Capabilities Enabled in Cislunar Space by Low Specific Mass Solar Electric Power and Propulsion Systems

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The influence of specific mass of the power and propulsion systems for electric propulsion systems is investigated in the context of cislunar applications. An analytical approach to mission analysis is presented to relate requirements such as transfer time and Δv to delivered payload and required system power. Two cases for specific mass are considered, 20 kg/kW to represent the state-of-the-art, and 2 kg/kW to represent next-generation technologies. Three classes of vehicle are considered including a cubesat, ESPA-class small sat, and small launch provider payload. It is shown that with with the proposed reduction of specific mass, propulsion systems of 750 W (cubesat), 5 kW (ESPA-class), and 40 kW (small launch provider) can enable delivery of finite payload to any orbit in cislunar space from a low earth orbit (LEO) within ~ 15 days. These results are discussed in the context of use cases for these classes of spacecraft and contrasted with maneuvers performed with chemical engines. It is shown that for similar maneuvers to geosynchronous orbit and beyond, chemical engines employing hydrazine do not close. This highlights the enabling features of fast-transits with propellant-efficient electric propulsion. The challenges with achieving an order of magnitude reduction in specific mass to achieve this paradigm of fast-transit vehicles is discussed in the context of the state-of-the-art and future technology roadmaps. The limitations and possible extensions of the analysis are also examined.

Nomenclature

α	= power and propulsion system specific mass, kg/kW
$\dot{m}$	= mass flow rate of propellant
η	= total system efficiency
f_{str}	= structural mass fraction
f_{tank}	= tankage fraction
g	= 9.8 m/s^2
I_{sp}	= specific impulse, s
m	= mass, kg
MF	= dry mass fraction
P	= power, W
PPS	= power and propulsion system
t	= time, s
v	= velocity, s

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I.   Introduction

While electric propulsion (EP) provides improved propellant efficiency for in-space maneuvers over chemical engines, long transit time remains an obstacle for this technology. This limitation stems primarily from the so-called "power supply penalty."[1] Although thrust of an EP system can be increased with higher power to the propulsion system, the mass of the power and propulsion system (PPS), e.g. solar arrays, thruster, and power processing units, also increases with power. As a result, the net acceleration, a balance between thrust and spacecraft mass, and subsequent transit times with EP are intrinsically linked to the mass scaling of the power and propulsion element. This scaling is typically quantified by the specific mass, the ratio of the power and propulsion mass to the power generated. In practice, because there are lower bounds on the specific mass of the PPS, the possible transit times with EP are longer than impulsive maneuvers accomplished with chemical systems.

If the specific mass of the PPS could be reduced, the combination of propellant savings and rapid transits would enable several potentially-game changing mission applications. For example, the impact of specific mass on the trade space of EP missions has been discussed extensively in the context of fast transit times to Mars[2] and the outer planets.[3] These deep space missions are particularly attractive for fast-transit EP where the effective propellant usage cost for the mission, the Δv, is high (> 10 km/s) and transfer times with less propellant-efficient, impulsive maneuvers with chemical engines are already inherently long (e.g. $\sim$7 months to Mars). Low specific mass EP invites the possibility that not only could these missions be achieved with reduced propellant compared to a chemical architecture, but in some cases, constantly accelerating EP systems could reach the destinations in the same or even faster time.

While there is a compelling case for fast transit EP in deep space, the need for fast transits with EP in cislunar applications has not historically been as evident. The Δv cost for missions is not as high in this region, and thus propellant budgets are not as prohibitive when using chemical engines. Similarly, the longest transits with impulsive maneuvers in this region require on the order of a few days. EP systems–even with low specific mass—when used for the conventionally-sized spacecraft for cislunar space ($\sim$1000 kg) do not have sufficient power to accelerate fast enough to improve upon the transfer times of chemical engines. It is in large part for these reasons that the primary use cases for EP systems in cislunar space have been station keeping, long-term orbit raising, and drag compensation.

There are two relatively recent developments in cislunar space that motivate a reconsideration of fast-transit EP systems enabled by low PPS specific mass. The first is the growing interest in returning to the moon and expanding operational capabilities in cislunar space. Proposed architectures for these strategic thrusts call for for more spacecraft is cislunar space with high speed (responsiveness) and propellant economy (cost effective). The second major development is the ascendancy of small spacecraft, i.e. those < 150 kg. With recent advances in sensor and communication technology, small sats are developing capabilities that can approximate larger payloads. At the same time, these small sats can be launched cost effectively as rideshares on large-scale deployments dropped into low earth orbit. With that said, unlike larger payloads (> 1000 kg) that conventionally have been employed in cislunar space, small sats are constrained in volume and mass. The available mass budgets for propellants in these small form factors often do not allow for transfers to higher operational orbits with chemical rockets from the low earth orbit drop off. While EP affords a solution with its higher propellant economy, the increase in transit time can mean months or years of delay until deployment and operational use. In light of these new applications in cislunar space and the new paradigm of small spacecraft, there is a need to revisit the capabilities enabled by solar-powered EP in cislunar space with low PPS specific mass.

To address this need, this paper is organized in the following way. In the first section, we describe an approach to simplified mission analysis for fast transits in cislunar space. In the second section, we present the results of our analysis for three classes of spacecraft: cubesat, small sat, and moderate payload. In the third section, we highlight three use cases enabled in cislunar space systems with low specific mass. In the fourth section, we discuss the technical challenges in achieving these architectures. In the fifth and final section, we re-iterate the general advantages of low specific mass EP systems, the limitations of our analysis, and future extensions.

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II. Approach

In this section, we present our approach to analyzing the mission space of EP systems with varying specific mass for the PPS. To this end, we first provide an overview of our key assumptions and governing equations. We then define the classes of spacecraft we consider, the nominal performance metrics we assume for the PPS system, and the values of specific mass we analyze for the PPS.

A. Mission analysis

The primary goal of our analysis is to characterize at a high level how the mission space for cislunar applications depends on the specific mass and performance metrics of the power and propulsion (PPS) element. To this end, our approach is similar to that by Jones[4] and Dankanich.[2] We employ a top-down mass scaling that allows for an analytical solution of mission and spacecraft parameters. The key assumptions from our analysis include

- The specific impulse, I_{sp}, thrust, T, and mass flow rate, $\dot{m}$ of the propulsion system are constant during the maneuver.

- The average solar power available to the thruster, P, is constant during the maneuver.

- The dry mass of the spacecraft is the sum of payload, PPS, structure, and tankage masses:

$$m_f = m_{pay} + m_{pps} + m_{str} + m_t. \tag{1}$$

- The wet mass of the spacecraft at the beginning of the maneuver includes the mass of propellant:

$$m_0 = m_{pay} + m_{pps} + m_{str} + m_t + m_p. \tag{2}$$

- The mass of the PPS scales linearly with the power per the specific mass parameter:

$$m_{pps} = \alpha P \tag{3}$$

- Per the approach of Dankanich,[2] we assume the structural and tank mass scale linearly with the spacecraft wet mass:

$$m_{str} = f_{str} m_0 \tag{4}$$
$$m_{tank} = f_{tank} m_0, \tag{5}$$

where f_{str} and f_{tank} are fractional coefficients less than 1. For this work, consistent with the example calculation performed in Ref. 2, we assume both fractions are 0.1 to yield a total scaling of $m_{str} + m_{tank} = 0.2 m_0$.

- The propellant mass, m_p, required for a maneuver with a cost of Δv can be inferred from the rocket equation:

$$m_p = m_f (e^{-\Delta v / I_{sp} g} - 1). \tag{6}$$

- The propulsion system thrusts continuously during the maneuver.

- The mass flow rate through the engine can be related to the input power and specific impulse through

$$\dot{m} = \frac{2\eta P}{I_{sp}^2 g^2}, \tag{7}$$

where η denotes the electrical efficiency of the thruster and PPU.

- The total transfer time for the maneuver, Δt, scales linearly with the total burn time of the thruster

$$\Delta t = \Gamma t_b = \frac{m_p}{\dot{m}}, \tag{8}$$

where Γ is an empirical correction factor on the order of unity.

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In our following analysis of mission space for cislunar applications, we assume nominal values for specific impulse, efficiency, specific mass of the PPS, and wet mass of the vehicle. We then parametrically solve the governing equations represented in the above assumptions to estimate delivered payload mass, m_{pay}, as a function of transfer time, Δt, and Δv for the maneuver. Before proceeding with this analysis, we remark here on two limitations in the fidelity. First, it is difficult to make a direct map of the Δv anticipated by this analysis to a maneuver. This stems from the fact that total Δv for a continuous thrust maneuver is dependent on the path of the trajectory, orientation of the engine, and duty cycle. Assessing the influence of these factors would require a higher fidelity solution for the orbit. For the sake of this parametric analysis then, we instead consider a range values for $\Delta v = 4000 - 8000$ m/s that captures nearly all non-impulsive maneuvers of interest in cislunar space. Second, the constant Γ is dependent on factors such as duty cycle, which in turn can be impacted by eclipses and deviations from peak-power tracking of the power system. For this analysis, we assume a correction factor of $\Gamma = 1$, recognizing that our transfer times may in some cases underpredict the actual transfer time.

B.   Spacecraft classes

We consider in our analysis three classes of spacecraft differentiated by their wet mass. We highlight these three because they broadly represent the range of systems enabled by the new small sat paradigm. These include

- **Cubesat class (24 kg)**: This mass is approximately representative of a 12 U spacecraft. This spacecraft is of broad interest to universities, as it is a low cost, small form factor system that can be flown as a ride share. With that said, cubesats continue to advance in capabilities with the miniaturization of sensor and communication technology. These systems have been leveraged for applications ranging from geostationary orbits to moon missions and even deep space. To date, however, the limited form factor and volume have made the implementation of EP systems on these devices challenging.

- **ESPA-class (150 kg)**: This class of spacecraft is predicted to be the most commonly flown small sat bus in the next decade. It is sufficiently small that can it be a secondary payload or deployed as part of a multi-spacecraft launch from an ESPA-class adapter. However, the larger mass budget than a cubesat opens the possibility for more sophisticated, larger payloads and larger propellant fractions. This in turn in principle may translate to more capabilities. As with cubesats, there are a number of commercial providers that provide buses in this range.

- **Small launch provider payload class (1000 kg)**: This wet mass represents the upper bound on launch capacity for the next generation of small launch providers. Unlike the ESPA and cubesats, which would be launched as rideshares or as part of larger groupings, this class could merit its own dedicated launch from a small vehicle. The larger mass enables ultimately increased capabilities such as more bandwidth for more communications as well as more available mass for propellant.

C.   Thruster performance

For this analysis, we assume a specific impulse of the propulsion system of $I_{sp} = 1000$ s. We adopt this value or two reasons. First, it is achievable with nearly all forms of electric propulsion at both high (> 1 kW) and lower powers.[5] The higher power systems likely would be employed on the ESPA-class and small launch provider payloads outlined in the preceding while the lower power is applicable to the cubesat class. Second, given that thrust to power scales inversely with specific impulse and faster transits are our target, this moderate value for specific impulse allows allows for a compromise between more rapid acceleration while still providing a three times higher value than conventional in-space chemical engines. For the overall efficiency, we assume a value of $\eta = 50\%$. This is a broad representation of the achievable efficiency for most modern types of of EP that operate at powers above 1 kW.[5] While lower power systems often exhibit efficiency lower than this,[6] several commercial developers have indicated capabilities that approach this mark.

D.   Specific mass of power and propulsion systems

We show in Fig. 1 the breakdowns for two cases for the specific mass we consider for the PPS: one to represent the state-of-the-art at $\alpha = 20$ kg/kW and one for next-generation systems of $\alpha = 2$ kg/kW. For

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the SOA, we inform our value based on a consideration of Hall effect thrusters, the most widely flown type of EP in cislunar space. Studies performed in the last decade on Hall thruster scaling[7,8] have indicated that conventional systems have ~ 2 kg/kW for the thruster and an additional ~ 2 kg/kW for the power processing unit. For the solar array specific mass for cislunar space, we baseline the Redwire Space ROSA arrays at 16 kg/kW demonstrated on the International Space Station. This is one of the lightest functional arrays demonstrated to date. Taken together, the summation of these demonstrated subsystems provides approximately $\alpha = 20$ kg/kW. Our next-generation value of $\alpha = 2$ kg/kW allows for an order of magnitude improvement. We elect to use this as it provides a stark comparison to the SOA and demonstrates several notable mission capabilities. We elaborate further in Sec. V about the feasibility of achieving this value in the near-term.

(a)
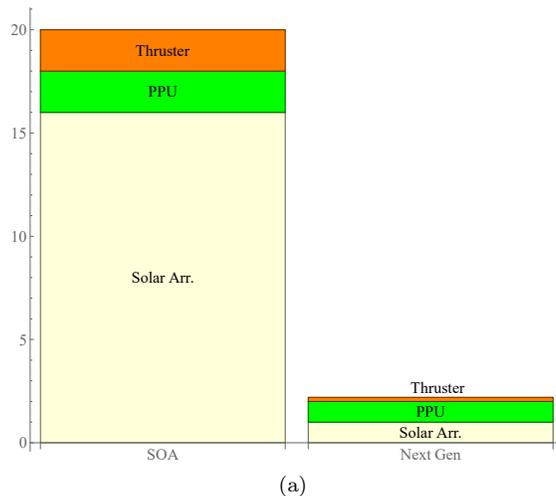
Figure 1: Breakdowns for state-of-the-art and next-generation assumptions for specific mass of the PPS.

III. Results

We show in Fig. 2 the results from our analysis described in the preceding section. Here we plot the delivered payload mass, m_{pay}, as a function of transfer time, Δt, and Δv for the three spacecraft classes we outlined in the preceding. We consider transfer times up to $\Delta t = 100$ days and $\Delta v = 4000 - 8000$ m/s. We chose this range of times as 100 days is on the order of the maximum time currently allowed for orbit raising for commercial EP systems from LEO to GEO with SOA systems. The range of Δv allows for a continuous thrust transfer to effectively any orbit in cislunar space. These include LEO to GEO with no plane change at the lower end (4000 m/s) to insertion into a low lunar orbit from LEO at the upper end (8 km/s).

Practically, these graphs provide a tool for first order mission planning. Given a targeted Δv, acceptable transfer time, and fixed wet mass, a user can determine the payload that can be delivered. Qualitatively, there are a number of general trends exhibited by all the figures in this result. First, we note that for fixed transfer time, the payload mass delivered decreases with Δv. Physically, this a direct consequence of the rocket equation driving the need for more propellant for a more energetically expensive mission. For sufficiently high values of Δv and short transfer times, the architecture does not close. This is represented in the plots by white space. Second, we see that for fixed Δv, payload mass generally increases with longer transit time. This stems from the fact that longer transit times relax the requirements on acceleration and thrust, thus requiring lower power. Reduced power translates to a less massive PPS, and consequently more budget can be allocated for the payload. Third, for fixed payload mass (contour lines), the transfer time increases nonlinearly with higher Δv. This ultimately can be explained by increasing propellant mass with more energetically-expensive missions, allowing less mass for the PPS. This translates to a lower power available for thrust and thus longer transit times. The trends ultimately asymptote because at a finite value of Δv, there is no mass left for power supply, and transfer times become undefined.

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(a) α=2 kg/kW, m_0=24 kg

(b) α=20 kg/kW, m_0=24 kg

(c) α=2 kg/kW, m_0=150 kg

(d) α=20 kg/kW, m_0=150 kg

(e) α=2 kg/kW, m_0=1000 kg

(f) α=20 kg/kW, m_0=1000 kg
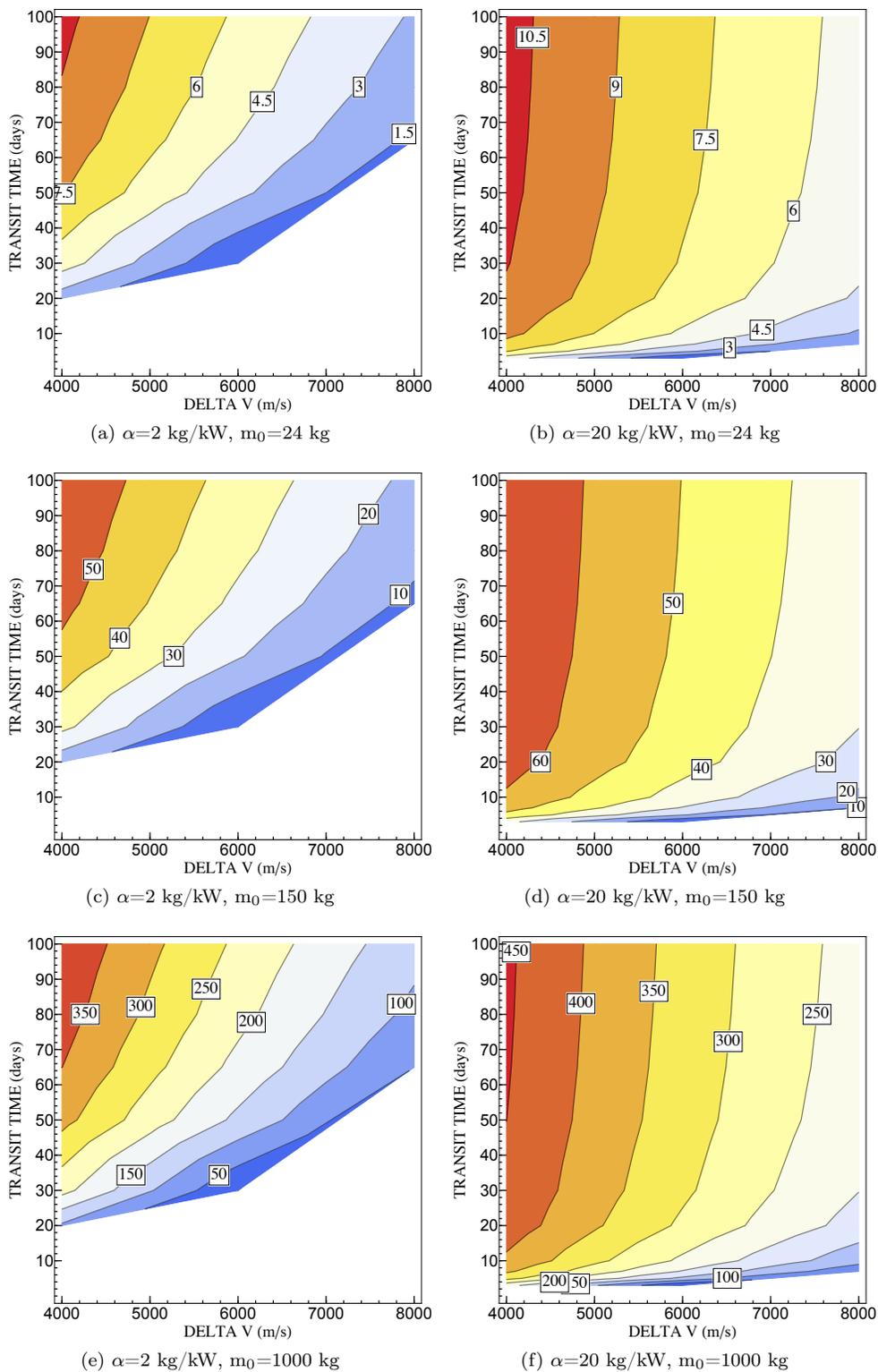
Figure 2: Payload masses (kg) as a function of burn time and Δv for two specific masses for the PPS and three classes of spacecraft wet mass. For all cases shown, $I_{sp} = 1000$ s, $\eta = 50\%$, $f_{tank} = f_{str} = 0.1$. White regions are inaccessible.

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(a) α=2 kg/kW, m_0=24 kg

(b) α=20 kg/kW, m_0=24 kg

(c) α=2 kg/kW, m_0=150 kg

(d) α=20 kg/kW, m_0=150 kg

(e) α=2 kg/kW, m_0=1000 kg

(f) α=20 kg/kW, m_0=1000 kg
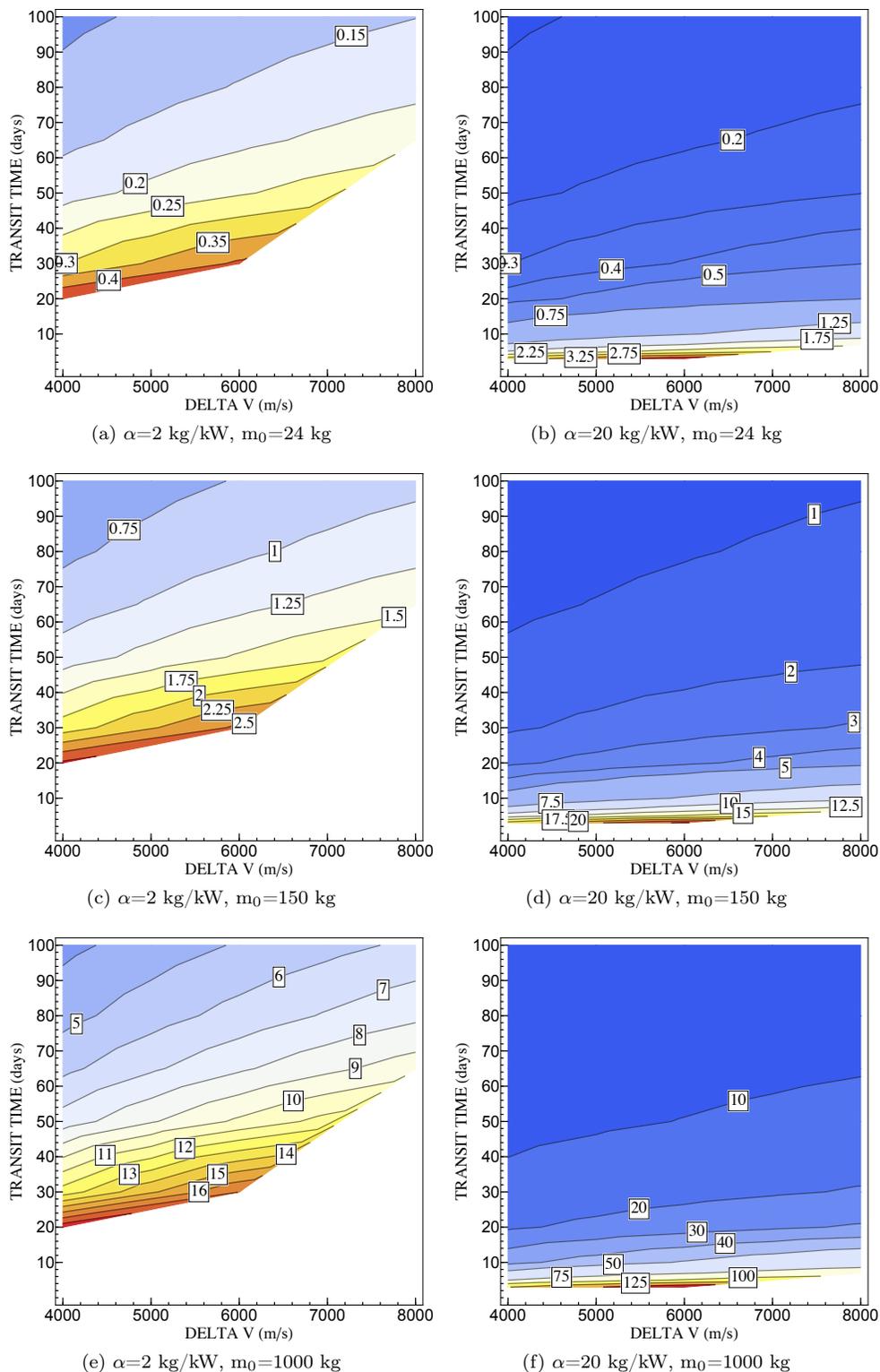
Figure 3: Power (kW) as a function of burn time and Δv for two specific masses for the PPS and three classes of spacecraft wet mass. For all cases shown, $I_{sp} = 1000$ s, $\eta = 50\%$, $f_{tank} = f_{str} = 0.1$. White regions are inaccessible.

In all cases, we note that the mass of payload improves with reduced specific mass of the PPS. This is a similar result as previously noted by Dankanich in the context of analyzing Mars missions.[2] Notably,

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extremely rapid transfers to high $\Delta v = 8$ km/s with finite payloads become possible in less than three days. This would translate to transfers to the moon in the same amount of time as a conventional Hohmann transfer with a chemical engine. The implication is that rapid transits rivaling chemical maneuvers thus do become possible with EP systems provided the PPS specific mass can be reduced.

We show in Fig. 3 the required power for the PPS for the same parameter space as plotted in Fig. 2. In all these cases, the general trend is that increasing Δv and reduced transit time both translate to higher power for the required mission. The increase in power for reduced transit time physically stems from the fact that higher thrusts are required to accomplish the maneuver. The increase in power with mission cost, Δv, for a fixed transit time is explained by the increasing amount of propellant required. Since the transit time remains constant, the flow rate must increase, thereby translating to higher thrust and required power. Notably, we see that with decreasing specific mass, e.g. from $\alpha = 20$ kg/kW to 2 kg/kW, the overall power levels increase for a given spacecraft class and mission parameter set. This is a consequence of the higher allowable mass budget for the PPS in each system, accommodating higher powers. The rapid transits in turn are enabled by the higher power levels.

With that said, we see that the dependence of the power on transit time is relatively weak, and as the transit time decreases, the contours become nearly horizontal. The implication is that for fixed power, a wide range of Δv can be accommodated with the same propulsion system for approximately the same transfer time. The trade, of course, is the payload mass, which necessarily decreases with Δv for fixed transit times (Fig. 2). In practice, this ability of a fixed power to access a wide range of missions motivates a key design consideration for cislunar space. Indeed, it suggests a path in which a propulsion system is designed for fixed power, and the mission is tailored by adding propellant at the expense of payload.

In this context, Fig. 3 provides insight into the power levels currently employed for SOA EP systems in cislunar space ($\alpha = 20$ kg/kW). We see that for 1000 kg payloads, a 5 kW system is able to achieve $\Delta v = 4000 - 6000$ m/s over a span of 70-100 days. This corresponds to orbit raising from LEO-GEO for GEO communication satellites ($\sim$1 ton wet mass) which has been demonstrated to date with EP systems. Indeed, for geostationary applications, the highest power propulsion systems that have been leveraged to date operate at less than 5 kW. In a similar vein, we see that a 1 kW thruster with SOA specific mass for the PPS is able to perform transfers over the complete wide range of Δv with comparable transfers times for higher mass payloads (60-90 days). This result may in part explain the focus of several new commercial interests in developing sub-kW class systems for the new small sat paradigm. A similar argument can be applied for a 150 W EP systems for cubesat (12 U) class missions.

The ability of a fixed power to accommodate the full wide range of Δv for approximately the same transfer time is also exhibited by the low specific mass cases. In particular, we draw attention to the near horizontal contours exhibited for each mission class at 40 kW (1000 kg), 5 kW (150 kg), and 750 W (24 kg). The key implication from these results is that for fixed power and each mission class, it is possible to *transit anywhere in cislunar space from a LEO with the EP system in less than ~ 15 days.* This ultimately speaks to how fast-transit, low specific mass systems are highly enabling for responsive yet low cost missions. To further highlight these advantages, we consider in the following section use cases for next-generation PPS systems for these three mission classes and power levels.

IV.   Case studies

Informed by our parametric results in the previous section, we consider here case studies with low specific mass PPS for three mission classes with EP powers of 750 W, 5 kW, and 40 kW. To highlight the ability of these missions to access the range of cislunar space, we make the strong assumption in this section that the Δv for the EP trajectories can be estimated with a quasi-circular approximation for co-planar maneuvers. This allows us to relate Δv for a mission starting at LEO to delivery to an arbitrary circular orbit from the earth. This quasicircular approximation provides for Δv up to 7000 m/s (at the moon's orbital radius) but does not include braking maneuvers or plane changes. For comparison, we also consider the payload mass and transit times for a chemical system, assuming $I_{sp} = 200$ s (typical of hydrazine) and a Hohmann transfer from LEO. The variation in the Δv for the Hohmann transfer with the destination orbital radius is considered in this analysis.

A.   Responsive cubesat in cislunar space

We show in Fig. 4 comparisons of the delivered payload mass and transfer times for a chemical system and low-specific mass PPS for the cubesat. As this result shows, for orbital radii extending from GEO ($\sim$ 5 times LEO orbital radius) to the moon (55 times the orbital radius), the EP system is able to transfer payloads approaching 6 kg to lunar orbit. The chemical system is not shown because it does not close. This is because the tankage fraction is too high, leading to negative payloads. We also see that while transfer times can be several days with EP compared to a pure Hohmann transfer, the time remains less than or on the order of 17 days.

This finding is significant as cubesats have traditionally been limited in application due to a combination of high specific mass and low thruster efficiency. This analysis shows that the low specific mass option opens up the possibility for delivering light weight payloads such as small cameras for planetary or space monitoring, or dedicated scientific instruments for research programs. Indeed, this class of spacecraft is particularly well suited for university engineering programs and for establishing low-cost situational awareness.

(a)

(b) α=20 kg/kW, m_0=24 kg
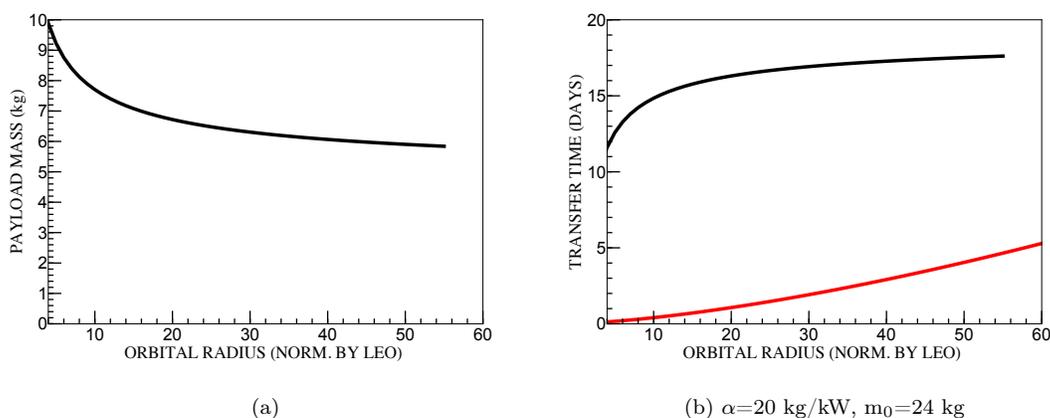
Figure 4: a) Payload delivered for EP (black) with $I_{sp} = 1000$ s, 750 W power, and specific mass of 2 kg/kW for a 24 kg wet mass starting in LEO at 200 km. b) Transfer time for the EP system (black) and chemical systems utilizing a Hohmann transfer (red). Note that payload mass for the chemical system is not shown as the mission does not close for transfers from LEO.

The fact that a 12 U with its small mass can be rapidly deployed in cislunar space from LEO also opens notable possibilities for large scale constellation deployments in this region. Indeed, in a similar way that megaconstellations can currently be deployed at the rate of 50-100 spacecraft per launch, fleets of low-cost, maneuverable cubesats could similarly be launched simultaneously to deploy and inhabit multiple orbits in the cislunar region. This could facilitate rapid and cost-effective expansion in capabilities for monitoring nd logistics.

B.   ESPA class to high moon orbit in 15 days

Fig. 5 shows the trade space for ESPA class spacecraft assuming a starting orbit in LEO. ESPA class spacecraft are well suited for 10-30 kg payloads and, like cubesats, can be regularly accommodated as rideshares. Payloads include low-bandwidth communications equipment for Earth and the moon, as well as high resolution sensing instruments. With low specific mass SEP, fast transits (approximately the duration of a Hohmann transfer) to lunar orbit become possible with 10-20 kg payloads using a 10-20 kW thruster. For less demanding trip times, power as low as 1 kW provides access throughout cislunar space with up to 70 kg payload. Fig. 5 shows payload mass and transfer time for a 5 kW EP system. This flexibility of trading payload for trip time is enabling compared to SOA EP, in which the minimum trip duration is much longer due to the large mass fraction of the PPS.

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(a)

(b) α=20 kg/kW, m$_0$=24 kg
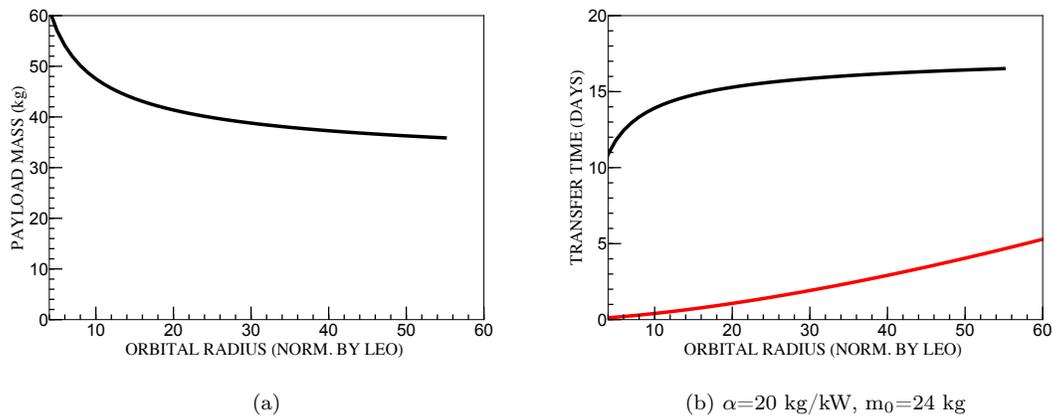
Figure 5: a) Payload delivered for EP (black) with I_{sp} = 1000 s, 5 kW power, and specific mass of 2 kg/kW for a 150 kg wet mass starting in LEO at 200 km. b) Transfer time for the EP system (black) and chemical systems utilizing a Hohmann transfer (red).

C.   Rapid response using small launch provider

Fig. 6 shows the trade space for the 1000 kg class spacecraft mass assuming a starting orbit in LEO. While GEO has traditionally been the domain of >1 ton dry mass spacecraft serving the telecommunications industry, the shorter time to station of smaller satellites combined with continued miniaturization of hardware, has led to recent activity by several spacecraft providers for <1 t products. An order of magnitude reduction in specific mass would enable 200-450 kg payloads. For example, a 1 month transfer to GEO with a 2 kg/kW PPS results in double the payload of a SOA: 440 kg vs 220 kg, respectively. Further reductions in transit time at the expense of payload enables operators to financially optimize their fleet.

(a)

(b) α=20 kg/kW, m$_0$=24 kg
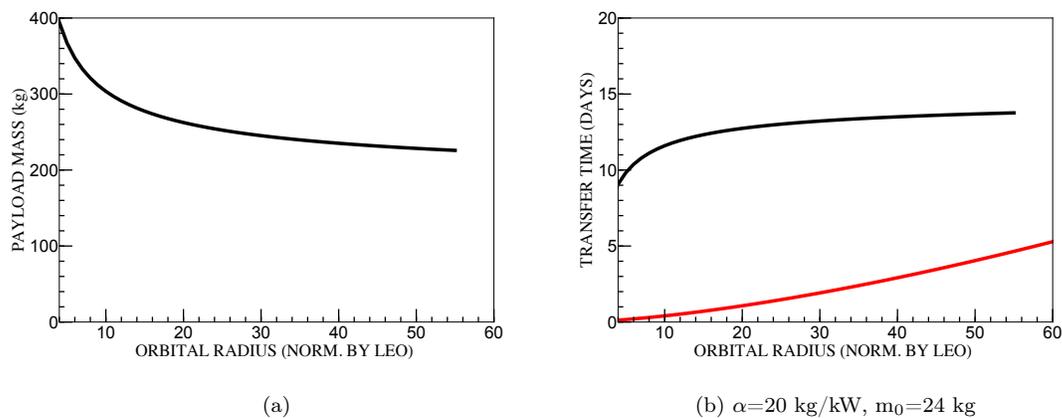
Figure 6: a) Payload delivered for EP (black) and chemical (red) system with I_{sp} = 1000 s for a 1000 kg wet mass starting in LEO at 200 km. b) Transfer time for the EP (black) and chemical (red) systems. The EP system is assumed to have 5 kW power and specific mass of 2 kg/kW.

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V.   Discussion of feasibility of low specific mass power and propulsion system

While there are several capabilities for cislunar applications enabled by an order of magnitude reduction in specific mass of the PPS, there remains the feasibility of achieving this metric. We discuss in the following possible strategies and roadmaps for the near-term.

A.   Thruster

For the propulsion element, as discussed in the preceding, state-of-the-art for Hall thrusters have a specific mass of approximately ~2 kg/kW.[7,8] While there are several factors that influence this value including the weight of the magnetic elements, the need for thermal margin, and the mass of the harnessing, a key and critical driving factor is the thrust density of the exhaust. Indeed, most Hall thrusters to date have been built to ensure an optimal current density–and by extension thrust density.[9] Observing this scaling in the thruster design inherently places a lower bound specific mass. With that said, experiments at the University of Michigan on a laboratory Hall thruster recently have demonstrated a factor of ten increase in power density at 45 kW total power and a factor of seven increase in thrust density (Fig. 7).[10] While outstanding challenges remain to translate this improvement to a flight-like system, e.g. challenges with thermal loading and erosion rates, this laboratory demonstration suggests a factor of ten improvement in specific mass may be a near term possibility for mid (5 kW) to high power (> 40 kW) applications outlined in the preceding. This would satisfy the requirement for our low specific mass architecture.

(a) Xenon

(b) Krypton
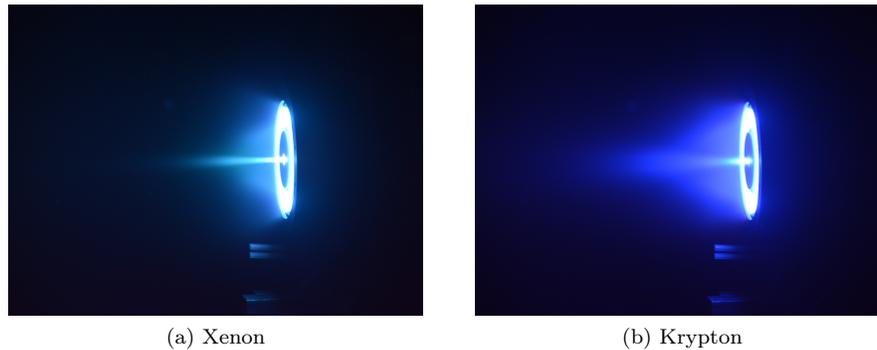
Figure 7: Hall effect thruster operating at ten times the nominal current density at the University of Michigan.[10]

Achieving a factor of ten reduction in specific mass for the Hall thruster becomes more problematic with decreasing power. This trade is highlighted in Fig. 8 where we show data extracted from Ref. 8 of thruster specific mass for a range of commercial and laboratory thrusters as a function of power. As can be seen, the specific mass generally increases at lower powers, exceeding 5 kg/kW below 500 W. While in principle operating at higher current density may also help mitigate this larger specific mass, extending this approach to lower power thrusters may be more problematic than at the 5 and 40 kW power levels. In particular, thermal rejection likely becomes an even more pressing challenge at these scales. With that said, extending the approach of ultra high thrust density towards lower powers is an ongoing research area, and the limits of achievable improvement is currently unknown. This leaves ambiguity about assessing the near term feasibility of achieving the target specific mass for the thruster with a Hall thruster for cubesat class missions.

Alternatives to Hall thrusters may show greater promise in meeting our goals at the low power range. Electrospray thrusters, for example, have high potential for low specific mass due to lack of a magnetic field, high propellant storage density, and lower volume since plasma generation is unnecessary. Maximum power per chip and manufacturing repeatability have thus far limited applications to secondary propulsion, but each of these is being actively addressed in research.[11]

B.   Power processing units

Historically, PPU specific masses for mid-to-high power Hall thrusters have been on par with thrusters masses. This is reflected in Fig. 8(b) where we show PPU specific mass for a range of commercial system

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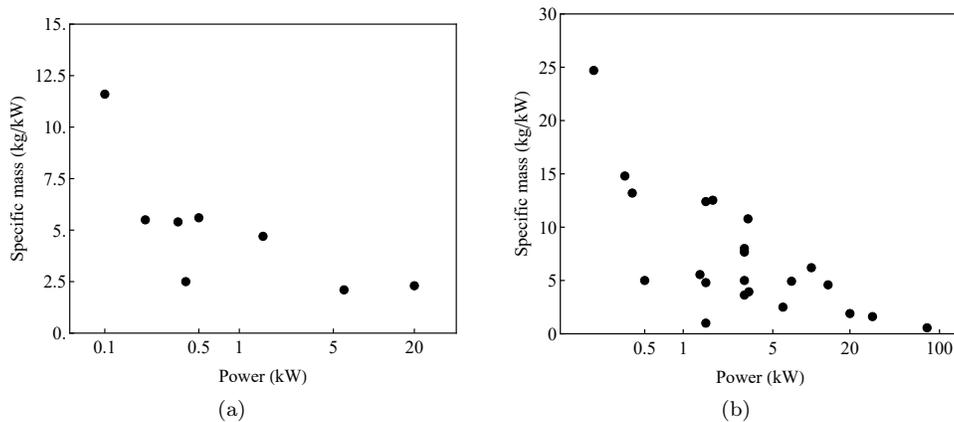
Figure 8: Specific masses of a) thrusters and b) power processing units. Data extracted from Ref. 8 and from manufacturer specification sheets.

as a function of power (data extracted from Refs. 8, 12). As can be seen here, as with the thruster, the specific mass improves with higher power, approaching 1 kg/kW for 100 kW systems and the ∼ 1.7 kg/kW for mid-power systems (>1 kW).

In order to achieve the order of magnitude reduction of specific power required for our proposed architectures at 5 kW and 40 kW, it will be necessary to explore new PPU architectures. Possible technical paths include utilizing fewer power supplies,[13] new materials for solid state switching (such as GaN in place of Si based MOSFETs),[14] and a flying capacitor multilevel converter topology.[14] Further reduction could occur if it is possible to run off of a single power supply. More radical approaches include direct drive operation,[15] in which the arrays are configured to develop high voltage directly. This in effect eliminates the PPU entirely–with the exception of minimal processing elements like passive capacitors.

As with thrusters, Fig. 8(b) shows PPU specific mass scales inversely with power. Reaching our target for the cubesat class spacecraft thus will pose the most difficult challenges for Hall thrusters. Incorporating the previously mentioned advances into ever smaller packages will be crucial towards this end. Adopting alternative thruster technologies may also provide a solution. For example, PPUs for electrosprays, which in principle may only require one, high voltage power supply, may be inherent lighter at this lower power range.

C. Solar arrays

The solar arrays pose the greatest challenge for the achieving the target specific mass as these are the dominant mass driver–almost an order of magnitude higher than the other two subsystems of the PPS. As discussed in the preceding, our justification for the SOA in solar arrays at higher power is based on conventional Si technology. For example, the ISS has an average value of 30 kg/kW while the recently-installed Redwire Space ROSA arrays on the ISS are 16 kg/kW. At the low power end, we see a range of specific masses from 5-25 kg/kW, as documented by a recent NASA report on SOA smallsat solar arrays,[6] shown below in Fig. 9. While the 7 kg/kW case shown in this result is compelling, it still higher than or target.

In terms of technical paths to achieve the required order of magnitude reduction for the soar power, Redwire has indicated that they anticipate 2-3 kg/kW for the Mega-ROSA arrays (for >25 kW systems).[16] Other relevant near-term technologies that could reduce specific mass include concentrator systems to increase the effective light-collecting area, lightweight composite or metallic collapsible support booms, and improved collection efficiencies. These may in effect achieve up to a factor of 5 reduction in specific mass. Additional future improvements may be enabled by third generation photovoltaic. For example, it has been proposed that thin-film copper indium gallium diselenide (CIGS) may be able to achieve with 1-3 kg/kW at the system level. Organic photovoltaic (OPV) cells, which have thicknesses <100 μm and specific masses <0.5 kg/kW, may provide any even more marked improvement. This stems from the fact that for these advanced cells, new storage and deployment systems will be possible, and stored volumes will drop dramatically. Indeed, CIGS cells can be rolled out like other SOA arrays, whereas OPVs offer even greater flexibility, potentially utilizing

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storage and deployment systems like those developed for solar sail applications (examples include NASA's NEA Scout[17] and ACS3[18]). With that said, these next-generation cells are still largely in the development stage and these specific masses have yet to be demonstrated in the context of EP systems on orbit.

As a notable caveat, while these advances in solar cell material will most readily be utilized for lowering specific mass of higher power systems of interest (> 5 kW), for the cubesat class, the specific mass of the solar cell is of secondary concern. The stowage and deployment system and the actuators for sun tracking, is relatively high, which could be the limiting factor for specific mass.
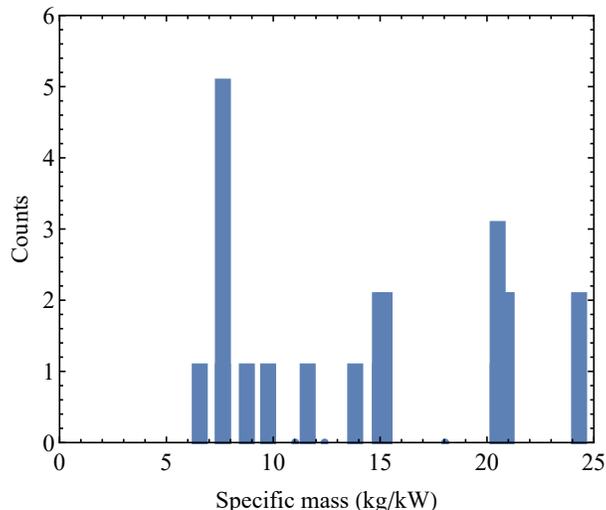
Figure 9: Histogram of specific masses of smallsat solar arrays. Data extracted from Ref. 6.

VI.   Discussion

As we have shown in the preceding, a lower specific mass enables lower mass PPSs, a more efficient use of volume, and higher power systems than have to date been available to small satellites. Indeed, if the proposed levels of specific mass could be achieved, transits to anywhere in cislunar space from LEO with $<$ 1 wet masses could be attainable. This opens new possibilities for leveraging cost effective rideshares to low earth orbit for massive deployments of constellations that can rapidly populate cislunar space or alternatively for facilitating rapid response to evolving events. The latter case represents a powerful power synergy with the on-going programs that are seeking to demonstrate launch capabilities and deployment within 24 h of notification. Indeed, low specific mass PPSs allow for a reduction in overall satellite mass and volume to the extent that highly capable satellites can be launched by the growing class of small launch providers.

We note here an additional advantage of rapid transits from the perspective of technology development. Indeed, the reduced trip times suggest that ground testing qualifications are proportionally reduced. Even allowing for a requirement of 50% demonstrated lifetime margin, the total test duration would not exceed a month. This could represent a substantial savings in time and cost over the current paradigm for flight unit qualification for new thrusters.[19]

With that said, we remark here that the analysis outlined in Sec. II is effective at first-order estimates and relative comparisons, but it does have several limitations. The estimated trip durations may be longer than the burn duration by a scaling factor proportional to the amount of time spent in eclipse. We have assumed unity for this factor in this analysis, for but slower burns, correction factors may be on the order of 1.3 depending on the time spent in eclipse. We established this factor by performing select and non-comprehensive higher fidelity orbit trajectories. A more detailed and higher future study must consider effects such as these, allowing for optimizing of thruster parameters such as specific impulse, and spacecraft parameters such as the structural and tankage fraction scaling. A bottoms up mass budget for each use case would be the subsequent step in mission analysis, particularly for the cubesat and ESPA class spacecraft, since each spacecraft class has unique scaling considerations.

Cost is another consideration for mission planning with high-power EP systems that was neglected here.

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SOA solar arrays currently scale approximately linearly with power for lower power systems, presenting another parameter to trade against trip duration and spacecraft mass. A thruster string cost model proposed by Hofer[7] proposes a square root scaling with power. Given the low TRL of several of the technologies discussed here, future studies accounting for cost scaling as data becomes available will be useful for making informed mission trades.

VII.   Conclusion

We have presented in this work a first order analysis of the capabilities enabled by solar-powered electric propulsion systems in cislunar space with low specific mass. We have shown that if the specific mass of the power and propulsion system can be reduced by an order of magnitude compared to the state of the art, it is possible for missions with < 1 t wet mass in low earth orbit to deliver finite payloads to anywhere in cislunar space in less than 15 days. We in turn have discussed the implications of the capability in the context of cubesat class, ESPA-class, and small launch provider payload class missions. Indeed, key capabilities that may be enabled by this advance in technology may include massive deployments of spacecraft into cislunar space and rapid response missions to target orbits of interest. In practice, achieving this new capability is contingent on the ability to reduce specific mass of all key elements of the PPS–thruster, power system, and power processing—by an order of magnitude. While there have been recent advances suggesting technical paths for achieving this reduction for the thruster at moderate power levels (> 5 kW), there remain several challenges for the other subsystems. With that said, we have discussed how innovations in power processing architectures and solar cell technologies may offer near-term solutions to reducing specific mass. In summary, while low specific mass EP systems historically have been considered more in the context of deep space applications, our results suggest that this class of technology could yield substantial paradigm-changing benefits closer to earth.

Acknowledgments

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