KEY PERFORMANCE PARAMETERS FOR AN OPERATIONAL CIS-LUNAR NTP VEHICLE

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ABSTRACT

Nuclear Thermal Propulsion (NTP) is in active development by NASA and DARPA. This paper presents an investigation into the effects of different values of key performance parameters (KPPs) of an NTP engine in the context of payload delivery missions between Earth and Lunar orbits. The high I_{sp} afforded by NTP combined with the relatively large engine mass is best applied to missions where the payload mass is large and required ΔV high. Two missions considered are the round-trip Earth to Moon tug and a NRHO to low Lunar orbit tug. Chemical propulsion vehicles can generally achieve higher propellant mass fractions, similar payload mass and similar ΔV performance on these missions. When re-use of the vehicle is considered, the reduced propellant mass required by an NTP vehicle begins to add up over multiple refilling launches and trades more favorably. Analysis illustrates the impact of variation in NTP engine performance values, especially I_{sp}, engine mass and thrust, on performance and the comparison to chemical propulsion. NTP transient and cooldown effects on the vehicle performance are examined. A variety of launch vehicles and hydrogen and ammonia NTP propellants are included in the analyses, and cryogenic fluid management system effects are accounted for.

INTRODUCTION

BACKGROUND

Nuclear thermal propulsion (NTP) is currently in development by NASA in the Space Nuclear Propulsion Program. The DARPA and NASA partnership in the DRACO project is pursuing development of a NTP engine for an initial in-space demonstration, ultimately extensible to the crewed Mars mission [1]. The applications for an NTP engine are multiple. NASA has long envisioned use of NTP for a crewed Mars transportation vehicle [2], [3], [4]. More recently NASA has, along with partners, investigated applications for NTP in the Earth-Moon system, cis-Lunar space, and for Earth-departure vehicles [5], [6], [7]. The appropriate thrust for an NTP engine has been investigated for the crewed Mars mission application, and some cis-Lunar applications studies have considered thrust level [8].

This paper considers in more depth the key performance parameters (KPPs) of an operational Earth-Moon or cis-Lunar vehicle. The thrust, mass, and specific impulse (I_{sp}) are all considered, as well as the impact on mission performance of different values of these KPPs. The relative contributions to performance of transient, steady state, and cooldown I_{sp} are considered for an Earth-Moon tug mission application. Additionally, operational capability KPPs such as maximum burn time are discussed. During the analysis, liquid hydrogen (LH_2) and ammonia (NH_3) propellants were considered for an NTP vehicle [9]. Comparison to an liquid oxygen / liquid hydrogen (LOX/LH2) chemical engine vehicle is also considered.

KEY PERFORMANCE PARAMETERS FOR AN NTP ENGINE

Two categories of KPP for an NTP engine are: operational capability parameters and performance parameters. These categories of KPP are not fundamentally distinct, the distinction is made simply to organize discussion.

Operational Capability

To meet the needs of a mission, the engine must restart a minimum number of times and be capable of firing for the duration of the longest required engine firing.

The mission concept of operations (CONOPS) drives the **number of restarts** that are required, but each of the CONOPS we consider in this paper require a minimum of four restarts. Six to seven restarts are required to give flexibility to the mission designer, enabling for example deep space maneuvers and separating an Earth-departure maneuver into an orbit raise and an Earth-departure firing. Such flexibility is important to enable trade-off of gravity losses with transient and cooldown effects on NTP engine performance. For an engine on a re-usable vehicle, 10-20+ restarts would enable 2-5 uses.

The **engine burn time** required is also a function of the mission CONOPS. However, in considering a KPP for Earth-Moon operations, the longest burn will happen during an Earth-departure or trans-lunar injection maneuver. There is a limit in burn time for this maneuver where gravity losses start to become quite high and a mission designer would likely trade an additional burn to first raise the orbit then complete the TLI or Earth-departure rather than extending the burn time. Roughly, 30 min is a maximum burn time for this type of CONOPS. Note that deep space maneuvers for Mars missions can be longer.

For a re-usable vehicle, the **mass of propellant required to re-fill** the vehicle may also be considered a KPP in some cases. Where the vehicle operates as an Earth-Moon tug and is re-filled in an Earth orbit similar to its initial launch orbit, the propellant mass itself is not a driving parameter, as the same number of launches needed to initially prepare the vehicle will suffice to re-fill it. However, for a cis-Lunar vehicle that operates for example between NRHO and LLO, the number of launches needed to re-fill it in NRHO (or Earth-Moon tugs transferring propellant to an NRHO depot) will be directly determined by the total required propellant mass.

Performance KPPs

Performance for an NTP stage can be considered as either the payload capability for a given mission or the ΔV capability for a given payload. This performance is determined by the basic rocket equation and additionally limited by the gravity and turning losses incurred through a CONOPS. The losses are driven by the length of the burns required due to the thrust to weight ratio of the vehicle and its payload. The basic capability is otherwise determined by the terms in the rocket equation: the propellant mass, the stage dry mass, and the specific impulse. The KPPs identified are listed in Table 1 and described below.

Table 1: Vehicle and Engine KPPs

Vehicle KPPs	Engine KPPs
Propellant Mass Available	Maximum Burn Time
Tank Mass per Propellant Mass	Number of Restarts
Other Dry Mass	Engine Mass
Total Thrust	Engine Thrust
Effective I_{sp} (for a given mission CONOPS)	Steady-State I_{sp}
	Transient Rate
	Cooldown mass per impulse

The **propellant mass available** is constrained by the size of the tank, which in this paper is limited by the capability of the launch vehicle that initially places the vehicle in orbit. For hydrogen propellant, the constraint is often the physical size of the launch vehicle fairing, where for chemical propulsion vehicles or ammonia NTP propellant, the mass limit of the launch vehicle limits the vehicle to a smaller size. Larger vehicles can be considered if multiple launches are used to fill the tanks to full, but the mass of the chemical vehicles can become quite large when compared to H-NTP due to the higher density of the propellants.

The dry mass of the vehicle consists of three elements: the engine mass, the tank mass, and the mass of other supporting subsystems. The **engine mass** is the differentiator between NTP and chemical systems and is large enough and uncertain enough (at this stage in development) to impact performance. For a launch vehicle, the engine mass is often represented in a KPP as the thrust to weight ratio of the engine itself. In the case of optimizing a multi-stage launch vehicle, this is a useful construction. However, for the tug vehicle considered here, which is constrained by the mass and fairing volume capability of a launch vehicle, the engine mass itself makes a more useful KPP.

The **tank mass per propellant mass** varies significantly among hydrogen, ammonia, LOX, and methane propellants due to the differences in density and cryogenic fluid management (CFM) systems (which we here include as 'tank mass'). The mass efficiency of a hydrogen CFM system will have a significant impact on vehicle performance. The remainder of the supporting subsystems mass will depend on the mission requirements and the specific design decisions of the vehicle manufacturer.

The **specific impulse (I_{sp})** of the engine is a key performance parameter. For an NTP engine, the specific impulse achieved in an operational burn is a composite of the I_{sp} ramp-up during start-up, the steady-state I_{sp}, the ramp-down during shutdown, and the small impulse provided by the cooldown propellant. The steady-state portion of the burn provides most of the total burn impulse (except for very short burns), but the other phases of a burn are all significant enough to affect the vehicle performance. In the results, each phase and its impact on the I_{sp} achieved for a maneuver are described. The impacts are great enough that the transient ramp rate (denoted as either MW/s or K/s) and the cooldown propellant mass (kg/N-s of impulse) can be considered sub-KPPs to the engine specific impulse. In the discussion and some plots below, we use the term "**effective I_{sp}**". This refers to the combined average specific impulse of an engine firing from start-up, steady-state operation, shutdown and cooldown.

APPLICATIONS FOR NTP

When considering applications for NTP, these fundamental attributes of the system must be considered:

- NTP has higher I_{sp} potential than chemical propulsion
- NTP engines will have a lower thrust to weight ratio (T/W) due to the large reactor mass
- Hydrogen storage for H-NTP will have higher tank mass per propellant mass than storage of other propellants / oxidizers that are more dense and are stored at less extreme cryogenic temperatures
- NTP engines will be more costly than equivalent chemical engines

Therefore, to best utilize the potential of NTP, it must be utilized in a mission that leverages the advantages and can live with the disadvantages. High I_{sp} means that, for a given ΔV and dry mass, much less propellant mass will be needed to complete a maneuver. This effect becomes magnified for large ΔV requirements. However, since the rocket equation works off the wet to dry mass ratio of a vehicle, the engine mass is a key parameter in determining the performance of an NTP stage. Tank mass per propellant mass is also important. This paper focuses on the parameters of concern to the engine developer, as ultimately, CFM technology and its effect on "tank" mass will be tradeable.

For a given stage, limited in size and mass by the launch vehicle that puts it into orbit, NTP will tend to outperform chemical to a greater extent when the T/W ratio of the vehicle is improved (either lower engine mass or lower tank mass) and when the payload mass is higher. Higher payload mass will more dramatically affect the vehicle mass fraction of the otherwise-lighter chemical vehicle more than the already-heavy NTP vehicle. Overall, for NTP to be the propulsion system of choice for a given mission, the ΔV requirement has to be high enough and the payload requirement high enough to make the trade of additional I_{sp} for additional stage dry mass worthwhile.

The higher cost of NTP means that it would be more attractive in a re-usable application where the cost could be amortized over multiple payload deliveries than in a single use case. It is worth noting that when considering refilling propellant tanks in a reusable vehicle that may be re-used within the Earth-Moon system, the amount of propellant mass needed for a refill mission will be an important parameter for multi-mission campaign design. Especially if the refill may take place in cis-Lunar space, the lift capability of available launch vehicles is severely limiting. Ability to maneuver with lesser propellant mass can be enabling.

MODELING APPROACH / METHODOLOGY

POTENTIAL CONOPS CONSIDERED

An NTP engine can benefit various mission CONOPS. This includes:

- Earth-Moon tug delivering payloads from LEO to LLO.
- Cis-lunar tug delivering payloads from NRHO to LLO or vice-versa.
- Earth departure stages for outer planet missions [10].

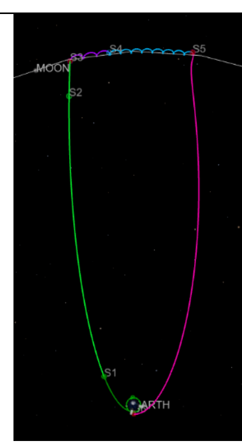
Hydrogen's low density compared to other propellant options results in most launch vehicles (LV) being volume limited, not mass limited, by an NTP system launching to a low Earth orbit (LEO). This means that within a LV fairing the volume occupied by the NTP stage is the limiting factor, and not the mass of the propellant, as is often the case for chemical propellant vehicles. To take full advantage of the LV capability, the NTP systems must in some cases be launched to a higher orbit than the minimum nuclear safe circular orbit.

Higher altitude starting orbits result in lower ΔV required for any Earth departure missions, whether Earth to Moon transfers or outer planet missions. Chemical stages performing similar CONOPS are limited by LV mass capability to orbits as low as 400 km circular. Launching to higher orbits would mean reducing the stage capabilities. Once an NTP stage is launched, it can be potentially re-filled, re-tanked, or docked to additional drop tanks. Any of these options allow a single main stage to be re-used for several tug missions. The highest starting orbits are possible if the main NTP stage with the engine is launched with minimal propellant, assuming secondary launches to fill the propellant tanks. Even lower ΔV missions are enabled in this use case, and each additional mission can be performed from the same orbits, assuming one propellant launch and one payload launch from Earth.

Table 2 outlines a round trip ΔV budget for a LEO to low Lunar orbit (LLO) mission. In this CONOPS, referenced throughout this paper, the tug's payload is launched separately from the tug itself. The first maneuver is the rendezvous and docking with the payload. The payload is taken to LLO and released. The tug vehicle returns to the Earth orbit to which it was initially launched and where it rendezvoused with the payload. The ΔV budget for trans-Lunar injection (TLI) and Earth orbit insertion (EOI) decreases as the launch orbit altitude increases resulting from lower mass NTP stages. For example, launching an empty core NTP stage, then utilizing a third launch (in addition to the dry vehicle and the payload) to fuel the stage will allow for a high-altitude initial orbit. This will result in decreased ΔV requirements and greater tug payload capability.

Table 2: Earth to LLO Re-usable Tug CONOPS and Example ΔV Budget from 1000 km Circular Orbit

Phase	Thruster Type	Magnitude of Δv (m/s)
1. Docking with Payload	RCS	90
2. Trans-Lunar Injection	Main	3,288
3. Lunar Orbit Insertion	Main	900
4. Orientation	RCS	50
5. Trans-Earth Injection	Main	927
6. Earth Orbit Insertion	Main	2,975



An NTP tug used in cis-lunar space could transport payloads from NASA's planned Gateway orbit, NRHO, to LLO. The ΔV between a reference Gateway NRHO trajectory [13] and low lunar orbit (LLO) of 100 km is approximated as 1640 m/s for the results below. An NTP tug will have much lower required propellant to make this trip than a comparable chemical propulsion tug. Several launches would be needed to provide the propellant to send a tug to NRHO and then re-fill its propellant tanks, so limiting the propellant mass needed to transport to NRHO is advantageous.

NTP stages could also be launched into elliptical orbits. These orbits are high energy and would take advantage of the Oberth effect on Earth-departure burns to reduce ΔV needs even further. The tradeoff in choosing such an orbit is the limited flexibility in launching payloads to rendezvous with the NTP stage. This may be acceptable for single Earth departure missions, but less desirable for a re-usable tug.

CRYOGENIC FLUID MANAGEMENT NEEDS

NTP systems typically require active Cryogenic Fluid Management (CFM) for the LH_2 propellant. Active CFM systems maintain the 20K set temperature necessary to keep the hydrogen as a liquid. This is especially important in timelines associated with the re-use CONOPS. To allow multiple trips between re-filling or re-tanking, active CFM is likely needed. However, in some cases where drop stages are more desirable, or only a single mission is being used for each stage, it may be possible to use only simple passive CFM. This may be more likely in Earth departure stages to outer planets, or for a mission CONOPS involving a single large payload delivery to lunar orbit.

It is possible to mix and match these CONOPS options in a re-use case. The main stage may utilize active CFM and be used for several missions, being augmented with passive CFM drop-tanks for some missions with larger payloads.

SPACECRAFT MODEL

A parametric spacecraft model was developed in MATLAB to automatically generate a wide variety of vehicles for the cis-lunar tug application. After a launch vehicle is selected a vehicle is generated such that the length is equal to the maximum allowable launch vehicle length capability (also referred to as volume limited) or the maximum launch vehicle mass capability (referred to as mass limited). The primary outputs are the dry and wet mass of the vehicle, and the maximum circular orbit that the launch vehicle can deliver the vehicle to. Three types of vehicles can be generated: Core, inline, and drop. The core has one or more engines, while the drop and inline exclude the engine and thrust structure and have a docking structure mounted on the side or top, respectively. The general designs were inspired by past multi-stage NTP vehicle design work by Aerojet Rocketdyne for NASA and ongoing work with NASA SNP [11].

The vehicle is divided into six subsystems: structures, avionics, propulsion, RCS, thermal, and power. Structures are sized based on expected mass and launch loads and design assumptions. Avionics has a fixed mass and power consumption value true for all vehicle configurations. RCS has a fixed mass for thrusters and piping, and an iteratively solved propellant tank to achieve the mission's RCS ΔV requirement. The thermal subsystem consists of radiator and cryogenic fluid management components, and are adjusted based on required thermal heat transfer, environmental heat loads, and CFM configuration. Power consists of solar arrays, batteries, and power management, which adjusts according to the power requirement of the vehicle and mission parameters such as expected time in shadow during low-Earth orbit. Finally, the propulsion subsystem consists of the engine, engine piping, and primary propellant tanks. Chemical or NTP propulsion choices will determine the mass and length of the engine, number of tanks required, and tank configuration. Figure 1 is an example vehicle layout within a launch vehicle fairing. The electronics, RCS, radiator, and solar panels are housed within the bus and docking structure.

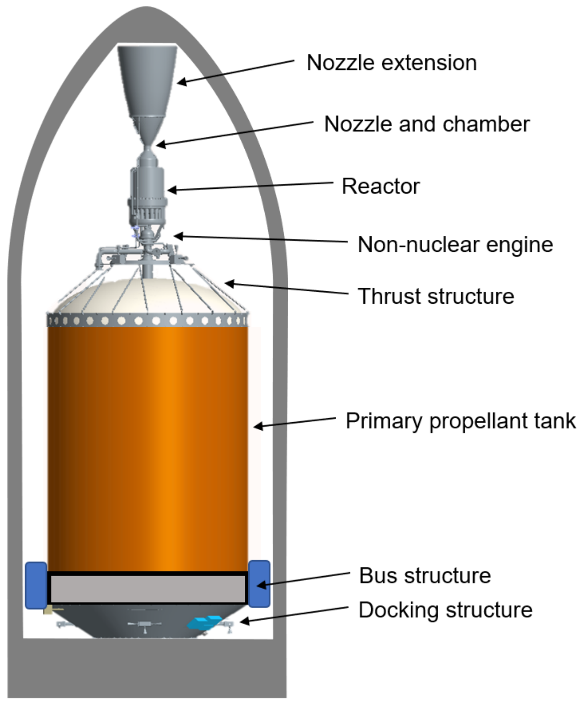


Figure 1: Example Volume-Limited NTP Core Vehicle Within Launch Vehicle Fairing

Due to interdependency between the vehicle dry mass, available propellant, and launch vehicle capability, there is a convergence process required, with the primary tank length iterated until convergence. The tank diameter is set to the maximum launch vehicle fairing diameter limit, and the length is adjusted until either the vehicle reaches the maximum length allowed, or the launch vehicle mass capability limit is reached. In the scenario where the volume limit is reached first, which is common for H-NTP, the launch vehicle insertion altitude is adjusted according to a mass versus altitude curve. The maximum altitude is selected to match the volume limited vehicle launch mass, taking full advantage of the launch vehicle's capabilities. Table 3 below lists the length stack-up of components for a core stage H-NTP vehicle.

Table 3: Volume-limited H-NTP Core Vehicle Length Stack-up within a SLS Block 2 Fairing

Component	Vertical Length [m]
Nozzle extension (stowed)	0
Nozzle and chamber	2
Reactor	2.5
Non-nuclear engine	2
Thrust structure	0.5
Primary propellant tank	17.75
Bus structure	0.5
Docking structure	1
Total length	**24.75**

If the vehicle reaches the launch vehicle mass limit before the volume limit the launch vehicle drop off altitude is decreased until the mass limit equals the volume limited vehicle launch mass, or a minimum limit of 400 km circular. The tank configuration can also be adjusted to either a cylindrical tank, two cylindrical tanks with a shared bulkhead, or spherical tanks. It's worth noting that the structural mass, RCS propellant tanks, and CFM heat load is adjusted during the convergence process as well. Further details about the vehicle model can be found in [6].

A large dataset of vehicles was generated for both core and drop stage vehicles. A summary of the vehicle options created are shown in Table 4, where all options were explored for each launch vehicle. It is important to note that at the time of analysis, Starship capability is not definitively known – a launch vehicle performance curve is approximated with ~108 t to 1000 km circular capability. This collection of core and drop stage vehicles are used in the mission trajectory and cis-lunar tug capabilities analyses.

Table 4: Trade Space of Vehicle Configurations Generated

Propulsion	CFM Options	Specific Impulse	Engine Mass	Number of Engines	Launch Vehicle
Hydrogen NTP	Active/ Passive	600 – 900 s	2525 - 5555 kg	1	SLS Block 2, Starship, New Glenn
Ammonia NTP	Passive	450 s	2525 - 5555 kg	1	
LOX/LH2 Chemical	Active/ Passive	466 s	359 kg	1-3	
LOX/CH4 Chemical	Passive	370 s	359 kg	1-3	

MISSION MODEL

An architecture model iterates the sequence of maneuvers as performed by a given vehicle concept, calculating the propellant for each, and converges to the payload capability of the vehicle for the given CONOPS. Alternately, the model can calculate the maximum ΔV potential of the vehicle with a given payload or can size the vehicle to provide a required ΔV budget with a given payload. For large payloads and ΔV budgets, multiple stages may be required; the model automatically converges to the require number of stages for a given mission within constraints (such as stage types and associated launch vehicles) provided by the user. The model can be run in two modes for simulating the ΔV maneuvers: an ideal rocket equation calculator with a user-defined I_{sp}, or a transient engine firing analysis tied to an engine cycle pseudo-transient performance database. The second method accounts for the I_{sp} variation over the start-up and shutdown engine operation with good precision. It also estimates the amount of cooldown propellant needed and the duration of cooldown flow.

A database of converged vehicle and mission results was created using the ideal rocket equation mode of the architecture model for multiple vehicle size constraints, active or passive CFM, multiple engine mass values, and over a range of engine I_{sp} values. The database primarily covers the Earth-LLO tug mission CONOPS, with drop-off of payload at LLO. A smaller number of cases were also converged for an NRHO-LLO tug to illustrate performance for a case that would be re-filled far from Earth.

A small number of cases were converged using the transient engine firing mode of the architecture model. These cases are used to illustrate the effective I_{sp} that results from accounting for the start-up, shutdown, and cooldown phases of each maneuver. The transient results combined with the database of ideal rocket equation results are used to illustrate the impact of variation in each of the performance KPPs for any effective I_{sp} achievable by the transient model.

RESULTS

Figure 2 illustrates the propellant mass fraction predicted for hydrogen NTP (H-NTP), LOX/LH2, and ammonia NTP (A-NTP) vehicles and the resulting ideal one-burn ΔV. The H-NTP vehicles plotted represent vehicles that are volume limited, whereas the A-NTP and chemical vehicles are constrained by the launch vehicles' lift capability to 400 km circular orbit. This plot assumes an H-NTP effective specific impulse of 875s, an optimistic assumption that effectively neglects transient and cooldown effects. The A-NTP vehicles have an assumed 370s effective specific impulse and an engine mass of 2525 kg. The chemical engine effective specific impulse is assumed to be 466s. The H-NTP vehicle has four data points to represent an engine mass range of 2525 kg, 3535 kg, 4545 kg, or 5555 kg. The series labeled "Ref." represents the ΔV capability of the H-NTP vehicle plus the additional ΔV provided by the launch vehicle as the H-NTP vehicles can be placed in a much higher orbit than 400 km before reaching the launch vehicle mass capability. The higher orbit for a Starship-launched vehicle may require that it be used in a disposable mode. A reusable H-NTP vehicle returns to the higher altitude for the "Adj." cases. This is done to take advantage of the launch vehicle's capability to supply propellant at a higher orbit as a hydrogen refill vehicle will encounter launch vehicle volume capability limitations before mass limitations.

Unsurprisingly, the more capable the launch vehicle, the larger the NTP vehicle and thus the better the propellant mass fraction that can be achieved. Chemical vehicles have much higher propellant mass fractions than do H-NTP vehicles, due to the much higher density of liquid oxygen, which represents most of the chemical propellant mass. The higher density results in a much smaller tank to store a greater mass of propellant. As can be seen from the magnitude of the ideal ΔV possible with each stage, the chemical Starship-sized vehicle outperforms the H-NTP Starship-sized vehicle for high H-NTP engine masses but is outperformed by low H-NTP engine masses when accounting for a higher drop off orbit. The SLS-sized H-NTP vehicles do relatively better compared to SLS-sized chemical vehicles because the additional fairing volume favors low-density hydrogen. A New Glenn H-NTP vehicle also outperforms the chemical New Glenn vehicle because of its relatively low mass limitations but high fairing volume. The A-NTP vehicles have similar propellant mass fractions as the chemical vehicles but with a monopropellant and have lower performance than both chemical and H-NTP by about 20%.

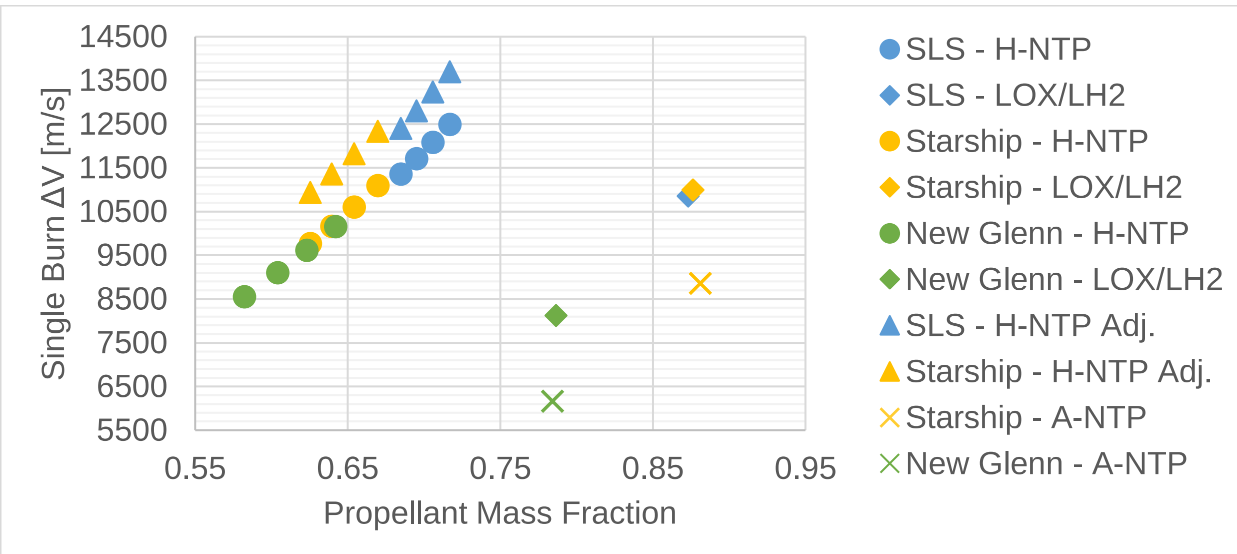


Figure 2: Propellant mass fraction and resulting ideal one-burn ΔV with no payload for a range of core NTP and chemical vehicles from the spacecraft model.

Figure 3 illustrates the payload capabilities of vehicles launched with Starship with various NTP engine masses and effective I_{sp} in terms of separately launched payload mass for the Earth-LLO mission concept. These results include gravity losses and RCS maneuvers. Also plotted are two comparable LOX/LH2 vehicles. The difference in capability between the 1-engine and 2-engine variants results from the difference in thrust and thus gravity losses between the two points. All cases assume 25 klb_f thrust per engine. The H-NTP engine is assumed to have an effective I_{sp} of 839 seconds. An 839 second effective I_{sp} is representative of what can be expected when accounting for transients and cooldown with a 900 second steady-state operation. (See section Transients and Cooldown below). The altitude that the vehicle is dropped off at by the launch vehicle varies based on if the vehicle is volume or mass limited.

Separately launched payload mass spanned the range of roughly 10 tons to 28 tons, with H-NTP performing both better or worse than the chemical equivalent vehicle depending on how optimistic the assumptions for the H-NTP engine are. Since the comparison between H-NTP and chemical vehicles is close for this reference mission, the trends in the plot should be interpreted as the important point. Whether a particular vehicle performs better than another will depend on the details of the design and mission requirements. For this example, H-NTP has similar performance to LOX/LH2 when the engine mass is low and the effective specific impulse is near the expected effective specific impulse for a 900 second nominal NTP engine.

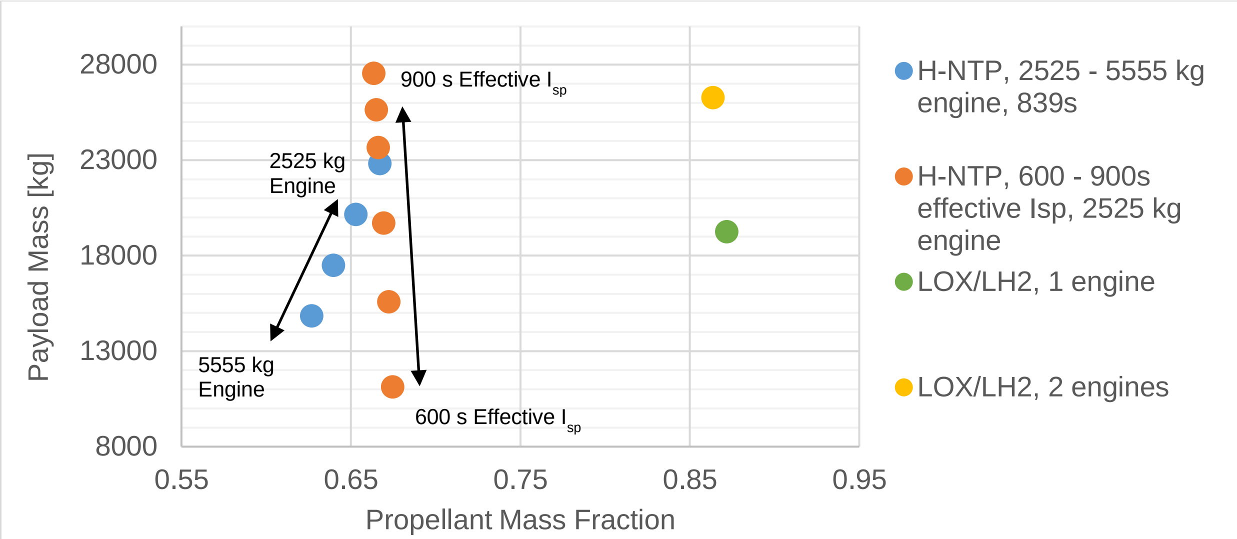


Figure 3: Payload for an Earth-LLO Starship launched tug and corresponding propellant mass fraction for various engine masses and effective I_{sp}

An alternative application of a reusable cis-lunar tug could be to transport payload between NASA's planned Gateway and a circular lunar orbit. As described in the CONOPS section, the ΔV between a reference Gateway NRHO trajectory [12] and low lunar orbit (LLO) of 100 km was approximated as 1640 m/s per direction. Listed in Table 5 is the separately launched payload mass that can be transported per trip such that the vehicle propellant tank will be empty after two roundtrip missions. The vehicles are fully filled at NRHO and match the vehicles created for Figure 2 with an H-NTP effective specific impulse of 839 seconds, an A-NTP effective specific impulse of 370 seconds, and a LOX/LH2 effective specific impulse of 466 seconds.

While the payload mass capability varies between launch vehicle options, the benefits of an H-NTP vehicle can be seen in the required propellant mass. The payload mass per propellant mass ratio is at least double for the H-NTP vehicles, which can increase in impact as the payload mass or number of roundtrip missions anticipated increases. Supplying propellant to NRHO will require many separate launches, so while the initial mass of an H-NTP vehicle is higher the long-term operating logistics of a reusable tug may favor H-NTP. The A-NTP vehicles had lower payload capability and higher propellant mass when compared to LOX/LH2 vehicles.

Table 5: NRHO to LLO and Back Tug H-NTP / LOX LH2 comparison

System	2 round trips Payload	2 round trips propellant mass	8x round trips propellant mass	Payload mass per propellant mass
H-NTP from New Glenn	9,542 kg	25,141 kg	100,566 kg	9.5%
LOX/LH2 from New Glenn	3,399 kg	31,119 kg	124,475 kg	2.7%
A-NTP from New Glenn	Cannot complete mission			
H-NTP from Starship	15,130 kg	34,200 kg	136,801 kg	11.1%
LOX/LH2 from Starship	24,233 kg	113,524 kg	454,095 kg	5.3%
A-NTP from Starship	11,358 kg	112,734 kg	450,934 kg	2.5%

SPECIFIC IMPULSE AND ENGINE MASS

Specific impulse and engine mass both have appreciable impacts on the payload mass capability that an NTP vehicle can deliver in the Earth-LLO reusable tug mission. Figure 4 illustrates that a difference of 3030 kg in the engine mass KPP affects the payload capability roughly the same amount as a difference of 90 seconds effective I_{sp}. Figure 5 illustrates the same comparison for a larger-sized vehicle with a higher propellant mass fraction. The impact of engine mass and I_{sp} is a lower percentage of the payload compared to the smaller vehicle due to engine mass being a smaller percentage of vehicle dry mass and a higher propellant mass fraction. Since the propellant mass fraction is higher, the difference in payload corresponding to the same 3030 kg difference in engine mass is equal to the impact of slightly less than 90 seconds I_{sp}.

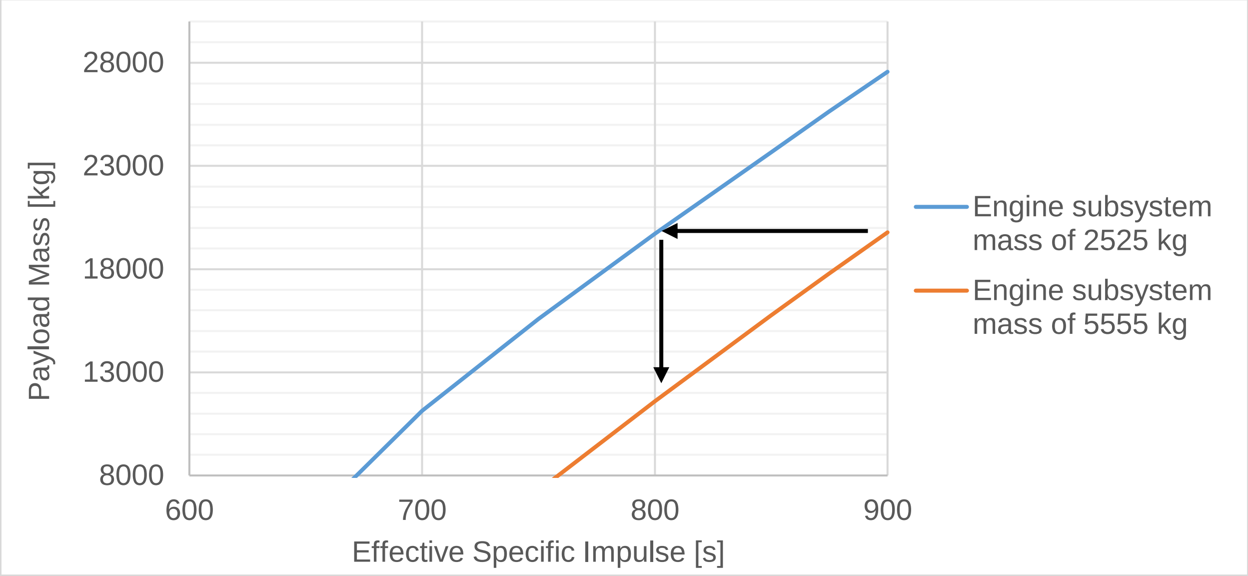


Figure 4: Payload mass vs. effective specific impulse for a Starship-launched NTP Core Stage, active CFM, 25 klb_f, altitude-optimized starting orbit.

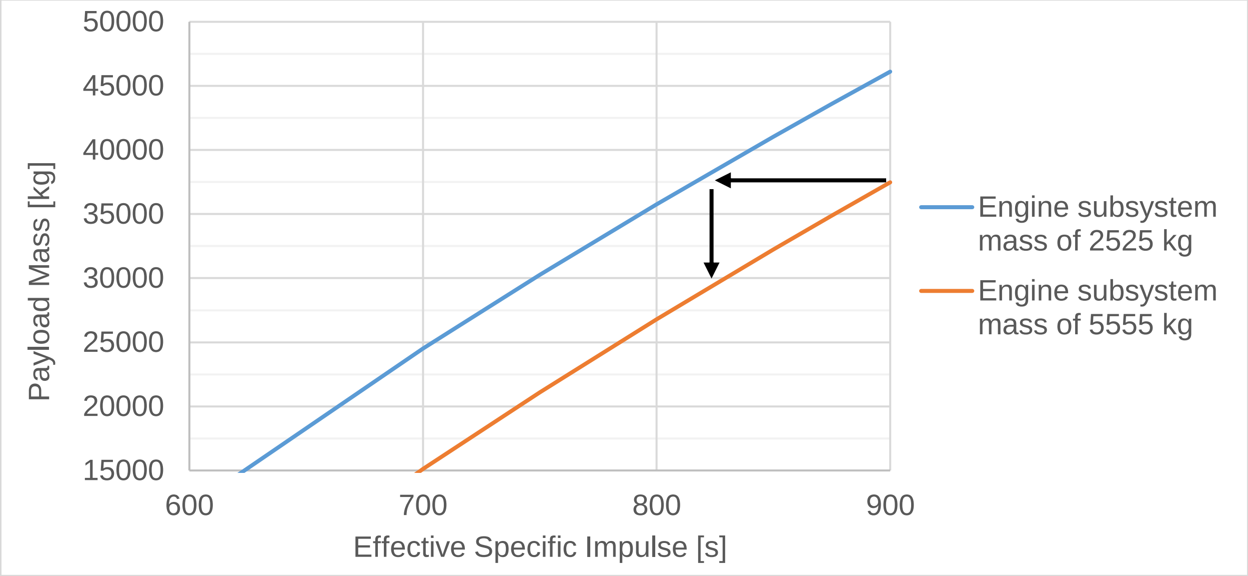


Figure 5: Payload mass vs. effective specific impulse for a ‘SLS B2’-launched NTP Core Stage, active CFM, 25 klb_f, altitude-optimized starting orbit.

Further exploring the impact of I_{sp} and engine mass KPPs over the range of interest, Table 6 applies weights to each KPP, with scores normalized to the lowest payload capability scenario of a specific impulse of 750 second and an engine mass of 5555 kg. The scores reported in this table represent the relative merit of engines with the KPP values as measured against the payload mass figure of merit. The scores show that developing an engine with 900 second I_{sp} rather than 700 second is more important than reducing the mass of an engine from 5000 to 2500 kg. A weighted comparison relative to the payload mass of a LOX/LH2 core stage on Starship is shown in Table 7, illustrating that both targets are likely required to outperform chemical vehicles on a straight payload mass basis for this mission. The specific scores in the table vary depending on the size of the vehicle being considered and the specific mission, but the trend shown will remain consistent.

Table 6: Weighted scores for different combinations of engine mass and I_{sp} for H-NTP Core on Starship

Engine Mass	I_{sp} 750 s	I_{sp} 900 s
5555 kg	1	2.7
2525 kg	2.1	3.8

Table 7: Weighted scores for different combinations of engine mass and I_{sp} for H-NTP core on Starship compared to 1-engine LOX/LH2 core stage

Engine Mass	I_{sp} 750 s	I_{sp} 900 s
5555 kg	0.4	1.0
2525 kg	0.8	1.4

THRUST

An H-NTP core stage has a much higher dry mass compared to a chemical stage of equivalent ΔV due to the reactor mass. For the reference mission considered, the propellant and payload masses can be greater than 10 times the engine mass. This thrust to weight ratio (TW) can have a significant impact on the ΔV budget of the mission due to gravity losses encountered during the burns. The first maneuver in an Earth-LLO tug mission is a trans-lunar injection (TLI) that requires at least 2 km/s of ΔV. Depending on the wet mass of the NTP vehicle and attached payload, performing this burn with 25 klb_f Thrust requires burn times in excess of 15 min and thus large gravity losses. For a Starship H-NTP core stage a thrust of 50 klb_f will reduce losses to less than 10%. Splitting the TLI into two burns is possible, but doing so results in a reduction of the effective Isp of the maneuver due to the added transients and increased cooldown requirement for two separate burns with an NTP engine.

Figure 6 illustrates the effect of thrust on separately launched payload mass for a Starship-launched H-NTP vehicle in for a variety of possible engine masses. For this example, the effect of doubling the thrust from 25 klb_f to 50 klb_f affects the delivered payload a similar amount as a difference of 600 kg engine mass. For larger vehicles and payloads, the effect of thrust is greater than the effect of engine mass.

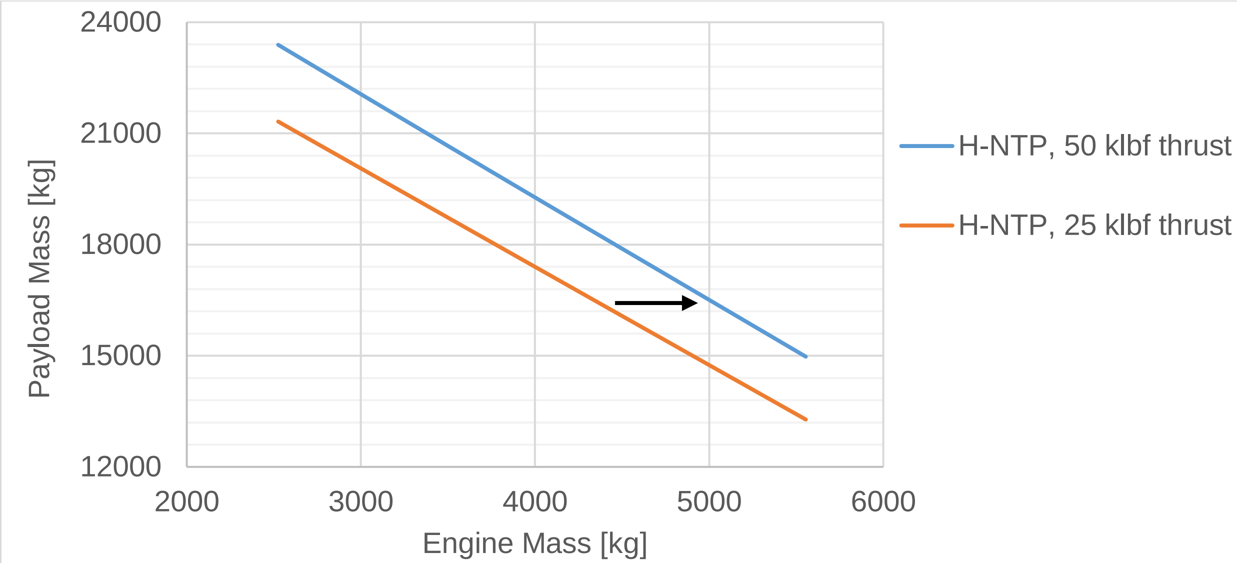


Figure 6: Effect on payload mass of various engine masses at two thrust levels.

For the TLI maneuver in the LEO-LLO mission the propellant tank is full and the payload attached, This results in TLI having the lowest T/W of all maneuvers and being most affected by gravity losses. Plotted in Figure 7 is the additional ΔV required compared to an instantaneous maneuver for TLI as a function of vehicle T/W. Specific example T/W values are labeled for a Starship core stage with the maximum payload mass included. The H-NTP engine is assumed to be 2525 kg each. The chemical vehicles incur higher gravity losses due to the much higher wet mass of the vehicle when compared to the H-NTP vehicles. Adding an engine increases the thrust from 25 klb_f to 50 klb_f and reduces in a substantial reduction in gravity losses for the chemical vehicle, and reasonable improvements for H-NTP. These results suggest that if a reusable tug may benefit from a thrust level higher than 25 klb_f.

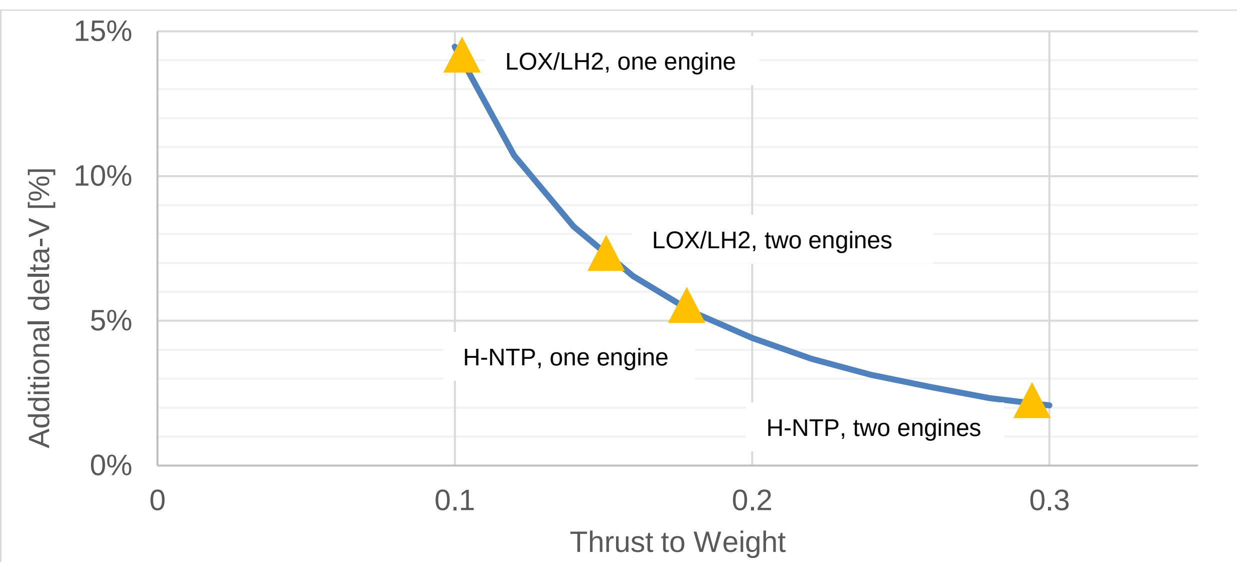


Figure 7: Additional ΔV required due to gravity losses for the TLI burn

ENGINE BURN TIME

The engine burn time (amount of time that the engine produces thrust) is another significant metric derived from the mission con-ops and vehicle thrust-to-weight ratio. In the case of a NTP engine it affects the lifetime of the fuel and the amount of cooldown required after the burn. Table 8 lists the ΔV and burn times for one cis-lunar tug mission, where the tug completes a round trip between the Earth and the Moon starting at an optimized orbit of 4058 km circular and transporting payload to a 100 km circular lunar orbit. The vehicle is an H-NTP core vehicle launched with Starship transporting the maximum payload mass with a fully filled propellant tank and an effective specific impulse of 839s. The first burn time is largest at nearly 23 minutes and experiences the highest gravity losses.

Table 8: Earth-Moon Tug Burn Times for an H-NTP Core Vehicle with Payload

Primary burn name	ΔV [m/s]	Time [s]
TLI	2645	1408
LOI	876	372
TEI	812	140
EOI	2501	352

TRANSIENT AND COOLDOWN

Each NTP engine burn has start-up, steady-state, and shutdown phases. During start-up and shutdown, the power level of the reactor is quickly ramping up or down (roughly to full power from zero in 30 seconds) [13]. As the power level rises (or drops) the temperature of the fuel and thus the flowing hydrogen is less than the steady-state temperature. This results in a reduction of the I_{sp} that can be achieved on average over the burn, as illustrated in Figure 8 [14].

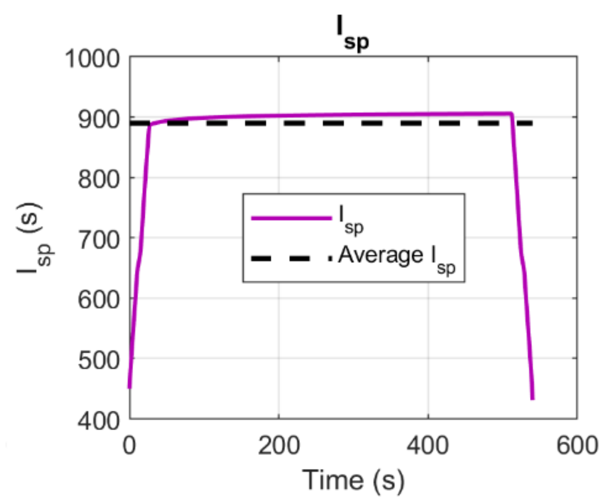


Figure 8: Example engine burn with start-up and shutdown.

Cooldown uses propellant at a low flow rate to maintain acceptable temperatures in the reactor after shutdown as fission products decay and release heat. Acceptable temperatures are important to avoid damage to the moderator that may otherwise prevent subsequent restarts. The amount of cooldown propellant required can be estimated by Equation 1.

$$m_{prop\ cooldown} = \frac{\int_{t_{sd_i}}^{t_{sd_f}} \dot{Q}_{fp}\left\{\frac{\rho}{\rho-\beta}\exp\left(\frac{\rho-\beta}{\Lambda}t\right)-\frac{\beta}{\rho-\beta}\exp\left(\frac{-\rho\lambda}{\rho-\beta}t\right)+0.1104\left[t^{-0.2436}-\left(t_{fp}+t\right)^{-0.2436}\right]\right\}dt}{c_p(T_o-T_i)} \quad \text{Eq. 1 [15]}$$

Table 9 shows that utilizing the cooldown specific impulse can have appreciable effects on average specific impulse. Some of the long duration maneuvers, such as TLI and TEI, where there is a long duration after shutdown when additional impulse can be applied to helpfully increase velocity, can incorporate cooldown impulse into the maneuver design. Other maneuvers, such as LOI, where the orbital period is short and a payload drop-off maneuver is desired, may not be able to beneficially take advantage of cooldown impulse. The effective Isp found here for TLI is consistent with analysis performed by others for an Earth-departure vehicle [10].

Table 9: Effective I_{sp} as including cooldown for a Starship-sized NTP vehicle

Maneuver	ΔV	Propellant Mass Used	Cooldown Propellant Mass	Non-Propellant Mass	Steady-State I_{sp}	Including Transient I_{sp}	Effective I_{sp} no cooldown impulse	Effective I_{sp} accounting cooldown impulse
TLI	2597	17474	1320	36945	900	894	**820**	**856**
LOI	889	4810	438	36945	900	881	**803**	**803**
TEI	813	1790	159	12782	900	854	**780**	**818**
EOI	2446	4271	391	12782	900	880	**794**	**837**
Average						887	**800**	**839**

Table 10 shows the partials of payload mass with respect to each KPP. I_{sp} is the most important KPP. The sub-KPPs of transient rate and cooldown per impulse are included, but technology development is not at this phase focused on directly improving them – to greatly affect cooldown mass would require significant design change (such as implementing bi-modal NTP-electric generation). The transient rate will be as fast as the materials that make the nominal I_{sp} possible will allow. Mass and thrust have the relative effect on payload discussed before, and while thrust is in the trade space, partials like these should be considered for a variety of missions.

Table 10: Partial derivatives of payload with respect to each KPP

KPP	δpayload / δKPP	With comparable magnitudes	Summary of Importance
Effective I_{sp}	80 kg / s	8 mt / 100 s	Highest
Mass	2.5-3 kg / kg	2.5-3 mt / 1000 kg	High
Thrust	0.3-0.4 kg / lb_f	3-4 mt / 10 klb_f (12.5 - 30 klb_f range)	Med
Transient Rate	Contributes to effective I_{sp}		Low
Cooldown average specific impulse	Contributes to effective I_{sp}. ~7 kg / s cooldown average I_{sp}		Med

SUMMARY AND CONCLUSIONS

NTP's high I_{sp} has the potential to make cis-Lunar operations much more efficient by limiting the amount of propellant required. The benefits for tug operations over a similarly-sized chemical vehicle are marginal for an Earth-Moon tug CONOPS due to the higher engine mass and tank mass per propellant mass required for a hydrogen NTP vehicle. However, when considering multi-use vehicles operating in cis-Lunar space (e.g., between NRHO and LLO), the amount of propellant mass required to complete tug missions is significantly less for an H-NTP vehicle than for a chemical vehicle. This lower propellant mass requirement could translate directly to fewer launches required to sustain the logistics of a sustainable cis-Lunar presence. Ammonia NTP vehicles can also provide significant ΔV performance, but with high propellant mass required due to the lower I_{sp}.

The relative importance of NTP engine I_{sp}, engine mass, and thrust has been assessed for tug CONOPS involving an Earth departure. Effective engine I_{sp} over the mission CONOPS is most important in the range of possible values considered; however, the engine mass has enough of an effect to determine the usefulness of NTP compared to a chemical system for a given mission. Thrust in the 25 klb_f class (rather than 12.5 klb_f) is desirable to limit losses and long burn times for Earth-Moon operations. The importance of considering both transient performance and cooldown mass requirements in the use of an NTP engine has been shown. These results can help in the evaluation of the relative merit of alternate engine concepts being considered for development, and in the assessment of risks in development against achieving mission-desired performance.

FUTURE WORK

There are many more mission CONOPS that can be assessed using the tools that produced the results presented. Variations on the CONOPS of re-usable vehicles operating between LEO/MEO and cis-Lunar orbits (LDHEO, NRHO, LDRO, LLO) should be investigated. The propellant logistics for a sustained Lunar presence can be investigated, including identifying the best CONOPS for providing propellant to cis-Lunar operating vehicles. This may be some combination of launch vehicles with on-orbit re-fueling as envisioned for Starship, or NTP or chemical vehicles operating as tugs. The significant propellant mass reduction resulting from use of H-NTP for cis-Lunar operations should be considered in full.

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REFERENCES

[1] L. Hall, "NASA, DARPA Partner with Industry on Mars Rocket Engine," NASA, 26 July 2023. [Online]. Available: https://www.nasa.gov/technology/nasa-darpa-partner-with-industry-on-mars-rocket-engine/.

[2] S. K. Borowski, D. R. McCurdy and M. L. Burke, "The Nuclear Thermal Propulsion Stage (NTPS): A Key Space Asset for Human Exploration and Commercial Missions to the Moon," in *AIAA SPACE*, San Diego, 2013.

[3] C. B. Reynolds, C. R. I. Joyner, T. Kokan, D. J. H. Levack and B. J. Musek, "NTP Robustness for Mars Conjunction and Opposition Class Missions," in *ASCEND 2020*, Virtual Event, 2020.

[4] C. B. Reynolds, C. R. Joyner, T. S. Kokan, D. J. Levack and B. J. Muzek, "Mars Opposition Missions Using Nuclear Thermal Propulsion," in *AIAA Propulsion and Energy*, Virtual, 2020.

[5] C. Foulds, M. Eades and P. Venneri, "A Versatile Minimum Viable Multi-Launch NTP Architecture for Space Exploration," in *Rocky Mountain AAS GN&C*, Breckenridge, 2024.

[6] C. Harnack, E. Grella, S. Greenhalge and M. Duchek, "Capabilities of Single Launch NTP and Chemical Spacecrafts for Cis-Lunar Tug Missions," in *Nuclear and Emerging Technologies for Space*, Idaho Falls, 2023.

[7] C. B. Reynolds, F. J. Horton, C. R. Joyner, T. Kokan and D. Levack, "Applications of Nuclear Thermal Propulsion to Lunar Architectures," in *AIAA Propulsion and Energy Forum and Exposition*, Indianapolis, 2019.

[8] C. B. Reynolds, C. R. Joyner, T. Kokan and D. Levack, "Nuclear Thermal Propulsion vs. LOX/LH2 Chemical Propulsion for Cislunar Architectures," in *Nuclear and Emerging Technologies for Space*, Idaho Falls, 2023.

[9] D. Nikitaeva, C. D. Smith and M. Duchek, Engine Cycle Comparison for Alternative Propellant Nuclear Thermal Propulsion Engines, Las Vegas, Nevada: https://doi.org/10.2514/6.2023-4715, 2023.

[10] C. Foulds, M. Eades, V. Patel and P. Venneri, "Fast Cadence Commercial Multi-Launch NTP Vehicle with Combined Boil-Off and Zero Boil-Off Tank Systems," in *Nuclear and Emerging Technologies for Space*, Sante Fe, 2024.

[11] M. G. Houts, C. R. Joyner, T. S. Kokan, B. Reynolds, J. Abrams, M. Eades, D. Beale, C. Denbrock and J. Easley, "Design Options for a Versatile Nuclear Thermal Propulsion (NTP) Stage," in *AIAA SPACE*, Orlando, 2018.

[12] D. Lee, "Gateway Destination Orbit Model: A Continuous 15 Year NRHO Reference Trajectory," NASA, 2019.

[13] F. B. Damerval, "XE-Prime EP-7A Startup Tests," Spear Report, Jackass Flats, NV, 1969.

[14] M. E. Duchek, D. Nikitaeva, C. Harnack, E. Grella and S. Greenhalge, Parametric Modeling of NTP Engine Performance for a Crewed Mars Mission, Las Vegas, Nevada: https://doi.org/10.2514/6.2023-4717, 2023.

[15] W. J. Emrich, Principles of Nuclear Rocket Propulsion, Kidlington, Oxford, United Kingdom: Butterworth-Heinemann, 2016.

[16] C. R. Joyner, D. J. Levack and J. Crowley, "Determining Mars Mission NTP Thrust Size and Architecture Impact for Mission Options," in *51st AIAA/SAE/ASEE Joint Propulsion Conference*, Orlando, 2015.