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ANALYSIS OF CISLUNAR TRANSFERS DEPARTING FROM A NEAR RECTILINEAR HALO ORBIT USING SOLAR ELECTRIC PROPULSION

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An analysis is completed to support the design of optimized trajectories of a massive spacecraft from an L2 Southern NRHO to a Distant Retrograde Orbit and a L2 Northern NRHO using a Solar Electric Propulsion System (SEP). An optimized trajectory is developed for each transfer for a 54 t spacecraft utilizing a 26.6 kW SEP system. A parameterization is developed for each reference transfer to allow analysis of the sensitivity of the trajectory to changes in vehicle mass, SEP power, and Ion Propulsion System (IPS) performance. Required Δv for each transfer is characterized by the initial acceleration of the spacecraft, thus allowing trajectories to be assessed over a wide range of vehicle mass and SEP power inputs. This approach is shown to be useful in identifying optimal IPS configurations for minimizing propellant requirements for the reference transfers. Additionally, the analysis identifies regions where increases in SEP power do not immediately result in a corresponding decrease in required propellant as well as highlights the relative sensitivities of propellant requirements to changes in IPS thrust and specific impulse.

INTRODUCTION

As part of humanity's return to the moon, a Gateway has been envisioned as an orbital outpost in cis-lunar space. The Gateway will utilize an L2 southern (L2S) Near Rectilinear Halo Orbit (NRHO) as a staging orbit from which to provide support to crewed and un-crewed missions in the lunar vicinity.[1] Utilizing a low thrust solar electric propulsion (SEP) system, the Gateway will be capable of transferring to a multitude of other orbits in cis-lunar space. Two destinations are presented in this paper, a 70,000 km Distant Retrograde Orbit (DRO) and an L2 northern (L2N) NRHO. These destinations are presented due to their ability to support varying mission objectives outside of the capabilities of the staging L2S NRHO.[1-2] Additionally, the use of the DRO as a potential disposal orbit at Gateway end of life (EOL) is considered. A one-way reference transfer has been designed for each of these destinations to minimize propellant usage. Two of these references (NRHO to DRO and L2S to L2N) are restricted to a one-way time of flight of approximately 6 months, while the EOL disposal trajectory is limited to 12 months one-way.

The design of optimized throw thrust trajectories for these transfers is complicated by continuously evolving vehicle configurations and analysis objectives. A parameterization-based method

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has been developed to allow for the assessment of these trajectories given varying spacecraft mass, power, and IPS configurations. This methodology allows for rapid trades to be conducted to understand the impact of varying design decisions on propellant requirements without the need for heavy computing resources.

Of particular interest when discussing the vehicle end of life disposal into a long-term storage orbit is the power available to the SEP system. The methodology presented in this paper offers unique insights into the sensitivity of the reference trajectories to decreases in available input power. These results are used to inform design decisions regarding expected system performance at varying stages in the mission lifespan.

GROUND RULES AND ASSUMPTIONS

As Gateway, and the Artemis program as a whole, continues to evolve, the ground rules and assumptions (GRA) do as well. The following outlines a current snapshot in time of the GRA that is used to conduct the analysis presented in this paper.

Spacecraft Assumptions

Over the duration of the Gateway mission, it is expected that the spacecraft configuration will grow and evolve to the current mission objectives. To develop the reference trajectories, however, a fixed configuration is assumed. An initial vehicle mass of 54,400 kg and an ion propulsion system (IPS) consisting of two 13.1 kW hall effect thrusters is assumed for the L2S to DRO and L2S to L2N transfers. These thrusters are continuously throttleable over their operating range and their performance data are listed in Table 2.[3] For the EOL disposal to DRO trajectory, a slightly lower vehicle mass of 53,000 kg and a 4x 6.5 kW class IPS system is used. A duty cycle of 90% is applied to the operation of these thrusters to account for possible thruster outages and any other non-thrusting periods.

Table 1. Summary of the spacecraft GRA

Parameter	Value	Notes
Initial Mass	54,400 kg	L2S to DRO and L2S to L2N
	53,000 kg	EOL to DRO
Power	26.6 kW	Maximum available power
IPS Configuration	2x 13.1 kW class	L2S to DRO and L2S to L2N
	4x 6.5 kW class	EOL to DRO
Duty Cycle	90%	

Table 2. 13.1 kW class performance with input power[3]

Power (kW)	Thrust (mN)	Specific Impulse (s)	Mass Flow Rate (kg/s)
13.1	579	2600	22.70
13	576	2588	22.69
12	546	2474	22.53
11	517	2359	22.35
10	488	2244	22.15

Cislunar Destinations

The staging orbit is assumed to be the Gateway reference, a 9:2 Lunar Synodic Resonance L2 southern NRHO with a period of roughly 6.6 days. This staging orbit was selected for numerous reasons, including its nearly stable behavior, avoidance of Earth eclipses, and low perilune altitude (to allow easier access to the lunar surface).[2,4-5]

The first destination is a 70,000 km radius Distant Retrograde Orbit (DRO) with an orbital period of roughly 13 days. DROs display long term stability, making them an attractive destination for both long term storage or vehicle disposal.[2]

The second destination is a northern L2 NRHO (L2N) with a perilune radius of 5,800 km and an orbital period of roughly 7.4 days. This orbit is similar in characteristics to the staging L2S, although mirrored into the northern hemisphere. This orbit provides offers the advantage of extended communications access to the lunar north pole.

The third destination is also the 70,000 km DRO, however it is targeted for end of life vehicle disposal. This transfer is restricted to 12 months one-way, as opposed to the 6 month one-way limit imposed on the other two reference trajectories. This is an attractive option for long term storage of the vehicle at end of life not only due to the stability of the orbit, as discussed above, but also due to the relatively low cost of the transfer.

REFERENCE TRANSFERS

Each reference transfer is designed as a one-way trajectory from the staging NRHO to the destination of note. The total time of flight for these trajectories is restricted to 12 months. For the roundtrip transfers this means a ToF of 6 months one-way, while the EOL disposal is restricted to 12 months one-way. Each transfer is designed in Copernicus[6] and utilizes a parallel monotonic basin hopping script to aid the optimization of the trajectory.[2,7]

L2S NRHO to 70,000 km DRO

The one-way reference trajectory for the staging NRHO to the DRO is shown in Figure 1. There are 5 thrust arcs (presented in red, coasts shown in blue) required to complete this trajectory. As the vehicle must move from a nearly polar to a nearly equatorial orbit, a large plane change occurs during this transfer. This is completed through a revolution about the Earth followed by a targeted lunar fly-by. Two thrust arcs are required to leave the NRHO and the lunar vicinity, a single thrust arc occurs during the Earth revolution to target the lunar fly-by, and a final arc is required to capture into the DRO. This trajectory requires a total of 97 m/s of Δv for 202 kg of propellant and a time of flight of 155.7 days.

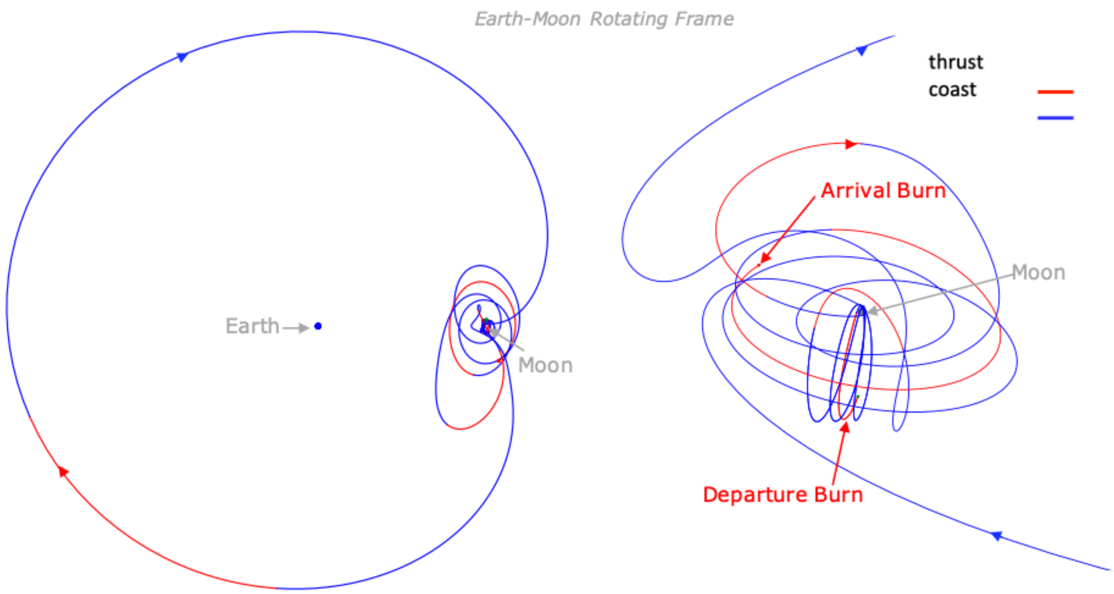


Figure 1. Transfer from the L2S NRHO to the DRO shown in the Earth-Moon Rotating Frame. Zoomed in view of the departure and arrival burns to the NRHO (right) and a wide view of the trajectory (left)[2]

Table 3. L2S NRHO to DRO trajectory summary

Parameter	Value
Time of Flight (days)	155.7
Propellant Mass (kg)	202
Δv (m/s)	97.1
Number of Thrust Arcs	5

L2S NRHO to L2N NRHO

The one-way reference trajectory for the staging L2S NRHO to the L2N NRHO is shown in Figure 2. There are 12 thrust arcs (presented in red, coasts shown in blue) required to complete this trajectory. The primary challenge to this trajectory is flipping the orbit over the x-y plane (as views in the Earth-Moon rotating frame). A nearly complete flip in plane angle is achieved through a single revolution about the Earth. Multiple targeted lunar fly-bys then lead to the vehicle arriving in the target L2N NRHO. This trajectory requires a total of 42.3 m/s of Δv for 88 kg of propellant and a time of flight of 145.5 days.

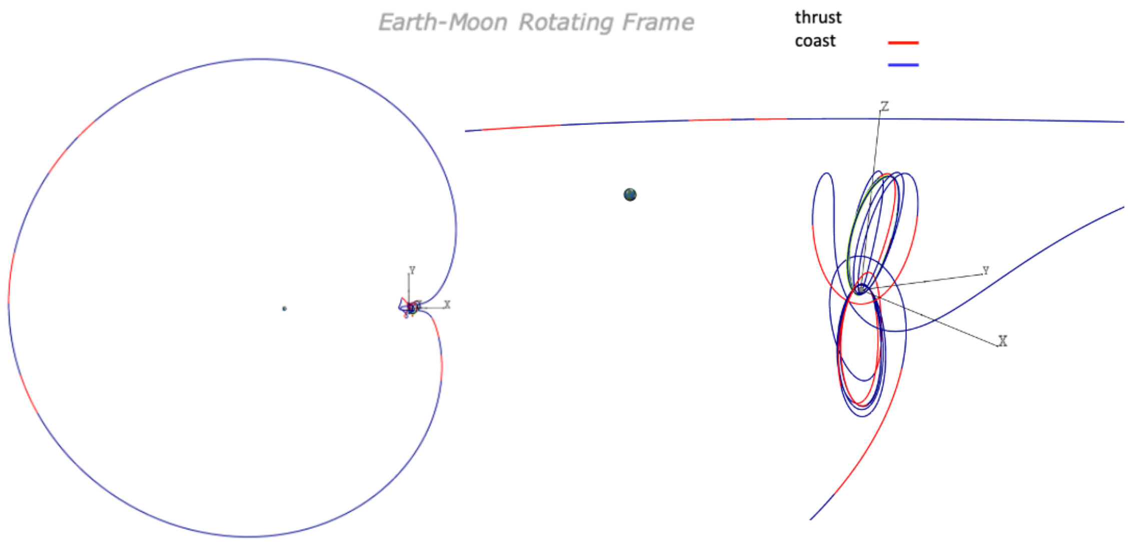


Figure 2. Transfer from the L2S NRHO to the L2N NRHO shown in the Earth-Moon Rotating Frame. Zoomed in view of the departure and arrival burns to the NRHO (right) and a wide view of the trajectory (left)

Table 4. L2S NRHO to L2N NRHO trajectory summary

Parameter	Value
Time of Flight (days)	145.5
Propellant Mass (kg)	88
Δv (m/s)	42.3
Number of Thrust Arcs	12

L2S NRHO to EOL Disposal

The one-way reference trajectory for the staging L2S NRHO to EOL disposal into the DRO is shown in Figure 3. There are 11 thrust arcs (presented in red, coasts shown in blue) required to complete this trajectory. As with the initial transfer to the DRO presented above, the plane change required to enter into a nearly equatorial orbit is a large challenge to this trajectory. The longer allowable time of flight of roughly 12 months allows for a much longer ballistic coast around the Earth as compared to the initial transfer to the DRO. The ballistic phase includes a lunar fly-by before kicking back out into figure 8 pattern seen in Figure 3. This translates into lower Δv costs of 37.3 m/s as compared to 97.1 m/s. This trajectory utilizes a lower initial vehicle mass of 53,000 kg (compared to 54,400 kg) and a lower performance IPS configuration, leading to slightly higher propellant costs versus the L2S to L2N (101.7 kg to 88 kg). The longer allowable TOF leads to a transfer time of 379.5 days for this trajectory.

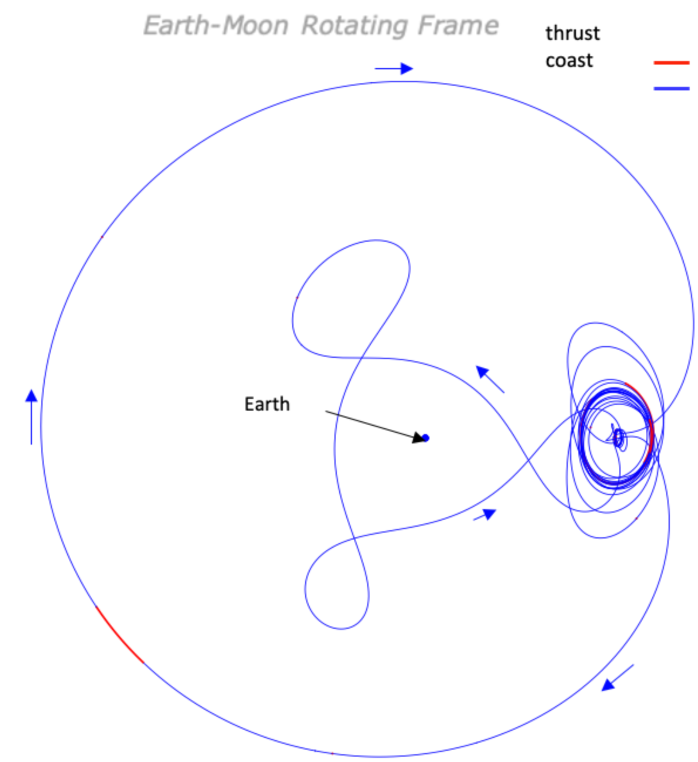


Figure 3. Transfer from the L2S NRHO to the DRO for end of life disposal shown in the Earth-Moon Rotating Frame

Table 5. L2S NRHO to EOL DRO trajectory summary

Parameter	Value
Time of Flight (days)	379.5
Propellant Mass (kg)	101.7
Δv (m/s)	37.3
Number of Thrust Arcs	11

IMPACT OF VEHICLE CONFIGURATION

The reference trajectories described in this paper were designed with a single vehicle design point in mind. A single configuration for vehicle mass, input power, and electric propulsion system was used to develop each transfer. These parameters, however, are not necessarily frozen. As part of the ongoing development of Gateway, the vehicle configuration continues to evolve. Likewise, Gateway is not a static vehicle and the makeup of the station while on-orbit in the NRHO is a function of time. As the buildup of Gateway continues and modules arrive at and depart from the station, the vehicle mass will change.[8] Over time, the availability of thrusters and power may also vary. As the vehicle approaches the end of its intended mission duration, systems such as solar arrays and thrusters may experience performance degredation. These factors all mean that it is desireable to assess the effects of varying spacecraft configurations on the reference trajectories.

Varying mass, power, and thruster combinations can produce significantly different performance conditions. Higher mass or lower input power would result in a lower initial vehicle acceleration along its trajectory. Significant variations in power may allow for a greater number of thrusters to be operated simultaneously (or force fewer thrusters to be utilized to reduce power consumption). The combination of so many variables makes it difficult to assess the impact on trajectory duration and required propellant without further analysis.

A methodology is thus developed to allow for rapid trades to be conducted for each reference transfer with respect to varying vehicle parameters. With data generated from high fidelity models, analysis of a wide range of mass, power, and IPS conditions can be conducted independently from the design of the trajectories themselves. This allows the mission designer to assess a wide range of possible spacecraft conditions without the need for intensive computing resources.

For each transfer, a parameterization of required Δv versus initial spacecraft acceleration is developed. Beginning with the initial conditions specified in Table 1, the initial spacecraft mass is increased in increments of 1,000 kg until a point is reached where the transfer is no longer feasible. This point represents the minimum vehicle acceleration required to complete the trajectory in the specified timeframe. A polynomial curve fit is applied to this data, which is then used to conduct further analysis. It is ensured that the geometry and time of flight for each solution remains similar throughout the sweep.

The relative sensitivities of each transfer to changes in initial spacecraft acceleration can be seen in Figure 4. The curves are bounded by the maximum reasonable spacecraft acceleration (set a vehicle initial input mass of 39,000 kg) and the feasible minimum limit (indicated by the blue box). It can be seen that the L2S NRHO round trip transfer to the DRO is by far the most costly of the three trajectories in terms of Δv. Because this transfer is significantly more difficult to achieve than the other two, the minimum feasible acceleration for this round trip to the DRO was used to provide a reasonable universal limit for all three trajectories. Thus, feasible solutions exist within the blue region of the chart for the L2N round trip and the EOL disposal trajectories. Likewise, it should be noted that solutions also exist below this limit for the DRO round trip if the one-way TOF is lengthened from the 6 month limit imposed in this analysis. From these curves, the Δv required for a given initial acceleration can then be used to determine the required propellant and time of flight for a specific mass and thruster configuration combination.

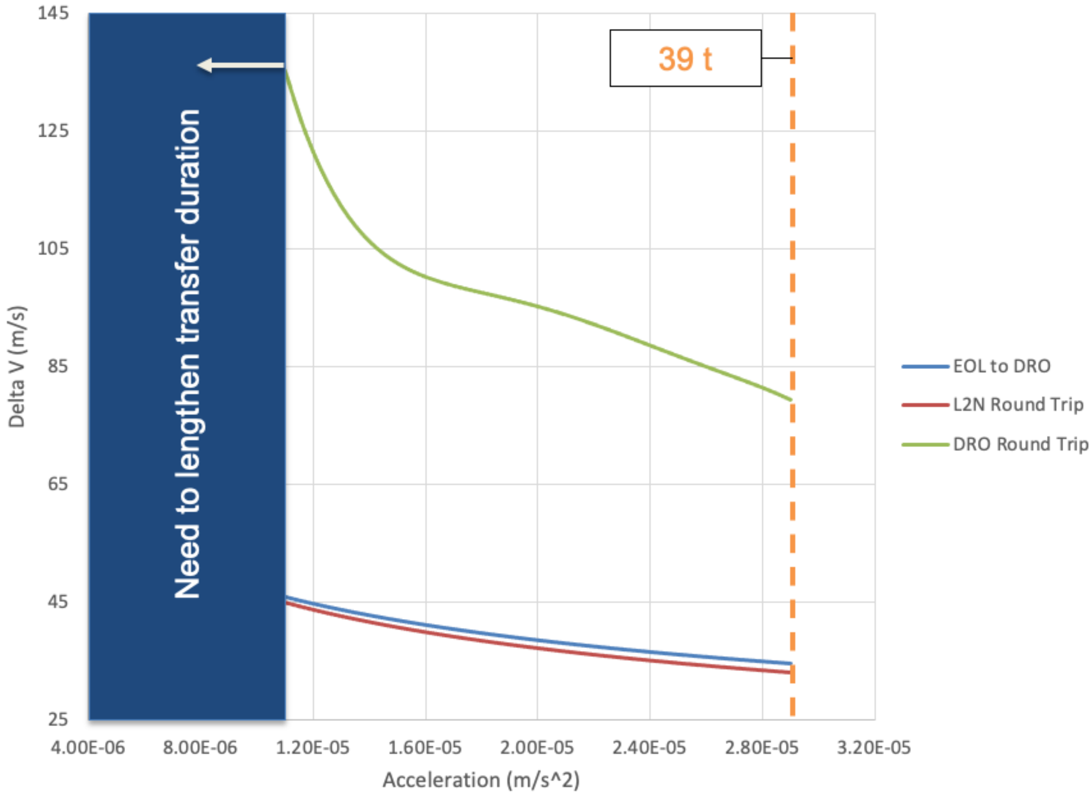
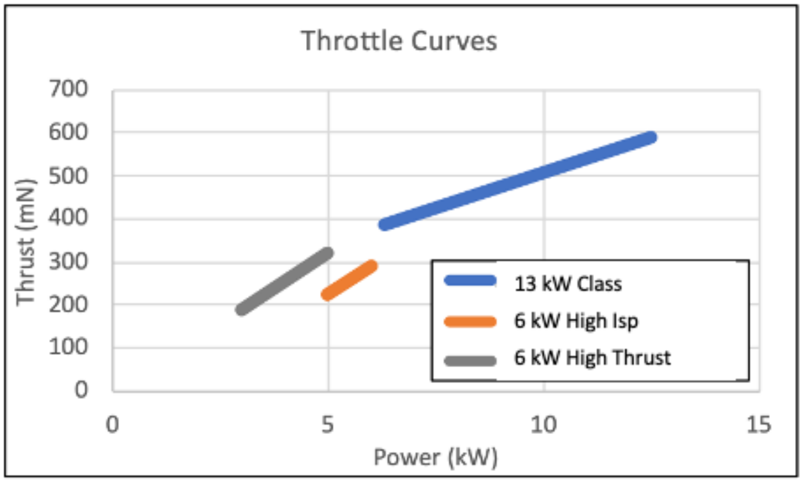


Figure 4. Δv vs Spacecraft Acceleration Curve Fits

For any spacecraft mass, the initial acceleration is based upon the total thrust of the IPS configuration. The chosen thruster combination also determines specific impulse and mass flow rate, which can be used to calculate required propellant and thruster on-time. Thus, thrust and mass determines *Δv*, and *Δv* in conjunction with specific impulse determine propellant usage. In the development of the reference transfers a straightforward IPS of two 13 kW class thrusters was utilized. The optimum combination of thrusters, however, can be much more complex.

Gateway is envisioned to utilize a combination of 13 kW and 6 kW class thrusters, the details of which have been published previously.[9] These thrusters have significantly different performance parameters as well as varying operating power ranges, as seen in Figure 5. Further complicating the analysis, the 6 kW class thrusters have two different operating potentials. This allows these thrusters to be operated in a 'High Thrust' or a 'High Isp' mode, indicated on the performance charts. The mission designer thus effectively has three different types of thrusters from which to choose an optimum IPS. Furthermore, 7 thrusters (3 of the 13 kW class and 4 of the 6 kW class) are expected to be available, resulting in a large number of possible thruster type and mode combinations from which to generate optimized results.



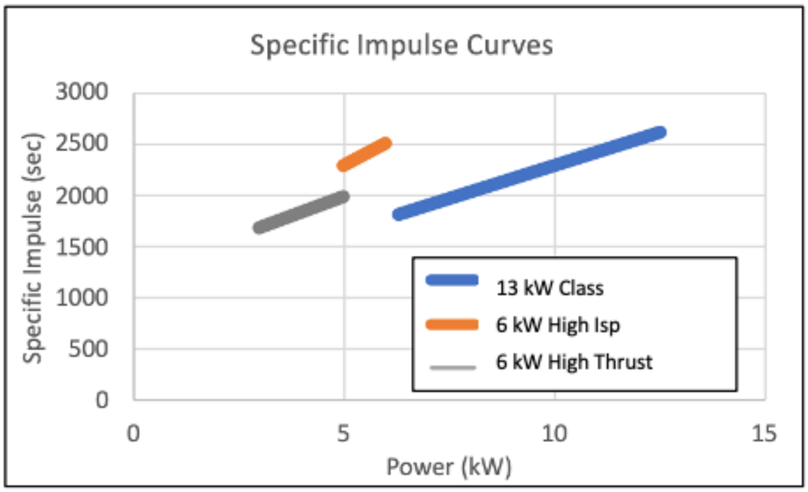


Figure 5. Performance curves (thrust, top and *Isp*, bottom) for 13 kW and 6 kW Class Thrusters[9]

The choice in thruster combination determines thrust and specific impulse. For any given IPS configuration, these two parameters are also functions of power (as seen in Figure 5). Acceleration is determined by thrust and mass, and so is truly a function of power and mass. With a simple rearranging of the rocket equation, and our aforementioned parameterization of Δv with respect to acceleration (power and mass), it can be seen that the methodology boils down to a relatively simple relation (Equation 1).

$$\Delta v = \mathrm{F(a)} = f(P, m)$$

$$I_{sp} = g(P)$$

$$M_{propellant} = M_0(1 - e^{\frac{f(P,m)}{g(P) \cdot g_0}}) \tag{1}$$

L2S NRHO to 70,000 km DRO

With the aforementioned approach, the plethora of possible thruster, initial mass, and SEP power combinations can be navigated. The results of this analysis for the roundtrip transfer from the L2S NRHO to the DRO are presented in Figure 6 (deterministic roundtrip results assume equivalent outbound and return one-way trajectories). From these results, it can be immediately seen that the optimum thruster configuration for limiting xenon propellant usage is a complex framework based on vehicle mass and available power. Feasible transfers exist within the region covered by the colormap, with the regions outside of the shaded boundaries represent infeasible vehicle parameters. A continuous solution field exists over a wide range of available powers stretching down to 10 kW, well below the reference power of 26.6 kW. Vehicle mass must be limited to under 45 t at the low end of available power, leaving a considerable negative margin on the reference mass of 54.4 t. The penalties for reduced power are evident in the xenon contours, where it can be seen that reducing available power from 26.6 to 10 kW results in a near doubling of propellant requirements. Modest reductions in available power, however, will incur modest penalties. For instance, a reduction of 6 kW from the reference value will result in an additional 40 kg of xenon usage for a 39 t vehicle. This effect is exacerbated for heavier vehicles, however, with the same power reduction adding a penalty of roughly 400 kg of xenon for an 80 t spacecraft.

Also of interest in these results is the behavior of the colormap itself. The amount of power available determines the number of thruster configurations that are feasible to operate. Given that the two types of thrusters envisioned for use on Gateway differ significantly in terms of thrust, power consumption, and efficiency, the thruster combination that will produce the lowest xenon usage is not straightforward. As discussed above, the mass of xenon required to complete the trajectory is a function of mass and power due to the parameterization of Δv with respect to thrust and spacecraft mass. This means that the highest specific impulse thruster combination is not necessarily the most optimum configuration to achieve the lowest propellant mass. This combination is a subtle balance of thrust and efficiency. It thus makes sense to utilize the 13 kW class thrusters, which are capable of superior thrust and specific impulse relative to the 6 kW class, as much as possible. The higher power consumption, and steeper throttle curves, of the 13 kW thrusters means that they are confined to the upper end of the solution field, where enough power exists to run these thrusters at a high enough power to take advantage of their capabilities. At lower powers, there are multiple solutions where it is advantageous to use the 6 kW class thrusters (in both operating modes) either in combination with a single 13 kW class thruster or by themselves. There are also multiple regions, notably in the deep blue region near 20 kW, where performance plateaus. These regions show points in the power spectrum where there is not enough power to run an additional 13 kW thruster at a throttle point high enough to produce better IPS performance than a combination of a single 13 kW and a single 6 kW class thruster. In these plateaus the excess power available does not provide any boost in performance to the trajectory.

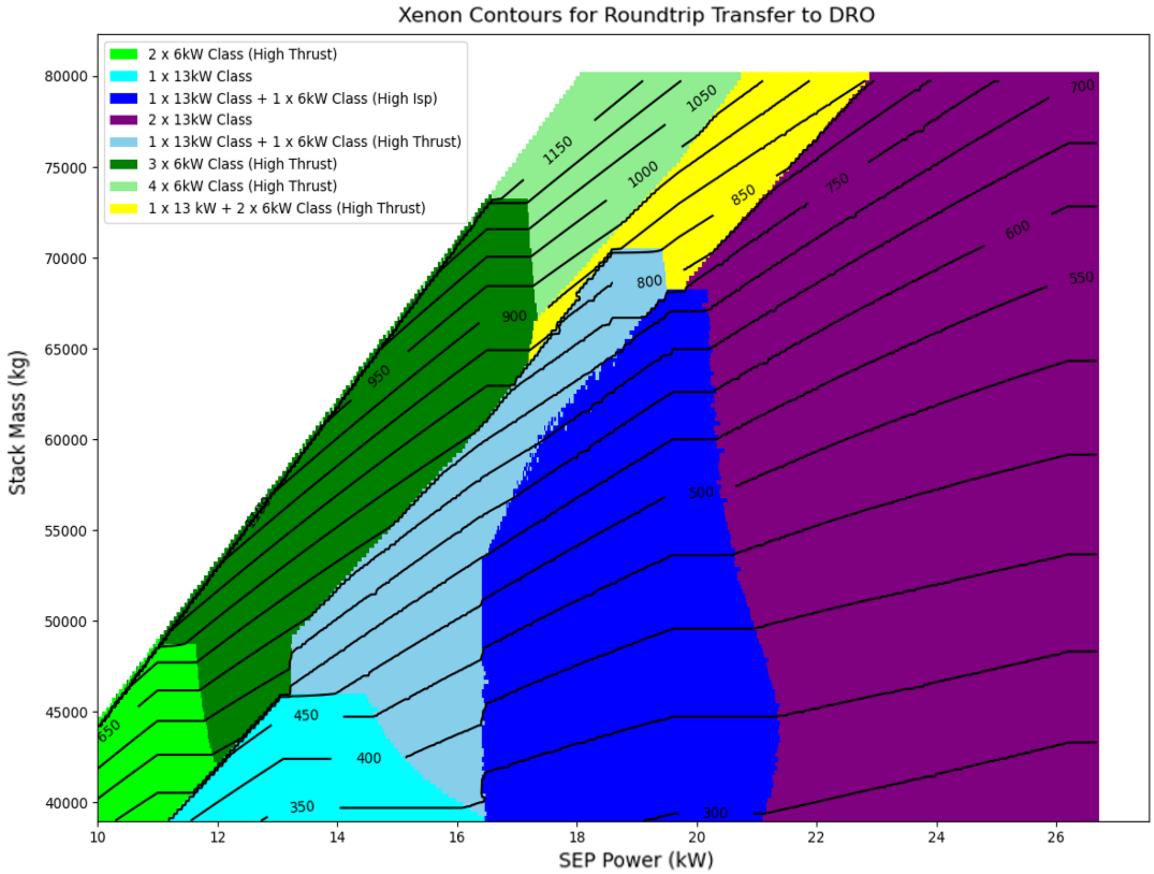


Figure 6. Contour plot for xenon required to complete a roundtrip transfer to the DRO with an optimal thruster configuration

L2S NRHO to L2N NRHO

The same methodology is applied to the roundtrip transfer from the L2S NRHO to the L2N, with results presented in Figure 7. These results are similar in appearance and behavior to those of the roundtrip to the DRO, with a few notable differences. As shown in Figure 4, the Δv required to complete the DRO trajectory is significantly higher than that required for either a transfer to the L2N or the end of life disposal to the DRO. As such, the propellant required to transfer to the L2N is significantly lower at any given power and mass combination. While the optimal thruster configurations are similar for both the DRO and L2N transfers, there is an extra IPS combination for the L2N. The lower Δv requirements translate into lower thrust requirements for any given mass and power combination. This allows for a small region at high initial masses and modest power to utilize a single 13 kW class thruster and three 6 kW class thrusters operating in high thrust mode (this region is shaded orange). This configuration has a slightly higher specific impulse than the neighboring configuration of four high thrust 6 kW class thrusters (pale green). In the DRO case, this entire region of the plot is shaded pale green as the higher thrust, lower specific impulse combination is needed due to higher Δv.

Additionally, the contour plot shows that the L2N transfer is less sensitive to changes in mass and power than the roundtrip to the DRO. This can be seen in the figures, with the relatively constant spacing between contour lines in Figure 7 as compared to a continuously shrinking spacing in Figure 6 as mass increases or power decreases. This behavior can also be observed in the Δv data in Figure 4. The DRO Δv requirements begin to grow dramatically as spacecraft acceleration is decreased whereas the same curves for the L2N and the EOL transfers remain much more flat in comparison.

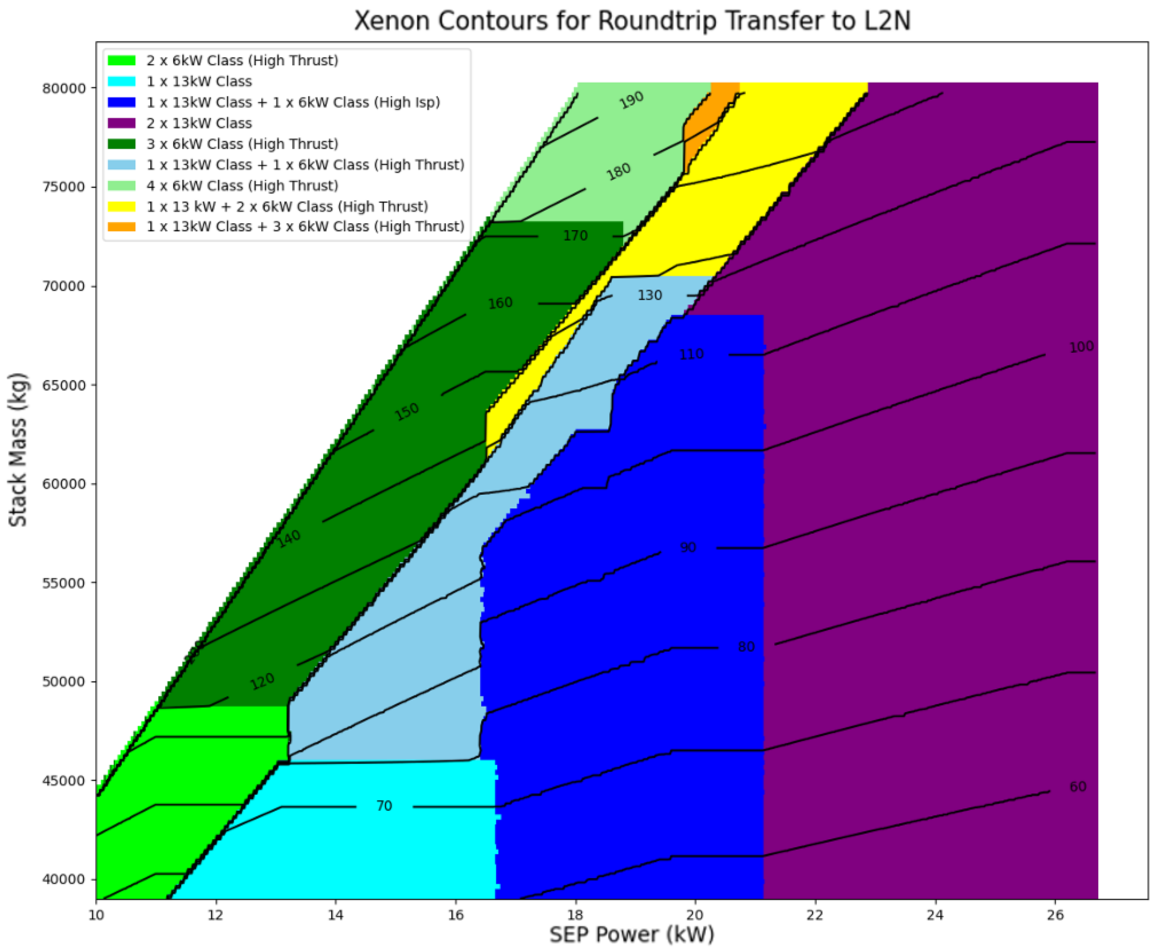


Figure 7. Contour plot for xenon required to complete a roundtrip transfer to the L2N with an optimal thruster configuration

L2S NRHO to EOL Disposal

Finally, the same analysis is conducted on the one-way transfer from the L2S NRHO to the DRO for end of life disposal of the spacecraft and presented in Figure 8. As discussed during the description of the reference trajectories, the key difference between this transfer and the other two is that this is a one-way trajectory confined to a transit time of 12 months. The remaining two trajectories are roundtrip transfers restricted to 6 months each way (for a total of 12 months to depart and arrive back at the L2S NRHO).

With this key difference in mind, it can be seen that the results for the EOL transfer are nearly identical to those of the L2N trajectory. In Figure 4, it can be seen that the Δv curves for these two transfers are extremely close, differing between 8 and 15 m/s across the acceleration bounds. Likewise, the propellant requirements and optimal thruster configurations are very similar for these two transfers.

As with the other two transfers, regions exist where additional power does not immediately result in increased performance for the vehicle. One such region exists in the blue shaded regions near 20 kW for all three transfers as well as at the upper end of the power spectrum where the two 13 kW class thrusters reach their maximum throttle settings.

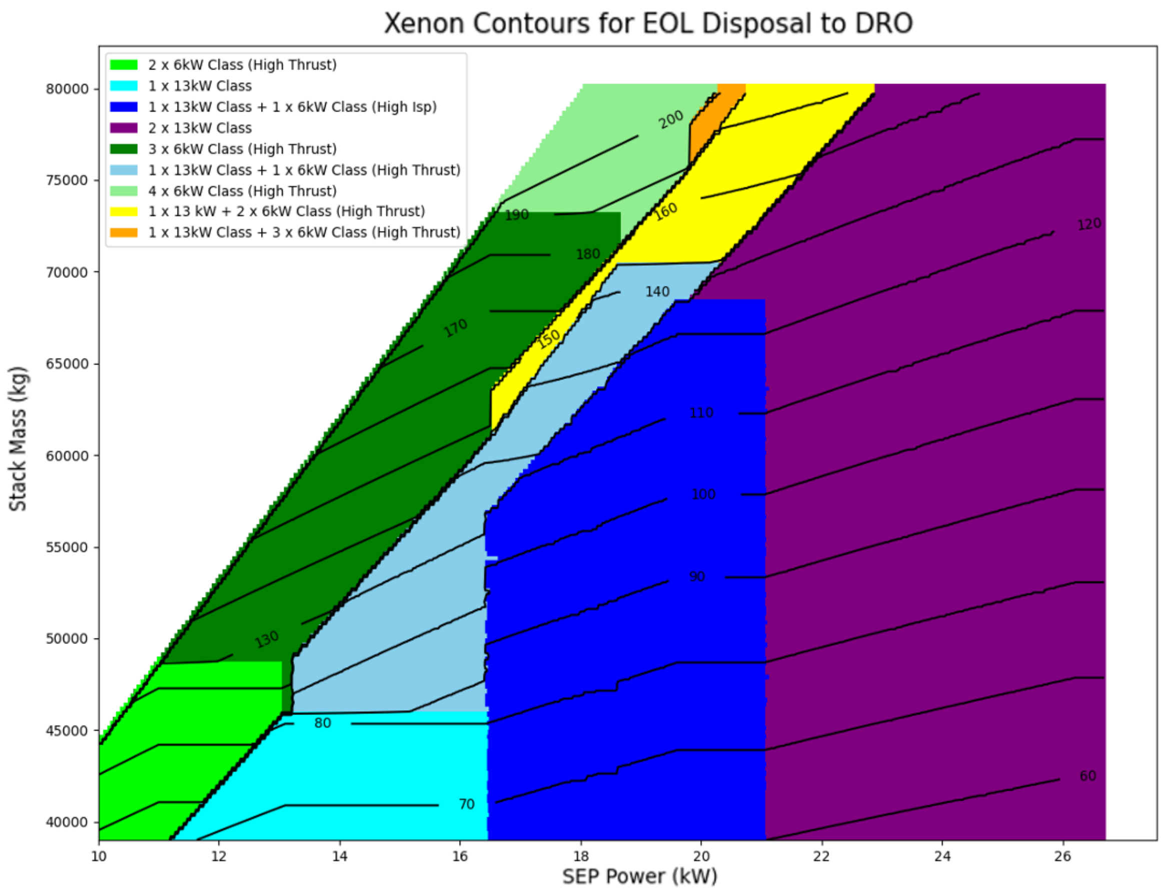


Figure 8. Contour plot for xenon required to complete the EOL disposal to the DRO with an optimal thruster configuration

Sensitivity to Power Reduction

Of particular note for the reference trajectories described in this paper is the dependency of propellant mass on not only specific impulse but also thrust. As power is reduced from the reference point of 26.6 kW the IPS will decrease in performance according to the throttle curves of the associated thrusters (Figure 5). Once power has reduced sufficiently, a different thruster combination must be used, meaning that the overall performance curves for thrust and specific impulse are not continuous for a given mass. As shown in the parameterization in Figure 4, Δv is dependent on initial vehicle acceleration. For a constant mass, this means that Δv is entirely (and inversely) dependent on thrust. As the available power decreases one can expect the thrust and specific impulse of the IPS to decrease as well. With both parameters contributing to the amount of propellant required it is not immediately clear which primarily the rise in xenon usage.

The propellant required to complete the L2S to DRO trajectory is shown in Figure 9 as a function of input power. Mass is held constant at 54.4 t for these results. Propellant usage has been normalized against the requirement at the reference power setting and is shown as the black line at the top boundary of the orange shaded region. The blue shaded region represents the Δv required for a particular power input, also normalized by the reference point. As mass is a constant 54.4 t for the results in this plot, Δv can then be seen as a representation of thrust. The remaining orange region, the space between the two curves, thus represents the contribution of the specific impulse of the IPS to the xenon cost (it should be noted that for a fixed specific impulse and varying thrust, or vice versa, these three contours would overlap exactly).

From Figure 9 several interesting behaviors are of note. Firstly is the effect of contributions as power to the IPS is reduced. Near the reference point, where the thrust of the IPS system is at its maximum for this analysis, reductions in thrust and thus increases in Δv drive the increase in propellant. As power is reduced, however, this behavior changes. As the power level drops below 20 kW and less efficient thruster combinations take over, the drop in specific impulse dominates. As the power drops down to 12 kW, the specific impulse contribution to the xenon load is overwhelming.

Also of note is the agnostic behavior of the propellant load with respect to changes in thrust, particularly at low power. As the power increases there are three distinct discontinuities in thrust output (indicated by the vertical red dashed lines). These points represent the change between IPS configurations and the thruster combination being utilized in each of these regions is indicated on the chart. At the first two IPS configuration changes, the normalized Δv significantly increases. This indicates a significant decrease in available thrust as the next combination is utilized. In these two regions the overall xenon usage continues a steady decline. In these IPS switches although the overall thrust of the system is decreasing (causing Δv to increase) the specific impulse increases as a more efficient thruster combination is utilized. At this lower end of the power spectrum, the specific impulse contribution is so overwhelming that decreases in thrust have very little overall impact on the trajectory.

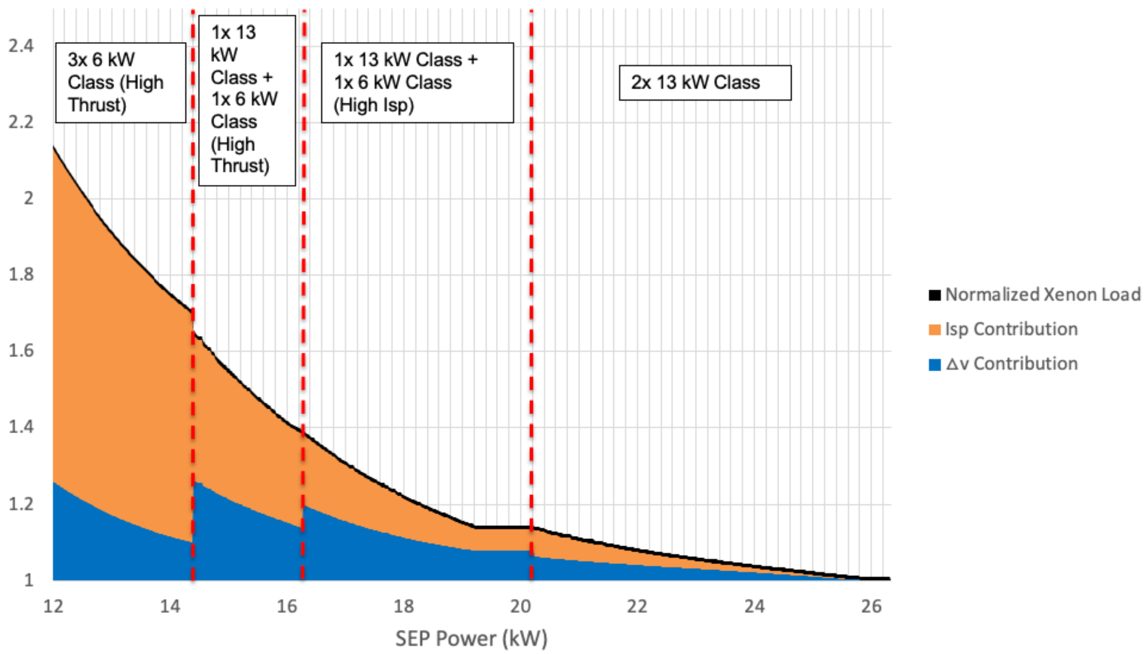


Figure 9. Sensitivity of the xenon load required for the L2S to DRO roundtrip transfer at 54.4 t to reductions in power. Contributions of reductions in IPS thrust and specific impulse are presented

CONCLUSION

The development of three reference trajectories presented in this paper show that a high power SEP system is capable of transfering a large spacecraft between multiple cis-lunar orbits for relatively low propellant costs. The use of an efficient and flexible SEP system is also shown to be capable of tolerating significant changes in both vehicle mass and available power without the loss of mission objectives. Reference transfers to a L2N NRHO and a 70,000 km DRO are demonstrated for a 54.4 t vehicle operating at 26.6 kW. It is then demonstrated that these transfers remain feasible without any changes to the baseline Gateway IPS while varying vehicle mass up to 80 t and reducing SEP power input down to 10 kW.

This analysis is made possible through the development of a methodology by which to characterize these transfers with respect to initial vehicle acceleration. The parameterization presented above allows the mission designer to complete rapid trades across a wide range of vehicle operating conditions and inputs without the need for intensive computing resources. Key insights are provided as to the amount of propellant required to complete a trajectory as well as the optimal thruster combinations for a given spacecraft mass and power. Additionally, the analysis shows functional limits to the IPS where an increase in available power may not immediately result in better performance. The sensitivity of these transfers to reductions in power is also presented, with further analysis showing the contributions of decreases in SEP thrust and specific impulse on propellant needs.

REFERENCES

[1] D. E. Lee, “White Paper: Gateway Destination Orbit Model: A Continuous 15 Year NRHO Reference Trajectory”, *NASA Johnson Space Center*, JSC-E-DAA-TN72594, 2019

[2] McCarty, S., Burke, L., McGuire M., “Analysis of Cislunar Transfers from a Near Rectilinear Halo Orbit with High Power Solar Electric Propulsion”, *AAS/AIAA Astrodynamics Specialist Conference*, 2018

[3] J. Jackson, S. Miller, J. Cassady, E. Soendker, B. Welander, M. Barber and P. Peterson, “13kW Advanced Electric Propulsion Flight System Development and Qualification”, *36th International Electric Propulsion Conference*, 2019

[4] R. J. Whitley, D. C. Davis, L. M. Burke, B. P. McCarthy, R. J. Power, M. L. McGuire and K. C. Howell, "Earth-Moon Near Rectilinear Halo and Butterfly Orbits for Lunar Surface Exploration", *AAS/AIAA Astrodynamics Specialists Conference*, 2018

[5] J. Williams, D. E. Lee, R. J. Whitley, K. A. Bokelmann, D. C. Davis and C. F. Berry, "Targeting Cislunar Near Rectilinear Halo Orbits for Human Space Exploration", *27th AAS/AIAA Space Flight Mechanics Meeting*, 2017

[6] C. Ocampo and J. Senet, “The Design and Development of COPERNICUS: A Comprehensive Trajectory Design and Optimization System”, *57th International Astronautical Congress*, 2006

[7] S. L. McCarty and M. L. McGuire, “Parallel Monotonic Basin Hopping for Low Thrust Trajectory Optimization”, *28th AIAA/AAS Space Flight Mechanics Meeting*, 2018

[8] J. Crusan, J. Bleacher, J. Caram, D. Craig, K. Goodliff, N. Herrmann, E. Mahoney, M. Smith, “NASA’s Gateway: An Update to Progress and Plans for Extending Human Presence to Cislunar Space”, *2019 IEEE Aerospace Conference*, 2019

[9] D. A. Herman, T. Gray, I. Johnson, K. Taylor, T. Lee and T. Silva, “The Application of Advanced Electric Propulsion on the NASA Power and Propulsion Element (PPE)”, *36th International Electric Propulsion Conference*, 2019