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CARGO LOGISTICS FOR A NOTIONAL MARS BASE USING SOLAR ELECTRIC PROPULSION

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Abstract

The current aim of NASA's Journey to Mars is a stepwise approach towards landing humans on the Red Planet, culminating in a sustained presence. There are many recent studies on how this can be achieved in an evolvable and affordable manner. Most architectures begin with crewed missions to Phobos or Mars orbit in the mid-2030's, progress toward short-stay missions on the surface, and then culminate with regular, long-stay missions at a permanent outpost in the 2040's. A common factor of these architectures is that many robotic launches are required in order to support the crew by prepositioning mission elements and other needed supplies. In this paper, the use of 150 kW reusable Solar Electric Propulsion (SEP) tugs as a means to deliver elements both to orbit and the surface is studied. The SEP tugs make use of technology currently being developed for the proposed Asteroid Redirect Robotic Mission (ARRM). They would also be used to deliver food and supplies to sustain the crews similar to resupply missions for the International Space Station. These SEP tugs would cycle (with loitering) between staging orbits in cislunar space and Mars orbit.

In order to characterize mission design parameters such as dates, masses, and durations, thousands of optimized trajectories were run using low-thrust optimization software. Solutions are found for all launch/arrival date pairs for the years 2038-2053. They can be displayed as contour plots called Bacon plots – the SEP equivalent of porkchop plots. Possible mission architecture concepts for a steady-state human presence on Mars along with the cargo missions needed to keep it functioning are described and the relevant mission parameters such as launch dates, masses, arrival dates, etc., are given. It was found that the reusable SEP tug architecture is highly beneficial to the logistics of a sustainable Mars outpost.

Keywords: Human Mars Exploration Logistics SEP

Acronyms/Abbreviations

ARRM - Asteroid Redirect Robotic Mission
DSH - Deep-Space Habitat
EDL - Entry, Descent, and Landing
EMC - Evolvable Mars Campaign
EUS - Exploration Upper Stage
HERMeS - Hall Effect Rocket with Magnetic Shielding
HMO - High-Mars Orbit (5-sol)
iCPS - interim Cryogenic Propulsion Stage
MALTO - Mission Analysis Low Thrust Optimizer
MAV - Mars Ascent Vehicle
MOI - Mars Orbit Insertion
MORT - Mars ORbiter Tool
mt - metric ton
NASA - National Aeronautics and Space Administration
NRHO - Near Rectilinear Halo Orbit
SEP - Solar Electric Propulsion
SLS - Space Launch System
TEI - Trans-Earth Injection
TLI - Trans-Lunar Injection
TMI - Trans-Mars Injection

1. Introduction

NASA's Evolvable Mars Campaign (EMC) has an end goal of a sustainable infrastructure at Mars that will be used by multiple human crews. In 2015, Price, et al. presented a stepwise approach towards landing humans on Mars [1, 2]. The pathway put forth was similar to others in that it made use of the capabilities of the Space Launch System (SLS) to slowly build up the human presence at Mars in a sustainable and affordable manner. This "minimal architecture" approach begins with a crewed mission to Phobos in the mid-2030's, progresses towards short-stay missions on Mars, and then culminates with regular long-stay missions at a permanent outpost in the 2040's.

Human spaceflight becomes sustainable through a robust supply chain that minimizes risk to mission and crew, regardless of the mission architecture. This is true for the International Space Station and would especially be true for humans on the Red Planet. A previous paper by the authors herein introduced early study results which support a hypothesis that such a supply chain is indeed achievable within the timeline outlined in the NASA EMC [3]. The current investments in the SLS and Solar Electric Propulsion (SEP) powered spacecraft form the foundation of the architecture analyzed.

In response to NASA's desire to have a Mars Campaign that is "evolvable" [4, 5], many researchers have proposed the use of hybrid architectures [6, 7], that take advantage of the strengths of both chemical and electric propulsion systems [8]. This can be in the form of either separate vehicles, with SEP used for cargo [9] and chemical used for crew [10, 11], or truly hybrid vehicles that use SEP in deep space and high-thrust chemical engines in critical regions where they may take advantage of the Oberth effect and greatly reduce trip times [12]. Another common desire of the Evolvable Mars Campaign is to make use of a lunar gateway [13, 14], as a staging point for missions beyond Earth. Common mission elements like propulsion, propellant, cargo, and habitats can be aggregated in stable cislunar orbits where they can then depart for various destinations by taking advantage of low-energy transfer techniques [15]. Along with the use of the lunar gateway, it can be beneficial to make use of reusable elements such as propulsion modules that return to the gateway to be refueled after delivering cargo to Mars. In this paper, the use of a reusable SEP tugs and their benefits in launch sequencing for a human Mars expedition are explored.

Various mission designs utilizing these SEP tugs were investigated in support of both orbital and landed cargo, as well as return supplies and other consumables [3]. An effort is made here to quantify the sizing of a SEP tug in terms of mass, power, and capabilities that optimizes the cargo mass along with suitable flight times. The results presented reveal a compelling case for their use to meet the needs of a Martian supply chain in future human missions to Mars.

2. Methods and Assumptions

Some assumptions are required in order to make the problem of designing an architecture of a sustained human outpost on Mars more tractable. None of the mission elements presented here are intended to be detailed designs, but rather serve as suitable placeholders that would allow for broad architecture design from which insights may be drawn.

2.1 SEP Tugs

A notional SEP tug utilizes up to 10 HERMeS Hall-effect thrusters [16] and has refillable xenon propellant tanks. It is a high-heritage follow-on to the technology development for the Asteroid Redirect Robotic Mission* (ARRM) which was a 50 kW SEP spacecraft propelled by 4 HERMeS engines. The SEP tug here would be powered by 150 kW (@ 1 AU) arrays and is roughly three times the size of the ARRM spacecraft, making use of many similar components (see Figure 1). It is capable of docking/undocking to support multiple round-trips. The dry mass of the SEP tug is approximately 8 mt. (This is consistent with other studies). A constant 10 kW is diverted for spacecraft systems and margin, leaving 140 kW for the propulsion system. Each HERMeS engine is assumed to provide 585 mN of thrust and 2660 seconds of Isp when receiving its maximum power of 14 kW. At Earth, there is enough power to run all 10 engines, diminishing to 3-4 engines at Mars as available solar power is reduced.

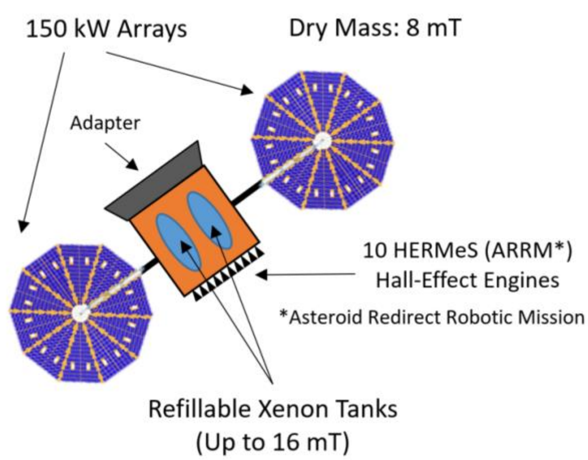

Figure 1 - The notional SEP tug would use up to 10 HERMeS engines and 150 kW of power. Nominally it would weigh 8 mt dry and have large, refillable xenon tanks.

One or more of these tugs would be delivered to the cislunar staging point to be mated with cargo modules bound for Mars. While there are many options for staging orbits [17], a lunar Near Rectilinear Halo Orbit (NRHO) [18] was chosen for this study. The basic properties of the NRHO are a low perilune near one of the poles (90° inclination), high apolune, a period of around 9 days, and an orbital plane facing Earth. This type of orbit balances the competing needs of a staging orbit, providing easier access to the lunar surface than a

* The ARRM mission to an asteroid was canceled earlier this year, however, the development of large Hall-effect thrusters continues.

Distant Retrograde Orbit, and easier access to deep space than a Low-Lunar Orbit [19]. Because the orbits are unstable (requiring ~10 m/s per year for station keeping), the tug departs the NRHO and vicinity of the Moon with minimal ΔV. A combination of solar perturbations and SEP thrusting increases the energy with respect to the Moon, so that a lunar gravity assist can cause the tug to escape Earth with a C3 of around 2 km^2/s^2. This energy raising process takes approximately 4 months and 100 m/s of ΔV. At the end of the resupply mission, this process is reversed to capture back into the NRHO. One important aspect of placing the SEP tug in cislunar orbit is that it is not necessary to use SEP to climb out of the Earth's gravity well, which requires months of spiraling.

2.2 Launch Vehicle

The NASA Space Launch System (SLS) is the agency's selected launch vehicle for exploration class crewed missions as well as potential deep space science missions. In its early launch configuration, scheduled for a 2019 launch, the SLS consists of a core stage using four RS-25 main engines, 2 five-segment solid rocket boosters, and a derivative of the Delta IV Heavy second stage known in the SLS program as an interim cryogenic propulsion stage (iCPS). The SLS configuration launching in 2019 is known as the 'Block 1' vehicle [20]. The Block 1 SLS is capable of sending approximately 25 mt to a trans-lunar injection (TLI). The SLS, along with other NASA human spaceflight hardware programs, is an evolvable vehicle; it is anticipated that it will quickly evolve to a Block 1b configuration (

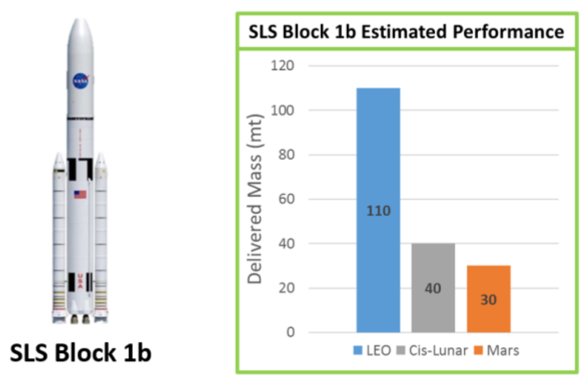

Figure 2) that would use a larger upper stage known as the Exploration Upper Stage (EUS). The Block 1b configuration is anticipated to deliver 40 mt to TLI; deriving from the typical lunar C3 value. The payload to a Trans-Mars Injection (TMI) likely would be ~30 mt. It is assumed that SLS Block 1b, or something with greater capability, would be available by the late 2030's.

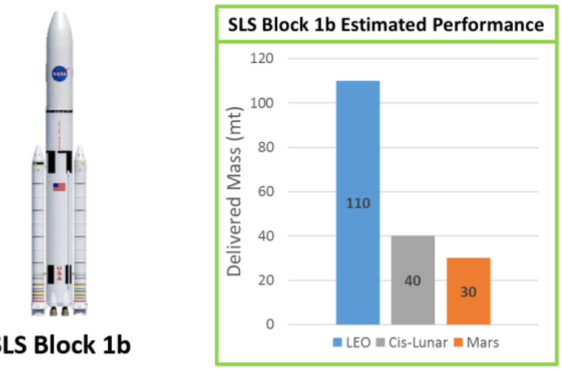

Figure 2 - Assumed Capabilities of the SLS Block 1b Launch Vehicle.

2.3 Mission Elements

Crewed missions to Mars require many elements no matter what architecture is employed. A sustained outpost on Mars would need many launches to send the infrastructure needed to assist the crew throughout their journey. They will need habitats, propulsion modules, landers, ascent vehicles, etc. There are virtually an unlimited number of ways to orchestrate the mission architecture in terms of types of mission elements, staging locations, and mission sequences. For the purposes of this study, we use element masses and an architecture somewhat similar to [1, 2, 21]. The specifics and feasibilities of the infrastructural elements are not crucial to the purpose of showing a robust method of cargo delivery and resupply.

Table 1 lists the mission elements used along with rough masses. For elements that are not part of the cargo supply chain, such as the Mars Ascent Vehicle (MAV) or surface habitat, the mass allocation is somewhat inconsequential to the resupply architecture. The in-space propulsive elements (TEI, MOI, and Mars Orbit booster) would be delivered and prepositioned by the SEP tug and therefore need to have an assumed mass. Each of these elements weigh 25-30 mt, which is near the limit of what the tug could deliver to High Mars Orbit (HMO – 5-sol for this study) under our assumptions. One of the great benefits of low-thrust missions is the ability to be flexible to change. When masses increase (or decrease), the trajectory can (and often must) be modified to accommodate changes and meet the new requirements.

Table 1 - Assumed Masses for Mission Elements needed for a Human Mars Outpost.

Mission Element	Mass Allocation	Includes Prop?
	Crew	
Orion (Command + Service)	20 mt	yes

Deep-Space Habitat (DSH)	30 mt	no
Surface Habitat (HAB)	35 mt	no
Propulsive		
TEI Stage	26 mt	yes
MOI Stage	28 mt	yes
LMO-to-HMO Booster Stage	22 mt	yes
Crew Lander/MAV	50 mt	yes
SEP Tug	8 mt	no
Resupply		
Orbital Resupply Module	15-30 mt	no
Surface Resupply Module	20-30 mt	yes

The orbital resupply module is a flexible cargo vessel that has an assumed dry mass of 8 mt. It is capable of carrying 7 to 22 mt of crew consumables (e.g. food, water, supplies) as well as other liquids and gases. Its purpose is to mate in HMO with the deep space habitat (DSH) and resupply it for the journey home. It could also serve as a resupply depot for other elements.

Delivering cargo to the surface requires more supporting mass to achieve entry, descent, and landing (EDL). A gear ratio of 3-to-1 was assumed for entry to useful landed mass. For a lander of 30 mt at entry, roughly 10 mt are allocated to the aeroshell and entry systems, 10 mt to the terminal landing and structure, and 10 mt to useful cargo for the crew.

3. Methods

The primary objective of this research was to quantify the benefits of SEP in a Human Mars architecture, specifically in the use of a tug to deliver logistics from a Lunar staging orbit to Mars. The flexibility and efficiency of low-thrust trajectories are quite beneficial and show promise in the creating a launch robust schedule to support a human Mars expedition.

In previous papers [3, 22], a notional 150 kW SEP tug was employed that weighed 8 mt and could deliver > 30 mt of useful mass to HMO. This is in comparison to ~22 mt via direct launch on an SLS 1b after accounting for propellant mass for MOI. An increase of nearly 50% in mass delivered, in addition to added timeline flexibility, was deemed worthy of the effort to build and use a SEP tug. On the other hand, the same SEP tug only led to marginal increases in delivered mass to the surface when compared to direct launch and entry. For that reason, the complexity of using the SEP tug was not employed for surface elements.

The ability to launch the requisite architectural elements with cargo launches adequately interspersed can be challenging. The SLS launch vehicle is rather large and would be difficult to launch too frequently. The greater the separation between launches, the easier it would be to sustain a Martian outpost. However, ballistic interplanetary transfers have a steep optimum where performance falls off quickly if not timed right. Launching more than 2-3 times per opportunity would be quite difficult if constrained to the few months of optimal planetary alignment, both from a launch operations perspective and a financial one.

Reusable SEP tugs are thus employed to alleviate the need for conventional direct launches. For orbital cargo missions (Figure 3) - an SLS launch vehicle, capable of lifting about 40 mt to NRHO, sends a cargo vessel to rendezvous with the SEP tug. The mission propellant for the round trip is transferred to the tug and the spacecraft begins its trajectory by performing energy raising maneuvers including a lunar gravity assist to send it towards Mars. The SEP engines then bring the cargo to Mars and spiral down to the 5-sol elliptical staging orbit. The detached SEP tug then begins its journey back to lunar NRHO and reverses the process.

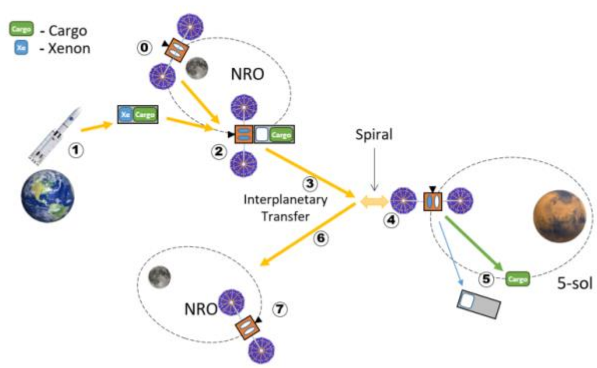

Figure 3 - Mission architecture concept using a SEP tug. Orbital cargo missions begin with a rendezvous of the tug and cargo in NRO, and then continue to high-Mars orbit where the cargo is left and the tug returns to NRHO.

3.1 Bacon Plots

In order to characterize mission design parameters (e.g. dates, masses, and durations) for the cargo missions, thousands of optimized trajectories were generated. By exploring a wide range of parametric combinations, a better map of the trade space can be constructed. This allows us to evaluate whether desired missions are feasible, as well as sensitivities to changes. Plotting performance parameter contours versus launch and arrival dates creates the SEP analog to a ballistic Porkchop plot, which is called a "Bacon Plot" [23].

Low-thrust mission design analysis was carried out using MALTO, a fast, medium-fidelity low-thrust optimizer developed at JPL [24]. MALTO stands for Mission Analysis Low Thrust Optimizer. This tool generally exhibits robust convergence and can be run in parametric mode with fast, accurate results. MATLO was used to generate thousands of trajectories by sweeping through all launch/arrival date pairs for 2038-2054. This covers a complete set of the 15-year (7 opportunity) Earth-Mars cycle. It was found that low-thrust trajectories do not vary as significantly from opportunity to opportunity as do ballistic transfers.

For Earth-to-Mars trajectories, the simulations begin with the SEP tug in NRHO mated with the mission element (cargo module or propulsive element) to be delivered to Mars orbit along with the requisite xenon propellant. This gives a maximum starting mass of 48 mt – 8 mt for the SEP tug dry mass and 40 mt from the maximum throw mass of the SLS 1b to a C3 of -2 km^2/s^2 (NRHO). The transfer begins with 4 months and a nominal ΔV of ~100 m/s to affect a series of gravity assist maneuvers and depart towards Mars with a C3 of +2 km^2/s^2. From this point the MALTO software finds the optimal thrust profile to minimize propellant usage over the range of dates to arrive at Mars and begin the short spiral down to a 5-sol elliptical staging orbit (HMO). The spiral would roughly require 750 m/s and 90 days. But with some assistance from the ACS thrusters this can be reduced to 250 m/s and 30 days.

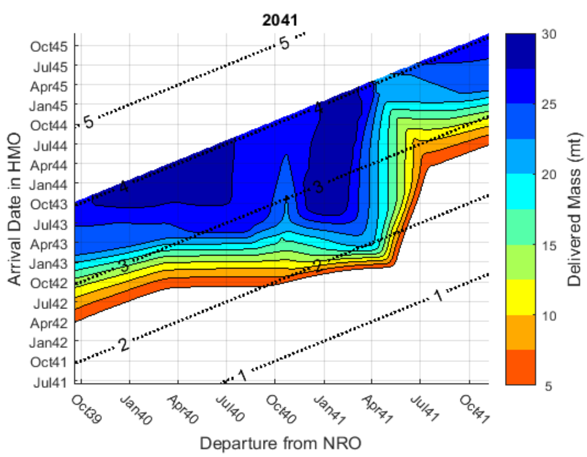

Figure 4 - Bacon Plot for Earth-to-Mars transfers around 2041. Colored contours show the maximum delivered cargo mass to HMO for any date pair over one synodic period. The diagonal dashed lines show constant transfers times in years. This includes 3 months to leave NRHO and 1 month to spiral down to HMO.

Figure 4 shows contours of the maximum cargo mass that can be delivered to HMO by the SEP tug in the 2041 opportunity. (Similar bacon plots exist for the other opportunities). Since this plot shows deliverable cargo, it does not include the 8 mt for the SEP tug and the 2 mt for the xenon needed for the tug to return to NRHO. We also allocate mass for 10% propellant margin on all xenon. The resulting mass is what can be delivered to HMO. In the case of cargo delivery, some fraction of it must be allocated for the mass of the container vessel and docking mechanisms.

One of the key features of the SEP bacon plot is that a feasible trajectory exists for any launch date. However, the effects of the planetary synodic period are still present. There are only certain times where fast transfers (~2 years) are possible. These dates roughly correlate with the natural ballistic opportunities. The other feature to note is the nearly constant arrival date for a given mass over a very long span of launch dates. Following the light blue contour (20 mt) in Figure 4 shows that the arrival date at Mars is around February of 2043 for launches from late 2039 until April of 2041. After that point the Mars arrival date jumps to mid-2045 and the pattern repeats. Additionally, note that the departure from NRHO need not correlate with the date of the actual launch, which could happen months or years earlier at the convenience of launch pad operations.

Note that the cut-off of data longer than 4 years of transfer time on the upper left of Figure 4 is simply due to the bounds of the parameters explored. Feasible trajectories exist for all durations longer than this, presumably with delivered masses in the "deep blue" range of near 30 mt as SEP transfers tend to get more efficient as time-of-flight increases. There is a natural asymptote as the transfer ΔV approaches that of a Hohmann transfer (which in this cases is very close to 30 mt).

3.2 SEP Tug Sizing

The 150 kW SEP tug used in previous studies was chosen as representative, having approximately the right size and thrust to push mission elements up to 30 mt. It was not, however, optimized for maximum delivered mass, minimum flight time, etc. The trades between power level, number of engines, thrust level, specific impulse, and the design of the spacecraft itself are all interrelated and highly dependent on the trajectories themselves – both outbound and inbound. In order to take a deeper look at SEP tug sizing for human missions, all of these trades need to be considered simultaneously.

A specialized tool (the Mars ORbiter Tool, or MORT) has been developed at JPL [23] that takes into account all of the interrelated sizing of the individual sub-system masses. The total mass then affects the low-thrust trajectories which needs to be re-optimized, and the process repeats. For this reason, it is necessary to create a large database of outbound and inbound trajectories, similar to the creation of Bacon plots. However, in this application we are only concerned with maximum delivered mass vs. time-of-flight, not specific dates. Also, power levels must also be varied in order to size the solar arrays and optimize that system.

At the core of MORT is a set of parametric relationships which estimate subsystem masses. The process of creating a mission begins with selecting some basic parameters such as a thruster set, initial power level, launch year, desired LV, etc. (see Figure 5). From there a first estimate of subsystem masses is made using parametric relationships. The trajectory module then builds up the spacecraft backwards in time. The spacecraft dry mass must be delivered to Earth (or NRHO in this study) at the end of a round-trip mission. This mass is passed to the trajectory database where a return trajectory is interpolated based on power and mass. The propellant required for this leg sets the mass in Mars orbit prior to departure for Earth. Mass for Mars maneuvers and the drop-off mass is then added to find the total mass that needs to be delivered to orbit from an outbound Earth-to-Mars trajectory. This trajectory is then interpolated based on power level and mass. After a loop of the trajectory module is completed, desired margins are added and the required propellant mass is sent to the spacecraft module as an input for another iteration. This process continues until a fully-converged, fully-optimized mission is found. Additional optimizations over power, thruster modes, payloads, etc, are performed automatically by iterating on the entire model.

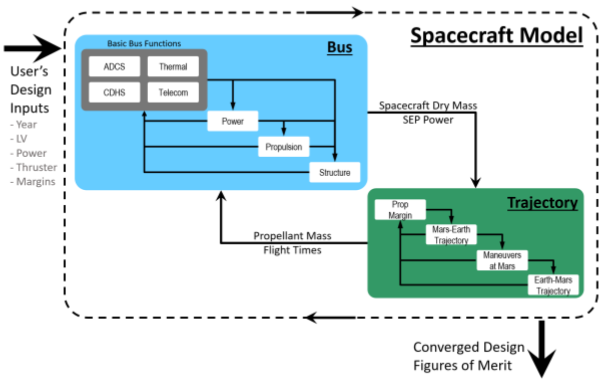

Figure 5 - Process flow for spacecraft mission design tool - MORT. Not only are the spacecraft sub-systems interrelated, but the SEP trajectories themselves are part of the iteration loops.

In typical MORT analyses, it is possible to run a complete sweep over all power levels to find the optimum, above which adding more power shows diminishing returns in terms of delivered mass. In this case, however, the starting point is a fixed 40 mt + tug dry mass at a C3 of 2 km^2/s^2. This means that the maximum drop-off mass will be reached when the low-thrust trajectories approach the theoretical lower limit on ΔV of about 4.5 km/s, each way. Power levels from 100 – 200 kW were all found to deliver a maximum payload of around 30 mt to HMO. For power levels much below 100 kW, the trajectories would begin to fail to converge and no solutions were found for reasonable flight times and maximal payload – effectively setting a lower limit for SEP tugs with ARRM-like Hall thrusters.

Low-thrust trajectories become more geometrically efficient with longer flight times, allowing them to use less propellant. This is evident in Figure 4 as the contour lines of delivered mass increase towards the upper-left (longer time-of-flight) of the plot. Since thrust scales almost linearly with power, higher power means that geometric efficiency is achieved for shorter times-of-flight. Figure 6 shows the total optimized transfer time-of-flight vs. power (from 100-200 kW) as calculated by MORT. In each case, the tug is pushing the maximum 30-32 mt to HMO. For the lowest powers, Earth-Mars transfers take nearly 3.5 years, whereas for the highest power that time can be as low as 2 years. For missions where time is important, and/or the SEP tug needs to return for the next mission within 4-5 years, a higher power system would be useful.

For the return leg, the tug is unencumbered by the large cargo element and can achieve a very high thrust-to-mass ratio and return quite quickly, sometimes cutting a year or more off of the interplanetary transfer, as shown by Figure 7. In fact, an example is shown in a previous paper by the authors [3] showing a round trip delivering 26 mt and returning to NRHO in less than 4 years. The bacon plots show that these type of "fast" missions are only available when the departure dates can be optimized, otherwise, round-trip times around 6 years are more common. The window for "fast" transfers with large payloads is only open for powers greater than ~150 kW, and is longer (many months) for the highest powers.

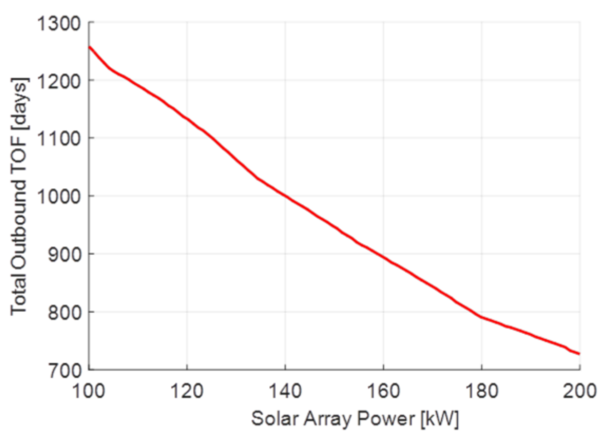

Figure 6 - Earth-Mars transfer times for varying SEP tug power levels. The mass delivered to HMO is 30 mt. Below 100 kW some trajectories fail to converge.

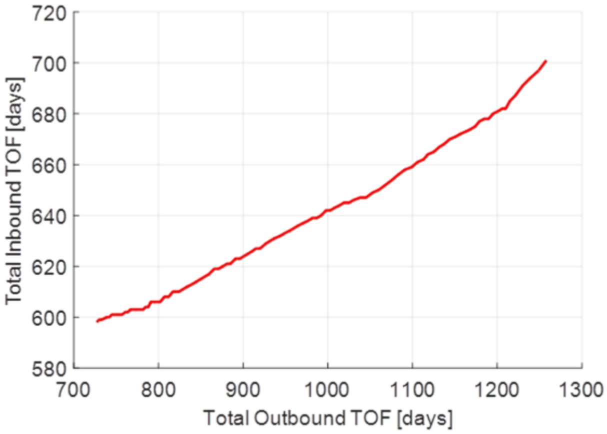

Figure 7 – Inbound (Mars-Earth) vs. Outbound (Earth-Mars) trip times for SEP tugs delivering 30 mt to HMO. Power varies from 200 kW (shortest times) to 100 kW (longest times).

For the reusable SEP tugs in this study, multiple missions with differing parameters and constraints will need to be performed with the same tug design. This is why it would be prudent to select a design that is robust and capable. The total dry mass of the tugs analyzed by MORT were fairly insensitive to the total power, ranging from 7.7 mt to 9.2 mt for 100 kW to 200 kW. The total xenon mass was consistently less than 10 mt for 30 mt of delivered payload, with more than 80% used on the outbound leg. For much faster trips with lighter payloads, it is possible that up to 15 mt of propellant could be needed.

Each SEP tug design carried 10 HERMeS engines, even though it might not be mass optimal to do so. For example, the power at Mars is roughly 1/3 of what it is at Earth, meaning that only 40-70 kW of power would be available at Mars to power the engines, which is just 3-5 of the 14 kW thrusters. The optimal number of engines is typically closer to the average mission power divided by the thruster max power. But for this application, the extra engines are useful for spares and lifetime requirements as the tug makes many trips.

4. Conclusions

Given the investments being made by NASA today, such as the SLS and HERMes thrusters, we find that the reusable SEP tug architecture is highly beneficial to the logistics of a sustainable Mars outpost. With the performance of SLS Block 1b and the expected performance of ARRM-derived SEP engines, both surface and Mars orbit logistic supply to Mars are enabled via an effective cadence of both ballistic and low-thrust transfers. With these vehicle investments and their respective capabilities for transferring cargo to Mars, a sustainable human Mars architecture becomes not only possible, but probable within the next twenty years.

For maximum flexibility and robustness, it is recommended that a reusable SEP tug have at least 140 – 180 kW of power, with margin. It should also be capable of carrying up to 15 mt of propellant. Our analysis shows that such a SEP tug would likely weigh between 8 and 9 mt. This tug would be capable of delivering up to 30 mt (out of 40 mt from an SLS 1b) from NRHO to HMO and return in 4-6 years. This is a 50% increase vs. a direct ballistic transfer, and is not constrained to the typical short (~weeks) launch opportunities every 26 months.

Acknowledgements

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