Orbital Power-Beaming: An Alternative for Power-Restricted Regions on the Lunar Surface

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One of the biggest challenges facing the expansion of lunar surface exploration is access to a reliable power source for purposes of surviving lunar night, exploring permanently shadowed regions (PSRs), and achieving long traverses. Many landers and rovers are currently limited in their range, performance, and lifetime due to power and recharging constraints. The concept of Space-Based Solar Power (SBSP) has been studied for decades for terrestrial applications and provides an attractive solution to these lunar surface missions by enabling flexible power delivery from orbit to surface assets. The Orbital Power-Beaming Assets for Lunar Applications (OPAL) project is employing systems analysis methodologies to study the systems-of-systems relationships between the orbital power beaming spacecraft and the surface assets or "clients" and to assess the cost, benefit, and complexity of employing this technology. A preliminary Concept of Operations (ConOps) has been defined and an orbital assessment resulted in the selection of a near-polar circular orbit with an altitude of 1,175 km based on station-keeping requirements, access to the lunar south pole, and distance to the surface. Solar Electric Propulsion (SEP) has been selected as the main propulsion system of each OPAL spacecraft to minimize the amount of propellant required. A power beaming trade resulted in the selection of an optical wavelength of 1064 nm, followed by a laser technology assessment that resulted in the recommendation of a fiber laser as the optical beam source for the OPAL spacecraft. Following the definition of three "power classes" that group clients based on similar power demands, a parametric sizing of the OPAL spacecraft was completed. A case study of an OPAL-compatible Commercial Lunar Payload Services (CLPS)–type lander was then performed, which included a high-level assessment of impacts to the power system mass and the operations of the lander. This integrated approach and the results of the analyses support the decision to continue the development of a concept for a power-tunable and focusable optical power beaming system that could accommodate the power demands of current and future lunar surface missions.

I. Introduction

A variety of power beaming technology demonstrations and concept studies have been performed in recent years by the California Institute of Technology (CalTech), the U.S. Naval Research Lab (NRL), the National Aeronautics and Space Administration (NASA), and others [1, 2, 3]. However, developments have been primarily focused on point studies that are only concentrated on a specific surface mission design. The Orbital Power-Beaming Assets for Lunar

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Applications (OPAL) project is employing systems analysis methodologies to enable an integrated study of the power beaming trade space from a systems-of-systems perspective.

Alternatives to power beaming include systems such as fission surface power, vertical solar array technologies, and others [4, 5]. These methods typically require considerable landed mass and intricate cabling systems, and they result in restricted range – offering success only in either power generation or power distribution. The proposed Space-Based Solar Power (SBSP) concept reduces landed mass and surface complexity while providing both power generation and power distribution. Although SBSP offers an increased level of flexibility for power on the lunar surface, it also has many challenges that can be overcome through improvements in a variety of technologies and subsystem, including, but not limited to, inefficient laser sources, large and complex optical systems, large photovoltaic arrays for power generation and radiators for laser waste heat rejection, and efficiency-limited photovoltaic cells for optical beam power conversion.

This study concludes with the recommendation to pursue the concept for an optical power beaming system capable of power tuning and spot size shifting, which has the potential to service multiple clients with a single payload design while maintaining higher efficiency and efficacy. Key considerations include a Concept of Operations (ConOps), mission design, orbital impact analysis, beamed frequency selection, preliminary laser system concepts, preliminary technology considerations, parametric sizing of the OPAL spacecraft, and a client impact assessment or case study.

II. Concept of Operations

The general ConOps for an OPAL spacecraft is represented in Fig. 1. This ConOps takes a client-centered approach in order to beam power to surface assets in a reliable and useful manner. This study focuses on servicing clients at or near the lunar south pole in power-restricted regions. Energy is collected on board the OPAL spacecraft and beamed to the surface via an optical power beaming system or payload. The surface asset is assumed to hold a 10° minimum tracking elevation in which no power is beamed. This accounts for both environmental effects from the lunar surface, such as multipath loss or terrain occultation, as well as any time required to slew and acquire the client. The client can then collect this power on the surface either with solar arrays or with an optical receiving system (receiver) to store and utilize on board as needed.

General operations, such as orbital maintenance maneuvers or communications down-link to Earth, have lower operational power requirements in comparison to power beaming activities. Within the scope of this study, the OPAL spacecraft Electric Power System (EPS) has been sized to accommodate power beaming during the entirety of the orbit (in both sunlight and eclipse) to consider the worst-case power generation demand.

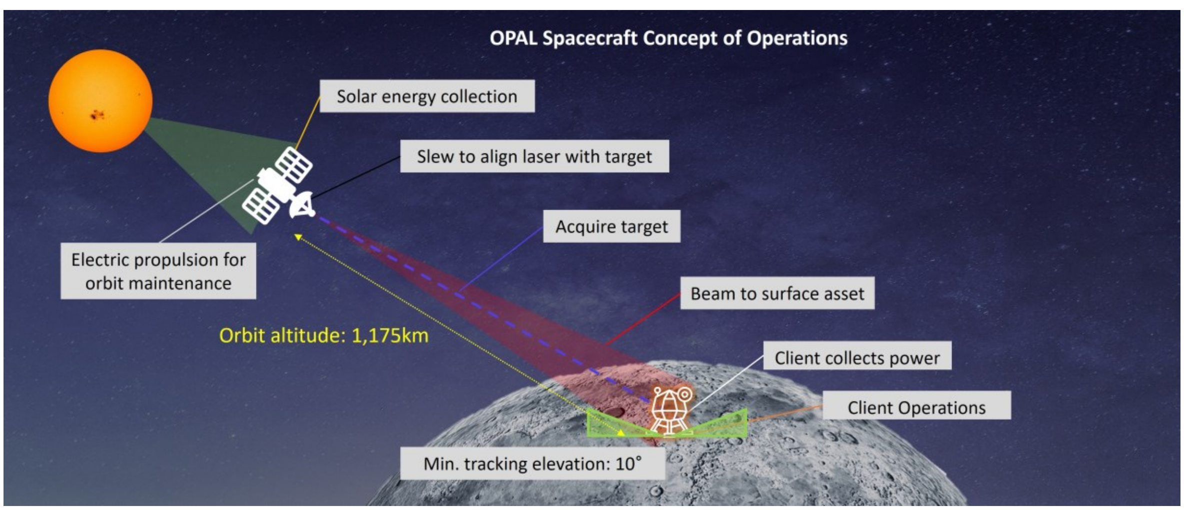

Fig. 1. ConOps for an OPAL spacecraft from solar energy collection to optical power delivery.

III. Methods & Assumptions

This section discusses the development and justification of methods and assumptions held within this study. This includes an orbit selection and preliminary propellant budget analysis, a power beaming study with frequency down-selection and preliminary beaming system conceptual design, and a technology assessment with a preliminary laser type selection to drive the spacecraft parametric sizing analysis.

A. Power Classes

In addition to a general ConOps, power classes were defined to better understand the sizing variation for a variety of power needs on the lunar surface. These power classes are based on specific spot size and irradiance assumptions discussed below. The proposed power-tunable and focusable payload would enable a power class to supply any power demand at or below its maximum threshold. These power classes are defined based on a combination of surface needs. They consider solar array sizing and irradiance on the surface and maintain some level of margin for pointing errors and unknown losses. The series of equations in Table 1 represent how the power classes are defined based on these parameters.

Table 1. Power class methodology, equations, and assumptions

Equation	Justification/Details
$A_{spot\ size} = \frac{\pi(l^2 + w^2)}{4} * 1.25$	Spot size area accounts for the surface of the receiver with an additional 25% margin held on the area for pointing error.
$P_{RX} = A_{spot\ size} * 1{,}365$	Received power on the surface is based on the spot size and assumes a 1,365 W/m^2 irradiance to equal that of the Sun.
$P_{beam} = P_{RX} * 1.5$	Losses are minimal due to the lack of atmosphere; however, the beamed power assumes an additional 50% loss for anything not included in the model. This beamed power is represented as the power of the beam as it exits the OPAL spacecraft.

This methodology allows the power classes to be defined based primarily on receiver sizing, demonstrating a flexible, client-based design approach. Table 2 outlines the power classes used throughout the entirety of this study. These classes yield three parametric spacecraft designs and facilitate an incremental approach to implementing SBSP at different scales.

Table 2. Power class definitions with example clients

	Class I	Class II	Class III
Client Receiver Size	$1\ m^2 = 1\ m \times 1\ m$	$3\ m^2 = 1.7\ m \times 1.7\ m$	$7.85\ m^2 = 2.8\ m \times 2.8\ m$
Beamed Power, P_{beam}	4 kW	16 kW	30 kW
Example Clients	Nova-C & Peregrine	VIPER & Unpressurized Rover (survival mode)	Unpressurized Rover & Large Rover (survival mode)

B. Orbital Assessment

STK analyses were conducted to better understand the implications of orbital selection on the overall system design and ConOps. The primary objectives of this orbital selection were to provide adequate access to the lunar south pole, remain at an altitude less than 5,000 km to minimize the beaming distance, provide a level of orbital stability to reduce propellant requirements, and maintain a circular orbit to minimize distance variations from the clients.

To assess access to the clients, a circular near-polar orbit (inclination = 86°) was assumed with a pure polar ground asset (lat = 0°, lon = -90°). Figure 2 represents a simulation conducted over one calendar year at varying altitudes. Access percentage increases as altitude increases, beginning to level off shortly after 1,000 km. Due to the associated increase in "no access" time (i.e., discharge time on the client side) with increased altitude, a distance near the knee of the percentage curve was deemed ideal, significantly narrowing the range of viable altitudes to between 1,000 km and 1,500 km. An orbital stability assessment of frozen orbits resulted in a selected orbit altitude of r = 1,175 km with eccentricity of e = 0.002 [6]. This orbit results in an orbital period of ~4 hr with access to the client of ~1 hr per orbit and was assumed for the client State of Charge (SOC) analysis within this study.

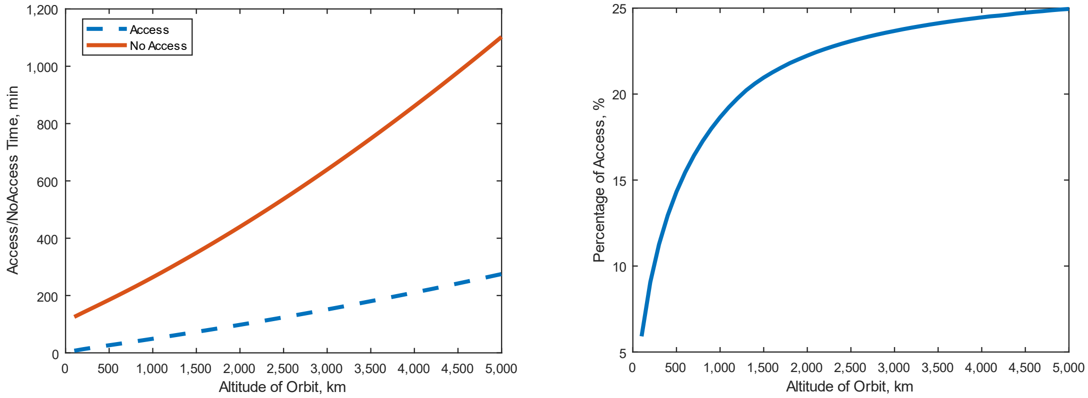

Fig. 2. Average access time vs no access time across one calendar year over a variety of altitudes (left). Percentage of total access across an orbit at varying altitudes (right).

To determine the implications of propellant mass required for orbital maintenance in the selected orbit, one-year simulations were conducted in STK for a chemical propulsion (hydrazine) case and a low energy Solar Electric Propulsion (SEP) case (Xenon). Table 3 outlines the assumptions held for each propulsion system.

Table 3. STK assumptions for propulsion subsystems for orbital maintenance simulation

	Chemical (Hydrazine)	SEP (Xenon)
Isp	220 s	1710 s
Mass flow rate (per thruster)	---	4.8×10^{-5} kg/s
Number of thrusters	1 (gimballed)	8

SEP was selected due to its considerable mass savings in comparison to a chemical propulsion system. Table 4 shows that despite the added operational complexity of using a low energy propulsion system, the mass savings associated with SEP make it a preferable option. A propellant mass of 200 kg was used for further mass analysis within this study to account for any additional maintenance and maneuvering.

Table 4. STK propellant results for hydrazine and Xenon electric propellant mass across 1, 5, and 10 years

Mission duration (years)	Chemical (hydrazine)	Chemical with 25% margin	SEP (Xenon)	SEP with 25% margin
1	176 kg	220 kg	11 kg	14 kg
5	880 kg	1,100 kg	55 kg	69 kg
10	1,760 kg	2,200 kg	110 kg	138 kg

C. Power Beaming Analysis

The primary frequencies considered for power beaming included those within the microwave and optical frequency spaces. Each frequency space has benefits and pitfalls within a respective beaming environment – Earth, Mars, or the Moon. This frequency selection focuses on minimizing the transmitter size, minimizing the spot size on the surface, and maximizing flexibility on the lunar surface. Microwave frequencies work well for terrestrial applications due to the lower attenuation experienced in comparison to higher frequencies. This attenuation poses less of an issue at the Moon due to its lack of atmosphere, opening opportunities to utilize the optical frequency space. Due to the high power requirements of the power beaming system, only parabolic transmitters were considered for microwaves. For optical frequencies, lasers were used for this analysis due to their availability at the high power required.

To reduce the beamwidth and therefore spot size, a significantly large microwave antenna would be required, trending towards a >150 m diameter transmitter to achieve a 50 m diameter spot size. The optical system is significantly smaller, and it was found that to achieve a desired spot size targeted to provide power to small optical receivers up to large solar arrays/farms, with diameters between 1 cm and 50 m, the optical transmitter can be <2 m in diameter. In addition to the significant reduction in system sizing, optical frequencies have the benefit of tunability. In utilizing a system of either adjustable lenses or reflectors as represented in Fig. 3, the spot size on the surface can be shifted to a desired size as distance changes along the orbit and to meet specific client needs [7]. Power is also tunable with either current or voltage variation to achieve a desired irradiance based on client needs. This tunability enables a variety of receivers to be utilized on the surface and creates longevity to accommodate technology improvements on the receiving side.

The optical frequency space was chosen for further assessment due to its smaller transmitters, ability to achieve a reduced spot size, and overall system flexibility. Further research shows that the wavelength of 1064 nm (370 THz) is compatible with Si solar arrays and 810 nm (282 THz) is compatible with both GaAs and Si solar arrays. Due to the higher beam quality, 1064 nm was assumed within the scope of this study, which allows Si solar arrays to function at a ~40% efficiency [8].

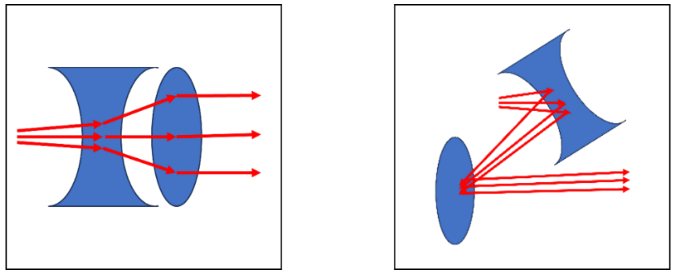

Fig. 3. Representations of conceptual focal length shifting using lenses (left) and using reflectors (right).

This tunable design also enables compatibility with the future lunar surface assets because of its adaptability to both larger receivers/solar arrays and smaller, lower mass optical receivers to be developed and utilized on the surface – resulting in continued reduction in landed mass.

D. Laser Trade Space & Technology Assessment

An assessment of the state-of-the-art of lasers was performed to inform the parametric spacecraft sizing, as well as to gauge the technology readiness levels of products on the market. The frequency selection and general ConOps provided bounding requirements for the down-selected laser. Additionally, Table 5 shows several laser technology characteristics that are highly desirable due to their reduction of on-orbit maintenance, improvements to overall system performance, and adaptability to client needs.

Table 5. Laser requirements (left) and ideal characteristics (right)

Laser Requirements	Ideal Characteristics
• ~1064 nm wavelength • Continuous wave capability • >30 kW maximum output • Power tunability • Fast wind-up to full operation • >30% energy efficiency	• Low relative mass • Diode emitter-pumped • Modifiable beam width • Built-in cooling • Resilient lasing media • High vibrational stability

Commercial off-the-shelf (COTS) products, military technologies, and scientific lasers that met the requirements were catalogued and compared based on performance. High-power candidates were either solid-state, chemical, fiber, or particle acceleration lasers pumped by a series of diode bars. Through this literature and market review, the Ytterbium (Yb)-doped laser was identified as an ideal candidate for integration into an OPAL spacecraft. These lasers use crystals or glass doped with Ytterbium, a rare-earth element, as a gain medium from which electron emission is induced. Commercial products such as the Yb-doped IPG Photonics YLS Laser Series deliver high power in the near-infrared range via fiber optics [9]. To estimate laser efficiency and size for spacecraft sizing, this COTS laser was notionally selected as the representative laser source for the optical power beaming payload. Technical specifications for this laser are found in Table 6. It is important to note that at the time of this writing neither this product nor any other high-power fiber laser has been designed or demonstrated for space applications.

Table 6. Technical specifications of IPG Photonics YLS Series

• 10-100% power tunability up to 120 kW • > 40% energy efficiency • 1070 $\pm$ 5 nm wavelength • Water-cooled	• High reliability and ruggedness • Excellent beam quality • Fast wind-up to full power • Flexible fiber delivery

E. OPAL Spacecraft Parametric Sizing

A parametric spacecraft design model was created following the definition of the power classes and the down-selection of frequency and laser type. A set of high-level ground rules and assumptions were established when creating the spacecraft model and are summarized in Table 7. Note that these parameters are not all inclusive and the values and decisions made are not final and remain the subject of future in-depth studies of the spacecraft design.

Table 7. Parametric Sizing Top-Level Assumptions

Parameter	Value / Decision	Rationale
Spacecraft Lifetime	10 years	To service clients with similar planned lifetime.
Solar Array Technology	Roll-Out Solar Arrays (ROSA)	Technology deployed on the International Space Station (ISS) and soon will be deployed on Gateway.
Vehicle Delivery	Commercially provided chemical stage	To reduce parasitic mass on vehicle after lunar orbital insertion.
Main Propulsion System / Thruster Model / Electric Power Required	SEP / 8x Busek BHT-1500 / 12 kW [10]	Each EP thruster requires 1.5 kW to operate.
Propellant Selection / Amount	Xenon / 200 kg	Computed value is ~110 kg. Set at 200 kg to accommodate growth and margin.
Laser Suite Mass (Optics, electronics & structure)	424 kg	Based on other optical power beaming concepts from industry plus growth and margin.
Spacecraft Bus Mass	380 kg	Assumption, to be refined in future analyses.
Thermal Control System - Acquisition	Cold-Plate	Several terrestrial lasers utilize this approach.
Thermal Control System - Rejection	Active Pumped-Fluid Loop Flexible Fabric Radiator 60% prop glycol, 40% water	Default model value, to be assessed.
Power Management and Distribution (PMAD) Efficiency	79.4%	Default model value, to be assessed.

The parametric sizing itself was completed by running full factorial analyses considering four input variables: 1) Laser DC to Optical Power Conversion Efficiency (η_{DC2O}), 2) Solar Array Cell Efficiency, 3) Solar Array Specific Mass, and 4) Battery Specific Capacity. Table 8 represents the value ranges in this analysis for the four input parameters from worst-case to best-case.

Table 8. Input parameters for parametric spacecraft design from worst-case to best-case

	Worst-case				Best-case
Laser DC to Optical Power Conversion Efficiency	25%	30%	35%	40%	45%
Solar Cell Efficiency	19.5%	23.5%	26%	29.5%	33%
Solar Array Specific Mass (kg/m^2)	5	3	1.0	0.5	0.1
Battery Specific Capacity (Wh/kg)	160	180	200	220	240

The η_{DC2O} was identified as the major driver for spacecraft sizing, whereas the other three variables affect the sizing to a lesser extent. The laser requires a large electrical power input and is highly inefficient, thus this results in a need for large power generation and waste heat rejection. In turn, this translates into large and relatively heavy solar arrays and radiator systems. Because the η_{DC2O} is one of the major sizing drivers, the spacecraft sizing results are presented as a function of this variable. Following the power class definitions in Table 2, a Class I spacecraft would require a laser with P_{beam} = 4 kW, which is a lower power demand than the SEP power demand of 12 kW. Therefore, the power generation for a Class I spacecraft is driven almost entirely by the SEP power requirement. The decision to implement SEP or chemical propulsion thus becomes a broader discussion including trades of spacecraft dimensions, mass, control, pointing stability, mission design, among other topics. To make a more direct comparison, Fig. 4

presents the power generation requirement based only on the laser power load for each spacecraft class after accounting for PMAD efficiency, transfer losses, and time that the OPAL spacecraft would spend in eclipse throughout its orbit. Under these conditions, the power generation requirement for a Class I spacecraft would range between 20 and 36 kW, well under the 60 kW produced by the Gateway Power and Propulsion Element (PPE) [11]. In contrast, a Class II would require higher power generation between 80 and 143 kW and a Class III spacecraft would require between 143 and 269 kW, narrowly above the International Space Station (ISS) power generation of 250 kW [12]. Note than when accounting for efficiency and degradation losses, as well as power loads from other subsystems, the actual power generation requirement and the power generation capacity at Beginning of Life (BOL) will be larger than shown in Fig. 4.

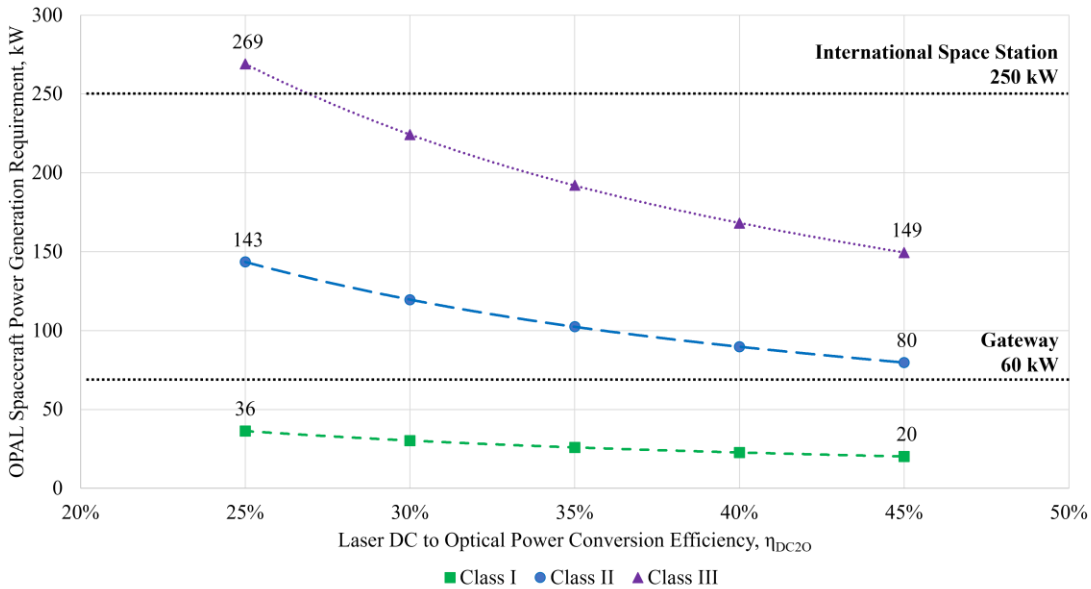

Fig. 4. Power generation required for Class I, Class II, and Class III spacecraft.

A similar trend is observed in the impacts of η_{DC2O} to the radiator sizing as shown in Fig. 5. For a Class I spacecraft, the radiator area required ranges from 169 to 251 m^2 with the former area being practically as large as the ISS photovoltaic radiators which have an area of ~170 m^2 [13]. A Class II spacecraft would require between 376 and 731 m^2 of radiator area with the former approximating the radiator area of the Jupiter Icy Moon Orbiter (JIMO) Technical Baseline (TB) 2.5 design of ~346 m^2 [14].

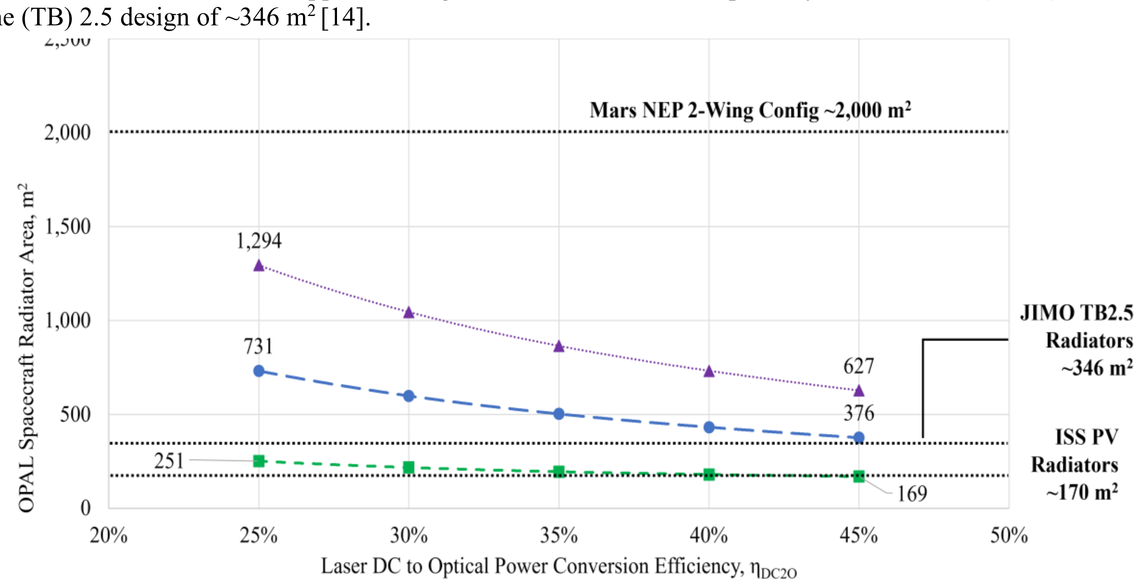

Fig. 5. Radiator area required for Class I, Class II, and Class III spacecraft.

Finally, a Class III spacecraft would require between 627 and 1,294 m^2 of radiator area that, although significantly large for this application, is not as large as the radiator area required of >2,000 m^2 for proposed Mars Nuclear Electric Propulsion (NEP) bi-wing spacecraft concepts [15].

Figure 6 shows significant spacecraft dry mass reductions between 20 and 48% are achieved when improving the η_{DC2O} from 25 to 45%. These results for spacecraft dry mass should be considered as first-order estimates since the data utilized for these calculations is sourced from historical data, parametric relationships, assumptions in each parametric subsystem model, and the top-level assumptions presented in Table 7. A common trend found across all spacecraft classes is that the heat rejection subsystem within the thermal control system – or radiators – comprise between 41 and 61% of the spacecraft dry mass. Further analysis, including a more intricate thermal control system design, could improve the radiator mass and other spacecraft subsystems.

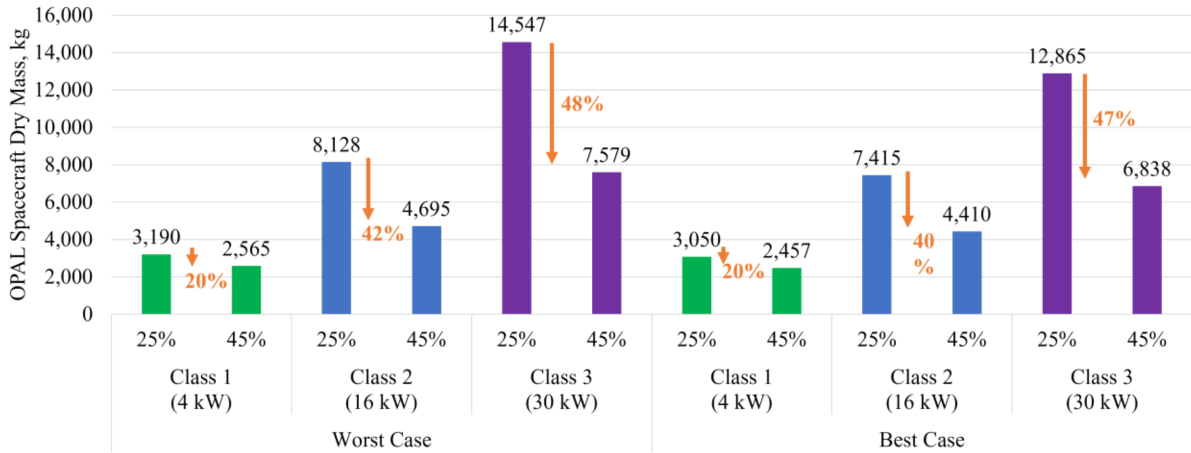

Fig. 6. Dry mass comparison for all spacecraft classes as a function of η_{DC2O} = 25% and 45%.

From Fig. 7 it can be noted that a Class I spacecraft appears to be a more feasible and immediate option as it resembles a more traditional single satellite design and would not necessarily require in-space assembly (iSA) to deploy its solar arrays or radiators. Additionally, a spacecraft of this size would be easier to control and enable coarse pointing, which is essential to achieving overall fine pointing when combined with the payload pointing subsystem. Most importantly, a Class I spacecraft - or similar - with a power-tunable and focusable optical power beaming payload has the potential to service clients operating at low power ranges such as 50 to 250 W as demonstrated in Section IV. Improvements to solar cell, laser, iSA, and other technologies could result in more viable Class II and III systems to further support higher power missions on the lunar surface. However, additional assessments regarding the cost, benefit, and complexity of building Class II and III spacecraft could provide a better understanding of the value of this concept at such scales.

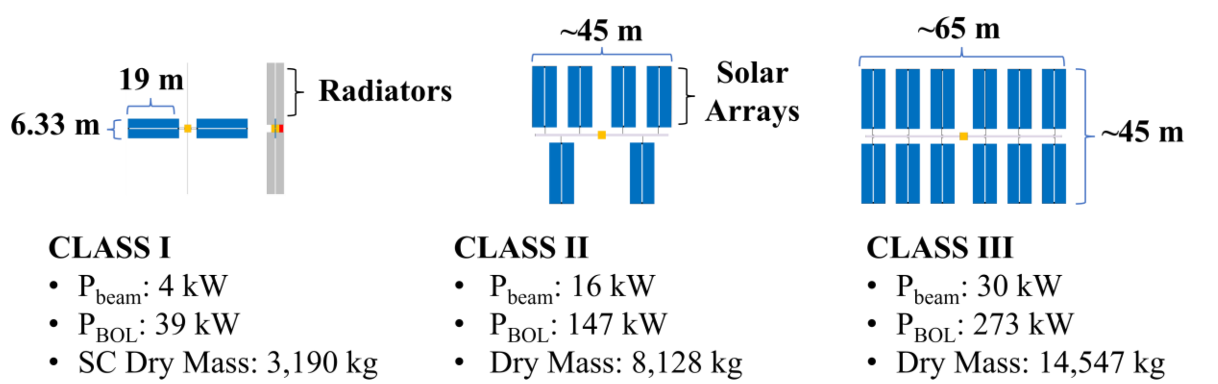

Fig. 7. OPAL worst-case spacecraft drawing with corresponding required beam output power, P_{beam}, power generation at Beginning of Life, P_{BOL}, and dry mass. Radiators only shown for Class I spacecraft.

IV. Case Study

A first order assessment, or case study, was completed to assess the potential impacts that an OPAL architecture could have on a Commercial Lunar Payload Services (CLPS)-type lander operating between 50 and 250 W. The first part of this analysis focused on sizing the battery required for the lander to operate through a lunar night (~360 hours) in equatorial regions or in equivalent shadow conditions at the lunar south pole. For study purposes, battery specific energy was traded between 160 and 200 Wh/kg. As shown in Fig. 8, the battery mass required to complete a lunar night survival mission far exceeds the payload mass capacity of the three example CLPS landers (Nova-C [16], Peregrine [17] , and Blue Ghost [18]).

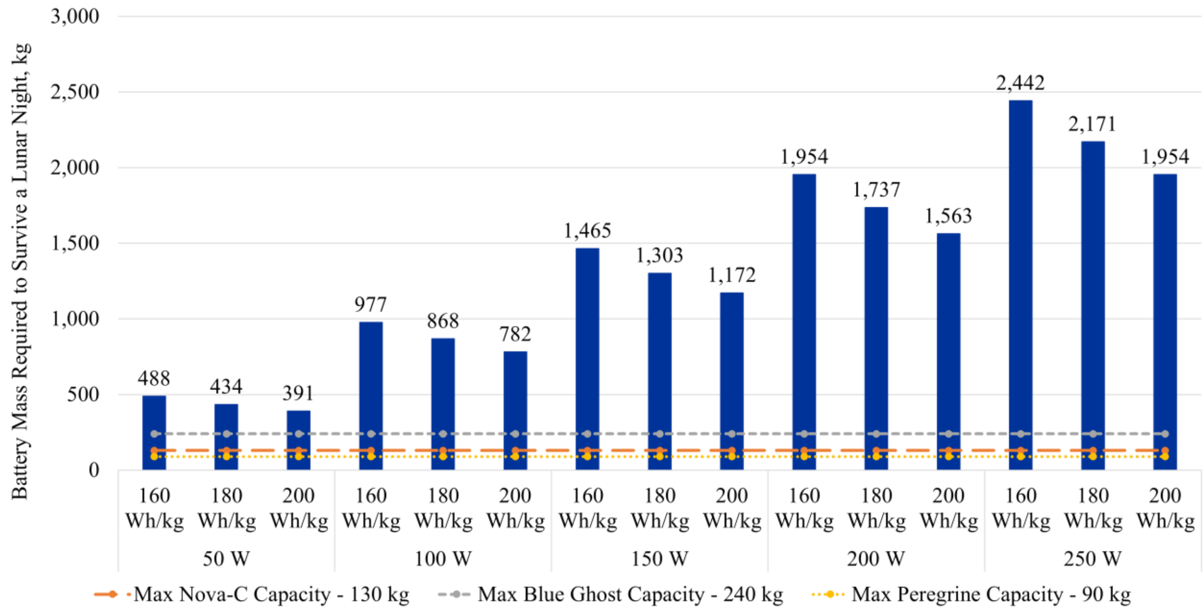

Fig. 8. Battery mass required for a CLPS lander to continuously operate for 360 hours at a range of lander average operating power from 50 to 200 W with different battery specific energies from 160 to 200 Wh/kg.

Furthermore, a charge-discharge analysis shown in Fig. 9 was conducted for a CLPS-type lander operating at 200 W continuously with a battery capacity of 1.3 kWh. This analysis compares the SOC of the lander when it is not serviced by an OPAL spacecraft against the SOC when the lander is serviced by a one, two, and three OPAL spacecraft constellation in the chosen near-polar circular orbit. Note that the size of the OPAL constellation determines the time the lander must operate on battery power – one spacecraft equals ~3 hr on battery power, two spacecraft equals ~2 hr, and three spacecraft equals ~0.33 hr (~20 mins of handover between spacecraft).

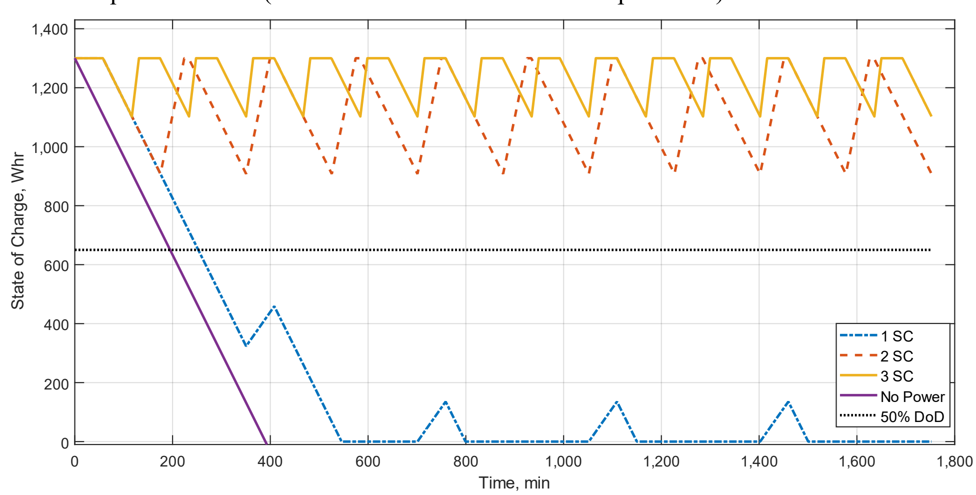

Fig. 9. State of Charge for a lander operating at 200 W with a 1.3 kWh battery.

A lander with the specifications listed above could only survive 6.7 hr into the lunar night or in a shadowed region with no additional power augmentation. In the case where only one OPAL spacecraft beams power to the lander, the lifetime of the lander increases to 9.2 hr. However, a lander could survive the ~3 hr in darkness in this scenario if its battery size was increased. In contrast, with two or more OPAL spacecraft the lander would remain well above a 50% SOC (or below 50% Depth of Discharge (DOD)) until the designed end of mission as shown in Fig. 9. The results presented in Fig. 8 and Fig. 9 show that for the assumed conditions it is prohibitive to continuously operate through lunar night or in shadowed conditions, but that doing so appears possible when serviced by an OPAL constellation.

To better assess the impacts of operating CLPS-type landers at different power levels while being serviced by OPAL constellations of variable size, the power system (including batteries, solar arrays, and PMAD) was parametrically sized. This exercise assumes that each lander is serviced by a dedicated constellation, that the lander is located somewhere in the south pole within range of the spacecraft, that the lander contains batteries with specific capacity of 160 Wh/kg, and that the lander carries a receiver with an optical to DC conversion efficiency, η_{O2DC}, of 40% and does not receive energy from the Sun.

As observed in Fig. 10, a constellation of three spacecraft results in the smaller battery sizing (0.45 to 2.24 kg), as the lander would only spend ~0.33 hr in darkness during handover from one spacecraft to another. In the case where only a single OPAL spacecraft is available to service the lander, it will spend ~3 hours in darkness thus resulting in a larger battery sizing (4.07 to 20.35 kg). When compared to the scenarios in Fig. 8, battery mass reductions from 95 to 99% would be achieved when the lander is serviced by an OPAL constellation.

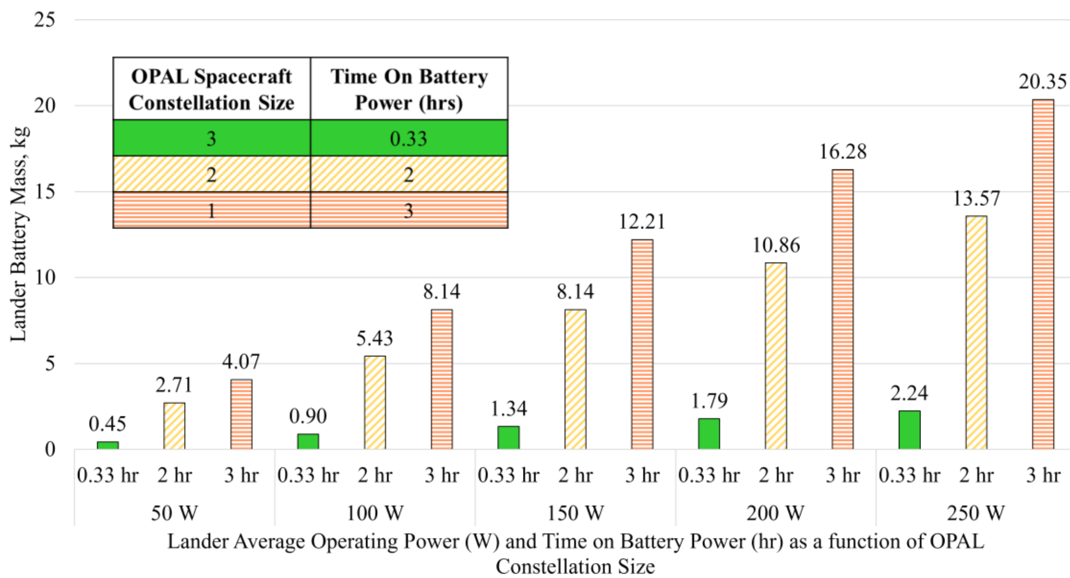

Fig. 10. Battery mass required to survive eclipse with an OPAL architecture.

The results from the lander power system sizing reveal, as expected, that higher power requirements yield larger receiver areas. Additionally, as shown in Fig. 11, these results also reveal that receiver area is inversely proportional to spacecraft in the constellation – more spacecraft in constellation equals more access and power beaming time thus reducing the amount of energy that must be captured per pass and reducing the receiver area required to satisfy the power demand.

Balancing the receiver size against the spot size achievable by the optical system on board the OPAL spacecraft is important to maintaining proper irradiance on the receiver thus ensuring efficient and effective energy transfer. This proper balancing is also essential to minimizing beam spillover thus preventing disturbances to other client subsystems such as sensitive payloads. Lastly, controlling the beam size can also help reduce or prevent potential distortions to the lunar environment such as volatilization of important resources inside permanently shadowed regions (PSRs).

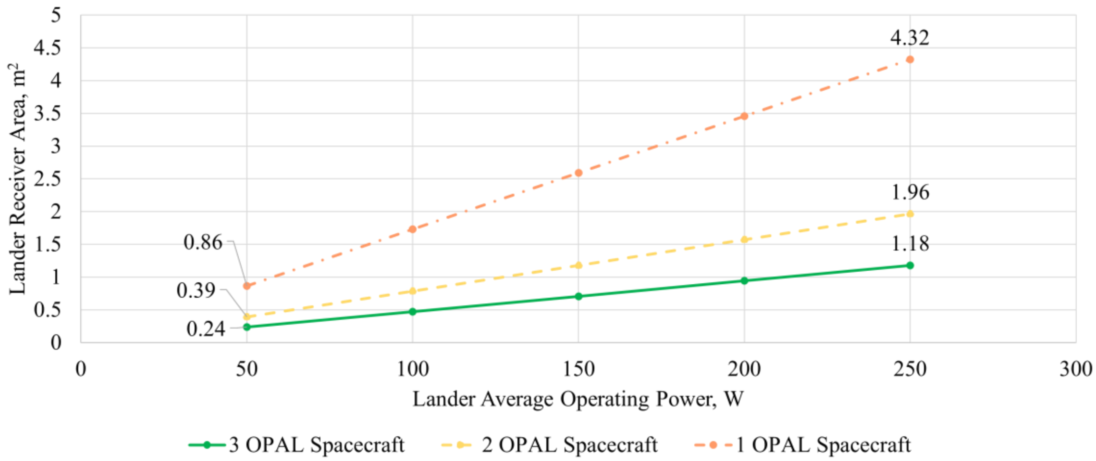

Fig. 11. Lander receiver area as a function of lander average operating power and OPAL constellation size.

Lastly, the beam power required to satisfy the power demands of the lander was computed for each constellation size following the equations from Table 1 to assess which cases could be serviced by which OPAL spacecraft class. As shown in Fig. 12, ten combinations (lander average operating power x constellation size) could be serviced by an OPAL Class I spacecraft with a $P_{beam} = 4$ kW. From Fig. 12 it is also seen that to satisfy more than 100 W of power with a single spacecraft would require a laser source with $P_{beam} > 4$ kW. Similarly, an OPAL spacecraft with a laser source with $P_{beam} \sim 11$ kW could theoretically service all combinations displayed in Fig. 12. The results in both Fig. 11 and Fig. 12 serve as further evidence towards the added value of developing a power-tunable and focusable optical power beaming system that can energize many clients according to their power demand, designs, and operational constraints.

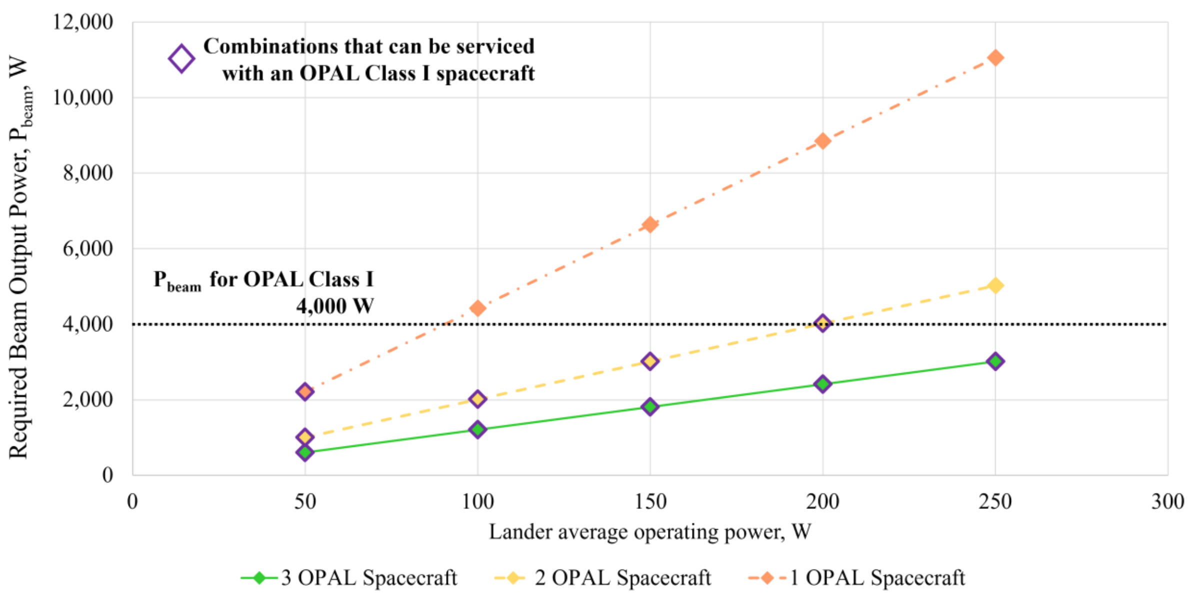

Fig. 12. Required beam output power, P_{beam}, as a function of lander average operating power and OPAL constellation size.

V. Conclusions & Future Work

The potential of orbital power beaming to enable significant mass savings and improved operations for lunar surface missions has been shown through the analysis presented in this paper. The operational deployment of an OPAL spacecraft constellation could contribute to extended mission lifetime and enable an increase in landed payloads (in lieu of energy storage mass), which could yield a higher Return on Investment (ROI). Although not discussed here, an OPAL spacecraft also has the potential to enable new operational paradigms such as powering assets as they explore into and through PSRs and other extreme regions on the lunar surface thus opening new real estate and relaxing otherwise strict mission planning.

The value of OPAL may be further improved with the implementation of a power-tunable and focusable optical power beaming system as this would enable power beaming to multiple assets with the same orbiting spacecraft. Ongoing research and analyses are aimed towards better understanding and defining top-level requirements for the power-tunable and focusable system, quantifying the impact of the system on both the OPAL spacecraft and client power system parametric sizing, and contextualizing the implications to the OPAL spacecraft and client ConOps. The current phase of the OPAL project also includes high-level Gaussian beam propagation calculations and preliminary optical system sizing to add fidelity to the analyses. Future steps may include additional work on the optical suite design (laser sourcing, beam conditioning, beam pointing, and optical surfaces design) and subsystem, spacecraft, and overall mission architecture cost analysis.

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