Capabilities of Single Launch NTP and Chemical Spacecrafts for Cis-Lunar Tug Missions

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Renewed interest in the region between the Earth and the moon suggests that a tug operating between these two destinations will be most likely required in the near future. This report details a parametric spacecraft model which sizes a nuclear thermal propulsion or chemical propulsion spacecraft and compares their payload transport capabilities for missions to the moon. A spacecraft sized to fit in a single launch vehicle is assumed, and six launch vehicles are investigated. Maximum spacecraft payload mass and minimum mission transit time are compared between nuclear and chemical propulsion spacecrafts. For vehicles sized to the maximum capability of a launch vehicle, payloads of at least five metric tons are possible with either propulsion option; with nuclear thermal propulsion able to provide additional mass or reduced transit time when the spacecraft is sized to the maximum capability of the heaviest-lift launch vehicles. Transit times to the moon with a five-ton payload could be reduced by at least 0.2 days with a nuclear propulsion system.

I. NTP Application for Cis-Lunar Tug

Government agencies and private industry have displayed a recent and growing interest in the cis-lunar environment. For example, NASA has objectives to bring humans back to the moon permanently and utilize in-situ resources (Ref. 1). Science or business activities will require spacecrafts that operate between Earth and the moon with regular cadence. A cis-lunar tug, defined here as a spacecraft that transports a payload to the moon, has been discussed and designed in the past, but no active vehicles currently exist (Ref. 2). Nuclear thermal propulsion (NTP) technology is currently being pursued as a propulsion option for crewed Mars missions and may be a promising candidate for such a Lunar tug (Ref. 3). One merit of NTP that applies is its high specific impulse (I_{sp}), which can in some cases reduce the amount of propellant needed for a mission, increase the deliverable payload mass, or decrease the trip time compared to a similar chemical propulsion vehicle. A recent study investigated the comparison of NTP and chemical propulsion systems for low Earth orbit to geosynchronous orbit with an upper-stage derived reference vehicle (Ref. 4). The focus of this paper examines the comparison of NTP and chemical propulsion options for higher ΔV, larger tug operations.

I.A. Cis-Lunar Concept of Operations

For this paper an example cis-lunar concept of operations (con-ops) was defined. A single launch vehicle transports the tug to a nuclear safe orbit of 1000 km at 28.5° inclination (Ref. 5). A second launch vehicle places a payload into the same orbit. The tug performs rendezvous and docking with the payload. The spacecraft travels from the Earth to the moon, drops off the payload, and returns to the nuclear-safe Earth orbit. A destination of a 100 km circular orbit at the moon was selected. Copernicus (Ref. 6) was used to calculate the change in velocity (ΔV) of the various maneuvers; the values for the minimum energy scenario are listed in Table 1.

TABLE I. Cis-Lunar Tug Mission ΔV

Phase	ΔV (m/s)
Trans-lunar injection	3288
Moon circularization	900
Trans-Earth injection	927
Earth circularization	2975
RCS maneuvers	90

II. Spacecraft Model and Design

A parametric spacecraft model was created in Excel which calculates the mass, volume, delta-V, power, and payload constraints of a single-launch spacecraft. A bottoms-up analysis was used to determine the subsystem masses, and the launch vehicle fairing geometry was used to constrain the vehicle. NTP and chemical propulsion subsystems were defined for various propellant options. Designs of the ESA Automated Transfer Vehicle and NASA Orion European Service Module were used as references for the spacecraft bus concept (Ref. 7 and 8).

II.A. Dimensions and Structure

For the analyses within this paper the spacecraft is assumed to be transported to LEO in a single launch. The spacecraft consists of an: Engine and nozzle, thrust structure, main propellant tank, spacecraft bus, and docking/mounting structure. A general layout is shown in Figure 1.

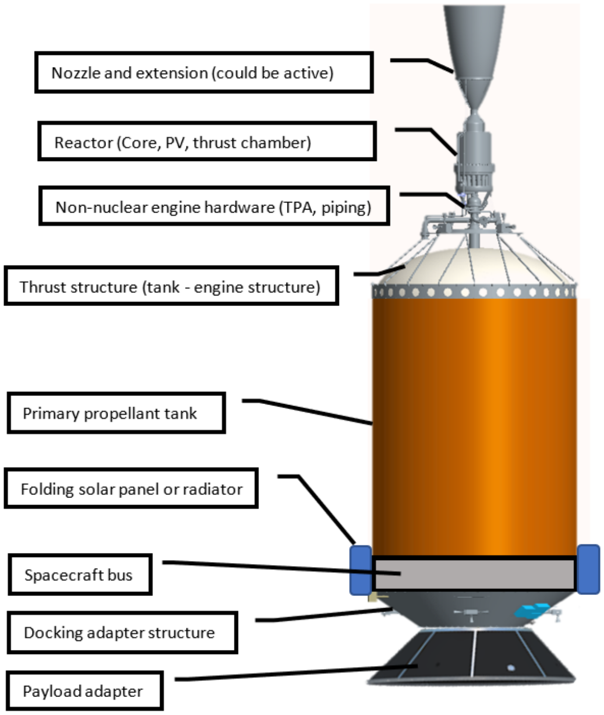

Fig. 1. General NTP spacecraft layout within launch vehicle fairing

The spacecraft bus contains the avionics, power management, RCS (reactor control system) tanks, and CFM (cryogenic fluid management) components. The fairing and mass-to-orbit capability used for each launch vehicle are shown in Table 2.

TABLE 2. Launch Vehicle Assumptions

Launch Vehicle	Fairing	Mass to 1000 km
SLS Block 2	8.4m PLF Long	96580
SLS Block 1B	8.4m PLF Short	89360
Falcon Heavy (Exp.)	5.2m Long	56878
Vulcan Centaur	5.4m Long	23481
New Glenn	7m High Capacity	44219
Starship	9m Standard	73132

II.B. Subsystems

II.B.1 Engine

An NTP engine derived from the NASA STMD reference designs (Ref. 9) was used, with the specifications described in Table 3. For the chemical propulsion option an RL-10 class engine with active nozzle extension was assumed (Ref. 10).

TABLE 3. Engine Specifications

Specification	NTP	Chemical
Thrust class	15,000 lb_f	15,000 lb_f
I_{sp}	900 s	451 s
Total engine mass	4550 kg	510 kg
Total engine length	5.0 m	2.3 m
Total engine diameter	1.9 m	1.9 m

II.B.2 Structures

The thrust structure, spacecraft bus, and docking structure components are assumed to be a cone or ring shape with supporting beams. The axial, lateral and buckling loads are derived from the mass of the above structure and loads during launch.

II.B.3 Power

Components for the communication, GNC and CDH were pulled from a collection of sources, including the Orion spacecraft, NASA Mars transit vehicle study, and COMPASS report (Ref. 11 and 12). The power requirements for the spacecraft were passed to the power management and then to the solar array and battery subsystem for sizing. Triple junction solar cells with up to 10 years of degradation was baselined, with the batteries being required to supply full power during eclipses and a depth of discharge of 25%.

II.B.3 Thermal and CFM

Heat removed from the spacecraft was assumed to be primarily from electrical system losses and the cryogenic fluid management system. For hydrogen propellant an active CFM was assumed due to the potential long loiter times for a cis-lunar tug mission. This system consisted of 20K and 90K cryocoolers as well as MLI and SOFI in order to reduce the propellant boil off to zero. For propellants with higher boiling points the active CFM system was removed. The maximum total flux from solar and albedo at LEO was used. Fold-out aluminum honeycomb radiators were assumed.

II.B.4 RCS

The RCS subsystem was required to provide an assumed delta-v needed during a single roundtrip mission to the moon. Bi-prop thrusters of MMH/MON were used. The RCS propellant tanks were assumed to be located in the spacecraft bus.

II.B.5 Primary Propellant Tank

The primary propellant tank was sized to maximize available propellant within the launch vehicle fairing while supporting launch loads and launch vehicle mass constraints. It was assumed to be a cylindrical tank with spherical end-caps and 3% ullage. Tank material and thickness was selected to minimize mass with a factor of safety of 1.2. The tank outer diameter is equal to the maximum allowable internal fairing diameter. The length was maximized during the model's iteration process while adhering to the limit of the internal fairing length and the mass constraints of the launch vehicle.

II.C Model Convergence Process

In order to maximize the amount of propellant in the spacecraft an iterative process is required due to subsystem dependencies.

1. Assumptions are used to calculate the engine and thrust structure specifications
2. Primary propellant tank length and mass are estimated
3. CFM, RCS, and other structures are calculated based on the mission profile and estimated spacecraft mass
4. Total spacecraft mass, length, and volume are compared to constraints and estimated values within the subsystems
5. If spacecraft exceeds mass or volume constraints, reduce tank size
6. Repeat process until estimated values converge with true values and tank material is minimized

A unique solution is found for each launch vehicle, propellant, and propulsion choice.

Note that a center of gravity (CG) constraint is not enforced on the spacecraft. Most launch vehicles' payload users guides specify a maximum CG location above the payload adapter. By neglecting this constraint, it is assumed that a custom payload adapter would be part of the spacecraft development, and that the spacecraft structures and guidance could support this. It is possible that imposition of a CG constraint would limit the propellant volume available to less than calculated in this paper.

III. Spacecraft Masses

The spacecraft model was used to generate spacecrafts for six launch vehicle options launching to the specified circular orbit at 1000 km. Payload mass capabilities for each launch vehicle at this orbit and inclination are from either the applicable payload user guide or NASA's launch services website (Ref. 13). Listed in Table 3 are the model results for an NTP spacecraft with hydrogen propellant. Almost all NTP spacecrafts were volume limited due to hydrogen's low density (that is, their mass is lower than the launch vehicle capability to the target orbit). The Vulcan Centaur was the only mass limited case. The largest available fairing was chosen for each launch vehicle.

TABLE 3. NTP-Hydrogen Spacecraft Masses

Launch Vehicle	Dry Mass (kg)	Wet Mass (kg)	Usable Propellant (kg)
SLS Block 2	22837	78742	51024
SLS Block 1B	16331	45074	25949
Falcon Heavy (Exp.)	10592	22268	10296
Vulcan Centaur	10800	23481	11226
New Glenn	13553	36643	20819
Starship	15544	43431	25194

The chemical spacecrafts have a shorter engine and lower dry mass due to the lack of a reactor, as well as reduced RCS propellant mass. All chemical spacecrafts are mass limited instead of volume limited due to the higher density of liquid oxygen. The chemical spacecraft masses for each launch vehicle are in Table 4.

TABLE 4. LOX/LH2 Spacecraft Masses

Launch Vehicle	Dry Mass (kg)	Wet Mass (kg)	Usable Propellant (kg)
SLS Block 2	12755	96580	77839
SLS Block 1B	12147	89360	71674
Falcon Heavy (Exp.)	7357	56878	45996
Vulcan Centaur	4923	23481	17103
New Glenn	7158	44219	34320
Starship	11197	73132	57401

Comparison of the two spacecraft dry mass datasets is shown in Figure 2. The NTP spacecrafts are at least 20% more massive than the chemical spacecrafts.

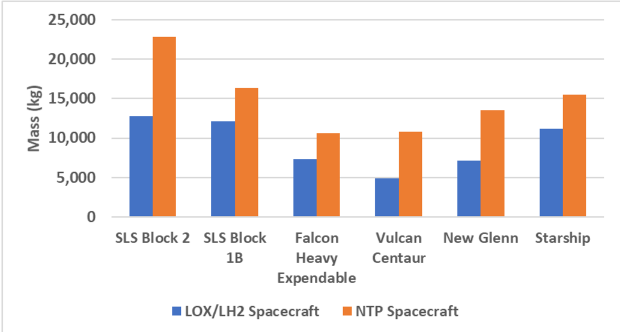

Fig. 2. NTP and Chemical Spacecraft Dry Masses for Each Launch Vehicle

Additional spacecrafts were modeled for NTP with ammonia propellant and chemical propulsion with LOX/CH4. The active CFM system was removed for these propellants due to their higher boiling points.

IV. Payload Mass Capabilities

The primary purpose of a cis-lunar tug is to transfer payload between Earth and lunar regions. Using the spacecraft propellant mass and ΔV requirements for this scenario a maximum payload mass that can be transported by the spacecraft can be calculated and compared. The result of the increased dry mass of an NTP vehicle but higher I_{sp} compared to a chemical vehicle can be evaluated.

IV. A. One-Way Low Energy Scenario

A low energy one-way transfer from the previously described LEO to LLO orbits represents the highest payload mass scenario and takes approximately three days of transit time. Plotted below in Figure 3 are the payload masses in kilograms for NTP and chemical spacecrafts.

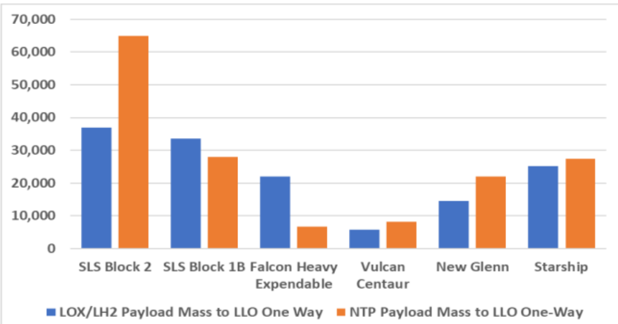

Fig. 3. Maximum Payload Mass for LEO to LLO for Three Day Transit

The vehicles sized for heavy launch vehicles generally benefit from the NTP's higher I_{sp} as they take advantage of the larger payload fairings, allowing for higher quantities of low-density propellant. The SLS block 2 provides the maximum payload mass at 65 tons. The Falcon Heavy has a relatively small volume fairing compared to its mass capability, and therefore a chemical spacecraft sized for it has higher payload capability instead. This suggests that NTP should be more heavily considered as the desired payload mass increases. Payload masses from 5 tons to 65 tons were found, with a median of 24 tons. A one-way trip could be used if capability to refill near the moon exists, or as a technology demonstration mission.

IV. B. Roundtrip Low Energy Scenario

Another scenario would require the cis-lunar tug to transport a payload mass from LEO to LLO, drop it, and return to LEO for re-filling and re-use. Using the low energy three-day transit for both portions of the mission yields the highest payload mass, which is shown in Figure 4.

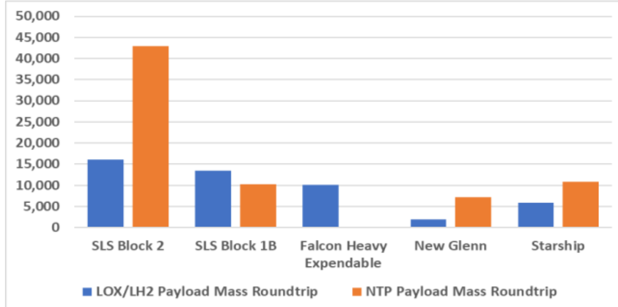

Fig. 4. Maximum Payload Mass for Roundtrip Scenario with Three Day Transit Times

Due to the higher delta-v requirements of a roundtrip mission the maximum payload masses are lower than for the one-way mission. A tug vehicle sized for launch in the Vulcan Centaur does not close the mission with either NTP or chemical, and a Falcon Heavy -sized tug only closes the mission with the chemical spacecraft. Spacecrafts sized for the other launch vehicles show similar trends as in the one-way scenario, where the SLS block 2 enables the highest capability tug and shows the largest difference between chemical and NTP tugs. New Glenn and Starship tugs also favor NTP, while the lower volume SLS block 1B variant and Falcon Heavy favor chemical spacecrafts. This data suggests that payload masses above 10 tons will be difficult with a tug launched by a single launch vehicle unless an SLS block 2 is chosen.

It is worth noting that because all but the smallest hydrogen NTP tug concepts are volume-limited by the launch vehicle fairings, they would have higher payload capability if the launch vehicle performance was fully utilized by launching them to a higher starting orbit, and if the payload could also be launched to the same orbit. This is not true for the mass-limited chemical tug concepts, as to launch into a higher orbit, they would have to be sized down by reducing propellant load.

IV. C. Transit Duration

There are scenarios in which shorter transit times of the inbound or outbound portion of a cis-lunar trip may be desirable, such as emergency supplies for a lunar base or prompt repair of a satellite. The required ΔV for transfer trajectories with durations of 0.5 to 3 days was calculated and then used to determine maximum payload mass deliverable by each spacecraft configuration. The payload mass for a one-way trip from LEO to LLO for an NTP spacecraft is shown in Figure 5. All spacecraft options have reduced payload mass of about 50% when the transit duration decreases from 3 to 1.4 days. The decrease in mass is a percentage of total mass so the SLS block 2 tug capability has the largest absolute change. The Starship and SLS block 1B spacecrafts have a similar maximum payload mass at 3 days as the SLS block 2 has at 1.3 days. A transit duration of 1 day is near the limit for almost all spacecrafts in this comparison, and is important to note for scenarios where time is crucial. Figure 6 shows the round-trip plot, where 1.5 days is the limit of most of the spacecrafts, with similar overall trends. For the chemical spacecrafts, Figure 7 shows the maximum payload mass for a roundtrip mission. A minimum time of 1.5 days is also shown, with Starship and New Glenn spacecrafts limited to about 2 days or more. Comparing Figures 7 and 8 it can be seen that NTP is capable of higher mass delivery, especially for SLS block 2, New Glenn, and Starship at the 1.5 day scenario.

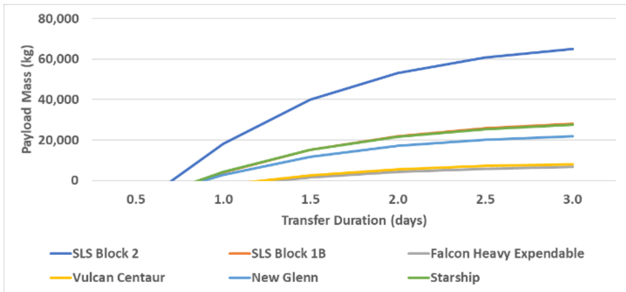

Fig. 5. Maximum Payload Mass for LEO to LLO for an NTP Spacecraft for Various Trip Durations

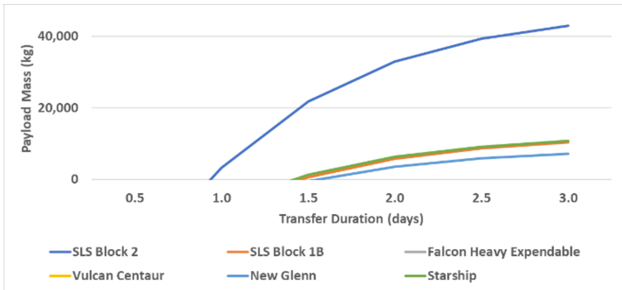

Fig. 6. Maximum Payload Mass Roundtrip for an NTP Spacecraft for Various Outbound Trip Durations

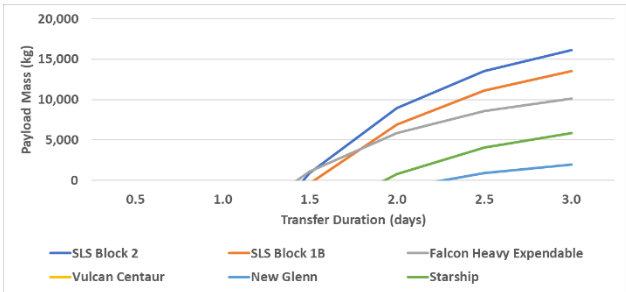

Fig. 7. Maximum Payload Mass Roundtrip for a Chemical Spacecraft for Various Outbound Trip Durations

Shown in Figure 8 is the difference in minimum transit time possible with the chemical tug design compared to the equivalent launch vehicle-sized NTP tug design for 5 ton and 10 ton payloads for a one-way mission. The NTP spacecraft generally can provide a shorter transit time, although the result is not consistent for tugs sized for all launch vehicles.

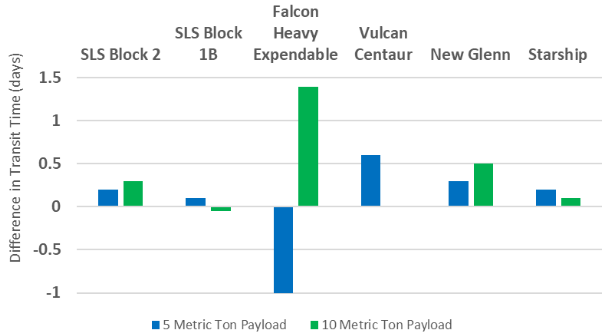

Fig. 8. Difference in transit time compared to NTP for two payload mass options

IV.D. Alternative Propellants

In addition to the hydrogen propellant for NTP and LOX/LH2 chemical spacecraft options, ammonia NTP and LOX/CH4 chemical was considered. Benefits of these propellants include reduced CFM and associated technology development, and avoidance of operational difficulties with hydrogen. Specific impulses of 400s and 363s were chosen for the ammonia NTP and LOX/CH4 options, respectively. The maximum payload mass for a three day one-way transit for the various configurations is shown in Figure 9. Only tugs sized for four launch vehicles close the one-way mission with all propellant configurations. The LOX/CH4 configuration has a lower payload mass for all launch vehicle options due to its lower I_{sp}, despite decreases in engine mass compared to NTP, and removal of the active CFM mass. The ammonia NTP configuration has the lowest performing payload mass due to both a lower I_{sp} than hydrogen and the higher mass NTP engine. While the alternative propellant options perform worse, three launch vehicle options provide over 10 tons of payload mass and could potentially fulfill the requirements of a cis-lunar tug spacecraft. The higher density of the propellants could also allow the use of smaller and less expensive launch vehicles like the SpaceX Falcon 9. If required technology development is considered, then a spacecraft without CFM may prove to be a viable candidate. It is worth noting that neither alternative propellant option was able to complete the roundtrip low energy scenario.

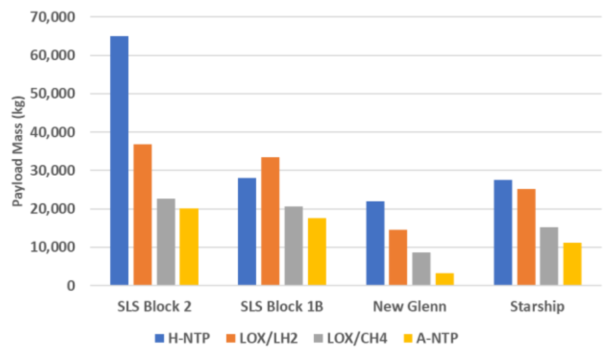

Fig. 9. Maximum payload mass for three day one-way transit from LEO to LLO

IV. E. NTP and Chemical Specific Impulse Break-even

The above analyses assumed a I_{sp} of 900s for the NTP engine, but near-term nuclear fuels may result in lower I_{sp} values (Ref. 14). For the roundtrip scenario with payload drop-off the I_{sp} assumption for NTP can be adjusted until the payload capability of the NTP tug equals that of the LOX/LH2 tug sized for the same launch vehicle. Shown in Figure 10 are the results of this analysis. A range of break-even values from 734s to 1371s are seen, with the SLS Block 2 associated tug requiring the lowest I_{sp} value to match chemical propulsion performance. Due to the lower volume fairing of the Falcon Heavy the required I_{sp} for its tug is the largest. The other four options are near the 900s I_{sp} value. If NTP I_{sp} is limited to less than 900s (e.g., for a near-term lower risk demonstration), a tug sized for a high volume launch vehicle could be paired with this engine and still provide more payload capability than a chemical propulsion tug.

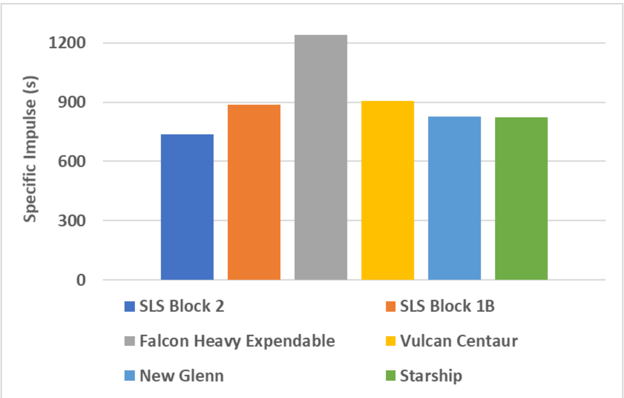

Fig. 10. Required NTP engine I_{sp} to equal LOX/LH2 spacecraft payload capability for roundtrip transit when sized for each launch vehicle

IV.E. Discussion

The results provided above illustrate the tradeoffs between I_{sp} and mass of the propulsion system and the density of propellants given a set of launch vehicles to launch the tug spacecraft. For any given tug volume and mass limitations, there exists a theoretical NTP I_{sp} at which NTP will outperform chemical propulsion for a given either ΔV or payload mass. Based on the ammonia NTP results this breakeven I_{sp} is higher than what was assumed for ammonia NTP. However, for very large ΔV missions (requiring multiple launches to assemble a multi-tank tug), ammonia-NTP's high I_{sp} compared to LOX/CH4 should result in ammonia NTP outperforming LOX/CH4.

V. Future Work

Continued work on the topics presented in this paper are planned, with the following high level goals:

- Investigate refilling and retanking con-ops and its effect on the spacecraft configurations and capabilities
- Refine spacecraft model to calculate states of the vehicle in pseudo-transient mission phases
- Integrate with higher fidelity engine model in order to capture NTP start-up and cooldown effects and thrust levels
- Expand capability of integrated spacecraft model to explore additional missions such as Mars trips
- Investigate sensitivity of tug capabilities to assumptions related to NTP engine mass, usable fairing volume, and starting/final orbit altitudes

While this work assumes a single launch to orbit scenario, a possible concept for a cis-lunar tug spacecraft would be to refill with propellant in orbit or to launch filled tanks separately and dock to the spacecraft. This would reduce the current total mass or volume limitations and allow for smaller and less expensive launch vehicles to be utilized. Larger payload masses or higher delta-v missions could be investigated as well. Some near-term improvements in the spacecraft model include factoring in the center of gravity limitations of the launch vehicle, which could limit the length of the spacecraft in the current vehicle layout as the nuclear reactor is currently assumed to be near the top of the fairing. Payload mass versus altitude curves could also be used for each launch vehicle in order to choose higher starting altitudes more favorable to NTP for configurations that were volume limited.

VI. Conclusions

This paper describes a parametric spacecraft model that was used to size nuclear thermal propulsion and chemical propulsion spacecrafts for launch in a variety of launch vehicles, assuming a single launch to orbit. Analyses and comparisons between the two propulsion options related to the maximum payload mass and minimum trip duration of a cis-lunar tug mission were described. For both one-way and roundtrip scenarios an

NTP spacecraft performed similarly or slightly better than a LOX/LH2 spacecraft for a majority of the launch vehicles investigated. The NTP tug sized for the large-volume fairing of SLS block 2 was able to deliver the largest payload mass to LLO, with almost double the payload mass compared to a LOX/LH2 tug sized for the SLS block 2. For launch of the tug in the smaller lift and lower volume Falcon Heavy the chemical propulsion tug performed better. When comparing performance for faster than minimum energy transit times, the NTP spacecrafts performed better for both 5 ton and 10 ton payloads, although this analysis showed high sensitivity to the launch vehicle chosen to constrain the tug design. Alternative propellants of ammonia for NTP and methane for a chemical system were briefly investigated and shown that they could close lower payload mass one-way missions but had reduced capabilities. Ammonia-NTP performed worse than chemical in all cases considered. For a tug designed to launch in the largest fairings and perform high ΔV missions, NTP can deliver much larger payloads and faster than a chemical propulsion tug subjected to the same constraints.

ACKNOWLEDGMENTS

This work was supported by NASA's Space Technology Mission Directorate (STMD) through the Space Nuclear Propulsion (SNP) project. This work was funded under Contract No. 80LARC17C0003.

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