Deep Space Gateway Design

Stefano Barra, Julian Grujicic, Hanna Kindberg, Dominic Roe, Ludovica Valentini

***Abstract*— This report is a discussion of a conceptual design of a Deep Space Gateway to support future manned lunar exploration starting in 2025 for a period of at least 5 years. The study has shown that the design of a Deep Space Gateway is possible, however there are still challenges that have to be dealt with; such as how to maintain a Closed Ecological Life Support System, where nearly everything has to be recycled and reusable, as well as new technologies to develop and to be tested in order to accomplish both the human and the mission operational requirements.**

I. INTRODUCTION

Most people agree that the next big step in human space exploration is going to Mars. Before this becomes reality, it is necessary to learn how a long time in space effects people, both physically and mentally. Some areas have already been investigated aboard the ISS, however, due to the relatively short distance to Earth, other areas can not be. Therefore a new space station, the Deep Space Gateway, DSG, has been proposed.

A. Background

The DSG concept has been explored for a number of years by NASA as successor to the ISS. It will exist to support further science and access to Deep Space for future manned exploration. The difference in this project being that, at least initially, the DSG will support manned missions to the Lunar Surface instead of Deep Space although this will also be considered as an "end of life" option.

B. Objectives

The objective of the report is to present different options and a conceptual design of the DSG. The following areas are to be considered:

- Orbit
- Communication
- Size and main elements
- Radiation
- Thermal control
- Propulsion system
- Power system
- Life support system
- Logistics
- Science aboard
- Off-nominal case

II. ORBIT AND ATTITUDE

A. Orbit

Part of the challenge of designing a space architecture for human exploration beyond Low Earth Orbit (LEO) is finding suitable orbits that meet multiple requirements and constraints imposed by the mission. First of all, Earth access is limited by the capability of the launcher; secondly, a suitable lunar access has to be provided to support lunar surface activities. Additional mission design drivers include communication coverage, station keeping cost (cf. Table VIII in Appendix A), exposure to the Sun, surrounding environment and other operational implications. Finally, the access to deep space has been also evaluated since this mission could be an intermediate stage to reach it, in particular Mars will be the next aim.

Different cis-lunar orbit types have been analysed based on the previous aspects, Low Lunar Orbit (LLO), Prograde Circular Orbit (PCO), Frozen Lunar Orbit, Elliptical Lunar Orbit (ELO), Near Rectilinear Orbit (NRO) (a.k.a. Near Rectilinear Halo Orbit, NRHO), Earth-Moon L2 Halo and Distant Retrograde Orbit (DRO); all of them, apart from LLO, are shown in Fig. 1.

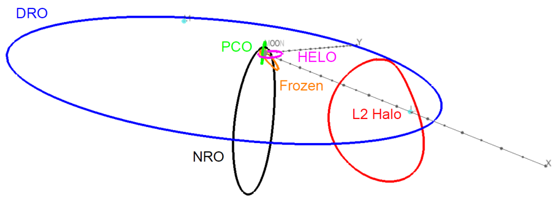


Fig. 1: Potential Cis-Lunar Orbits. [1]

This evaluation indicates that the NRHO is the best trade-off, as Fig. 2a shows.

NRHO are a sort of bridge between L1 and L2 halos in the Earth-Moon system and can be considered an extension of the Halo family [1], but they are much closer to the Moon; they are elongated and resemble an ellipse in the CR3BP (Circular Restricted Three-Body Problem) rotating frame and have large amplitudes over either the north or the south pole with shorter periods close to the opposite one and are therefore called L1/L2 North/South NRHO (Fig. 2b). They remain relatively fixed in the Earth-Moon plane, rotating at the same rate as the Moon around the Earth and the Moon around its own axis.
Finally, the choice has been the Earth-Moon L2 South NRHO in 9:2 resonance with the lunar synodic cycle [2]. The selection of L2 was mainly due to the visibility to the far side of the Moon and better stability than L1; however these benefits require more propellant for the Orion to reach the DSG. However, South orbits require less propellant for the returning trip since a splashdown in the northern hemisphere is assumed. A deep analysis and comparison among various orbits characterized by different periselene values (e.g. 9:2 orbit with $r_p = 3233$ km, 4:1 orbit with $r_p = 5931$ km, an $r_p = 4500$ km NRHO L2 South Family, ...) led to the 9:2 orbit since it presents no eclipses by the Earth over 19 years, implies less propellant for station keeping, a lower landing time and fuel consumption for the Lunar Transport System and feasible opportunities on a daily or near daily basis for Orion missions to NRHO. On the other hand it is difficult to avoid eclipses by the Moon, there are on average 10.5 per year, but since they are short, about $1.2 - 1.3$ h, they are manageable. Table VII in Appendix A presents the parameters that characterize the chosen orbit.

B. Attitude

Before describing its attitude the DSG body frame has to be specified; the X-axis is in the forward direction of the vehicle, the Y-axis is to starboard (right) and the Z-axis is to Nadir (down), as Fig. 8 in Appendix A shows. The main attitude of the DSG is the XVV (X-axis in the Velocity Vector) with respect to the LVLH (Local Vertical Local Horizontal) frame of reference and it has been chosen to maximise the power, minimise the negative thermal effects and meet the requirements of communication, i.e. keeping the antennas always pointing at the Earth and the Moon. It also provides several additional advantages; firstly, solar panels are always exposed to the Sun so that radiation comes perpendicular to their surface; moreover, the main direction of the heat sinks is parallel to that of the solar radiation so that they can get rid of excess heat in the most efficient way. Finally, in this way the station can always point at the Moon allowing the possibility of continuous communication; on the other hand in order to keep contact with the Earth the antenna has to be moved, but since the Earth is much further the movement of the antenna will be much smaller. Nevertheless, sometimes the attitude can vary, depending on the type of activities and manoeuvres it is expected to accomplish; for instance, during visiting vehicles docking/undocking phases it can move to a specific attitude relatively to the approaching/departing vehicle (this kind of operations can be 180° yaw or 90° pitch and so on). Once the manoeuvre is completed the DSG will go back to its normal attitude.

III. Communication

The DSG has good communication access to Earth due to the chosen orbit. However, the communication with parts of the moon is limited, see figure 3. Therefore, it was decided to place a communication satellite in the NRHO north orbit that mirrors that of the DSG. The communication satellite ensures that the transport system has sufficient communication access to earth during the time when the communication access to the DSG is limited. More about this can be read in the transportation group report, however, an overall description of the communication system can be seen in Figure 4.

A. System

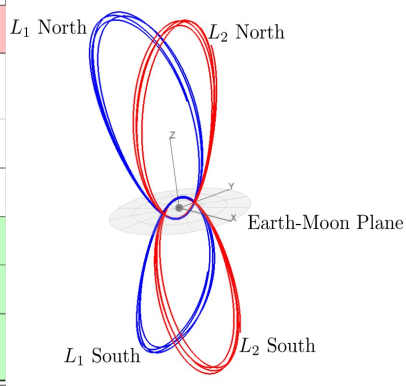
When choosing the communication system, two options were investigated, either to use radio frequency or optical communication, also known as

[1] A halo orbit is a periodic, three-dimensional orbit near the L1, L2, L3 Lagrange points in the 3-body problem of orbital mechanics.
[2] There are 9 NRHO revolutions per 2 lunar months.

Orbit Type	Earth Access (Orion)	Lunar Access (to Polar LLO)	Crewed Spacecraft Stationkeeping	Crewed Spacecraft Communication	Crewed Spacecraft Thermal
Low Lunar Orbit (LLO)	Infeasible	$\Delta V = 0$ m/s, $\Delta T = 0$	50 m/s + per year	50% Occulted	Radiators Insufficient
Prograde Circular Orbit (PCO)	Marginally Feasible	$\Delta V < 700$ m/s, $\Delta T < 1$ day	0 m/s for 3 years	Unknown	Unknown
Frozen Lunar Orbit	Marginally Feasible	$\Delta V = 808$ m/s, $\Delta T < 1$ day	0 m/s	Frequent Occulation	Unknown
Elliptical Lunar Orbit (ELO)	Marginally Feasible	$\Delta V = 953$ m/s, $\Delta T < 1$ day	>300 m/s	Frequent Occulation	Unknown
Near Rectilinear Orbit (NRO)	Feasible	$\Delta V = 730$ m/s, $\Delta T = .5$ day	<10 m/s per year	No Occulation	Radiators Sufficient
Earth-Moon L2 Halo	Feasible	$\Delta V = 800$ m/s, $\Delta T = 3$ days	<10 m/s per year	No Occulation	Radiators Sufficient
Distant Retrograde Orbit (DRO)	Feasible	$\Delta V = 830$ m/s, $\Delta T = 4$ days	0 m/s	Infrequent Occulation	Radiators Sufficient

Legend	Favorable	Marginal	Unfavorable

(a)



(b)

Fig. 2: Staging Orbit Summary Comparison (2a) [1]. Cis-Lunar NRHO Family (2b).

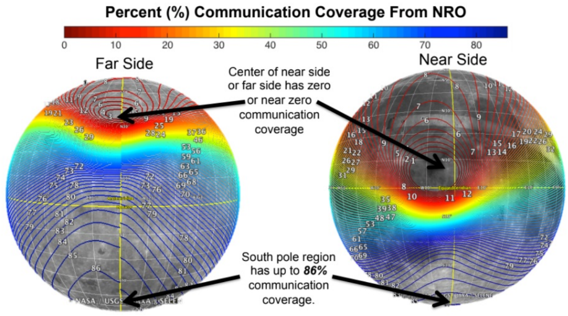


Fig. 3: Moon communication availability. [1]

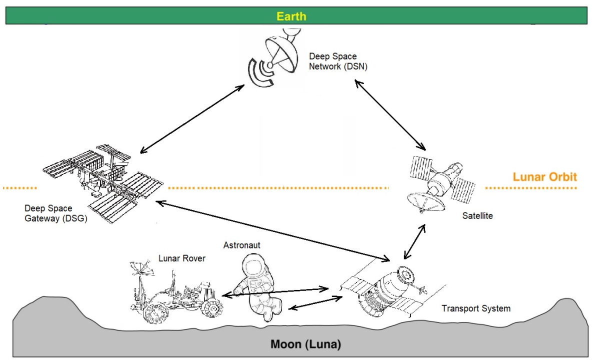


Fig. 4: The DSG communication system.

laser communication. Radio frequency has long been used for satellite communication, while laser satellite communication is a quite new technology. Due to the importance of the system the choice is therefore to use radio frequency for the main communication system. However, the idea is to have an experimental laser communication system to test it for use when further exploring the solar system.

The different frequency bands used for radio communication were also compared. The advantages of a higher frequency band is that they have wider bandwidths and that the antennas are smaller. On the other hand lower frequency bands are less affected by rain fading, i.e. absorption of the radio signals in the atmosphere due to rain and snow. [2] After a trade-off between these aspects the final choice was to use the X-band for communication with earth and the lunar transport. The band has a uplink frequency of 7.9-8.4 GHz and a downlink frequency of 7.25-7.75 GHz.[3]

B. Link budget

Link budget calculations were made and the required power transmit, P_t, was calculated according to

$$P_t = P_r - G_t - G_r + L_{fs} + L_{rf} \tag{1}$$

where P_r is the power received, G_t the transmitter gain, G_r the receiver gain and L_{fs} the free space loss. The loss due to rain fading, L_{rf}, is only dominant for frequencies above 10 GHz [4] and is therefore set to zero in this case. The power received is calculated by

$$P_r = \frac{E_b}{N_0} + P_n + 10\log_{10}\frac{DR}{B} \tag{2}$$

where $\frac{E_b}{N_0}$ is the required energy per bit relative to the noise power, DR is the data rate, B is the bandwidth and P_n is the noise power given by

$$P_n = kTB. \tag{3}$$

Here k is Boltzman's constant and T is the receivers temperature. The antenna gain is calculated by

$$G = 10\log_{10}(n(4\pi\lambda^2)(\pi R^2)) \tag{4}$$

where n is the aperture efficiency, λ the wavelength and R is the radius of the antenna. Finally, the free space loss is given by

$$L_{fs} = 20\log_{10}(4\pi\lambda) + 20\log_{10}D \tag{5}$$

where D is the distance between the two antennas. All the requirements for the communication system were based on the requirements suggested by ESA for communicating with the DSG [5] and the specific values can be seen in Appendix B The results for the link budget can be seen in table I and all of the parameters used can be found in appendix B.

TABLE I: Results from the link budget

Link	**Earth → DSG**	**DSG → Earth**	**Transport → DSG**	**DSG → Transport**
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 transmit [dB]	-12.1	5.7	5.5	7.5
Power transmit [W]	0.1	3.7	3.6	5.7
EIRP transmitter [dBW]	49.1	40.8	40.8	43.2

IV. Propulsion system

For missions beyond LEO, spacecraft size and mass can be strongly prohibitive and constitute a large part in the overall mass budget; in order to minimize this effect, an advance Solar Electric Propulsion (SEP) system, along with advanced solar array technology, has been chosen as the main propulsion system for the DSG. In this way 10 times less propellant is required than a comparable, conventional chemical propulsion system that can constitute even more than 50% of spacecraft mass. Additional significant exploration benefits are high specific impulse and high power, low fuel consumption, cost saving, safety and long life, since the minimum estimated operational lifetime is 15 years in cis-lunar space. [6]

Since 2012 NASA has been developing a 13.3 kW Hall thruster electric propulsion string that represents the building block for a 40 kW-class SEP capability. The development is led by NASA's Glenn Research Center and Jet Propulsion Laboratory (JPL) and it was extended to Aerojet Rocketdyne through the Advanced Electric Propulsion System (AEPS) contract. The Power and Propulsion Element (PPE) will be one of the first elements to be launched for the assembly of DSG due to its primary importance in the DSG mission.

The main task during the mission is the orbit maintenance, i.e. station keeping manoeuvres. The PPE will also provide non-propulsive attitude control using RCS (Reaction Control System), i.e. momentum wheels, and SEP thrust vectoring; in this case the propulsion system will be used to desaturate the RCS when the storage of further angular momentum is prevented by the maximum velocity, since other methods, like magnetorquers or gravity gradient torques are not as effective due to weak magnetic and gravity fields of the Moon.

Additional secondary, but still important, capabilities regard emergency/avoidance and rendezvous/docking manoeuvres if, and when they are necessary; the former are more likely to be faster and more demanding so that the SEP is not always feasible and a chemical propulsion is needed. On the other hand, keeping a chemical propulsion system would have some drawbacks, so in this kind of situation one of the spare Transport Vehicles, that are generally docked to the station, will be used. In any case, it would be better to use the SEP if the required Δv is small and if it is known far enough in advance, in order to consume less fuel and not to reach high accelerations that could be dangerous for the deployed arrays. The SEP will also allow the insertion from Translunar injection into NRHO, this aspect will also be determining in the amount of propellant to be stored, even if the PPE could also be refueled on-orbit. Moreover, it can also be used for orbit transfers that could be necessary to support some Moon operations or future missions to further deep space. In fact, the PPE is a readily scalable technology with a clear path to much higher power systems and it will demonstrate that is suitable for future human Mars class missions. Finally the PPE will be capable of transferring electrical power to the external hardware, like providing communications to Earth, visiting vehicles and crew on EVA, thanks to the high-voltage power management and distribution (Power Processing Units, PPUs).

The Ion Propulsion System (IPS) includes four identical independent electric propulsion strings made up of a Flight Thruster (FT), i.e. the Hall Thruster, a PPU, a Xenon Flow Controller (XFC) and Interconnecting Cable Harnesses each. [6] It also includes two 2000 kg-class Xenon Composite Overwrap Pressure Vessels (COPVs), filled up to 1.25 t each, at the optimal storage condition of $p = 150$ bar and $T = 25°C$. [8] A deeper analysis for the propellant consumed for the annual station keeping is attached in Appendix F.

The overall mass of the propulsion system is slightly more than 3 t, it includes around 0.5 t for all the PPUs and FTs [7], 2.5 t of Xenon and 0.1 t for the empty tanks (cf. Appendix F). The thrusters can reach a specific impulse of over 2600 s at a discharge voltage of 600 V and provide a thrust up to 589 mN with a mass flow rate of 22.9 mg/s. [6]

The AEPS programme has completed the first phase of the design process through the Preliminary Design Review (PDR) and successfully completed an early integration system test demonstrating the required control and throttling capabilities of the thruster, PPU and XFC. [7] Since the PDR corresponds to a Concept Maturity Level CML 8, this means that all the parts/technologies have reached a TRL of at least 6. [9]

V. Power system

The major power subsystems are the Power generation/conversion, the Power management and distribution and the Energy storage. Generally, it is desired to utilize power systems that have high power capabilities with high specific power α $[\frac{W}{kg}]$ and low cost.

For this mission a photo voltaic power system has been chosen to supply the solar electric propulsion system with power. It works by converting input solar illumination to electricity.[30]

Key components are: substrate, solar cells, array structure, deployment mechanism and energy storage. This type of system is widely used within the space industry, for example at the ISS. Power systems are usually around 20-30 % of the spacecraft's or space station's mass and costs 20 percent of its budget, which corresponds well with achieved values in Table XII and Fig. 13.[30]

A. Substrate

When deciding the solar arrays and their structure, several options were studied; the Mega-Rosa and the Mega-flex, both with a high specific power of 150 W/kg achieving all SOA-related goals including 4x rad tolerance, 1.7x power/mass kW/kg, 4x stowed volume efficiency, and 20x deployed strength.[31]

Moreover both also have a TRL level 6, which means that they have been through the main laboratory tests, but have yet to be tested in space.[30]

However, Mega-flex having the smaller cross-sectional area (taking into account 2 block 1a of 15m in diameter) as can be seen in Table II was chosen for further analysis.[32]

TABLE II: Solar array deployed Cross-sectional area

Solar array comparison:	**Mega-Flex**	**Mega-Rosa**
Type	2pcs $d = 15m$	2pcs $4.2m \cdot 24m$
Total power $[kW]$	200	200
Cross Sectional Area $[m^2]$	353.25	403.2

Mega-Flex - like Ultra-Flex its predecessor, which was used on the Mars Phoenix Lander - is an accordion fanfold flexible blanket solar array comprised of interconnected isosceles-triangular shaped lightweight substrates, or gores.

Photo voltaic cells are bonded to the weave mesh blankets [33]. Mega-Flex can generate electrical power using any type of space-grade solar cell [34].

B. Solar cells

The industry is currently developing 4 to 6 junction cells with funding, 33 to 36% efficiency projected with qualification by 2017, the goal is to reach an efficiency of about 39 %. Multi junction solar cells of the type XTJ triple junction solar cells have today an efficiency of about 29% for the solar cells today's technology was considered.[30]

C. Array structure and deployment mechanism

When stowed, the solar array is configured as a flat-pack to produce a compact launch volume and high system frequency. The circular membrane structure, which contains radial spar elements, becomes tensioned similar to an umbrella, resulting in a highly efficient, strong and stiff structure.[33]

This level of performance, which is 4 times higher than typical planar arrays has a store ability of $>$ than $40kW/m3$ [34].The deployment process is shown in figure 5 [29].

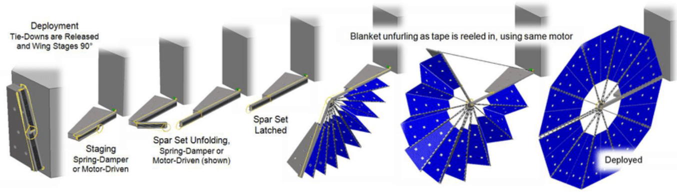


Fig. 5: Deployment mechanism of Mega-flex.

D. Energy storage

Furthermore, discussing the energy storage one has to consider what type of storage to use depending on what kind of mission requirements one has. In this case three different types of batteries were chosen: thermal batteries, primary batteries and secondary batteries.

To begin with, the thermal batteries are high power with minutes of discharge time, one time use and will be used for critical events such as emergency maneuvers[30].

Moving on to the primary batteries having hours of discharge time, also one time use, but offer relatively wide temperature operation when the solar arrays are not available [30].

Lastly, there are the secondary batteries operating as main energy storage. They have hours of discharge time, but are reusable with a cycle life time of more than a 1000 cycles[30]. The secondary batteries are of lithium-ion battery type and offer a specific energy level of 200 Wh/kg. The operating temperature is between -10 degrees to 30 degrees Celsius. But since they will be stored inside the space station, where they will be temperated by the Life Support System, this will not be a problem [30] .

E. Calculations

To estimate the mass of the solar arrays M_{El} that are needed to supply the total power P_{tot} which had been estimated in an earlier step (Appendix G), was simply divided by the the specific power here written as α [8].

$$M_{El} = \frac{P_{tot}}{\alpha} = 1330kg \tag{6}$$

The battery mass could be calculated using equation 7 below where α_b is the specific energy level and 1.25 represents the mass of the wiring box and 1.05 is the maturity margin [35].

$$M_{batteries} = \frac{P_{tot} * t * 1.25 * 1.05}{\alpha_b} = 3937.5kg \tag{7}$$

VI. Life Support Systems

The Life Support Systems exist in order to maintain conditions on-board the DSG to support the continued health and wellbeing of the crew as well as provide a comfortable working environment. Although not a system, the provision of Food for the crew is also considered within this section.

A. Requirements

Human Requirements: The performance of the Life Support Systems is defined by the consumption and output of the astronauts defined for a single astronaut in table III below. The overall system shall be capable of supporting a maximum crew of 6 in normal conditions.

Environment: Due to the longevity of missions comfort in the environment is as critical as sustaining life. The easiest way of providing psychological comfort is to replicate the earth bound environment which, conveniently, is the environment provided on the ISS. This means that systems

TABLE III: Mass of Human Consumables [10]

	Consumable	Mass
Input	Oxygen	0.8
	Water	3.6
	(Dry) Food	0.6
Output	Carbon Dioxide	1.0
	Water	3.9
	Solids	0.1

that have previously been qualified for flight will continue to be so. Furthermore, this environment provides increased safety, by selecting pressures similar to earth; oxygen concentration required will not have to be increased thus will provide no added flammability risk. The internal temperature maintained will be the in the human comfort range, or normal room temperature, specified in Table IV below.

TABLE IV: Internal Environment of the ISS/DSG [11]

Factor	Min	Max	Unit
Pressure	97.9	102.7	kPa
Temperature	20	24	°C
Humidity	55	65	% [12]

B. Air Systems

Pressure: Estimating based on ISS data, the number of days from maximum to minimum pressure is directly related to the size of the station, 899m^3 of free air volume giving 146 days between levels at a leak rate of 0.45kg/day. A smaller station would require more frequent resupplies of air and nitrogen, to maintain pressure. However, these leaks are small compared to the loss due to EVAs and docking with resupply vehicles.

The only existing method of replenishing Nitrogen and Oxygen is the Nitrogen/Oxygen Recharge System (NORS). Eventually, the system will require full closure but for the lifetime of the DSG this lack of closure is not an issue.

Air Quality: Current CO_2 systems trap the gas in solids for ejection from the station. In 2017 a paper was presented showing a proposal for a Ionic Liquid based system for recovering CO_2 and water from the air. The filtration system design was based on multiple existing components in current Flight Hardware which would suggest it to be around TRL4/5, with a fully functional system having been sufficiently developed in the project time-frame. The Sabatier system that complements CO_2

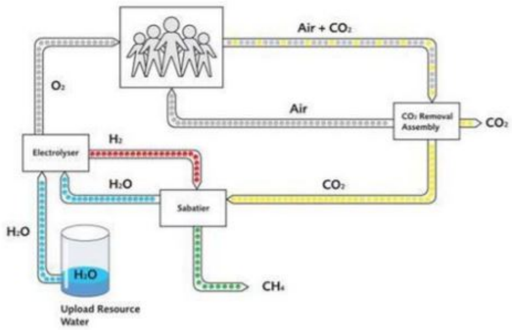


Fig. 6: The ACLS Schematic [14]

filtration has been under development by ESA for a number of years as part of its Advanced Closed Loop System. This collects CO_2 and converts it to Methane and Water, the water is in turn converted to Hydrogen for further production of Methane and Oxygen. This is graphically presented in Fig 6. This system has now been designed to fit into a standard rack utilised on the ISS, and will fly to the ISS to be installed in the Destiny node during 2018, on board the HTV 7 mission [14].

C. Food & Water

Food: When planning the food for a mission in space there are different aspects that are important to consider. Firstly, it must ensure that the crew get all the nutrients needed to stay healthy. Secondly, it must be possible to store most of the food for a longer period of time. Thirdly, the food must be safe in regards to microorganisms. Lastly, it must be packaged in such a way that it has the minimum possible mass and volume. [15]

The areas of nutrients, shelf life and microorganisms have been well researched during the ISS missions and a lot of the knowledge gained can be transfered to the DSG missions. Regarding the mass and volume of the packaging it is even more important that it is as minimized as much as possible as the transport time and cost is larger to the DSG than to ISS as the initial strategy is to provide food from earth.

Since the DSG is also meant to be a practice before traveling to Mars, where the possibilities to re-supply food are limited, the plan is to start producing food aboard the DSG. The production of food will increase gradually after it has been confirmed that the environment aboard the DSG does not pose any problems. Being able to produce a part of the food would not only decrease the resupply mass but also make it easier to ensure that the astronauts get all of the nutrients needed.

Water: There are 3 stages to the water cycle: potable water which is consumed by the crew, yellow water which is the product of crew and grey water, that being used for washing and the like. The aim of the water system within the Life Support System is to have a closed system requiring minimal resupply.

Current systems on the ISS recycle 93% of waste water so would be ideal for the closed loop cycle, however their size and mass makes them inefficient for deep space. The most effective solution capable for use on grey water and yellow water would be a forward osmosis system based on a membranes which is both smaller and lighter than the current system. A trial system was flown on the ISS in 2011 however the feasibility of the system is yet to be determined. It is still deemed that this system could be developed in time for the mission launch.

Alternate Biological processes are being developed in programmes such as Melissa at ESA. These systems are not as advanced as Forward Osmosis system however the biological processes could prove advantageous in the breakdown of the other products of the cleansing produce. This process, due to its lower readiness level would be more suited to testing on the DSG due to the lack of understanding in how the bacteria will behave in a higher radiation environment.

D. Waste Management

Currently all waste is stored on-board before being ejected to burn up in earth's orbit. This has several issues: firstly, the waste will not necessarily return to the desired orbit and secondly, to ensure it does so requires unnecessary fuel usage. It also requires large upmass to be sent from earth or the lunar surface so it not ideal. The current main sources of waste are packaging, clothing and human waste. By increasing the use of these resources the upmass can be reduced and the ECLSS loop on the space station can be closed further.

Although it may be impossible to fully reuse the waste based on today's available technology it is possible, experimentally, to recover a large part of the nutrients using nitrate based organics[16], some of these nitrates can be produced from the solids present in urine.

The most efficient way of recovering nutrients from faeces is by using bacteria to decompose to produce methane gas, 'microbial goo' [17] or simply using the dried nutrients as fertilizer for food production.

E. Exercise & Wellbeing

Exercise is critical to maintain the wellbeing of the astronauts supporting positive mentality as well as reducing the muscle loss experienced during micro-gravity flight. Currently exercise takes up 2 hours of a crews daily schedule compressing time available for science and maintenance.

In order to try and reduce this there has been development of a Gravity Loading Countermeasure Skinsuit. The purpose of the skinsuit is to increase dynamic loading during exercise. As this is still in the development stages this has TRL 5/6 [18] having been test flown as an early prototype [19].

On the ISS the main exercise equipment is the Advanced Resistive Exercise Device which takes up a large volume and mass. In order to reduce both of these as well as improve performance the DSG will use the Flywheel Exercise Device introduced to the ISS in 2009 and although only currently certified for certain exercises [18] giving TRL 7/8, although it is assumed that by the launch of the DSG it will be fully certified.

VII. On-board Science

A secondary objective of the DSG is to facilitate experiments developing technology to support long duration exploration in harsher radiation environments than those posed in LEO as well of the Physiological & Psychological effects that accompany this.

A. Science Facilities

The ISS currently uses a system of internal racks which hold experimental modules to ensure efficient use of space and allow easy interchange. The DSG will contain 8 of these racks to utilize previously gained knowledge of the use of these.

For external science facilities the JAXA modular system will be utilized as it allows each experiment to interchanged easily and simplistic integration with the DSG, each one of these is capable of providing up to 3kW.

B. *Proposed Science*

The proposed science on-board is designed to support the long term goal of the human exploration/possible colonisation of Mars. To do this further research into the physiological and psychological effects of long duration spaceflight is necessary, thus this will be the main focus of human research.

The other key area of research and development is the technology required to support these long duration flights. Whilst the DSG systems have been selected in order to minimise the resupply requirements, there is still a sizeable requirement for resupply in order to sustain the mission. Some suggested experimental research would be on further improving the efficiency of the Sabatier system as well as water recovery systems. NASA has previously developed the Veggie growth system[20] on board the ISS, objectives would be to assess the performance of this in the Deep Space environment as well as expand the variety of food certified for consumption from this system.

Externally, as previously mentioned, the primary system to test would be a laser communication system to assess the impact that the distance and atmosphere has on the ability to transfer data. Further payloads could include sensors to detect micrometeoroids or radiation spikes heading towards the gateway.

VIII. Resupply

Despite the high closure of the Life Support Systems and the efficiency of the propulsion systems there is still a certain amount of uplift required in order to maintain the efficient operation of the Gateway and support the lunar exploration missions. This is defined in table V below.

In order to reduce costs the number of resupplies should be minimized. To determine the duration between resupply missions a number of future launch vehicle capabilities were studied, namely the SpaceX Falcon Heavy and a number of configurations of the NASA SLS.

TABLE V: Monthly Resupply Masses

Group	System	Commodity	Mass (Kg)
Deep Space Gateway	**Propulsion**	Xe	0.5
	Life Support Systems	H_2O	2000
		Food	365
		N_2	2
	Science	Payload	$\leqslant$500
Lunar Transport	**Propulsion**	CH_4	14500
Lunar Exploration	**Science**	Payload	$\leqslant$5000
Total			**22,367.5**

TABLE VI: Comparison of Orbital Launchers [21]

Launcher	Mass to LEO	Mass to TLI
SpaceX Falcon Heavy	63,800kg	-
NASA SLS Block 1B	105,000kg	39,100kg
NASA SLS Block 2	130,000kg	50,000kg

Based on the launch capabilities in table VI it can be taken that the Falcon Heavy would require monthly resupply missions whereas either SLS would only require a bi-monthly mission. Therefore the two options can be used freely dependent on mission requirements and cadences.

Based on the bi-monthly SLS resupplies, 6-monthly crew changes as defined in the Overall Co-ordination Group report and the monthly Lunar Exploration Missions Figure 9 in Appendix C was generated to demonstrate the logistical manoeuvres at the DSG.

IX. Deep Space Gateway Design

A. *Assembly*

The design of the DSG was based on the selection of modules, some of which are similar to those existing on the ISS. The modules were then assembled in the final construction of the station, and their mass and power required were taken into account in final mass and power budgets, shown in Appendix G. The modules that will be part of the DSG are listed below:

- Habitation Module
- Science Module
- Power Module
- LSS Module and Main Storage Module
- Main Docking Module
- Secondary Docking Ports
- Observation Dome
- Airlock
- 2x Pressurized Truss

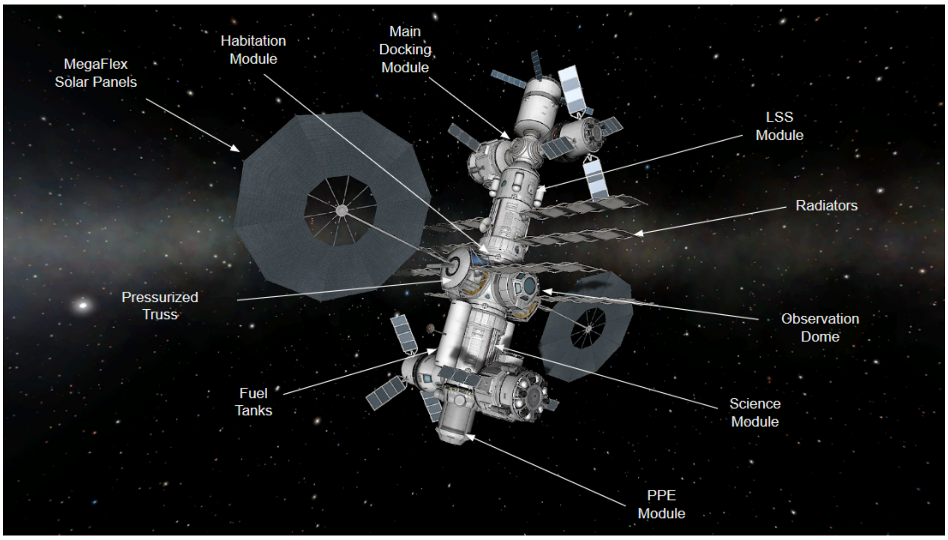


Fig. 7: DSG Assembly

These modules have been selected due to their functionality within the station, as described in the previous sections. Due to the long duration of stay and the distance in time between cargo mission, the DSG will employ two storage modules, a primary and a secondary. The implementation of a robotic arm hasn't been considered in the initial design due to the expectation that by the launch date, all the visiting capsules will have docking capabilities. The robotic arm was considered as a "sub-system" of a more advanced structural and operational design, including EVA procedure design. A more advanced study could discuss the possible interactions of a robotic arm have with the rest of the DSG.

The final assembly of the station, including the docking capsules configuration, can be seen in Figure VIII. The back view of the station, together with the dimensions and details of the docking system, are shown in Appendix D.

The final consideration of the DSG design involved the analysis of an off-nominal scenario. Specifically, a possible hazardous situation caused by fire on board. While such an event could be treated in a similar way as it is currently done on the ISS, this might not be the case if a fire would happen in the central sector of the station. If the crew were not able to extinguish a fire in such circumstances, it might have to isolate the central core and depressurise this part of the station. Such an event would force the crew to leave the DSG and head back to Earth. The worst possible situation, but not the most unlikely one, might cause the crew to be stranded in different parts of the station while the isolation of the central core occurs.

To provide a solution to this dangerous off-nominal case, the docking ports have been assembled in two different parts of the station, as a form of redundancy. This way, if part of the crew are separated from the others, they would still be able to use a docked capsule to abandon the DSG. Once the crew had left the station with such capsules, they would be expected to perform a docking maneuver in order to transfer the whole crew in one capsule suitable for Earth re-entry.

B. Radiation Environment

NASA has studied several limits for the radiation doses that an astronaut can safely absorb during their career. These limits are higher then the ones considered for the general public, differ between man and woman, and are summarized in Table XI in Appendix E.

It can be assumed that an astronaut on a mission in the DSG would absorb the maximum dose of radiation allowed during their whole career, with a security factor of $\sigma = 1.2$. In other words, for every astronaut this would be both the fist deep space mission and the last.

This assumption allows to use the career total radiation limit reported in Table XI. To make a conservative estimation on this limit, the design of the DSG should take into account the worst-case scenario on the radiation exposure, such as a female astronauts of age 35 (assuming this to be the minimum age for starting their career). The exposure limit is given by the smallest value divided by the security coefficient:

$$E_{limit} = \frac{1.75 \text{ Sv}}{\sigma} = 1.46 \text{ Sv} \tag{8}$$

This limit can be used for further analysis on the radiation shielding systems of the DSG and to give an estimate on the maximum mission length. It's important to note that this limit should be considered as a total exposition over the duration of the mission, thus decreasing the overall biological effect.

The radiation environment usually found in interplanetary space is sometimes referred as benign when compared to planetary poles or the Van Allen Belts. Nonetheless, the radiation level in L2 could be hazardous for long duration missions. The main source of radiations in this point that are, in order of relevance:

- Solar Radiation
- Cosmic Radiation

Several studies have been conducted in order to estimate the amount of radiation an interplanetary mission would have to stand. In particular, relying on the data acquired by the MSL mission to Mars, a paper published on Science [23] has provided an estimate of the total radiation for a Earth-Mars trip. Due to the similarity of radiation environment between the Earth-Marts transit, such study can be used to obtain an estimate of the combined effect of solar and cosmic radiation, that is (pag. 1082):

$$E_{cosmic} + E_{solar} = 1.84 \ \pm \ 0.33 \ 10^{-3} \ \text{Sv/day} \tag{9}$$

That would give, for a one year mission:

$$E_{total} = 0.87 \text{ Sv} \tag{10}$$

This result suggests that a shielding method might be used to reduce the radiation dose, but since this value is lower than the absorbed dose limit, the sole structure of the DSG might provide enough shielding.

During a period of relatively calm solar wind intensity, the structure of the station could lower the radiation risks to an acceptable level. Despite that, the solar wind is not constant and our prediction capability of the solar wind weather is limited. Furthermore, the biggest concern for a living habitat in deep space would be the radiation caused by Coronary Mass Ejection (CME). These high energy events can increase the risk of cancer for astronaut on long mission.

The use of a shield in a specific area of the station might help to reduce the overall risk. Such a shield would be placed in the external structure, and it would be more efficient if placed as external as possible to avoid the "ricochet" radiation effect of charged particle hitting metal surfaces, known as neutron activation. Such a material could be more easily implemented if the structure would be made by composite materials.

A material commonly used for radiation shielding is the Graded-Z laminate. The Graded-Z is a composite made by several materials with different atomic numbers, and it's commonly used in satellite-based particle detectors.

A study conducted by NASA et al [24] shows how different designs of Graded-Z provide shielding against radiation. These values have been validated during different solar activities level. On the event of the solar minimum of 1977, it has been shown that a shield of 150 mm of different aluminum-based Graded-Z laminates could absorb a daily dose of about 0.0243 Sv. The same laminate could be used with a much smaller thickness, e.g. on a range between 0.5 cm to 2 cm, only on a specific area of the DSG, to provide shielding to the astronauts during CME.

C. Materials

In the design of a space station, the main role of the structural elements is to withstand the atmospheric pressure, as well as keeping the correct temperature and granting a minimal radiation shielding. In the ISS, the first two are relevant factors for the structure design, while for the DSG the radiation shielding will become more relevant. This consideration allows to list the requirement for the materials to be used for the DSG:

- Lightweight
- Sensor integration
- Radiation shielding
- Micro-meteoroid protection

The main materials used for space applications are aluminum alloys, mainly due to their very low specific weight, but they allow few protection from radiation and from micrometeoroid impacts. The future of aerospace structures is based on the increased use of composite materials. These material have been tested since the 80's [25] and are used in current mission for many parts of launcher vehicles and some modules of the ISS.
Particularly interesting is the possibility of implementation of wires, pipes and sensors within the material itself: a more advanced design of such material could allow the use of sensors to detect strain increase, micrometeoroid, measure vibrations, radiation and temperature variation to help diagnose the material status and health. This will also reduce the number of sensors required for the DSG and increase the overall safety of the station.
On a more advanced level, electrical wires and pipes containing water could pass within the external shells, to improve thermal control, radiation shielding and allowing more space for the crew inside the station [26].

D. Thermal Control

A rough evaluation of the heat exchanged by the DSG and the space environment can be estimated using an heat balance equation,

$$\sum Q_{IN} + Q_{OUT} = 0 \tag{11}$$

where Q_{IN} and Q_{OUT} are the total absorbed heat and the total emitted heat. The main problem in heat control of space system is the removal of waste heat using radiative emission. In order to find the required heat sink, in can be conservatively assumed that the total heat absorbed and generated by the DSG has to be re-emitted. In reality, part of this will be used for maintaining a livable temperature, so the total heat to be disposed will be lower. Such control on the total irradiated heat can be achieved by using movable heat sink, allowing to change the radiative surface. These values can be broken down to smaller components, such as the heat absorbed by the DSG from the Sun radiation, Q_{SUN}, and the heat generated within it by electrical components or LSS, Q_{GEN}, while on the right hand side there is the heat re-irradiated by the station surface including solar panels, Q_{SURF}, and by the heat sink system, Q_{HE}. Using these data it's possible to evaluate the heat that needs to be disposed [27]. Such heat can be used to evaluate the size of the required heat sink, using the following equation:

$$A_{HS} = \frac{2}{S\varepsilon_{HS}}\left(Q_{SUN} + Q_{GEN} - Q_{SP} - Q_{SKIN}\right) \tag{12}$$

It can be assumed that the heat generated by the DSG is mainly caused by it's electric systems and the on board computers. These systems, when used, convert electric power into heat due losses, that can be estimated as 5% of the total power. that gives the requested radiative surface,

$$A_{HS} = 206.8\ m^2 \tag{13}$$

that can be arranged as eight heat sink panel of 1.5 m side and 9 m length, placed on the station in a direction normal to the solar panels to reduce the incident radiation from the Sun. Intermediate calculations are shown in Appendix H

X. CONCLUSIONS

To begin with, one can see from the performed study that the challenges to create a completely independent and self-supporting environment still remain, however the advances in previous years have made the goal more achievable and we believe that it will become more accessible with the construction of the DSG. Furthermore, some acceptance will have to be made adapting these systems to a new environment, some specifications that exist currently would be critically limiting to operations and experience. Therefore it remains important that the political and societal drive exists to accept this and work in unison to combat this.

Overall, this study suggests that the technology to create a sustainable support network for future human exploration exists, although some areas require development for future missions. The DSG itself could launch within the 10 year expected time-frame continuing to provide the opportunities of the ISS in a new environment whilst being at the forefront of technological advancements as the human race looks to expand beyond earth and explore the galaxy.

Appendix

A. Orbit and attitude

TABLE VII: Orbital parameters.

Orbital parameters:	**Values**
Aposelene [km]	66071
Periselene [km]	3233
Semi-major axis [km]	34652
Eccentricity	0.9067
Inclination [deg]	89.995
Selenographic longitude of ascending node [deg]	289.49
Argument of periselene [deg]	90
Period [days]	6.7

The calculations and considerations about the annual costs for stationkeeping are based on the result of a Monte Carlo analysis computed by NASA and shown in Table VIII. [28] The force model includes the Sun, Earth, Moon, and Jupiter, whose motions are modeled using the ephemeris model.

TABLE VIII: Annual station keeping costs for noisy spacecraft.

	Max ΔV [m/s]	**Min ΔV [m/s]**	**Average ΔV [m/s]**
10 **[km],** 10 **[cm/s]***	3.37	1.53	2.26
100 **[km],** 100 **[cm/s]****	51.48	13.27	20.27

* Medium navigation errors
** Very high navigation errors

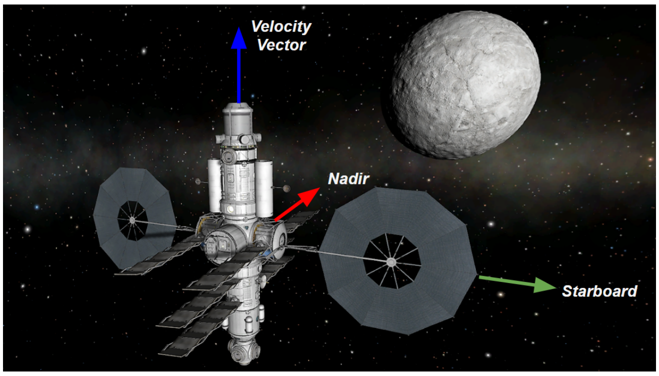


Fig. 8: The DSG Body Frame.

B. Communication

TABLE IX: Requirements for the data budget [5]

Transmitter → Receiver	**Down-link [Mbps]**	**Up-link [Mbps]**	**Band-width [MHz]**	**Gain [dBi]**	**EIRP [dBW]**
DSG → Earth	10	2	10	>28	>35
DSG → Transport	1	1	2	>35	>40

TABLE X: Parameters used for the link budget

Link	**Earth → DSG**	**DSG → Earth**	**Transport → DSG**	**DSG → Transport**
Frequency interval [GHz]	7.25-7.75	7.9-8.4	7.25-7.75	7.9-8.4
Chosen frequency [GHz]	7.5	8.3	7.75	8.4
Distance [m]	$3.91 \cdot 10^8$	$3.91 \cdot 10^8$	$7.18 \cdot 10^7$	$7.18 \cdot 10^7$
Temp. (receiver) [K]	20	298	20	373
E_b/N_0 [dB]	14	14	14	14
Aperture efficiency (receiver) [-]	0.75	0.94	0.90	0.90
Data rate [Mbps]	2 +	10	10	1
Bandwidth [MHz]	10	10	2	2
Antenna diameter (receiver) [m]	0.75	15	0.73	0.75

C. Resupply

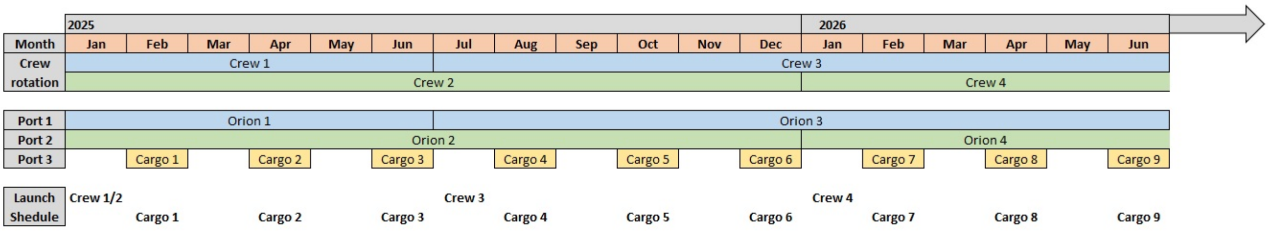


Fig. 9: Logistical Movement Schedule at the DSG

D. Assembly

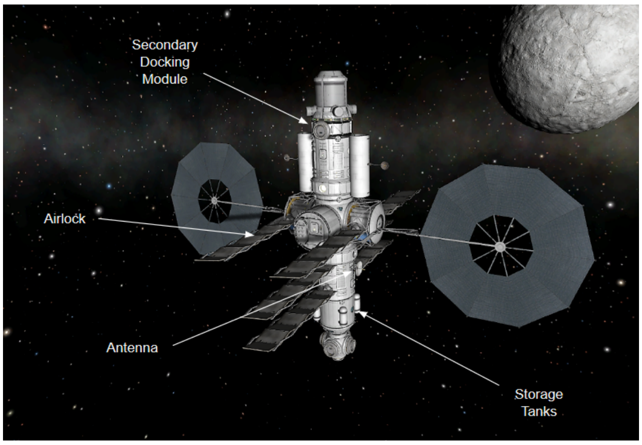


Fig. 10: Back view of the DSG.

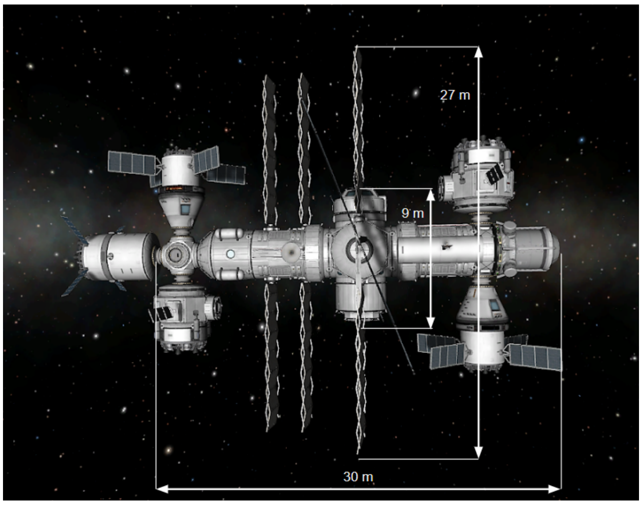


Fig. 11: Estimated size of the DSG.

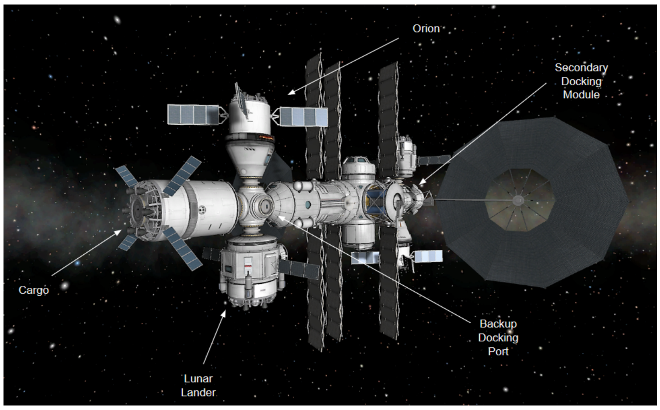


Fig. 12: Docking system of the DSG.

E. Radiation

TABLE XI: Career exposure limit for astronauts.[22]

Age (years)	25	35	45	55
Male	1.50 Sv	2.50 Sv	3.25 Sv	4.00 Sv
Female	1.00 Sv	1.75 Sv	2.50 Sv	3.00 Sv

F. Propulsion System

Calculations for the mass of the empty tanks [8]:

$$m_{PSS} = m_P + m_T = m_P\left(1 + \frac{zRT}{KM}\right) = 2616 \text{ kg}$$

$$m_T = m_{PSS} - m_P = 2616 - 2500 = 116 \sim 0.1 \text{ t}$$

where:

- m_{PSS} = combined mass of propellant and tankage [kg];
- m_P = mass of propellant [kg]: 2500;
- m_T = mass of tank [kg];
- z = gas compressibility factor: 0.3;
- R = gas constant: 8314 [J/(kmolK)];
- T = tank temperature at filling of tank [K]: 298.15;
- K = tank performance factor [m^2/s^2]: $12.20 \cdot 10^4$;
- M = molecular mass [kg/kmol]: 131.3

The mass of the propellant used in the worst case (very high navigation errors and maximum mass to be moved) for the annual station keeping is:

$$m_P = m_S\left(1 + e^{-\frac{\Delta V}{v_e}}\right)$$

where m_S is the mass of the station in kg for the configuration with the maximum number of docked vehicles (2 Lunar Modules, 2 Orion and 1 cargo) and it is:

$$m_S = m_{DSG} + 2m_L + 2m_O + m_C = 226670 \text{ kg}$$

- m_{DSG} = mass of the DSG [kg]: 75970 (see Table XII);
- m_L = mass of the Lunar Transport Vehicle [kg]: 24500 (as specified in the report of the Transport System Design);
- m_O = mass of Orion [kg]: 25850; [36]
- m_C = mass of the cargo vehicle [kg]: 50000 (rough estimation based on the amount of cargo to be transported).

The ΔV is the maximum annual velocity increment for the worst case, i.e. 51.48 m/s, and v_e is the exit velocity calculated from the maximum specific impulse:

$$v_e = I_s p \cdot g_e = 2600 \cdot 9.81 = 25506 \text{ m/s}$$

So the very conservative value for the mass of propellant used for station keeping for year is:

$$m_P = 457 \text{ kg} \sim 0.5 \text{ t}$$

G. Power System

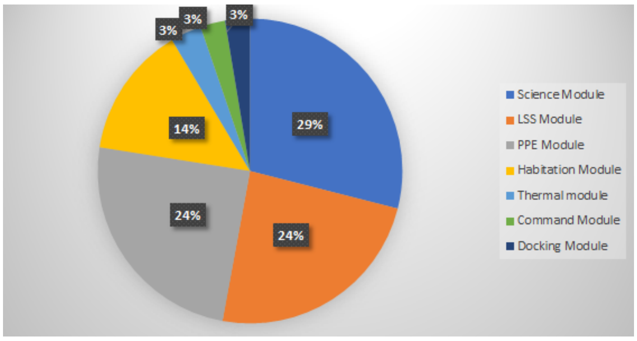


Fig. 13: A pie chart of the mass budget

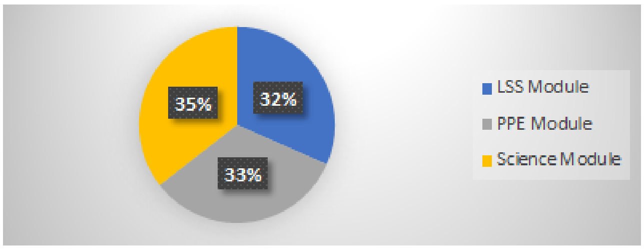


Fig. 14: A pie chart of the power budget

TABLE XII: Mass budget

Module:	**Mass: [ton]**
Science Module	22
LSS Module	19.19
PPE Module	19.68
Habitation Module	10-6
Thermal Module	2.50
Command Module	2.00
Docking Module	2.00
Total	**75.97**

TABLE XIII: Power budget

Module:	**Power [kW]**
Science Module	54
LSS Module	48
PPE Module	50
Habitation Module	54
Total	**152**

H. Thermal Control

The values of the coefficients have been found on the volume *Satellite Mission Design and Analysis*.

$$Q_{SUN} = S \cdot \frac{A_{TOT}}{2} = 602.8\ kW \tag{14}$$

where S is the solar heat flux, 1370 W/m^2 and A_{TOT} is the total surface of the station, estimated as 880 m^2.

$$Q_{GEN} = \eta P_{GEN} = 7.6\ kW \tag{15}$$

where η is the electrical efficiency of the station, estimated as 5%.

$$Q_{SP} = 2\varepsilon_{SP} S \frac{A_{SP}}{2} = 239.8\ kW \tag{16}$$

were A_{SP} is the area of the solar panels and ε_{SP} is their emissivity, 0.25.

$$Q_{SKIN} = 2\varepsilon_{SKIN} S \frac{A_{SKIN}}{2} = 237.0\ kW \tag{17}$$

were A_{SKIN} is the surface of the station without solar panels, estimated as 188 m^2 and ε_{SKIN} its emissivity, 0.92.

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