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INTERNATIONAL SPACE SOLAR POWER STUDENT COMPETITION PAPER NO. 1

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Abstract

The purpose of this research is the definition of a lunar orbiting system that provides energy on the lunar surface through wireless power transmission. To achieve the goal of supplying the energy demand of a lunar base a constellation of satellites is used, placed in particular stable orbits also called *frozen orbits*. The altitude is then obtained from a trade-off study optimizing for eclipse time, system efficiency, coverage, receiver surface area, spacecraft weight. Every satellite of the constellation is composed of solar arrays that provide energy to a laser that transmits power toward the receivers on the lunar surface. The receivers use photovoltaic cells optimized for the laser's monochromatic light and a shape that minimizes the cosine losses of the transmission. A study is conducted considering a receiver on the lunar surface in the strategic position of the landing site for the Artemis missions, the Shacketlon crater near the lunar south pole. The aim is to provide the night-time energy requirement of a Lunar base estimated to be more than 22 MWh per year.

Keywords: space solar power, moon, laser transmission, rectenna, monochromatic photonic power converters, wireless power transmission, satellite constellation, spherical shell receiver.

Acronyms/Abbreviations

- EPS: Electrical Power System
- DoD: Depth of Discharge
- AODCS: Attitude and Orbit Determination and Control System
- PPC: Photonic Power Converter
- COTS: Commercial Off-the-Shelf
- CW: continuous-wave
- QCW: quasi-continuous-wave
- NASA: National Aeronautics and Space Administration
- ORiS: Orbital Recharge in Space

1. Introduction

1.1 Background

ORiS was born in march 2022 thanks to five MSc students in aerospace engineering at Politecnico di Torino who have decided to dive into the world of entrepreneurship by following their passion for Space.

1.2 ORiS' objectives

The urge to explore space and especially our solar system has led some government agencies, including the National Aeronautics and Space Administration (NASA) with three partner agencies: the European Space Agency (ESA), Japan Aerospace Exploration Agency (JAXA), and Canadian Space Agency (CSA), to invest in the Artemis program. One of the program's goals is to establish a Lunar base. The permanent base camp on the Moon will be used as a springboard for the exploration of Mars, indeed the psychophysical behavior of astronauts will be studied and new technologies will be tested. The energy required for a lunar or martian base is a much higher amount than for usual space missions; One of the main solutions to this problem is to oversize solar panels on the lunar surface to guarantee energy demand even at night. This involves bringing much more mass (more solar panels and energy storage technology) to the lunar surface, increasing launch costs in proportion to the increase in mass. For this reason, the solution proposed by NASA to reestablish a human presence on the Moon since the Apollo 17 mission in 1972 is to place the base on the rim of the Shackleton crater; indeed the location is optimal for its minimal eclipse time, and solar radiation provides enough energy for the Base. This solution proposed by NASA is not applicable to any situation, but is specifically designed for the Artemis mission. Contrarily, the solution proposed by ORiS is a satellite constellation that has the advantage of energetically supplying lunar bases positioned at any point on the lunar surface, allowing for further exploration to places where it is otherwise impractical. The single satellites receive energy from the Sun and convert it to power a laser that transmits wireless energy to the lunar surface, where receivers such as rectennas or monochromatic PPCs will convert it to electrical power. The solution will be adaptable to other celestial bodies such as Mars, Phobos, and Deimos. The objective of this paper will be to demonstrate that a constellation of *n* number of satellites is able to provide for the annual nighttime energy demand of a Moon base. The system's main features, such as dimension and shape of the receiver, type of receiver, the satellites' mass, number of satellites *n*, laser output power, satellite solar array dimensions, and altitude will be discussed.

2. Material and methods

Software such as STK, Matlab and Simulink were used for simulations and calculations in this paper.

3. Theory and calculation

3.1 Orbit Selection

The candidate orbits for the ORiS constellation are:

1. Earth-Moon libration point L1 circular halo orbit
2. Earth-Moon libration point L2 circular halo orbit
3. Low Lunar Near – Circular Orbit
4. Medium Lunar Near – Circular Orbit
5. High Lunar Near – Circular Orbit

In terms of choosing an orbit that maximizes the energy stored on lunar surface and minimizes overall cost excludes L1 and L2 Halo orbits, because of the distance from the lunar surface. Such a big distance leads to having very large receiver surfaces, which means more mass to bring from the earth to the moon and consequently a much higher mission cost. The amount of station-keeping propellant is also a disadvantage for these orbits. Despite the advantage of being almost always in sunlight, these orbits were preliminarily excluded from the trade-off study. The Low Lunar Orbit, below 100 km of altitude, suffers from gravitational perturbations effects. The gravity field of the moon makes most LLO unstable. In recent years, however, several studies demonstrate the existence of low-eccentricity frozen orbits in all the range of inclinations [34]. These orbits have low coverage of receivers sites, which means having to consider more satellites inside the constellation and consequently a greater cost of the whole mission, and/or to consider a more powerful laser for each satellite (with a lower TRL). The Medium Lunar Near – Circular Orbits, 100 to 500 km range, suffer less for lunar gravitational perturbations but more from gravitational perturbations relative to the earth. For long periods of time, the evolution of these orbits is a victim of the Lidov-Kozai effect [1], and for this reason significant station-keeping operations are required for a long mission. High Lunar Near – Circular Orbit (altitude > 500 km) are similar to the previous orbits, although with increasing altitude the pointing accuracy requirement is higher, increasing cost and receiver surface.

After this analysis, a Lunar Near – Circular Orbit has been selected, with an altitude range between 300 and 700 km.

3.2 Altitude Selection

A trade-off study has been conducted to understand the best operative altitude on the basis of four high level Figures of Merit:

- Energy delivered to a lunar base in one year
- Receiver dimensions, without considering pointing error
- Pointing Accuracy
- Charging times and EPS mass

Each figure of merit is analyzed in detail in the following subparagraphs.

3.2.1 Energy delivered to a lunar base in one year

The power generated on board the satellite is converted in electric power (η=0.95) and then laser output power (η=0.60). The laser beam reaches the target surface on the lunar soil equipped with multiple receivers that convert it back to electric power. The transmission is possible when the elevation angle is $\alpha \geq 30°$ and satellite is not in its eclipse zone. The receiver of this case study is located at the same latitude and longitude of the *Artemis III* landing site $[-89.54°, 0°]$.

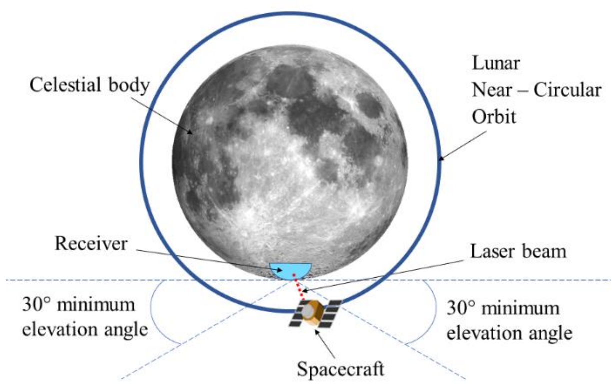

Fig. 1 Laser Power Beaming Architecture Concept

3.2.1.1 Transmitter

When it comes to having high power and a good ability to focus the beam at long distances, only certain types of lasers can be used, such as *diode lasers* and *fiber lasers*. The recent decade has seen the rapid development of compact high-energy lasers, and the increasing selection of these COTS devices will benefit the development of a wireless power beaming system.

Since the laser must be capable of supplying a high amount of average power, on the order of 10s of kilowatts, either continuous wave (CW) or rapidly pulsed lasers (QCW) must be used. Diode lasers and

fiber lasers are both able to accommodate these two different operative modes.
Quasi-CW operation devices are sometimes even designed specifically for pulsed operation: they are designed for a smaller heat load and different emitters can be packed closer together, in order to obtain a higher brightness and beam quality. QCW operation allows a relatively high output power during limited time intervals, thus strongly reducing the heating and related thermal effects.

COTS lasers usually have a very small aperture (in the order of *mm*) and relatively high divergence (< 8 *mrad*). The laser beam then needs to be shaped and collimated to reach distances of hundreds of kilometers with good beam quality.

The power at distance *z* from the laser, passing through a circle of radius *r* is given by:

$$P(r,z) = P_0\left(1 - e^{-\frac{2r^2}{w^2(z)}}\right) \qquad (1)$$

Where

$$P_0 = \frac{1}{2}\pi I_0 w_0^2$$

is the total power transmitted by the beam. In this sense, the beam spot sizes used in the calculations refer to the $1/e^2$ radius.

3.2.1.2 Beaming and Eclipse Time analysis

An STK analysis on a 1-year case study was conducted to quantify the time when the satellite is in eclipse form the moon or the earth, and simultaneously the laser is ON.
In calculating the total energy sent to the lunar surface, it was decided to not power the laser during eclipses, in order to not fall back on oversizing the batteries. In figures 2a and 2b are shown the trends in two different periods of the mission time of a satellite at two different altitudes.
In the first case we can see that there is no overlap between the time in eclipse and the time when the laser is ON, while in the second it's evident that an overlap is possible.

3.2.1.3 Receiver

Two different receivers have been considered for converting the laser beam into electrical energy: *rectennas,* which are devices containing a rectifier circuit and an optical antenna, have been researched for decades and have shown theoretical efficiencies of 80-100%. However, rectennas are still only in the research phase with a low TRL, and are not yet ready to be mass-produced for such an application. Another option for a receiver is photovoltaic cells specially designed for the laser's monochromatic light, also called *monochromatic photonic power converters* (PPCs). Photovoltaic conversion efficiency for solar cells is fundamentally limited by transmission and thermalization losses, because they operate under the broad-band solar spectrum. For monochromatic light, these losses can be minimized by matching the photon energy and the absorber material's bandgap energy. Unprecedented photovoltaic conversion efficiency of 68.9% was recently obtained by researchers at Fraunhofer ISE using GaAs PPCs for 858 nm monochromatic light.

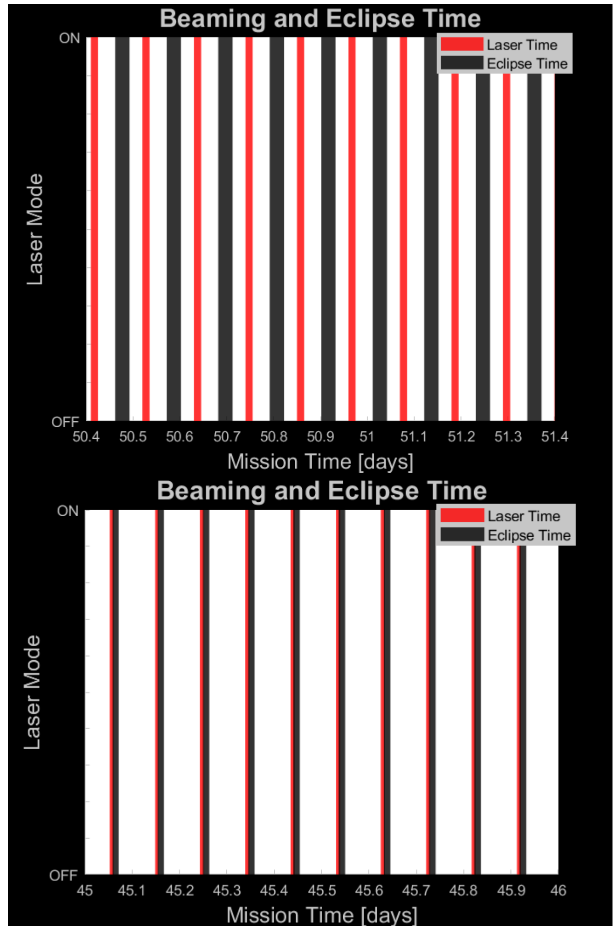

Fig. 2a, 2b Beaming and Eclipse times at different

In this initial study, we will then consider the receivers as photovoltaic panels for monochromatic light, converting 858 nm monochromatic light, and transmitting 10 to 20 kW of power. The geometry of the PV cell arrays doesn't have to be traditionally flat, as we see in traditional solar panels on earth or in space. A flat surface with raised edges is proposed. Based on cost and manufacturing challenges, the edges can either be round or pyramidal. The rounded shape, after a process of surface optimization, leads to the most efficient geometry design for the receivers, allowing them to capture the most energy from the laser beam throughout the power link between satellite and station.

One of the main requirements for the design of transmission and reception is the *irradiance* (or brightness) I of the beam reaching the PV cells. PV cells usually have an ideal value for which they work at peak efficiency. The PPCs considered show 68.9% efficiency for $I_{ref} = 11.4\ \text{W/cm}^2$, which will be considered as a reference value of irradiance in this paper.

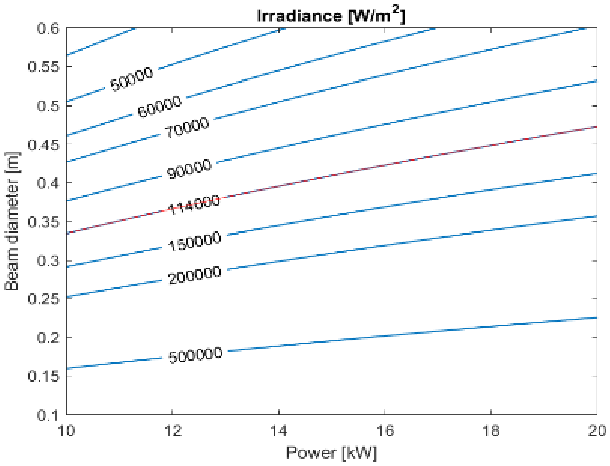

Fig. 3 Irradiance vs beam diameter and laser output power

In fig. 3 is the irradiance I for various combinations of laser transmitting power and beam diameter (at the receiver), calculated with equation (2)

$$I(r,z) = I_0 \left(\frac{w_0}{w(z)}\right)^2 e^{-\frac{2r^2}{w^2(z)}} \tag{2}$$

The beam's radius at a distance z from the *beam waist* w_0 is calculated with the following equation:

$$w(z) = w_0\sqrt{1 + \left(\frac{z\lambda}{\pi w_0^2}\right)^2} \tag{3}$$

The graph shows that to maintain I around the reference value, the laser beam at the lunar surface hitting the

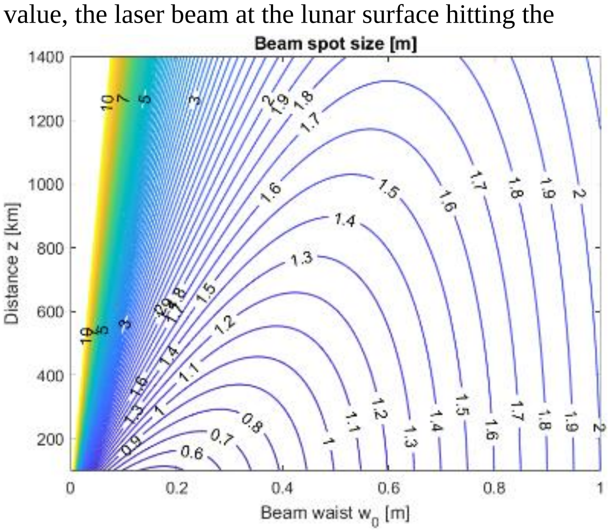

Fig. 4 Irradiance vs beam diameter and laser output power

receivers must have sizes between 33 – 47 cm, depending on power P which in this study has been arbitrarily limited to 20 kW. The beam spot size is conventionally measured at a radius for which the intensity has dropped to $1/e^2$ of its maximum value. This means that around the beam, 13.5% of power is still being transmitted. This will be one of the margins to take account of when designing the receiver surface, with a ~50% radius augmentation. At this stage of design, a lower irradiance must be considered, since much higher power is needed to achieve I_{ref} at the receiver. This is not necessarily limiting, since the PPCs considered are only one existing prototype of the technology.
With the values estimated in paragraphs *3.2.1.1*, *3.2.1.2*, and *3.2.1.3*, we can now estimate the amount of energy transmitted in one year from 5 ORiS satellites to a receiver on the lunar surface. In Table 1 is shown the amount of energy in function of the altitude chosen:

Table 1. Energy transmitted from 5 satellites in one year

Altitude [km]	Energy transmitted [MWh]
300	15.0
400	26.1
500	23.6
600	37.3
700	35.3

3.2.3 Receiver dimensions

The receiver surface area must be optimized for the laser beam, to minimize weight and costs and obtain the best scenario for system efficiency. Considering the case of 12kW (the system can be scaled for various values) of output power from the laser and 60° minimum incidence angle, the beam spot diameter has certain maximum and minimum values, given by certain requirements. The highest orbit considered is 700 km, so the maximum absolute distance will be around 1400km.

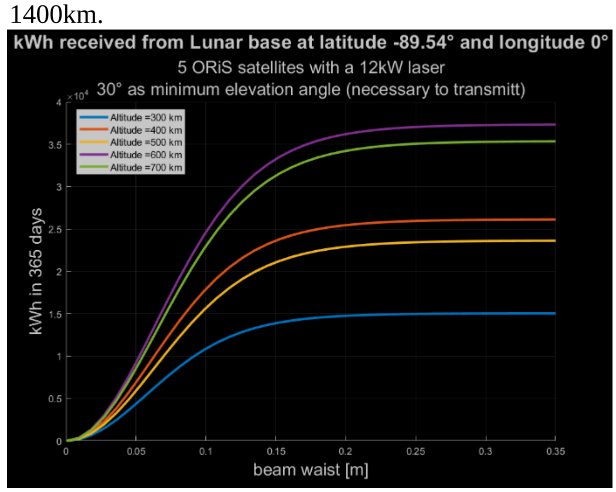

Fig. 6 Beam diameter in function of beam waist w_0 and distance z from w_0

As can be seen from fig. 6, for each orbit there is a lower limit to w_0 for the beam diameter to be kept under $3\ m$ at the receiver. This value is not an absolute limit but is taken considering that after the 3 m curve receiver dimensions start to increase very rapidly, increasing cost, weight and complexity of the system. It should be kept in mind, in fact, that these structures should be easily transported and installed on the lunar surface, possibly in places that have not yet been explored. From fig. 5 we can also find what the beam's maximum diameter will be at the receiver, for transmission from each altitude. The results are reported in Table 2.

Table 2. Minimum laser beam waist w_0 and receiver diameter, for every altitude

Altitude [km]	300	400	500	600	700
$w_{0,min}$ [cm]	11	15	19	22	26
D_{rec} [m]	1.17	1.44	1.73	2.03	2.33

Adding the previously mentioned 50% margin to D_{rec}, the estimated receiver dimensions which allows transmission from the highest altitude (worst-case) is:

$$\begin{cases} D_{rec} = 3.5\ m \\ A_{rec} = 9.6\ m^2 \end{cases}$$

These dimensions do not take into consideration margins to be added due to pointing errors and they consider a flat receiver with no edges.
W_0 is directly correlated to the quality of the laser and collimating lens, and from its value depend basically all other transmission parameters. Clearly, only one laser will be used to transmit from all distances, which will have a characteristic beam waist. For calculations, a beam waist greater than all minimum values will be used.

3.2.3 Pointing accuracy

In order to determine and control the spacecraft attitude, to stabilise the spacecraft and to orient it in the desired directions during the missions, considering the importance of the required re-pointing, the complex mission goals and the inertial characteristics of the spacecraft, a 3-axis stabilisation technique has been chosen for the ORiS satellites. The ADCS consists of:

- Sun sensor;
- Star sensor;
- Solar cells;
- Reaction wheels;
- Attitude thrusters;

and could achieve an accuracy of 0.36 arcsec. To have such fine accuracy, it is necessary to define and to configure a more sofisticated and complex ADCS system. On one hand this means higher costs, but on the other, this angle is useful in order to minimize receiver surface and thus minimize costs from another point of view. This structure was determined by a careful analysis of costs and benefits in their totality.
The radius of the receiver increases as shown in Table 3.

Table 3. Receiver radius increment vs orbit altitude

Altitude [km]	300	400	500	600	700
r_+ [cm]	52	70	87	105	122

3.2.4 EPS weight

The work carried out is useful to sizing the solar panels and batteries on board the ORiS satellites. To do this, the power requirements of all the satellite subsystems are identified and divided into two categories: power required by the laser and power required by all the remaining subsystems of the satellite.
To size the solar panels, the power required for the satellite is defined, considering an efficiency of the solar cells equal to 34%, through the equation (4).

$$P_{SA} = \frac{\left(\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}\right)}{T_d} \tag{4}$$

All lengths of time in which the ORiS satellites are in daylight or eclipse were calculated. This was necessary to take account of differents operation modes: in daylight, both categories will be powered, while only the subsystems in eclipse. Terms X_e and X_d represent the efficiency of the paths from the solar panels through the batteries to the individual loads and from the arrays to the loads, respectively. During operative life, a degradation factor of 3.75%/yr was considered, and sizing was done on EOL power, thus at the end of the 5-year expected operative life the EPS will still have an adequate performance. The surface area of the solar panel is then calculated as P_{SA}/P_{EOL}. To estimate its mass, a power to mass ratio of $165\ W/kg$ was considered. P_{SA} divided by this ratio will give an estimate of the solar panel's mass.
The strategy used to optimize the EPS mass uses as main parameter the power percentage given by the solar panels. This percentage is calculated in the case of the satellite's maximum energy consumption, as the portion of power the panels are able to supply in relation to the rest coming from the spacecraft's batteries. This strategy had a positive outcome, in the sense that the condition of maximum energy consumption occurs during only a small fraction of the orbit, namely during laser operation. For the remaining time, the power demand is much smaller and well within the capabilities

of the solar arrays, which in this case can power all subsystems and charge the batteries.
The specific energy parameter is a measure of how much energy a battery contains in comparison to its weight. The higher the battery's specific energy, the lower the EPS mass will be. If this value isn't high enough, supplying energy through batteries would weigh more with respect to generating the same energy only through a solar array.
The DOD of the battery is taken as 25% to preserve its operational lifetime. The operative modes considered when sizing the battery are eclipses and laser operation, with their respective power demands. The mass of the batteries was estimated for each altitude as shown in Table 4 considering the number of cells, power generated by each cell, and the aforementioned specific energy. A model of the battery was created using the Simulink software, to control the duration of charge and discharge cycles throughout an orbit.

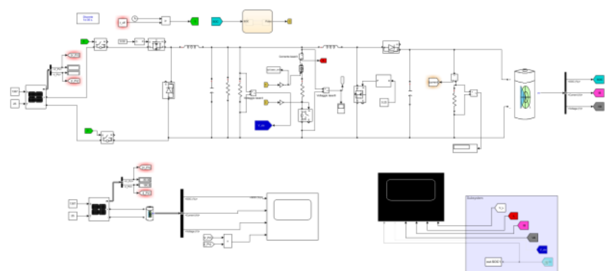

Fig. 7 Simulink battery model

According to the series and parallels configuration of its cells, the output voltage from the solar panel is 30V. The loads consisting of satellite subsystems and laser are powered with 28V, so a buck converter is needed for proper operation. The battery is charged at a constant voltage of 30V. The model takes as inputs the times when the satellite is in daylight or eclipse, and the times when it is inside the *transmission cone* of the receiver on the lunar surface. In this manner, every scenario for the satellite can be taken account of: battery supply, battery + solar panel supply, solar panel supply.
Following simulations with this model, the optimization strategy used can be said to have given a positive outcome, in that the solar panels are able to charge the batteries every time the satellite is in a position to transmit power to the user. Charging times and EPS mass are taken into consideration together with the remaining figures of merit for the optimization of the mission architecture as a whole.

Table 4. Charging times and EPS mass vs orbit altitude

Altitude [km]	Capacity [Ah]	Percentage solar panel – battery [%]	Time for recharge DoD [min]	Panel + battery mass [kg]
300	543	20-80	48	119
400	753	20-80	66	151
500	263	80-20	8	166
600	320	80-20	10	175
700	385	80-20	11	185

3.3 Optimal Receiver Shape
A flat receiver has an efficiency due to cosine loss of 82.7% (considering a minimum elevation angle of 30°). A hemispherical surface (fig. 8) was considered to minimize transmission cosine losses.
In this way we increase the energy transmitted on lunar surface of about 14%.

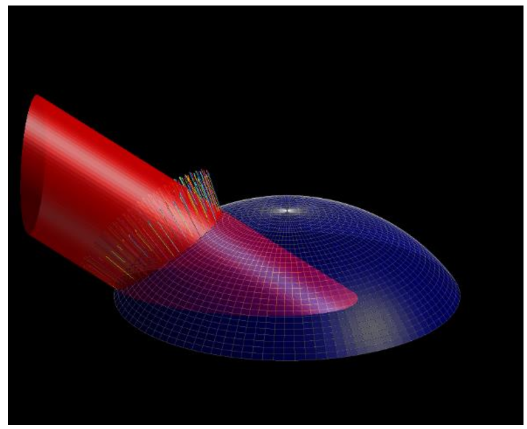

Fig. 8 Optimized spherical shell geometry

The geometry was optimized for each altitude and for a a minimum elevation angle of 30°. If this angle can be lowered to 15° it's possible to increase the energy transmitted to the lunar soil of 30-40%; at the same time the shape will be different: the structure will be higher, the diameter and total area will increase.

4. Results and Discussion
For each case we define a value (from 1 to 10) which will determine the final score in function of the weights of the Figure of Merit considered (from 1 to 5):
Energy in one year: 5/5
Receiver dimensions: 3/5
Pointing accuracy: 4/5
EPS weight: 3/5

Table 5. Altitude trade-off study

Altitude	Energy in one year	Receiver dimensions	Pointing accuracy	EPS weight	*Score*
300	3	9	9	9	105
400	6	8	8	7	107
500	5	8	7	6	95
600	**9**	**7**	**6**	**6**	**108**
700	8	7	5	5	96

After a trade-off study based on some parameters shown in the previous table, it is possible to say that the altitude of 600 km is the best choice.

Power-supply requirements for a lunar base are to be estimated. Requirements given by NASA in Ref. [38] for a future wireless laser power transmission system are 5kW for night operation, so conservatively 12h per day as opposed to the 40kW of the lunar habitat's daytime consumption. Given these values, 21900 kWh are needed in a year. Following the optimization process presented in this paper, an appropriate combination of laser power output and number of satellites in the constellation can be obtained.
For example, to power the lunar habitat in daytime as well, 15 satellites of the ORiS constellation would be needed, with laser peak power outputs of 21kW.

In conclusion, 5 satellites of the ORiS constellation, placed at 600 km of altitude shown in figures 9 and 10, can provide 37304 kWh to a Moon base placed at lunar south pole. The satellites are equipped with a 12-kW laser: the weight of the laser diode module, DC to DC

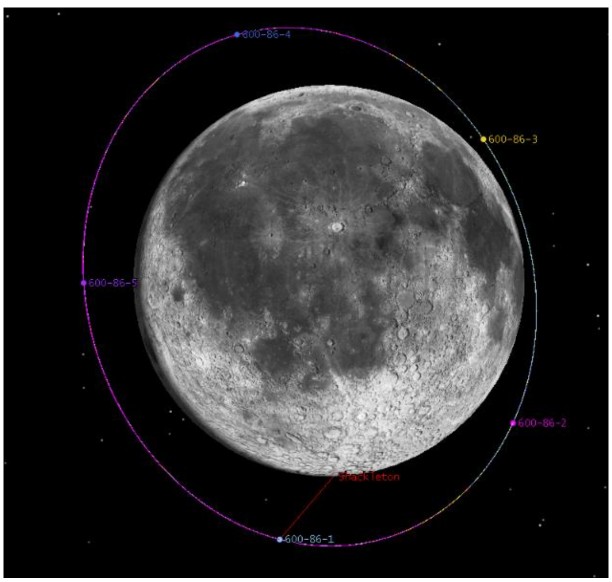

Fig. 9 Layout of ORiS Constellation at 600km altitude

converter unit and structures for integration with the satellite is about 240 kg. The weight of the solar panels and batteries is about 175 kg. The surface of the solar panels, considering an operational life of 5 years, is 70 m^2. It is possible to estimate the weight of the entire satellite to around 700-800 kg.

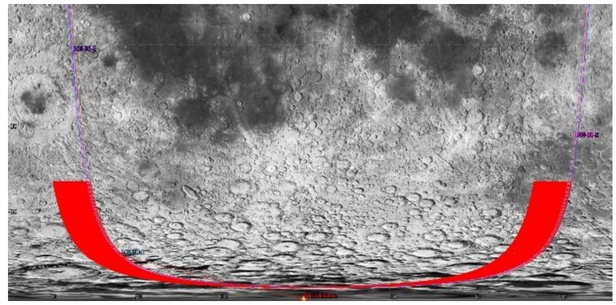

Fig. 10 Lunar ground track

A spherical shell with a maximum radius of 3.6 m and a height of 2.72 m, as shown in figure 11, minimizes cosine loss ($\eta = 0.98$) in the 600 km – altitude case. If a minimum elevation angle of 15° is considered and the flat receiver is replaced with a spherical shell, the system can provide about 50 MWh in one year. Therefore, each satellite in lunar orbit can potentially deliver about 10 MWh to the lunar poles from an altitude of 600 km and it is possible to calibrate the number of satellites according to the energy requirement.

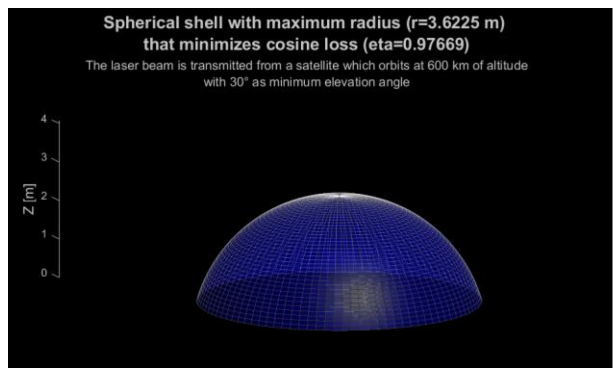

Fig. 11 Optimal receiver shape – 600 km case

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