

Parametric Modeling of NTP Engine Performance for a Crewed Mars Mission

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Nuclear thermal propulsion (NTP) has been studied extensively for use on a crewed mission to Mars, and NASA and DARPA are partnering in the development and demonstration of an NTP engine as part of the DRACO project. The specific impulse and thrust achieved by an engine in the context of a mission determines the propellant amount required to close the mission and the size of the vehicle required. A NTP engine has a slower start-up/shut-down transient than traditional chemical engines and requires post-firing cooling, and thus the average I_{sp} for a burn is a function of the power ramp-up rate and the duration of the burn. An initial demonstration engine may target a lower I_{sp} than the 900 sec that has been extensively studied by NASA. This paper shows the effect on the mission closure and vehicle sizing of the transient ramp rate performance and of lower than 900 sec specific impulse. Ammonia is considered as an alternate NTP propellant, with mission performance estimated.

I. Background

Over the past 50 years, Nuclear Thermal Propulsion (NTP) has been developed inconsistently starting from the ROVER/NERVA program, and later Timberwind [1, 2]. Assessment of NTP as an option for a crewed Mars mission continued with the DRA 5 Addendum 2 [3]. NASA and Aerojet Rocketdyne (AJRD) have been studying NTP-powered crewed Mars missions through the Game Changing Development (GCD) NTP project and now the Space Nuclear Propulsion (SNP) project [4]. NASA has also studied NTP for crewed Mars missions in the Mars Transportation Assessment Study [5]. The authors of this paper have been supporting the project with system-level modeling and parametric modeling of NTP systems for use in cis-Lunar space [6]. Recently, DARPA and NASA have partnered to pursue development of a NTP engine for an initial in-space demonstration, but ultimately extensible to the crewed Mars mission.

II. Introduction

A NTP engine has a few key performance parameters (KPPs) that are indicators of its ability to close a mission and the performance of the spacecraft in executing that mission. For a crewed Mars mission, the initial mass and volume of the spacecraft are critical parameters, as they essentially determine the number and type of launches required and indirectly determine the cost of the spacecraft (apart from propulsion development costs).

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The engine key performance parameters are:

- Steady-state specific impulse (I_{sp})
- Start-up power ramp rate
- Shutdown power ramp rate
- Engine thrust
- Engine weight

The first three together determine the average I_{sp} achievable for a given engine burn. The engine thrust and weight together with the effective tankage mass parameter (mass of tank and propellant management systems per mass of propellant) and the average I_{sp} determine the vehicle thrust to weight. The vehicle thrust to weight, I_{sp} , and the change in velocity (ΔV) needed to complete a mission maneuver, determine the propellant requirement for the maneuver. For a given mission opportunity and mission design (e.g., payload mass, stay time, transfer durations, flyby sequence), the key performance parameters determine the total vehicle/architecture mass needed to close the mission.

Past research has investigated the performance of NTP for the crewed Mars mission, with various assumptions for I_{sp} , thrust-to-weight ratio, tankage mass fraction, etc. This paper aims to answer a specific set of questions that have received less attention in the past and that are useful for setting requirements or KPP targets for the development of a NTP engine system applicable to the crewed Mars mission. The questions are:

1. What is the impact on the mission of less than 900 sec average I_{sp} ?
 - Resulting from startup/shutdown profile of a burn
 - Resulting from lower peak temperature limit in the reactor (e.g., 700-850 sec I_{sp} of a potential demonstration mission)
 - Resulting from decay-heat removal H_2 flow
2. What is the impact of using a different propellant (such as ammonia)?
 - Can ammonia alternative propellant-NTP (A-NTP) close the same Mars mission opportunities?
 - How does ammonia A-NTP compare to hydrogen-NTP (H-NTP) for the Mars mission?

III. Models Used and Methodology for Analysis

A. Mission & Architecture Models

The mission and campaign model includes the trajectory and optimization of the initial mass of the vehicle based on the ΔV needed to close the mission. There are multiple assumptions that affect the model and the results. These include the type of mission (opposition or conjunction-class, Venus flyby options, abort scenarios), the mission opportunity (e.g., 2039, 2041), the aggregation orbit for a multiple-stage vehicle, constraints on the Earth departure orbit (e.g., requirement to rendezvous in near-rectilinear halo orbit (NRHO) with a habitat module), the mass of payload to be delivered to Mars and back to Earth, any constraint on maximum transit or total mission duration, and any constraint on the duration of operations in the Mars sphere of influence (SOI), among others. The analysis in this paper is based on a set of these assumptions, provided in Table 1.

The vehicle I_{sp} and mass assumptions were taken from Aerojet Rocketdyne's work on opposition class missions [4]. The mission opportunities represent two different mission concepts where a Venus flyby is used in the inbound or outbound segment of the mission. For the analysis purposes of this paper, the stages are sized for both mass and volume for a H-NTP vehicle that would be aggregated in low Earth orbit (LEO), but the mission is evaluated starting at NRHO. The stage types used in this analysis are an inline stage that is carried throughout the mission, core stage that has 2-6 1000-5000 kg NTP engines, and drop tanks that can be resized and incrementally added to fulfill the mission ΔV . Another assumption for this analysis is that either extra vehicle components such as strap-on stages and additional drop tanks are leveraged for raising from LEO to NRHO or filling of partially filled stages in NRHO by commercial launch vehicles occurs. By evaluating the vehicle at NRHO, this paper can evaluate engine performance parameter importance making fewer assumptions about aggregation and LEO operations.

The stack-up of stages assumed in this paper for the following results is similar to vehicle architectures examined by NASA in the Mars Transportation Assessment Study (MTAS) [5], but not exactly the same. MTAS included additional elements to the stack (a power and propulsion module and second truss section) not included in the reference stack for this paper. It does not represent the most optimal configuration, as finding the optimum is not the focus of this work. Rather, this work assumes a reference vehicle to illustrate the impact of variations in engine performance, for the purpose of informing key performance parameter requirements for engine development. The exemplary mass

values for the vehicle architecture should not be used to compare an NTP engine to other propulsion systems, as further work is needed to perform a “best-on-best” or “apples-to-apples” comparison. The mass values here should be interpreted to inform about the relative importance of the engine parameters in the range of performance considered.

The 2039 opportunity is a more challenging mission to close and thus encompasses many opportunities across the synodic cycle. This is not necessarily the mission that NASA is most likely to design for; however, the mission does make a good reference case in which to illustrate the impact of engine performance.

Table 1. Mission analysis ground rules and assumptions.

Ground rule or assumption	Parameter value
Mission class	Opposition
Mission year	2039
Mars SOI Stay Duration	50 days
Total Trip Duration	729 days
Starting and Ending Orbit	Starting: NRHO, Ending: (401 km x 317,000 km)
Mars Parking Orbit	5 Sol (250 km x 112,000 km)

Table 2. Vehicle assumptions. [4]

Ground rule or assumption	Parameter value
RCS I_{sp}	329 sec
OMS I_{sp}	450 sec
Inline Inert Mass	15720 kg
Drop Tank Inert Mass	8727 kg
Core Stage Inert Mass	15720 kg

Copernicus, a NASA developed tool for spacecraft trajectory design and optimization, was used to analyze different missions to determine timelines and delta-V needs. Copernicus models demonstrate the effects of varying payload and spacecraft sizes, by solving trajectories to allow for various mission needs. For example: minimizing ΔV of a given mission segment to maximize propellant available for a following engine burn, minimizing mission duration based on available propellant, or identifying feasibility of mission architectures. In addition, Copernicus was used to calculate gravity losses based on spacecraft size and parameterized thrust and I_{sp} .

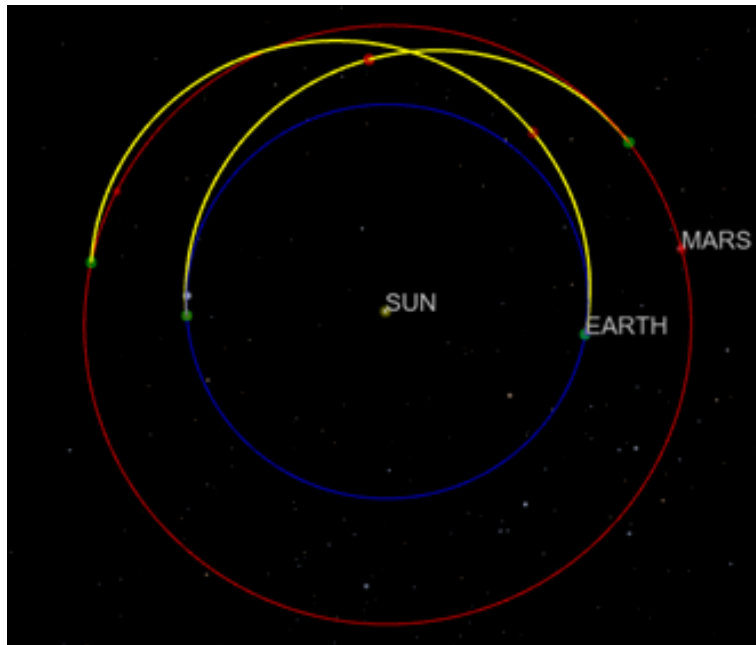


Figure 1: Example of Mission Model from Copernicus for a Conjunction Class Mission

B. Spacecraft Architecture Sizing Model

The spacecraft architecture sizing model uses a physics-based first principles approach instead of the ideal rocket equation. This involves tracking the vehicle propellant mass and ΔV using time steps and tracking the engine thrust and propellant mass flow rate. The outputs of the engine model, described in detail in another paper [7], are used to extract the engine performance at different operating conditions. This time step approach unlocks the capability to assess NTP engine transients, as well as the resulting decay heat based on the total burn time of each major thrusting maneuver. Due to the inherently dynamic nature of NTP engine transients caused by the slow start-up and shut down times in comparison with chemical rocket engines, the burn time is varied until ΔV convergence is achieved with respect to the desired values obtained from a model in Copernicus and validated against Aerojet Rocketdyne's work [4]. The last burn exhausts all the remaining propellant regardless of the resulting ΔV , after which the total propellant mass is scaled based on the total vehicle ΔV and the mission is performed again until propellant mass convergence is reached when the resulting vehicle ΔV matches the desired ΔV . [8] This methodology was developed and thoroughly described in previous work [9].

Although convergence is rapid when single staged vehicles are scaled, due to launch vehicle payload capability limitations which include a maximum propellant tank volume, it does not accurately represent reality. Therefore, the total propellant mass must be broken up into multiple stages, each with its own RCS mass and inert mass to manage the on-board propellant. Figure 2 shows the flow of the vehicle analysis model where the Reference Vehicle Class is fed into the model which specifies the vehicle components, and the mission structure represents the architecture of all the different ΔV s and payload mass drop offs. For this purpose, at the end of each full mission simulation iteration, the total propellant mass is divided into several tanks which are equally scaled. If it so happens that more propellant than the volume limitation is needed, then an entire stage is added and the tanks are rescaled. Similarly, if the propellant is scaled down enough to be able to eliminate an entire stage, then a stage is dropped, and the tanks are again resized to accommodate the total propellant volume. When the vehicle stages are varied, their associated corresponding additional masses are also varied. The tank mass is calculated using the tank mass correlations from Eq. 1 for hydrogen (multi-layer insulation is considered) and Eq. 2 for ammonia [3].

$$m_{LH_2 \text{ tank}} = 9.09V + 2.88 \left(\frac{V}{4\pi/3} \right)^{1/3} \quad \text{Eq. 1}$$

$$m_{\text{other tank}} = 12.16V \quad \text{Eq. 2}$$

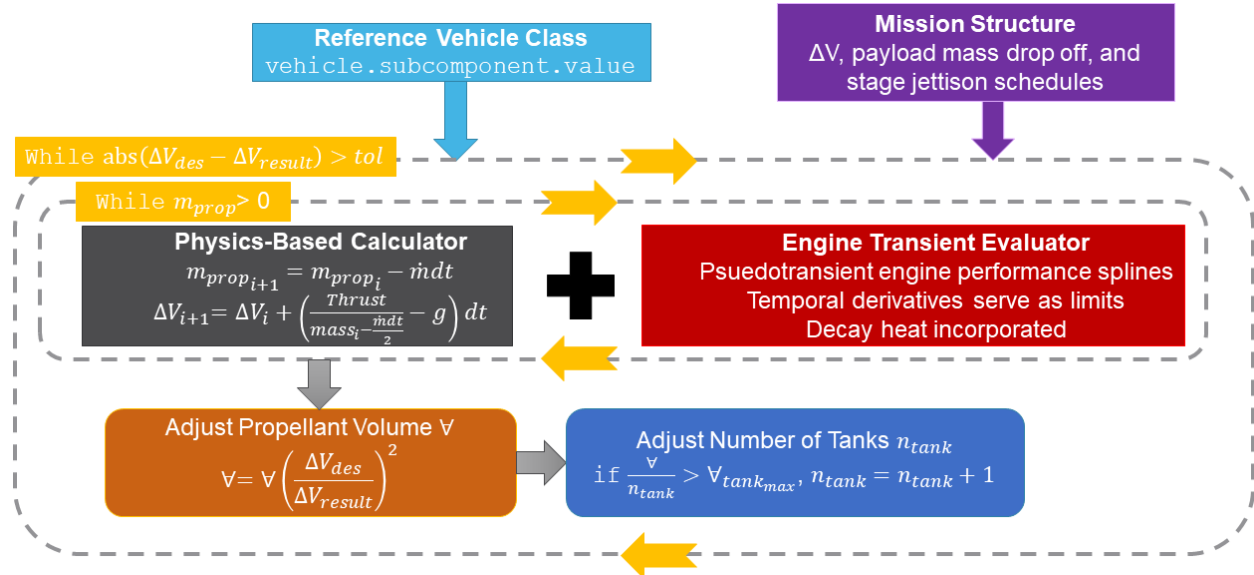


Figure 2: Spacecraft Model Code Architecture

C. Finite Burn Gravity Losses Methodology

A suite of Copernicus decks was created for use in tandem with the architecture sizing model to investigate gravity losses. Gravity losses are defined as the additional impulse, in excess of that required for an impulsive burn, required by a spacecraft to achieve a targeted v_∞ . This additional impulse is required due to the engine burning on-axis with the nearby planet's gravitational vector. In **Error! Reference source not found.**, the red lines represent the engine burn vector. This figure demonstrates how the burn does not occur directly perpendicular to the gravity vector, as in an ideal impulsive burn at periapsis. Spacecraft with low thrust to weight ratios are particularly susceptible to these effects, as they require a longer burn time to accelerate to the required velocity and therefore are accelerating against the gravitational pull for a longer period of time. To facilitate rapid iteration, four separate Copernicus decks were created.

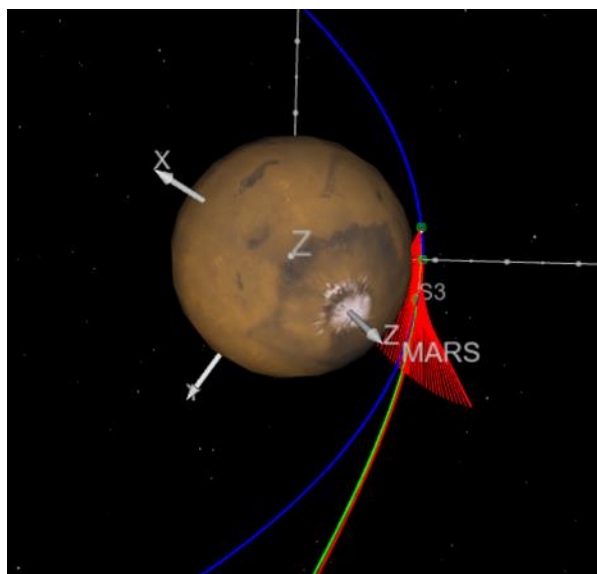


Figure 3. MOI Gravity Loss Model in Copernicus

By decoupling each burn, the convergence process was much easier between iterations. Each burn was verified with test cases. One deck was created for each major NTP burn:

- TMI: Trans-Mars Insertion
- MOI: Mars Orbit Insertion
- TEI: Trans-Earth Insertion
- EOI: Earth Orbit Insertion

The gravity loss Copernicus decks worked as follows:

1. The reference mission's impulsive delta-v is input
2. Spacecraft parameters are input (this can be T/W ratio, specific wet masses, or even desired burn times)
3. Copernicus determines the v_∞ achieved by the impulse burn
4. Copernicus solves for a finite burn resulting in the same v_∞ as the impulse burn

A python script was utilized to converge several missions at once, with the resulting gravity losses being iterated through the architecture vehicle sizing model until convergence was reached.

IV. Results

A. Transient Engine Performance

Figure 4 shows an example of a transient burn which details the slow power ramp rate based on NERVA information of 2.5 MW/sec or about 200 sec to full power [10]. Here, the long start up and shutdown time of the NTP engine translates to a significantly lower overall I_{sp} than the steady state, suggesting that both short start up and shutdown times and longer burn durations will benefit the overall average I_{sp} of the vehicle. Figure 5 shows the same burn but

with transients of 70 K/s, which is fast but demonstrated in XE-Prime test experience [11]. The ΔV and steady state maximum reactor power levels were kept the same; however, the decay heating adds to the total reactor power resulting in slightly higher power levels towards the end of the steady state portion of the burn. The most notable difference, aside from the difference in the ramp up/down times, is the average I_{sp} , which is shown to closely match the steady state I_{sp} value in Figure 5. Furthermore, slower transients also increase the total burn time of the engine.

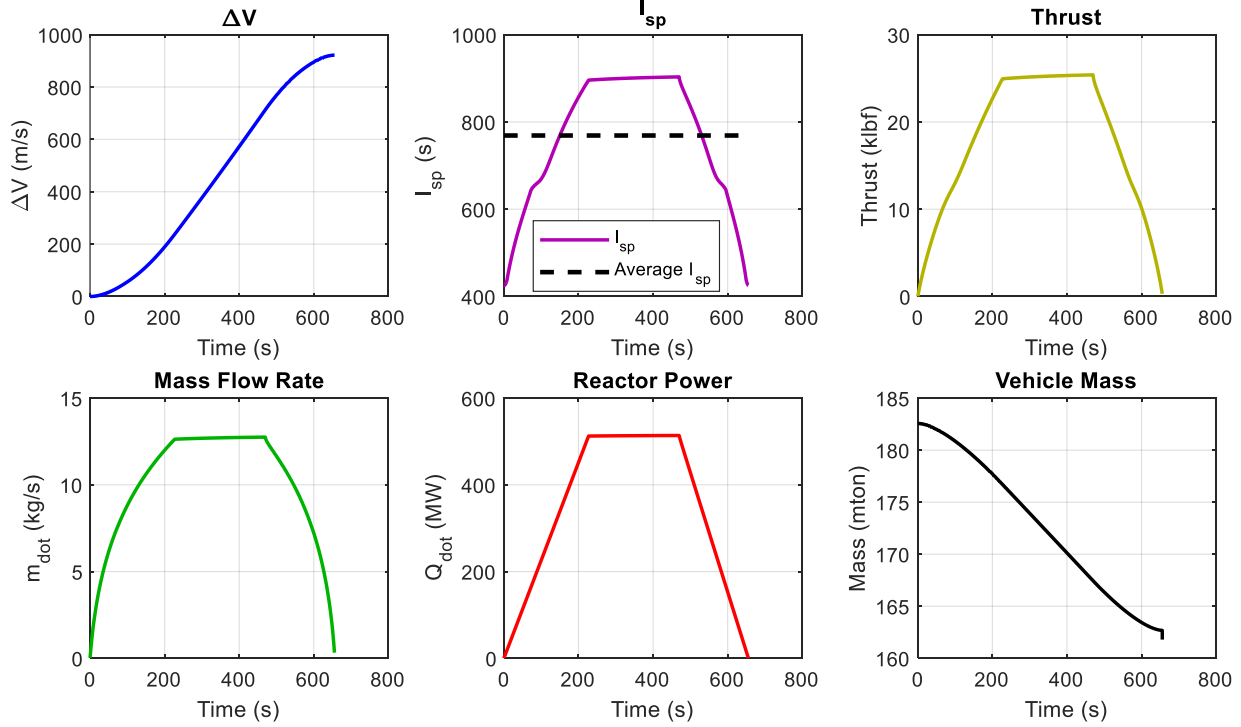


Figure 4: Example Slow Transient Performance Results for a slow ramp rate

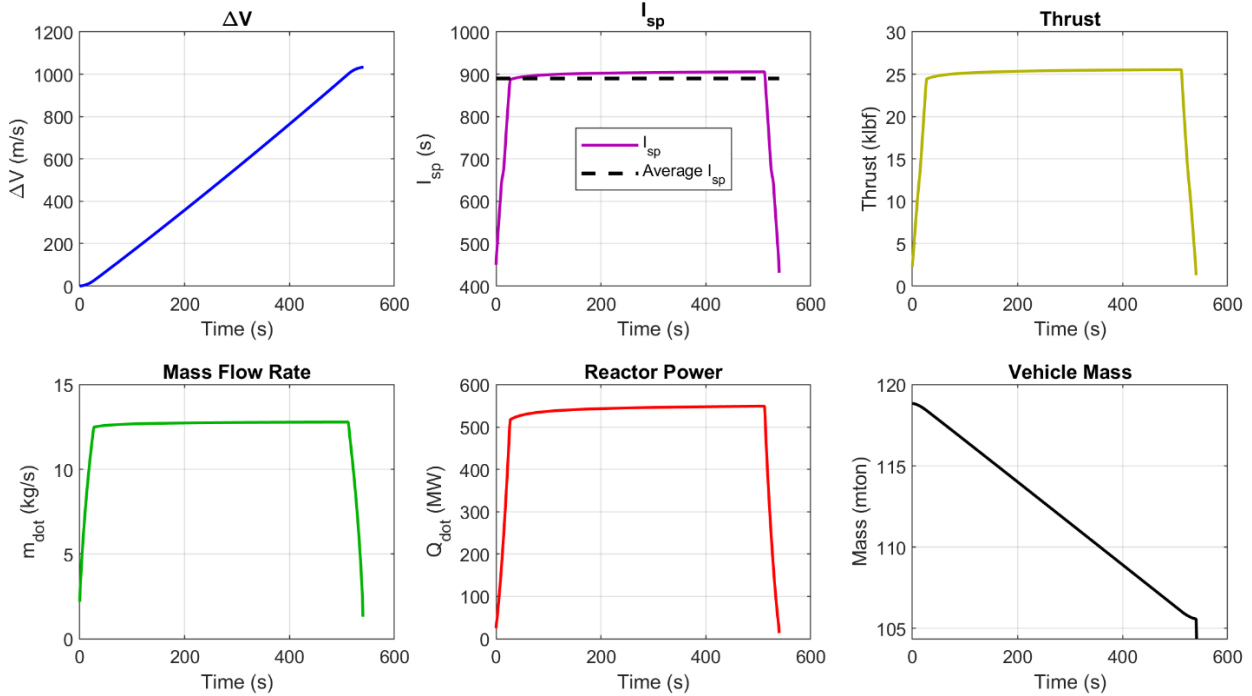


Figure 5: Example Fast Transient Performance Results

An estimate of the cool-down propellant mass needed is made as a function of burn duration based on point kinetics and decay heat as suggested by Emrich [12] and shown in Eq. 3 with the values of the constants informed by neutronics modeling.

$$P_{sd}(t) = P_{fp} \left\{ \frac{\rho}{\rho - \beta} \exp\left(\frac{\rho - \beta}{\Lambda} t\right) - \frac{\beta}{\rho - \beta} \exp\left(\frac{-\rho\lambda}{\rho - \beta} t\right) + 0.1104 \left[t^{-0.2436} - (t_{fp} + t)^{-0.2436} \right] \right\} \quad \text{Eq. 3}$$

B. Transient Impact on Burn Performance

The transient engine ramp rate affects the average I_{sp} for each burn. Tables 3-6 illustrate the burn performance and resulting architecture mass when the ramp rate is fast or slow. The ramp rate is more consequential to the shorter burns in the CONOPS due to the smaller percentage of time spent at or near steady-state. The ΔV budgets represent the actual change in velocity applied, while the burn time and average I_{sp} account for the total impulse required to achieve the ΔV . The Section IV.C discusses variation in gravity loss due to various assumptions of steady-state I_{sp} and total engine thrust.

The results demonstrate that the ramp rate range considered has an effect on the architecture in the reference mission roughly equal in magnitude to a 20 sec difference in steady-state I_{sp} . The reference mission CONOPS mitigates the effect of a slow transient compared to other possible CONOPS that utilize shorter and multiple MOI and TEI burns performed by the NTP as in [4]. With more shorter burns, the effect of the ramp rate in the range considered can be as large as that of a 100 sec difference in average I_{sp} . Splitting into smaller, more optimal burns may be worthwhile if the transient ramp rate is fast.

Table 3: Opposition Class CONOPS RCS, OMS, H-NTP Burn Details, 70 K/sec Ramp Rate

Burn Name	ΔV (m/s)	Time (s)	Avg I_{sp} (s)	Propellant Used (kg)	Cooldown Mass (kg)	Payload Mass (kg)	# Drop	# Inline
LDRO to LDHEO (RCS)	66		329	0	0	48000	8	1
LDRO to LDHEO (OMS)	93		450	9476	0	48328	8	1
TMI	909	1787	901.3	45146	3237	47339	6	1
DSM	3332	4730	905.7	120691	6931	37904	0	1
MOI	1463	1075	897.6	26914	2135	37904	0	1
Plane Changes (OMS)	50		450	1582	0	37904	0	1
TEI	1962	1146	898.2	28719	2252	38899	0	1
EOI	997	501	887.9	12227	1091	38571	0	1
LDHEO to LDRO (RCS)	66		329	0	0	38571	0	1
LDHEO to LDRO (OMS)	93		450	1942	0	38571	0	1

Table 4: Opposition Class H-NTP Vehicle Parameters at Earth Departure from NRHO

Parameter	Value
# Drop Tanks	8
# Inline Stages	1
Total Propellant Mass (mton)	262
Total RCS Mass (mton)	11.3
Total Dry Mass (mton)	190
Total Vehicle Mass (mton)	464

Table 5 illustrates the resulting burn times and average I_{sp} for the same mission CONOPS performed with a much slower ramp rate of 2.5 MW/sec, and Table 6 shows the resulting architecture-level result.

Table 5: Opposition Class CONOPS RCS, OMS, H-NTP Burn Details, 2.5 MW/sec Slow Ramp Rate

Burn Name	ΔV (m/s)	Time (s)	Avg I_{sp} (s)	Propellant Used (kg)	Cooldown Mass (kg)
LDRO to LDHEO (RCS)	66		329	0	0
LDRO to LDHEO (OMS)	93		450	9594	0
TMI	909	1972	878.3	46878	2591

DSM	3333	4947	896.2	123162	6258
MOI	1464	1246	860.6	28306	1522
Plane Changes (OMS)	50		450	1596	0
TEI	1963	1313	863.0	30019	1625
EOI	997	659	814.0	13332	578
LDHEO to LDRO (RCS)	66		329	0	0
LDHEO to LDRO (OMS)	89		450	1870	0

Table 6: Opposition Class H-NTP Vehicle Parameters at Earth Departure from NRHO

Parameter	Value
# Drop Tanks	8
# Inline Stages	1
Total Propellant Mass (mton)	267
Total RCS Mass (mton)	11.4
Total Dry Mass (mton)	191
Total Vehicle Mass (mton)	469

C. Mission Impact of Variation in Average I_{sp}

The following results are produced by varying the assumptions for engine steady-state I_{sp} , thrust, and engine mass and calculating the resulting finite burn performance for each major engine firing. The vehicle mass is converged upon by iterating the finite burn performance with the spacecraft architecture sizing model.

Figure 6 shows gravity losses for the worst-case (due to highest mass and thus lowest thrust-to-weight ratio) Trans-Mars Injection (TMI) burn. The additional impulse required becomes significant where the thrust-to-weight ratio (T/W) is less than 0.07 and becomes relatively insignificant with four engines operating at 900 sec I_{sp} . The gravity losses for each burn were investigated using the same method. Within the chosen reference mission, gravity losses are most significant for the TMI, when the vehicle is most massive. For the MOI and TEI, the gravity losses are significant only for cases with two engines, and the EOI does not have significant losses due to being the last burn which has the highest thrust to weight ratio.

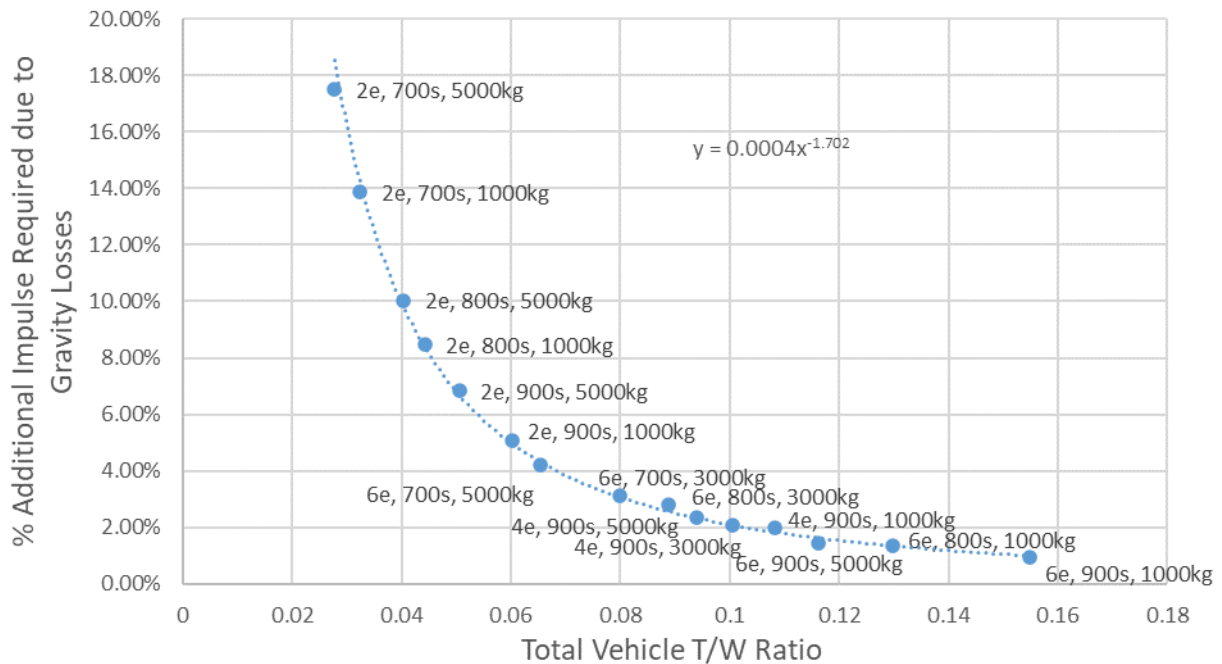


Figure 6. Gravity Losses for an 853 m/s TMI Burn from the Reference Mission for Multiple Engine Quantity and Performance Cases

The T/W of the vehicle determines the gravity loss. Thrust is determined by the number of engines. Differences in mass (weight) among the cases are determined by the engine I_{sp} and the engine mass. Figure 7 shows the T/W resulting from variations in these parameters. Higher I_{sp} and lower engine mass result in reduced gravity loss due to the associated mass savings. More engines and thus higher thrust result in lower gravity losses. However, Figure 8 shows that the effect of gravity losses on the vehicle mass is dominated by variation in I_{sp} and total engine mass.

Figure 8 shows the effect of I_{sp} , engine count, and engine mass on the total vehicle mass at Earth departure. An engine mass decrease from 5000 kg to 1000 kg provides a similar benefit to vehicle mass as a 100 sec I_{sp} increase. The effect of changing the engine mass is greater than 100 sec I_{sp} with 6 engines, and less with only 2 engines. In nearly all cases, reducing engine count and thus dry mass down to two engines results in a total mass decrease; the only exception is with the lightest 1000 kg and 900 sec I_{sp} engine case, where two or four engines are both near optimal. Increased I_{sp} and decreased engine mass always lead to reduced total vehicle mass.

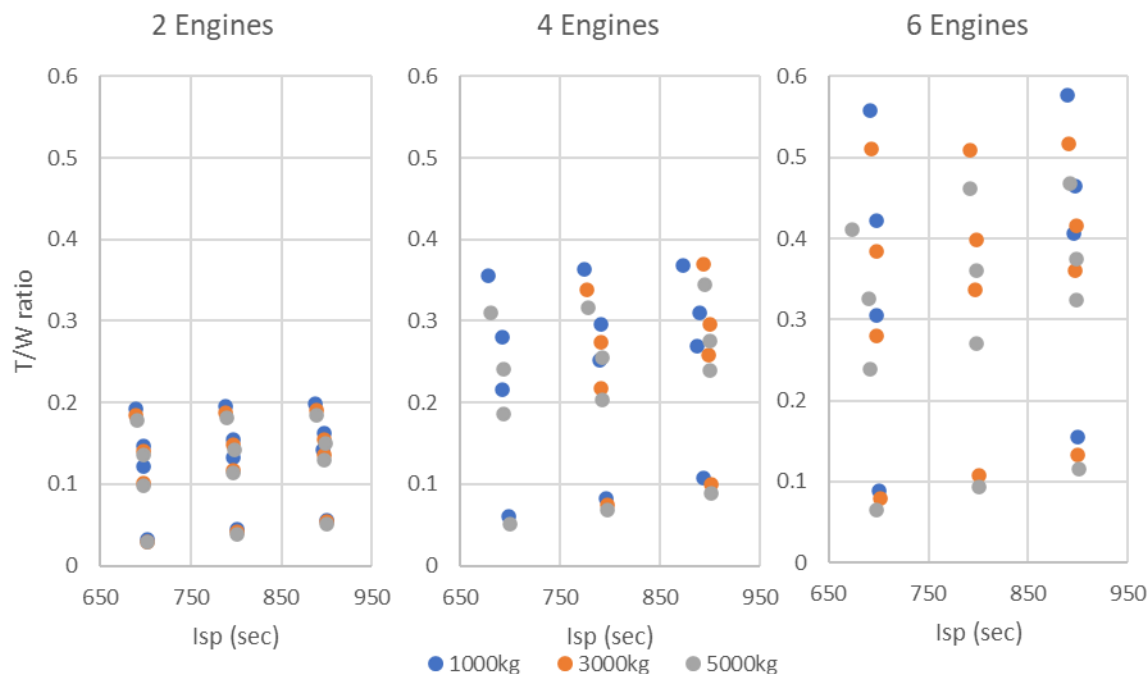


Figure 7: T/W vs I_{sp} for Various Engine Architectures – Reference Mission

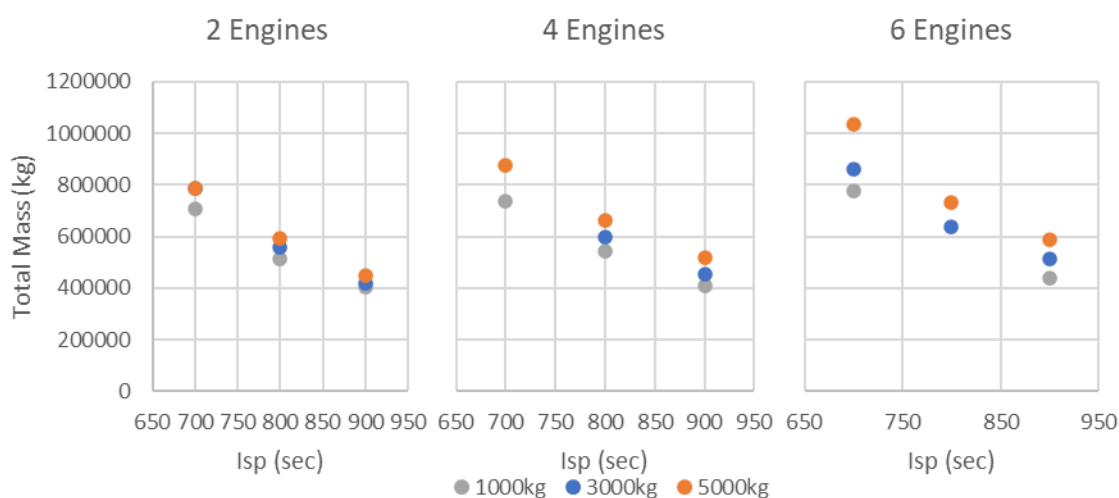


Figure 8: Total Vehicle Mass vs. I_{sp} for different engine quantities – Reference Mission

Note that for all the above, if the ramp rate of the engine is constrained to be as slow as 2.5 MW/s, the effective I_{sp} drops by approximately 20 seconds compared to a 70 K/s ramp rate. For a four-engine vehicle, this is the same

magnitude of effect as an increase in engine mass of about 1000 kg. For less favorable mission CONOPS with shorter burn durations, the effect can be as much as an engine mass increase from 1000 to 5000 kg or an I_{sp} decrease of ~100 sec. To maximize the I_{sp} over the mission and thus reduce the effect of slower engine transients on propellant mass, a CONOPS with fewer, longer burns is desirable. In summary, the considered ranges of transient ramp rate, steady-state I_{sp} , and engine mass are roughly equally important figures of merit for selection of reactor/engine technology or KPPs for a technology development program.

D. Ammonia Propellant Mission Performance

Using the same assumptions and methodology for ammonia as for hydrogen, the vehicle architecture resulted in only two drop tanks required to close the 2039 Mars Opposition-class mission. However, the total initial A-NTP vehicle mass is more than three and a half times that of the H-NTP vehicle. The gravity losses are also higher for the ammonia A-NTP vehicle as the burn times are longer and the T/W is lower, resulting in higher overall required ΔV .

Tables 7 and 8 show the results of this model. This analysis assumes stages sized for NRHO delivery. This results in higher dry mass and thus higher overall vehicle mass compared to larger stages sized for LEO departure. Furthermore, with the current assumptions for the 2039 Mars Opposition mission, even with engine masses of 1000 kg, the ammonia A-NTP vehicle will converge to over 1000 mton. Other mission opportunities requiring less ΔV would be more attractive to consider with an A-NTP engine, however this is outside the scope of this paper. Nevertheless, it is notable that although total mass is more than double the equivalent H-NTP case, the dry mass of the ammonia vehicle is actually less than that needed for H-NTP due to the much higher density of the propellant resulting in far fewer tank stages.

Table 7: Opposition Class CONOPS RCS, OMS, A-NTP Burn Details

Burn Name	ΔV (m/s)	Time (s)	Avg I_{sp} (s)	Propellant Used (kg)	Cooldown Mass (kg)	Payload Mass (kg)	# Drop	# Inline
LDRO to LDHEO (RCS)	66		329	27536	0.0	48000	2	1
LDRO to LDHEO (OMS)	73		450	27785	0.0	48328	2	1
TMI	853	6173	451.0	349837	30369	47339	0	1
DSM	3332	9763	451.0	550914	43086	37904	0	1
MOI	1569	2285	451.0	129925	13876	37904	0	1
Plane Changes (OMS)	50		450	2873	0.0	37904	0	1
TEI	2078	1931	451.0	109680	12090	38899	0	1
EOI	1009	634	450.9	34724	4519	38571	0	1
LDHEO to LDRO (RCS)	66		3290	2175	0.0	38571	0	1
LDHEO to LDRO (OMS)	73		450	2201	0.0	38571	0	1

Table 8: Opposition Class A-NTP Vehicle Parameters

Parameter	Value
# Drop Tanks	2
# Inline Stages	1
Total Propellant Mass (mton)	1206
Total RCS Mass (mton)	29.7
Total Dry Mass (mton)	124
Total Vehicle Mass (mton)	1360

V. Discussion and Conclusions

The steady-state I_{sp} , engine mass, and the start-up ramp rate are important performance parameters for determining the mass required to close the mission. The burn duration and average thrust are affected by the ramp rate and resulting

transient performance. Propellant quantity required to close a particular mission is also affected, as the difference in T/W determines the finite burn losses.

An engine with less than 900 sec average I_{sp} would certainly be able to close the crewed Mars mission, though at a mass penalty. Ammonia A-NTP can close the crewed Mars mission also, with fewer drop stages and lower dry mass, though at higher ($\sim 3x$) total mass. If filling of the ammonia tanks can be completed cost-efficiently by commercial launch vehicles, the ammonia A-NTP architecture may trade well at the total mission level due to the far fewer number of stages needing aggregation compared to H-NTP.

It is demonstrated that the ramp rate of the engine is as important of a KPP as the steady-state I_{sp} . Development must account for the start-up transient risks and consider the trade-offs in performance risk for both ramp rate and peak temperature/ I_{sp} . In development, the risk accepted by pushing to higher goal KPPs can be traded against the mass impact of those KPPs, and total risk balanced by considering all KPPs. For example, an engine that has 50% lower mass and 50 sec lower I_{sp} is equally good as a heavier, higher I_{sp} engine (measured by the total architecture mass figure of merit). The analysis shows also that a 10% increase in I_{sp} results in a much larger mission performance improvement than a 10% reduction in mass. Also, development that results in a fast transient rather than a very slow transient may prove more valuable than 10's of sec higher I_{sp} .

Forward work includes refinement of this analysis and assessment of uncertainty in effective tankage mass fractions using a higher-fidelity parametric vehicle model that the authors have developed. Additionally, other mission opportunities should be investigated with similar methods, as the total mission ΔV does have an impact on the observed results. Sensitivity analysis of the engine ramp rate's effect on the architecture mass may identify a "knee in the curve" that indicates likely values for threshold and target KPPs for the engine development.

E. Acknowledgements

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