

Power Beaming from Lunar Orbit for Small Science Landers

Geoffrey A. Landis¹, Steven R. Oleson², Ben Abshire³, Bushara Dosa⁴, Brent Faller², Ryan McDonough⁵, Lucas Shalkhauser³, Peter Simon³, Elizabeth Turnbull², Natalie Weckesser⁵
NASA John Glenn Research Center, Cleveland, OH 44135, USA

Anthony Colozza⁶, John Gyekenyesi⁶, Thomas Packard⁶, David Smith⁶
HX5, LLC, Fort Walton Beach, FL 32548, USA

James Fittje⁷
Science Applications International Corporation, Inc., Brunswick, OH 44212, USA

David Squires⁸
Ashford, CT 06278, USA

A proposal for future small science mission to the moon envisions a network of small landers spread across the lunar surface, at latitudes ranging from equatorial to near polar landing sites. To provide power across the lunar nighttime, when solar power is not available, we analyze a proposal to power such small landers from orbit, using a laser to direct power from an orbital power station to photovoltaic arrays on the landers that are tuned to the laser wavelength. The approach is seen to be feasible, and conops for the system were outlined and a design for the spacecraft put together. To provide power to landers at any location, three orbital stations are required, each carrying a 3-kW laser.

I. Introduction

A proposal for future small science mission to the moon envisions a network of small landers spread across the lunar surface, at latitudes ranging from equatorial to near polar landing sites [1]. A difficulty for such a network, however, is providing power across the lunar nighttime, when solar power is not available. Due to the length of the lunar night, batteries power for even low power (~50 W) night operation would require addition of 250 kg or more of battery and thermal insulation mass, making such a solution infeasible, and although isotope thermal power systems might be a possible solution to this problem, the cost and availability of plutonium would make this impractical.

¹ Researcher, Power Division, NASA Glenn Research Center. AIAA Associate Fellow.

² System Engineering and Architecture Division, NASA Glenn Research Center.

³ Power Division, NASA Glenn Research Center.

⁴ Communications and Intelligent Systems Division, NASA Glenn Research Center.

⁵ Program Planning and Control Office, NASA Glenn Research Center.

⁶ Senior Research Engineer, Cleveland Program Office, HX5.

⁷ Aerospace Engineer Science Applications International Corporation, Inc. AIAA Senior Member.

⁸ Optical Systems Consultant, Ashford, CT 06278.

An attractive alternative would be to power such small landers from orbit, using a laser to direct power from an orbital power station to photovoltaic arrays on the landers that are tuned to the laser wavelength. In this work we analyze the feasibility of such a power system.

II. Summary

A. Baseline Mission

The baseline mission was to analyze use of a power and relay “beamcraft” that could beam power to a number of science landers on the surface, to support nighttime power for operation of at any location on the moon, with a target operation timeframe of the early 2030s. A 50-Watt night power requirement was adopted as representative need of a 100-kg science on a CLPS-style lander. The requirement was that the orbital power stations would be able to service multiple landers at any location of the moon, with single-fault tolerant operation. The primary figure of merit for the analysis was the reduction of science lander landed mass (assumed to cost roughly \$1.2M/kg delivery). An added benefit of the orbital beamcraft stations was the ability to use the beamcraft as communications relay stations, giving high bandwidth communications for both lunar near- and far-side operations.

While beamed power architectures for the moon have been examined before [2], this approach takes advantage of recent advances in laser technology, and proposes a much smaller system than previously proposed, suitable for small landed science stations.

B. Technology Choice

Several technologies for the power beam were examined. Laser power beaming was chosen for the small spot size capability, minimizing the size of the beam director on the spacecraft and the receiving array. Laser power beaming also has the advantage that photovoltaic receivers tuned to the laser wavelength are also capable of being operated in sunlight, and hence the same array can provide both daytime power (from sunlight) and night power (from the laser beam).

Similar to the requirements for surface beamed power applications [3], the requirements on the laser were:

- High efficiency (optical power out/electrical power in)
- Good beam quality
- Available at high continuous-wave power levels
- High Technology Readiness Level (TRL)
- Availability of photovoltaic cells at the laser wavelength

Based on these requirements, the laser technology chosen was a diode-pumped fiber laser, with a high beam quality at a wavelength of 1.07 microns.

For a laser with wavelength matched to the peak efficiency of the photovoltaic receiver, power conversion efficiency of the receiver is typically on the order of 50%.

A minimum of three laser spacecraft are required to provide power to any location on the moon. Within this constraint, it was desirable to keep line of sight from the beamcraft to the lander to shortest reasonable distance, in order to minimize the spot size on the surface. Two geometries met the criteria, either two orbiters in orthogonal polar orbital planes plus another in equatorial orbit, or three spacecraft in lunar polar orbits with orbital planes separated by 60°. The second of these gave superior view times for mid-latitudes and was chosen for detailed analysis, although both geometries met the requirements. For both cases, the orbital altitude of 800 km above the surface was chosen, with an orbital period of 3.17 hours. A line of site visualization was done using the Satellite Orbit Analysis Program (SOAP) software tool [4]. Figure 1 shows this visualized, with the three spacecraft orbits labeled o1, o2, and o3, and the field of view from the spacecraft to the ground shown. The maximum distance for the beam to travel is 1600 km, while the minimum distance is the orbital altitude 800 km. The 800-km polar orbits were propagated forward for a ten-year period using the Copernicus software [5], showing that perturbations give small changes in eccentricity and inclination, but the orbits remain stable even with no propulsive stationkeeping.

In the nominal concept of operations, the orbiting satellite comes into view of the lander, and initiates the power beaming during the pass, charging the batteries of the lander to provide power until the next pass occurs. The pass duration ranges from over 35 minutes, for the best-case geometry of the satellite passing directly overhead, to a worst-case pass duration of 10 to 12 minutes. This reduces the battery needed to survive and operate during the lunar night from requiring a duration of 354 hours of operation down to a slightly less than 3 hours, a reduction by a factor of a hundred in capacity and hence required battery mass.

For near-equatorial sites, much of the time only one satellite will come into view, with the others not rising above the horizon. Figure 2 shows the envelope beam range and angle for a satellite in one of the three orbital planes over

the course of a single lunar day, showing the 14-day periodicity. As the moon rotates, the minimum range to the satellite on each pass will decrease and then increase, until the satellite is not available (a 10° horizon mask is assumed), and the beaming task is passed off to the satellite in the next orbital plane. The hand-off between orbital planes is shown in figure 3, for the worst case, an equatorial ground station.

For sites farther from the equator, two of the satellites will be able to provide power on each orbit, and for sites within about 30 degrees of the pole, all three satellites will be in view on every orbit. The laser is used to power the landers only during the local night; during the daytime, the lander array that receives laser power is used as a solar array, although at slightly lower conversion efficiency.

The need for three orbital spacecraft is due to the requirement for the system to be able to power landers at all latitudes as the moon rotates. If the landers to be powered were distributed only to near-equatorial (within $\sim 30^\circ$ latitude of the equator) sites, a single beamcraft in equatorial orbit would be sufficient; likewise, to distribute power only to near-polar sites (within $\sim 30^\circ$ latitude of the poles), a single polar-orbiting beamcraft would be sufficient. This suggests a deployment strategy of deploying a single beamcraft satellite to serve the initial landers, and then adding additional orbital planes as more landing locations are needed.

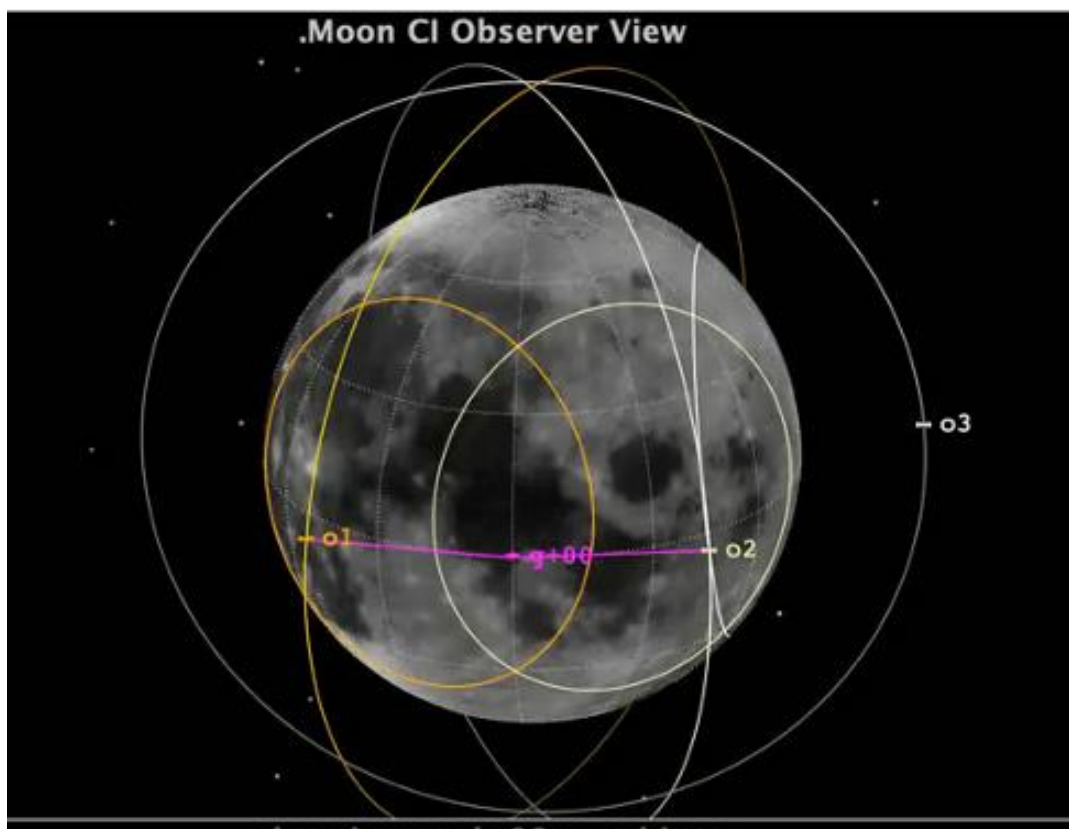


Fig. 1 Three polar lunar orbital planes, showing the overlap of the field of view of the lasers to the surface.

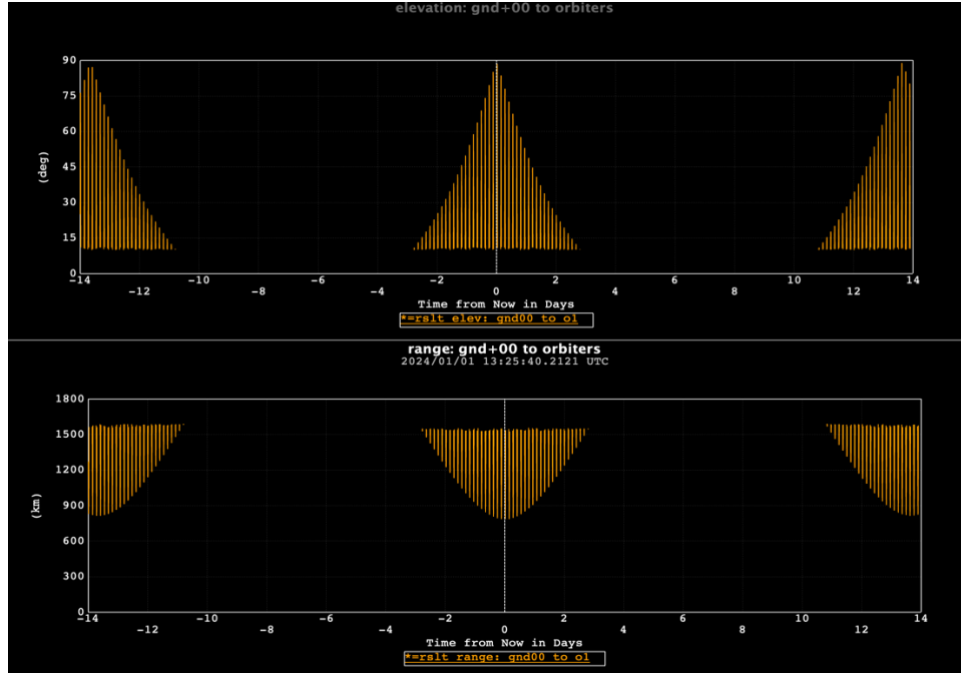


Fig. 2: SOAP analysis of the view factor from the orbiter to an equatorial ground station (top: elevation angle, bottom: range).

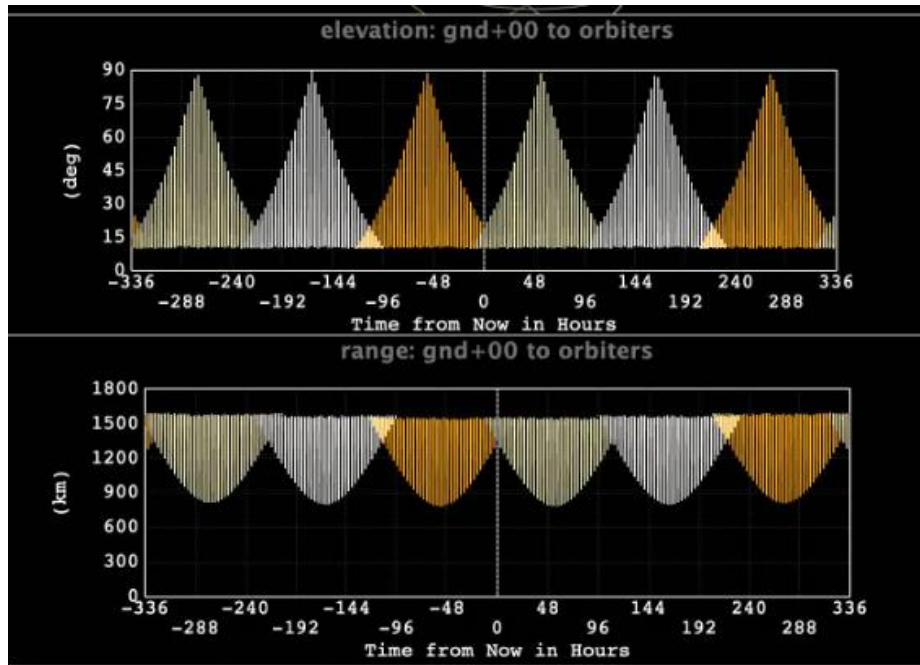


Fig. 3: SOAP analysis of the view factor from all three orbiters, showing hand-off from the orbiter in one orbital plane to the next as the moon rotates. (top: elevation angle, bottom: range).

C. Spacecraft design

A commercially-available fiber laser was selected. Although the laser selected does not have space heritage, it is available at power levels up to 6 kW, with an electrical to optical conversion efficiency of 38%. For fault tolerance, two identical lasers were assumed. A thermal radiator is used to radiate away waste heat to keep the laser within operating thermal limits. The beam is emitted from the fiber and directed via a lens to an offset parabolic mirror, which

collimates the beam and directs it to the surface. A 1.4-meter diameter mirror was selected; this is the same diameter as the mirror used on other missions such as the Kepler spacecraft [6]. While lighter-weight mirrors have been proposed, a mirror modeled on the Kepler mirror was baselined for risk reduction, because the technology for mirrors of this size is well developed and proven in space [7].

Figure 5 shows the profile of the beam in the far-field, defining the spot size on the surface. The main mirror was sized such that, at the maximum beaming distance of 1500 km, a 1.5-meter diameter array will equal the diameter of the full-width half maximum (FWHM) of the beam, which includes roughly half of the beam power. This is a compromise between minimizing array area and maximizing beam collection; doubling the array diameter to include the majority of the Airy disk would increase power collection by about 68%, but would increase the required array area by a factor of four (and also lead to a less uniform illumination on the array). As the spacecraft approaches closer, the beaming distance decreases and the minimum spot-size on the surface proportionately decreases; the beam is slightly defocused to keep the illumination filling the array and avoid a high-intensity hot-spot.

Using these parameters, a conceptual design for the orbital beamcraft was done (Figure 4), along with the design of the receiving array to be incorporated on the lander.

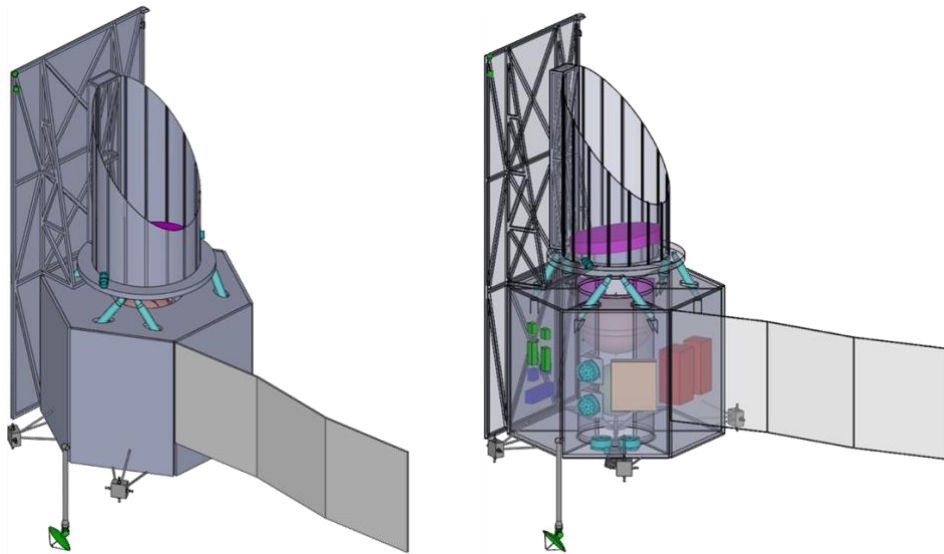


Fig. 4. Solidworks visualization of the orbital spacecraft carrying the laser and mirror, showing exterior view (left) and internal components (right).

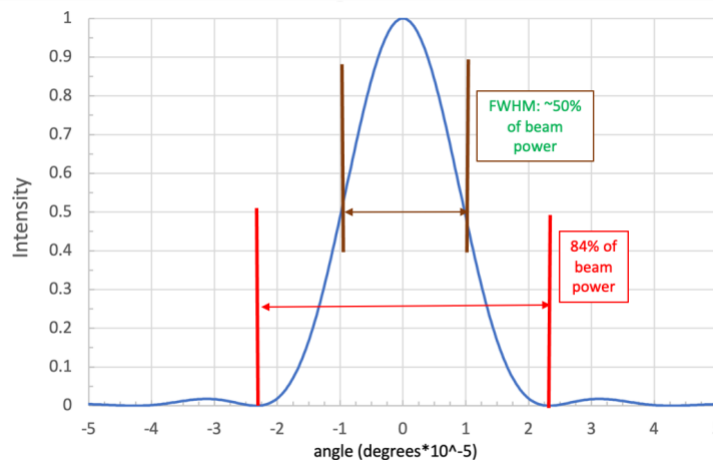


Fig. 5. Diffraction-limited far-field beam spread of the laser in degrees, showing the intensity at the full-width half-maximum (FWHM) diameter, as well as the diameter of the Airy disk, containing 84% of the beam power.

Description				
Case 1_Lunar_Beam_Craft CD-2023-207	Basic Mass	Growth	Growth	Total Mass
	(kg)	(%)	(kg)	(kg)
Lunar Beam Craft				
Spacecraft	2818	14%	401	3220
Laser and optics	186.3	24%	44.6	230.9
Attitude Determination and Control	108.0	7%	7.7	115.7
Command & Data Handling	39.2	36%	14.1	53.3
Communications and Tracking	12.8	11%	1.4	14.2
Electrical Power Subsystem	450.5	35%	157.7	608.2
Thermal Control (Non-Propellant)	197.1	18%	35.5	232.6
Propulsion (Chemical Hardware)	114.1	5%	6.0	120.1
Propellant (Chemical)	964.0	0%	0.0	964.0
Structures and Mechanisms	746.4	18%	134.3	880.7

Table 1. Orbital spacecraft mass breakdown by top-level subsystem.

Table 1 shows the mass breakdown of the spacecraft, including the mass growth allowance per AIAA mass estimation standards [8]. Total mass of the orbital spacecraft is 3220 kg, including the mass of the propulsion system, 1084 kg, needed to insert the spacecraft into the required lunar orbit. (Note that while propellant growth is not included explicitly; this is incorporated as propellant margin in the propulsion subsystem analysis.) This is well inside the capability of a reusable Falcon Heavy or equivalent to launch to the translunar trajectory, with about 40% additional payload capacity available for other lunar payloads, such as a CLPS-style lunar lander [9]. If a slightly more capable launch vehicle becomes available by the timeframe required, it could be possible to launch two beamcraft on a single launch. Future work could include an analysis of alternatives to the direct transfer approach taken here. One such alternative would be to use a weak stability layer transfer which can have significant ΔV and mass savings, reducing required ΔV by hundreds of m/s (with associated decrease in propellant load), at the expense of transfer time and operational complexity [10].

III. Conclusions

A design was made for a power system for a network of lunar landers, in which a lunar orbital beamcraft provides power to the landers for operation during the lunar night. Analysis shows that the approach is feasible. Since each beamcraft can provide night power to multiple landing sites, this approach becomes increasingly attractive as the number of landing sites needing power increases.

IV. References

- [1] Haviland, H. F., Weber, R. C., Neal, C. R., Lognonné, P., Garcia, R. F., Schmerr, N., ... & Bremner, P. M., "The lunar geophysical network landing sites science rationale," *The Planetary Science Journal*, 3(2), Feb. 2022, 40.
- [2] Fay, E. H., and Cull, R. C., "Lunar orbiting microwave beam power system", *25th Intersociety Energy Conversion Engineering Conf.*, Reno, NV, August 12-17, 1990; NASA-TM-103211.
- [3] Landis, G. A., "Laser Power Beaming for Lunar Polar Exploration," paper AIAA-2020-3538, *2020 AIAA Propulsion & Energy Forum*, New Orleans LA, August 24-26, 2020.
- [4] Stodden, D.Y. and Galasso, G.D., "Space system visualization and analysis using the Satellite Orbit Analysis Program (SOAP)," *1995 IEEE Aerospace Applications Conference. Proceedings Vol. 1*, February, 1995, pp. 369-387.
- [5] Williams, J., Kamath, A.H., Eckman, R.A., Condon, G.L., Mathur, R. and Davis, D.C., "Copernicus 5.0: Latest Advances in JSC's Spacecraft Trajectory Optimization and Design System," *AAS/AIAA Astrodynamics Specialists Conference*, Portland, ME, August 2019.
- [6] Kosi, K., "Focus Mechanism for Kepler Mission," *39th Aerospace Mechanisms Symposium, NASA/CP-2008-215252*, p. 373; <https://ntrs.nasa.gov/api/citations/20080023060/downloads/20080023060.pdf#page=373>
- [7] Stahl, H. P., "Advanced ultraviolet, optical, and infrared mirror technology development for very large space telescopes," *Journal of Astronomical Telescopes, Instruments, and Systems*, 6, No. 2, 025001, 2020.

<https://www.spiedigitallibrary.org/journals/Journal-of-Astronomical-Telescopes-Instruments-and-Systems/volume-6/issue-2/025001/Advanced-ultraviolet-optical-and-infrared-mirror-technology-development-for-very/10.1117/1.JATIS.6.2.025001.full>

- [8] AIAA S-120A-2015, *Standard Mass Properties Control for Space Systems*, 2019.
- [9] von Ehrenfried, M., “Spacecraft, landers, rovers and payloads,” *The Artemis Lunar Program: Returning People to the Moon*, Springer, 2020, pp.48-74.
- [10] Parker, J. S. and Anderson, R. L., *Low-energy lunar trajectory design (Vol. 12)*. John Wiley & Sons, 2014.