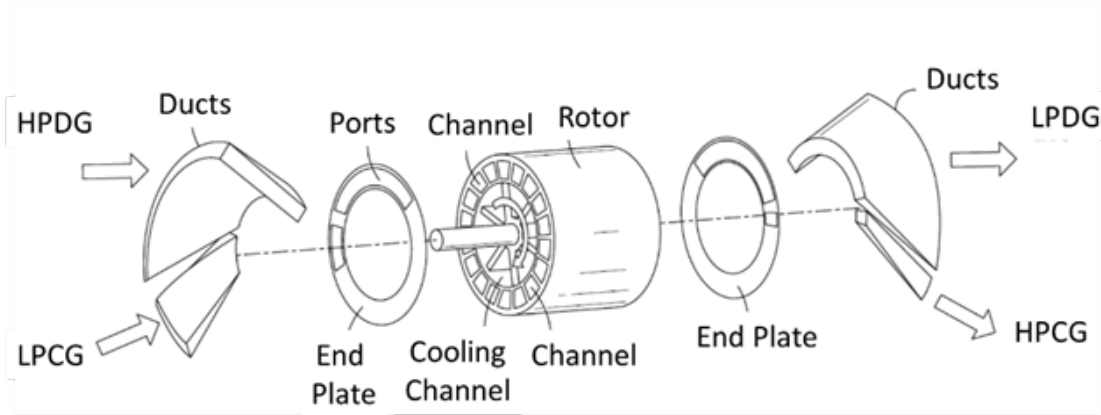


Figure 5: Exploded view of a typical four port wave rotor. HPDG - High Pressure Driver Gas, LPDG - Low Pressure Driver Gas, LPCG - Low Pressure Compressed Gas, HPCG - High Pressure Compressed Gas.

Fig.6 shows a diagram of the compression and expansion waves in a four port wave rotor. In this figure the rotor from Fig.5 is cut and unwrapped from a cylinder to a flat plane with periodic boundaries. For visualization purposes, we assume that the rotor has a large number of channels making the volume of each infinitely small. This assumption allows for visualizing the fluid as a continuous domain. This visualization is similar to an x-t diagram for tracking the evolution of shock tubes. Inside the channels the blue color denotes the compressed gas and the gold color is the driver gas. At the interface of the two gases, a contact discontinuity exists. When the driver gas inlet port is opened, the contact discontinuity initiates a shock compression wave (red arrow). This wave travels at the convective velocity of the fluid plus the speed of sound. It therefore travels faster than the driver gas and compresses the compressed gas as it passes through it. When the shock hits the right wall of the channel (before the compressed gas outlet port opening) the shock reflects back further increasing the pressure of the compressed gas. Again the reflected shock

travels faster than the velocity of fluid. Therefore, the compressed gas outlet port can be opened without changing the nature of the compressing shock. Once the reflected shock reaches the left side of the channel, the driver gas inlet port is closed. An expansion wave is initiated by the end of driver gas injection, causing the driver gas velocity to go to zero. When the contact discontinuity reaches the compressed gas outlet port on the right, the port is closed. This traps the driver gas in a stagnate state. The driver gas outlet port is then opened to remove the driver gas. Another expansion wave is created that imparts the velocity to the gas for removal. When the expansion wave reaches the left side, the compressed gas inlet port is opened. The driver gas outlet port is then closed when the contact discontinuity between the two gases reaches the right side. When the driver gas outlet port is closed, another compression shock is initiated stagnating the compressed gas. When the shock reaches the left side of the channel, the compressed gas inlet port is closed. With the compressed gas stagnated, we reach a periodic boundary that enables the unsteady operation of the wave rotor.

5.2 Brayton Cycle with Topping Wave Rotor

Wilson and Paxson [23] along with Welch [24] conducted research on wave rotors for use as a topping cycle for jet engines. Their research found that the macroscopic balances of the volume-averaged thermodynamic properties in the rotor passage control volume can provide a simple system level analysis of performance. Fig.7 illustrates the use of a four port wave rotor in a Brayton cycle with a nuclear reactor core. We start at station 1.0 where low pressure working fluid enters the compressor. The air is compressed using a traditional axial or centrifugal compressor. The gas at station 2.0 enters the wave rotor and is further compressed by the hot gas entering at station 4.1. The gas then passes through station 3.0 at a higher pressure and travels to the nuclear reactor core (station 3.1). The nuclear reactor core heats the gas up to station 4.0. The hot gas then travels to 4.1 where it compresses the incoming low pressure gas from station 2.0. The gas then enters the turbine at 5.0 to extract the rest of the work from the gas. In between stations 1.0 and 6.0 the gas is cooled by a radiator to reject the excess enthalpy from the cycle. The radiator area sets the bottom temperature of the cycle and the nuclear reactor outlet temperature sets the top of the cycle.

Treating the wave rotor as a single control volume, we see that neglecting viscous effects, the work produced by the fluid entering and exiting the ports must be equal.

$$\dot{m}_{3.0-2.0} \int_{T_{2.0}}^{T_{3.0}} c_p(T) dT = \dot{m}_{4.1-5.0} \int_{T_{5.0}}^{T_{4.1}} c_p(T) dT \quad (3)$$

For this case the mass flow rate is the same throughout the system and can be neglected. Assuming we can model the compression and expansion of the gas as an overall polytropic process and that the heat capacity is only a function of gas temperatures by taking the average over the ports, the work between the inlet and outlet ports can be found by defining pressure ratios between the ports,

$$\dot{m}_{3.0-2.0} \int_{T_{2.0}}^{T_{3.0}} c_p(T) dT = \dot{m}_{3.0-2.0} \frac{c_{p,avg} T_{2.0}}{\eta_{3.0-2.0}} [PR_{3.0-2.0}^{\frac{\gamma-1}{\gamma}} - 1] \quad (4)$$

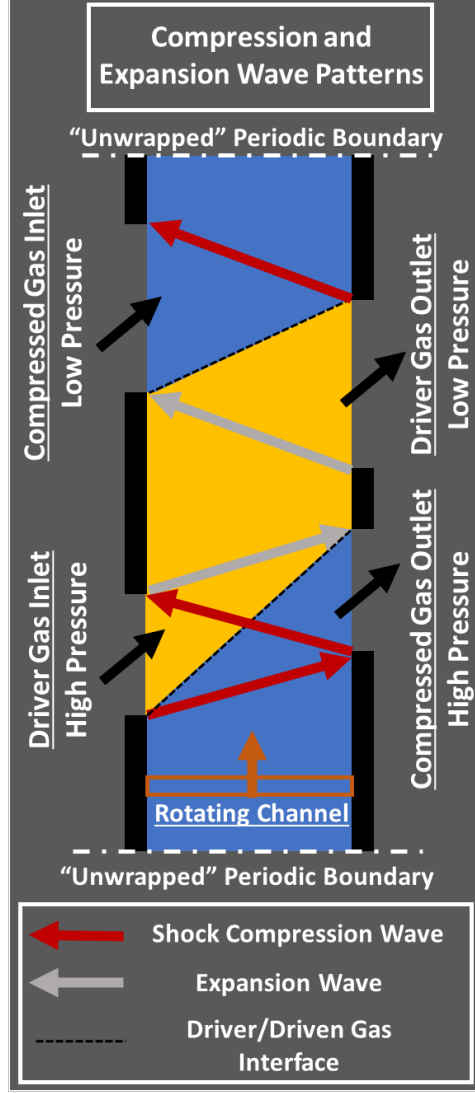


Figure 6: Compression and expansion wave patterns in a four port wave rotor.

$$\dot{m}_{4.1-5.0} \int_{T_{5.0}}^{T_{4.1}} c_p(T) dT = \dot{m}_{4.1-5.0} \eta_{4.1-5.0} c_{p,avg} T_{4.1} \left[1 - \left(\frac{PR_{2.0-5.0}}{PR_{3.0-2.0}} \right)^{\frac{\gamma-1}{\gamma}} \right] \quad (5)$$

Where now the work can be computed using the inlet port temperatures and the pressure ratios. Equating the right hand side of Equations 4 and 5, the required pressure ratio from 2.0-5.0 can be computed if we assume the mass flow rate is the same through all ports as is the case for a nuclear powered Brayton cycle as opposed to one that involves a combustion process and the addition of fuel.

$$PR_{2.0-5.0} = PR_{3.0-2.0} \left[\frac{1 - \frac{(1-\eta_{4.1-5.0})}{\eta_{4.1-5.0} \eta_{3.0-2.0}} \frac{T_{2.0}}{T_{5.0}} \left[PR_{3.0-2.0}^{\frac{\gamma-1}{\gamma}} - 1 \right]}{1 + \frac{1}{\eta_{3.0-2.0}} \frac{T_{2.0}}{T_{5.0}} \left[PR_{3.0-2.0}^{\frac{\gamma-1}{\gamma}} - 1 \right]} \right]^{\frac{\gamma}{\gamma-1}} \quad (6)$$

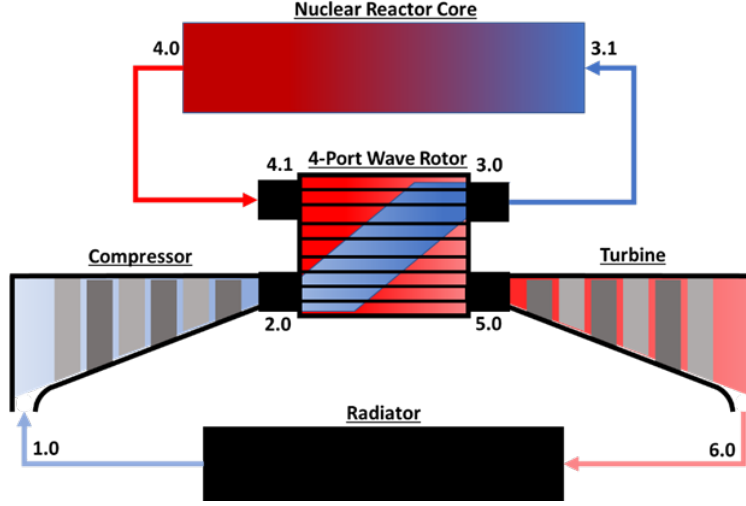


Figure 7: Diagram of Brayton cycle with topping 4-port wave rotor.

For completeness, the compressor power can be modeled using,

$$W_{\text{comp}} = \dot{m}_{1.0-2.0} c_{p,\text{avg}} \frac{T_{1.0}}{\eta_{1.0-2.0}} \left[PR_{1.0-2.0}^{\frac{\gamma-1}{\gamma}} - 1 \right] \quad (7)$$

And the turbine can be modeled using,

$$W_{\text{turb}} = \dot{m}_{5.0-6.0} c_{p,\text{avg}} \eta_{5.0-6.0} T_{5.0} \left[1 - \left(\frac{1}{PR_{5.0-6.0}} \right)^{\frac{\gamma-1}{\gamma}} \right] \quad (8)$$

Balancing the work of the system, we see that the wave rotor can be viewed as removing the required compression from the compressor by using thermal energy from the reactor above the operating temperature of the turbine system. Due to the simple channel design and low rotation speeds compared to a traditional compressor, the wave rotor provides only a small addition of mass to the system compared to the significant increase in system net power. On a traditional Brayton cycle for space propulsion, the radiator rejects heat at average temperatures of at least 500 K for mass efficiency. One strategy of adding the wave rotor is pushing the rejection temperature up to a much higher value. Instead of adding extra power, it may be possible to significantly reduce the mass of the other propulsion system components resulting in a lower mass to power ratio. We will discuss this later as part of the bi-modal design.

5.3 Motive Gas Heating with Wave Rotor

NTP propulsion based on the NERVA concept is an elegantly simple expander cycle. The top of Fig.8 shows a detailed build up of the system based on recent work at NASA Glenn [25]. The bottom part of the figure shows the simplified system components that are relevant for our analysis. The cycle consists of a compressor that takes the motive gas from a fuel tank and pressurizes the gas using turbomachinery. For the NERVA expander cycle, the turbomachinery is run by expanding a portion of the motive gas through a turbine and

exiting the spacecraft out a separate small exhaust nozzle. The compressed gas then flows through the NTP reactor core where it is heated up to the desired temperature. The gas leaves the core and then is exhausted through the rocket nozzle. Again, the limiting factor maximizing I_{sp} is the maximum temperature of the NTP reactor core.

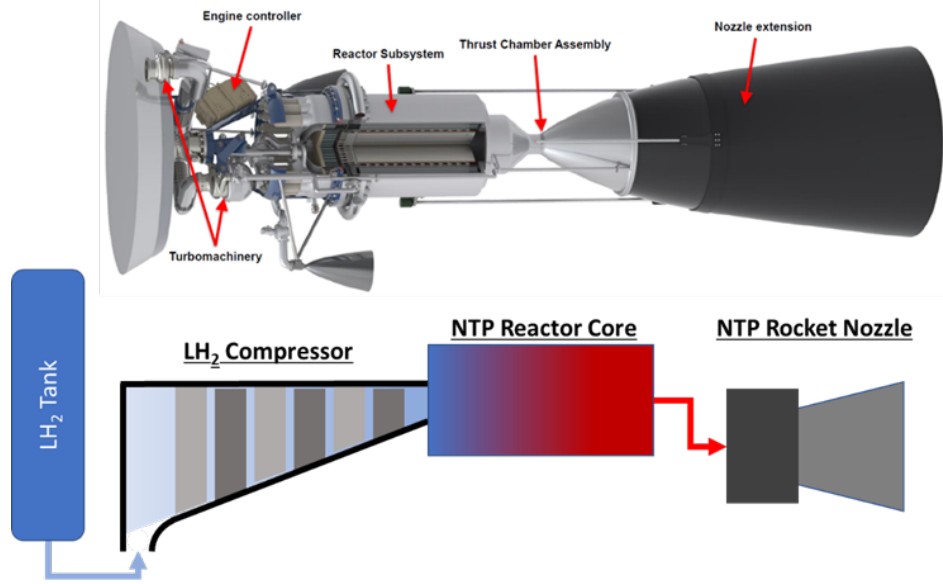


Figure 8: NTP expander cycle. Detailed drawing from NASA Glenn (top) [25]. Simplified representation (bottom).

To increase the performance of the NTP mode, a wave rotor can be placed in between the NTP reactor core and the rocket nozzle as shown in Fig.9. In this configuration, the motive gas is compressed by a driver gas. Motive gas denotes that gas is expelled by the rocket to produce motive force. Driver gas denotes the gas used to compress the motive gas (similar in reference to a driver gas in a shock tube). The driver gas in our case is also referred to as the Brayton cycle gas. The purpose of the driver gas is not to increase the pressure of the motive gas (as was the case for the Brayton cycle wave rotor), but to increase the motive gas temperature to values significantly beyond the traditional NTP NERVA design. Equations 4 and 5 can be recast into,

$$\dot{m}_{10.0-9.1} \int_{T_{9.1}}^{T_{10.0}} c_p(T) dT = \dot{m}_{10.0-9.1} \frac{c_{p,avg} T_{9.1}}{\eta_{10.0-9.1}} [PR_{10.0-9.1}^{\frac{\gamma-1}{\gamma}} - 1] \quad (9)$$

$$\dot{m}_{4.2-4.3} \int_{T_{4.2}}^{T_{4.3}} c_p(T) dT = \dot{m}_{4.2-4.3} \eta_{4.2-4.3} c_{p,avg} T_{4.2} [1 - (\frac{PR_{10.0-9.1}}{PR_{4.2-4.3}})^{\frac{\gamma-1}{\gamma}}] \quad (10)$$

Where the mass flow rate between the two do not have to be the same. This increases the number of unknowns, but we are now free to select the pressure ratio for both flow paths as an input parameter. The temperature inputs are also set by the reactor temperatures. Given a required thrust, the mass flow rate of 10.0-9.1 is fixed and we find the required mass flow rate for 4.2-4.3. We also point out that the temperature of the driver gas at 4.3 can be defined as the temperature required for the Brayton cycle wave rotor temperature

at 4.1. Using this constraint, the pressure ratio of 4.2-4.3 can be found. Therefore, all the requirements for the NTP wave rotor can be defined.

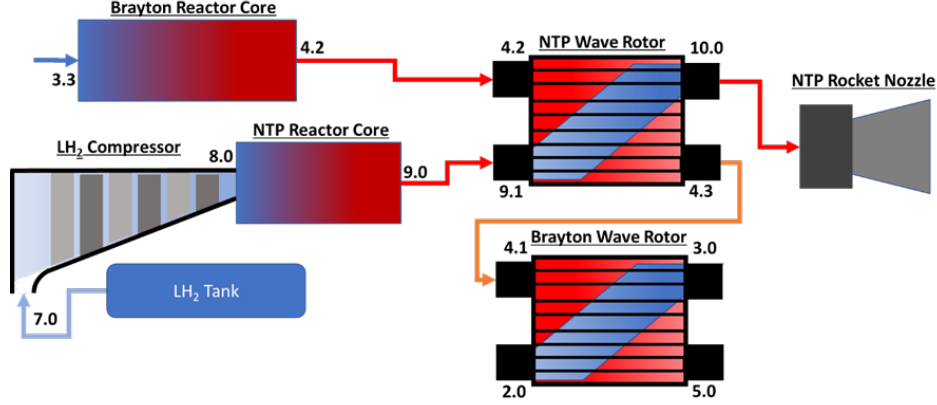


Figure 9: NTP mode with wave rotor placed between reactor core and rocket nozzle. Brayton cycle components from Fig.7 are shown to describe where the power to heat the motive gas comes from.

5.4 Wave Rotor Sizing

The macroscale model presented in the previous section allows for estimations of the performance of the wave rotor at a systems level. To start working on a detailed design, Computational Fluid Dynamics (CFD) is typically used to define the sizing of the wave rotor. This methodology starts with a simple one-dimensional modeling as a first round of analysis. The sizing of the port opening and closing is highly dependant of the stagnation properties of the ports. Typically a optimization method is coupled to a CFD solver in order to find correct values. To reduce the number of possible solutions, analytical models have been developed. They are very limited in the assumptions made and tend to give incorrect answers, but their value is found in helping reduce the number of possible solutions for the CFD optimization methodology.

5.4.1 Analytical Model

As shown in Fig. 6, the operation of the wave rotor consists of a series of shock waves and expansion fans. If we consider it an ideal one-dimensional flow, the normal shock waves and expansion fans can be modeled using a series of analytical equations. To model shock waves, we apply a control volume around the vicinity of the shock wave. Assuming the flow is inviscid and applying the Euler equations results in

$$\rho_L(w_s + u_L) = \rho_R(w_s + u_R) \quad (11)$$

$$p_L + \rho_L(w_s + u_L)^2 = p_R + \rho_R(w_s + u_R)^2 \quad (12)$$

$$h_L + \frac{(w_s + u_L)^2}{2} = h_R + \frac{(w_s + u_R)^2}{2} \quad (13)$$

Assuming the fluid is calorically perfect (not always valid at high temperatures) the above equations can be solved for local temperature and density ratios.

$$\frac{T_R}{T_L} = \Pi_s \left(\frac{\frac{\gamma+1}{\gamma-1} + \Pi_s}{1 + \frac{\gamma+1}{\gamma-1} \Pi_s} \right) \quad (14)$$

$$\frac{\rho_R}{\rho_L} = \frac{1 + \frac{\gamma+1}{\gamma-1} (\Pi_s)}{\frac{\gamma+1}{\gamma-1} + \Pi_s} \quad (15)$$

where

$$\Pi_s = \frac{p_R}{p_L} \quad (16)$$

Inspecting the above equations, the local thermodynamic properties across a moving normal shock are physically independent of the flow field velocities. This allows for the solution of the velocity induced by the shock

$$u_{is} = \frac{a_L}{\gamma} (\Pi_s - 1) \sqrt{\frac{\frac{2\gamma}{\gamma+1}}{\Pi_s + \frac{\gamma-1}{\gamma+1}}} \quad (17)$$

where a_L is assumed to be the upstream side of the moving shock. The post shock velocity can then be found

$$u_R = u_L - u_{is} \quad (18)$$

The shock velocity is

$$w_s = a_L \sqrt{\frac{\gamma+1}{2\gamma} (\Pi_s - 1) + 1} \quad (19)$$

The shock Mach number is defined by

$$M_s = \frac{w_s + u_L}{a_L} = \sqrt{\frac{\gamma+1}{2\gamma} (\Pi_s - 1) + 1} \quad (20)$$

For modeling expansion waves, we assume they are isentropic in behavior in this simplified model. This allows the method of characteristics to be used to find the flow properties across an expansion wave. Casting this method in a one-dimensional form.

$$u + \frac{2a}{\gamma-1} = \text{Constant} \quad (21)$$

where u is the flow velocity and a is the local speed of sound. The characteristic can also run in the opposite direction resulting in

$$u - \frac{2a}{\gamma - 1} = \text{Constant} \quad (22)$$

Both constants are known as the Riemann invariants. For the case of a wave rotor port closing, this induces an expansion wave. We can apply the backward characteristic for the situation where the closing occurs on the right hand side of the one-dimensional flow. This gives

$$u_L - \frac{2a_L}{\gamma - 1} = u_R - \frac{2a_R}{\gamma - 1} \quad (23)$$

Noting that the wall enforces $u_R = 0$, the ratio of the speed of sounds can be found

$$\frac{a_R}{a_L} = 1 - \frac{\gamma - 1}{2} \frac{u_L}{a_L} \quad (24)$$

allowing for the temperature, pressure and density ratios for an isentropic flow to be found

$$\frac{T_R}{T_L} = \left[1 - \frac{\gamma - 1}{2} \frac{u_L}{a_L} \right]^2 \quad (25)$$

$$\frac{p_L}{p_R} = \left(\frac{T_R}{T_L} \right)^{\frac{\gamma}{\gamma - 1}} \quad (26)$$

$$\frac{\rho_R}{\rho_L} = \left(\frac{T_R}{T_L} \right)^{\frac{1}{\gamma - 1}} \quad (27)$$

The induced velocity due to the expansion wave can be found by using Eq.26

$$u_{ie} = \frac{2a_L}{\gamma - 1} \left[1 - \left(\frac{p_R}{p_L} \right)^{\frac{\gamma - 1}{2\gamma}} \right] \quad (28)$$

Again the induced velocity is independent of the flow velocities to the right and left of the wave. The relations for an expansion fan created by a port opening can similarly be derived. To close the relations for the shock, we also include the fact that the mass flow rate of each of the outlets must be consistent with the inlets. This allows for defining the timing for the compressed gas inlet and driver gas outlet as shown in Fig.6.

A Matlab code was written for a four-port wave rotor using the above relations. Fig.10 shows the output of the port timing for a 0.3 meter wave rotor. The results are shown in a typical x-t diagram used for mapping one-dimensional shock problems. We ran into issues getting consistent port times by closing out the set of equations using continuity of the mass flow rate. Use of these equations for other applications found similar issues [26] when compared to CFD and experiment. As the CFD modeling capability is matured this methodology will be integrated into the solver as a possible way to reduce the number of simulations required to find correct port timings even if computed timing values are off.

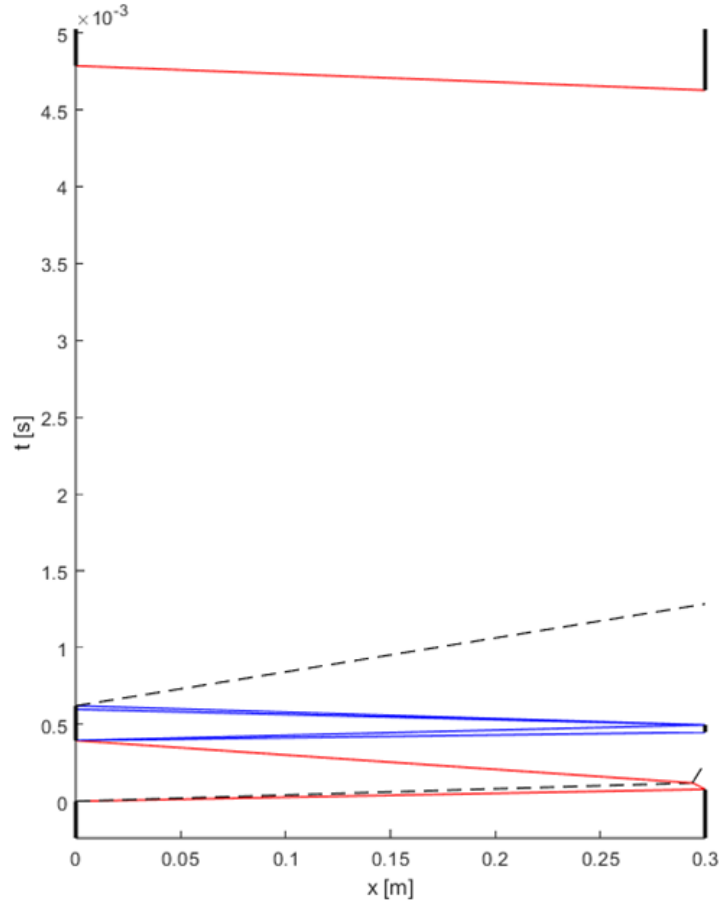


Figure 10: Example analytical result for port timing. Red lines are compression shock and blue lines expansion wave. Dotted line is contact discontinuity between driver and motive gas.

5.4.2 Computational Fluid Dynamics Model

During the phase I study, we did not anticipate applying effort developing a CFD solver to study the behavior of the different wave rotors. During the kick-off meeting, it was noted that having visuals of the dynamic processes inside the wave rotor would help explain how it operates. We therefor started some preliminary development of a CFD solver to be used for modeling. An undergraduate honors student at the University of Flordia spent time writing a one-dimensional CFD code. The code was developed based on the inviscid Euler equations to help us better understand how the analytical port timing model has issues replicating the port timing. During phase I the student was able to develop the baseline solver and implement proper boundary conditions for the flow. We did not have time to integrate an iteration loop to solve for port timing by the end of the project. The iterative loop taken from [27] is shown in Fig.11.

The Euler equation for a one-dimensional flow is given by,

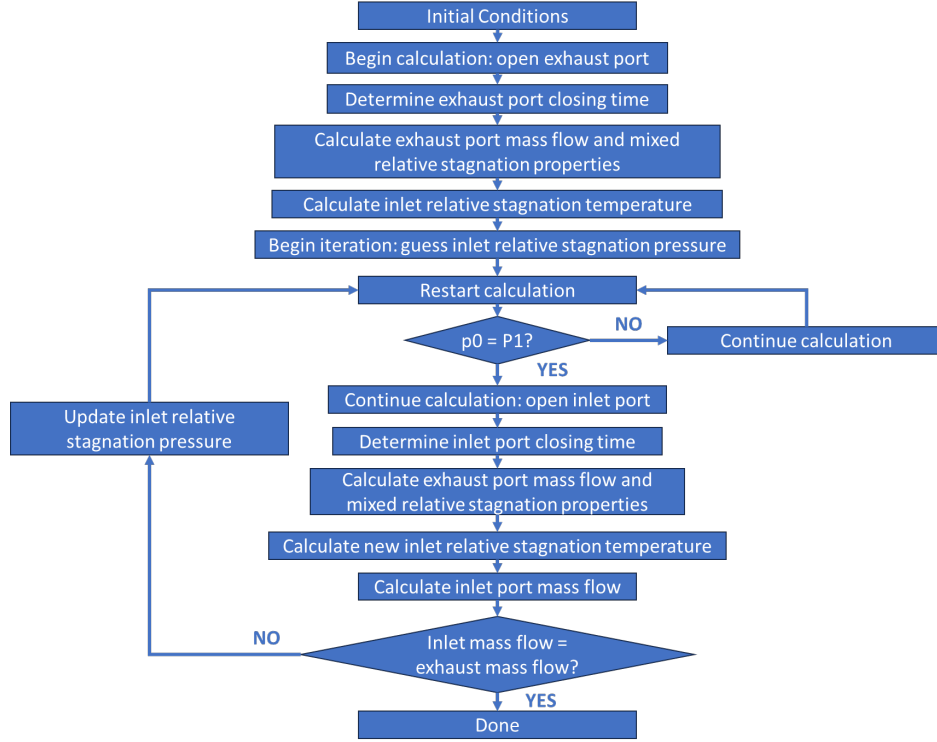


Figure 11: Four port iteration flow chart.

$$\frac{\partial U}{\partial t} + \frac{\partial F}{\partial x} = 0 \quad (29)$$

where

$$U = \begin{bmatrix} \rho \\ \rho u \\ e \end{bmatrix} \quad (30)$$

$$F = \begin{bmatrix} \rho u \\ \rho u^2 + p \\ (e + p)u \end{bmatrix} \quad (31)$$

For the initial development we assume a perfect gas

$$p = \frac{\gamma - 1}{p\epsilon} \quad (32)$$

with

$$\epsilon = c_v T = \left(\frac{e}{p} \right) - \frac{1}{2} u^2 \quad (33)$$

The unsteady flow was discretized using the Jameson Runge-Kutta-like scheme [28]

$$\begin{aligned}
U_i^{(0)} &= U_{i,j}^n \\
U_i^{(k)} &= U_i^n - \alpha_k \Delta t \left(\frac{D_0}{\Delta x} F_i^{(k-1)} \right) \\
U_i^{n+1} &= U_i^{(4)}
\end{aligned} \tag{34}$$

where D_0 denotes it's use with finite difference methods and for each stage,

$$\alpha_k = \frac{1}{5-k} \text{ for } k = 1, 2, 3, 4 \tag{35}$$

Inflow and outflow boundary conditions were implemented using the characteristic methods as developed by [29].

5.5 Bi-Modal Core Operation

For the bi-modal core operation, the conditions for the NTP reactor core materials are extremely harsh. They must operate at temperatures ranging from 1000 K to nearly 3000 K with high temperature gradients and temperature cycling over time, as well as exposure to high-temperature flowing hydrogen. However, the propulsion cycles are short with a total irradiation time of only two days for a 45-day mission and less than 10 hours for a 65-day mission. Thus, they will not experience significant amounts of radiation damage or fission product buildup.

Three material systems are being considered for use in the fuel for NTP reactors: two composite fuel systems and one monolithic system. They all have strengths and weaknesses regarding use in NTP reactors. They are compared in Fig. 12, which shows the reactor exit temperature vs. projected endurance in hydrogen. Each is discussed in more detail, below.

In the ceramic-metal (cermet) composite concept, particles of a ceramic fuel are embedded in a metallic matrix. Early cermet concepts used a cermet fuel composed of uranium dioxide (UO_2) particles embedded in a tungsten (W) matrix [30]. Both of these materials has a melting temperature above 3000 K. However, this design required high enrichment to generate the desired temperatures due to the low uranium density of UO_2 and the high neutron cross-section of W. A modern cermet concept uses uranium nitride (UN) particles embedded in a W-molybdenum (Mo) matrix. UN has a higher uranium density than UO_2 and still has a melting temperature above 3000 K. Increasing the Mo content in the W-Mo alloy decreases the neutron cross-section but also decreases the melting temperature. One major limitation of this concept is the decomposition reaction that occurs in UN fuel above 2120 K that is likely to result in fuel loss during operation [31].

In the ceramic-ceramic concept, ceramic particle fuel is embedded in a ceramic matrix [32]. The plan is to take advantage of the tristructural isotropic (TRISO) fuel particle technology in which a small fuel kernel is covered by various layers of carbonaceous materials. TRISO particles have demonstrated excellent performance for power reactors [33]. The fuel kernel could be made from either UN or a carbide fuel and zirconium carbide (ZrC) will be used in place of the SiC layer used in TRISO fuel. The fuel particles will be embedded in a ZrC or niobium carbide matrix [34]. ZrC has a melting temperature above 3000 K and a very low neutron cross-section. If UN is used as the fuel kernel, the decomposition reaction

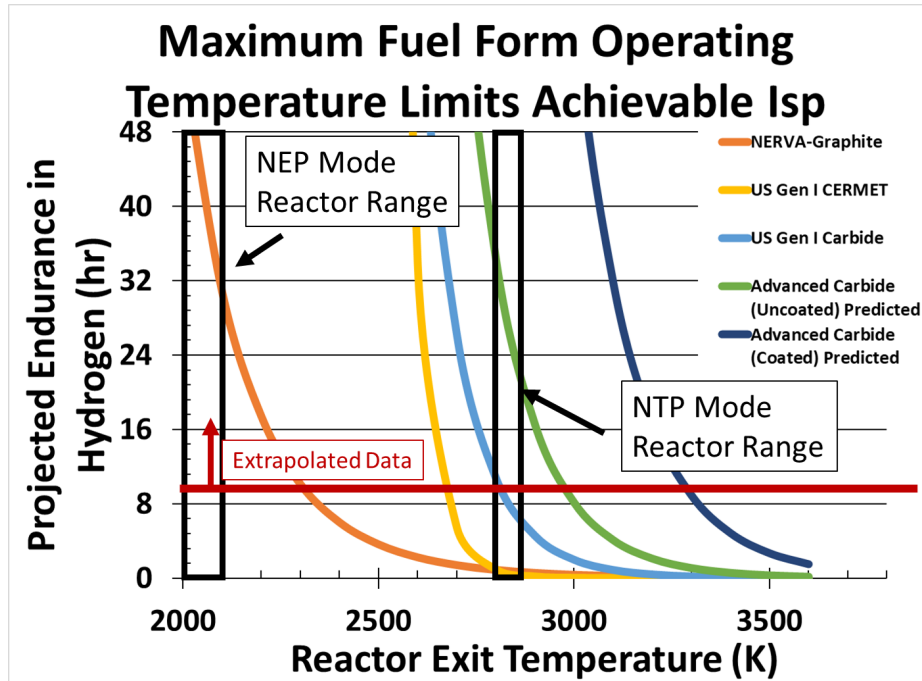


Figure 12: Projected Endurance of nuclear core material technology vs. reactor exit temperature.

discussed in the previous paragraph would still be an issue. If uranium carbide (UC) is used as the fuel material, it would need to be mixed with refractory carbides such as ZrC to raise the melting temperature.

In the advanced carbide concept, a solid solution of ceramic materials is used as the fuel material [35]. Possibilities include a bicarbide (UC and ZrC) [36] and a tricarbide (UC, ZrC, and a third carbide such as tantalum, niobium, or W carbide) [37]. The melting temperature is raised above 3000 K by mixing with refractory carbides, and the cross-section is fairly low. The primary limitation of the advanced carbide concept is lack of data from high-temperature operation.

NEP reactor conditions are somewhat different than those for NTP. The temperatures would only reach up to around 2000 K, but the core may need to operate for much longer time. Due to the lower maximum temperature, some of the limitations of the fuel materials would not be present for an NEP core. With UN, the decomposition reaction would not have a significant effect, making that fuel a better option. Also, the operating temperatures would be below the melting temperature of UC so it would not need to be mixed with a refractory carbide. However, the longer operating times would add additional complications. There may be significant swelling and fission product buildup that could degrade the performance of the reactor.

A mixed mode reactor, that would operate for both the NTP and NEP propulsion modes, would pose the largest problem for the material performance in the core. The NTP modes would have the high temperature requirements and the total operating time would be the combined time of both NTP and NEP modes, meaning that even more swelling and fission product buildup could occur. The benefits to the design would be significant, but the actual design of the reactor materials would be more difficult.

Due to the harsh conditions within the NTP and NEP cores, it is difficult to solely use testing to ensure its performance. Modeling and simulation provide a useful tool for analyzing the thermomechanical behavior of the core materials during reactor operation, and reducing the required number of tests. Advanced simulation tools developed for modeling the fuel performance of power reactors are being taken advantage of to accelerate the development of models for propulsion reactors. The BISON fuel performance code [38] is being used to simulate the temperature throughout NTP cores [39, 40]. The mesoscale MARMOT tool is being used to determine effective properties of the core materials [39] and to investigate specific material behaviors such as fuel loss [31]. Such methods should be applied to investigate the reactor designs for NTP and NEP cores, as well as for mixed mode cores.

5.6 WREN Propulsion Bi-Modal Cycle

The generalized concept for the bi-modal design is shown in Fig.13. This schematic, as well as all following subsection descriptions, directly reflect the layout and function of the aforementioned OOP macro system. The main focus of this work is the improvement in NTP performance. In this regard, there are two main lines of fluid flow. One is the motive gas line that is expelled from the NTP rocket nozzle to produce thrust and the other is the driver gas line as part of a Brayton cycle to supply compression work on the motive gas. For this work we chose to use hydrogen gas for both the motive and Brayton cycle gas. Depending on mission requirements, using different gas species for the motive and Brayton cycle gas may lead to more optimal designs.

The motive gas is compressed by a turbopump and sent to a recuperator, providing the reservoir temperature for cooling the driver gas in the Brayton cycle loop. Afterwards, the motive gas is sent to the NTP core (nuclear reactor) for heating. It then goes to the wave rotor for further heating and then expelled out the nozzle.

The driver gas goes through a Brayton cycle similar to Fig.7, but with an additional compressor to supply the required pressure increase for the NTP Wave Rotor compression process. This places an additional compression load on the Brayton cycle when operating in the NTP mode. A series of divert valves change the flow of the system between NTP and NEP modes. In this concept, the second compressor is bypassed when the NTP Wave Rotor is not in use. Fig.13 does not show the divert valves. Instead, two gas loops are shown to denote the Brayton cycle gas flow during the NTP and NEP modes. The NTP mode loop is denoted by the subscript Bt (Brayton thermal) and the NEP mode loop is denoted by the subscript Be (Brayton electric). It can be seen that the main difference between the two loops is the second compressor and NTP Wave Rotor is bypassed when in NEP mode. Additionally, we note that Radiator 2 is shown as Radiator 2.1 and 2.2. This is to denote how the power is rejected differently between the two modes. A series of control valves couple the radiator to the respective gas loop locations. During NEP mode, heat rejection of the alternator and Power Management and Distribution (PMAD) system is added to the rejection system. Therefore the heat load on Radiator 2 is different between NTP and NEP modes. When in NEP mode, the Alternator is engaged to extract excess power into electricity. This power is conditioned by the PMAD system and sent to the NEP thrusters.

For this concept the wave rotor cycle operates at a temperature significantly above the limits of the in-space turbine technology. The Brayton nuclear reactor core operates at two

temperature points. One at extremely high core temperature, but for a relatively short period during operation of the NTP mode. The second at a more moderate temperature, but still relatively high core temperature compared to other NEP modes. As discussed in Section 5.5, it is not clear if a single core or multi-core reactor is required.

Again in this work, we focus on a practical improvement to the NTP system by maximizing the I_{sp} for a given thrust. This was conducted with out regard to mission requirements except for improving the NEP system mass to power ratio defined as the parameter α in section 5.7. In our work we define alpha as the power of the propulsion cycle divided by all of the systems shown in Fig.13. Compared to a pure NEP mode, this makes our NEP alpha larger.

5.6.1 Propulsion System Macroscale Model

In order to model the system as a whole, an object oriented programming (OOP) MATLAB program was developed. Each component is modeled as an independent object allowing the application of different constraints on the propulsion system for optimization. These object classes were established as either inheriting or being composed of other classes dependent on modeling needs. For example, all of the engine sub-components which contain some sort of gas flow (i.e. wave rotor, turbine, rocket nozzle, etc.) are given the thermochemical qualities of the gas class to determine work processes. The gas class contains 9-coefficient curve fit functions for the specific heat (C_p), enthalpy, and ratio of specific heats (γ) of gas species based on empirical data derived from the NASA Chemical Equilibrium and Applications (CEA) code [41][42]. These curve fits are functions of temperature and are therefore thermally perfect. This is useful for our model because both the driver and motive gas are not chemically reacting except for hydrogen dissociation in the NTP Wave Rotor. All heat and work exchanges are assumed volume-averaged, ideal systems, and the high number of coefficients ensure high result fidelity. The following sections go over the assumptions made for each of the subsystems.

5.6.2 Wave Rotors

As was outlined in section 5.2 and 5.6.1, each wave rotor within this system concept can be modeled using simple macroscopic balances of the volume-averaged thermodynamic properties within the rotor passages. Again in this concept, both the driver gas and compression gas in the wave rotors are hydrogen. The primary motivation for this choice is the lack of fidelity to capture mixing of the motive and driver gas inside the wave rotor using the macroscale model. Note that the shock processes in the wave rotor are initiated by the gas contact discontinuities. The process is inherently driven by inviscid forces, but the viscous diffusion forces of the high gradient regions will produce a small amount of mixing. Additionally, the opening and closing of port doors is not instantaneous. If the open/close process is slow, mixing may also occur. Such analysis requires the use of CFD modeling. These losses are included in wave rotor efficiency parameters found in equations 4 and 5. For this model, all efficiencies were fixed at 80 percent, both for the compression and expansion process.

The mass of the wave rotor is negligible compared to the whole system. However as a demonstration of how tenuous it's mass is compared to its power conversion contribution,

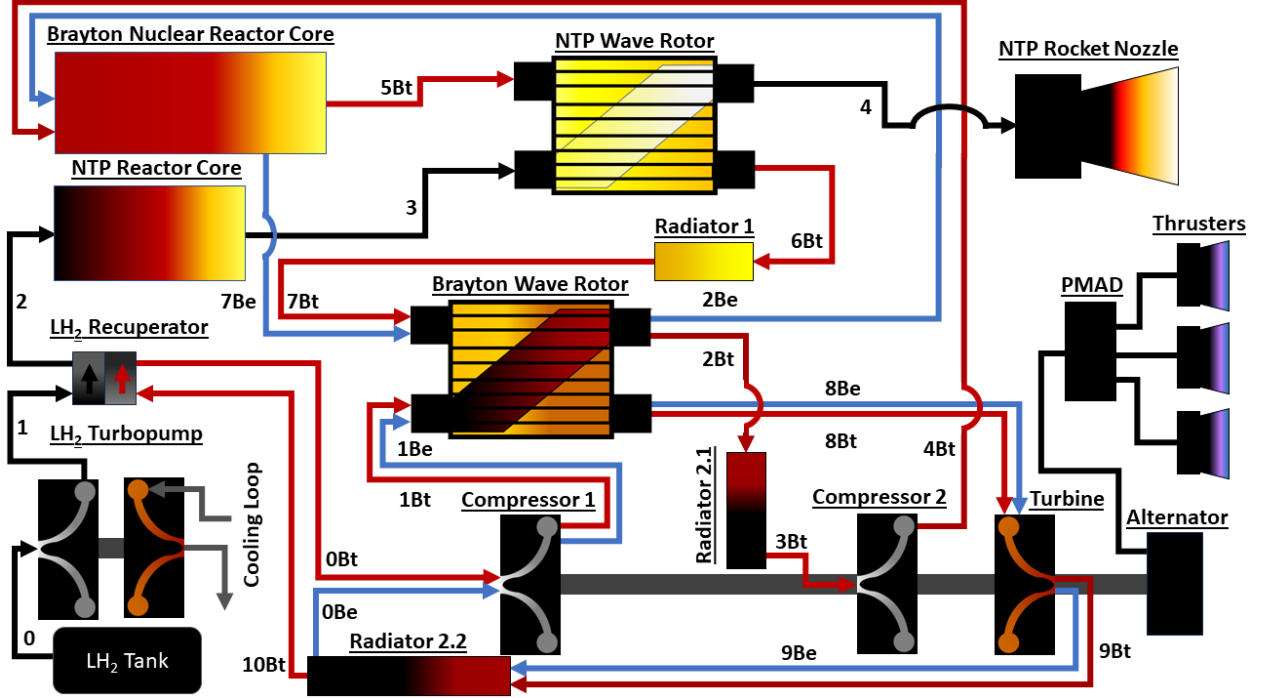


Figure 13: Preliminary system concept for bi-modal NTP/NEP propulsion system with wave rotor topping cycle. Sub-components are colored in visible black body radiation spectrum to give qualitative description of the gas temperature. Different cycle and mode lines are indicated by colored arrows: black for the motive gas cycle during NTP mode; red for Brayton cycle during NTP mode; blue for Brayton cycle during NEP mode.

a simple solid cylinder of high density carbon carbon composite (C/C) was included to represent the rotor, its housing, and motor. Because the wave rotor does not directly apply work to the gas, very little power is required for rotation, and so the motors are quite small. With a minimum density of 1.8 g/cm^3 , the wave rotor's mass would barely constitute a fraction of a percent of the engine mass given current estimated sizes of less than a 0.5 meters in height and diameter.

The choice of high density C/C was for the composite's ability to retain its mechanical properties for temperatures at or above 2800K. Though exposure to hydrogen will require additional protective surface coatings. The wave rotors uniquely unsteady flow environment could theoretically allow for exceptionally high gas temperatures, resulting in a service temperature closer to the exit temperature of the driver gas. This is because walls of the rotor channels are constantly exposed to hot and cold flows in approximately equal periods over a full rotation [43]. In addition to the flow timing analysis, a focus on thermal loads will be necessary in order to determine points of active cooling if passive cooling does not suffice.

5.6.3 LH_2 Turbopump

The work of the LH_2 compressor was found using a scaling parameter for the Space Shuttle main engine (SSME) LH_2 turbopump as conducted in similar NTP studies [4]. Previous work on turbopump designs by NASA was also available [44]. For their NERVA derived design

with a thrust of 111.2 kN, the mass of the turbopump was approximately 100 kg. The power of the pump was functionally coupled with the NTP wave rotor and its compression process (ports 3 to 4). This design makes up for the pressure losses of the motive hydrogen when it passes over the recuperator and the NTP reactor core, allowing the gas to attain the required pressure balance.

The turbine would derive power from a regenerative cooling loop similar to the liquid rocket cycle of the SSME. This line of hydrogen would be used for active cooling of the reactor cores, the NTP rocket nozzle, NTP Wave Rotor, etc. We assume this cooling would be conducted at station 2 after cooling the Brayton gas loop and hence the grey open loop arrows in Fig.13 to simply the description for this work. Details of the cooling lines will also be reserved for future research, as this system overview is only concerned with the engine dry mass. However, future research into a valved tap-off cycle has been considered where the tapped hydrogen could be bypassed from the turbopump into the NTP rocket nozzle to induce some throttleability during the NTP phase.

5.6.4 Recuperator

A recuperator is included to provide additional cooling to the Brayton cycle when operating in NTP mode. The lightweight mass model is adapted from a micro shell-and-tube model for use in turbo-Brayton cryocoolers of air and space vehicles [45]. Although a recuperator is able to operate in the temperature and pressure ranges we introduce for the exchanging lines with a high efficiency, it should be noted that an improved model will be necessary as Zagarola and McCormick [45] utilized a recuperator meant for monatomic noble gases. Regardless, this heat exchanger is assumed to be extremely light; comparable to LH₂ turbopump in mass.

The recuperator will not be in operation during the NEP mode because liquid hydrogen is no longer being fed from the LH₂ tank. For this reason, balancing the cycle gasses surrounding the Brayton Wave Rotor will be met by routing the driver gas through the specific required acreage including both radiator 2.1 and 2.2 to meet cooling needs. This will be done by bypassing station 10Bt and 0Bt surrounding the recuperator. Station 0Be will be the new driver gas line which passes over radiator 2.1 following radiator 2.2 (note: this line switch is not shown in fig.13 for simplicity). Estimates indicate that a little under half of the radiator 2.1 acreage will be devoted to driver gas cooling during NEP mode, the other half will provide cooling to the Alternator and PMAD.

5.6.5 Turbine

The purpose of previously published mass-power scaling factors is motivated by the ease of preliminary estimates of system mass contribution. The change in enthalpy of the driver gas as it passes over the turbine determines the work done given its mass flow rate. As such, an isentropic expansion of a thermally perfect gas model is used to determine the work output of the turbine based on equation 8. The turbine in this model is assumed to have an efficiency of 85% The mass scaling parameter is the average specific mass of turbines outlined in previous space nuclear power systems studies [46]. This parameter was originally published in 1987, thus it can be inferred that our model overestimates the actual weight of

the turbine since turbomachinery technology and materials have improved since then. The same assumption goes for the compressors and alternator. For example SiC/SiC Ceramic Matrix Composites (CMC) have a density that is a third of superalloys[47]. Recent work has also referenced the values from Walter [48][49].

The inlet temperature of a turbine is set to 1650 K based on material limit estimations. Current in-space turbines operate at 1200 K for He-Xe gases and 973 K for CO₂[50]. Work by Duchek[51] suggested a turbine inlet temperature of 1400 K to be a key performance parameter. The in-space Brayton cycle study by Morrison[49] suggested that an turbine inlet temperature of 1800 K would be feasible based on advanced materials and film cooling. NASA Glenn has focused on a 1750 K SiC/SiC Ceramic Matrix Composite (CMC) for aviation turbine use[52], but with a target strength of 20 ksi for only 300 hours. Recent work supported by NASA Glenn [53] showed a consistent cycle life for SiC/SiC CMC blade material between 1500 F (1088 K) and 2400 F (1588 K) as part of their high cycle fatigue test program. This technology used in conjunction with a thermal barrier coating could achieve an operational temperature of 1650 K. Hence we use this value as the upper limit for the WREN turbine technology.

5.6.6 Compressor

Similar to the Turbine, the two compressors are modeled based on an average mass-power scaling [46] and subject a thermally perfect driver gas to isentropic compression, i.e. Eq. 7. The compressors in this model are assumed to have an efficiency of 85%. The purpose of Compressor 1 and Compressor 2 are quite different from one-another despite sharing the main drive shaft. Compressor 2 serves to balance the Brayton loop when it is coupled with the NTP Wave Rotor during NTP mode, while Compressor 1 functions as a typical Brayton cycle compressor for both NTP and NEP mode. During the NEP mode Compressor 2 will be shut off and the driver gas will bypass it.

The mode change from NTP to NEP has introduced some notable challenges in terms of pressure and temperature balances. A primary requirement for the bi-modal system is to maintain the same operating conditions of the Brayton Wave Rotor for both modes. This is because of the extremely limited range of conditions that the wave rotor can properly operate. The state-of-the-art wave rotor technology still uses a fixed port timing fixing port conditions. Therefore, the compressor system is designed having two-stages. It must also be noted that the wave rotor compression process is set by the speeds of the compression and expansion waves inside the channel. Cooler gas temperatures during the NEP mode, may required an adaptive port timing system or the addition of a third wave rotor to handle the different port conditions. We currently ignore such issues in this systems level study. To better understand this issue, detailed computational fluid dynamics modeling of the wave rotor will be required.

5.6.7 Radiator

Radiators are used to reject the excess heat from each of the main systems to space. The radiators are sized based on energy balance approach for rejecting excess heat to the surroundings. Heat inputs and view factors to surrounding components of an environmental

body such as the sun and planets are required to be taken into account in this energy balance approach. For a Earth to Mars mission, the Earth environment during NTP mode operation is considered the worst case scenario. The required radiator area is determined by the thermal power that needs to be rejected. For the WREN propulsion system, two main radiators are employed. Radiator 1 is used to moderate the temperature between the two wave rotors. It serves to limit the temperature of the Brayton cycle gas at the turbine inlet. Radiator 2 is the first stage of heat rejection for the two compressors. Fig. 13 shows a Radiator 2.1 and Radiator 2.2 subsystems. This is to denote the different heat rejection requirements of the two modes and the interface to Radiator 2. During NTP mode, Radiator 2.1 acts as an inter-cooler between the Brayton Wave Rotor and Compressor 2. In NEP mode, Compressor 2 is by-passed and additional heat can be rejected by Radiator 2.2. This radiator is then used to cool the alternator and PMAD systems. Any additional area is given to Radiator 2.1 to allow for reducing Compressor 1 inlet temperature.

We used the same design methodology as for the NASA Compass study of the NEP-CP 1.2 reference vehicle [7]. The design uses a pump loop cooling system due to the high heat loads. The pump loop system is broken into segments and is controlled by a series of valves and sensors. H_2 gas mixing from Radiator 2.1 and 2.2 can then be controlled. Leakage of H_2 during operation was not considered due to the expected short transit times compared to the total H_2 required for NTP motive gas. Future work will focus on these second order effects. It was found that the high speed of sound of H_2 provided improved heat transport properties compared to HeXe gas typically used for the Brayton cycle. A range of expected power loads for the NTP and NEP modes was approximated and used to create three radiator designs. The resulting radiator mass and areas were fit to curves based on the average radiator temperature.

$$m_{rad} = 8.26815 \cdot 10^{13} \cdot T_{avg}^{-4.025} \cdot PW_{reject} \quad (36)$$

$$A_{rad} = 2.756 \cdot 10^{13} \cdot T_{avg}^{-4.024} \cdot PW_{reject} \quad (37)$$

Radiator 1 operates at extremely high temperatures. We assumed that it would be made out of a series of graphite or ceramic tubing. Control of heat rejection would be conducted by a series of rotating insulating covers to control the heat rejection area. Additional low temperature radiator panels are required for the PMAD system for needs beyond the cooling from processing the alternator electrical power. These were considered as included in the mass budget of this subsystem. Other heat rejection requirements for the fuel tanks, crewed module, ect. do not use Radiator 1 or 2 and are included in their respective systems.

There are also potentially lighter mass radiator system technologies such as reported by Tombuilian [54] that is based on carbon fiber fabric. Incorporating such technology could potentially double the specific heat rejection (W/kg) of the individual heat pipes leading to a significant reduction in the radiator mass. For this study we chose the higher Technology Readiness Level (TRL) materials to show the benefits of adding a wave rotor vs. NASA's current baseline systems.

5.6.8 Nuclear Reactor

As described in section 5.5, the optimal design of a nuclear core for use in the WREN system has not been explored. We decided to use an exit temperature of 2850 K for the nuclear core when in NTP mode. This is close to the desire by NASA for a 900 second specific impulse I_{sp} traditional NTP design [55]. Referring to Fig. 12 higher temperatures could be used, but we expect the NTP mode to operate longer due the extra propulsion mass vs. available thrust of the bi-modal system. The NEP reactor exit core requirements will be much lower due to our desire to have the same gas mass flow rate through the Brayton cycle for both modes. This assumption is applied with the desire to maximize the electric power of the NEP mode. Fig. 13 shows two reactor cores. This does not preclude that individual cores are required but to capture the worst case scenario of the different thermal and power load requirements between the needs of the Brayton cycle loop and the heating of the motive gas. The total power requirement of each mode is used to estimate the required reactor mass. To estimate the mass requirements of the core, we relied on NTP core designs with low power [56][57]. These designs resulted in mass to thermal power ratios ranging from 0.006 to 0.01 kg/kW_{th} . For conservatism we chose a value of 0.1 kg/kW_{th} . For the gas flowing through the reactor core, we assumed a pressure loss of 2%.

5.6.9 Nuclear Reactor Shield

Referring to the NEP-CP 1.2 reference vehicle study [7], the design of the nuclear core reactor shield is a complicated process. Modeling and simulation code is used to predict the radiation environment in the shadow of the shield with respect to the nuclear reactor core. For our level of detail we relied on past work [58] to develop a scaling parameter. They suggested a human at most should receive a dose of 5 rem for a mission. A relationship between required shield mass and crew dosage was given for a 75,000 lbf NERVA-derivative engine. This power is in the same class as the WREN system. We decided to lower the dosage requirement to 2 rem. This resulted in a mass to thermal power ratio of 0.008 kg/kW_{th} .

5.6.10 Alternator

The power of an alternator is directly coupled with the output work of a turbine, i.e. a turboalternator. Similar to the turbine and compressors, a mass-power scaling is used (0.2 kg/kW)[46]. The operational efficiency is assumed to be 97% [59][60]. The maximum continuous operating temperature of the alternator for this design is 500 K [61]. For an alternator processing 50 MW of input work, the power rejection requirements would be 1.5 MW. If it operates continuously at 450K, it will require approximately 1600 m^2 of Radiator 2 for cooling.

5.6.11 Power Management and Distribution (PMAD)

After work is extracted from the Brayton turboset by the alternator in NEP mode, the electric power must be correctly conditioned for distribution to the electric thrusters. The largest contribution to mass is the direct current (DC) to DC converter. The Kilowatt project for distributing power on the Moon was found to be one of the most recent works

providing data for actual systems[62]. The largest units produced for extraterrestrial use reach 200 kW. A linear relationship was given to compute the specific power (kW/kg).

$$SP_{PMAD} = 0.0279 \cdot (PW_e \cdot 10^{-3}) + 0.6816; \text{ for } (PW_e \cdot 10^{-3}) \leq 200kW \quad (38)$$

The listed actual systems can handle input power up to 200kW we therefore limit the specific power to this maximum value our multi-megawatt system. It is envisioned multiple converter units are used. We assume these units are designed to high efficiency standards and use a value of 98% [63][64][65][66]. If we assume the spacecraft generates 50 MW for the NEP system approximately 1 MW must be dissipated by Radiator 2 for the PMAD system. We assume that the maximum operating temperature of the system is 250 °C (523 K). This configuration is designed to operate at 200 °C (473 K) using SiC power converters. This results in approximately 900 m² in radiator area when converting 48.5 MW coming from the alternator. Together with the Alternator, this results in 2500 m² in radiator area. A value close to the entire radiator size of the NASA NEP-CP 1.2 spacecraft.

5.6.12 Power Distribution Cable

The power distribution cable is the main cable to send electricity from the power converter in the PMAD to the electric thrusters. We separated this part from the PMAD system due to modeling for the correct voltage based on the electric propulsion thruster used, sizing the wire for self-cooling, and ensuring the transmission efficiency was over 99.2%. For our concept we assumed 200 meters of cabling would be required. This length is based on the estimated size of the radiator constrained by its triangular shape to stay in the shadow of the reactor shield. A relationship for the mass of the cable was found given the line voltage and transmission power specific to our spacecraft design.

$$m_{cable} = 2.349 \cdot 10^{10} \cdot V_{cable}^{-1.984} \cdot (PW_{cable} \cdot 10^{-6}) \quad (39)$$

5.6.13 Electric Propulsion Thrusters

Nuclear electric propulsion (NEP) is a critical subsystem for this mission. A high specific impulse NEP system arranged along with a high thrust delivering NTP system allows for a fuel efficient, faster transit time to be feasible. As such, the NEP subsystem needs to be selected such that one gets the most optimal system performance for a given mission. In regards to Fast-Transit missions to Mars, the most important parameter is to effectively package the electric thruster into a low mass to power ratio. Other parameters to consider in system integration and selection are heat and power, size duty cycle, stability, propellant storage, firing time, life time, and complexity. Additionally, the ability of the thruster to produce the required thrust becomes an important factor. For electric propulsion thrusters, the I_{sp} is related to the thrust by,

$$\frac{T}{PW_e} = \frac{2 \eta_{et}}{I_{sp} g} \quad (40)$$

We see that a high I_{sp} thruster may not provide the required thrust to quickly transit between two planets. Fig. 14 shows the notional operating range of different thruster

technologies. The figure has a red line to denote the designed for NTP I_{sp} of the WREN system (Table 1). Any electric thruster technology to the left of the line could be replaced by the WREN NTP system. The most important parameter becomes the thruster efficiency that can be designed to for each thruster. We see that magnetoplasmadynamic (MPD) thrusters have the ability to provide a high thrust to power ratio. The thrusters can be configured as Self-Field (SF) or Applied Field (AF). SF-MPD technology can generally provide a better thrust per power performance. We see that the SF-MPD is in a similar I_{sp} regime as the WREN NTP system in Fig. 14. Therefore, AF-MPD was explored for this work. With the state-of-the-art thrusters such as MPD thrusters, having a high input power results in high current density causing the cathode meltdown, thus bringing the thruster to the end of its lifetime at an alarming rate [67]. We therefore need to model both the thruster and the required thruster subsystems to keep the thrusters cool. This is true for all thruster technologies, Fig. 14 shows the efficiency of such systems range from 30 to 70 % efficient. Another issue is that most state-of-the-art thruster technology is only scaled to 100s of kW due to limitations of current NEP electrical power generation systems. Specifically, having a high input power results in high current density causing the cathode meltdown due to thermodynamic constraints in dimensional scaling. This results in a "scaled up" thruster having a significantly shorter lifetime [67]. We therefore have to rely on already conducted scaling studies to predict the performance of parameters of the MPD thrusters instead of simple extrapolation. A recent review on key performance parameters for Mega-Watt NEP elements [51] listed a power per thruster target and a threshold value of 1.0 and 0.5 MW_{et} respectively. Additionally, NASA Glenn's VF-1 vacuum chamber can test up to 1 MW MPD thrusters [68]. We therefore limit our per thruster to a 1 to 1.5 MW_{et} value out of practicality. Although Mega-Watt thrusters have been demonstrated, the majority of tests have been relatively short compared to transit times to Mars [69]. Significant research is required to enable use for the high-power requirements for a Mars Fast-Transit as discussed in later sections of this paper. Work by Gilland [70] conducted modeling work for MPD thrusters ranging in I_{sp} from 2,000 to 10,000 seconds. The paper is unique in that the thruster sub-components such as the PMAD and electrical cabling are removed from the power to mass ratio calculation and supplied a value just for the MPD thruster and the required radiator for cooling. The study evaluated thrusters designed for 1, 2.5 and 5 MW_{et} . For a power of 1 MW_{et} and an I_{sp} of 3000 sec, α is listed as 0.5 kg/kW_e. The motive gas studied was hydrogen, but we will assume the scaling parameter closely approximates other fuel types. A recent review paper [71] on nuclear space propulsion, Table 3.3 lists a lithium MPD thruster having an α of 0.5 kg/kW_e with a power rating of 245 kW. We therefor use this value of α for our analysis.

As explained in following sections, our current strategy is limited to a constant acceleration burn for the NEP system with a resulting constant I_{sp} . Other systems such as VASIMR [72] found that a variable I_{sp} provided mass savings over the trajectory burn. We plan to alternatively look at a mix of high and low power thrusters to provide variation in the thrust profile of the NEP burns as our trajectory modeling capabilities are improved.

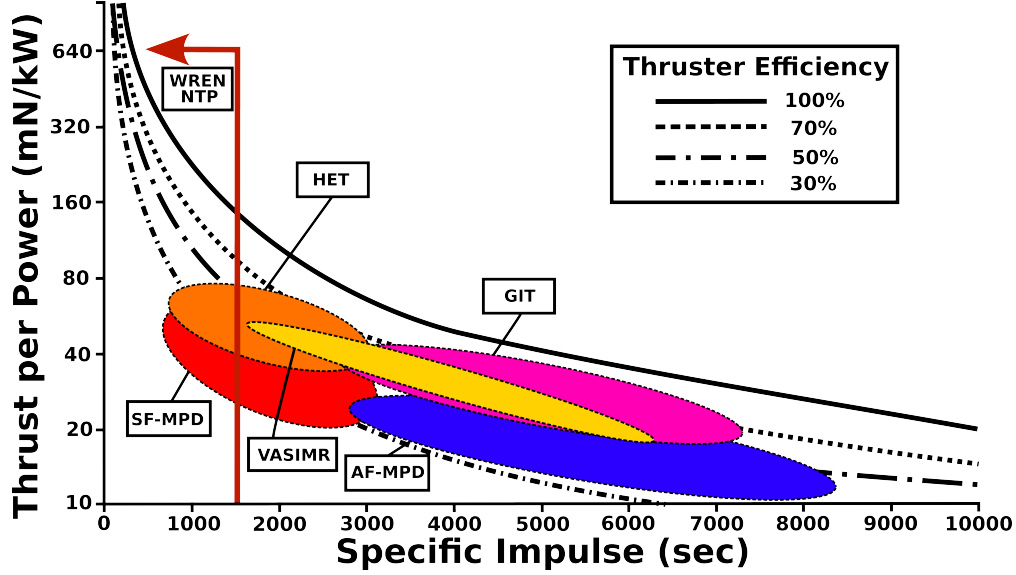


Figure 14: Approximate relationship between I_{sp} and specific thrust for a range of electric thruster technologies. Adapted from [73].

Table 1: Initial Mars mission WREN cycle performance metrics.

NTP Mode		NEP Mode	
Exhaust Temperature(K)	4768	NEP Power (MW)	53.1
Exhaust Velocity (km/s)	14024	Thruster Efficiency (%)	50
Isp (sec)	1430	Isp (sec)	3000
Thrust (kN)	37.3	Thrust (kN)	1.80
NTP Thrust Power (MW)	245	Thruster Power (MW)	26.56
Specific Mass (kg/kWth)	0.44	Specific Mass (kg/kWe)	2.01
Thrust/Weight	0.036	Thrust/Weight	0.002

Table 2: Other important metrics of the initial WREN system.

	Mass (MT)		Area (sq. m)	Mass (MT)
NTP Components	15.24	Radiator 1	44.00	0.132
NEP Components	46.55	Radiator 2.1	5414	16.24
Shared Components	45.13	Radiator 2.2	4542	13.63
Total System	106.9	Total	10000	30.00

Core Power (MW)	NTP Mode	NEP Mode		Mode	Mass Flow Rate (kg/s)
NTP Reactor Core	101.12	Shut Down	Motive Gas	NTP	2.657
Brayton Reactor Core	359.1	205.3	Driver Gas	NTP/NEP	10.90

5.7 Performance of Initial Bi-Modal Concept Design

As an initial design of the propulsion system, we assumed that the I_{sp} of the NTP should be maximized while being within materially manageable, dissociated hydrogen exhaust temperatures. We also specified an NTP thrust above 34.7 kN as this was on the low end of past NTP rocket designs. The radiator acreage was constrained to just below 10,000 m². As a comparison, the NASA NEP-CP Vehicle 1.2 was designed with a radiator on the order of

2,500 m². We assume that a larger radiator, by a factor of 4, could be justified if the transit time and other system masses could be significantly reduced. The compression ratio of the NTP Wave Rotor was set to enable an I_{sp} of 1430 sec for the NTP mode as shown in Fig. 1. The NEP alternator is not engaged ensuring that the power generated by the Brayton cycle goes into the NTP wave rotor for heating the motive gas. During NEP mode, Compressor 2 is disengaged from the turboset and the Brayton cycle gas bypasses the NTP wave rotor. It was found that 53.1 MW of electricity is available at the end of the power distribution cable for the NEP thrusters. The total propulsion mass was just above 100 MT (metric tons) (Table 2). This led to a mass to power ratio of 2.01 kg/kW_e for the NEP propulsion mode. The NEP components make up the greatest contribution to mass as shown in Fig. 15. In the figure, "Other" refers to the LH₂ Turbopump, LH₂ Recuperator, and the mass of both wave rotors. If significant mass savings are sought, improvements to the electric thruster power density are most significant after the radiators. For completeness, a full list of pressures and temperatures of the NTP motive and Brayton cycle gas (both NTP and NEP mode) are listed in the Appendix.

Finally we must note that the radiators operate at high inlet temperatures. A large amount of nuclear reactor energy is then wasted. We plan to explore the use of thermoradiative thermal energy conversion devices for production of additional electrical energy that could be used to recover waste heat from the Brayton cycle [74][75]. The temperature of Radiator 1 is also high enough that high temperature solar cells may be able to be effectively employed to recover the waste heat. We plan to add these subsystems to our modeling as the Matlab OOP program is matured.

5.8 Comparison to Other Rocket System Technology

Fig. 16 shows the performance of many different flown and studied in-space nuclear electric power systems. The x-axis lists the total system electric power provided and the y-axis lists the specific power of the energy generation/propulsion system. We see that the inclusion of a wave rotor to the Brayton allows for the WREN NEP performance to pass what was believed to be the NEP Turbo-Generator Limit. This is mainly due to the ability of the Brayton Wave Rotor to process the gas from the nuclear reactor core at almost 2000 K (see Table 11 in the Appendix). This high temperature along with significantly bringing up the Brayton cycle bottom temperature to 371 K and Radiator 2 inlet temperature of 790 K. This significantly reduces the area requirements of Radiator 2 compared to the colder radiator used for the NASA NEP/CP 1.2 design. Referring back to Fig. 16, only the notional Low- α Fusion concept provides both more total power and specific power capability for a NEP propulsion system.

Comparing both WREN NTP and NEP modes to other propulsion systems, Fig. 17 organizes them according to their vehicle acceleration vs. I_{sp} . Due to the ability of the Brayton cycle to provide a large electric power load to the electric thrusters, the system significantly falls out side of the vehicle acceleration of other Electro- systems. The WREN NTP conversely due to its increase in I_{sp} through a Brayton cycle process, has a significantly lower vehicle acceleration compared to other demonstrated thermal fission technologies. In Fig. 17 the power density is also mapped to the x-y coordinate systems. We see that the WREN NTP and NEP performance parameters line up with expected values between

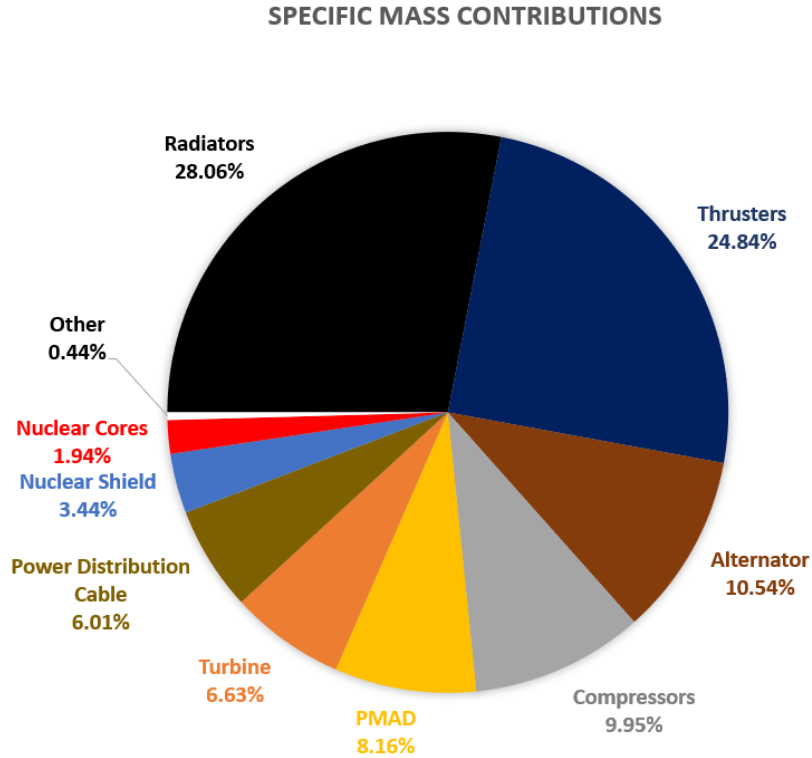


Figure 15: Specific Mass Contributions of the WREN System sub-components.

I_{sp} , vehicle acceleration, and power density. This gives confidence in the verification of the WREN propulsion modeling conducted for this work.

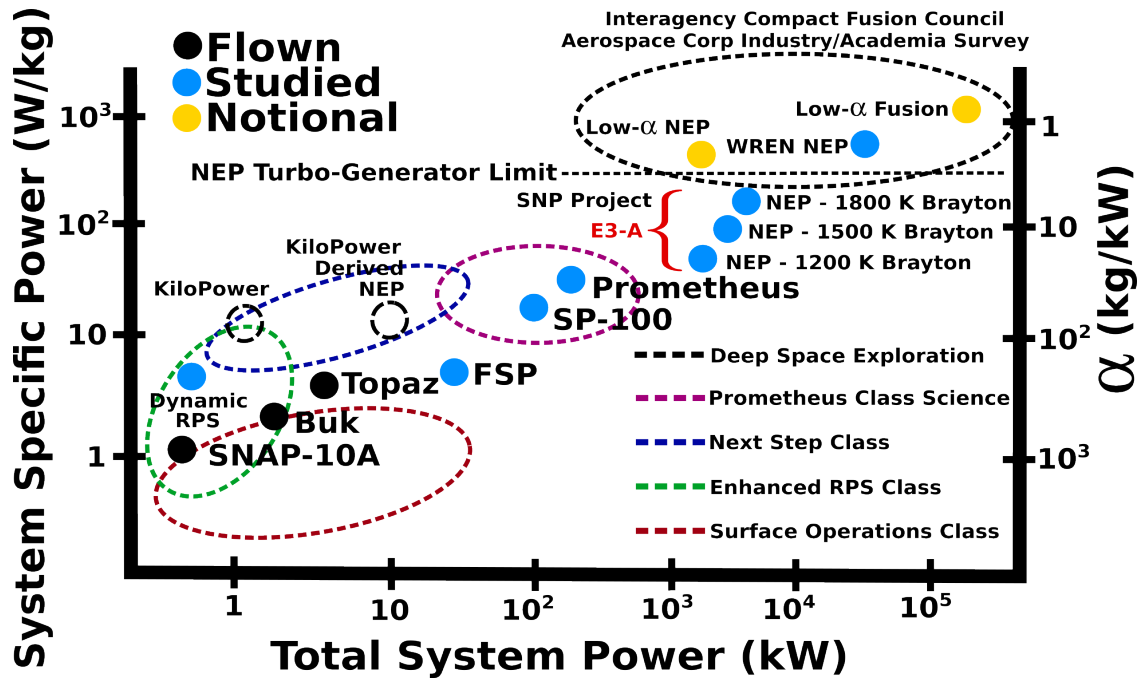


Figure 16: Comparison of WREN NEP mode to other NEP system classes.

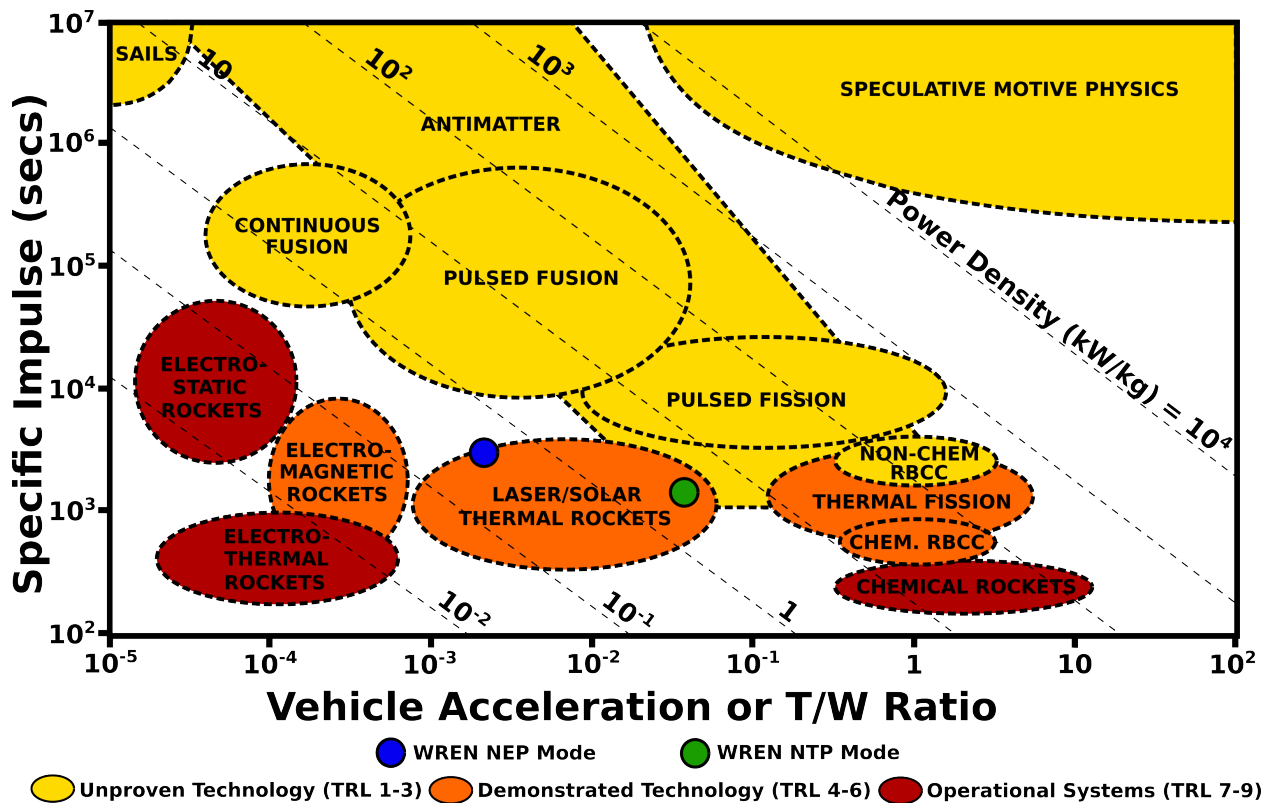


Figure 17: Comparison of WREN NEP mode to other rocket propulsion technologies.

6 Reference Mission

6.1 Spacecraft Overview

The inspiration for the development of this propulsion cycle was to significantly improve the NTP aspect of the bi-modal design. The NTP human mission to Mars report by Borowski[76] computed trajectories for a fast conjunction mission in the 2033 time frame. Table 3 lists the assumptions that they used for the NTP technology. Most impactful is the I_{sp} capability of 906-941 seconds. Compared to traditional chemical rocket propulsion (300-450 seconds) this is a substantial improvement. Their study proposed the NTP technology could reach Mars in a 150-220 day transit time. Additionally, the NASA Space Technology Research Grants Program issued a solicitation topic entitled "Revolutionary Propulsion for Rapid Deep Space Transit." One of the research objectives was developing propulsion systems enabling "Traversing the distance between Earth orbit and Mars orbit in no more than 45 days." [77] Our focus was to see how far we could improve this transit time with the WREN technology with the possibility of achieving the 45 days to Mars demonstration goal.

The WREN propulsion outlined in the previous section was further integrated into a Mars mission concept spacecraft. Fig.18 shows a rendering of the concept spacecraft orbiting Mars. Considerations such as placement of each sub-component, volume of fuel, and size of crew module were developed. Sub-component placements were dictated by the location of the Reactor Shield. Systems sensitive to radiation dosage, were placed in the cone shaped shadow of the reactor shield. This lead to the trapezoidal shape of the radiators and placement of the electric propulsion opposite of the rest of the propulsion system. In this analysis we considered a spacecraft using a current NASA crew habitat design. The habitat consisted of 45 MT of mass and can support a crew of 6 with supplies for 1000 days. The excess supplies are included as an emergency measure in case of propulsion system failure during transit to support potential rescue mission scenarios for the crew. The stored fuel is only for the Earth to Mars transit. A slower, more efficient cargo transport would bring the return fuel necessary for return to Earth. Since we are only concerned in the NASA demonstration of a 45 day transit, analysis of the requirements for the return to Earth leg was not conducted.

Table 3: 2033 NTP human mission to Mars NTP system characteristics.

NTR System Characteristics	
Engine/Fuel Type	SNRE-derived / UC-ZrC "Composite"
Propellant	LH2
Thrust Level	25 klbf ("Pewee-class" baseline; 304 engine cluster on "Core" NTPS
Fuel Element Length	1.32 m
Exhaust Temperature	2790 - 2940 K
Chamber Pressure	~1000 psi
Nozzle Area Ratio	300:1
Isp Range	906 s (2790 K) - 941 s (2940 K)

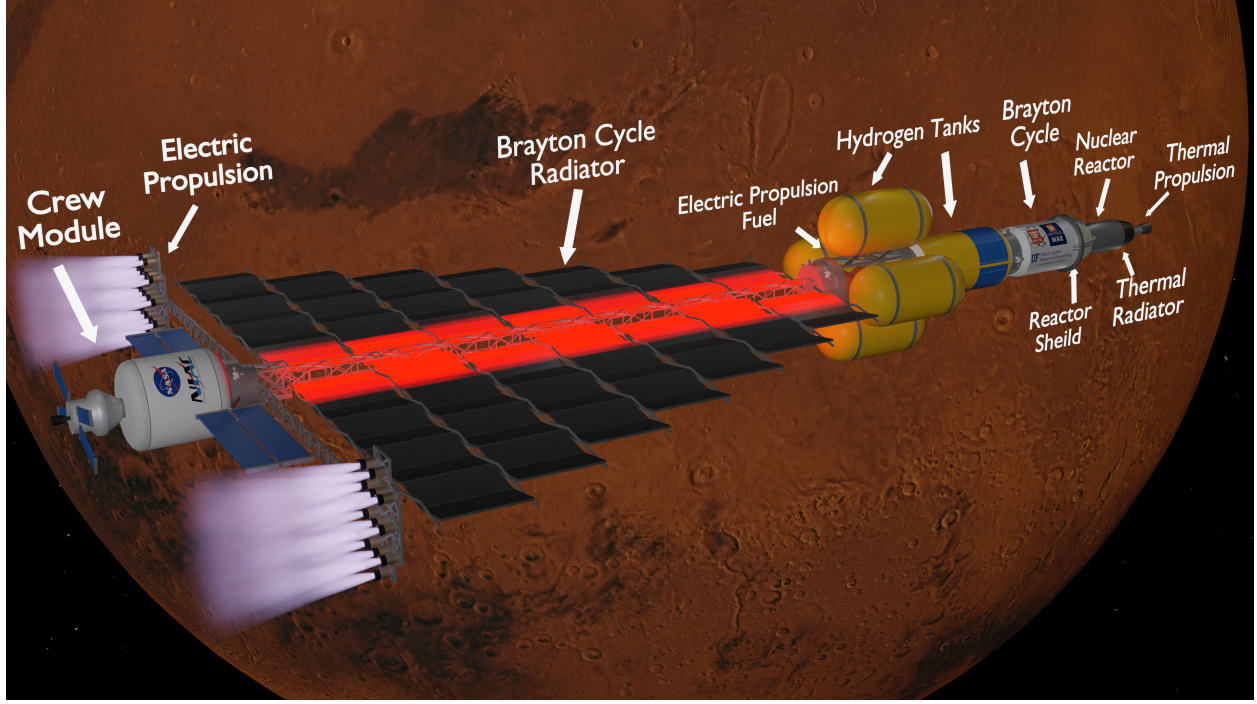


Figure 18: WREN human Mars mission concept spacecraft.

6.2 Fast-Conjunction Class Mars Mission

In this section we focus on the transfer to Mars. For low I_{sp} propulsion systems like chemical propulsion, missions are constrained to using a Hohmann transfer orbit that typically gives the lowest ΔV . Occasionally the planets are aligned such that a fly-by of Venus can reduce this expense even further. Fig.19 shows such a case. These types of maneuvers are considered Opposition class missions. These missions unfortunately consist of long transit times (200 - 400 days). Shorter transit times can be made using larger impulsive burns. This class is referred to as Fast-Conjunction missions. Here the spacecraft leaves when the Earth starts to overtake Mars. The timing sets the offset between the planets as shown in Fig.19.

For the Mars mission, the NTP mode is used to quickly deorbit/orbit a planet due to its relatively large thrust compared to the NEP systems (Table 1). The maneuver is typically assumed to be impulsive in nature and can be approximated by the Tsiolkovsky rocket equation.

$$\Delta V = v_{exit} \ln \frac{m_i}{m_f} = I_{sp} g \ln \frac{m_i}{m_f} \quad (41)$$

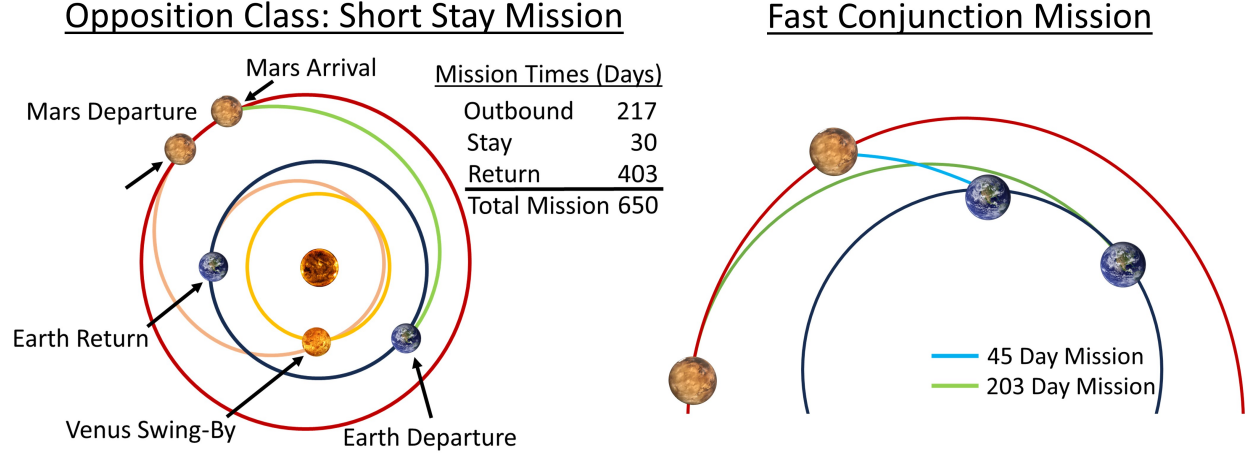


Figure 19: Opposition and Fast-Conjunction Class Mission Trajectories.

The NTP Human Mission to Mars report by Borowski[76] computed trajectories for a fast conjunction mission in the 2033 time frame. They explored transit times ranging from 60-220 days for the outbound Earth to Mars leg. We used the ΔV values from these results to extrapolate the required values for a 45 day mission. For Earth departure a ΔV of 10.4 km/s was estimated to be required and for Mars arrival the value was 11.4 km/s. Comparatively, a 220 day mission only required 3.6 km/s for Earth departure and 1.3 km/s for Mars arrival. Referring to equation 41, if we want to have the same initial spacecraft mass as a chemical rocket ($I_{sp} = 450$ sec) the advanced propulsion system would require an I_{sp} of 2000 sec for the 45 day transit. If we want to have the same initial mass as current NTP technology ($I_{sp} = 900$ sec) the advanced propulsion system would require an I_{sp} of 4000 sec. The NTP mode of the current WREN design is only 1430 sec. Therefore an NTP only WREN system would not provide a practical initial spacecraft mass for the 45 day mission.

We now consider an NEP only system for the Earth to Mars leg. It would require the space vehicle to conduct a spiraling orbital transfer maneuver, which can take many days to complete. As an example, the Variable Specific Impulse Magnetoplasma Rocket (VASIMR) technology is a multi-metawatt advanced nuclear reactor power source. A study was conducted to analyse if the technology would be able to travel to Mars in 39 days.[72] Their concept spacecraft required an approximately 200 MW NEP propulsion system to quickly leave Earth's orbit. Their mission relied on a direct entry of the Mars landing craft that would slow down using an aerobreaking maneuver. The main spacecraft conducted an orbiting maneuver lasting 30 days. The crew module ranged in size from 20 to 30 MT to support a crew of 4 astronauts. Their concept indicated that the NEP system would require a α_{NEP} of 0.5 to 1.0 kg/kW_e depending on the departure orbit from Earth. Referring to the performance specifications of the current WREN configuration, the NEP mode alone can not practically close a mission similar to VASMIR.

In order to approximate fuel requirements, a systems level trajectory model was developed to handle the bi-modal nature of the WREN propulsion system. Due to the low thrust of the NEP subsystem, this analysis requires the inclusion of the acceleration capability of the space vehicle during the use of the NEP subsystem. This parameter can be derived by

inference of α_{NEP} , shown in Eq. 42.

$$\alpha_{NEP} = \frac{2\eta_{et}}{\lambda} \left(\sqrt{1+k} - \sqrt{k + \frac{m_{PL,NEP}}{m_i}} \right)^2 \quad (42)$$

where η_{et} is the efficiency associated with power conversion of the NEP thrusters, λ is the trajectory characteristic parameter, k is the structural coefficient of the space vehicle, and $m_{PL,NEP}$ and m_i are the masses associated with the payload carried by the NEP system and initial state of the vehicle at the start of the operation of the NEP system, respectively. The structural coefficient is defined by

$$k = \frac{m_S}{m_p} \quad (43)$$

where m_S is defined as the tanks structure mass and m_p is the propellant mass.

The mass budget of the spacecraft is defined as a cumulative total of all of the subsystem masses, represented in Eq. 44.

$$m_{i,tot} = m_{PL} + m_p + m_S + m_{PP} \quad (44)$$

For the bi-modal concept, special care must be taken in assigning the proper system masses. In efforts not to modify the governing equation, Eq. 42, the propellant mass is accounted as three separate values.

$$m_p = m_{p,Ed} + m_{p,Ht} + m_{p,Ma} \quad (45)$$

where $m_{p,Ed}$ is the propellant used during Earth departure, $m_{p,Ht}$ is the propellant used during Helio-centric transit, and $m_{p,Ma}$ is the propellant used during Mars arrival. When applying Eq. 42, $m_{p,Ma}$ is included as part of $m_{PL,NEP}$. We also need to add the mass of propellant tanks for the Earth departure burn to m_i if they are not ejected as part of a staging strategy. We use the same value for the tank structural coefficient for both NTP and NEP motive gas tanks. The propulsion system mass contains all the components to enable operation of both modes. Traditionally α_{NEP} is defined as

$$\alpha_{NEP} = \frac{m_{PP}}{P_e} \quad (46)$$

where it is assumed that m_{PP} completely scales with the electric power required for the NEP mode. In our case,

$$m_{PP} = m_{PP,NTP} + m_{PP,NEP} + m_{PP,shared} \quad (47)$$

we have sub-components that are exclusive to the NTP mode ($m_{PP,NTP}$) and the NEP mode ($m_{PP,NEP}$). There are also sub-components that are shared between the modes ($m_{PP,shared}$). For instance, the Brayton cycle components are used by both modes. From a fundamental perspective it is unclear how α_{NEP} defined in Eq. 42 should be treated. For instance, the NTP mode may require an excessively large power from the Brayton cycle compared to the NEP model or vice versa. In our analysis we ignored this issue and assume that the Brayton cycle would be fully utilized by the NEP system providing theoretically maximum electric power for the NEP mode. Referring to Fig. 15 we see that subsystems used only by the NEP

make up a significant mass of the propulsion system. If the power required by the NEP to satisfy the trajectory requirements is in reality lower, the assumptions in Eq. 42 may be too conservative in regards to the ratio of payload mass to initial mass. This completes making Eq. 42 consistent for use in a bi-modal propulsion system for the Fast-Conjunction mission. The final mass of the spacecraft is found by

$$m_f = m_{i,tot} - m_p - m_{S,staged} \quad (48)$$

where $m_{S,staged}$ is the mass of any tanks ejected after a propulsive burn. We ignored this parameter in this work.

In Eq.42, the trajectory characteristic parameter is a non-trivial parameter to compute. It represents the integral of the acceleration squared over the spacecrafts trajectory, given in Eq. 49.

$$\lambda = \int \gamma_{acc}^2 dt \quad (49)$$

An ideal estimate of the ΔV required by the NEP mode is found by integrating the acceleration profile and draws analogy to the classic rocket equation.

$$\Delta V_{ideal} = \int \gamma_{acc} dt = v_{exit} \cdot \ln \left(\frac{m_f}{m_i} \right) \quad (50)$$

We can also link the mass flow rate of the NEP motive gas ($\dot{m}_{et}$) and I_{sp} to the required electrical power from the Brayton cycle alternator (PW_e) using:

$$\frac{1}{2} \dot{m}_{et} \cdot (g \cdot I_{sp,NEP})^2 = \eta_{et} \cdot PW_e \quad (51)$$

Where η_{et} is the efficiency of the electric thrusters and g is the Earth gravity acceleration at sea-level. A variant of the rocket propulsion equation can be formed for power-limited systems.[78]

$$\int_{t_o}^{t_f} \gamma_{acc}^2 dt = 2\eta_{et} \cdot PW_e \cdot \left(\frac{1}{m_f} - \frac{1}{m_i} \right) \quad (52)$$

Using optimization methods,[79] the payload mass fraction can be found using the expression:

$$\frac{m_{PL}}{m_i} = 1 - \frac{k+1}{1 + \left(\frac{2\eta}{\lambda} \right) \left(\frac{PW_e}{m_i} \right)} - \alpha_{NEP} \cdot \left(\frac{PW_e}{m_i} \right) \quad (53)$$

If we assume that the payload to initial mass ratio is independent of the other variables, Eq. 53 is used to produce Eq. 42.

The VASIMR study conducted full trajectory analysis varying the transit time to more exactly compute the ΔV and initial mass requirements for their concept spacecraft. It was found that the trajectory characteristic for their mission was able to be represented as a simple function with only minor uncertainty by,

$$\lambda(t_{trans}) \approx a \cdot t_{trans}^b \quad (54)$$

where $a = 5.22 * 10^7$ and $b = -2.9183$ are coefficients fit to their trajectories and t_{trans} is the transit time measured in days. This allowed for further studies on the VASIMR system using a similar system of equations presented in this section for their NEP burns. We used these values to verify the results of our system of equations with only using the NEP mode of the WREN cycle. For analysing the bi-modal system, we can't use the VASMIR data due to the spiral maneuver conducted by the NEP thrusters at Earth departure. As the desired transit time is shortened from hundreds of days to 39 day, the spiral maneuver significantly contributes to the required trajectory characteristic. We plan to use the NTP mode to quickly leave the Earth-Moon system using an impulsive burn. We therefore just require a representative acceleration profile during the heliocentric phase of the mission trajectory. If we assume that the NEP mode is employed using constant acceleration over the trajectory, Eq. 49 becomes a simple linear equation. We also note that for the Fast-Conjunction mission, Earth and Mars are always fixed distances over a given transit period. Therefore heliocentric phase of Eq. 54 may be correlated through the distance formula.

$$x = v_i \cdot t + \gamma_{acc} t^2 \quad (55)$$

If we include v_i we are able to compute the required acceleration for the heliocentric phase. These assumptions will not led to optimum performance values, but are suitable for understanding the performance of our system without conducting full trajectory calculations. Solving for the trajectory characteristic,

$$\lambda = (x - v_i \cdot t)^2 \cdot t^{-3} \quad (56)$$

The form is relatively close to the VASMIR correlation form. Referring now to the work of Borowski [76] that studies a pure NTP propulsion system, the acceleration is zero giving us a way to solve for x providing a leading coefficient that can be corrected with different initial velocities. It must be noted again this is not finding the correct distance traveled, but a distance that correlates to a required λ given an initial departure velocity. We found that this approximation gives consistent results for transit times less than 65 days. For longer travel times losses due to gravity and the thrust vector become too large. This can be visually seen by the trajectories computed by Dankanich [80] for the VASIMR design. Additionally, the VASIMR work showed that for mission Earth to Mars legs with transit times in excess of 100 days, the NEP spiral orbits become less challenging suggesting a pure Multi-Megawatt NEP system may provide a theoretically simpler solution than a bi-modal NTP/NEP design. Our next steps were to conduct full trajectory calculations for the Earth to Mars leg using the Copernicus software[81]. Results will be compared to this simplified model to understand the uncertainty in our assumptions.

6.3 Earth to Mars Leg

In our work we are not currently focusing on an entire Mars mission, but just the Earth to Mars mission leg to demonstrate the technology capability. This is a similar practice used when initially studying the VASIMR technology. This isolates propulsion system performance metrics removing impacts such as Mars stay times and return leg strategies that complicate the analysis. We consider transporting a crew of 6 with enough supplies for 1000

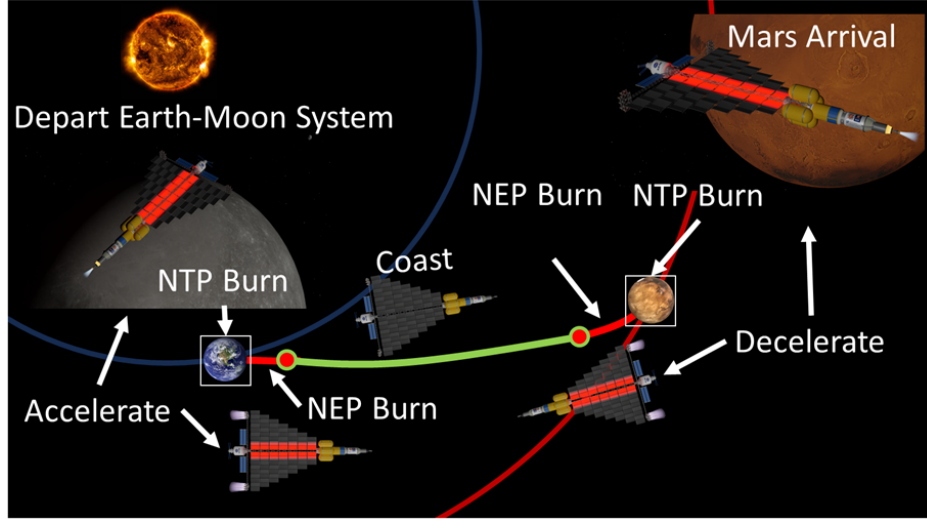


Figure 20: Concept of operations Earth to Mars mission leg.

days. The excess supplies are included as an emergency measure in case of propulsion system failure during transit to support potential abort and rescue scenarios. This habitat is similar to the one being used for NASA baseline mission studies[7]. Fig. 20 shows the concept of operations of the spacecraft. The spacecraft departs Earth from a Near-Rectilinear Halo Orbit (NRHO) after assembly as shown in Fig. 21. As the spacecraft passes Earth, a NTP mode burn is conducted. After the burn maneuver, the NTP mode is turned off and a slow-spin maneuver is conducted over a period of a few hours. Small reaction control jets are used for this low ΔV spin maneuver and considered part of the structural mass. During the maneuver NTP components are cooled and the nuclear reactor is moderated in preparation for the NEP mode. Once the electric thrusters are positioned correctly, the NEP burn starts. Fig. 20 shows a coast period where the NEP mode is turned off. In this work we consider a constant thrust strategy resulting in no coasting period. We include this state of the spacecraft since work on other NEP propulsion systems find a coasting period results in a minimized initial departure mass. Next the spacecraft positions the electric thrusters to start a NEP burn to slow the spacecraft down as it approaches Mars. It then performs another spin maneuver in order to ready the spacecraft for the final NTP burn to orbit Mars. In comparison, the VASMIR mission assumed the crewed landing module would conduct a direct entry into Mars atmosphere while the spacecraft would slow down over a multi-day period to park in Mars' orbit.

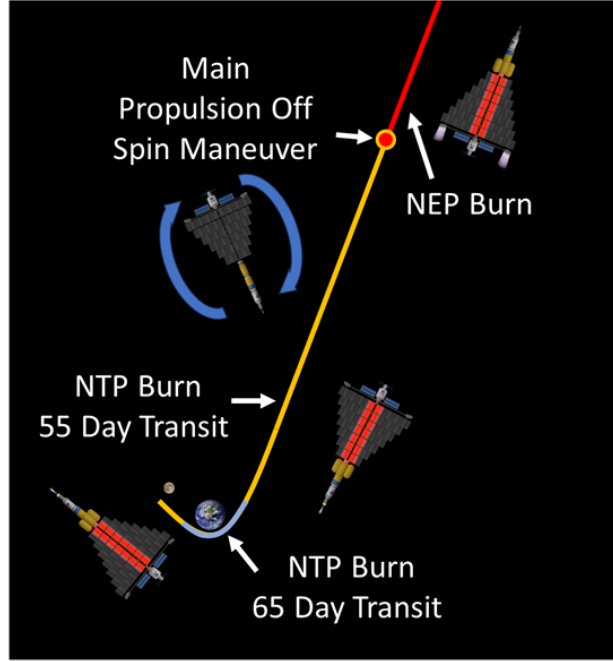


Figure 21: Concept of operations for departure from Earth-Moon system.

Table 4 shows estimates for a notional 45 day transit using the equations outlined in this work. The initial mass of the spacecraft was found to be 530 MT. This is heavier than the NASA NEP-CP Vehicle 1.2 (382 MT). Though the NEP-CP system has a transit time of approximately 382 days. To obtain similar masses, the crewed module would need to be reduced to 4 with a resulting redesigned module to realize mass savings. A pure NTP spacecraft studied by NASA had an initial mass of over 600 MT resulting in a 297 day transit. Exact comparison is hard to make due to the two NASA reference mission spacecraft are designed for a full mission scenario. The main take away is the 45 day transit using WREN propulsion produces a spacecraft in the range of initial mass similar to NASA baseline designs.

Table 4: Summary of spacecraft masses and ΔV burns for 45 Day Earth to Mars leg.

Bi-Modal Trajectory Estimate from Power Limited Propulsion Equation						
Transit Time	Crew Vehicle (MT)	Propulsion (MT)	Payload (MT)			
45	45	100	145			
Traj. Stage	Cycle	Initial Mass (MT)	Final Mass (MT)	Fuel Mass (MT)	Fuel Tank Mass (MT)	ΔV (km/s)
ED Burn	NTP	530	396	134	15	4.1
Transit Burn	NEP	381	257	124	14	14.5
MA Burn	NTP	253	156	97	11	6.8

The main motivation for a fast-transit trajectory is to reduce the risk of human crew. Learning from experiences of astronauts living on the International Space Station (ISS), many health risks arise due to zero-gravity. Additionally, radiation from the Sun and cosmic

space significantly contribute to the total radiation exposure of the crew over a mission. Reducing the transit to 45 days significantly reduces these risks compared to the traditional long duration transits, but how much risk is removed between a 45 day vs. 65 day transit? Fig. 22 shows estimates of initial departure mass vs. transit time. By making the trip just 5 days longer, significant initial mass savings can be achieved. For a 65 day transit the initial mass is almost half. The figure also shows the impact of different NTP mode ΔV burns for Earth departure. To remove our concerns with the rough estimation of the NEP trajectory characteristic, we see that trading NTP vs. NEP burn time is relatively insensitive due to the high I_{sp} of the NTP mode. It should be noted again that if we use a pure NTP burn as in Borowski [76] it is clearly impractical.

Finally a main weakness of the presented performance of the WREN system is the low thrust to weight ratio of the NTP mode. For the initial spacecraft masses and ΔV required for NTP burns, the burn can not be assumed to be instantaneous. Referring to other NTP spacecraft [76] the burn times should be on the order of 30 minutes to 2 hours for impulsive maneuvers. This analysis shows the WREN NTP burn times are on the order of 8 to 15 hours in duration. This has a large impact on the requirements of nuclear core technology where NERVA class systems have never been tested for such duration. Future trajectory analysis will require finite burn modeling of the NTP modes to understand the impact of gravity losses. Fig. 21 presents a notional description of the difference in burn time with respect to the trajectory for a 55 and 65 day Earth departure NTP burn. In a worst case scenario, Earth departure and Mars arrival would have to be conducted using a spiral maneuver. For the 39 day VASIMR mission, it required on the order of 200 MW electric power to quickly leave Earth's orbit for their 39 day Earth to Mars leg. The WREN NTP mode has a thrust power of approximately 200 MW equating to approximately 400 MW electric power of VASIMR is operating at 50% thruster efficiency. Unfortunately the trajectory characteristic reported in the VASIMR studies lumps in the contribution of Earth departure, Heliocentric transfer, and Mars arrival burns. Making it impossible to be used for a bi-modal system trajectory analysis. We therefore plan to look into detailed trajectory modeling to better understand this issue.

To help verify the above results, a simple study of the Earth to Mars mission leg was conducted with the NASA Copernicus software. It should be noted that the NTP/NEP ConOps includes an initial sequence departing from an initial staging point in the NRHO orbit. The leg of the mission from the NRHO to the TMI is not modeled in the analysis; however a NRHO starting mass is estimated based on certain NRHO departure assumptions used for other Copernicus studies. The propulsion system performance parameters are slightly different than results presented Table 1. The propulsion performance inputs are shown in Table 5. Rev. 1 was taken as a snapshot of the current baseline design when the Copernicus study was initiated. The α for the NEP was slightly higher.

The optimized results from Copernicus returned different ED ΔV values than the simplified model derived for the majority of this study. The constraints of the Copernicus input deck produced a burn-coast-burn maneuver for the NEP cycle compared to the assumed continuous burn used by the simplified model. This resulted in a larger NTP burn for the Copernicus results. Fig.23 shows that this did not result in a large difference in results in regards to initial Earth departure mass. It may be that the high I_{sp} of the NTP is fortuitous in producing similar results. We will continue to study how best to come up with simplified

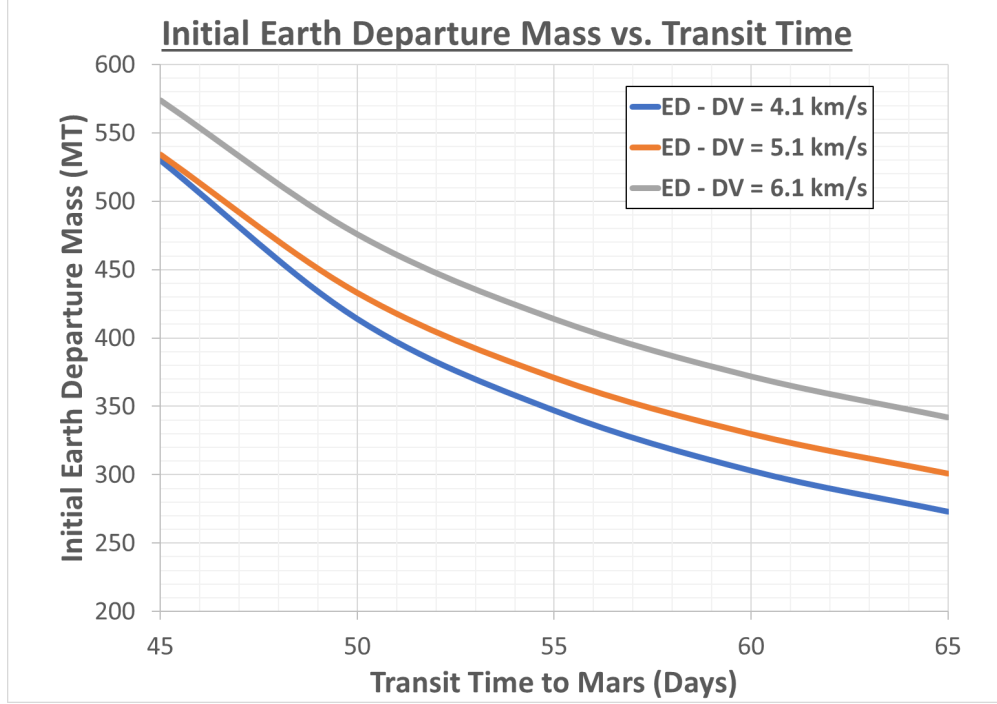


Figure 22: Initial Earth Departure (ED) Mass vs. Transit time for different Earth departure burns of the NTP mode.

mission leg analysis.

Table 5: Input parameters used for Copernicus study.

Rev 1: delivered mass to Mars = 146 MT

NTP Cycle		NEP Cycle	
Isp (sec)	1430	Isp (sec)	3000
Thrust (kN)	34.7	Thrust (kN)	1.57
Alpha (kg/kWth)	0.038	Alpha (kg/kWe)	2.25

For transit times approaching 45 days, the similar results coming from Copernicus indicate that the time for the Earth departure burn is too large to assume an impulsive burn. Therefore, future work will require tightly coupling the optimization of mission trajectory with the optimization of the propulsion system. We have been discussing with NASA Glenn on the requirements for conducting such a coupled analysis.

Ultimately, these results may be indicating that the NTP I_{sp} and the NEP thrust power maybe too large for this mission. In regards to NTP burns, a similar phenomena was found with the LO_2 -Augmented Nuclear Thermal Rocket (LANTR)[82]. In this concept additional LO_2 was added in an after-burner to enhance the thrust performance of the NTP rocket for operation in the Earth-Moon system. In the case of the WREN system, the compression ratio of the NTP Wave Rotor can be designed to mission requirements trading I_{sp} for thrust. Referring to Fig. 15 the NEP only subsystem components make up a significant contribution

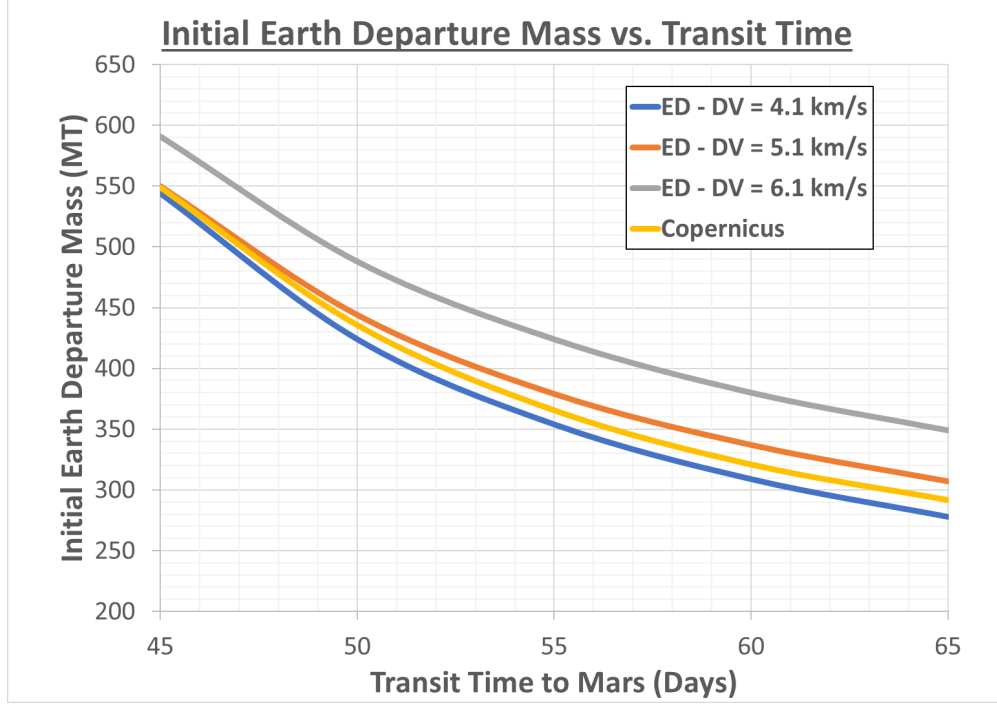


Figure 23: Initial Earth Departure (ED) Mass vs. Transit time for different Earth departure burns of the NTP mode compared to optimized Copernicus simulation.

to the propulsion mass. Full trajectory analysis is needed to explore if such a high NEP thruster power is required. Fig. 22 shows that trading between Earth departure ΔV (NTP) and required NEP acceleration provide similar results as we reach a 45 day transit time. Removing NEP propulsion mass will help with the NTP Earth departure burn and may offset the losses of less power available during the heliocentric phase. Again detailed trajectory modeling is required to better understand these issues.

7 Application to Other Missions

7.1 Sensitivity to TRL

The standard method by which a novel vehicle is deemed feasible in early systems engineering design is its individual subsystem technology readiness level (TRL). The TRL is a NASA rating system which quantifies the maturity of a technology, and ranges from basic observation of natural phenomenon to flight-proven, mission ready hardware. Figure 24 features the full TRL chart and each level's description. This section will address several low-TRL aspects of the WREN system and develop a basis for performance feasibility compared to other available, studied, or theorized technologies. A simple numerical analysis was performed using the OOP macroscopic systems model to understand potential system performance over a range of subsystem TRL levels.

Another attractive feature of the WREN system is driven by the fact that lesser advanced technologies, and therefore higher TRL systems, could be implemented to the Brayton power

Technology Readiness Level	
9	Actual system “flight proven” through successful mission operations
8	Actual system completed and “flight qualified” through test and demonstration (ground or space)
7	System prototype demonstration in a space environment
6	System/subsystem model or prototype demonstration in a relevant environment (ground or space)
5	Component and/or breadboard validation in relevant environment
4	Component and/or breadboard validation in laboratory environment
3	Analytical and experimental critical function and/or characteristic proof-of-concept
2	Technology concept and/or application formulated
1	Basic principles observed and reported

Figure 24: Technology Readiness Level definitions adopted by NASA

cycle design of a Martian transit vehicle. This may expedite maturation from systems concept to prototyping within the short term. The wave rotor itself has shown benefits of extracting high cycle power while allowing the turbine inlet temperature of a Brayton cycle to remain at flight-rated temperatures [23], for example. Additionally, a comparison can be made with middle and lower TRL studied and notional nuclear propulsive systems to show performance can be met or exceeded against proof-of-concept systems.

In order to conduct the analyses of the following subsections, a set of constraints was established to avoid inconsistent, imaginary, or unwanted characteristic values. These constraints are detailed in table 6. Focus was set primarily on the thermodynamic management of power imbalance within specific subsystems. For example, excess power on the turboshaft during the NTP mode was to be kept to close as zero as possible within the code fidelity. This value would unlikely result in exactly zero, and a negative value indicates losses on the shaft, hence an established constraint range. A slightly positive value could be used as a safety margin of power generation or as additional power for electronics and running the wave rotor motors. This particular constraint was controlled via changes to the bottom temperatures of the radiators and LH₂ recuperator, which of course had direct consequences in terms of mass gain if these bottom temperatures had to be decreased. This bottom temperature control was also used to minimize the excess power difference across the recuperator (sign indicates power flow direction). These two thermodynamically driven constraints were found to be quite effective in determining stable solutions of the macro model for demonstration of TRL parameter behavior. Some of these constraints were changed for the sake of their own individual analysis as well, see the “Analysis Changes” column in table 6. All other parameters are unchanged from the base system in section 5.7.

Table 6: General TRL Analysis Constraints

Analysis Constraints			
Description	Constraint	Controlling Parameters	Analysis Changes
Excess Power of Turboshaft on NTP mode	(0, 0.1) kW	Bottom Temperature of the Radiator and LH_2 Recuperator	N/A
Power difference across LH_2 Recuperator	(-0.05, 0.05) kW	Bottom Temperature of the Radiator and LH_2 Recuperator	N/A
Total Radiator Area	10,000 m^2	NTP Thrust after previous two constraints are met	Radiator Area as multiple of 2500 m^2
NTP Specific Impulse	1430s	NTP Wave Rotor Pressure Ratio	Isp between 1300s and 1600s
Turbine Inlet Temperature	1650K	N/A	Turbine inlet between 1200K and 1650K
Mixed Driver Gas Molar Ratio (H_2/Xe)	1/0	N/A	Gas Mixture Analysis between 1/0 and 0.4/0.6
Outlet Temperature of both Nuclear Cores	2850K	N/A	Core Outlet between 2612K and 2850K
NEP Power Power	53.1 MW	Alternator, Power Distribution Cable, and PMAD size constrained to base system	Unconstrained for all analyses besides specific impulse analysis

7.1.1 Turbine Operating Temperature

As mentioned previously, wave rotors allow for a thermodynamic cycle to extract more power while simultaneously keeping the working fluid below or at maximum rated Material Operating Temperature (MOT). Fig. 16 shows how a higher MOT of a Brayton-type NEP is required to produce higher total cycle power. The three NEP-CP MOT systems listed, which are in the "E3-G1" cluster in the upper right of the figure, were analyzed by the SNP (Space Nuclear Propulsion) Project Office at three temperatures of interest. 1200 K is a common and materially manageable MOT for gas turbines, requiring no direct cooling. This temperature is then considered a high TRL of 7 to 9 for space vehicle Brayton power units. NASA Glenn work on SiC/SiC blades justifies a TRL of at least 4 for 1500 K. While temperature ranges of 1650 to 1800 K MOT would be considered TRL 1-3 for application in space systems. Figure 25 shows a history of MOT progress of both material operating temperatures and commercially produced engine MOTs. Notice the required cooling gap which forms between the material limits, which are primarily super-alloy materials, and the commercial engines which operate above 1300 K. While liquid and air cooling for turbine blades has vastly improved for commercial jets, the ability for a Brayton unit to cool its turbine in space is severely limited. Without the luxury of open-air cooling to minimize the weight and complexity of the engine, an NEP Brayton system requires either a cooling loop which can exchange the turbine heat to its radiators or an advanced non-brittle ceramic or composite which can withstand high temperatures [34]. The latter option is considered more desirable as it will heavily reduce weight and complexity to the system, and will ensure maximum efficiency of the turbine. Fig. 25 also indicates that turbine material operating temperatures increases around 5 K on average each year and commercial engine turbines with cooling systems increase by around 19 K on average [83]. This shows a steady progression in terms of both material robustness and cooling unit sizing. Projections of refractory alloys

could see above 1500 K MOTs by the 2030s.

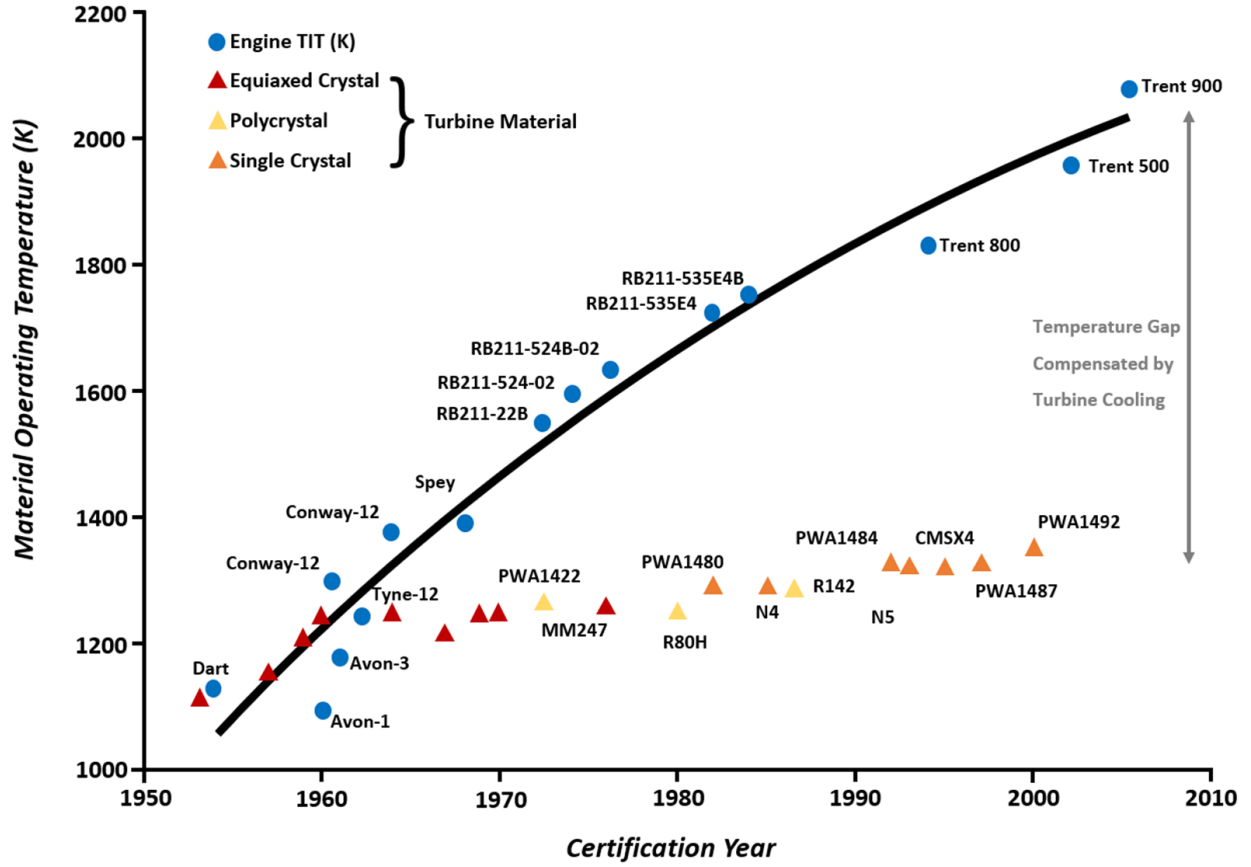


Figure 25: History of Material Operating Temperature for commercial jet engines and operating temperatures of crystalline alloy turbine materials. The black line indicates the trend in MOT. Adapted from [83].

Table 7: WREN performance based on turbine inlet temperature

Turbine Inlet Temperature (s)	1200 K	1500 K	1650 K
NEP Power (MW)	16.3	37.2	53.1
NEP Specific Mass (kg/kWe)	3.33	2.29	2.01
NEP Thrust (kN)	0.56	1.27	1.80
NEP Thrust/Weight	0.001	0.002	0.002
NEP Mode Brayton Core Outlet Temperature (K)	1430	1788	1966
NTP Thrust Power (MW)	101	187	245
NTP Thrust (kN)	15.3	28.5	37.3
NTP Thrust/Weight	0.029	0.034	0.036
Total Dry System Mass (MT)	54.48	84.36	106.9
Motive Gas Mass Flow Rate (kg/s)	1.091	2.031	2.657
Driver Gas Mass Flow Rate (kg/s)	4.48	8.34	10.90

Table 7 shows the performance of the WREN system given these temperatures as a changing constraint. The results indicate that all turbine inlets will produce electric power for the NEP mode in the tens of MWs. This is above the range of theoretical performance of all SNP Project systems (Fig. 16), including the 1800 K Brayton. Additionally, the specific mass of the full, dry-mass of the engine does not go above 3.3 kg/kW_e. This places the WREN NEP system above the theorized NEP Turbo-Generator Limit and into the range of deep space exploration requirements outlined by the Interagency Compact Fusion Council [6] (see Fig. 16). The reason for this low of a specific mass is likely due to the high energy output of both reactor cores, where both operate with a gas outlet temperature of 2850 K during NTP mode and the Brayton core operating at 1430 to 1966 K in NEP mode. Because of this relatively high performance regardless of turbine system TRL, the NEP mode is considered "insensitive" to the MOT TRL compared to other NEP system classes.

In order to give a full picture of system behaviour based on MOT TRL, an analysis was conducted in reference to the WREN base-system from section 5.7. The outlined system to compare to is given in tables 1 and 2. Normalizing all system performance metrics to this reference will visualize TRL sensitivity as a function of the chosen parameter. In the case of Fig. 26, this parameter is MOT for this section. If it is appropriate, this same visualization will be done for other sections. Function resolution is done case-by-case depending on available computation time. All metrics converge to 1 at the base system (1650 K turbine). In this we can see at what percentage and rate the parameters change as the turbine temperature is decreased to a higher TRL.

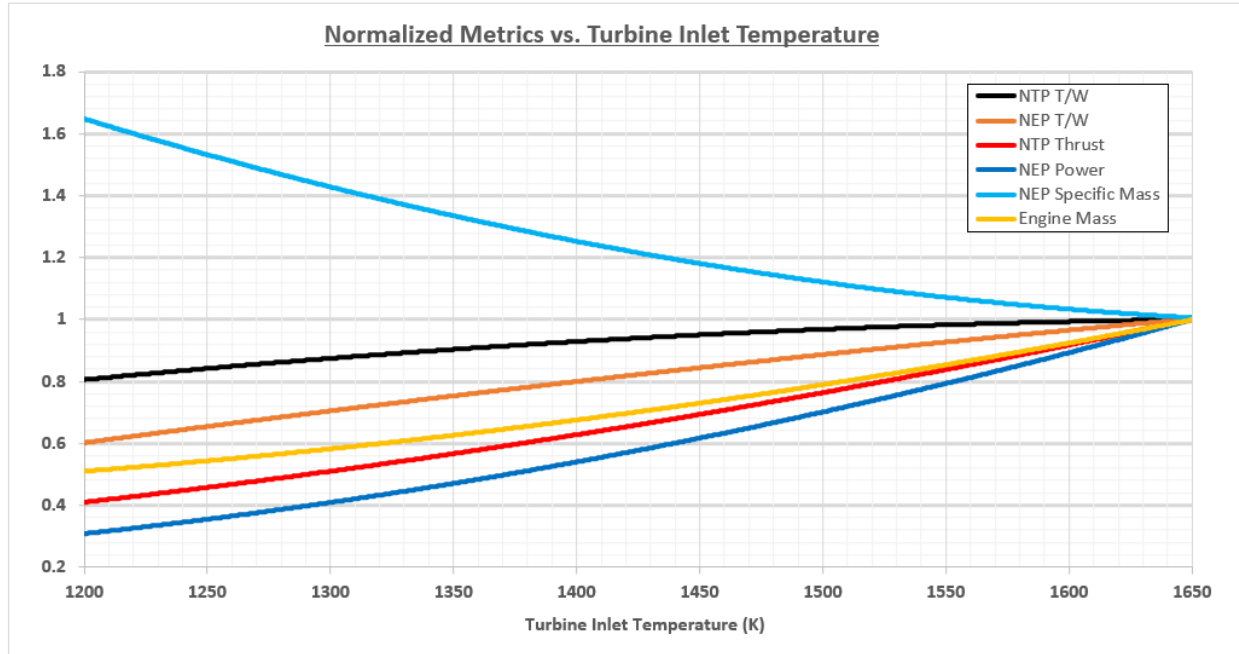


Figure 26: Normalized WREN performance metrics as a function of MOT. Normalized with respect to the base WREN design at 1650K inlet.

7.1.2 NTP Specific Impulse

The two contentious parameters that prevent the WREN system NTP mode I_{sp} from reaching a high TRL operationally is the limitations in material thermal characteristics and the necessary T/W (Thrust-to-Weight) ratio for a timely Earth departure maneuver. Figure 17 shows the approximate technological performances of different propulsion systems and the estimated locations of WREN NEP and NTP modes. The high TRL items (7-9) tend to hug the perimeter of low specific impulse or low thrust. Noticeably, the WREN NTP falls below most typical chemical rockets in terms of T/W ratio, but shows drastic improvements in I_{sp} ; above other demonstrated thermal fission technology of TRL 4-6.

Earlier T/W results in section 5.7 place the dry WREN NTP around 0.03-0.04. For comparison, the T/W of an empty Apollo S-IVB upper stage is two orders of magnitude above the dry WREN system (approximately 3.5; drops to 0.8 with gross initial mass) [84]. This ratio was necessary to get Apollo astronauts through the Van Allen radiation belts in as little time as possible starting from low Earth orbit. Assuming the WREN system would start its mission from NRHO, this high of T/W ratio is not necessarily required to ensure shortened flight time within the belts during an Earth gravitational assist, however there is still a motivation to maximize it for a faster trajectory profile.

We are constrained to using Hydrogen motive gas using the assumption of maximizing I_{sp} potential. We are also power limited to the added Brayton cycle used with a wave rotor to increase the propellant I_{sp} . We can therefore either increase the mass flow rate of the propellant or reduce the Brayton cycle power. The Brayton cycle power reduction for a given propellant flow rate will lower the sub-component masses. We therefore indirectly trade the ratio of the Brayton reactor core to NTP reactor core power requirements. The additional constraint of having the radiators be below $10,000\ m^2$ ensures comparison to the WREN baseline configuration.

Given the constraints in table 6, the following results in table 8 were calculated for changing NTP I_{sp} . A normalized set of system behavioural curves was also generated, see figure 27. As predicted, when I_{sp} is increased via NTP compression ratio, the T/W of the NTP mode drops growth in NTP subsystems. When specific impulse is decreased, T/W exceeds laser/solar thermal rockets performance (see fig. 17), however the specific mass of the NEP is only impacted by a relatively small amount. This also has the consequence of requiring both reactor cores to increase power output, especially in the case of the Brayton Reactor Core. This parameter variation shows that is a decent trade space to consider when developing mission.

7.1.3 Driver Gas Species

The gas species which has been used for each section thus far has been molecular Hydrogen for both the motive gas as well as the driver gas of the NEP Brayton cycle. This was explored as a worst case scenario if the NTP wave rotor would require both driver and motive gas to ensure high I_{sp} . Using another gas for the Brayton cycle small, amounts of mixing may occur in the NTP wave rotor. This would have potential to significantly reduce the I_{sp} of the NTP system. Conversely if a different gas is used for the Brayton cycle, hydrogen may mix with the driver gas. How detrimental would that be? Using pure hydrogen for the Brayton

Table 8: WREN performance based on I_{sp}

NTP Specific Impulse (s)	1300	1400	1500	1600
NEP Specific Mass (kg/kWe)	2.34	2.05	1.96	1.92
NEP Thrust/Weight	0.001	0.002	0.002	0.002
NTP Thrust Power (MW)	482	270	207	175
NTP Thrust (kN)	81.8	42.0	29.8	23.5
NTP Thrust/Weight	0.07	0.04	0.03	0.02
NTP Exhaust Temperature (K)	3941	4570	5246	5969
Total System Dry Mass (MT)	124	109	104	102
Motive Gas Mass Flow Rate (kg/s)	6.4	3.1	2.0	1.5
Driver Gas Mass Flow Rate (kg/s)	17	12	9.9	9.2
NTP Reactor Core Power (MW)	239	116	77.5	57.8
Max Brayton Reactor Core Power (MW)	552	378	331	309

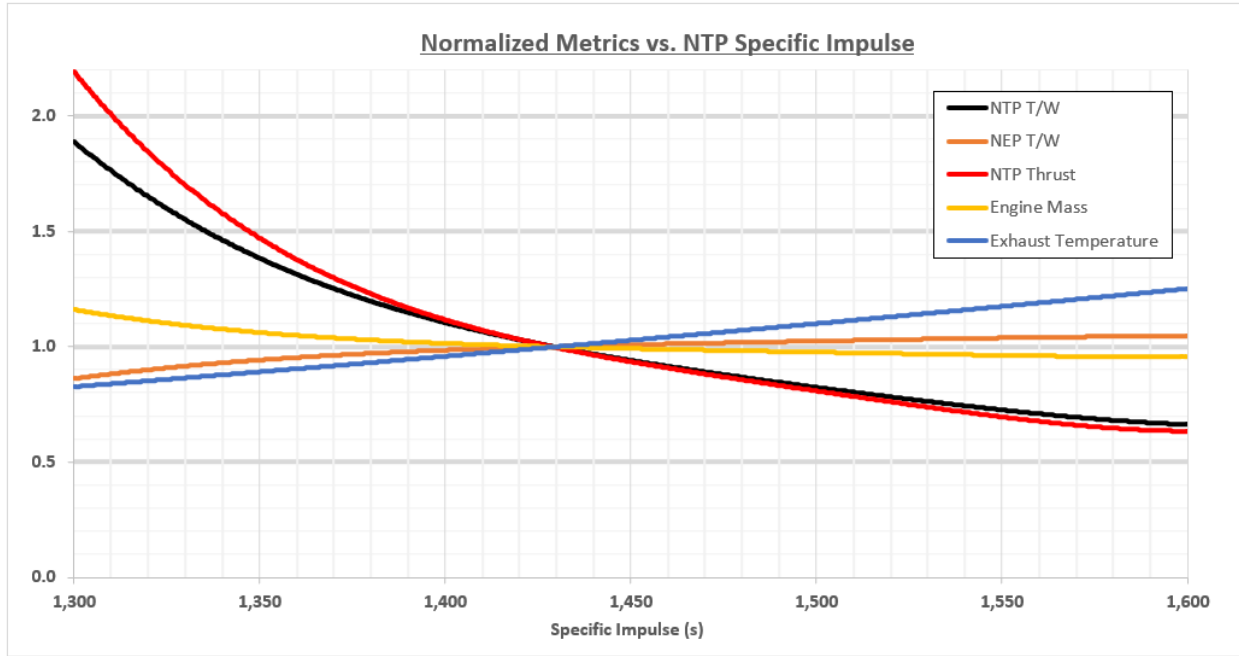


Figure 27: Normalized WREN performance metrics as a function of NTP I_{sp} . The NEP specific mass is not included because it follows the exact curve as total engine mass (yellow). NEP Power is constrained and is also not included.

cycle should expectantly give a worst case scenario. For the baseline hydrogen system we also assumed that leakage would be relatively small and would incur a small excess fuel mass penalty. Such considerations were beyond our ability to study under this Phase I effort.

In order to bound the problem, we also looked at an ideal system using a mixture of gases for the Brayton cycle. For the presented analysis we decided on using an H_2 -Xe mixture, starting from a molar fraction of 1:0 up to 0.4:0.6. Going up to a ratio of 0:1 was not possible as anything above 60 percent Xenon resulted in a number of imaginary system values, such as negative radiator size. Additionally too much Xenon could result in too much neutron

absorption, increasing the difficulty of controlling the Brayton reactor core. Similar studies have been done with He-Xe mixtures have already been studied in nuclear Brayton power conversion systems [85]. Additionally, monoatomic noble gases are easy to model as they are generally non-reactive at high temperatures, even Hydrogen cannot stably bond with Xenon as it is not electronegative enough.

CEA curve fit functions for the enthalpy of Xenon and other gases were also adapted from Gordon et al. [41] and a new subclass was generated for the OOP which mixed simple gas thermochemical properties by molar fraction. For enthalpy this was done with a simple molar average of mixed fluids. For ratio of specific heats, γ , this was done with a weighted harmonic mean formulation shown in the following equation:

$$\frac{1}{\gamma_m - 1} = \sum_{i=1}^n \frac{\phi_i}{\gamma_i - 1} \quad (57)$$

where ϕ is the molar ratio of each species in the fluid, and thus for an H₂-Xe mixture $n = 2$.

Figure 28 shows the normalized metric results as a function of molar fraction of Xe (0 to 60 percent) from our base WREN system. The results show a clear advantage of introducing Xenon to the driver gas. NTP thrust and NEP power increase with increasing Xenon ratio at a rate faster than the engine mass, providing a T/W ratio for both modes that is greater than the base design. This greater extraction of power is because the addition of Xenon increases the Prandtl number of the driver gas. This improves the momentum transfer of the fluid as it passes over the turbomachinery. However, because Xenon has a tendency to absorb neutrons, it is known that Xenon should typically not exceed around a 0.3 molar fraction as it will often force a core shutdown [86].

While this is a good sign of high TRL for a fraction of ≤ 0.3 Xe, research done by Mason [86], Mock and Davis [85], and many others typically used a He-Xe mixture rather than H₂-Xe. This mixture may function slightly different thermodynamically, thus more work is needed to determine the feasibility of this mixture. Additionally, looking at a H₂-He-Xe mixture may prove beneficial due to the potential 0.3 Xe mole fraction limit.

7.1.4 Core Outlet Temperature

Nuclear propulsion derives its power from the thermal energy produced by the critical decay of a fissile fuels. A propellant or working fluid which passes over it extracts this energy and can use it to perform work. While solid reactor cores provide virtually unlimited energy vs. chemical propulsion, the rate of extraction is however difficult to increase as it is limited by core fuel geometry and material qualities. Reactor cores are therefore referred to as "power limited" systems. Fortunately, a large number of core geometries have been designed which maximize the amount of heat conduction surface area to the working fluid without structural or volumetric losses to spacecraft. The difficulty in power increase then arises in the material capabilities of the fuel elements themselves. These elements are typically composed of longitudinal tubes of a nuclear composite, a moderator, and a filler [57]. As the power output is increased by limiting the size of the moderators, the temperature can go beyond the uranium ceramic's melting point and make the fuel susceptible to viscous erosion. This therefore imposes a tight coupling to reactor design, power requirements, and Brayton

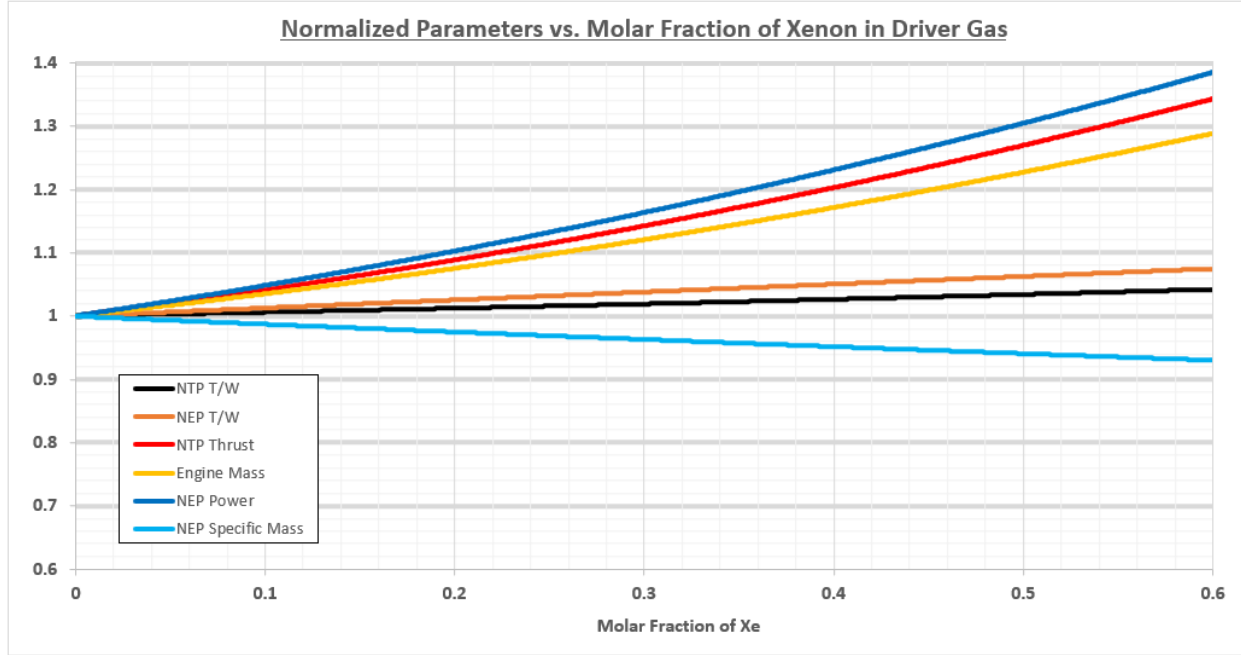


Figure 28: Normalized WREN performance metrics as a function of Xenon Molar Ratio.

cycle gas mixtures. It is therefore necessary to include detailed reactor design modeling with the Brayton cycle gas design space trade.

Figure 12 shows the projected maximum endurance time of a reactor at a given power output of Hydrogen propellant, expressed as temperature of the outlet flow. Several core material types are featured including predicted advanced carbide ceramics. NERVA-Graphite and both US Gen I CERMET and Carbide cores have been tested up to around 2 hours, all data above that burn time is extrapolated. This places the TRL for these longer burns around 1 to 3. The other predicted carbides would be of a lower TRL and highlights the importance into reactor material development research.

Both nuclear reactors for the WREN macro model are set to produce an outlet temperature of 2850 K in the NTP mode. If the Hydrogen motive and driver gas were run through a US Gen I carbide reactor core, then this would result in theoretical max burn time less than those predicted in section 6.3. During the NEP mode the Brayton reactor drops its outlet temperature, see table 7, and can achieve a much longer theoretical burn time; nearly indefinite if the extrapolated values from Fig. 12 can be trusted with high certainty. Again research in this area is greatly needed.

For TRL analysis, the core power output was brought down to an outlet temperature that greatly increases NTP burn time and places it left of the Fig. 12 curve projections. The lowest NTP mode temperature chosen was 2612 K based on current system constraints, any lower and the macro model produces imaginary system numbers. Results indicate that very minimal change occurs in the system metrics as core outlet temperature is lowered from the base system. Metrics associated with the NEP mode performance and the engine mass fluctuate by less than 1 percent, but the NTP metrics such as I_{sp} , T/W, and thrust all reduce by around 5 percent. This is to be expected because regardless of NTP mode

outlet temperature, the NEP mode is still maintained at a much lower power output. While performance is slightly lacking for the NTP system, this is indicative of a general insensitivity to TRL. However, because these high endurance projections have yet to be tested at large time scales, TRL still remains low.

7.1.5 Radiator Acreage

As stated by the NASA Compass team, the maximum radiator area which can be fit into a single 8.6 m diameter SLS fairing is approximately 2500 m^2 . This assumes the maximum area for the fewest complex folds, structural components, and actuators necessary to unfold and construct the radiators in orbit [7]. With the successful construction of the ISS and its approximately 500 m^2 radiator acreage in orbit, it can be stated that radiator construction is already at a TRL of 8-9. However this excludes the issue in stacking large radiators together and properly connecting their shared plumbing. For each launcher vehicle needed to construct this system up to the 10000 m^2 maximum acreage, complexity of construction continues to move the TRL lower.

This section looks at the performance of the system designed around the radiator if the number of launches was limited. This can be done by adjusting the NTP thrust of the base design until a quarter of the initial area is removed, and then again until reaching the 2500 m^2 SLS launch maximum.

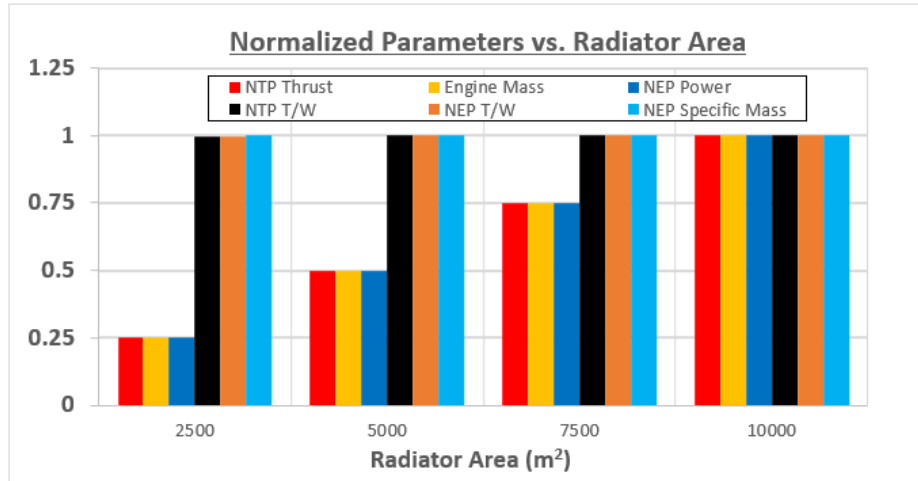


Figure 29: Normalized WREN performance metrics as a function of Radiator Area

A very linear correlation forms from this constraint once performance metrics are normalized to the base WREN system. Figure 29 shows that for each quarter that is taken away, the NTP thrust, NEP power, and total engine mass decrease by a quarter of the original results. What this implies is that radiator area is correlated to the performance as a direct scaling factor for the whole bimodal macro system. While ratios such as T/W and α remain the same, the NTP thrust and NEP power must be balanced with the payload delivered for the required ΔV for the 45-65 day Martian trajectory.

7.2 Deep Space Mission to Outer Planets

In early 2003, a NASA Science Definition Team (SDT) was formed to develop a concept for a follow on to the successful Galileo project. They proposed the Jupiter Icy Moons Orbiter (JIMO) mission to explore Europa and the Jupiter system and the search of extraterrestrial life. The study identified three crosscutting themes:

”oceans (finding their locations, studying the structure of their icy crusts, and assessing active internal processes); astrobiology (determining the types of volatiles and organics on and near the surfaces, and the processes involved in their formation and modification); and Jovian system interactions (studying the atmospheres of Jupiter and the satellites and the interactions among Jupiter, its magnetosphere and the surfaces and interiors of the satellites, including Io).” [87]

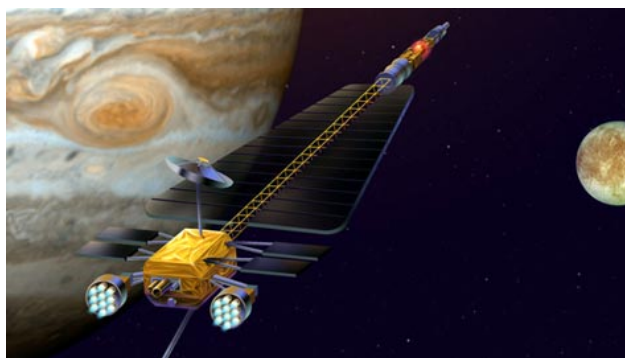


Figure 30: JIMO mission spacecraft concept.

The JIMO mission was soon integrated into the Prometheus Project that focused on a nuclear systems and technology theme. This project was to develop a Deep Space Vehicle (DSV) for outer solar system robotic exploration missions that would combine a safe, reliable, Space Nuclear Reactor with electric propulsion. The vehicle included a Payload Accommodation Envelope (PAE) with a mass capability for science instruments and supporting resource of not less than 1500 kg. There was also the exploration of commonality of nuclear power for Lunar and Mars surface power and cargo transport missions. The first mission was to continue the vision of JIMO and explore the Jovian system. The DSV was also designed to enable multiple follow on missions listed in Table 9.

Table 9: Proposed follow on mission for the Prometheus DSV.

Follow-On Missions	Description
Saturn and its moons	Comprehensive exploration of Saturn and Titan
Neptune and its moons	Comprehensive study of the Neptunian system
Kuiper Belt Rendezvous	Rendezvous with and study multiple Kuiper Belt objects
Interstellar Precursor	Reach 200 AU at the heliopause nose
Comet Cryogenic Sample Return	Return a cryogenically preserved sample from a comet
Multi-Asteroid Sample Return	Study multiple asteroid types and return samples from each

In regards to our NIAC project, the DSV called for an electric thruster power of 180 kWe designed with Xenon electric thrusters providing an I_{sp} in the range of 6000 to 8000

seconds.[88] Compared to the Mars mission we studied, the mission design time was estimated to be able to be implemented at a maximum 20 year operational time. The missions are also obviously not manned, making the transit time for mission not as high priority. Therefore, a fully NEP system was only considered. A NASA report led by Bailey [89] provided a listing of relevant space nuclear power systems as shown in Table 10. The JIMO systems has a reactor temperature at 1150 and 1300 K for two designed using a Brayton cycle for power conversion. The α_{NEP} of the two systems ranged from 35 to 45 kg/kWe. This was significantly higher than NEP concepts under the SNP project as shown in Fig 16 along with our WREN concept.

Table 10: Relevant Space Nuclear Power Systems.

System	Fuel (Coolant)	Thot/ Tcold	Power Conv.	Thermal Power	Electrical Power	System Mass	Design Life
GPHS RTG*	Pu-238	1273/ 573 K	SiGe TE	4.5 kW	285 W	56 kg	10+ yrs
MMRTG	Pu-238	811/ 483 K	PbTe/ TAGS TE	2 kW	110 W	45 kg	14 yrs
ASRG	Pu-238	1123/ 363 K	Stirling	500 W	140 W	25 kg	14 yrs
SNAP 10A*	UZrH (NaK)	790/ 560 K	SiGe TE	40 kW	500 W	432 kg	1 yr
BUK/ RORSAT*	UMo (NaK)	973/ 623 K	SiGe TE	100 kW	3 kW	1200 kg	6 mo
Topaz I*	UO ₂ (NaK)	1723/ 923 K	Thermionic	150 kW	6 kW	1000 kg	1 yr
Topaz II	UO ₂ (NaK)	1800/ 743 K	Thermionic	135 kW	5.5 kW	1060 kg	3 yrs
SP-100	UN (Li)	1350/ 850 K	SiGe TE	2400 kW	100 kW	4500 kg	7-10 yrs
JIMO (TB2.5)	UN (Li)	1300/ 500 K	Brayton	500 kW	135 kW	6200 kg	15-20 yrs
JIMO (PB1)	UO ₂ (HeXe)	1150/ 400 K	Brayton	1000 kW	200 kW	6800 kg	15-20 yrs
FSP	UO ₂ (NaK)	850/ 425 K	Stirling	200 kW	40 kW	5800 kg	8 yrs

* These systems have been launched and operated in space.

Focusing on the Jovian mission, work by Parcher and Sims [90] studies Venus and Mars gravity-assist trajectories to Jupiter with NEP. They found that mission ΔV requirements ranged from 10 to 20 km/s depending on different gravity-assist combinations. The ΔV begins to increase exponentially when the time of flight is less than 5.5 years for what they defined as a direct transfer to Jupiter. In this type of mission, the spacecraft does not first orbit Earth and is directly injected into an orbit destined for Jupiter. Using these results, we looked at a 95 kW design with NEP thruster technology providing an I_{sp} of 6000 seconds and electrical conversion efficiency of 70%. This produced a maximum thrust of 2.26 N with a mass flow rate of 3.48×10^{-5} kg/s. The low α_{NEP} of the WREN concept would be able to provide a better acceleration profile and significantly reduce the required Injection C3 for a notional spacecraft.

Another possible option is to consider what to do with a the WREN spacecraft designed for human transport to Mars. After performing a number of missions, would the risk of failure for outer planet missions be allowed to be higher than a human rated mission to

Mars? If so, the end of life use case for the Mars transport vehicle, could be to send the spacecraft to the outer planets for a final exploration mission as part of a decommissioning strategy.

8 Recommended Future Steps

The inclusion of wave rotors into the design of a bi-modal NTP/NEP system enables fast-transit missions to Mars. Our proposed study goal of completing a 45 days to Mars leg was found to be achievable in providing similar initial masses as current NASA mission concept spacecraft. It was found that the NTP T/W was too low to perform a similar maneuver leaving Earth as the NASA NEP-CP concept spacecraft (Fig. 2). Either a different departure strategy must be used or further performance capability must be developed for the WREN cycle. More importantly, this study found that significant saving in initial Earth departure mass can be realized for a 65 day transit time. Additionally, only looking at the Earth to Mars transit leg is not obviously a full mission. The results of this study indicate that a full mission scenario study would lead to similar significant improvements compared to current NASA reference mission designs limited to opposition class short stay missions.

To further develop the wave rotor technology, digital modeling using Computational Fluid Dynamics (CFD) is required to understand the system performance limits when include into the WREN cycle. Most importantly, is the ability of the wave rotor to process gases at temperatures significantly above material limits due to the unsteady shock heating process. CFD coupled with conjugate heat transfer analysis is required to find estimates on materials limits. Additionally, a testing apparatus similar in nature to the NASA Glenn wave rotor design (Fig. 31) is required to verify capabilities in a lab environment to push the TRL level of the concept. After developing a digital model with CFD, requirements for experimental testing can be developed.

We also recommend the continued development of high temperature nuclear fission reactors with focus on extending their endurance to long duration. With significant improvement in I_{sp} of the NTP cycle, long duration burns (> 2 hours) may be advantageous. In regards, to the Brayton cycle, there is need to explore how temperatures may be further increased for the Brayton core outlet temperature as shown in Table. 8. This can be done by significantly increasing the inlet temperature capabilities of in-space turbine systems and is therefore a significantly important area of research to be supported. Finally, the theoretical power levels achievable for the WREN cycle enable MW NEP designs. Research needs to be focused on electric propulsion thruster technology that can handle these extreme power levels. NASA currently has the ability to test up to 1 MW scale thrusters in their vacuum chamber test bed. Therefore it is not unreasonable to assume a program for their development can be conducted without requiring significant infrastructure investment.

9 Conclusion

A systems level model of a bi-modal NTP/NEP propulsion system was developed. Through the use of wave rotor technology, both NTP and NEP propulsion parameters were signifi-

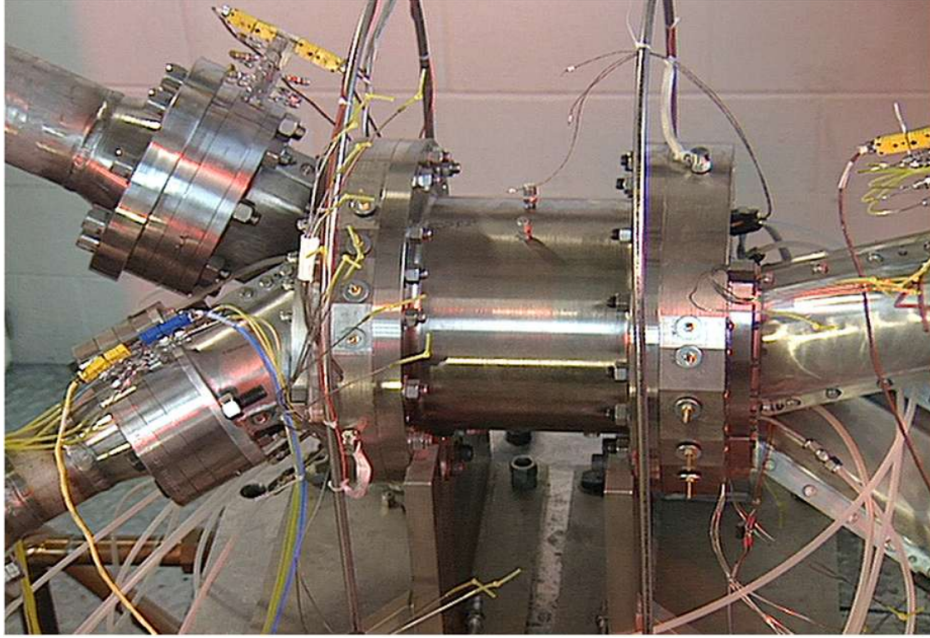


Figure 31: NASA Glenn wave rotor testing apparatus.

cantly improved. A simplified trajectory analysis model was developed to explore the capability of an non-optimized configuration. It was found that the system provided significant improvement in its ability to conduct a Fast-Transit mission to Mars. Though the system results in reasonable initial masses for a concept spacecraft traveling to Mars in 45 days, significant mass saving can be realized by increasing the transit time by just 5 to 20 days. It is suggested that a full mission profile should be explored to understand the implications of having to minimize transit time for both the outbound and return leg of a human Mars mission. Either way, this propulsion system technology demonstrates the ability to easily conduct Fast-Transit missions to Mars.

10 Acknowledgments

We would like to acknowledge Prof. William Lear, Dr. Gerard Welch, Dr. Dan Paxson, and Robert Kielb for their discussions on wave rotor technology. To Dr. Kurt Polzin for his discussions on nuclear reactor and electric propulsion technology development. To Prof. Christopher Petersen for his discussions on mission trajectory optimization. Also, the NASA NIAC program personnel (Michael LaPointe, John Nelson, Ron Turner, Kathy Reilly, Frank Spellman, Mau Nguyen, and Micah Bullard) and external council (Mae Jemison, David Brin, Louis Friedman, Amy Kronenberg, Edgar Zapata, and Merri Sanchez) for their discussions on the technology during the NASA NIAC Phase I activities.

11 Appendix

The pressure and temperatures of the two propulsion gas loops are listed here in the Appendix in Table 11 for completeness. Station positions are labeled in Fig. 13. As detailed in the figure, yellow (motive) and orange values (driver) specify the port conditions surrounding the NTP Wave Rotor during the NTP mode which results in the optimized I_{sp} of the thermal rocket. Values highlighted in red and blue represent the port conditions surrounding the Brayton Wave Rotor for the NTP and NEP mode, respectively. Note that these value counterparts between modes surrounding the Brayton Wave Rotor are equivalent (e.g. $p_{1Be} = p_{1Bt}$), indicating the OOP model successfully balanced the modes. Pressure values were set based on requirements for the NTP rocket nozzle and nuclear reactor cores.

Table 11: Pressure and temperature values at their respective stations based on Fig. 13.

Station Pressures (kPa)		Station Temperatures (K)	
NTP Mode Motive Gas Cycle			
p0	139.27	T0	20.39
p1	700.12	T1	21.00
p2	686.39	T2	519.76
p3	672.94	T3	2850.00
p4	2541.39	T4	4768.08
NTP Mode Driver Gas Brayton Cycle			
p0Bt	88.49	T0Bt	359.99
p1Bt	179.64	T1Bt	454.68
p2Bt	858.67	T2Bt	770.96
p3Bt	841.84	T3Bt	519.76
p4Bt	4217.70	T4Bt	870.92
p5Bt	4135.00	T5Bt	2850.00
p6Bt	859.36	T6Bt	2151.75
p7Bt	842.17	T7Bt	1966.28
p8Bt	347.25	T8Bt	1650.00
p9Bt	92.07	T9Bt	1241.03
p10Bt	90.26	T10Bt	519.76
NEP Mode Driver Gas Brayton Cycle			
p0Be	90.26	T0Be	362.71
p1Be	179.64	T1Be	454.68
p2Be	858.67	T2Be	770.96
p7Be	842.17	T7Be	1966.28
p8Be	347.25	T8Be	1650.00
p9Be	92.07	T9Be	1241.03

References

- [1] Clare Skelly. *NASA Announces Nuclear Thermal Propulsion Reactor Concept Awards*. 2021. URL: <https://www.nasa.gov/press-release/nasa-announces-nuclear-thermal-propulsion-reactor-concept-awards>. (accessed: 1.16.2024).
- [2] Clare Skelly. *Nuclear Thermal Propulsion: Game Changing Technology for Deep Space Exploration*. 2018. URL: <https://www.nasa.gov/directorates/stmd/tech-demo-missions-program/nuclear-thermal-propulsion-game-changing-technology-for-deep-space-exploration/>. (accessed: 1.16.2024).
- [3] W Robbins. “An historical perspective of the NERVA nuclear rocket engine technology program”. In: *Conference on Advanced SEI Technologies*. 1991, p. 3451.
- [4] Michael L Belair, Charles Sarmiento, and Thomas Lavelle. “Nuclear thermal rocket simulation in NPSS”. In: *49th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*. 2013, p. 4001.
- [5] NASA STMD. *GO: Space Nuclear Propulsion*. 2023. URL: <https://techport.nasa.gov/file/142857>. (accessed: 1.16.2024).
- [6] NASA STMD. *GO: Advanced Propulsion*. 2023. URL: <https://techport.nasa.gov/file/142854>. (accessed: 1.16.2024).
- [7] Steven R Oleson et al. *Compass Final Report: Nuclear Electric Propulsion (NEP)-Chemical Vehicle 1.2*. Tech. rep. 2021.
- [8] Seymour T Nelson, Jerry Grey, and Peter M Williams. “Conceptual study of a liquid-core nuclear rocket”. In: *Journal of Spacecraft and Rockets* 2.3 (1965), pp. 384–391.
- [9] W LOUIS BARRETT JR. “Specific impulse of a liquid-core nuclear rocket”. In: *AIAA Journal* 1.11 (1963), pp. 2649–2650.
- [10] Jimmy Allen et al. “Overview of High-Performance Centrifugal Nuclear Thermal Rocket Propulsion System”. In: *ANS Virtual Winter Meeting*. 2020.
- [11] Michael G Houts, L Dale Thomas, and Bahram Nassersharif. “Centrifugal Nuclear Thermal Rocket Challenges and Potential”. In: *45th Annual AAS Guidance, Navigation and Control Conference*. AAS 23-175. 2023.
- [12] Lawrence Thomas et al. “Establishing the Feasibility of the Centrifugal Nuclear Thermal Rocket”. In: *AIAA Propulsion and Energy 2021 Forum*. 2021, p. 3604.
- [13] Yebing Zhang. “Advanced nuclear power engine: A brief overview of gas core reactor for space exploration”. In: *International Journal of Advanced Nuclear Reactor Design and Technology* (2023).
- [14] Joel Krakower and Michael G Houts. “Thermal Challenges in High Performance Nuclear Thermal Propulsion Systems”. In: *AIAA SCITECH 2024 Forum*. 2024, p. 1236.
- [15] Pezhman Akbari, Razi Nalim, and Norbert Mueller. “A review of wave rotor technology and its applications”. In: *Journal of engineering for gas turbines and power* 128.4 (2006), pp. 717–735.

- [16] Iuliana Costiuc and Anghel Chiru. “Evolution of the pressure wave supercharger concept”. In: *IOP conference series: materials science and engineering*. Vol. 252. 1. IOP Publishing. 2017, p. 012081.
- [17] G. E. Welch. “Wave engine topping cycle assessment”. In: *35th Aerospace Sciences Meeting and Exhibit*. 1997, p. 707.
- [18] Mohamed Razi Nalim et al. “Hybrid Wave Rotor Electric Aero Propulsion”. In: *50th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*. 2014. URL: <https://api.semanticscholar.org/CorpusID:209964013>.
- [19] HE Weber. “Shock-Expansion Wave Engines—New Directions for Power Production”. In: *Turbo Expo: Power for Land, Sea, and Air*. Vol. 79283. American Society of Mechanical Engineers. 1986, V001T01A025.
- [20] M Bellenoue et al. “New combustion concepts to enhance the thermodynamic efficiency of propulsion engines”. In: *Aerospace Lab* 11 (2016), 13–pages.
- [21] P H Rose. “Potential applications of wave machinery to energy and chemical processes”. In: *Proceedings of the 12th International Symposium on Shock Tubes and Waves* (1979), pp. 3–30.
- [22] R. Kielb, A. Castrogiovanni, and R. Volland. “Wave Rotor for Continuous, Vitiating-Free, High-Enthalpy Test Gas”. In: *JANNAF*. 34th ASP. 2014.
- [23] Jack Wilson and Daniel E Paxson. “Jet engine Performance Enhancement Through Use of a Wave-Rotor Topping Cycle”. In: *NASS-TM-4486* (1993).
- [24] Gerard E Welch. “Macroscopic balance model for wave rotors”. In: *Journal of Propulsion and Power* 13.4 (1997), pp. 508–516.
- [25] Joseph Banks. *Nuclear Thermal Propulsion Systems*. 2023. URL: <https://www1.grc.nasa.gov/research-and-engineering/nuclear-thermal-propulsion-systems/>. (accessed: 11.16.2023).
- [26] Florin Iancu, Janusz Piechna, and Norbert Müller. “Basic design scheme for wave rotors”. In: *Shock Waves* 18 (2008), pp. 365–378.
- [27] Jack Wilson and Daniel E Paxson. *Optimization of Wave Rotors for Use as Gas Turbine Engine Topping Cycles*. Tech. rep. SAE Technical Paper, 1995.
- [28] Antony Jameson, Wolfgang Schmidt, and Eli Turkel. “Numerical solution of the Euler equations by finite volume methods using Runge Kutta time stepping schemes”. In: *14th fluid and plasma dynamics conference*. 1981, p. 1259.
- [29] Victor Granet et al. “Comparison of nonreflecting outlet boundary conditions for compressible solvers on unstructured grids”. In: *AIAA journal* 48.10 (2010), pp. 2348–2364.
- [30] C Haertling and RJ Hanrahan Jr. “Literature review of thermal and radiation performance parameters for high-temperature, uranium dioxide fueled cermet materials”. In: *Journal of nuclear materials* 366.3 (2007), pp. 317–335.

- [31] Jacob Hirschhorn et al. “Review and preliminary investigation into fuel loss from cer-mets composed of uranium nitride and a molybdenum-tungsten alloy for nuclear thermal propulsion using mesoscale simulations”. In: *JOM* (2021), pp. 1–16.
- [32] F Homan, J Napier, and C Caldwell. “Particle fuels technology for nuclear thermal propulsion”. In: *Conference on Advanced SEI Technologies*. 1991, p. 3457.
- [33] Nicholas R Brown. “A review of in-pile fuel safety tests of TRISO fuel forms and future testing opportunities in non-HTGR applications”. In: *Journal of Nuclear Materials* 534 (2020), p. 152139.
- [34] Caen Ang, Lance Snead, and Kelsa Benensky. “Niobium carbide as a technology demonstrator of ultra-high temperature ceramics for fully ceramic microencapsulated fuels”. In: *International Journal of Ceramic Engineering & Science* 1.2 (2019), pp. 92–102.
- [35] Travis Knight and Samim Anghaie. “Advanced carbide fuels for space nuclear thermal propulsion”. In: *37th Joint Propulsion Conference and Exhibit*. 2001, p. 3510.
- [36] R Matthews et al. “Carbide fuels for nuclear thermal propulsion”. In: *Conference on Advanced SEI Technologies*. 1991, p. 3455.
- [37] Samim Anghaie and Travis Knight. “Status of advanced carbide fuels: Past, present, and future”. In: *AIP Conference Proceedings*. Vol. 608. 1. American Institute of Physics. 2002, pp. 852–856.
- [38] Richard L Williamson et al. “BISON: A flexible code for advanced simulation of the performance of multiple nuclear fuel forms”. In: *Nuclear Technology* 207.7 (2021), pp. 954–980.
- [39] Marina Sessim and Michael R Tonks. “Multiscale simulations of thermal transport in W-UO₂ CERMET Fuel for nuclear thermal propulsion”. In: *Nuclear Technology* 207.7 (2021), pp. 1004–1014.
- [40] Mark D DeHart, Sebastian Schunert, and Vincent M Labouré. “Nuclear thermal propulsion”. In: *Nuclear Reactors: Spacecraft Propulsion, Research Reactors, and Reactor Analysis Topics* (2022), p. 3.
- [41] Sanford Gordon and Bonnie J McBride. *Thermodynamic data to 20,000 K for monatomic gases*. Tech. rep. 1999.
- [42] Bonnie J McBride. *Thermodynamic data for fifty reference elements*. National Aeronautics and Space Administration, Glenn Research Center, 2001.
- [43] Daniel E. Paxson. “A general numerical model for Wave Rotor Analysis”. In: vol. 92. National Aeronautics and Space Administration, 1992, p. 1.
- [44] Shu-cheng S Chen, Joseph P Veres, and James E Fittje. “Turbopump design and analysis approach for nuclear thermal rockets”. In: *AIP Conference Proceedings*. Vol. 813. 1. American Institute of Physics. 2006, pp. 522–530.
- [45] Mark V Zagarola and John A McCormick. “High-capacity turbo-Brayton cryocoolers for space applications”. In: *Cryogenics* 46.2-3 (2006), pp. 169–175.

- [46] Carl E Walter. “Gas-cooled reactor power systems for space”. In: *4th Symposium on Space Nuclear Power Systems*. 1987, p. 4.
- [47] J Douglas Kiser et al. *Overview of CMC (Ceramic Matrix Composite) research at the NASA glenn research center*. Tech. rep. 2016.
- [48] Nathaniel Read and Eugene Shwageraus. “Megawatt class space fission power system mass modelling”. In: *Annals of Nuclear Energy* 182 (2023), p. 109594.
- [49] Christopher G Morrison. “Temperature and Power Specific Mass Scaling for Commercial Closed-Cycle Brayton Systems in Space Surface Power and Nuclear Electric Propulsion Applications”. In: *Nuclear Technology* 206.8 (2020), pp. 1224–1239.
- [50] Christopher Harnack et al. “Brayton Cycle Power Conversion Model for MW-Class Nuclear Electric Propulsion Mars Missions”. In: *NETS 2022*. 2022.
- [51] Matthew E Duchek et al. “Key Performance Parameters for MW-Class NEP Elements and their Interfaces”. In: *ASCEND 2022*. 2022, p. 4273.
- [52] Joseph E Grady et al. “Overview of ceramic matrix composite research at NASA Glenn Research Center”. In: (2017).
- [53] Dane Dale and Alex Ruschau. “Nasa Hytec CMC Turbine Blade Durability”. In: (2023).
- [54] Briana N Tomboulia. “Lightweight, High-temperature radiator for in-space nuclear-electric power and propulsion”. In: (2014).
- [55] John W Dankanich and Ronald J Litchford. “NASA Envisioned Future Priorities for In-Space Transportation”. In: *International Astronautical Congress (IAC 2022)*. 2022.
- [56] Samantha B Rawlins and L Dale Thomas. “Feasibility of Low-Enriched Uranium Fueled Nuclear Thermal Propulsion in the Low-Thrust Region Below 16klbf”. In: *Annals of Nuclear Energy* 179 (2022), p. 109368.
- [57] Bruce G Schnitzler. *Small reactor designs suitable for direct nuclear thermal propulsion: interim report*. Tech. rep. Idaho National Lab.(INL), Idaho Falls, ID (United States), 2012.
- [58] Paul H Sager. “Radiation shield requirements for manned nuclear propulsion space vehicles”. In: *AIP Conference Proceedings*. Vol. 246. 1. American Institute of Physics. 1992, pp. 1251–1258.
- [59] Co Huynh et al. “Design and Development of a Two-Megawatt, High Speed Permanent Magnet Alternator for Shipboard Application”. In: *Naval engineers journal* 117.4 (2005), pp. 23–29.
- [60] K Sivasubramaniam et al. “Development of a high speed HTS generator for airborne applications”. In: *IEEE Transactions on applied superconductivity* 19.3 (2009), pp. 1656–1661.
- [61] Rodger Dyson et al. *Nuclear electric propulsion brayton power conversion working fluid considerations*. Tech. rep. 2022.
- [62] Anthony J Colozza. *Small lunar base camp and in situ resource utilization oxygen production facility power system comparison*. Tech. rep. 2020.

- [63] Lothar Heinemann. “Analysis and design of a modular, high power converter with high efficiency for electrical power distribution systems”. In: *2002 IEEE 33rd Annual IEEE Power Electronics Specialists Conference. Proceedings (Cat. No. 02CH37289)*. Vol. 2. IEEE. 2002, pp. 713–718.
- [64] Niraja Swaminathan and Yue Cao. “An overview of high-conversion high-voltage DC–DC converters for electrified aviation power distribution system”. In: *IEEE Transactions on Transportation Electrification* 6.4 (2020), pp. 1740–1754.
- [65] Ralph Jansen et al. “High efficiency megawatt motor conceptual design”. In: *2018 Joint Propulsion Conference*. 2018, p. 4699.
- [66] Di Zhang et al. “Development of megawatt-scale medium-voltage high efficiency high power density power converters for aircraft hybrid-electric propulsion systems”. In: *2018 AIAA/IEEE Electric Aircraft Technologies Symposium (EATS)*. IEEE. 2018, pp. 1–5.
- [67] John Blandino et al. “Overview of electric propulsion research in U.S. academia”. In: *39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit*. 2003.
- [68] Michael LaPointe and Pavlos Mikellides. “High power MPD thruster development at the NASA Glenn Research Center”. In: *37th Joint Propulsion Conference and Exhibit*. 2001, p. 3499.
- [69] Robert H Frisbee, Nathan J Hoffman, and Kathy H Murray. “SP-100 Dynamic Power and Lithium-Propellant MPD Nuclear Electric Propulsion Technology Requirements”. In: *AIP Conference Proceedings*. Vol. 301. 1. American Institute of Physics. 1994, pp. 1251–1260.
- [70] James Gilland et al. “Mission Performance of High-Power Electromagnetic Thruster Systems”. In: *AIP Conference Proceedings*. Vol. 813. 1. American Institute of Physics. 2006, pp. 201–208.
- [71] Robert D Brown and Roger M Meyers. “Nuclear Electric Propulsion”. In: *Space Nuclear Propulsion for Human Mars Exploration*. National Academies Press. 2021, pp. 30–54.
- [72] Andrew V Ilin et al. “VASIMR® human mission to Mars”. In: *Space, Propulsion & Energy Sciences International Forum*. 2011, pp. 1–12.
- [73] Marcus R Collier-Wright et al. “High-temperature superconductor-based power system architectures as enablers for high power missions”. In: *ASCEND 2021*. 2021, p. 4105.
- [74] Geoffrey A Landis. “Analysis of Thermoradiative Thermal Energy Conversion”. In: *APS Eastern Great Lakes Section (EGLS) Fall 2023 Meeting*. 2023.
- [75] Loura Hall. “Radioisotope Thermoradiative Cell Power Generator”. In: *2023 NASA NIAC Phase I Selection*. 2023. URL: <https://www.nasa.gov/general/radioisotope-thermoradiative-cell-power-generator/>.
- [76] Stanley K Borowski, David McCurdy, and Thomas Packard. “Nuclear Thermal Propulsion (NTP): A Proven, Growth Technology for ‘Fast Transit’ Human Missions to Mars”. In: *AIAA SPACE 2013 Conference and Exposition*. 2013, p. 5354.

- [77] NASA. *NASA Research Announcement (NRA): Space Technology – Research, Development, Demonstration, and Infusion 2018 (SpaceTech-REDDI-2018), 80HQTR18NOA01*. May 2018.
- [78] Nicolas Bérend et al. “How fast can we go to Mars using high power electric propulsion?” In: *48th AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit*. 2012, p. 3889.
- [79] Nicolas Bérend. “Optimum design of power-limited propulsion systems with application to fast Earth-to-Mars transfer”. In: *Acta Astronautica* 79 (2012), pp. 107–117.
- [80] John Dankanich, Bob Vondra, and Andrew Ilin. “Fast transits to Mars using electric propulsion”. In: *46th AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit*. 2010, p. 6771.
- [81] Jacob Williams et al. “Overview and software architecture of the copernicus trajectory design and optimization system”. In: *4th International Conference on Astrodynamics Tools and Techniques*. JSC-CN-20553. 2010.
- [82] Stanley K Borowski et al. *Robust Exploration and Commercial Missions to the Moon Using NTR LANTR Propulsion and Lunar-Derived Propellants*. Tech. rep. 2017.
- [83] Feijia Yin and A Gangoli Rao. “Performance analysis of an aero engine with inter-stage turbine burner”. In: *The Aeronautical Journal* 121.1245 (2017), pp. 1605–1626.
- [84] RonaldG Toelle. *A performance study for the application of the Saturn V to high energy earth escape missions*. Tech. rep. 1967.
- [85] EA Mock and JE Davis. *Nuclear Brayton turboalternator-compressor (TAC) conceptual design study*. Tech. rep. NASA, 1972.
- [86] John L Mason. “Working gas selection for the closed Brayton cycle”. In: *AGARDograph, Volume Date* 81 (1964), pp. 223–252.
- [87] Ronald Creely and Torrence Johnson. “The Jupiter icy moons orbiter project: The scientific rationale”. In: *Eos, Transactions American Geophysical Union* 85.36 (2004), pp. 337–343.
- [88] Randall Taylor. “Prometheus Project final report”. In: (2005).
- [89] Sterling Bailey et al. “Small Fission Power System Feasibility Study Final Report”. In: (2010).
- [90] Daniel W Parcher and Jon A Sims. “Venus and Mars gravity-assist trajectories to Jupiter using nuclear electric propulsion”. In: *Advances in the Astronautical Sciences* 120 (2005), pp. 169–184.