Conceptual Study of Transport System between Deep Space Gateway and Moon for Lunar exploration

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***Abstract*— This document is the final report for the Deep Space Gateway Lunar Transport system. The document will include the structure, trajectory, and technology of a single Transport mission between the DSG and the Lunar surface.**

***Index Terms*— Spaceflight, Rocket, Transport, Trajectory, Life Support, Human-rated, Environment, Cost, Communication, Power, Aerospace.**

I. INTRODUCTION

This report provides information about a conceptually study on a transport mission from the deep space gateway to the lunar surface. The design and construction of a deep space gateway is seen as the next goal for international human space exploration. It is planned to be used for explorations on the lunar surface but also as a base for future Mars missions. Next to the deep space gateway this requires a transport system to finally reach the lunar surface. Consequently this report deals with the conceptual design of a transport system which covers a proper trajectory analysis and the design of the mission and the vehicle. More over the power and mass budget, as well as the problem of communication coverage and the providing of a life support system is discussed within the scope of this report.

II. TRAJECTORY ANALYSES

After the final decision of the Near Rectilinear Halo Orbit 9:2, numerous simulations (apart from the simplest analysis with the Konstantn Tsiolkovskis equation,

$$\Delta V = I_{sp} g_0 \log \frac{m_S + m_P + m_L}{m_S + m_L} \tag{1}$$

have been carried out in order to obtain an accurate estimation of the amount of fuel needed to perform the transfer from the Deep Space Gateway to the Moon, and vice versa. We have checked that these values agree with the initial estimations with the Tsiolkovskis equation. The most efficient strategy with this elliptic orbit turns out to be reaching the DSG always at the perilune (except off-nominal cases that we will analyse at the Design Mission section IV) since it is the closest point to the Moon and it will take less travel time.

To simplify the analysis, some assumptions have been made.

- The undocking and separation of the Space Station and the Vessel will be accomplish before the DSG achieves the perilune (closest point to the Moon).
- At the perilune, the vessel will have got the correct and appropriate attitude to start the manoeuvre of descent. All the required modifications will be performed then a bit earlier in time.
- At the launch site, on ground, the vessel will have got the correct and appropriate attitude to start the manoeuvre of ascent. All the required modification will be perform then a bit earlier in time.
- The first analysis will not include change of orbital plane. These calculations will be studied after this analysis.
- For the descent, the initial point has the following features:
  - The speed is the orbital speed of the DSG at the perilune, i.e. 1700.4 m/s.
  - The altitude (with respect to the Moons surface) is 1496 km.

The equation of launcher dynamics (shown below) have been used during the whole simulation with Matlab.

$$\dot{V} = \frac{T}{m} - \left(g - \frac{\dot{X}^2}{R_m + H}\right) \sin\gamma \tag{2}$$

$$\dot{\gamma} = -\frac{1}{V} \left(g - \frac{\dot{X}^2}{R_m + H}\right) \cos\gamma \tag{3}$$

$$\dot{X} = V \cos\gamma \tag{4}$$

$$\dot{Y} = V \sin\gamma \tag{5}$$

$$\dot{m} = -\beta(t) \tag{6}$$

$$\tag{7}$$

The analysis has been split in two parts: The descent and the ascent. In order to minimize the amount of propellant that the vessel will use (and hence, carry with), the differential equations have been solved backwards, i.e. starting with the ascent, on ground, we have calculated the minimal propellant mass that the vessel needs to be able to lift-off and achieve the NRHO 9:2 successfully. Once the amount of fuel for the ascent is known, this mass will be taken into account for the descent part since the fuel for the two segments have to be carried on the vessel from the DSG (there is no possibility, so far, of getting propellant on the Moons surface).

A similar backwards strategy will be performed in this report in order to facilitate the process followed with the simulations.

A. Ascent

For the ascent phase, the vessel will be carrying an estimated structural mass of 5000 kg plus a payload mass of 2000 kg (notice that part of the payload has already been deposited on Moons surface, specifically 3000 kg). These 2000 kg payload will include the crew members weight, their belongings (space suits, remaining food, supplies, etc.) as well as eventual Moons rocks or samples needed for the research itself in the DSG or on Earth.

With this total mass of 7000 kg, an analysis with Matlab states that a 4500 kg of propellant is required for the ascent phase (in comparison with the 4486 kg result from Tsiolkovskis equation, taking a ΔV of 1700.4 m/s). The analysis was performed backward, following the same strategy as before.

The initial condition of the ascent phase are:

- Speed close to 0. Speed different to zero is used in order to avoid singularity of the launcher dynamic equations.
- Altitude equals to 0 km (representing Moons surface, on ground).
- Pitch angle (between vertical axis of the vessel and the local horizon) close to 90 degrees, i.e. vertical position.

The ascent phase is over and the vessel is considered to be in orbit with the DSG when it achieves next state:

- Same speed as the DSG at the perilune. V = 1700.4 m/s
- Same altitude as the DSG at the perilune. H = 1496 km.
- Pitch angle close to 0 degrees, i.e. horizontal position.

The ascent part will take around 1 h and 5 min. Gravity turn is not enough to get a good trajectory and hence a small thruster is required to control the attitude (specifically the pitch angle γ, the inclination angle of the spacecraft with respect to the local horizon).

Only a first burn (of 225 s) is required to lift-off and achieve the NRHO orbit again. After this burn (altitude of 45 km and speed of 1585 m/s), the vessel will stay ascending (for 3689 s). As we mentioned before, a system controlling the attitude is necessary to achieve the perilune with the correct parameters.

More details about this phase can be seen on the Launch profile figure 8 in Appendix.

Taking into account that the mass flow rate of the propulsion system is 20 kg/s it is easy to derive the amount of fuel used for any phase of the trajectory.

As it can be seen in Figure 8e. Vehicle angle with horizon, the thrusters have to be used after around 1000 s to correct the pitch angle γ and achieve an inclination of 0 when the vessel is close to the DSG.

B. Descent

For the descent phase, the vessel will be carrying an estimated structural mass of 5000 kg plus a payload mass of 5000 kg plus a propellant mass of 4500 kg for the ascent phase. This 5000 kg payload will include crew members, their belongings as well as materials, tools and equipment needed for the lunar exploration.

With a total mass of 14500 kg, an analysis with Matlab states that roughly 10 000 kg of propellant is required for the descent phase (in comparison with the 9263 kg result from Tsiolkovskis equation, taking a ΔV of 1700.4 m/s). The analysis was performed backward, following the same strategy as before.

The initial condition of the descent phase are:

- Same speed as the DSG at the perilune. V = 1700.4 m/s
- Same altitude as the DSG at the perilune. H = 1496 km.
- Pitch angle close to 0 degrees, i.e. horizontal position.

The descent phase is over and the vessel is considered to be on ground when it achieves next state:

- Speed close to 0, to avoid extreme structural forces or stresses that can damage the vessel and its landing gear during the touch-down.
- Altitude equals to 0 km (representing Moons surface, on ground).
- Pitch angle close to 90 degrees, i.e. vertical position.

The descent part will take around 2 h and 25 min. Gravity turn is not enough to get a good trajectory and hence a small thruster is required to control the attitude (specifically the pitch angle γ, the inclination angle of the spacecraft with respect to the local horizon).

At the beginning of the descent phase (close to perilune), thrusters are used to change the inclination angle and orient the vessels speed towards the lunar surface, starting a free falling (a system controlling the attitude is necessary to leave the orbit). It will stay descending (for 8250 s). Once we are close to the surface (around 85 km, speed of 1573 m/s), there is the main burn to slow down the vessel and perform a soft touch-down.

More details about this phase can be seen on the Launch profile figure 9 in Appendix.

Taking into account that the mass flow rate of the propulsion system is 20 kg/s it is easy to derive the amount of fuel used for any phase of the trajectory.

As before, it can be seen in Figure 9e. Vehicle angle with horizon, that the thrusters have to be used the first 4000 s to correct the pitch angle γ and start the descent phase with an inclination of 180.

C. *Mass budget*

A mass diagram is shown below to ease the understanding of the vessel and the heaviest parts of it.

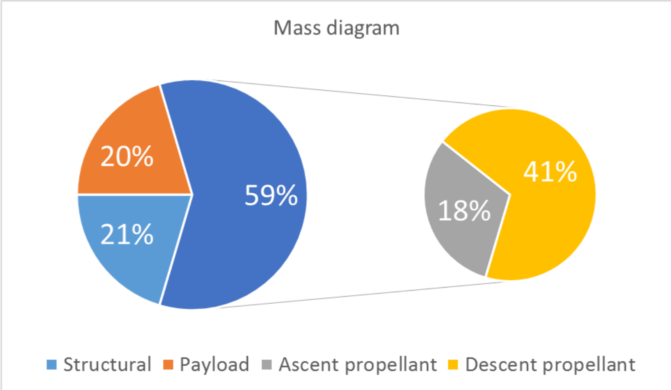

Fig. 1: Mass Diagram

III. Targets on Lunar surface

Since the interest of this mission is to research and explore as much as possible the Earths satellite, our investigation team have planned to go to different places on the Moons surface.

Some of these points of interest are quite close to the orbital plane, i.e. longitudes close to 90 E and 90 W (remember that the orbit is facing the Earth (orbital normal vector and vector pointing from the Moon to the Earth are coincident) and roughly polar, inclination of 90) and hence there will be no problem at all to achieve them because the vessel will stay orbiting the Moon until the desired latitude to land.

Nevertheless, for some points, whose longitudes, close to 0, are way far from the orbital plane, the vessel will need to perform an inclination change of the orbit to achieve them. For those cases, there are two possibilities:

- Either the inclination change is accomplished at the apolune (the farthest point of the orbit) where the amount of fuel required is very low (inclination change of 45, ΔV = 63.68 m/s

and hence 459 kg of fuel, for instance). In this case, the crew will have to travel inside the vessel for half a period of the orbit (around 3.35 days).
- Or the inclination change is accomplished at the perilune (the closest point of the orbit) where the amount of fuel required is extremely high (inclination change of 45, ΔV = 1301.4 m/s and hence 11291 kg of fuel, for instance) because of the high velocity at that point. In this case, the vessel will stay attached to the DSG until the undocking, close to the perilune.

The second option is not doable according to the design of our spacecraft since there is no enough volume and weight capacity to contain that excessive amount of fuel. Consequently, as a preliminary strategy, we will consider the first option to accomplish those missions. However, there are as well different orbits possibilities which start changing the inclination angle at points between the apogee and perigee but that analysis would be going beyond the project scope.

IV. Design Mission

Once we know good estimations of ascent and descent time as well as the mass budget, we can start designing our vessel. We can affirm again that it does not make sense trying to catch the DSG at a point which is not the perilune since it will take much more time (and more propellant mass in the vast majority of cases). Consequently, the journey on the lunar surface must last, at least, one period of the orbit (exactly 6.7 days).

But what happens, for example, when the crew detects a malfunction on the vessel and they cannot lift-off to reach the DSG at the perilune on time. There are two possibilities:

- Either we try to catch it in a close point to the perilune with a bigger amount of propellant. As we said, it is not the most efficient solution at all but it will be still possible because our vessel is lightly over-sized and there is around one ton of extra propellant. Nevertheless, this possibility is still doable when the repair of the malfunction did not take too long and the DSG is still close to the Moon (it has been estimated less than 4 hours after the perilune passage).
- Either we stay another period on the Moon, we fix and repair the required systems and we catch the DSG at the next perilune passage. We considered that this strategy can be the most realistic solution since the crew will not be in a hurry to fix everything in a short time as well as the least propellant consuming. This is the reason why we will size our system to be fully DSG-independent during two weeks: one week for the nominal journey on the Moon and one week of contingencies.

V. Transport Design

A. Introduction

This section will contain the size and volume of each component of the transport design system. Components in the subsections will be ordered from top to bottom based on the landing configuration on the lunar surface.

B. Docking Adapter

The transport vehicle uses the International Docking Adapter which has a diameter of 800 millimeters. [22] The adapter is located at the top of the transport system to allow for more space on the side walls at landing. It contains a retractable hatch that folds outwards in three parts. The hatch protects the adapter from accumulating moon regolith which would be harmful to the Deep Space Gateway.

C. Pressurized Cabin

The transport system's structure is hexagonal to allow for an airlock, two inflatable modules, and two solar panel arrays. The system is symmetrical with the airlock opposite to a window for viewing. The cabin is $2.5m$ tall and each wall is $1.5m$. The total volume of the cabin is $14.62m^3$ but after including life support systems, payload, and the crew, the pressurized cabin space is $8m^3$ of space during the ascent and descent phases. When the transport system has landed, the inflatable modules can expand linearly outward from the wall. The dimensions of the modules are $1.5m$ wide, $1.5m$ long, and $2m$ tall. With both inflatable modules pressurized, the total pressurized volume is $17m^3$.

D. Propulsion System

There are 16 reaction control system (RCS) thrusters placed in groups of four spaced 90° around the transport system, just under the pressurized cabin. The RCS thrusters will provide thrust to orientate the transport system during lunar landing and DSG docking. More information about the propulsion system can be found in Section VI.

Fuel storage is located between the RCS thrusters and the main thrusters. The available space is 7.07m^2 area circle that is 3.5m tall, resulting in a total volume space of 24.75m^3. The required fuel volume for the entire mission is 20.0m^3, leaving contingency room for the tank shapes and possible room for unpressurized storage.

Below the fuel storage are the four main thrusters for the ascent and descent stages. The thruster nozzles are one meter off the ground when the landing legs have fully deployed to allow for uneven lunar surface and minimal risk to the engines performance.

E. Landing System

The landing system uses four hydraulic legs to land on the lunar surface without conflict. The landing legs connect to the transport system at the fuel storage. The legs fully extend at a 20° angle with a length of 3 meters. During the landing the legs will compress accordingly to keep the transport system stable.

VI. Propulsion System

A. Main Thrusters

The main thruster system contains four thrusters that use methane, CH_4, as a fuel and liquid Oxygen, LOx, as the oxidizer. Each thruster produces 22.24kN of thrust for a total of 88.96kN. The propulsion system can complete the ascent and descent stages of each mission using only three engines. Using four engines provides a more reliable safety system for the crew. The added risk of using four engines is smaller than the benefit of saving the crew if one thruster fails. This engine has been tested and is rated PRL-9 on the propellant readiness level. The Isp is 363s.[23]

B. RCS Thrusters

The RCS thrusters system contains 16 thrusters that will use the same fuels as the main thrusters: CH_4 and LOx. These thrusters have 450N of force per engine with an Isp of 303s. The force provided by these engines is sufficient to perform landing and docking maneuvers. The total fuel needed for each mission is 500 kg for all thrusters.

C. Fuel

To calculate the fuel needed, the mass of the fuel, 14500kg, was broken into the masses of CH_4 and LOx. I used stoichiometry to solve for the fractions of each chemical was needed.

$$CH_4 + 2O_2 = CO_2 + 2H_2O$$

From above we can say that the reaction needs 1 part CH_4 and 2 parts LOx. We can take the molar mass of each chemical, 32.02 g/mol for LOx and 20.05 g/mol for CH_4. Solving for the masses of each chemical using the above total mass we get the total mass 8917kg for LOx and 5583kg for CH_4. Using the densities of each chemical at their given temperatures we find the total volume of each chemical to be 8.5m^3 for LOx and 11.5m^3 for CH_4.

VII. Power Budget

A. Introduction

This chapter deals with the power supply for any system of the transport vehicle relying on electricity like for example sensors, active heating and cooling, telemetry, control and life support systems. For the reason that the spacecraft is operating in the inner solar system the best source of energy is sunlight to generate electricity with photovoltaic solar panels.

B. Solar Cells

Solar arrays used on spacecraft are made of thousands of solar cell to generate the required energy [2]. Therefore gallium arsenide-based solar cells are typically favoured over crystalline silicon in industry because they have a higher efficiency and a slower degradation. [3] Fabricated single-junction GaAs thin-film crystal solar cells reached an efficiency of 28.8 % [4]. The efficiency can be greatly improved by multi-junction photovoltaic cells which combine several layers of gallium arsenide, indium gallium phosphide, and germanium to match the solar spectrum, not only including visible light but also energy from IR- and UV-spectrum. [3] The multi-junction GaAs cells are capable of exceeding an efficiency of 38.8 % [4]. For the best research-cell efficiencies please see the results of the National Renewable Energy Laboratory (NREL) in Appendix IV

C. Solar Array Design

The travel time from the DSG to the lunar surface is with approximately two hours comparatively short. Due to the higher loads the solar arrays should not be deployed during docking at the DSG and landing on the moon. Even though, traveling with deployed solar panel is possible due to no atmosphere the space environment poses a risk of damaging the solar array. Therefore it is advantageous to have a back-up battery system available covering the whole time of travel from the DSG to the moon and back. Consequently, the solar array is only deployed after landing on the lunar surface or after docking to the DSG to recharge the batteries. This requires a deployable solar array design and lead to the space light proven UltraFlex System from Orbital ATK.

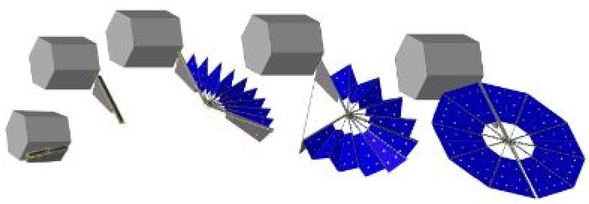

Fig. 2: MegaFlex System [5]

This system is very lightweight (150 W/kg at the BOL) has a high strength and a compact stowage volume. [5] Furthermore the solar panels always need to face the sun to provide maximum power. Therefore the array need to be rotatable and requires a sun-tracking system and the installation on gimbals.[2]

D. Energy Storage

To store the generated energy batteries are used which are distinguished between primary and secondary batteries. The current state-of-art primary and rechargeable batteries are listed in Figure 3.

Hereby primary batteries are non-rechargeable, single discharge batteries and for our application Lithium primary batteries will be used as the emergency destruct battery [8] for the transport vehicle in case of a malfunction of the rechargeable batteries or other failures. However the research focus was on the secondary batteries which are recharged with the generated solar power during docking at the DSG and the stay on the moon. The stored energy can then be used for the flight to the lunar surface or the fly back to the DSG. Lithium-Ion batteries are the state-art-technology in space flight applications especially due to high energy density and low self-discharge [9]. The specific energy of lithium-ion batteries is in the range of 100 to 265 Wh/kg [11]. Lithium-Sulphur batteries seem to be the next generation of battery technology as they already achieved 400 Wh/kg at cell level. As Lithium-Sulphur batteries are not yet proven for space applications, for now there was a space proven Lithium-ion battery from ABSL (ABSL 8s104p 28V 156Ah [10] chosen.

E. Dimensioning of Battery

Given the data of the selected type of battery the required number of batteries was calculated based on the assumed total power consumption of 3.01 kW including life support and control systems. For the battery dimensioning the worst case scenario of undeployable or damaged solar panel was considered. This resulted in a maximum battery use of 12 hours (2 hours to the moon and 10 hours back to the DSG). With these numbers and a storage capacity of 4.492 kWh per battery a total number of nine batteries is calculated. Finally this leads to a total battery mass of 449 kg and 0.34 m^3 volume. The detailed steps of the calculation can be seen in Appendix V.

F. Dimensioning of Solar Array

For the calculation of the solar panel size the solar constant, a flux a flux density measuring the mean solar electromagnetic radiation per unit area. The solar constant includes all types of solar radiation and has a value of 1.37 kW/ m^2 on Earth [12]. But as this value is measured outside the Earths atmosphere and the moon is orbiting this solar

SOP System	Technology	Mission	Specific Energy, (Wh/kg)	Energy Density, (Wh/l)	Operating Temp. Range, (°C)	Cycle Life	Mission Life (yrs)	Issues
Primary Batteries	Ag-Zn, $Li\text{-}SO_2$, $Li\text{-}SOCl_2$	Delta Launch Vehicles, Cassini Probe, MER Lander, Sojourner Rover	90-250	130-500	-20 to 60	1	1-9	• Limited operating temp range, • Voltage delay
Rechargeable Batteries	Ni-Cd, $Ni\text{-}H_2$	TOPEX, HST, Space Station	24-35	10-80	-5 to 30	> 50,000, @25% DOD	>10	• Heavy and bulky, • Limited operating temp range
Advanced Rechargeable Batteries	Li-Ion	MER: Spirit & Opportunity Rovers	100	250	-20-30	> 400 @50% DOD	>2	Cycle Life

Fig. 3: Comparison Battery Systems [7]

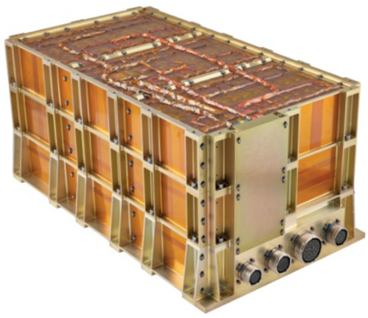

Fig. 4: Lithium-Ion Battery from ABSL [10]

constant can be used as an average for the solar panel design. In contrast to the Earth the Moon has the advantage of not having weather conditions and thereby profiting of a nearly constant value. Given the solar constant and the EOL efficiency factor of 25.8 % [6] of the Gallium arsenide-based multi-junction solar cells this results in a maximum output of 353.5 W/ m^2. Knowing the total power consumption by the life support and control systems and the mission duration the required power which need to be generated from the solar array are 462.1 kWh. This results in a total solar array area of 8.1 m^2. For the reason that two circular solar arrays of the above mentioned system are used the diameter per array is 2.27 m with a mass of 14.5 kg. The detailed steps of the calculation can be seen in Appendix V.

G. *Off-nominal cases*

Although calculation for the dimensioning of the batteries and the solar arrays has been conducted it need to be particularly highlighted that especially in space technology high safety factors are considered to enable as high safety standards as possible in the dangerous environment. Consequently, several off-nominal cases need to be considered like for example:

- Damage of solar panel through space environment
- Solar Panels are not deployable on the moon
- Efficiency loss
- Material fatigue
- Malfunction of sun tracking control
- Malfunction of batteries

and the calculations updated in terms of safety factors.

VIII. Communication

A. *Introduction*

As mentioned by the DSG group in their report the communication from the DSG to Earth is fully covered through the choice of the NRH L2 South Orbit. The NRHOs are nearly polar orbits the small inclination of the NRH L2 South Orbit provides the best communication coverage on the moon and shows in comparison to near rectilinear L1 halo orbits an advantage for the communication on the far side of the moon. However, there is only limited communication coverage on the lunar surface and communication may be mission critical without a lunar space network. The NRH L2 South Orbit favours the south pole and the far side, but however only with up to 86% coverage of the south pole. [18]

Consequently, an assessment to identify sites of interest and required coverage is required.

B. *Decision*

Even though the astronauts will not do explorations on the north far side of the moon because

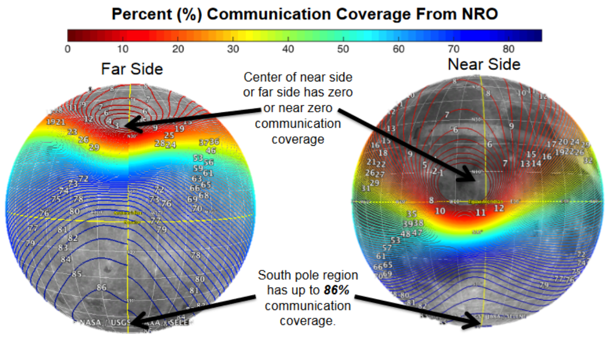

Figure 7. NRO Communication to Surface Coverage

Fig. 5: Communication coverage to moon [18]

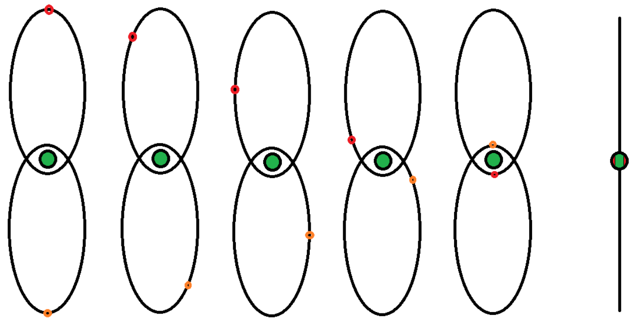

Fig. 6: Communication coverage with satellite

there are no points of interest, the nearly full communication coverage should be enabled at least for the south far side. Even though the NRO L2 South is advantageous for exploration on the far side of the moon the communication coverage of 86 % which corresponds to one day of no coverage was considered as not acceptable, which is why a satellite constellation is planned to enable a higher coverage on the lunar surface. Additionally, in respect to the future development of human space flight and the aspired success of traveling to the mars an International Lunar Network (ILN) would be advantageous for communication and navigation of international missions. An additional satellite in the NRH L2 North Orbit (mirrored DSG-orbit with same parameters) is considered to be the best compromise of a nearly full communication coverage on the lunar surface and costs. As seen in the Figure 6 there is still short-term occultation especially at the moons equator. This problem can be avoided by not landing directly at the equator but by exploring this area of the moon by the rover with communication contact to the transport system.

C. Communication to other systems

Next to the communication via radio bands spacecrafts also use optical navigation techniques and on-board sensors like star trackers and sun sensors for navigation and communication. For example during docking at the DSG navigation sensors are required. The X-band system can provide range measurements beginning at a distance of 400 km but thereafter laser radar rendezvous sensors are used for relative measurements between 500 and 2 m [17]. Optical proximity sensor can be used for determining the relative position and relative attitude between two meters and contact.

D. Antenna Design

For communication the X-band is used as analysed by the DSG group in chapter .. Furthermore a parabolic reflector dish antenna is the chosen antenna type. Together with a horn antenna as a feed which is pointed towards the parabolic reflector this antenna type has advantageous characteristics as for example a high gain and low cross polarization. It can be used for high frequencies both as transmitter and receiver [16].

TABLE I: Link Budget Communication

	Transport → DSG	DSG → Transport
Chosen frequency [GHz]	7.75	8.4
Distance [m]	$7.18 \cdot 10^7$	$7.18 \cdot 10^7$
Temperature receiver [K]	20	373
Data Rate [Mbps]	10	1
Diameter receiving antenna [m]	0.73	0.75
Gain transmitter [dBi]	35.2	35.7
Gain receiver [dBi]	35.0	35.9
EIRP (transmitter) [dBW]	40.8	43.2

With kind support of the DSG group the main antenna parameters where calculated and summarized in Table I. According to the calculations the antenna diameter at the transport system will have a diameter of 0.75 m. The resulting transmitting gain of 35.2 dBi, receiving gain of 35.9 dBi and transmitting EIRP of 40.8 dBW also correspond to the ESA research requirement of at least 35 dBi and 40 dBW [19]. More parameters can be found in Appendix VI. Moreover, next to the high gain antenna reflector and the x-band feed also a sun sensor and a tracker for the satellite and DSG need to be installed as further antenna equipment. Hereby the sun senor is used to make sure that the antenna is never directly facing the sun because otherwise the very high temperatures from the sun will determine the noise temperature of the receiving system [13].

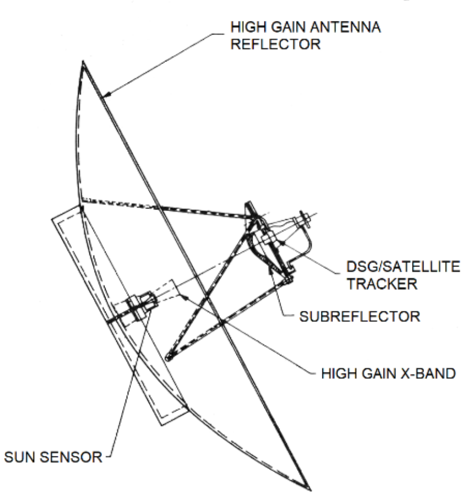

Fig. 7: Antenna Design [15]

IX. Life Spport Systems

A. Introduction

Life support systems (LSS) are essential for human to live in space. It can be said that International Space Station (ISS) is the most successful continuously manned spacecraft in our history, and its LSS has high reliability. Thus, LSS on our transport vehicle are mostly based on LSS on ISS. There are two reasons why we chose to use ISS systems for our transport vehicle. Firstly, its safety has been already proved. The other is that estimation of mass, volume and power is easier by using published documents. Technology Readiness Level (TRL) is used to assess their feasibility. The specifications of LSS referenced from LSS on ISS is based on [20]. On the other hand, to reduce the mass and power needed, several new technologies are used. These technologies have been tested in laboratory but still under developed.

TABLE II: Overview of LSS

	Mass [tons]	Volume [m^3]	Power [kW]
Total	**1.52**	**3.32**	**2.26**
Environmental Control	1.06	2.07	1.19
Supply & Waste	0.29	0.55	0.51
Others	0.16	0.49	0.46

The LSS on the transport vehicle is not completely independent but partly dependent to LSS on Deep Space Gateway (DSG) since the DSG is the base of the transport vehicle and the transport vehicle returns to the DSG all the time after missions on the lunar surface. The mission duration and number of crew used for calculation of specification of LSS is three weeks (21 days) in total and three crew members . This duration includes extra time to cope with troubles which can happen. LSS needs 1.5 tons of mass, 3.3 of volume and 2.2 kW of power in total.

LSS is roughly divided into three sections which are environmental control, Supply & Waste management and Others. In environmental control section, atmosphere control, for example carbon dioxide removal, is mainly considered. Here, temperature and humidity control are also taken into account. Supply & Waste section considers input and output for human body such as water, food and waste. In Others section, for example, PC for control of LSS, light and fans are considered. Others also includes equipment for safety and hygiene.

B. Environmental Control

Environmental control is critical for crew lives. For atmosphere control, CO2 removal, trace gas removal temperature control and humidity control are considered. It is decided that Oxygen is not recovered and CO2 removed from cabin air is just corrected in a tank and brought back to the DSG. A technology for CO2 removal is a new technology using an electrochemical membrane to separate CO2 from cabin air, and whose TRL is 4. How the technology works is stated in [21]. The other environment control technology are just as same as what used for ISS and these are stated in [20].

The technologies are already used on ISS, thus it has good reliability and its TRL is 8/9.

C. Supply & Waste Management

Thinking about food and water supply and waste management is necessary. Human consume 5kg of food, water, and oxygen in a day in total. Water and oxygen is not recovered on the transport system, thus enough amount of oxygen, water and food have to be carried to the moon's surface. Table 3 shows mass of supplies needed for one person to live a day. The amount of supplies is calculated based on the table and mission duration. Kitchen and toilet is almost as same as those on ISS, however, there is a gravity on the moon's surface, and some improvement will be needed to cope with both of environment with and without gravity. Regarding this, TRL is going to be 4.

D. Other LSS

Safety and hygiene is also one of the important issues. This includes fire detectors and extinguisher, body wash and clothes for example. The most of the systems uses technologies for ISS to make the spacecraft safer by using current systems. There is gravity on the moon's surface, then it is a possible choice to use a shower in the transport vehicle. However, if shower is used, much more water is used and it means more water have to be carried in the situation without water recovery. Thus, to reduce the system mass and complexity, wet towels are used to wash body during moon missions. When crews come back from outside, dust on the space suit will be a problem. Dust is small and can cause malfunction of space suit and other equipment. To avoid it, when air lock is filled with air, fan with filter works to get rid of dust in the air lock. It will take several minutes.

Other technologies used for lights or system management and so on are the same ones used for ISS. To be redundant, all equipment for environment control which is critical for crews to keep their lives are carried with its spare.

X. CONCLUSIONS

Even though this is a preliminary study and conceptual, all the systems and technologies used have a medium to high level of Technology Readiness (TRL4 to TRL9) and hence we can affirm that the proposed system would be feasible.

Due to the orbit selection, we have some constraints and drawbacks we still need to work on to solve them in a efficient and safe way but this would be out of the scope of the project.

XI. DIVISION OF LABOUR

- Abstract and Introduction: Alvaro Cano, Svenja Wanner
- Trajectory Analyses: Alvaro Cano
- Mass budget: Alvaro Cano
- Targets on Lunar surface: Alvaro Cano
- Design Mission: Alvaro Cano
- Transport Design: David Salmi
- Propulsion System: David Salmi
- Power Budget: Svenja Wanner
- Communication: Svenja Wanner
- Life Support Systems: Masahiro Yamamoto
- Conclusion: Alvaro Cano

APPENDIX I
ABBREVIATIONS

– DSG: Deep Space Gateway
– LSS: Life Support System
– TRL: Technology Readiness Level
– ISS: International Space Station
– NRHO: Near rectilinear halo orbit
– L1: Lagrangian point 1
– L2: Lagrangian point 2
– BOL: Beginning of Life
– EOL: End of Life
– GaAs: Gallium-Arsenide

APPENDIX II
LIST OF SYMBOLS

V - Rocket speed in direction of motion.
X - Arc length distance from launch.
Y - Altitude from the surface of the Moon.
γ - Rocket inclination, 90° is vertical.
R_m - Radius of the Moon.
m - Mass of the rocket.
β - Mass rate function.
D - Drag force.
T - Thrust force.
I_{sp} - Specific impulse.
V_{eff} - Effective exhaust velocity.
G - Gain

η	-	Aperture Efficiency
R	-	Radius of Antenna
P_n	-	Noise Power
k	-	Boltzman's Constant
T	-	System Temperature
B	-	Bandwidth
P_r	-	received Power
DR	-	Data Rate
E_o/N_o	-	Energy per bit to noise density ratio
P_t	-	transmitted Power
RF	-	Rain fade

APPENDIX III
TRAJECTORY FIGURES

Figure 8 belongs to the ascent phase.
Figure 9 belongs to the descent phase.

APPENDIX IV
SOLAR CELL EFFICIENCIES

For Solar Cell Efficiencies, please see Figure 10.

APPENDIX V
DIMENSIONING OF BATTERIES AND SOLAR ARRAY

For Dimensioning of the Batteries, please see Figure 11.

APPENDIX VI
LINK BUDGET CALCULATIONS

For Link Budget, please see Table **??**.

Calculations conducted by DSG group on basis of [14].

Calculation of Gain:

$$G = 10 \times \log_{10}(\eta \times (4\pi/\lambda^2) \times (\pi \times R^2)) \quad (8)$$

Calculation of Noise Power:

$$P_n = 10 \times \log_{10}(k \times T \times B) \quad (9)$$

Calculation of received Power:

$$P_r = (E_o/N_o) + P_n + 10 \times \log_{10}(DR/B) \quad (10)$$

Calculation of transmit power:

$$\begin{aligned} P_t = P_r - G_t - G_r + 20 \times \log_{10}(4\pi/\lambda) \\ + 20 \times \log_{10}(R) + RF \end{aligned} \quad (11)$$

	Earth → DSG	DSG → Earth	Transport → DSG	DSG → Transport
Frequency Interval [GHz]	7.25-7.75	7.9-8.4	7.25-7.75	7.9-8.47.75
Chosen frequency [GHz]	7.5	8.3	7.75	8.4
Distance [m]	$3.91 \cdot 10^8$	$3.91 \cdot 10^8$	$7.18 \cdot 10^7$	$7.18 \cdot 10^7$
Temperature receiver [K]	20	298	20	373
E_b/N_0 [dB]	14	14	14	14
Data Rate [Mbps]	2	10	10	1
Bandwidth [MHz]	10	10	2	2
Diameter receiving antenna [m]	0.75	15	0.73	0.75
Received power [dB]	-138.6	-119.9	-131.6	-128.9
Free space loss [dB]	221.8	222.6	207.4	208.0
Gain transmitter [dBi]	61.2	35.0	35.2	35.7
Gain receiver [dBi]	34.2	62.0	35.0	35.9
Power transmitter [dB]	-12.1	5.7	5.5	7.5
Power transmitter [W]	0.1	3.7	3.6	5.7
EIRP (transmitter) [dBW]	49.1	40.8	40.8	43.2

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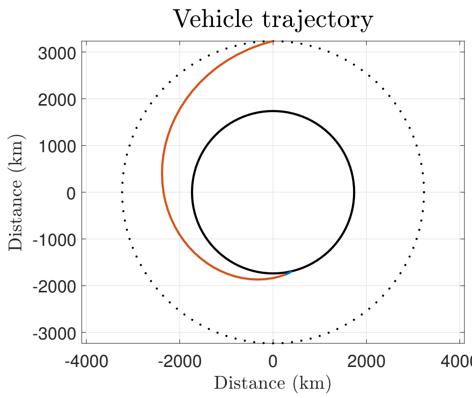
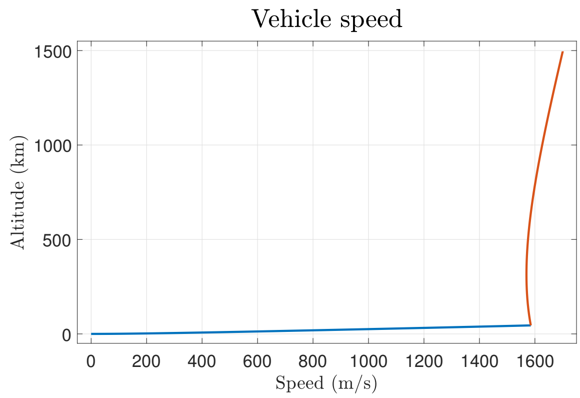
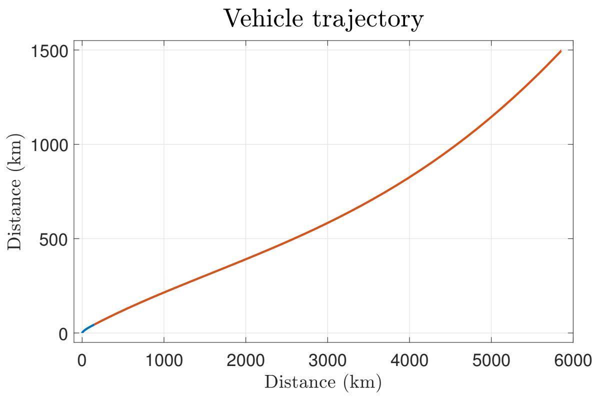


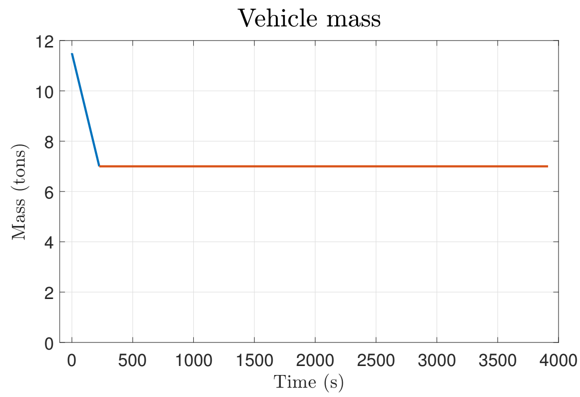
(a) Ascent Orbital trajectory



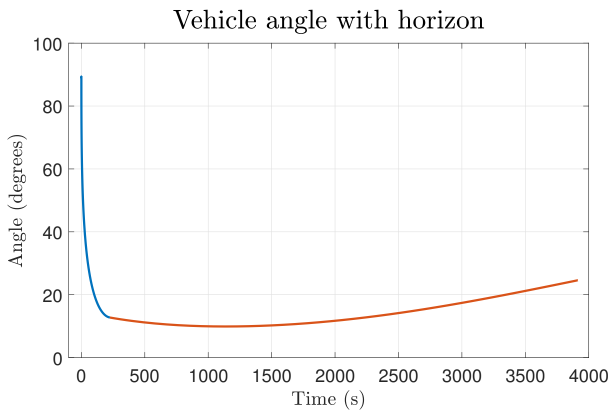
(b) Ascent Speed



(c) Ascent Trajectory

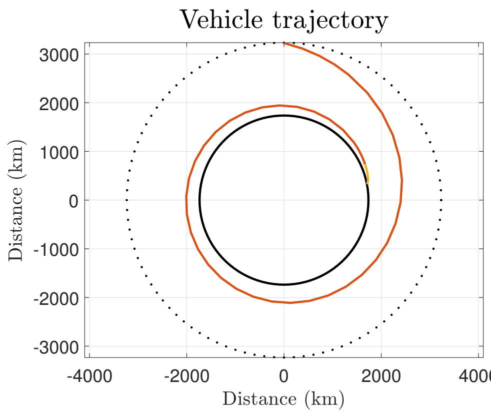


(d) Ascent Mass evolution

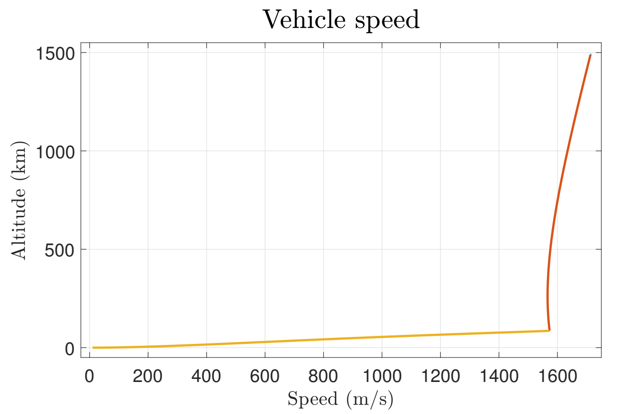


(e) Ascent Pitch angle

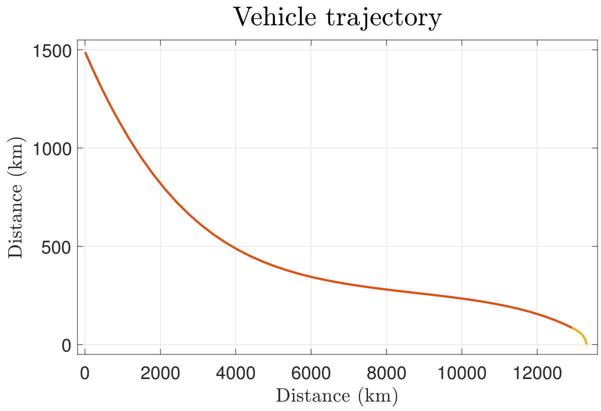
Fig. 8: Ascent phase trajectory



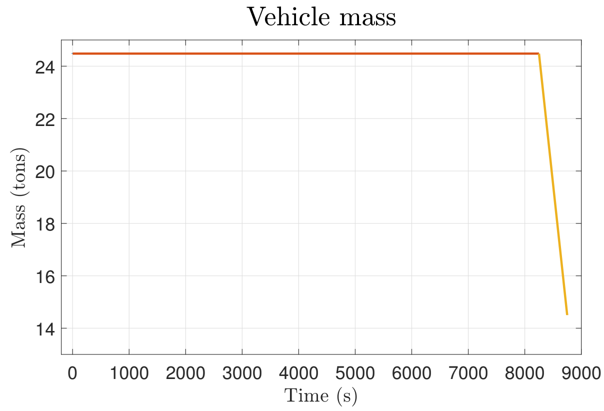
(a) Descent Orbital trajectory



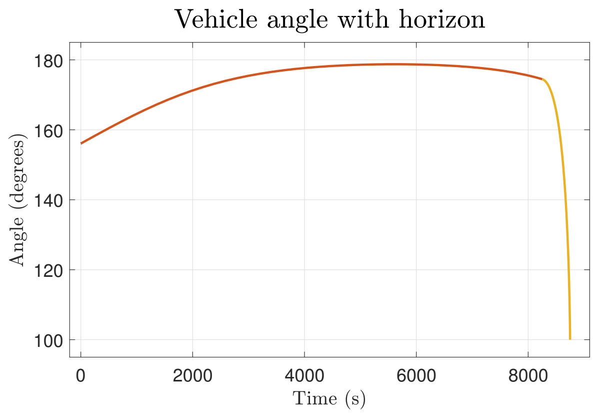
(b) Descent Speed



(c) Descent Trajectory



(d) Descent Mass evolution



(e) Descent Pitch angle

Fig. 9: Descent phase trajectory

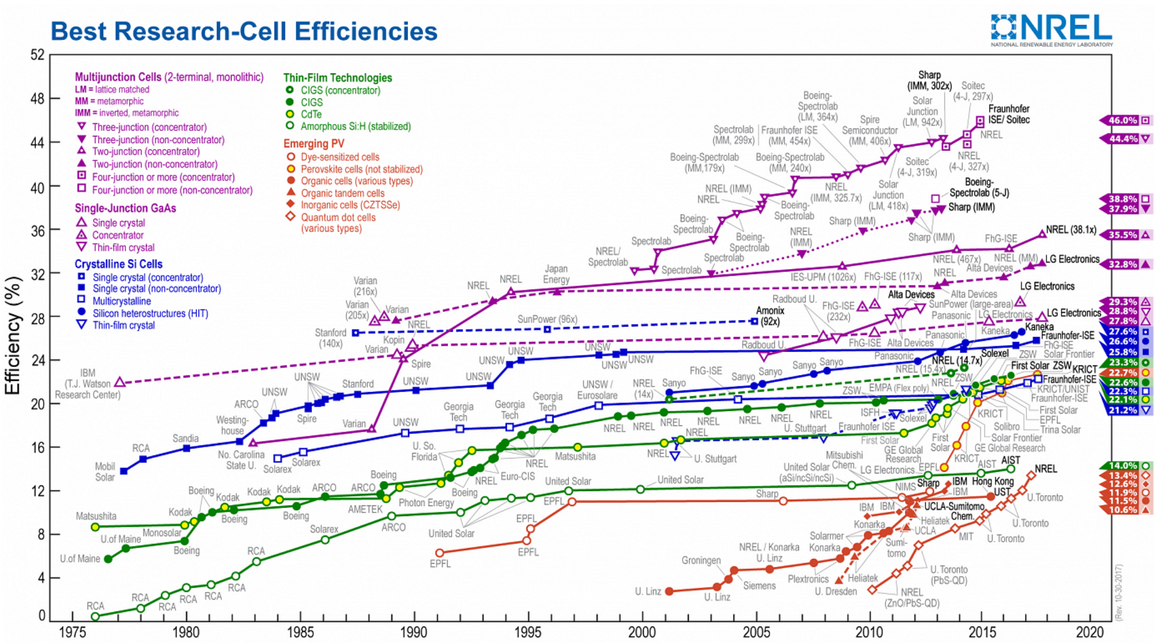


Fig. 10: Cell Efficiencies [4]

Emergency Scenario - Dimensioning of Batteries

Power Consumption		
Life Support system	2.26	kW
Computer, Controllsystem	0.75	kW
Total Consumption	**3.01**	**kW**

Maximal battery use		
Flight to Moon	2	hours
Stay on Moon	0	hours
Flight to DSG	10	hours
Total battery use	**12**	**hours**

Energy storage capacity		
Energy storage capacity	**36.1**	**kWh**

Battery data		
Battery storage capacity	4.492	kWh
Number of Batteries	9	
Length	530	mm
Width	300	mm
Height	240	mm
Total height	2160	mm
Volume	**0.34**	**m^3**
Mass per battery	49.9	kg
Total mass	**449**	**kg**

Normal Case - Dimensioning of Solar Panels

Power Consumption		
Life Support system	2.26	kW
Computer, Controllsystem	0.75	kW
Total Consumption	**3.01**	**kW**

Duration of Power Consumption		
Flight to Moon	2	hours
Power Consumption on Moon	100.5	hours
Flight to DSG	2	hours
Total Duration	**104.5**	**hours**

Additional charging on Moon		
EMU	0.85	kW
Amount	3	units
Operation time	9	hours
Storage capacity for EMU	22.95	kWh
Drill	0.5	kW
Operation time	9	hours
Storage capacity for Drill	4.5	kWh
Time on Moon	6.7	days
Total charging capacity	**183.92**	**kWh**

Energy storage capacity		
Storage capacity battery	36.1	kWh
Power consumption	498.1	kWh

Solar Panel Size/Mass		
Required power from solar panel	462.1	kWh
Charging on Moon	160.8	hours
Charging Power	2.87	kW
Power per area	**353.5**	W/m^2
Total Size	**8.1**	**m^2**
Number of Panels	2	
Size per Panel	4.06	m^2
Durchmesser	2.27	m
Mass per kW (MegaFlex)	0.1	kW/kg
Mass	**29**	**kg**

Fig. 11: Dimensioning of Batteries and Solar Array